06

TECHNICAL MANUAL

STRUCTURAL REPAIRS

USAF SERIES

F-106A AND F-106B

AIRCRAFT

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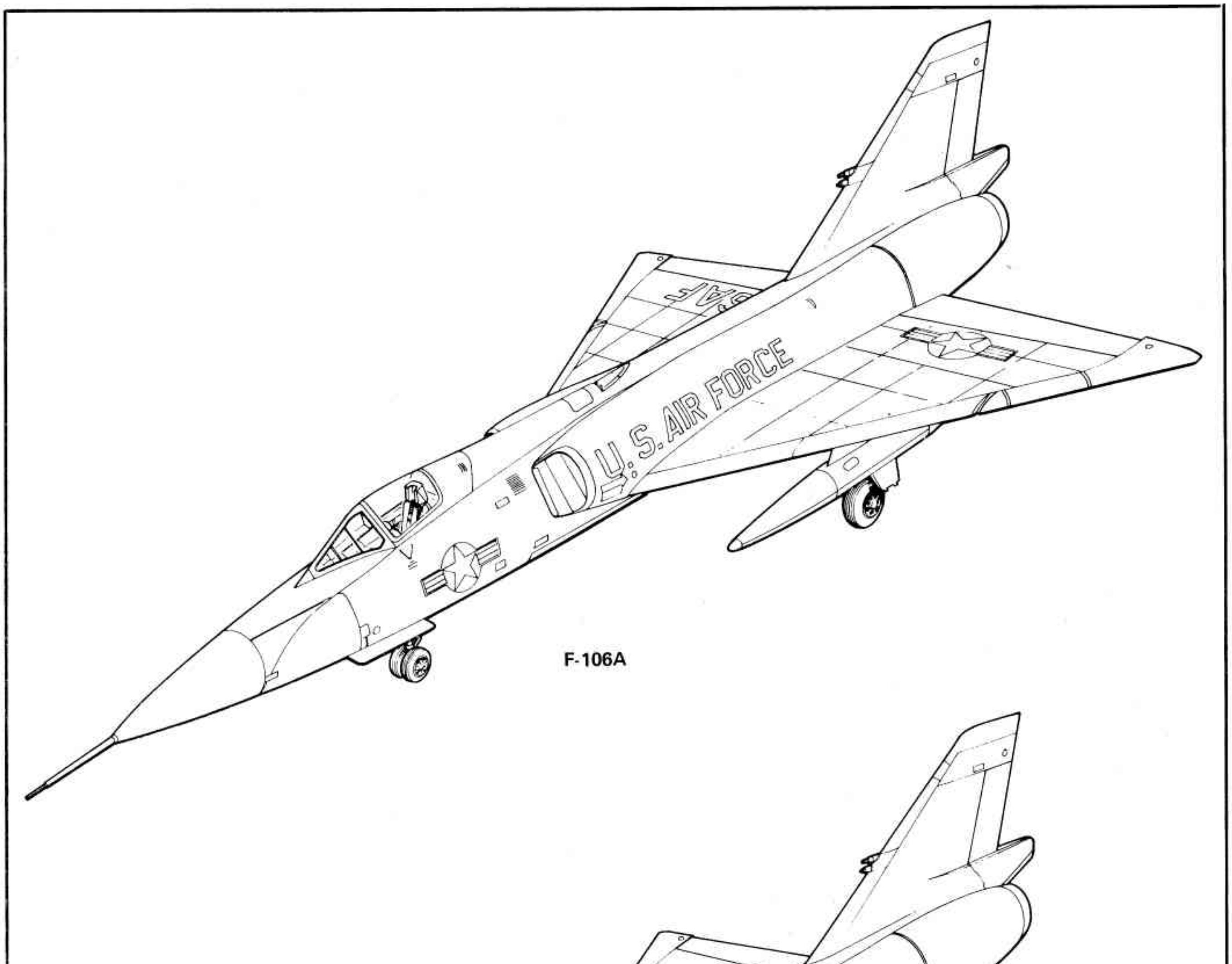


F-106A and F-106B Aircraft

INTRODUCTION

This manual is published to assist Air Force personnel in inspecting and repairing the structure of the F-106A and F-106B airplanes. The repair principles given will enable the airframe repairmen to make repairs that will be satisfactory for most of the damages experienced in normal operation.

In most cases, the repairs shown are within the capabilities of base organization level. Damage to a particular member will usually be accompanied by damage to the attaching parts, such as skin and web. A major repair may be accomplished by combining repairs for skin, webs, and stiffening members. If the possibility of successfully combining repairs seems doubtful, an aeronautical structures engineer should be consulted. A survey of the damage — the type, difficulty, and number of repairs required — and the available facilities will indicate when depot organization level assistance is required. Section I deals with the evaluation of damage, methods of repair, and describes repair materials and fasteners. Procedures for airplane alignment checks and use of optical instruments are also given in Section I. Sections II through VI describe the separate components of the airplane, call attention to points of special importance, and prescribe repairs for the particular types of structure found in each area. Specific repairs of assumed damage are located in the section applicable to the individual component. Typical repairs, which are applicable to elements found in more than one component, are contained in Section X. Section VIII lists the extrusions and roll-formed sections used in the construction of the airplane. Section IX is devoted to repair evaluation and procedures for damage occurring after landing gear failure. Section XI contains a summary of all the repair materials which may be required to fabricate the repair parts described in this manual. Specifications of these items are also contained in this section to avoid calling them out repeatedly in the text of the preceding sections.

The following publications provide supplementary information on the F-106A and F-106B airplanes:

T.O. 1F-106A-1	Flight Manual
T.O. 1F-106B-1	Flight Manual
T.O. 1F-106A-2 series	Maintenance Manuals
T.O. 1F-106A-4	Illustrated Parts Breakdown
T.O. 1F-106B-4	Illustrated Parts Breakdown
T.O. 1F-106A-5	Basic Weight Check List and Loading Data
T.O. 1F-106B-5	Basic Weight Check List and Loading Data
T.O. 1F-106A-6	Inspection Manual
T.O. 1F-106A-10	Power Package Buildup Instructions
T.O. 1F-106A-23	System Peculiar Corrosion Control
T.O. 1F-106A-36	Nondestructive Inspection Procedures
T.O. 1-1-2	Corrosion Control and Treatment for Aircraft
T.O. 1-1A-1	Aircraft Repair — General Manual for Structural Repair
T.O. 1-1-3	Repair of Integral and Removable Metal Fuel and Oil Tanks
T.O. 33B3-1-1	Inspection of Material, Radiography
T.O. 00-85-16	Preparation of Fighter, Trainer, Liaison and Helicopter Aircraft for Shipment
T.O. 1-1-1	Cleaning of Aeronautical Equipment
T.O. 1-1-4	Exterior Finishes, Insignia and Markings applicable to Aircraft and Missiles
T.O. 1-1-24	Maintenance Repair and Electrical Requirements of Fiber Laminate and Sandwich Constructed Airborne Radomes

Section I

GENERAL

1-1. GENERAL DESCRIPTION.

1-2. The F-106A and F-106B airplanes, manufactured by Convair, a Division of General Dynamics Corporation, are high-performance, land-based, delta-wing, all-weather interceptors. The F-106A is a single-place airplane whose primary mission is interception and destruction of attacking hostile airplanes or airborne missiles that operate within the performance capabilities of the airplane. The F-106B interceptor is a two-place, tandem version of the F-106A with pilot training provisions. Both airplanes are equipped with fully retractable tricycle landing gear and are powered by the Pratt and Whitney J75-P-17 continuous flow gas turbine engine. Externally, the F-106A and F-106B are identical in dimensions except for a slight increase in the cross-sectional area of the F-106B cockpit. Figure 1-1 shows the principal dimensions of the two airplanes. The fuselage fuel tank of the F-106B is located farther aft than the tank in the F-106A to accommodate the lengthened cockpit. In addition, the equipment located in the electronic bay of the F-106A is moved aft to the forward missile bay area on the F-106B. The lower halves of the missile bay doors on the F-106B are shortened to conform to this change in equipment location. The access and inspection provisions are shown on figures 1-2 thru 1-5, refer to paragraph 1-335 for structural breakdown of F-106A and F-106B major airplane components.

1-3. TYPE OF CONSTRUCTION.

1-4. Wing Group.

1-5. The F-106A and F-106B wings are of the full cantilever, stressed skin construction with a delta configuration. This configuration has a 60-degree sweepback of the leading edge and a 5-degree sweep forward of the elevon and wing tip trailing edges. The right and left wing panels are attached to the fuselage with special high-strength bolts through the main spars and fuselage bulkhead fittings, and by drag angles riveted to the inboard edge of the wing and attached to the fuselage structure by means of screws. Each wing panel is equipped with removable cambered leading edge sections, a cambered wing tip, an elevon, a main landing gear and gear wing fairing, and provisions for the external mounting of droppable fuel tanks. Refer to Section II for additional information on the wings and for wing repairs.

1-6. Tail Group.

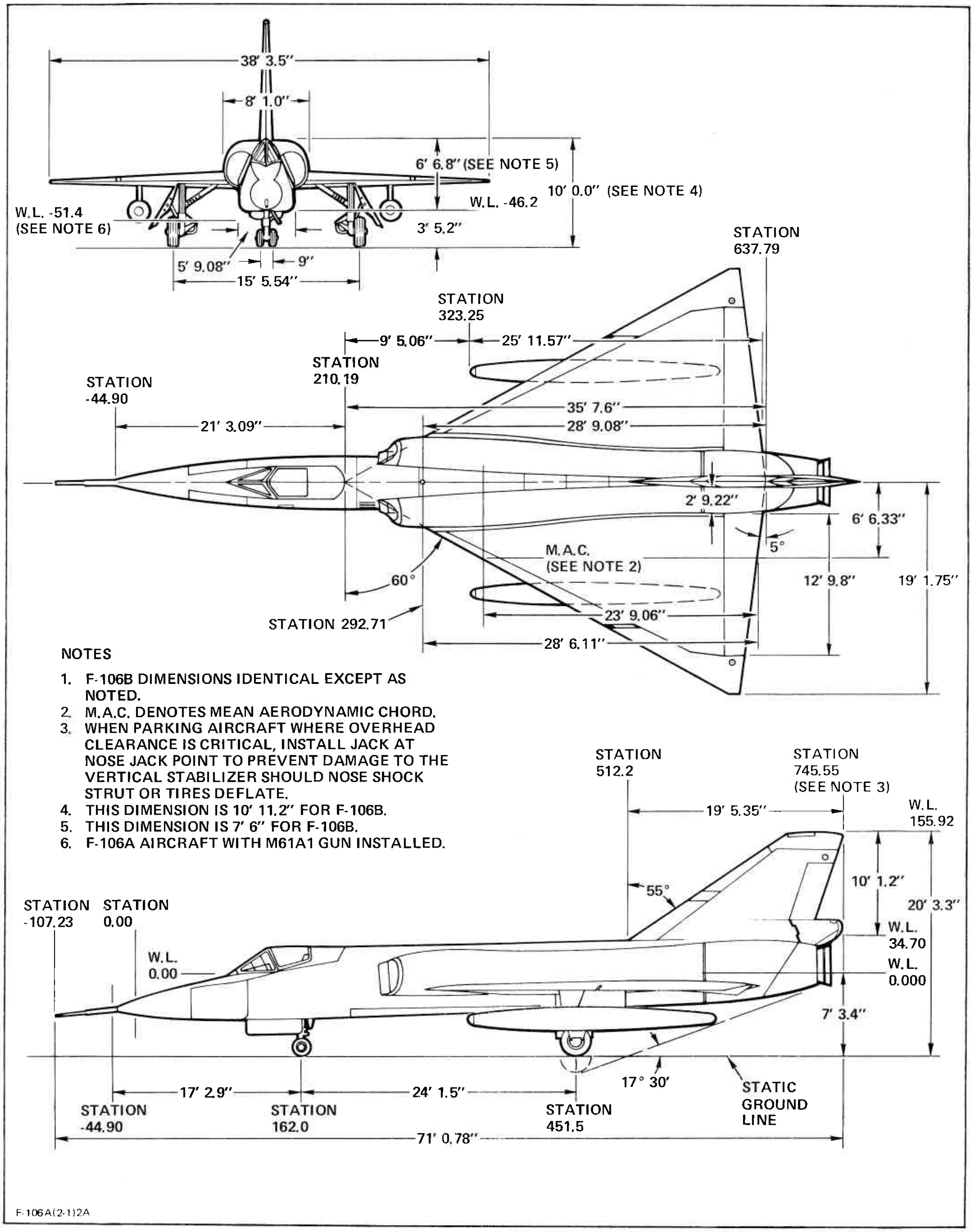
1-7. The tail group consists of a vertical fin and rudder. The fin assembly is an integral part of the fuselage structure and cannot be detached for repair or replacement without extensive separation from the fuselage bulkheads. The fin is equipped with a removable leading edge, a fin tip, and a rudder. The fin loads are carried by four vertical forged spars which are an integral part of the fuselage bulkhead assembly. The rudder is of the aluminum honeycomb core, sandwich-type construction with detachable hinge fittings on the leading edge spar. Refer to Section III for additional tail group information and for repairs to the fin and rudder.

1-8. Body Group.

1-9. The fuselage design is of semi-monocoque construction type with provisions for the radar equipment, the pilot's cockpit, an integral fuel tank, a missile bay, engine air intake ducts, main and nose landing gear wheel wells, electronic and accessory compartments, and the engine compartment. The fuselage structural framework consists basically of a series of semicircular zee and channel spliced beltframes, forged and built-up bulkheads, longerons, gussets, and intercostals. The fuselage framework is enclosed by stressed skins made of aluminum, magnesium, and titanium alloys that are attached with flush-head rivets and other types of fasteners. The fuselage structure depends heavily on the attached wing for longitudinal stiffening; adequate fuselage supports must be provided before a wing may be removed. Refer to Section IV for additional fuselage information and for repairs to the fuselage.

1-10. Landing Gear.

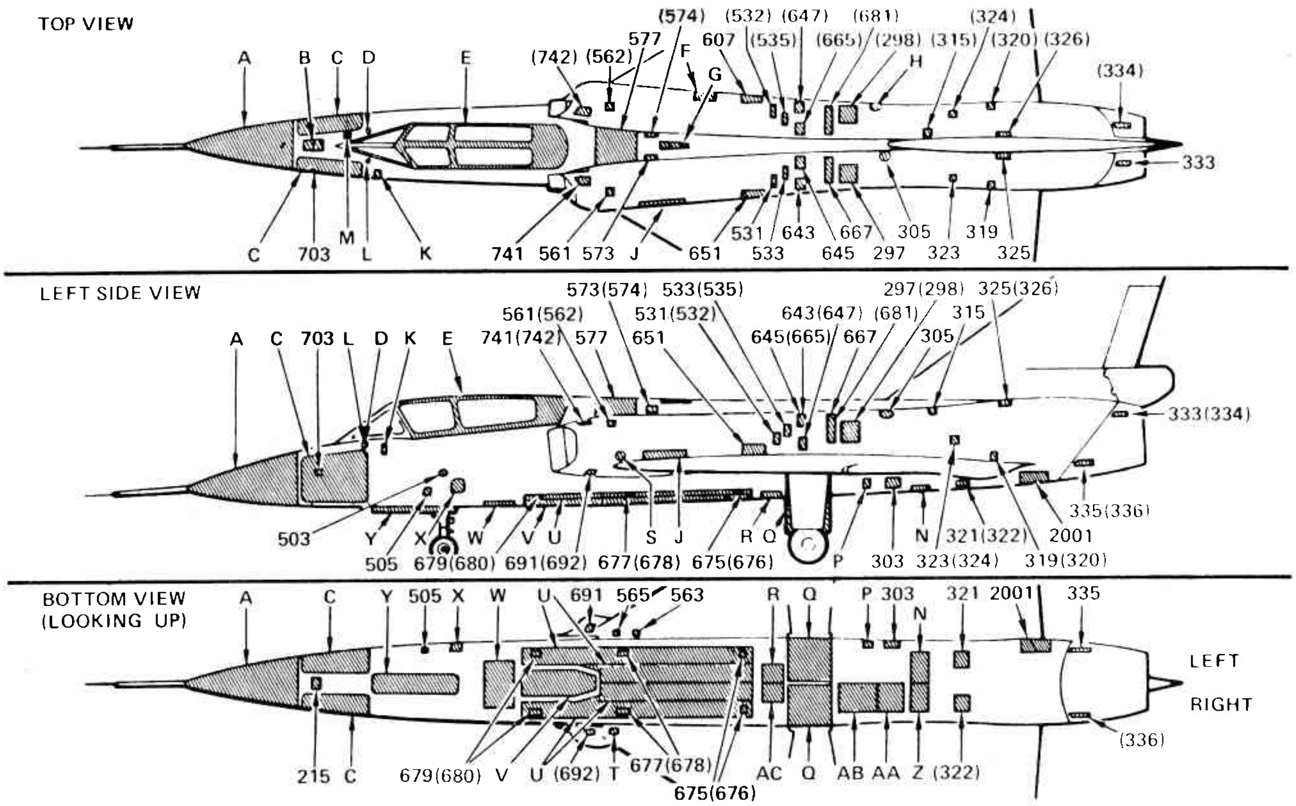
1-11. The F-106A and F-106B airplanes are equipped with electrically controlled, hydraulically operated, fully retractable, tricycle landing gear. The main landing gear retracts inboard and up into the wheel wells in the wing-fuselage area, while the nose landing gear retracts forward and up into the wheel well in the fuselage nose. The dual wheels on the nose gear are steerable during ground operations of the airplane. The nose and main gear assemblies are covered in the retracted position by hydraulically operated doors in the fuselage; wing fairings attached to the main gear struts enclose the main gear wheel well area in the wing



NOTES

1. F-106B DIMENSIONS IDENTICAL EXCEPT AS NOTED.
2. M.A.C. DENOTES MEAN AERODYNAMIC CHORD.
3. WHEN PARKING AIRCRAFT WHERE OVERHEAD CLEARANCE IS CRITICAL, INSTALL JACK AT NOSE JACK POINT TO PREVENT DAMAGE TO THE VERTICAL STABILIZER SHOULD NOSE SHOCK STRUT OR TIRES DEFLATE.
4. THIS DIMENSION IS 10' 11.2" FOR F-106B.
5. THIS DIMENSION IS 7' 6" FOR F-106B.
6. F-106A AIRCRAFT WITH M61A1 GUN INSTALLED.

Figure 1-1. Principal Dimensions



ACCESSIBLE EQUIPMENT

TOP VIEW

A	RADAR ANTENNA
B	INFRARED RECEIVER, MECHANISM AND ACTUATING CYLINDER
C	FORWARD ELECTRONIC EQUIPMENT
D	BRAKE SYSTEM COMPONENTS, WINDSHIELD ANTI-ICE TRANSFORMER
E	COCKPIT FURNISHINGS
741(742)	DUCT LIP ANTI-ICING LINES, UPPER
561(562)	VARIABLE RAMP PITOT-STATIC COMPONENTS
577	UPPER-AFT ELECTRONIC EQUIPMENT, AIR CONDITIONING COMPONENTS, ANTI-ICING TANK
573(574)	HEAT EXCHANGER EXHAUST DUCT ANCHOR BOLTS
F	ENGINE HOT SECTION ANALYZER COMPONENTS
G	AERIAL REFUELING RECEPTACLE
(607)	CURRENT TRANSFORMER, MAIN AC CONTACTOR EXTERNAL POWER MONITOR
531(532)	STRUCTURE ACCESS
(535)	CSD OIL COOLER
643(647)	OIL COOLING AIR CONTROL
645(665)	DUCT VALVE BEARING
(681)	CSD OIL COOLER
297(298)	ENGINE MOUNT, FORWARD
H	EQUIPMENT DELETED
(315)	BLEED MANIFOLD
323(324)	ELEVON ACTUATOR, FORWARD END
319(320)	ELEVON ACTUATOR, AFT END
325(326)	ENGINE MOUNT, AFT
333(334)	TAIL CONE LATCH, UPPER
305	ENGINE OIL TANK FILLER CAP
667	ENGINE OIL COOLER
645	TUBING OIL COOLING DUCT BEARING
533	ENGINE AIR-OIL COOLER
651	MA1 POWER TRANSFER RELAY BOX (SEE NOTE 2)
J	HYDRAULIC VALVES
K	CANOPY CONTROLS, EXTERNAL
L	BRAKE SYSTEM COMPONENTS, WINDSHIELD ANTI-ICE TRANSDUCER
M	NITROGEN ACCUMULATOR ADSORBER AND COMPRESSOR. IR COOLING SYSTEM
703	ANGLE OF ATTACK TRANSDUCER

LEFT SIDE VIEW (DOORS SHOWN IN TOP VIEW NOT LISTED)

335(336)	TAIL CONE LATCH, LOWER
321(322)	ELEVON VALVE
N	ENGINE
303	HYDRAULIC PUMP
P	FIRE ACCESS DOOR
Q	MAIN LANDING GEAR WHEEL WELL
R	RAM AIR TURBINE
675(676)	MISSILE BAY DOOR MECHANISM, AFT
S	CANOPY EXTERNAL JETTISON HANDLE
677(678)	MISSILE BAY DOOR MECHANISM, MID
691(692)	DUCT LIP ANTI-ICING, LOWER
V	LOWER-AFT ELECTRONIC EQUIPMENT
679(680)	MISSILE BAY DOOR MECHANISM, FORWARD
W	05 DOOR, LOWER-MID ELECTRONIC EQUIPMENT
X	EXTERNAL POWER RECEPTACLE
505	OXYGEN FILLER CONNECTION
Y	NOSE LANDING GEAR WHEEL WELL
503	CONTROL STICK TORQUE TUBE

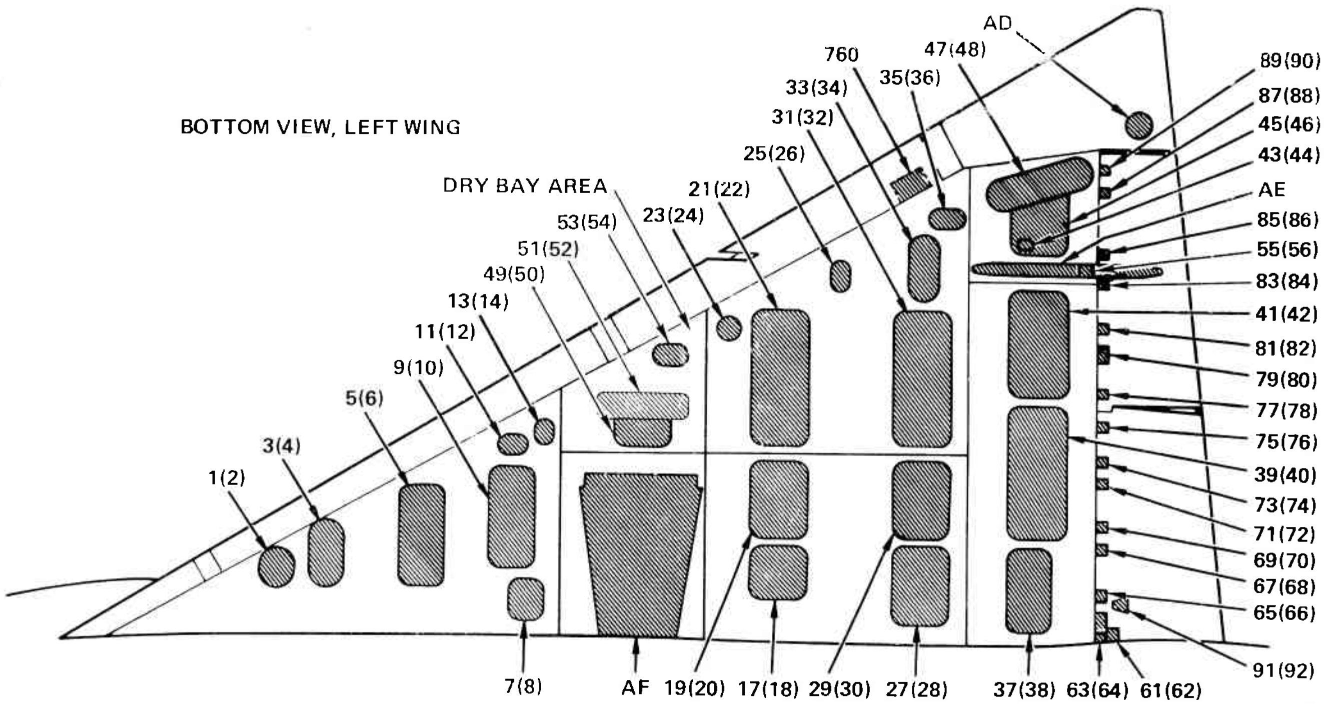
BOTTOM VIEW (DOORS SHOWN IN LEFT SIDE VIEW NOT LISTED)

565	COCKPIT-ELECTRONIC COMPARTMENT GROUND CONDITIONING CONNECTION
563	MISSILE BAY GROUND CONDITIONING CONNECTION
2001	AIR TURBINE GENERATOR
Z	ENGINE AND RAPID TUNE MAGNETRON HYDRAULIC POWER EQUIPMENT
AA	ENGINE ACCESSORIES
AB	CONSTANT SPEED REMOTE GEARBOX
AC	HYDRAULIC SYSTEM COMPONENTS
T	REFUELING ADAPTER
215	ANTENNA

NOTES

1. IDENTIFICATION NUMBERS SHOWN ARE ACCESS DOOR STENCIL NUMBERS. WHERE TWO NUMBERS ARE SHOWN IN DOOR LISTING, THE FIRST NUMBER APPLIES TO THE ACCESS DOOR IN THE LEFT WING, AND THE SECOND NUMBER (IN PARENTHESIS) APPLIES TO THE ACCESS DOOR IN THE RIGHT WING.
2. IDENTIFICATION LETTERS HAVE BEEN ASSIGNED TO THOSE DOORS OR AREAS THAT DO NOT HAVE A STENCILED NUMBER.
3. DOORS ARE LISTED BY VIEW IN CLOCKWISE ORDER.

Figure 1-3. Access and Inspection Provisions; F-106 B Fuselage



ACCESSIBLE EQUIPMENT

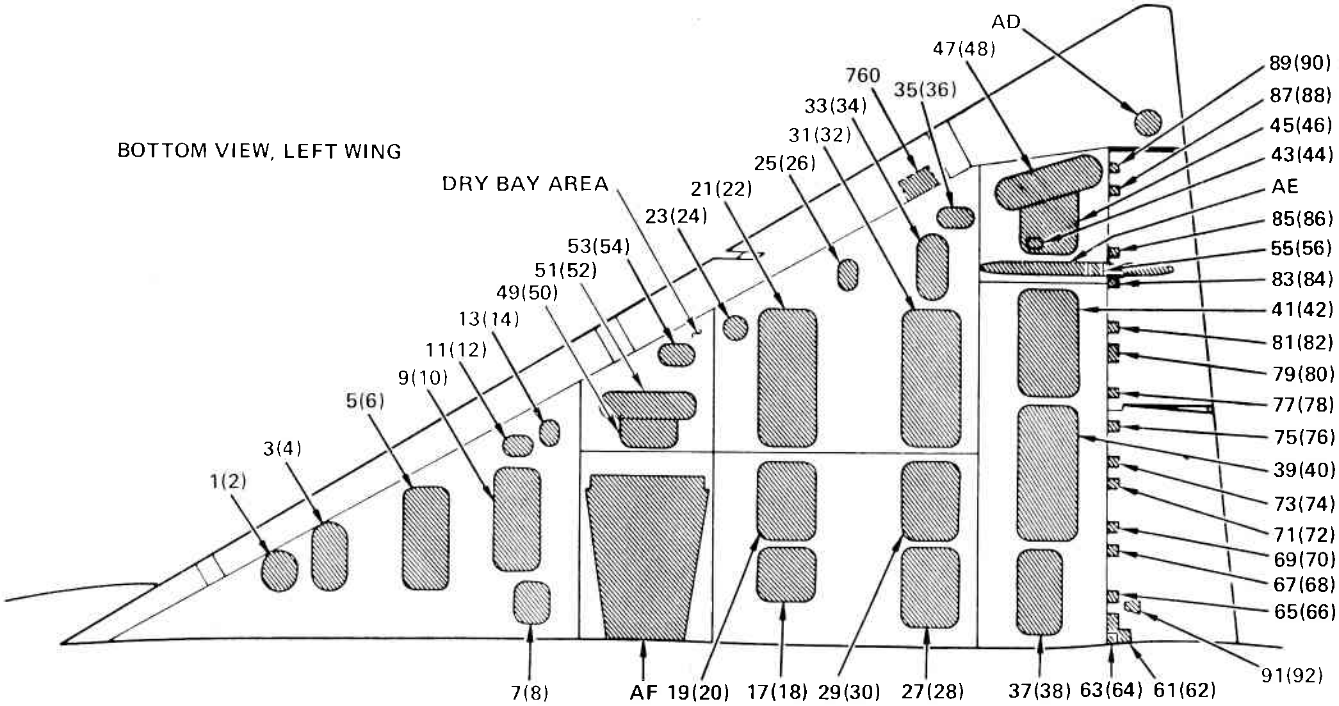
1(2)	FUEL SYSTEM TANK 1.
3(4)	FUEL SYSTEM TANK 1. C.G. TRANSFER LINE
5(6)	FUEL SYSTEM TANK 1. REFUEL SHUTOFF VALVES. REFUEL PILOT VALVES. VENT AND PRESSURE RELIEF VALVES. C.G. TRANSFER LINES, AIR OPERATED SHUTOFF VALVES, REFUEL PRESSURE OPERATED VALVES
9(10)	FUEL SYSTEM TANK 1. TANK PRESSURE CHECK VALVE. REFUEL VENT VALVE. C.G. TRANSFER LINE, REFUEL PILOT VALVE
11(12)	FUEL SYSTEM TANK 1
13(14)	FUEL SYSTEM TANK 1. EXTERNAL TANK SHUTOFF VALVE. TRANSFER LINE
49(50)	TANK PRESSURE SWITCH, BOOST PUMP PRESSURE SWITCH. TRANSFER LINE. EXTERNAL TANK PRESSURE REGULATOR AND FUEL CHECK VALVE
51(52)	PYLON SUPPORT BEAM
53(54)	VENT LINE
23(24)	FUEL SYSTEM TANK 2. VENT LINE
21(22)	ELECTRICAL FUEL QUANTITY RECEPTACLE. FUEL SYSTEM TANK 2. PRESSURE RELIEF VALVE. VENT VALVE. VENT OUTLET FITTING
25(26)	FUEL SYSTEM TANK 2 STRUCTURE ACCESS
31(32)	FUEL SYSTEM TANK 2. FLOAT VALVE. VACUUM RELIEF VALVE. VENT LINE. BELLMOUTH
33(34)	FUEL SYSTEM TANK 2 STRUCTURE ACCESS
760	REMOTE COMPASS TRANSMITTER (UPPER SURFACE OF RH LEADING EDGE ONLY)
35(36)	FUEL SYSTEM TANK 2 STRUCTURE ACCESS
47(48)	STRUCTURE ACCESS
AD	POSITION LIGHT
89(90)	ELEVON HINGE BOLT. BONDING WIRE
87(88)	ELEVON HINGE BOLT. BONDING WIRE
45(46)	STRUCTURE ACCESS. HYDRAULIC LINES
43(44)	ELEVON ACTUATOR BOLT
AE	ELEVON ACTUATOR
85(86)	ELEVON HINGE BOLT. BONDING WIRE
55(56)	ACTUATOR FITTING
83(84)	ELEVON HINGE BOLT. BONDING WIRE
41(42)	FUEL SYSTEM TANSFER TANK. C.G. TRANSFER LINE. FUEL SHUTOFF VALVE. LOW LEVEL PILOT VALVE
81(82)	ELEVON HINGE BOLT. BONDING WIRE
79(80)	ELEVON HINGE BOLT. BONDING WIRE
77(78)	ELEVON HINGE BOLT. BONDING WIRE
75(76)	ELEVON HINGE BOLT. BONDING WIRE
39(40)	FUEL SYSTEM TANSFER TANK. C.G. TRANSFER LINE. VENT VALVE. REFUEL SHUTOFF VALVE. LOW LEVEL PILOT VALVE. VACUUM RELIEF VALVE. ELECTRICAL FUEL QUANTITY RECEPTACLE
73(74)	ELEVON HINGE BOLT. BONDING WIRE
71(72)	ELEVON HINGE BOLT. BONDING WIRE
69(70)	ELEVON HINGE BOLT. BONDING WIRE
67(68)	ELEVON HINGE BOLT. BONDING WIRE
65(66)	ELEVON HINGE BOLT. BONDING WIRE
91(92)	ELEVON FITTING
61(62)	ELEVON HINGE BOLT. BONDING WIRE. (ELEVON LOWER SURFACE)
63(64)	ELEVON HINGE BOLT. BONDING WIRE. (ELEVON UPPER SURFACE)
37(38)	FUEL SYSTEM TRANSFER TANK. REFUEL PILOT VALVE, REFUEL VENT VALVE, PRESSURE RELIEF VALVE, AIR CHECK VALVE, LOW LEVEL PILOT VALVE, FUEL SHUTOFF VALVE, REFUEL PRESSURE OPERATED SHUTOFF VALVE
27(28)	FUEL SYSTEM TANK 3. BOOST PUMP. ENGINE SUPPLY LINE. T-FLAPPER CHECK VALVE
29(30)	FUEL SYSTEM TANK 3. C.G. TRANSFER LINE. SOLENOID OPERATED FUEL SCAVENGE VALVE, AIR CHECK VALVE, REFUEL SHUTOFF VALVE
17(18)	FUEL SYSTEM TANK 3. BELLMOUTH. VENT VALVE.
19(20)	FUEL SYSTEM TANK 3. FUEL BOOST PUMP. C.G. TRANSFER LINE, REFUELING PILOT FLOAT VALVE. TRANSFER LINE
AF	MAIN WHEEL WELL
7(8)	FUEL SYSTEM TANK 1. BELLMOUTH

NOTES

1. IDENTIFICATION NUMBERS SHOWN ARE ACCESS DOOR STENCIL NUMBERS. WHERE TWO NUMBERS ARE SHOWN IN DOOR LISTING, THE FIRST NUMBER APPLIES TO THE ACCESS DOOR IN THE LEFT WING AND THE SECOND NUMBER (IN PARENTHESIS) APPLIES TO THE ACCESS DOOR IN THE RIGHT WING.
2. IDENTIFICATION LETTERS HAVE BEEN ASSIGNED TO THOSE DOORS OR AREAS THAT DO NOT HAVE A STENCILED NUMBER.

F-106A(2-5-2)14B

Figure 1-4. Access and Inspection Provisions, Wing, F-106A



ACCESSIBLE EQUIPMENT

1(2)	FUEL SYSTEM TANK 1.
3(4)	FUEL SYSTEM TANK 1.
5(6)	FUEL SYSTEM TANK 1. AAR VENT VALVE, PRESSURE RELIEF VALVE, REFUEL PILOT FLOAT VALVE, PRESSURE RELIEF VALVE, TRANSFER LINE
9(10)	FUEL SYSTEM TANK 1. TANK PRESSURE CHECK VALVE. REFUEL VENT VALVE. RE-FUEL PILOT FLOAT VALVE,REFUEL PILOT VALVES EXTERNAL TANK REFUEL VALVE TRANSFER LINE.
11(12)	FUEL SYSTEM TANK 1.
13(14)	FUEL SYSTEM TANK 1. EXTERNAL TANK SHUTOFF VALVE. VACUUM RELIEF VALVE. TRANSFER LINE.
49(50)	TANK PRESSURE SWITCH. BOOST PUMP PRESSURE SWITCH. TRANSFER LINE. EX-TERNAL TANK PRESSURE REGULATOR AND AIR SOLENOID VALVE.
51(52)	PYLON SUPPORT BEAM
53(54)	VENT LINE
23(24)	FUEL SYSTEM TANK 2. VENT LINE
21(22)	FUEL SYSTEM TANK 2. ELECTRICAL FUEL QUANTITYRECEPTACLE. PRESSURE RELIEF VALVE. VENT OUTLET FITTING.
25(26)	FUEL SYSTEM TANK 2 STRUCTURE ACCESS.
31(32)	FUEL SYSTEM TANK 2. VENT LINE BELL-MOUTH
33(34)	FUEL SYSTEM TANK 2 STRUCTURE ACCESS
760	REMOTE COMPASS TRANSMITTER (UPPER SUR-FACE OF RH LEADING EDGE ONLY)
35(36)	FUEL SYSTEM TANK 2 STRUCTURE ACCESS
47(48)	STRUCTURE ACCESS
AD	POSITION LIGHT
89(90)	ELEVON HINGE BOLT. BONDING WIRE
87(88)	ELEVON HINGE BOLT. BONDING WIRE
45(46)	STRUCTURE ACCESS. HYDRAULIC LINES
43(44)	ELEVON ACTUATOR BOLT
AE	ELEVON ACTUATOR
85(86)	ELEVON HINGE BOLT. BONDING WIRE
55(56)	ACTUATOR FITTING
83(84)	ELEVON HINGE BOLT. BONDING WIRE
41(42)	FUEL SYSTEM TRANSFER TANK. FUEL SHUTOFF VALVE. LOW LEVEL PILOT VALVE
81(82)	ELEVON HINGE BOLT. BONDING WIRE
79(80)	ELEVON HINGE BOLT. BONDING WIRE
77(78)	ELEVON HINGE BOLT. BONDING WIRE
75(76)	ELEVON HINGE BOLT. BONDING WIRE
39(40)	FUEL SYSTEM TRANSFER TANK. ELEC-TRICAL FUEL QUANTITY RECEPTACLE. REPLENISH PILOT FLOAT VALVE.
73(74)	ELEVON HINGE BOLT. BONDING WIRE
71(72)	ELEVON HINGE BOLT. BONDING WIRE
69(70)	ELEVON HINGE BOLT. BONDING WIRE
67(68)	ELEVON HINGE BOLT. BONDING WIRE
65(66)	ELEVON HINGE BOLT. BONDING WIRE
91(92)	ELEVON FITTING
61(62)	ELEVON HINGE BOLT. BONDING WIRE. (ELEVON LOWER SURFACE)
63(64)	ELEVON HINGE BOLT. BONDING WIRE. (ELEVON UPPER SURFACE)
37(38)	FUEL SYSTEM TRANSFER TANK. TRANSFER LINE PILOT FLOAT VALVE. ELECTRICAL FUEL QUANTITY RECEPTACLE.
27(28)	FUEL SYSTEM TANK 3. BOOST PUMP EN-GINE SUPPLY LINE. DUAL T CHECK VALVE
29(30)	FUEL SYSTEM TANK 3.
17(18)	FUEL SYSTEM TANK 3, BELLMOUTH ANTI-G VENT VALVE.
19(20)	FUEL SYSTEM TANK 3. FUEL BOOST PUMP. REFUEL SHUTOFF VALVE, SOLENOID-OPERATED SCAVENGE VALVE. TRANSFER LINE
AF	MAIN WHEEL WELL
7(8)	FUEL SYSTEM TANK 1. BELLMOUTH

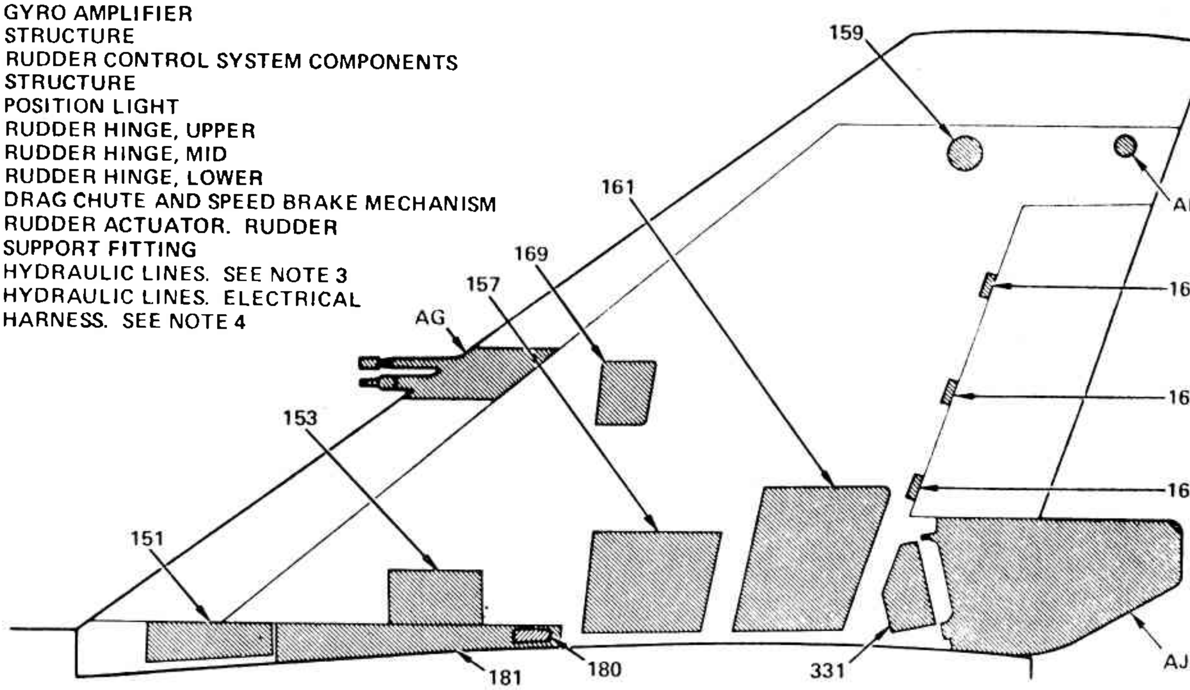
NOTES

1. IDENTIFICATION NUMBERS SHOWN ARE ACCESS DOOR STENCIL NUMBERS. WHERE TWO NUMBERS ARE SHOWN IN DOOR LISTING, THE FIRST NUMBER APPLIES TO THE ACCESS DOOR IN THE LEFT WING, AND THE SECOND NUMBER (IN PARENTHESIS) APPLIES TO THE ACCESS DOOR IN THE RIGHT WING.
2. IDENTIFICATION LETTERS HAVE BEEN ASSIGNED TO THOSE DOORS OR AREAS THAT DO NOT HAVE A STENCILED NUMBER.

Figure 1-5. Access and Inspection Provisions, Wing, F-106B

ACCESSIBLE EQUIPMENT

151 ENGINE BLEED AIR DUCT. SEE NOTE 3
153 STRUCTURE
AG "Q" (ARTIFICIAL FEEL SYSTEM) INTAKE
157 GYRO AMPLIFIER
169 STRUCTURE
161 RUDDER CONTROL SYSTEM COMPONENTS
159 STRUCTURE
AH POSITION LIGHT
163 RUDDER HINGE, UPPER
165 RUDDER HINGE, MID
167 RUDDER HINGE, LOWER
AJ DRAG CHUTE AND SPEED BRAKE MECHANISM
331 RUDDER ACTUATOR. RUDDER SUPPORT FITTING
180 HYDRAULIC LINES. SEE NOTE 3
181 HYDRAULIC LINES. ELECTRICAL HARNESS. SEE NOTE 4



NOTES

1. IDENTIFICATION NUMBERS SHOWN ARE ACCESS DOOR STENCIL NUMBERS.
2. IDENTIFICATION LETTERS HAVE BEEN ASSIGNED TO DOORS OR AREAS THAT DO NOT HAVE A STENCILED NUMBER.
3. APPLICABLE TO F-106A AIRCRAFT 56-453,-454, 56-456 THRU 57-2506 AND F-106B AIRCRAFT 57-2508 THRU 57-2531.
4. APPLICABLE TO F-106A AIRCRAFT 58-759 AND SUBSEQUENT AND F-106B AIRCRAFT 57-2532 AND SUBSEQUENT.

F-106A(2-5-2)16

Figure 1-6. Access and Inspection Provisions, Fin.

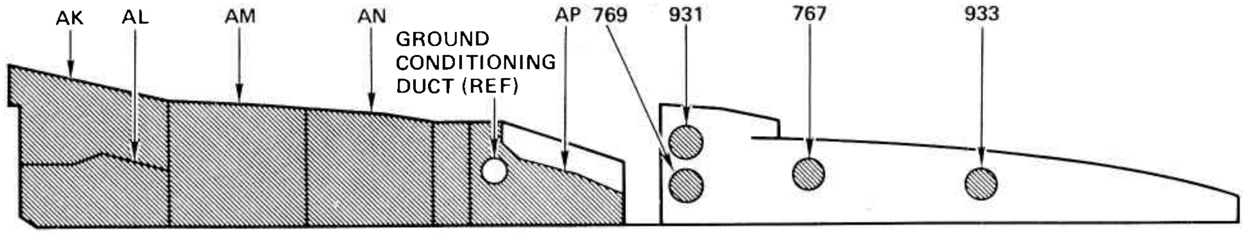
lower surface. Refer to Section V for additional landing gear information and for repairs to the landing gear.

1-12. Engine Compartment.

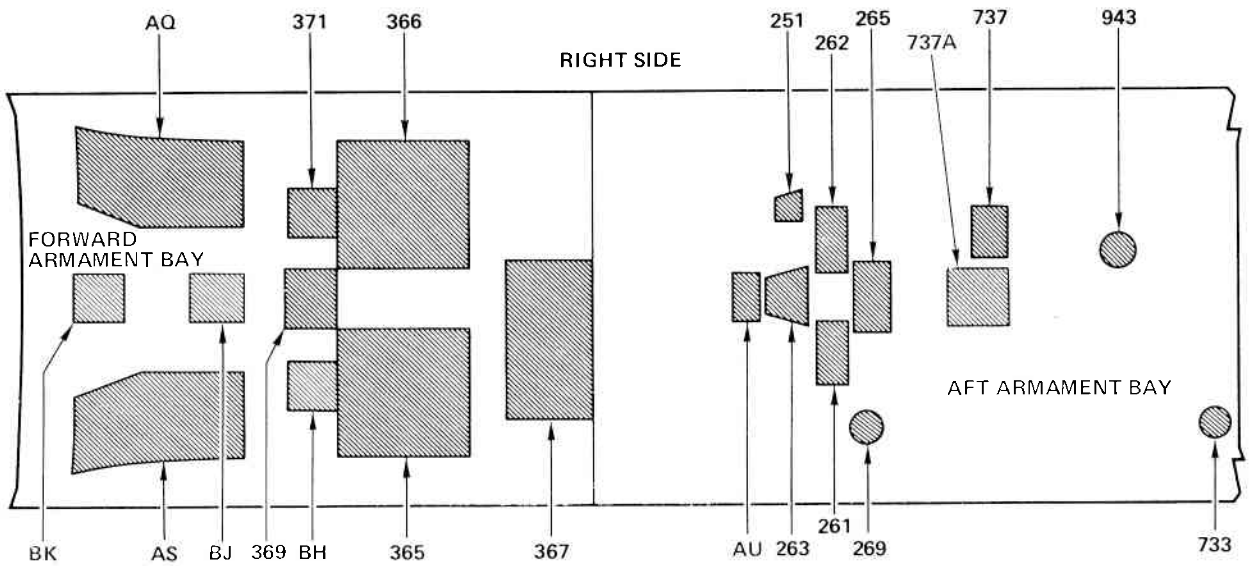
1-13. The aft section of the fuselage from sta 472.00 to the aft end of the tail cone is designed to accommodate the engine. The engine is enclosed by a thin gage titanium shroud containing insulation blankets to provide a fireseal and a barrier to protect the structure from the effects of engine heat. Several access doors are located in the aft section of the fuselage for inspection and maintenance of the engine and engine accessories. The engine is supported within the fuselage by one thrust mount at the forward right-hand side, one rigid link on the forward left-hand side, and by two adjustable links at the aft end. Refer to Section VI for engine section repair information.

1-14. STRESSES.

1-15. A high rate of performance is required of the F-106A and F-106B airplanes, their armament, and their associated electronic equipment in the execution of a mission under all weather and visibility conditions. Because these airplanes must operate at extremes of speed and altitude, it is highly important that the structural integrity be maintained. The airplanes are designed with a predetermined safety margin to provide each structural component with sufficient strength to withstand the varying loads imposed on it, but not over-strength because of the unnecessary increase in weight. For this reason it is important that any damaged structural component be repaired to fully restore its original strength. The importance of the damage and the type of repair required will require an individual decision for each damaged airplane after a thorough inspection and evaluation of the damaged area. The first inspection and evaluation will usually be conducted by advanced base personnel and a decision made as to whether the damage is negligible, can be repaired by using information found in this manual, or will require the services of an aeronautical structures engineer. The stresses which can be developed safely in a member are usually dependent on three factors: cross-sectional area, shape, and type of material. The shape, cross-sectional area, and type of material of the structural members used in the F-106A and



VIEW LOOKING OUTBOARD, LEFT SIDE



LEFT SIDE
VIEW LOOKING DOWN, ROOF

ACCESSIBLE EQUIPMENT

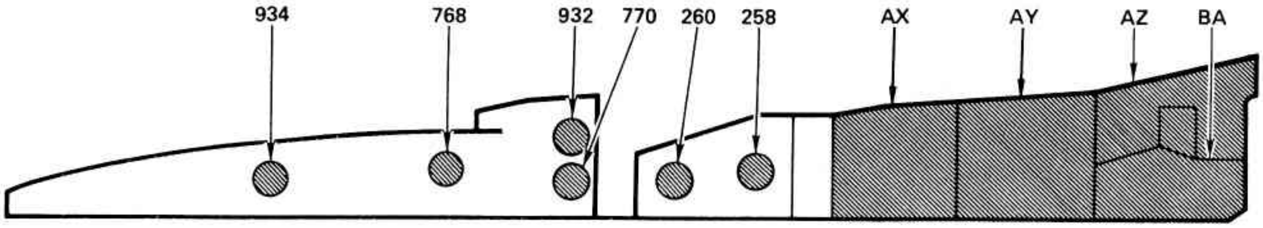
AK	PNEUMATIC LINES. HYDRAULIC LINES
AL	ELEVATOR CONTROL
AM	ELEVATOR CONTROL. PNEUMATIC LINES. HYDRAULIC LINES
AN	ELEVATOR CONTROL. FUEL TRANSFER LINE. PNEUMATIC LINES. HYDRAULIC LINES
AP	ELEVATOR CONTROL. GROUND CONDITIONING DUCT. PNEUMATIC LINES. HYDRAULIC LINES
769	ELEVATOR CONTROL
931	FUEL LINE
767	ELEVATOR CONTROL
933	ELEVATOR CONTROL
AQ	ELECTRICAL HARNESSES
371	ARMAMENT CONTROL RELAY BOX. ELECTRICAL HARNESSES
366	ARMAMENT SYSTEM RELAYS. ELECTRICAL HARNESSES. RUDDER CABLE PULLEY. VARIABLE RAMP JACKS
251	ELECTRICAL HARNESSES
262	FUEL TRANSFER LINE
265	PNEUMATIC SYSTEM AIR FLASK FITTING
737/737A	REFUEL PRESSURE LINE
943	ELECTRICAL HARNESS CONDUIT
733	THROTTLE TELEFLEX CONDUIT
269	THROTTLE TELEFLEX CONDUIT
261	FUEL TRANSFER LINE
263	PNEUMATIC SYSTEM AIR FLASK FITTING
AU	PNEUMATIC SYSTEM AIR FLASK FITTING
367	AIR CONDITIONING SYSTEM COMPONENTS. PNEUMATIC SYSTEM AIR FLASKS. VARIABLE RAMP JACKS
365	AIR CONDITIONING DUCT. RUDDER CABLE PULLEY. VARIABLE RAMP JACKS. MOISTURE SEPARATOR INLET THERMOSTAT
BH	VARIABLE RAMP PITOT STATIC SHUTTLE VALVE
369	MISSILE INTERVALOMETER
BJ	AIR CONDITIONING DUCT
AS	AIR CONDITIONING SYSTEM DUCTING AND MOISTURE SEPARATOR
BK	INTERVALOMETER

NOTES

1. IDENTIFICATION NUMBERS SHOWN ARE ACCESS DOOR STENCIL NUMBERS.
2. IDENTIFICATION LETTERS HAVE BEEN ASSIGNED TO THOSE DOORS OR AREAS THAT DO NOT HAVE A STENCILED NUMBER.

F-106A(2-5-2)17-1C

Figure 1-7. Access and Inspection Provisions, Armament Bay, F-106A (Sheet 1 of 2)



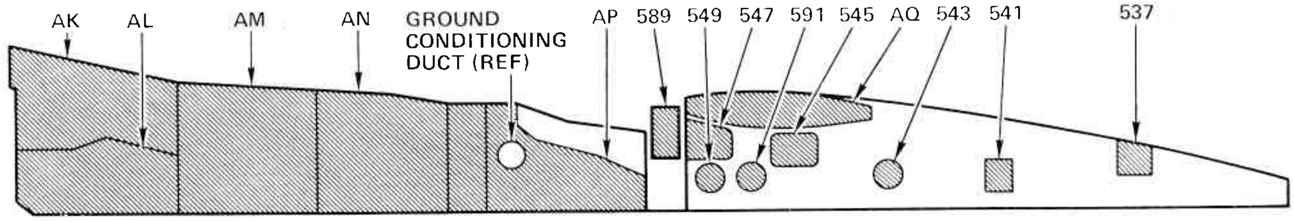
VIEW LOOKING OUTBOARD, RIGHT SIDE

ACCESSIBLE EQUIPMENT

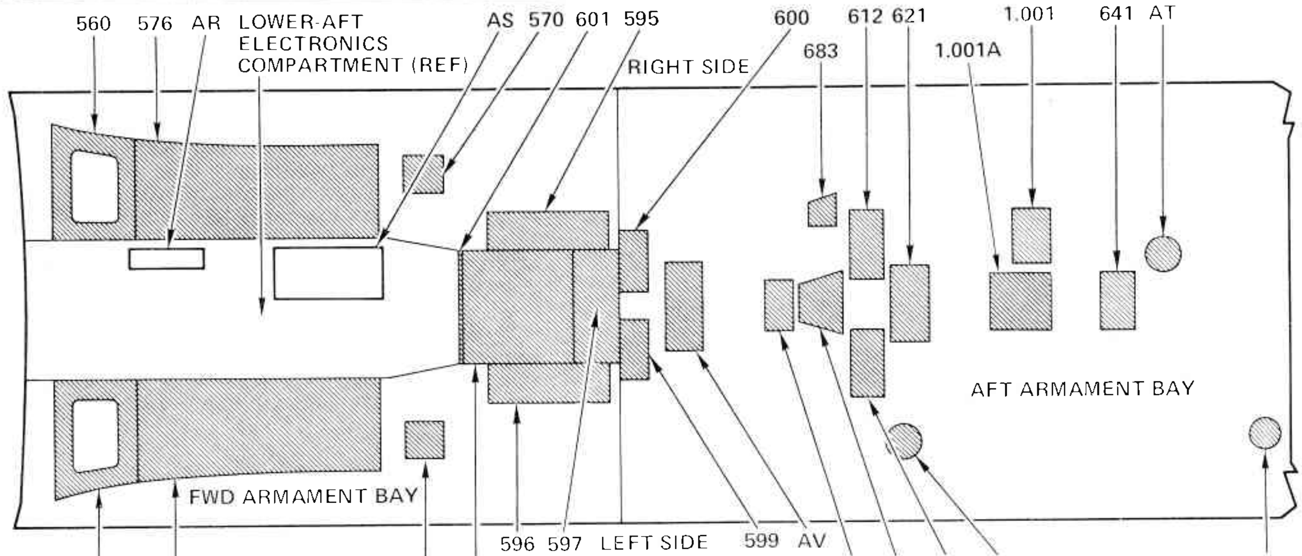
934	AILERON CONTROL
768	AILERON CONTROL
932	FUEL LINE
770	AILERON CONTROL
260	AILERON CONTROL
258	AILERON CONTROL
AX	FUEL TRANSFER RELAYS NOS. 1, 2, 3, AND 4. AILERON CONTROL BELL CRANK. FUEL TRANSFER LINE
AY	M.W.W. DOOR CLOSE RELAY. IGNITION RELAY. IGNITION ARMING RELAY. ANTI-ICE CONTROL RELAY. EXTERNAL FUEL TANK EJECTION RELAY. HYDRAULIC FLASHER RESET RELAY. DC POWER INTERLOCK RELAY. DC POWER FAILURE WARNING RELAY. AC EMERGENCY CONTROL RELAY. AC EXTERNAL POWER INTERLOCK RELAY.
AZ	AC POWER DISCONNECT RELAY. DC POWER DISCONNECT RELAY. OVERHEAT TEST RELAYS, LOOPS 1 AND 2. OVERHEAT FLASHER. HYDRAULIC PRESSURE WARNING FLASHER. PITCH "G" LIMITER INTERLOCK. ELEVATOR CHANNEL RELAY. AILERON CHANNEL RELAY. AC GENERATOR CONTROL PANEL. OVERHEAT DETECTION CONTROL. FIRE DETECTION CONTROL.
BA	AILERON CONTROL

F-106A(2-5-2)17-2A

Figure 1-7. Access and Inspection Provisions, Armament Bay, F-106A (Sheet 2 of 2)



VIEW LOOKING OUTBOARD, LEFT SIDE



VIEW LOOKING DOWN, ROOF

ACCESSIBLE EQUIPMENT

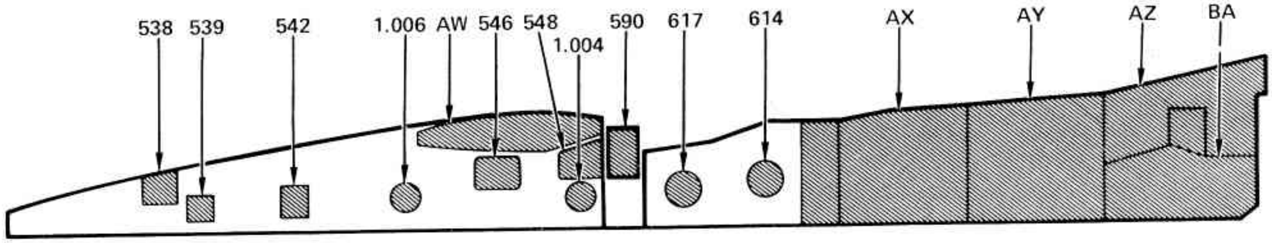
AK	PNEUMATIC LINES, HYDRAULIC LINES
AL	ELEVATOR CONTROL
AM	ELEVATOR CONTROL PNEUMATIC LINES, HYDRAULIC LINES
AN	ELEVATOR CONTROL, FUEL TRANSFER LINE, PNEUMATIC LINES, HYDRAULIC LINES
AP	ELEVATOR CONTROL, GROUND CONDITIONING DUCT, PNEUMATIC LINES, HYDRAULIC LINES
589	AIR CONDITIONING DUCTING
549	ELEVATOR CONTROL, FUEL VENT LINE
547	FUEL VENT LINE
591	ELEVATOR CONTROL, VARIABLE RAMP SYSTEM DRAIN VALVE
545	FUEL VENT LINE
AQ	VARIABLE RAMP, LOWER-AFT ACTUATOR JACK, RUDDER CABLE PULLEY
543	ELEVATOR CONTROL, LUBRICATION
541	ELEVATOR CONTROL, FUEL DRAIN VALVE
537	ELEVATOR CONTROL, FUEL DRAIN VALVE
560	ELECTRICAL HARNESSES, PNEUMATIC LINES, HYDRAULIC LINES, AIR CONDITIONING DUCTING, RUDDER CABLES
576	RUDDER CABLES, FUEL SYSTEM DRAIN LINES
AR	AIR CONDITIONING DUCTING
AS	MOISTURE SEPARATOR, MOISTURE SEPARATOR SERVO BYPASS VALVE
570	VARIABLE RAMP, LOWER-FORWARD ACTUATOR JACK
601	ELECTRICAL HARNESSES, AIR CONDITIONING DUCTING (SEE NOTE 3)
595	ELECTRONIC COOLING AIRFLOW CONTROL
600	VARIABLE RAMP AIR FLASK, DUCT LIP ANTI-ICE REGULATOR, GROUND COOLING CHECK VALVE
683	ELECTRICAL HARNESS
612	FUEL TRANSFER LINE
621	PNEUMATIC SYSTEM AIR FLASK FITTING
1.001/1.001A	REFUEL PRESSURE LINE
641	COOLING LINE
AT	ELECTRICAL HARNESS CONDUIT
627	THROTTLE TELEFLEX CONDUIT
623	THROTTLE TELEFLEX CONDUIT
611	FUEL TRANSFER LINE
619	PNEUMATIC SYSTEM AIR FLASK FITTING
AU	PNEUMATIC SYSTEM AIR FLASK FITTING
AV	TURBINE DISCHARGE, AIR TEMPERATURE CONTROL VALVE
599	GROUND COOLING CHECK VALVE, DUCT LIP ANTI-ICE REGULATOR
597	HEAT EXCHANGER
596	CABIN AIR SENSOR AND SHUTOFF VALVE, CABIN TEMPERATURE CONTROL VALVE
373	AIR CONDITIONING SYSTEM COMPONENTS, PNEUMATIC SYSTEM AIR FLASKS, VARIABLE RAMP, UPPER ACTUATOR JACKS
569	VARIABLE RAMP, LOWER-FORWARD ACTUATOR JACK
575	PNEUMATIC LINES, HYDRAULIC LINES, BALLISTIC HOSES, THROTTLE TELEFLEX CONDUIT
559	ELECTRICAL HARNESSES, AIR CONDITIONING DUCTING, RUDDER CABLES, PNEUMATIC LINES, HYDRAULIC LINES, THROTTLE TELEFLEX CONDUIT

NOTES

1. IDENTIFICATION NUMBERS SHOWN ARE ACCESS DOOR STENCIL NUMBERS.
2. IDENTIFICATION LETTERS HAVE BEEN ASSIGNED TO THOSE DOORS OR AREAS THAT DO NOT HAVE A STENCILED NUMBER.
3. DOOR IS LOCATED ADJACENT TO AFT FACE OF ELECTRONICS COMPARTMENT AND ROOF OF MISSILE BAY.

F-106B(2-5-2)5-1B

Figure 1-8. Access and Inspection Provisions, Armament Bay, F-106B (Sheet 1 of 2)



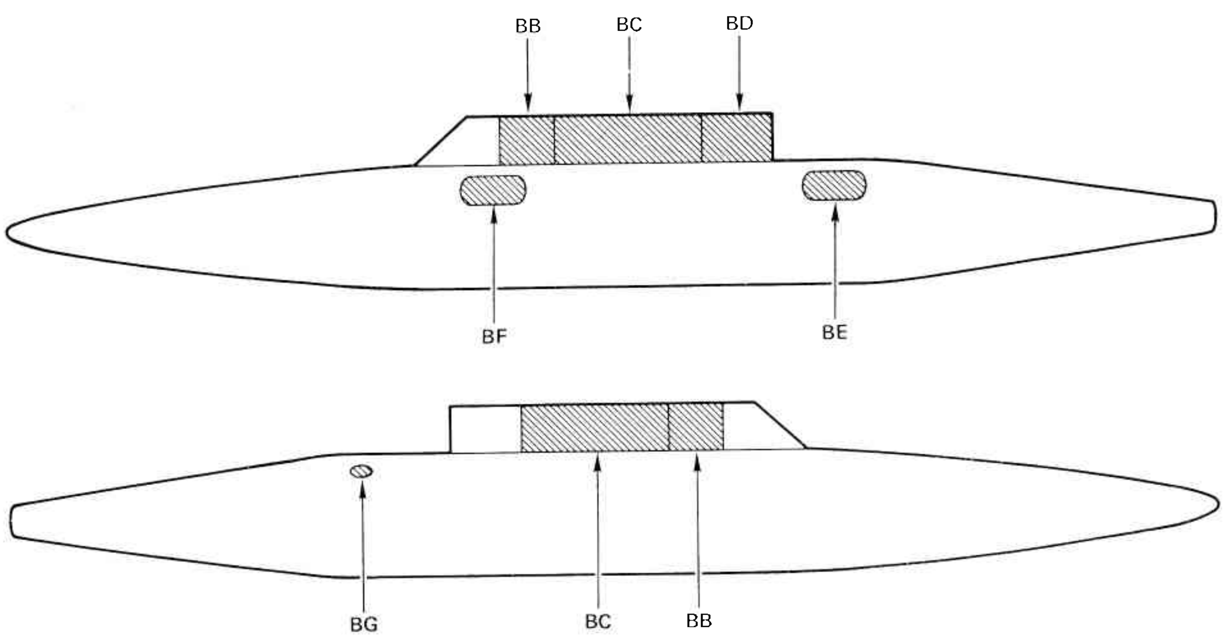
VIEW LOOKING OUTBOARD, RIGHT SIDE

ACCESSIBLE EQUIPMENT

538 AILERON CONTROL
539 AILERON CONTROL. FUEL DRAIN LINE
542 AILERON CONTROL. FUEL DRAIN VALVE
1.006 AILERON CONTROL
AW VARIABLE RAMP, LOWER-AFT ACTUATOR JACK. RUDDER CABLE PULLEY
546 FUEL VENT LINE
548 VENT VALVE SENSE LINE
1.004 AILERON CONTROL
590 AIR CONDITIONING DUCTING, GROUND REFUELING REGULATOR
617 AILERON CONTROL
614 AILERON CONTROL
AX AILERON CONTROL. ELECTRICAL HARNESS
AY AC EMERGENCY CONTROL RELAY. AC EXTERNAL POWER INTERLOCK RELAY. DC EXTERNAL POWER INTERLOCK RELAY. DC POWER FAILURE WARNING RELAY. M.W.W. DOOR CLOSE RELAY. EXTERNAL FUEL TANK EJECTION RELAY. IGNITION RELAY. IGNITION ARMING RELAY. ANTI-ICE CONTROL RELAY. AILERON CONTROL
AZ AIR CONTROL TIMER RELAY CONTROL BOX. FORWARD MISSILES MISFIRE RELAYS. AFT MISSILES MISFIRE RELAYS. MISSILE INTERVALOMETER. AUXILIARY AFT MISFIRE RELAY. SPECIAL WEAPON MISFIRE RELAY. AC GENERATOR CONTROL. AC POWER DISCONNECT RELAY. AC EXTERNAL POWER DISCONNECT RELAY. FIRE DETECTION SYSTEM RELAYS. OVERHEAT FLASHER. OVERHEAT DETECTORS, LOOPS 1 AND 2. HYDRAULIC PRESSURE WARNING FLASHER. HYDRAULIC FLASHER RESET RELAY.
BA AILERON CONTROL

F-106B(2-5-2)5-2A

Figure 1-8. Access and Inspection Provisions, Armament Bay, F-106B (Sheet 2 of 2)



ACCESSIBLE EQUIPMENT

BB FUEL AND AIR LINE DISCONNECT
BC EJECTOR RACK
BD ELECTRICAL DISCONNECT
BE VENT VALVE. REFUELING SHUTOFF VALVE. CHECK VALVE. LOW LEVEL FLOAT SWITCH
BF HIGH LEVEL FLOAT SWITCH. PILOT FLOAT VALVE. VACUUM RELIEF VALVE
BG FUEL FILLER CAP

F-106A(2-5-2)18

Figure 1-9. Access and Inspection Provisions, External Fuel Tank/Pylon

F-106B airplanes have been selected to produce as strong and light a structure as possible, and at the same time meet aerodynamic and production requirements. Consequently, it is very important that any repair return the damaged structural member to its original shape and strength as nearly as possible, and any loss of strength due to change of shape must be compensated for by an increase in cross-sectional area.

Paragraphs 1-16 thru 1-31 and figures 1-10 thru 1-13, deleted.

1-32. Damage Importance Relative to Location.

1-33. Particular attention should be directed to the location of the damage on the member because less damage may be tolerated in certain areas considered critical on each member. When a force is applied which tends to bend a member, stresses are set up in it. The stresses are greatest in the parts of the member which are farthest from its center. For this reason, ribs are built with the greater part of their cross-sectional area at the edges. These reinforced edge sections are called flanges or rails. Because of this concentration of stresses, these flanges or rails are the points where the least damage may be tolerated. Typical rib construction is shown in figure 1-14. Care should be used in the size and location of rivet holes drilled in the flanges, rails and stiffeners. Holes drilled by the manufacturer are the best guide for the mechanic. Remember, a member loses a portion of its cross-sectional area when a hole is drilled in it. If a rivet is properly installed, the strength of the member in compression is largely returned; however, its strength in tension is still reduced. These factors should be considered in evaluating damage and in planning repairs.

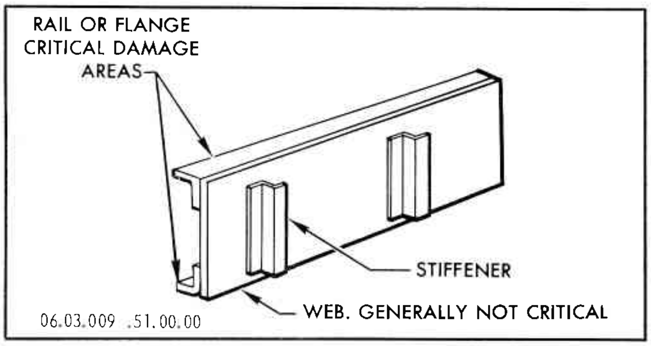


Figure 1-14. Typical Rib Elements

1-34. Weight of Repair.

1-35. To provide maximum operating efficiency, the gross weight of the airplane must be contained within stipulated limits. Therefore, the weight of materials used to repair the existing airplane structure should be held to a minimum consistent with strength requirements. An aircraft structural engineer should be consulted to determine the effect of weight addition through major repairs. In any case, an accurate record must be kept of all weight additions involved in repairs to the aircraft structure. Refer to T.O. 1F-106A-5 and T.O. 1F-106B-5 for airplane center of gravity limits and design gross weights.

1-36. CONTROL SURFACE BALANCING.

1-37. All control surfaces are hydraulically operated and will not require balancing. Repairs to these units shall be designed so as to restore the maximum surface smoothness and trueness of its original contour.

1-38. COCKPIT PRESSURE LEAKAGE, TEST AND REPAIR.

1-39. During normal operation, the cockpit is cooled and pressurized with temperature-regulated engine bleed air that is refrigerated by passing through a refrigeration unit. Cockpit pressure is automatically controlled by a pressure regulator that discharges excess volumes of cockpit air into the nose wheel well. From 31,000 feet to the upper altitude limit of the airplane, cockpit pressure is maintained at a constant differential of 5 psi over ambient atmospheric pressure. It follows that excessive leakage of cockpit air pressure could lower the pilot's maximum altitude limit. Pressurization leakage can also be an important factor in total drag. Even in the low-speed ranges, 300 to 400 miles per hour, pressurization leakage from the outer surface of the fuselage adds to drag. At supersonic speeds, each leak is in effect a subsonic jet projecting from the fuselage, and is as effective as a metallic projection in producing drag. Leaks may be repaired from the inside by stationing a man inside the cockpit while under pressure. Repairing leaks while under pressure is recommended since results may be checked immediately and leaks may be repaired as soon as they are found. After major repairs to the primary structure or replacement of the windshield or canopy panels, the cockpit should be pressure leak tested as outlined in T.O. 1F-106A-2-6. If the flow of air necessary to maintain a constant pressure is excessive and system analysis has isolated the cause to a leak in the structure, check for leaks in accordance with T.O. 1F-106A-2-6-2-1 and as described in the following paragraphs.

1-40. Equipment Requirements.

1-41. The following equipment is required in addition to the equipment listed for the cockpit pressure and leak test in T.O. 1F-106A-2-6-2-1:

a. Stethoscope or rubber tube (listening device to detect leaks.).

b. Bubble fluid (Specification MIL-L-25567).

c. Fillet sealer (MIL-S-8802).

1-42. Deleted.

1-43. Deleted.

1-44. Procedure.

WARNING

Personnel stationed inside the cockpit while it is under pressure should be examined, approved and physiologically indoctrinated (altitude physiology), prior to the test, by medical authority. An experienced operator should be stationed at the test rig control panel at all times while the cabin is under pressure.

a. Station a mechanic, equipped with sealant repair materials and a stethoscope or rubber tube, inside the cockpit. Close and lock canopy.

b. Pressurize cockpit to required leak test pressure, following procedure given in T.O. 1F-106A-2-6-2-1. Rate of pressure change should *never* exceed ½ psi per minute (1000 feet per minute altitude change).

c. With cockpit pressurized, locate leaks on outside of airplane with bubble fluid or castile soap solution. Isolate leaks on inside by running stethoscope or section of rubber tube along seams in floor and bulkhead areas, listening for a whistling sound when tube is moved over a leak.

d. Repair leaks as shown in figures 1-15 and 1-16 until it is no longer possible to detect any change in sound when stethoscope or rubber tube is passed over repair location.

e. Slowly decrease cockpit pressure to zero gage pressure. Maintain rate of pressure change at ½ psi per minute.

f. Open canopy and remove test equipment. Restore air-conditioning and pressurization system to flight condition as directed by procedure in T.O. 1F-106A-2-6-2-1.

1-45. AIRFRAME FABRICATION MATERIALS.

1-46. Titanium.

1-47. The data presented on this relatively new type of material employed in airplane construction provides practical suggestions and limitations to be used in fabricating parts and repairs from titanium sheet. There are three specific reasons for the employment of titanium in structure: High stress and overall strength by weight comparison with other materials, high heat resistance values, and high corrosion resistant properties.

1-48. Types of Titanium.

1-49. The types of titanium most commonly used in the F-106A and F-106B airplanes are:

a. AMS 4901 sheet and strip, replaced by MIL-T-9046, Type I, Comp B.

b. AMS 4908 sheet and strip, replaced by MIL-T-9046, Type III, Comp A.

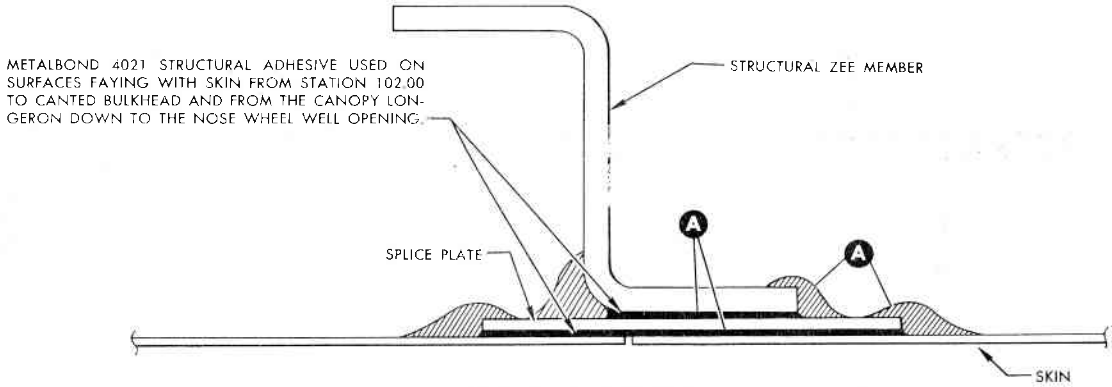
c. Convair Specification 0-01014 sheet and strip, replaced by MIL-T-9046, Type II, Comp A.

d. AMS4925 bar or forging, 4AL-4MN, replace with MIL-T-9047, 6AL-4V.

e. Convair Specification 0-01015 bar or forging, replaced by MIL-T-9047, 5AL-2.5SN.

1-50. Identification of Titanium.

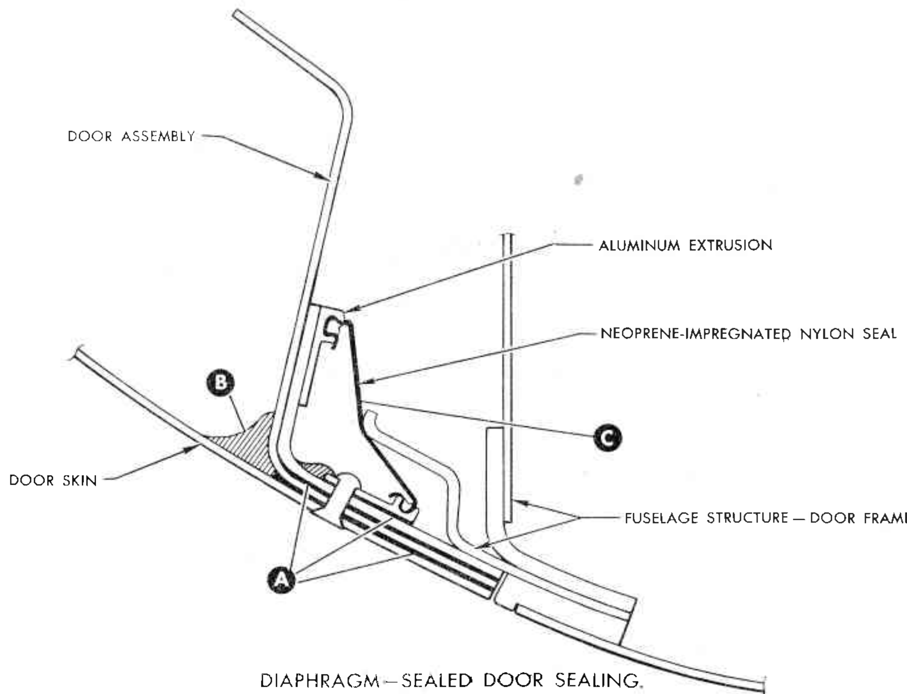
1-51. Titanium can be identified in two ways. Touched on a grinding wheel, it gives off bright white traces ending in brilliant bursts. Titanium may also be identified by applying a drop of 48 percent concentrated hydrofluoric acid. The following procedure should be used when making a chemical test to prevent damage to the titanium part:



COCKPIT SKIN SPLICE SEALING.

A Apply Sealer, Military Specification MIL-S-81733 to faying surfaces of original assembly.

Apply a fillet of sealant, Military Specification MIL-S-8802, along edges of faying surfaces.



DIAPHRAGM—SEALED DOOR SEALING.

A Apply sealer, Militray Specification MIL-S-81733 to faying surfaces of original assembly.

B Repair leaks in original sealant by application of fillet sealer, Military Specification MIL-S-8802.

C Repair leaks in seal as shown on sheet 2 of this illustration.

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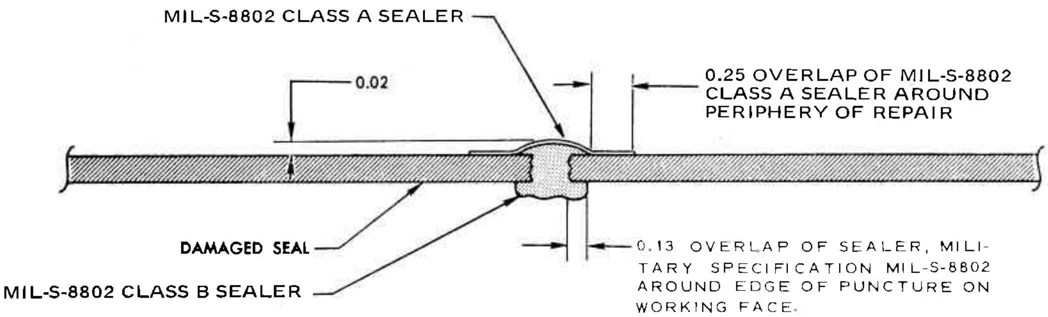
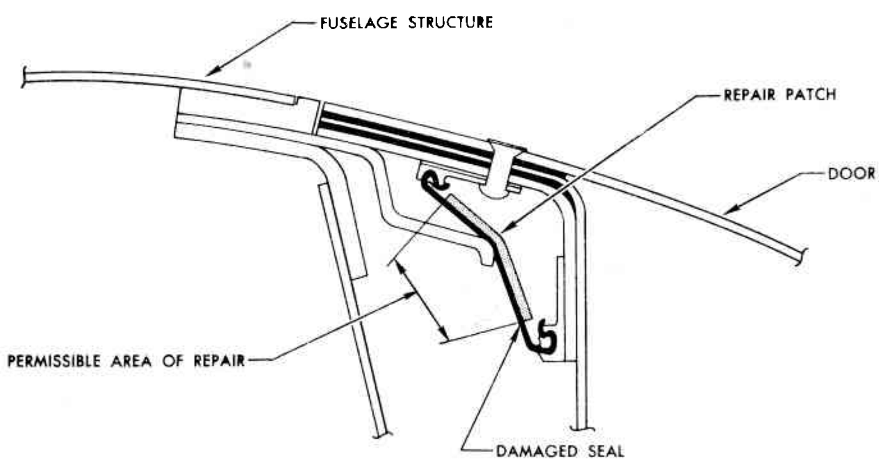
NOTE:
REFER TO PARAGRAPH TITLED "COCKPIT PRESSURE LEAKAGE TEST AND REPAIR" IN THIS SECTION.

Figure 1-15. Cockpit and Forward Electronic Compartment Sealing (Sheet 1 of 3)



TYPE I REPAIR OF DAMAGED DIAPHRAGM SEAL

A Measure puncture. If puncture exceeds 3/32 x 3/16-inch, repair as shown for type II repair below.
B Clean surface of seal with methyl ethyl ketone in a ½ inch wide area around edge of puncture.
C Inject puncture with enough sealer, Military Specification MIL-S-8802 Class B to close opening. Smooth and form sealer with spatula on working face as shown above.
D Allow sealer, Military Specification MIL-S-8802 to cure until firm and track free.
E Apply a coating of MIL-S-8802 Class A sealer over working face of repair as shown above.



TYPE II REPAIR OF DAMAGED DIAPHRAGM SEAL

A Measure puncture. If puncture exceeds ½ inch in length or diameter, replace seal.
B Remove seal and apply Camel Vulcanizing Patch or equivalent to inside of seal as shown. Manufactured by H. B. Egan Mfg. Co., Muskogee, Oklahoma. Apply as directed in manufacturer's instructions. Overlap patch ⅜ inch around edge of puncture.
C Re-install seal.

NOTE:
TYPE I AND II REPAIRS ARE LIMITED TO A MAXIMUM OF SIX FOR ENTIRE DOOR WITH A MINIMUM SPACING OF TWELVE INCHES BETWEEN REPAIRS.

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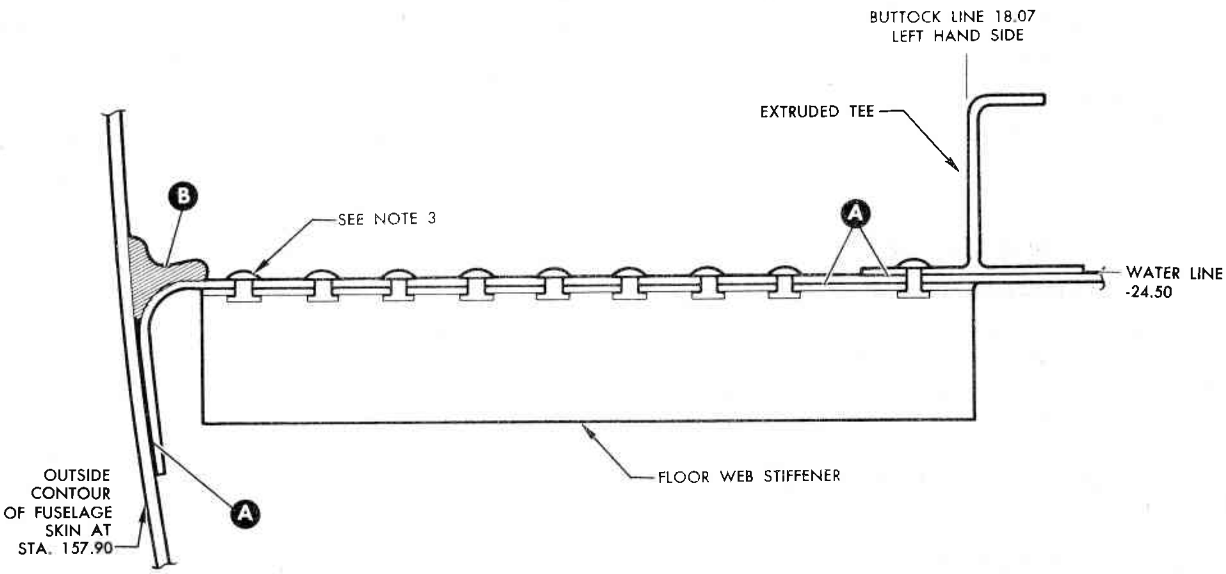
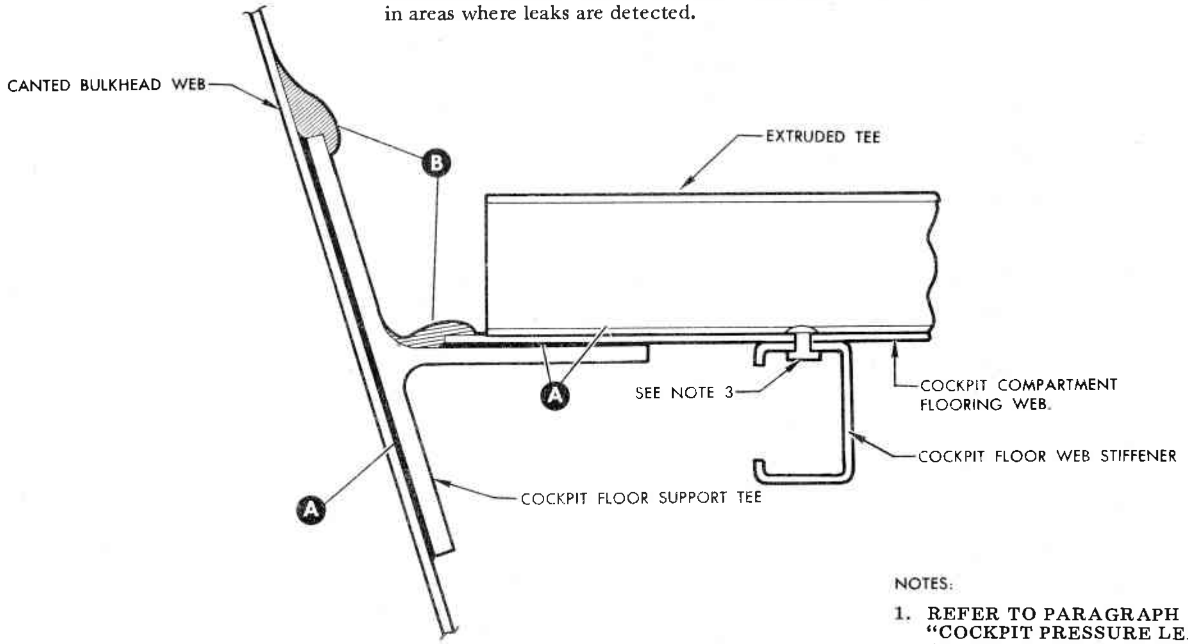
Figure 1-15. Cockpit and Forward Electronic Compartment Sealing (Sheet 2 of 3)



COCKPIT SIDE FLOOR ATTACHMENT

Ⓐ Apply sealant, Military Specification MIL-S-81733, to all faying surfaces when making repairs. (See note 2.)

Ⓑ Apply a fillet of sealant, Military Specification MIL-S-8802, along edges of faying surfaces and in areas where leaks are detected.



COCKPIT AFT FLOOR ATTACHMENT

Ⓐ Apply sealant, Military Specification MIL-S-81733, to faying surfaces when making repairs. (See note 2.)

Ⓑ Apply a fillet of sealant, Military Specification MIL-S-8802, along edges of faying surfaces and in areas where air pressure leaks are detected.

NOTES:

1. REFER TO PARAGRAPH "COCKPIT PRESSURE LEAKAGE TEST AND REPAIR" IN THIS SECTION.
2. METALBOND 4021 STRUCTURAL ADHESIVE TAPE IS USED ON FAYING SURFACES IN ORIGINAL FABRICATION.
3. ALL RIVETS ARE TO BE COATED WITH SEALER, MILITARY SPECIFICATION MIL-S-81733, PRIOR TO INSTALLATION.

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Figure 1-15. Cockpit and Forward Electronic Compartment Sealing (Sheet 3 of 3)

a. Cut a very small piece from a corner or edge of the part to be tested.

b. Place the test piece on a piece of plate glass.

c. Apply one or two drops of 48 percent concentrated hydrofluoric acid, Specification O-H-795, to the test piece.

WARNING

If hydrofluoric acid comes in contact with the skin, immediately rinse the affected area with cold, clear rinse water. If the burning continues, report to the dispensary immediately.

d. If the test part is titanium, a boiling action will take place immediately.

1-52. Corrosion Resistant Steel.

1-53. Types 301 and 302 corrosion resistant steel are used in the ¼ and ½ hard conditions. Their uses are restricted to structural applications requiring high corrosion resistant properties plus toughness and ductility. High tensile strength may be attained by moderate or severe cold working. These materials are nonmagnetic in the annealed condition but become magnetic when cold worked. Gas or arc welding is not advised in hardened materials due to their tendency to crack. These materials should not be used where temperatures exceed 426°C (800°F).

1-54. Types 321 and 347 Corrosion Resistant Steel.

1-55. Types 321 and 347 corrosion resistant steel are intended for applications requiring high heat resistance values as well as high corrosion resistance. They contain good weldability and are completely resistant to weld decay due to intergranular corrosion. Annealing is not required after welding provided welding is accomplished with rod similar to parent material.

1-56. Magnesium.

1-57. AZ31A magnesium sheet possesses a combination of good physical properties and formability. Forming operations should be confined to moderate contours since cracks are likely to occur. Material may be hot-formed at 148.9°C (300°F) at the expense of a slight drop in physical properties. Considerable annealing takes place at temperatures above 148.9°C (300°F). Fair weldability may be attained with helium arc. Type 5056 countersunk rivets are generally used for attachment of magnesium plating.

1-58. Aluminum Alloys.

1-59. The various aluminum alloys used as repair materials include 2014-T, 2024-T, 6061-T, 7075-T, and 7178-T. The two primary materials which constitute the major part of the airplane structure are 7075-T and 7178-T, both clad and bare. 7178-T is a high-strength aluminum alloy which contains physical properties superior to 7075-T and may be used as a substitute material for 7075-T. In some instances, heavier gages of 7075-T may serve as a substitute material for 7178-T. Refer to material substitutions listed in Table 1-1.

1-60. Plastics, Fiberglass Reinforced.

1-61. Fiberglass reinforced plastics are a new engineering material with specific properties that are unique and cannot be realized by the use of any other available material. Fiberglass reinforced plastics are employed in the F-106A and the F-106B to utilize the following properties: relatively low interference with radio and radar transmission; good electrical insulation; light weight; high strength-weight ratio; high chemical resistance; low cost of fabrication; and formability to complex shapes. Parts such as the radome, radio antenna housings, pneumatic system compressed air storage bottles, rigid plastic air ducts, cockpit trim panels, and a portion of the pilot's glare shield are constructed of fiberglass reinforced plastic. The fabrication of these parts is based upon glass fibers in loosely matted form, or woven fiberglass cloth, combined with a resin, usually belonging to an organic chemical group known as polyesters or alkyds. Certain applications may require use of phenolic, epoxide, or silicone resins, but the polyester resins are most general in use. Prior to use, these resins are relatively thick, syrupy liquids capable of flowing between the glass fibers and forming to the shape of the mold. When properly treated with agents known as catalysts the resins change, through a process called polymerization, from liquids to hard infusible solids. Parts thus formed are then trimmed to the necessary shape and fastened to the airplane with screws, bolts, or rivets passed through holes drilled in the finished part and the attaching structure. Repairs to damaged fiberglass reinforced plastics must be planned with the original design purposes in mind. For example, the pilot's glare shield is designed to be rigid enough to retain its shape when subjected to G loads during airplane turning movements, but it must be flexible enough to bend away from the pilot's knees and feet during an ejection. Consequently, repairs that would tend to stiffen or reduce the flexibility of the glare shield are not advisable. Fiberglass reinforced plastic parts that carry a structural load or have an exposed surface which is a portion of the airplane exterior are classified as structural plastics. Field repairs are not recommended for the compressed air storage bottles. Minor and major repairs may be made to the structural plastic components by following the repair procedures outlined in the applicable sections of this manual. Refer to paragraph 1-197 for repairs to the rain erosion protective coating on plastic exterior surfaces, and to paragraph 1-199 for general information concerning fiberglass laminate skins.

1-62. Material Substitutions.

1-63. When repairs or replacement of parts are required, a duplication of the original materials should be used whenever possible. When duplicate materials are not available, refer to Table 1-I for substitution materials.

1-64. CATEGORIES OF REPAIR CAPABILITIES.

1-65. Repair capabilities are divided into two levels, advanced base level and depot level. In most instances, the repairs shown in this manual are for use by the advanced base level organization. However, these repairs have not been identified as either advanced base or depot level because the repair level requirements can best be determined by surveying all the factors affecting the decision, such as the total damage, the complexity of the required repairs, and the local facilities available for making the repair. If a survey of a damaged airplane indicates the need for special support and alignment tools, a large quantity of machined repair parts, an excessive number of manhours, or the services of structural engineering personnel not available at the advanced base, the repair should be considered in the category of depot level capability. Although the depot level capability is much greater than the advanced base level due to better repair facilities, larger stock of repair materials, and a large pool of skilled personnel, the repair may better be accomplished in some cases by transporting the necessary facilities and personnel to the airplane. The base commander may arrange for depot level assistance, as outlined in Area Support Maintenance Assistance procedures, if the advance base level repair capabilities are not adequate for timely repair of the damaged airplane.

1-66. CLASSIFICATION OF DAMAGE AND TYPES OF REPAIR.

1-67. All damage is divided into five standard classes:

a. Negligible damage.

b. Damage reparable by patching.

c. Damage reparable by insertion.

d. Damage necessitating the replacement of parts.

e. Damage requiring repair exceeding limits specified in this manual shall be designed by an authorized aeronautical structures engineer.

1-68. Negligible Damage.

1-69. The negligible damage classification consists of any minor damage that does not affect the structural requirements of the component involved, or damage that can be corrected by a simple procedure without placing restrictions on flight operations. Listed below are minor damages which may be classified as negligible, depending on location and dimensions. To determine the proper category for each of these damages and the structural members of the airplane, see figures 1-17 thru 1-21 and refer to the negligible damage tables in Sections II, III, IV, V and VI.

a. Dents. Minor dents with no damage to structure and which have no adverse effect on operations may be permitted to remain. Heavier dents that show evidence of stretching or wrinkling of the metal should be treated as a repair.

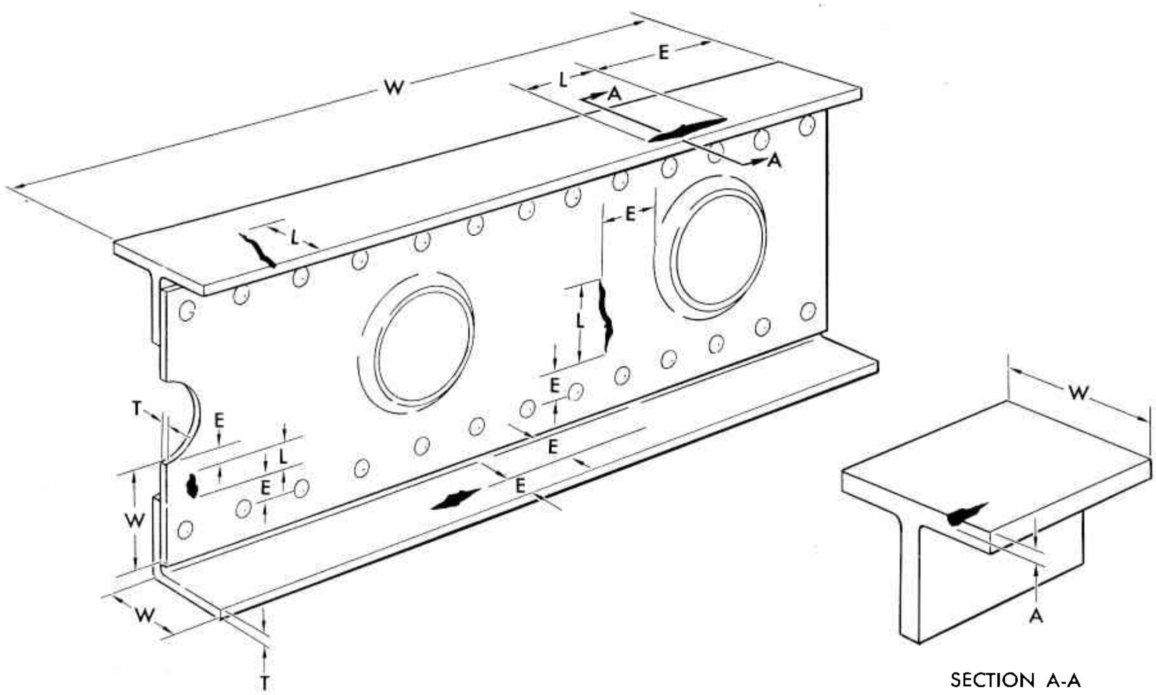
b. Bends. Structural members that have been slightly bent in connection with other more serious damage and which can be fully restored to their original configuration without cracks or wrinkles need not be replaced. 2024-T3 and 2024-T4 materials usually can be bumped out or straightened with good results. In other types of materials, extreme care should be exercised to insure that no cracks remain undetected. Parts which cannot be fully restored must be repaired or replaced.

c. Cracks. This type of damage usually originates at edges, holes, radii, or points where concentrated loads are applied or abrupt changes occur in the cross-sectional area. This type of damage shall not be considered negligible without taking retarding measures. All cracks must be stop-drilled at the ends, and inspection of the area should be made at frequent intervals thereafter to determine the condition of the component involved. See figure 1-20 for information concerning stop-drilling.

d. Holes. Holes which reduce the cross-sectional area of a structural member creating areas of elevated stress, and which interfere with the function of a component, including pressurized or fuel-tight areas, must be repaired. Sharp nicks or cracks projecting into the material from holes must be stop-drilled, rounded or faired into the hole, as shown in figure 1-20. Holes in outer plating of the airplane will be limited to dimensions given in the applicable sections of this manual.

e. Scratches and Nicks. Scratches or nicks which do not exceed the depth of the alclad surface of a part are considered permissible. To determine if alclad has been penetrated, thoroughly clean the surface with methyl-ethyl-ketone, Specification TT-M-261. Apply a 10 percent solution of sodium hydroxide, Specification O-S-60S, to the scratch. If the scratch penetrated the alclad it will be indicated by a black or dark brown discoloration. The solution should not remain longer than two minutes, and due to its corrosive action should be thoroughly washed from the area with water. Scratches and nicks should be burnished and treated with a protective chemical film before priming and painting. Refer to paragraphs 1-188 and 1-190 and figures 1-20 and 1-21 for procedures. Deep scratches may reduce the cross-sectional area of a member and produce localized stress concentrations. This condition can lead to fatigue cracks and possible failure of the component involved.

f. Corrosion. Corrosion damage which does not exceed the prescribed limits for scratches and nicks may be considered as negligible; however, the existing products of corrosion should be removed in accordance with T.O. 1-1-2 and part should be refinished as outlined in paragraphs 1-188 and 1-190.



CODE IDENTIFICATION

A. Depth of nick or scratch.
E. Distance of nick or scratch from an edge, hole, rivet or radius.
C. Cross-sectional area or T x W.
L. Length of nick or scratch.
N. Cross-sectional loss in area due to damage.
T. Gage of part at point of damage.
W. Width of part at point of damage.

NICK DAMAGE CLASSIFICATIONS

KEY	CLASS I	CLASS II	CLASS III
A	0.10 X T OR 0.010 INCH (USE LOWER VALUE)	0.25 X T OR 0.025 INCH (USE LOWER VALUE)	0.50 X T OR 0.050 INCH (USE LOWER VALUE)
E	2 X L OR 0.25 INCH (USE HIGHER VALUE)	2 X L OR 0.25 INCH (USE HIGHER VALUE)	2 X L OR 0.375 INCH (USE HIGHER VALUE)
N	0.02 X C	0.05 X C	0.07 X C

SCRATCH DAMAGE CLASSIFICATIONS

KEY	CLASS I	CLASS II	CLASS III
A	0.05 X T OR 0.005 INCH (USE LOWER VALUE)	0.10 X T OR 0.010 INCH (USE LOWER VALUE)	0.20 X T OR 0.025 INCH (USE LOWER VALUE)
E	0.25 INCH	0.25 INCH	0.375 INCH
L	0.5 X W OR 3.0 INCH (USE LOWER VALUE)	0.5 X W 3.0 INCH (USE LOWER VALUE)	0.25 X W OR 3.0 INCH (USE LOWER VALUE)
N	0.02 X C	0.05 X C	0.07 X C

NOTES:
1. "N" SHALL INCLUDE METAL REMOVED IN RADIUSING OUT NICK.
2. WHEN A NICK OR SCRATCH OCCURS AT AN EDGE, HOLE, OR RADIUS, REDUCE "N" VALUES BY 50 PERCENT.
3. DAMAGES IN EXCESS OF NEGLIGIBLE LIMITS SHOWN MUST BE TREATED AS REPAIRS.
4. WHEN "L" IS GREATER THAN 4 X "A," DAMAGE SHALL BE TREATED AS A SCRATCH.

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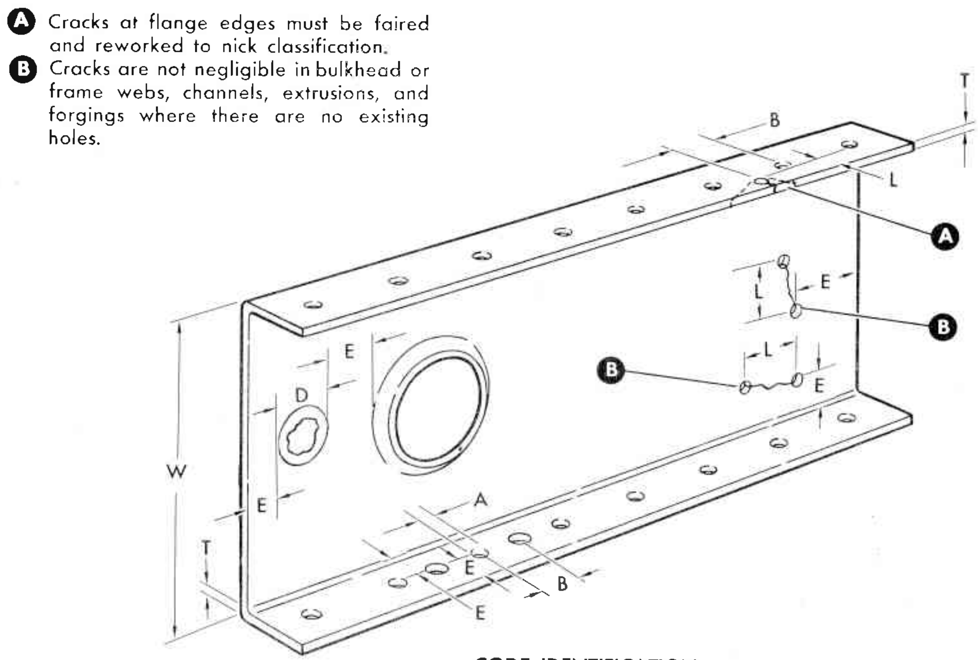
Figure 1-17. Negligible Damage Classification—Nick and Scratch



CODE IDENTIFICATION

A. Diameter of largest existing rivet or bolt hole.
B. Distance of hole or crack from existing hole.
C. Cross-sectional area or T x W.
D. Diameter of hole.
E. Distance of hole or crack from edge, rivet or radius.
L. Length of crack.
N. Cross-sectional loss in area due to damage.
T. Gage of part at point of damage.
W. Width of part at point of damage.

HOLE DAMAGE CLASSIFICATION

KEY	CLASS I	CLASS II	CLASS III
D	1.0 X A OR 0.25 INCH (USE LOWER VALUE)	2.0 X A OR 0.50 INCH (USE LOWER VALUE)	4.0 X A OR 1.00 INCH (USE LOWER VALUE)
B	4.0 X D	4.0 X D	4.0 X D
E	2.0 X D	4.0 X D	4.0 X D
N	0.02 X C	0.04 X C	0.07 X C

CRACK DAMAGE CLASSIFICATION

KEY	CLASS I	CLASS II	CLASS III
L	1.0 X A OR 0.25 INCH (USE LOWER VALUE)	2.0 X A OR 0.50 INCH (USE LOWER VALUE)	4.0 X A OR 1.00 INCH (USE LOWER VALUE)
B	4.0 X L	4.0 X L	4.0 X L
E	2.0 X L	4.0 X L	4.0 X L
N	0.02 X C	0.04 X C	0.07 X C

NOTES:
1. "N" SHALL INCLUDE METAL REMOVED BY STOP-DRILLING CRACKS.
2. WHEN CRACKS OR HOLES OCCUR AT AN EDGE, HOLE OR RADIUS, REDUCE "N" VALUE BY 50 PERCENT.
3. DAMAGES IN EXCESS OF NEGLIGIBLE LIMITS SHOWN MUST BE TREATED AS REPAIRS.

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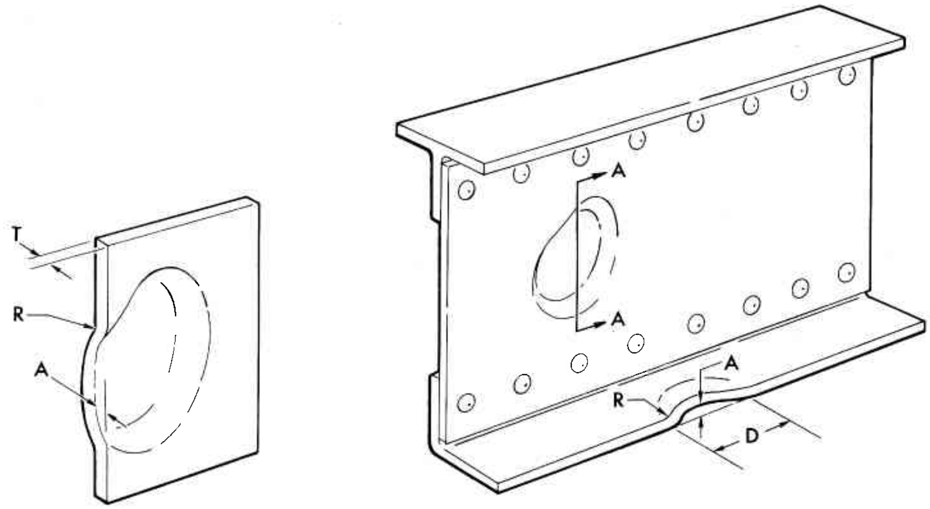
Figure 1-18. Negligible Damage Classification—Hole and Crack



CODE IDENTIFICATION

A. Deflection from contour (maximum).
D. Diameter of dent (smaller direction).
R. Radius of bend at worst point.
T. Gage of part at point of damage.

DENT DAMAGE CLASSIFICATION

KEY	*CLASS I	*CLASS II	**CLASS III
A	2 X T OR 0.050 INCH (USE LOWER VALUE)	4 X T OR 0.10 INCH (USE LOWER VALUE)	10 X T OR 0.25 INCH (USE LOWER VALUE)
D	20 X A MINIMUM	10 X A MINIMUM	5 X A MINIMUM
R	10 X T MINIMUM	5 X T MINIMUM	2 X T MINIMUM

NOTES:
1. * INDICATES DENTS WITH NO VISIBLE STRETCHING OF METAL, AND WHICH CAUSE NO INTERFERENCE WITH OPERATIONS.
   ** INDICATES DENTS WITH SOME EVIDENCE OF METAL STRETCHING, CREASING AND WRINKLING BUT WITHOUT CRACKS OR TEARS AND WHICH CAUSE NO INTERFERENCE WITH OPERATIONS.
2. DENTS IN 2024-T3, T-4, AND T-6 AND TITANIUM SKINS AND WEBS WHICH CAN BE BUMPED OUT TO MEET GIVEN DIMENSIONS MAY BE CLASSIFIED AFTER REWORK.
3. DENTS IN 7075-T6 OR MAGNESIUM ALLOY MATERIAL (FORGINGS AND CASTINGS INCLUDED) SHALL NOT BE BUMPED OUT AND MUST BE TREATED AS REPAIRS WHEN IN EXCESS OF DIMENSIONS GIVEN.

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Figure 1-19. Negligible Damage Classification—Dent

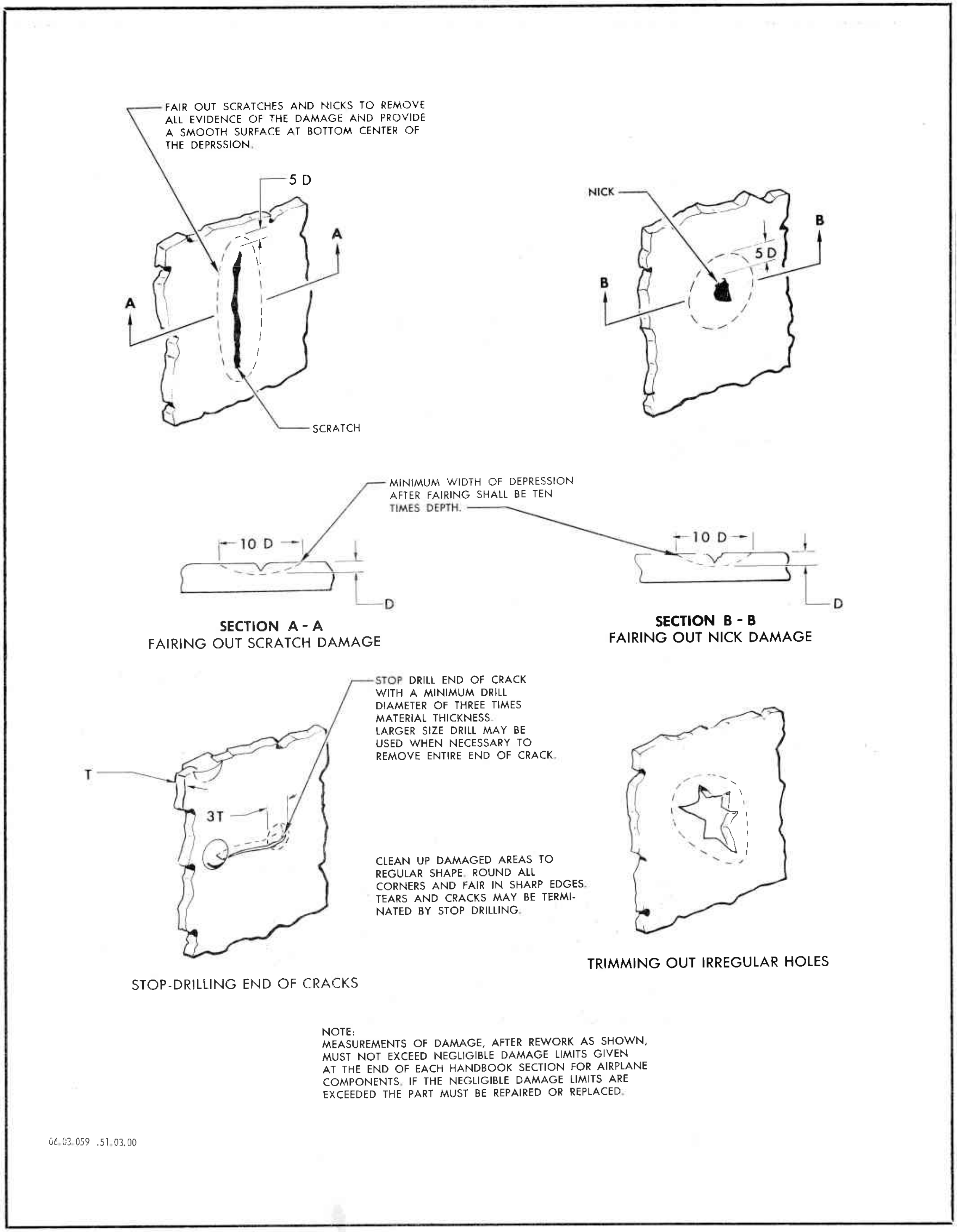
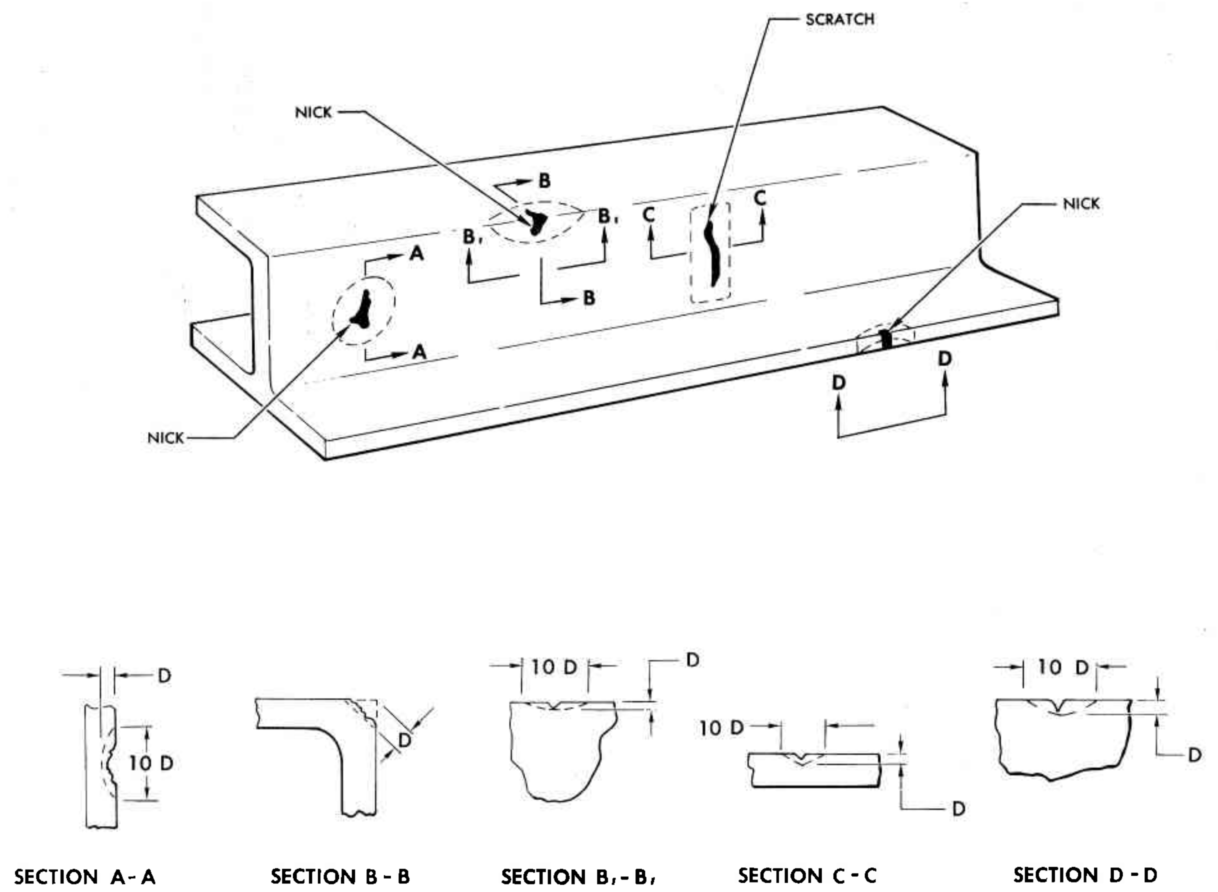


Figure 1-20. Fairing of Minor Damage in Plating



NOTES:
1. FAIR ALL NICKS AND SCRATCHES IN FITTINGS, INCLUDING EXTRUDED AND ROLL-FORMED STRUCTURAL MEMBERS BY REMOVING METAL AS INDICATED BY DOTTED LINES. REMOVE ALL EVIDENCE OF DAMAGE BY FAIRING OUT AN EQUAL AMOUNT AT EACH SIDE OF DEPRESSION (SEE SECTION A-A).
2. DEEP NICKS WHICH PENETRATE MORE THAN 50 PER CENT OF THE THICKNESS OF A STRUCTURAL MEMBER MAY BE FAIRED BY DRILLING A HOLE THROUGH THE MEMBER TO REMOVE SHARP CORNERS. THIS METHOD MAY BE EMPLOYED ONLY WHEN EDGE DISTANCE PERMITS AND HOLE DIAMETER IS CLASSIFIED AS NEGLIGIBLE.
3. MEASUREMENTS OF DAMAGE, AFTER REWORK AS SHOWN, MUST NOT EXCEED NEGLIGIBLE DAMAGE LIMITS GIVEN AT THE END OF EACH HANDBOOK SECTION FOR AIRPLANE COMPONENTS. IF THE NEGLIGIBLE DAMAGE LIMITS ARE EXCEEDED, THE PART MUST BE REPAIRED OR REPLACED.

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Figure 1-21. Fairing of Minor Damage in Fittings

1-70. Damage Repairable by Patching.

1-71. Damage repairable by patching is any damage that can be repaired by bridging the damaged area of a component with splice material of the same type as the original component. Holes and cracks in any of the various structural components that exceed the prescribed limits for negligible damage may be repaired by this method. Filler plates may be used for bearing purposes or for returning the part to its original contour when necessary. See figure 1-22 for formula and method for patch repair. The method of attaching the patch to the undamaged portion of the original structure is very important in developing the strength of the repair, which must be equal to or greater than the strength of the original structure to withstand stresses up to the maximum values for which it was designed.

1-72. Damage Repairable by Insertion.

1-73. Damage repairable by insertion is any damage that can be repaired by splicing in a section of material identical in shape and type to the damaged part. When a portion of a structural component has been severed or otherwise mutilated, not exceeding 50 percent of its total area, the repair requires cutting out the damaged portion and inserting the repair material into the space resulting from the removal of damaged portion. The inserted material is riveted in place by means of splice material of the same type, but of the next heavier gage or a suitable substitute, bridging both sides of the damaged area. The splice connections to the original structure at each end of the insertion provide for load transfer continuity between existing structure and the inserted part. This type of repair is generally used where the damaged part is relatively long or when interference of other structural components can be avoided. See figure 1-23 for formula and method for insertion repair. The method of attaching the inserted part through splice members attached to the undamaged portion of the original structure is highly important since the repair must restore the full load-carrying characteristics of the structure to the maximum values for which it was designed.

1-74. Damage Necessitating Replacement of Parts.

1-75. The type of repair necessitating a replacement of parts is generally made when a component cannot be economically repaired on the airplane or when the "down time" of the airplane can be appreciably reduced by replacing rather than repairing a part. Small nonstructural attaching parts such as clips, angles, gussets, etc., for which no stock level has been established, should be locally manufactured, using the damaged part as a pattern. When a damaged component is to be replaced it should be removed carefully to avoid damage to adjacent or attaching structure. Damage to forged or cast fittings beyond the prescribed limits for negligible damage will require the replacement of the fittings unless otherwise advised by structural engineering authorities. In all cases requiring replacement, it is preferable to use a new part from spare stock; however, when such parts are not available, a new part may be fabricated from the same gage and type of material as the original. Dimensions should be taken from the damaged part, if possible, or from the same part on another airplane of the same series. Extrusions, when not available, may be replaced by equivalent sections fabricated from flat sheet stock or may be machined from bar stock. When the same gage and type of material used in the original design is not available, refer to Table 1-I for a substitute. The original rivet pattern should always be used when possible; however, should the extent of the damage prohibit this procedure, Tables 1-II through 1-XVII and paragraph 1-120 should be consulted. In all damage cases requiring the replacement of parts, the materials and methods of attachment should restore the structure to its original strength.

1-76. Combined Repairs.

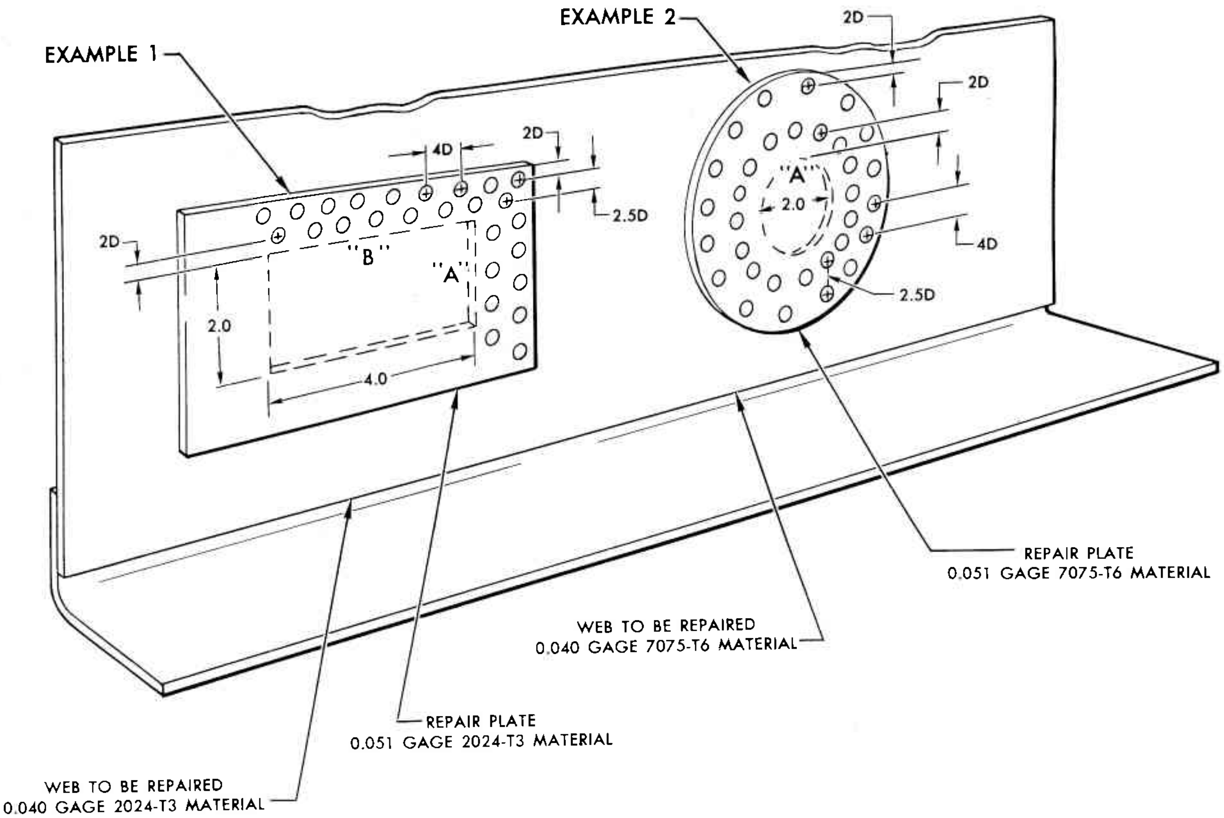
1-77. Since most damage is likely to involve two or more structural components, it follows that most repairs will be combination repairs involving one or more structural components and the skin. In this case the damage may require both the patch and insertion type of repair. See figure 1-24 for the formula and method for a combination repair. Special care must be exercised in making this type of repair since the skin loads and stresses must be transmitted through the internal structure while retaining its structural continuity. The combined repair must be designed to restore the strength of the repaired structure equal to or greater than the original structure.

1-78. AERODYNAMIC SMOOTHNESS.

1-79. Skin Gap and Rivet Protrusion Limitations.

1-80. When making repairs to the exterior surface of the airplane, it is essential that the aerodynamic contour shall be maintained within certain limitations. For data concerning tolerances permissible in skin gaps and the protrusion of rivet heads, see figure 1-25. Surface discontinuities will have an adverse effect on performance; therefore, it is necessary to maintain contours as close to the designed configuration as possible. To maintain aerodynamic smoothness between skin splices and along fairing angles, apply aerodynamic smoothing compound as shown on figure 1-27. In general, the outer surface is most critical aerodynamically in the forward section of the airplane and becomes progressively less critical toward the after section. If large areas of skin are replaced, waviness or bulges must not exceed the following limits:

a. Fuselage area, forward of duct inlet: Contour deviations at points of skin attachment shall not exceed ± 0.030 inch. Height of wave between skin attachment points shall not exceed 0.030 inch or 1 percent of wave length, whichever is smaller.



EXAMPLE 1

RECTANGULAR PATCH REPAIR

A. Identify material to be repaired, which in this case is 0.040 gage 2024-T3.
B. Use same material of next heavier gage for repair plate. For suitable substitute see Table 1-I.
C. To select rivets required to make the repair, refer to Table 1-II and note the various sizes of rivets required per inch of seam opposite 0.040 in the gage column applicable to 2024-T3 material.
D. Assuming that AD5 rivets are best suited to the conditions of the repair, multiply the dimensions of one side and one end of the repair by 3.5 (the number of rivets required per inch of seam) to determine the number of rivets required for each end and sides of the damaged area "A" and "B".

"A" $2 \times 3.5 = 7.0$ Use 7 rivets
"B" $4 \times 3.5 = 14.0$ Use 14 rivets

21 rivets required for one side and one end of the repair.

E. For complete repair, multiply above product by 2 and continue same rivet pattern in corners of repair.

EXAMPLE 2

CIRCULAR PATCH REPAIR

A. Identify material to be repaired, which in this case is 0.040 gage 7075-T6.
B. Use same material of next heavier gage for repair plate. For suitable substitute see Table 1-I.
C. To select rivets required to make the repair, refer to Table 1-II and note the various sizes of rivets required per inch of seam opposite 0.040 in gage column applicable to 7075-T6 material.
D. Assuming that AD5 rivets are best suited to the conditions of the repair, multiply the diameter of the repair by 4 and multiply the result of this figure by 4.1 (the number of rivets required per inch of seam) to determine the number of rivets required for the complete repair.

"A" $2 \times 4 = 8 \times 4.1 = 32.8$ or 33 rivets required

NOTES:
1. WHEN NUMBER OF RIVETS REQUIRED EXCEED THE NUMBER ALLOWED IN 2 ROWS, IT WILL BE NECESSARY TO MAKE 3 ROWS.
2. RIVET PATTERNS NOT SHOWN IN THIS ILLUSTRATION THAT EXCEED 2 ROWS SHALL MEET STANDARD MINIMUM RIVET SPACING REQUIREMENTS.
3. SEE RIVET NOMOGRAPH IN THIS SECTION FOR QUICK REFERENCE IN PLANNING REPAIRS.

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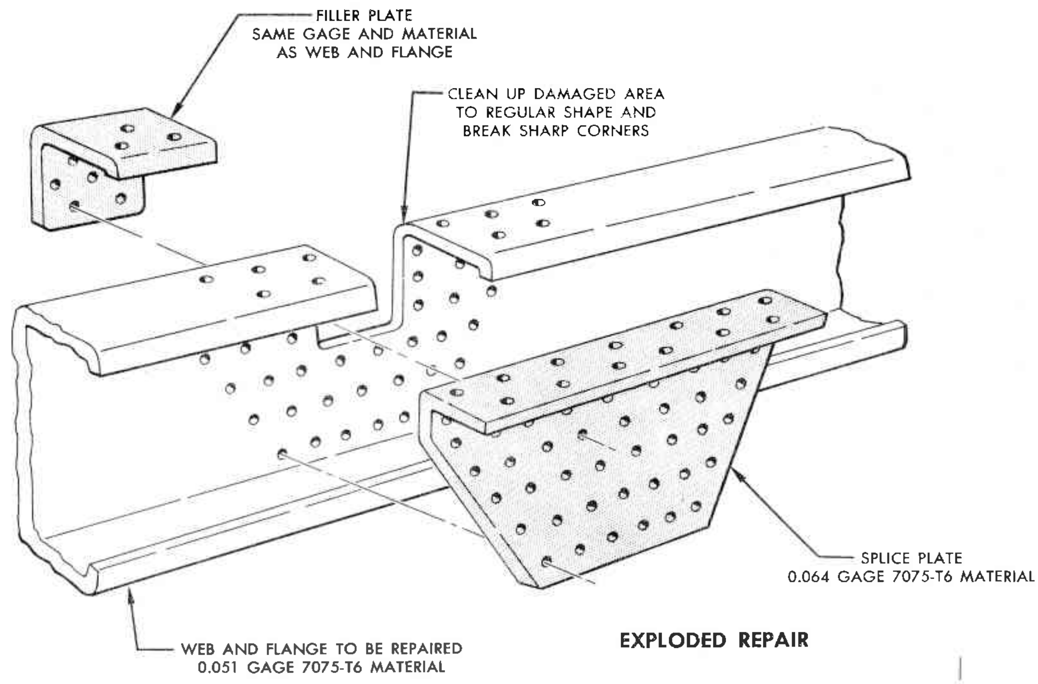
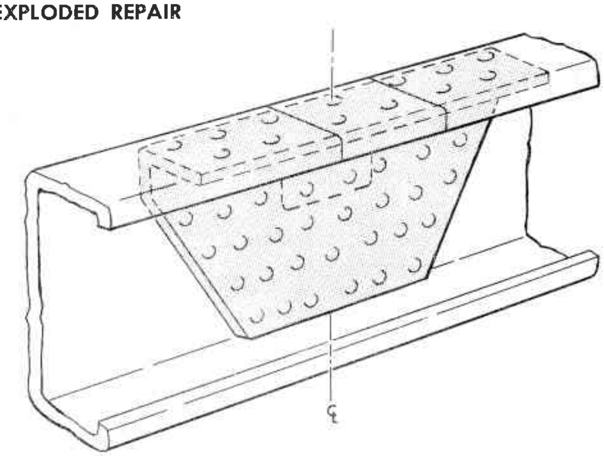
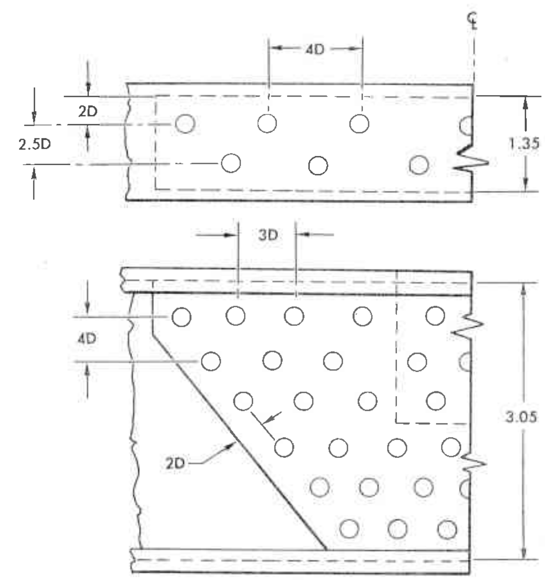
Figure 1-22. Formula and Method for Patch Repair



EXPLODED REPAIR





VIEW A

RIVET PATTERN TO CENTERLINE OF REPAIR – OPPOSITE SIDE IDENTICAL

EXAMPLE

USE FOLLOWING STEPS TO DETERMINE METHOD OF REPAIR.

A. Identify material to be repaired which in this case is 0.051 gage 7075-T6.

B. Use same material and gage for filler plate but next heavier gage for splice plate. For suitable substitute see Table I-I.

C. To select rivets required, refer to Table I-II and note various sizes of rivets required per inch of seam opposite 0.051 in the gage column applicable to 7075-T6 material.

D. Assuming that the AD6 rivets are best suited for the flange repair, multiply the width of the flange by 3.7 (the number of rivets required per inch of seam) to determine the number of rivets required for each side of the damaged area.
$1.35 \times 3.7 = 4.99$ Use 5 rivets each side.

E. For web area use same procedure as for flange repair except assume AD5 rivets best suited for the repair.
$3.05 \times 5.4 = 16.4$ Use 16 rivets each side.

F. Continue same rivet pattern through filler plate. Rivet pattern shall meet standard minimum rivet spacing requirements.

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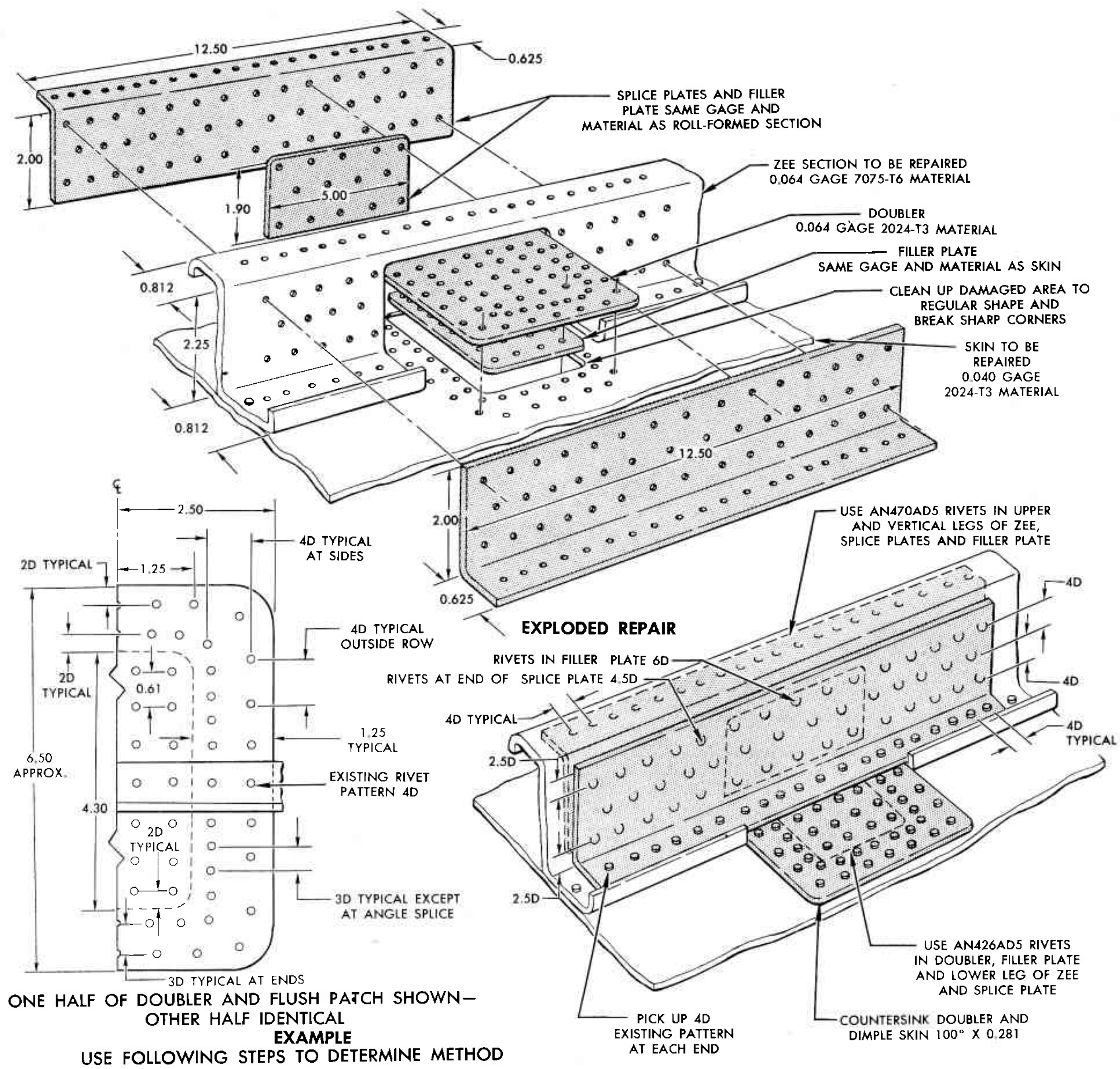
Figure 1-23. Formula and Method for Insertion Repair



EXAMPLE

USE FOLLOWING STEPS TO DETERMINE METHOD OF REPAIR

A. Identify material to be repaired and select repair materials as shown. For suitable substitute see Table I-I.

B. To select rivets required for skin repair refer to Table I-II and note various sizes of rivets required per inch of seam opposite 0.040 in the gage column applicable to 2024-T3 material using 100° countersunk rivets in dimpled skin.

C. Assuming that AN426AD5 rivets are used in this area, multiply the width and the length of the skin repair by 3.5, to determine the number of rivets required.
2.50 x 3.5 = 8.750 Use 9 rivets on each end.
4.30 x 3.5 = 15.05 Use 15 rivets on each side.

Note that standard minimum rivet spacing, 4D rivet pitch, will not permit this number of rivets and 3D is used on inner row. Minimum row of spacing of 2.50 was not used due to existing rivet pattern of 4D rivet as shown. Flush patch is not a load carrying member and a loose rivet pattern may be used if held within prescribed limits.

D. For zee section use same procedure as for skin except use rivets required for 0.064 gage 7075-T6 material.

E. Assuming that AN470AD5 rivets are best suited for the repair, multiply widths of zee section legs 0.812 x 6.7 = 5.44. Pick up 6 existing AN426AD5 rivets and use same pattern with AN470AD5 rivets in upper leg of zee section. 2.25 x 6.7 = 15.07. Use 15 rivets on each side of repair in vertical leg and use 4.5D rivet pitch to fill out rivet pattern to end of splice as shown. D6 may be used in filler plate as its primary purpose is to serve as filler only.

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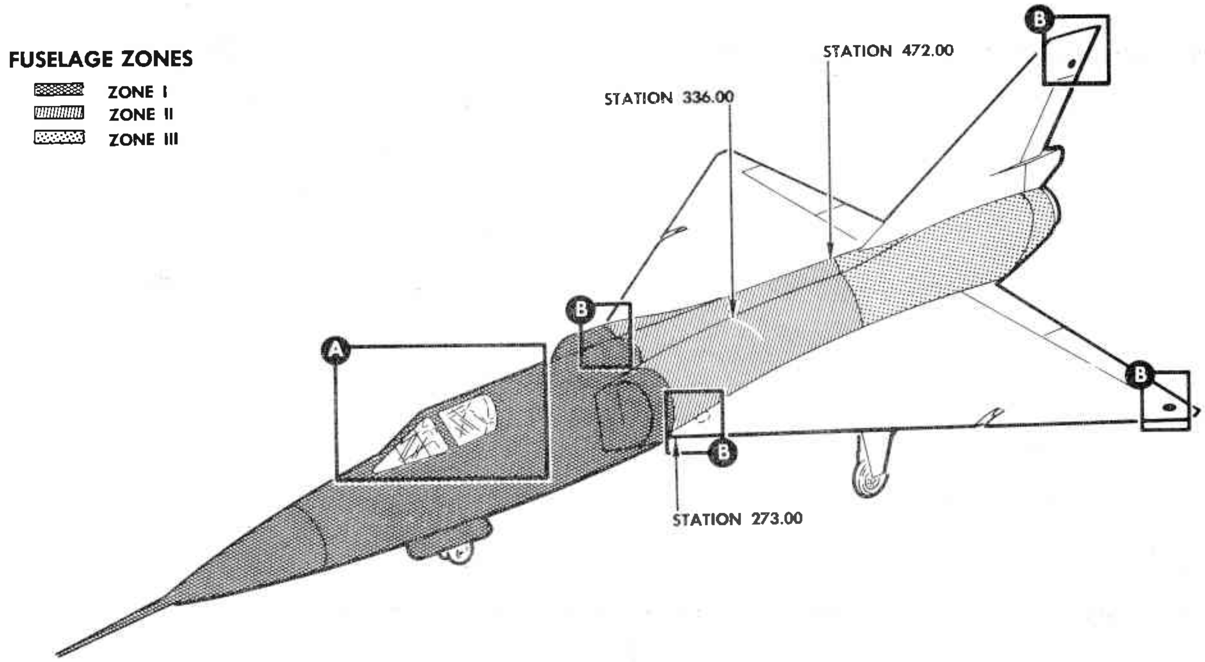
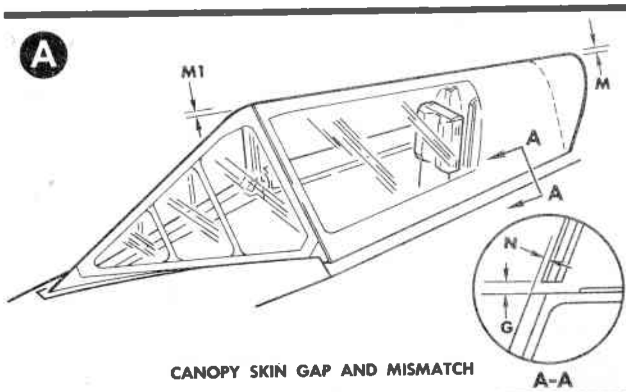
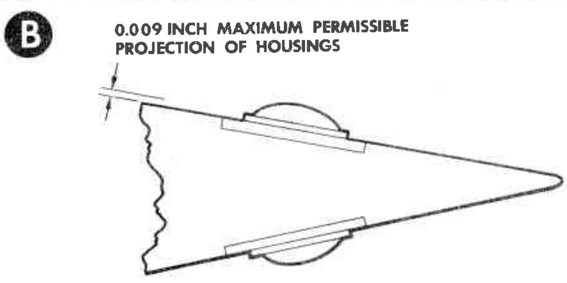
Figure 1-24. Formula and Method for Combination Repair



NOTE
REFER TO SHEET 3 OF THIS ILLUSTRATION FOR SYMBOL DEFINITION AND GAP AND MISMATCH TOLERANCES.



CANOPY SKIN GAP AND MISMATCH



TYPICAL CLEARANCE FOR ALL RECOGNITION LIGHTS

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RIVET HEAD PROTRUSION TOLERANCES	
AREA	MAXIMUM HEAD HEIGHT
INSIDE ENGINE AIR INLET DUCT, FROM INLET LIP AFT TO STA 336, FLAT SURFACE, TOP AND BOTTOM.	FLUSH TO 0.004 INCH BELOW SURFACE
FUSELAGE ZONE I.	0.002 INCH
WING LEADING EDGE.	
FIN LEADING EDGE.	
INSIDE ENGINE AIR INLET DUCT, FROM WEDGE LEADING EDGE AFT TO STA 336, FLAT SURFACE, SIDE.	
FUSELAGE ZONE II AND III.	0.004 INCH
FIN AREA, AFT OF LEADING EDGE.	
WING AREA, AFT OF LEADING EDGE.	
INSIDE ENGINE AIR INLET DUCT, AFT OF STA 336, FLAT SURFACE, SIDE.	
INSIDE ENGINE AIR INLET DUCT, FROM INLET LIP AFT TO STA 336, CURVED SURFACE, SIDE, NO MILLING PERMITTED.	
INSIDE ENGINE AIR INLET DUCT, FROM STA 336 AFT TO STA 472, CURVED SURFACE, SIDE, NO MILLING PERMITTED.	0.008 INCH

Figure 1-25. Aerodynamic Smoothness Requirements (Sheet 1 of 3)

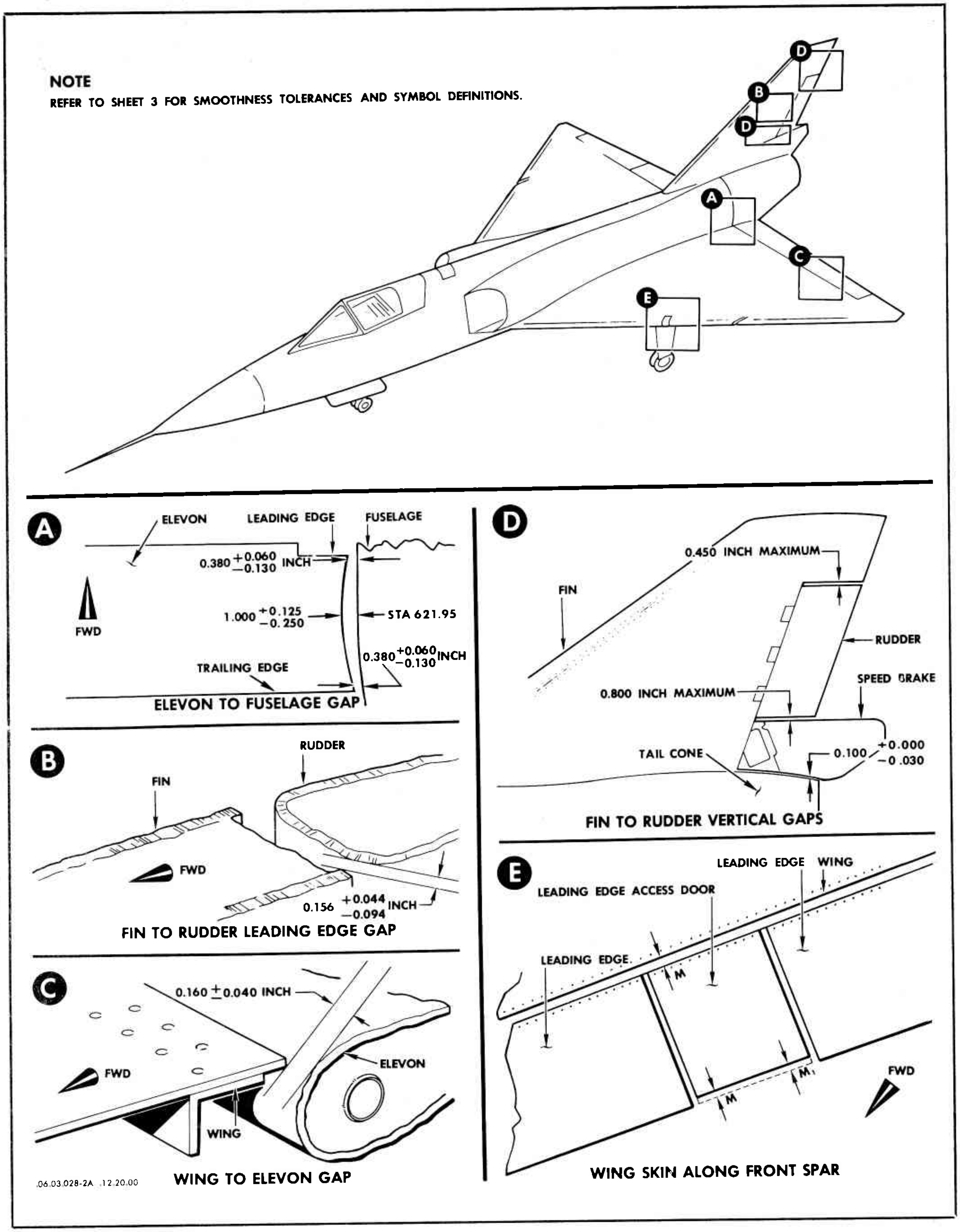
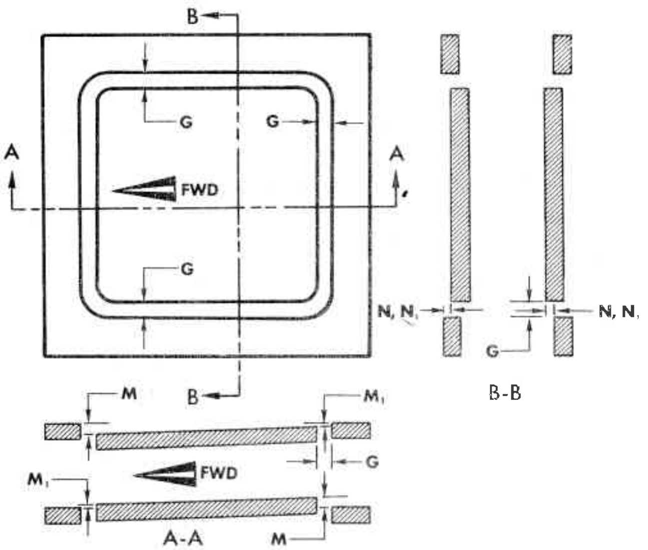
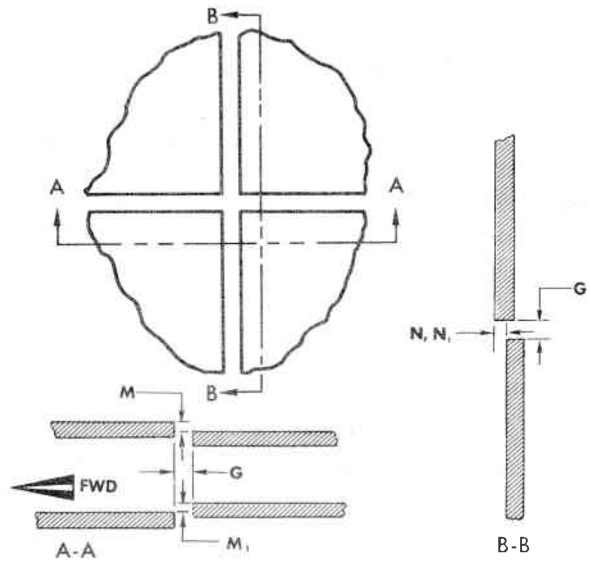


Figure 1-25. Aerodynamic Smoothness Requirements (Sheet 2 of 3)





AREA	SYMBOL	BUTT JOINTS	
		DOOR	SKIN
FUSELAGE ZONE I	G M M_1 N	0.040" MAX 0.008" MAX 0.004" MAX 0.015" MAX	SAME
FUSELAGE ZONE II	G M M_1 N	0.040" MAX 0.012" MAX 0.006" MAX 0.015" MAX	SAME
FUSELAGE ZONE III	G M M N	0.040" MAX 0.015" MAX 0.007" MAX 0.015" MAX	SAME
RADOME	G M M_1	0.040" MAX 0.030" MAX 0.015" MAX	
CANOPY	G M M_1 N	0.030" MIN—0.090" MAX 0.030" MAX 0.015" MAX FLUSH TO 0.060" BELOW FUSELAGE SURFACE	
FORWARD ELECTRONICS COMPARTMENT DOORS AND NOSE GEAR DOORS	G M M_1 N	0.030" MIN—0.090" MAX 0.030" MAX 0.015" MAX 0.030" MAX	
MISSILE BAY DOORS	G G M M_1 N N_1	0.030" MIN—0.090" MAX (LONGITUDINAL GAP) 0.31" MAX FROM BL 0 TO DOOR SPLIT LINE, TAPERING TO 0.06" MAX AT UPPER HINGE LINE (TRANSVERSE GAPS, BOTH ENDS) 0.030" MAX 0.015" MAX 0.030" MAX 0.045" MAX	

AREA	SYMBOL	BUTT JOINTS	
MAIN LANDING GEAR DOOR	G M M_1 N	0.030" MIN—0.090" MAX 0.030" MAX 0.015" MAX 0.030" MAX	
TAIL CONE	G M M_1	0.040" MAX 0.030" MAX 0.030" MAX	
SPEED BRAKES	G M M_1 N	0.030" MIN—0.090" MAX 0.030" MAX 0.015" MAX 0.030" MAX	
		DOOR	SKIN
WING LEADING EDGE (SEE NOTE 2)	G M M M_1	0.020" MAX 0.010" MAX (ALONG FRONT SPAR) 0.004" MAX 0.004" MAX	SAME
WING AFT OF LEADING EDGE	G M M_1 N	0.040" MAX 0.010" MAX 0.010" MAX 0.010" MAX	0.030" MAX 0.010" MAX FLUSH 0.010" MAX
WING LANDING GEAR FAIRING	G M M_1	0.080" MAX 0.010" MAX 0.010" MAX	
FIN	G G M & M_1 N N N	0.020" MAX (LEADING EDGE AND VERTICAL SPLICES) 0.030" MAX (DOORS AND HORIZONTAL SPLICES AFT OF LEADING EDGE) 0.010" MAX (TYPICAL) 0.010" MAX (FWD OF SPAR 5) 0.040" MAX (AFT OF SPAR 5) 0.060" MAX (FINTIP AND TOP OF STREAMLINED RUDDER)	

NOTES:

1. G = GAPS BETWEEN SKINS OR SKINS AND DOORS.

   M = MISMATCH AT TRANSVERSE JOINTS, PROTRUDING SKIN FACING AFT.

   M_1 = MISMATCH AT TRANSVERSE JOINTS, PROTRUDING SKIN FACING FORWARD.

   N = MISMATCH AT LONGITUDINAL JOINTS.

   N_1 = MISMATCH BETWEEN MISSILE BAY DOORS ALONG BL 0.0 EXCEPT AT FORWARD AND AFT END OF DOORS.

2. A CHAMFER (0.5" WIDE) IS PERMISSIBLE FOR WING LEADING EDGE DOORS WHICH ARE UP TO 0.016" MAXIMUM IN EXCESS OF THE 0.004" MAXIMUM MISMATCH TOLERANCE.
3. LEADING EDGE OF RAM AIR TURBINE DOOR MAY PROTRUDE A MAXIMUM OF 0.020".
4. SCREW HEAD MAXIMUM PROTRUSION IF 0.004" IN ALL AREAS.
5. EXCEPT AS NOTED IN RIVET PROTRUSION TABLE, ALL DRIVEN RIVETS MUST PROTRUDE A MEASURABLE AMOUNT ABOVE THE SKIN SURFACE UP TO 0.008" MAXIMUM BEFORE MILLING.
6. ALL FUSELAGE DOORS WHICH HAVE METAL TO METAL CONTACT REQUIRE "FORMED GASKET" TYPE SEAL.
7. ALL GAPS IN NON-REMOVABLE BUTT JOINTS OF THE AIRPLANE EXTERIOR SURFACE ARE FILLED WITH AERODYNAMIC SMOOTHING COMPOUND. REFER TO SECTION XI. THE INTAKE DUCTS ARE CONSIDERED EXTERIOR SURFACES.

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Figure 1-25. Aerodynamic Smoothness Requirements (Sheet 3 of 3)

b. Fuselage area aft of duct inlet: Contour deviation at points of skin attachment shall not exceed ±0.030 inch. Height of wave between skin attachment points shall not exceed 0.050 inch or 2 percent of wave length, whichever is smaller.

c. Wing leading edge area: Height of wave between skin attachment points shall not exceed 0.020 inch or 1 percent of any wave length less than 2.00 inches.

d. Wing area aft of leading edge: Height of wave between skin attachment points shall not exceed 0.050 inch or 1 percent of any wave length less than 5.00 inches.

1-81. Aerodynamic Limitations for One-Time Flight Repairs.

1-82. An airplane damaged beyond "in service" limits may be temporarily repaired for a "one-time" flight to a repair depot, provided the following limitations are observed:

a. All outside patches shall be regarded as temporary repairs which will be replaced with conventional flush repairs. Refer to figure 1-26 for temporary patches.

b. Damage to critical structural components shall be carefully evaluated and if no damage limits are listed in this manual for such components, a qualified aeronautical engineer shall determine airworthiness of the airplane.

c. Flight of the airplane shall be restricted to a maximum of 3 g's.

WARNING

When a "one-time" flight repair is made to an airplane, the cockpit shall be placarded to limit the flight of the airplane, and appropriate entries made in Form 781, Part II.

Under emergency conditions, the aeronautical engineer may approve the use of materials which are not normally approved (not listed in the Material Substitution Tables) or which might be unsuitable for permanent repair. Regardless of the type of material selected, the original strength of the damaged member must be restored; no exceptions may be made. Outside patches must be as smooth as possible with all edges tapered or faired with smoothing compound as shown in figure 1-26. One-time flight repairs should be designed by a qualified aeronautical engineer.

1-83. GENERAL SHOP PRACTICES.

1-84. Fundamental Principles of Repair.

1-85. The fundamental principle of structural repair is to design the strength of the repair equal to or greater than the original structure. Therefore, it is most desirable, though not always necessary, that the repairs have the same configuration as the original structural members. The type of materials, gage, and method of attachment of a structural component must be carefully considered since the weight must be kept to a minimum to permit the maximum load-carrying capacity and operating efficiency consistent with safety. Close observation of the existing structure will indicate the stress-strain ratio of the structural design in each area; for example, a component or group of components made of heavy material and high-tension fasteners indicates a high stress loading in this area and these factors must be considered in designing the repair. Whenever possible the same type material should be used; however, suitable substitutes are listed in Table 1-I. When practicable, pick up existing rivet patterns using the same type and size of rivet as used in the original design. Rivet sizes and patterns may be revised when advisable; for prescribed limits see figure 1-28 and refer to paragraph 1-120. Shop methods for general practice are discussed in the General Manual for Structural Repair, T.O. 1-1A-1.

1-86. Clean Working Practice.

1-87. Establishment of a clean working practice is necessary to eliminate the possibility of damage from corrosion caused by contact of "loose hardware" with unlike metals, damage from loose objects thrown about during flight, and jamming of control mechanisms by foreign objects. This practice will insure that chips, broken parts, and discarded fasteners do not remain in the airplane. Generally, a thorough cleaning of the area at the end of each work period is sufficient to maintain safety standards. During the progress of the repair, broken parts and loose hardware should be picked up before they become lodged between the faying surfaces of repair parts or fall into hard-to-reach recesses. At the completion of a repair, the repair site and surrounding areas of the airplane must be carefully inspected for foreign matter and thoroughly cleaned. Inspect for loose hardware with flashlight and long-handled mirror. The presence of loose hardware in hidden recesses may be detected by lightly tapping lower surfaces of the area with a soft rubber mallet and listening for a rattling sound. Pick up small chips and dirt with a vacuum cleaner, or whisk broom and dust pan. Refer to T. O. 1F-106A-2-2-2-2 for approved airframe cleaning materials and procedures.

1-88. SHEET METAL FORMING.

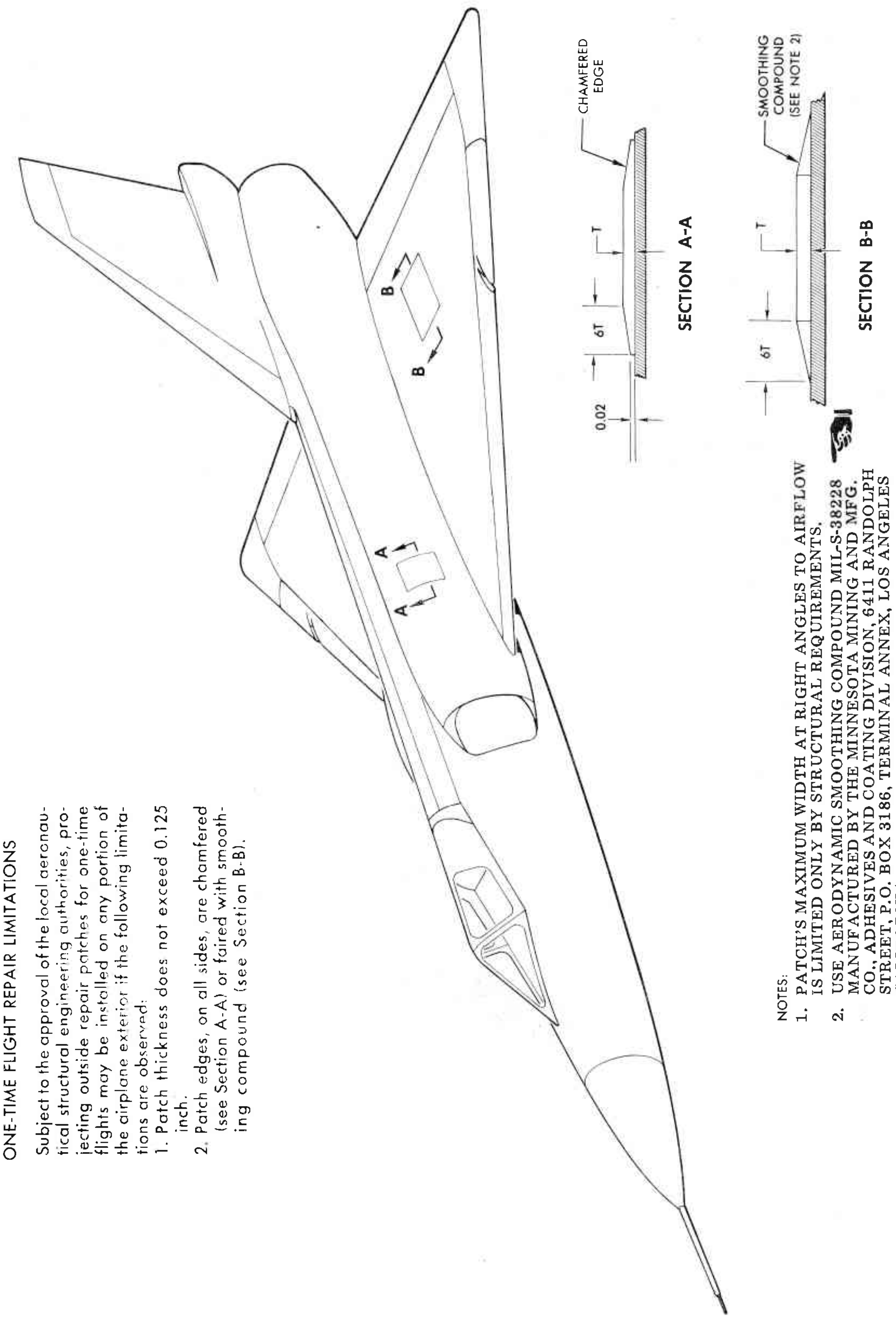
1-89. Materials Used for Fabrication and Repair.

1-90. The basic materials employed in the construction of these airplanes and to be considered for structural repair purposes are: aluminum alloys, titanium alloy, commercially pure titanium, corrosion resistant steels, and magnesium. When fabrication and repair require the employment of these materials, three important factors should be considered. These factors are shape, alloy, and condition or temper designation. In all cases

ONE-TIME FLIGHT REPAIR LIMITATIONS

Subject to the approval of the local aeronautical structural engineering authorities, projecting outside repair patches for one-time flights may be installed on any portion of the airplane exterior if the following limitations are observed:

1. Patch thickness does not exceed 0.125 inch.
2. Patch edges, on all sides, are chamfered (see Section A-A) or faired with smoothing compound (see Section B-B).



NOTES:
1. PATCH'S MAXIMUM WIDTH AT RIGHT ANGLES TO AIRFLOW IS LIMITED ONLY BY STRUCTURAL REQUIREMENTS.
2. USE AERODYNAMIC SMOOTHING COMPOUND MIL-S-38228 MANUFACTURED BY THE MINNESOTA MINING AND MFG. CO., ADHESIVES AND COATING DIVISION, 6411 RANDOLPH STREET, P.O. BOX 3186, TERMINAL ANNEX, LOS ANGELES CALIFORNIA.

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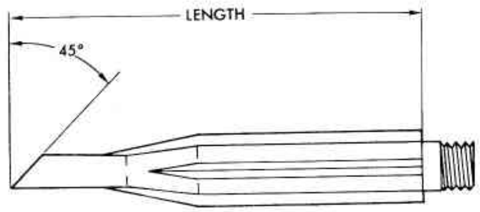
Figure 1-26. Aerodynamic Smoothness Requirements—One-Time Flight



ENLARGED VIEW OF SEALANT NOZZLE

SEALANT APPLICATION PROCEDURE

a. Pump trigger until pressure from ram is exerted against plunger.
b. Relax grip on trigger completely.
c. Place nozzle in position where sealant or fairing compound is to be applied.
d. Squeeze trigger into handle of gun and hold.

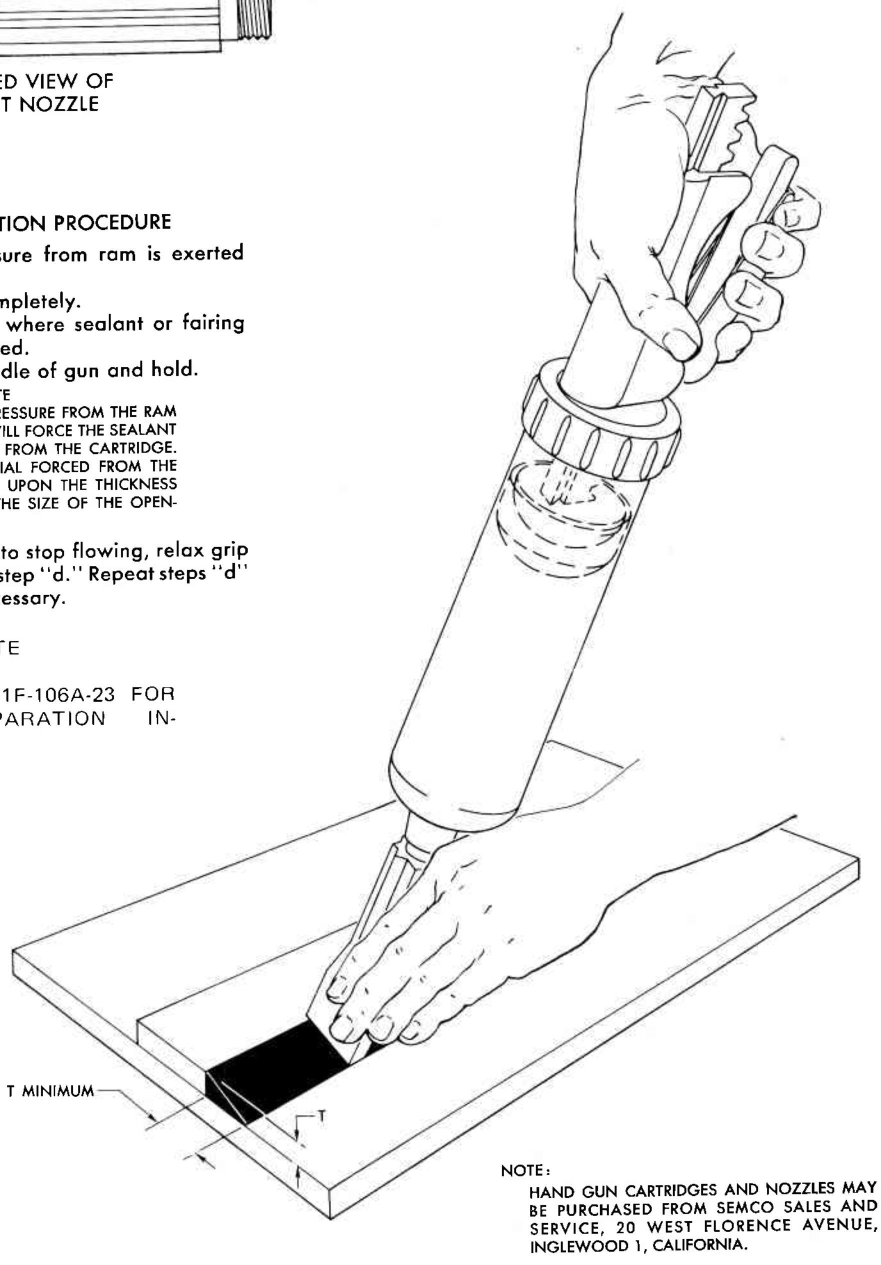
NOTE

IN THIS POSITION, THE PRESSURE FROM THE RAM AGAINST THE PLUNGER WILL FORCE THE SEALANT OR FAIRING COMPOUND FROM THE CARTRIDGE. THE AMOUNT OF MATERIAL FORCED FROM THE CARTRIDGE WILL DEPEND UPON THE THICKNESS OF THE MATERIAL AND THE SIZE OF THE OPENING IN THE NOZZLE.

e. When material is about to stop flowing, relax grip completely, then repeat step "d." Repeat steps "d" and "e" as often as necessary.

NOTE

REFER TO T.O. 1F-106A-23 FOR SURFACE PREPARATION INSTRUCTIONS.



NOTE:
HAND GUN CARTRIDGES AND NOZZLES MAY BE PURCHASED FROM SEMCO SALES AND SERVICE, 20 WEST FLORENCE AVENUE, INGLEWOOD 1, CALIFORNIA.

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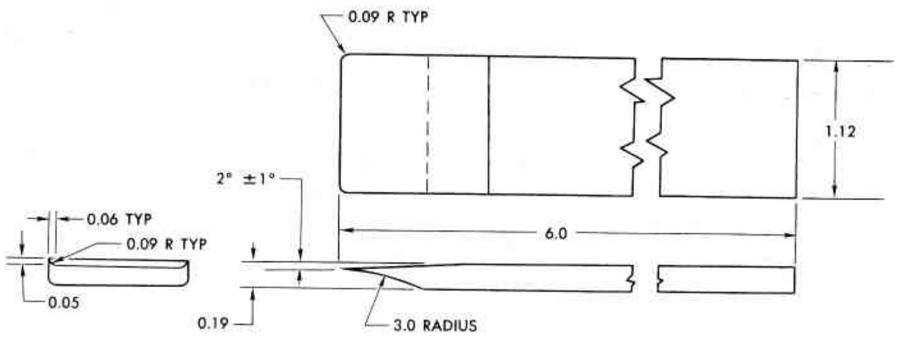
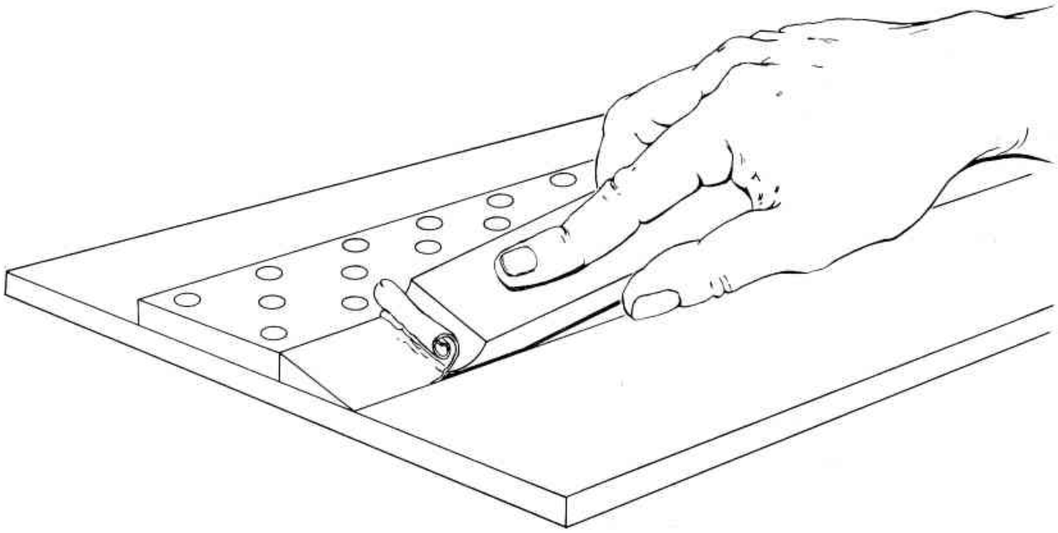
Figure 1-27. Application of Fairing Compound (Sheet 1 of 2)



DETAIL OF
TRIMMING TOOL

TOOL MAY BE MADE FROM 301 TOOL STEEL HEAT TREATED TO ROCKWELL READING OF 60-63, OR FROM A DISCARDED 12-INCH MILL FILE.



TRIMMING PROCEDURE

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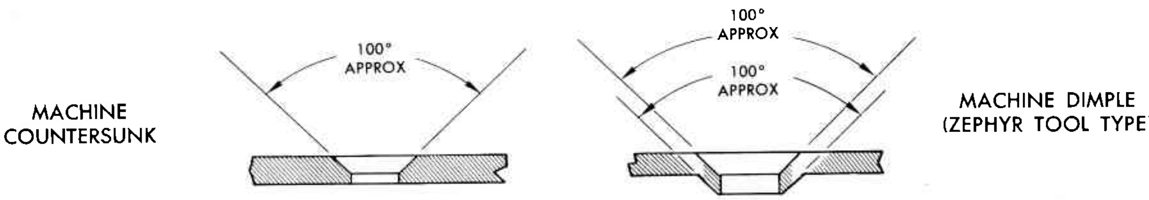
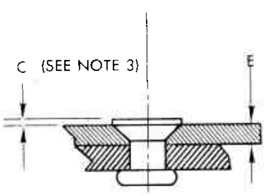
Figure 1-27. Application of Fairing Compound (Sheet 2 of 2)

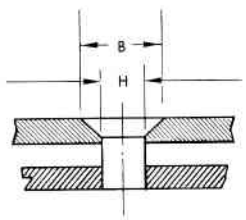


NOTES:

1. THE FIGURES ABOVE INDICATE THE TYPICAL ANGULAR DIMENSIONS OF TOOLS USED TO INSTALL COUNTERSUNK RIVETS AS REQUIRED BY THIS STANDARD. ALL DIMPLED AND COUNTERSUNK RIVET INSTALLATION DATA ARE BASED ON THE AN426 TYPE (100°) RIVET AND THE TOOLS USED SHALL BE DESIGNED ACCORDINGLY. IN ADDITION, DESIGN AND USE OF THE TOOLS SHALL BE SUCH AS TO ACCOMPLISH THE REQUIREMENTS GIVEN BELOW, SHOWN IN THE FOLLOWING TABLES AND ILLUSTRATIONS.
2. PROVIDED THE INDICATED LIMITATIONS ARE OBSERVED, CHOICE BETWEEN THE VARIOUS METHODS OF INSTALLING COUNTERSUNK RIVETS IS OPTIONAL.
3. TABULATED VALUES FOR DIMENSION "C" IN TABLE 1 ARE APPLICABLE TO INNER STRUCTURE ONLY. FOR VALUES APPLICABLE TO OUTER PLATING, REFER TO AERODYNAMIC SMOOTHNESS REQUIREMENTS IN THIS SECTION.
4. WHEN INSTALLING A HEAVY GAGE PATCH WHICH PICKS UP A MACHINE COUNTERSUNK HOLE IN THE ORIGINAL MEMBER, USE METHOD SHOWN BELOW.



AFTER INSTALLATION



BEFORE INSTALLATION

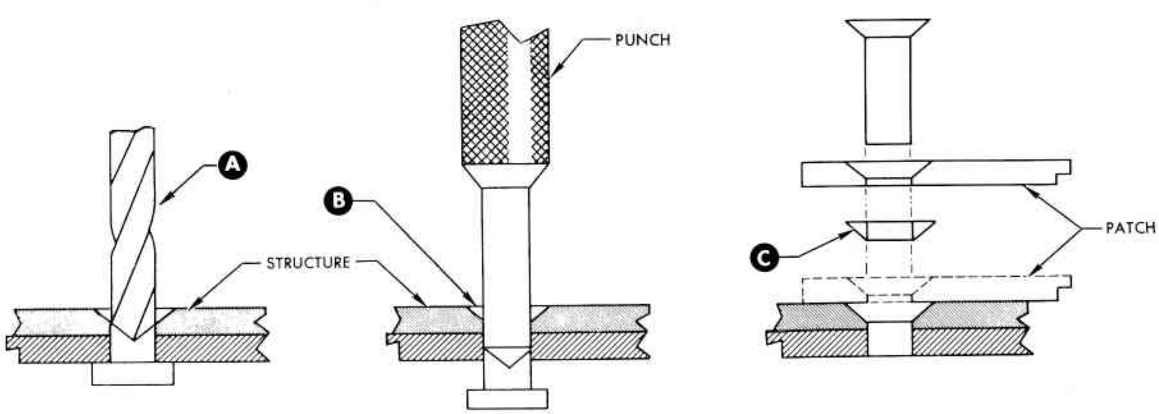
TABLE I

NOMINAL RIVET DIAMETER		1/16	3/32	1/8	5/32	3/16	1/4	5/16	3/8
B (NOMINAL)		0.114	0.179	0.225	0.287	0.355	0.478	0.569	0.697
C (SEE NOTE 3)	MAX	0.008	0.008	0.008	0.008	0.008	0.008	0.008	0.008
E	MIN	0.032	0.040	0.051	0.064	0.064	0.064	0.064	0.064
H	MIN	0.067	0.099	0.128	0.161	0.191	0.257	0.323	0.386
	MAX	0.073	0.105	0.134	0.167	0.199	0.265	0.331	0.394

A Remove rivet head using drill of the size prescribed for installation of the rivet to be installed. Use care to center on and not drill beyond the rivet head.

B Remove rivet and save the head.

C Use the rivet head as a tapered filler. Cement in place with sealant, Military Specification MIL-S-8802.



METHOD FOR FILLING DIMPLED OR COUNTERSUNK HOLES

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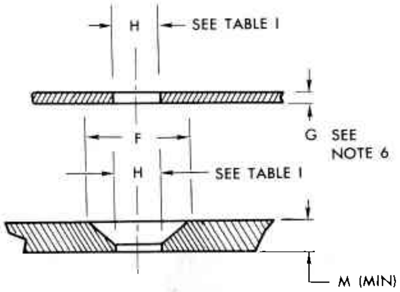
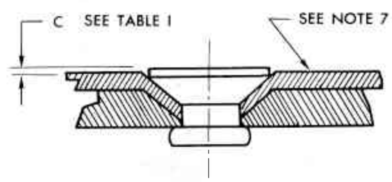
Figure 1-28. Flush Rivet Installation (Sheet 1 of 3)

DIMPLED SKIN



BEFORE INSTALLATION



AFTER INSTALLATION

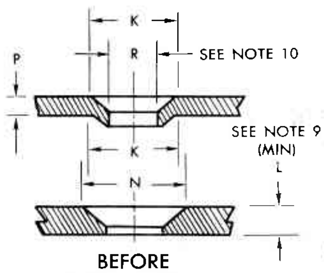
TABLE II

G	NOMINAL RIVET DIAMETER				
	3/32	1/8	5/32	3/16	1/4
(NOMINAL)	F (NOMINAL) SEE NOTE 5				
0.020	0.194	0.240	0.302	0.370	0.493
0.025	0.198	0.244	0.306	0.374	0.497
0.032	0.202	0.248	0.310	0.378	0.501
0.040		0.254	0.316	0.384	0.507
0.051			0.325	0.393	0.516
0.064				0.403	0.526
0.072					0.537
0.081					0.549
M (MIN)	0.051	0.064	0.081	0.102	0.125

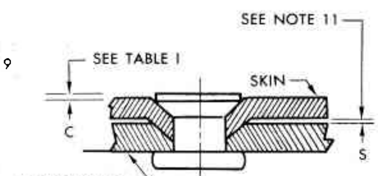
MACHINE DIMPLED SKIN

TABLE III

NOMINAL RIVET DIAMETER	3/32	1/8	5/32	3/16	1/4
N NOMINAL DIA OF MACHINE CSINK	0.189	0.235	0.297	0.365	0.488
K NOMINAL DIA OF MACHINE DIMPLE	0.179	0.225	0.287	0.355	0.478
L (MIN)	0.040	0.051	0.064	0.072	0.091
P (MAX)	0.032	0.040	0.051	0.051	0.051
R (MAX)	0.107	0.140	0.175	0.216	0.280
R (MIN)	0.099	0.128	0.161	0.191	0.257



BEFORE INSTALLATION



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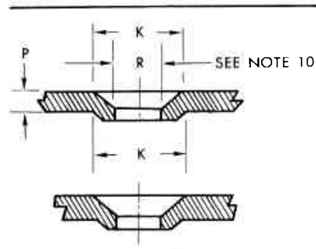

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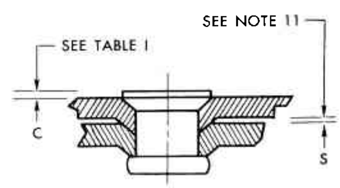

AFTER INSTALLATION

NOTES:

5. VALUES IN TABLE II ARE FOR USE ONLY WHEN THE TOP SHEET IS DIMPLED WITH THE RIVET HEAD. WHEN TOP SHEET IS MACHINE DIMPLED, USE TABLE III.
6. TABULATED VALUES FOR DIMENSION "G" ALSO APPLY AS A TOTAL VALUE WHEN MORE THAN ONE SHEET IS DIMPLED SIMULTANEOUSLY INTO A MACHINE COUNTERSINK WITH THE RIVET HEAD.
7. USE OF THIS INSTALLATION METHOD IS RESTRICTED TO APPLICATIONS WHERE THE TOP SHEET HAS FORMING CHARACTERISTICS EQUAL TO OR BETTER THAN 2024-T.
8. VALUES IN TABLE III ARE FOR USE IN THE FOLLOWING APPLICATIONS:
   A. WHEN MACHINE DIMPLED SHEET (OR SHEETS) IS RIVETED TO MACHINE COUNTERSUNK SHEET (OR SHEETS SEE NOTE 7).
   B. WHEN INDIVIDUALLY MACHINE DIMPLED SHEETS ARE RIVETED TOGETHER. (DIMENSION "K" IS APPLICABLE EVEN THOUGH MORE THAN TWO MACHINE DIMPLED SHEETS ARE JOINED.)
9. THE MACHINE COUNTERSUNK SHEET INDICATED BY DIMENSION "L" MAY CONSIST OF MORE THAN ONE SHEET PROVIDED THE TOTAL THICKNESS IS NOT LESS THAN "L."
10. WHEN REQUIRED BY TOOLING LIMITATIONS, VALUES FOR DIMENSION "R" IN MACHINE DIMPLES MAY BE MET AFTER DIMPLES ARE FORMED.
11. TO PROPERLY TRANSMIT SHEAR, DIMPLES SHOULD NEST TIGHTLY IN COUNTERSINKS. THIS MAY RESULT IN GAPS BETWEEN SHEETS WHICH WILL BE ACCEPTABLE SUBJECT TO THE FOLLOWING CONDITIONS:
   A. THE GAP(S) SHALL NOT EXCEED 0.004 INCH ADJACENT TO 3/32 RIVETS, 0.006 INCH ADJACENT TO 1/8 INCH RIVETS, 0.008 INCH ADJACENT TO 5/32 INCH, 3/16 INCH OR 1/4 INCH RIVETS.
   B. THE AREAS AFFECTED ARE NOT INTENDED TO BE LIQUID TIGHT.
   C. THE GAPS DO NOT CAUSE RIPPLES THAT ARE OBJECTIONABLE FROM AN APPEARANCE STANDPOINT.

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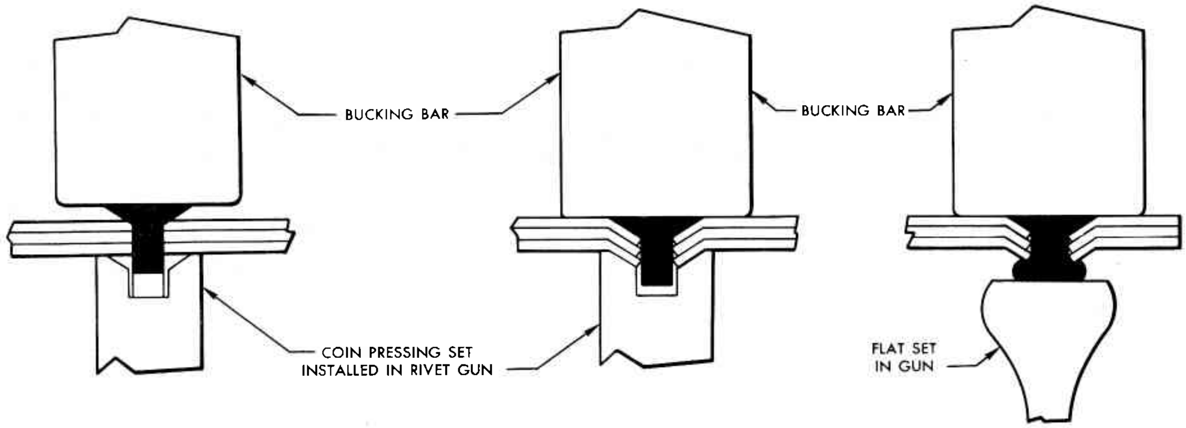
Figure 1-28. Flush Rivet Installation (Sheet 2 of 3)

COIN DIMPLING

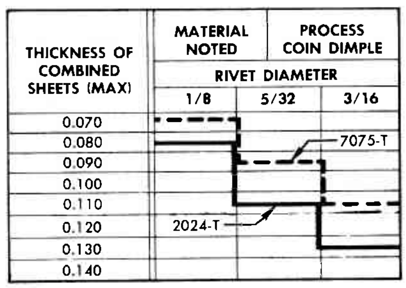


BEFORE DIMPLING AFTER DIMPLING RIVET HEADED

COIN DIMPLING TOOL NUMBERS

RIVET SIZE	MARTIN TOOL NUMBER	ZEPHYER TOOL NUMBER
3/32	MM-215-1	ZT1851-3/32
1/8	MM-215-2	ZT1851-1/8
5/32	MM-215-3	ZT1851-5/32
3/16	MM-215-4	ZT1851-3/16

TABLE IV



NOTES:

12. TABLE IV IS APPLICABLE ONLY WHEN MORE THAN ONE SHEET IS DIMPLED SIMULTANEOUSLY BY THE COIN DIMPLING PROCESS. WHEN THE SUM OF THE COMPONENT SHEETS FALLS ABOVE THE HEAVY LINE OF THE MATERIAL INDICATED, THE SHEETS MAY BE DIMPLED SIMULTANEOUSLY PROVIDED THE APPLICABLE LIMITS ARE OBSERVED.
13. COIN DIMPLING IS A PROCESS WHERE THE RIVET IS USED AS A TOOL FOR DIMPLING TWO OR MORE SHEETS SIMULTANEOUSLY. THE SHEETS ARE BACKED UP BY A RECESSED TOOL DURING THE DIMPLING OPERATION, THEN A FLAT TOOL FORMS THE SHOP HEAD. THIS METHOD HAS THE ADVANTAGE OF MAKING IT POSSIBLE TO INSTALL 100° COUNTERSUNK RIVETS IN DIMPLED SKIN APPLICATION WHEN DEALING WITH 7075-T WITHOUT HOT DIMPLING EQUIPMENT. IT IS SUBJECT TO THE FOLLOWING LIMITATIONS:
    - A. IT SHALL BE USED ONLY WHEN NECESSARY AS THE RIVET HEAD DEFORMS SO THAT THE REPLACEMENT WITH A STANDARD 100° RIVET IS NOT FEASIBLE.
    - B. IT SHALL NOT BE USED WHERE THE RIVET MATERIAL IS SOFTER THAN 2117.
    - C. THE GAGE OF ANY SINGLE SHEET OR A COMBINATION OF SHEETS SHALL NOT EXCEED 1/2 OF THE MAXIMUM VALUE ALLOWABLE FOR THE COMBINATION AS DETERMINED FROM TABLE IV.
14. TABLE IV IS SUBJECT TO THE FOLLOWING LIMITATIONS:
    - A. WHERE 2024-T IS INDICATED, THE FOLLOWING MATERIALS ARE TO BE INCLUDED: 2024-0, 7075-0, AND 7075-W (WITHIN 2 HOURS AFTER QUENCH).
    - B. WHERE 7075 IS INDICATED, THE FOLLOWING MATERIALS ARE TO BE INCLUDED: 2014, 7075-W, 2024-RT ("Y" SECTIONS), 2024-RT, 2024-T80, 2024-T81, 2024-T84, AND 2024-T86.
    - C. WHERE MORE THAN ONE MATERIAL IS USED SO THAT BOTH 7075-T AND 2024-T LIMITATIONS ARE APPLICABLE, THE 7075-T LIMITATIONS SHALL BE USED.
15. A 3X GUN SHOULD BE USED FOR 1/8, 5/32 AND 3/16 INCH RIVETS WHEN THE COMBINED MATERIAL THICKNESS IS 0.080 INCH OR LESS. A 4X GUN SHOULD BE USED WHERE THE COMBINED THICKNESS IS BETWEEN 0.064 AND 0.150 INCH. WHERE THICKNESS LIMITATIONS OVERLAP, EITHER SIZE GUN MAY BE USED.

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Figure 1-28. Flush Rivet Installation (Sheet 3 of 3)

of repair or replacement the original factors should be duplicated; however, should conditions prohibit this procedure, suitable substitutes are listed in Table 1-I.

1-91. Material Shapes.

1-92. There are three basic shapes: sheet stock, extruded shapes, and drawn bar stock.

a. Sheet stock is obtainable in various gages and in clad or bare types. This type of material is used in fabricating patches for skin, panels, rib and bulkhead webs, and may be shaped to produce splice and replacement channels or angles. Refer to "Y" sections listed in Section VIII.

b. Extruded shapes are used extensively throughout the structure, particularly in the heavily stressed areas. When repairs require the replacement of these parts, the same die number extrusion should be used; however, if the same die number is not available, other extrusions of a larger dimension may be machined to the desired dimension. Refer to list of extruded shapes in Section VIII.

c. Drawn bar stock is a material from which parts may be machined. Extruded shapes machined from bar stock are slightly weaker than the original extruded shapes of the same alloy; therefore, the cross-sectional area of the substitute part should be approximately 5 percent greater than the original to compensate for this lower strength factor.

1-93. Bend Radii.

1-94. The standard inside bend radii for the various standard gages of sheet aluminum, magnesium, steel, and titanium are given in Table 1-XIX. See figure 1-29 for the computation of bend reduction and flat pattern layout. The bend radii for aluminum alloy tubing is given in Table 1-XX.

1-95. Temper of Aluminum Alloy.

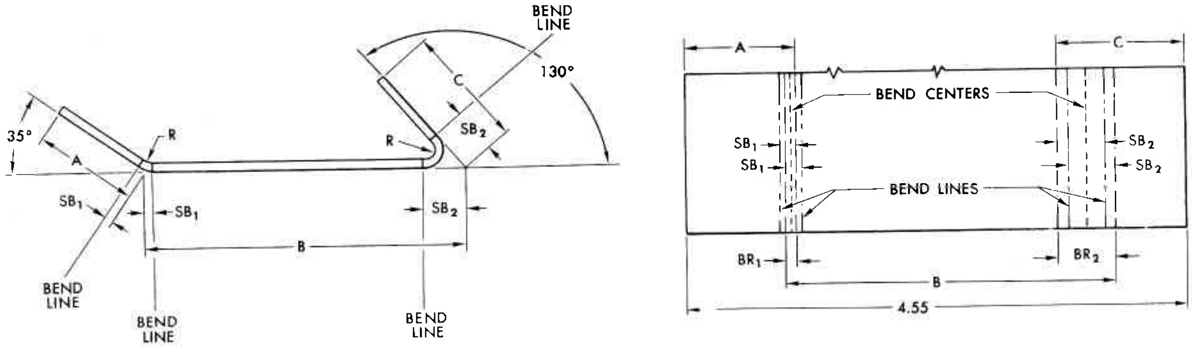
1-96. The aluminum alloys employed in the primary structure are heat-treatable and in a "T" hard condition. It follows, therefore, that materials used in designing and fabricating repairs or making replacements must also be in the "T" hard condition when the repair is completed. All aluminum alloys are heat treated and aged in accordance with Specification MIL-H-6088, except in cases where bare materials have been aged to conditions for clad materials in accordance with Specification AN-A-42.

a. 2014-T4 material is not commonly used; however, structural parts requiring moderately high strength comparable to 2024-T3 or 2024-T4 may be fabricated and employed as a substitute should the necessity arise. This material has good forming qualities in the "O" condition, but is very limited in the "-T4" condition. It possesses good machinability in either the "O" or "-T4" condition.

b. 2024-T sheet stock in the clad and bare condition is used, although not extensively, in the structure in various "-T" or temper designations. Parts fabricated from "-T3" stock without the reheat treatment will retain this rating. If reheat treated and formed in the "W" or unstable condition, they will be designated as 2024-T4. This designation applies when the material is not cold worked or artificially aged after heat treatment. Both 2024-T3 and 2024-T4 materials contain slightly more strength than the 2024-T4 extrusions. A few roll-formed sections are fabricated from 2024-T3. Refer to Section VIII for roll-formed sections and extrusions. Roll-formed sections of 2024-T3 reheat treated to facilitate additional forming will be designated as 2024-T4 after hardening. Extrusions of 2024-T4 will retain their designated rating after being reheat treated. The designation 2024-T6 (formerly known as 24S-T80) indicates material that has been artificially age hardened from the "-T4" condition to increase its mechanical properties. The designation 2024-T81 indicates material that has been artificially age hardened in accordance with Specification AN-A-42 from the "-T3" condition to increase its mechanical properties. These materials are used where a moderate amount of increased strength over the "-T4" and "-T3" materials is required. 2024-T material in increased gages may be used as a substitute material for 7075-T6; however, this practice is not considered advisable except in cases of emergency. When 2024-T is substituted in highly stressed areas it shall be used as a temporary repair only and must be replaced as soon as 7075-T6 material is available. 2024-T36 material that has been solution heat treated and strain hardened becomes 2024-T86 when age hardened. This material is considerably stronger than 2024-T3 or 2024-T4 material. The use of 2024-T36 material should be restricted to plating or machined parts due to its relatively poor forming characteristics.

c. 6061-T6 material may be employed in lower or nonstressed areas. This material contains excellent forming characteristics in the heat treated condition and possesses good machinability and weldability qualities.

d. 7075-T sheet stock in the bare condition is employed extensively in the wing structure, and has a temper designation of "-T6" only. Extrusions, bar stock, and bare roll-formed shapes carry the same temper designation; however, they possess slightly more strength than 7075-T6 clad material, as shown in Table 1-I. When 7075-T6 material is used for repairs and a moderate amount of forming is required, the material should be reheat treated and formed in the "W" or temporarily unstable condition due to the hardness of the material. Following this operation the material must be artificially aged to regain its "-T6" rating or condition. Forming of 7075-T6 material should be confined to moderate or gentle contours, although with care a bend radius of 5T may be obtained (see Table 1-XIX). Ordinary cold dimpling methods are not advisable in 7075-T6 material since cracks are likely to occur. Refer to paragraphs 1-175 through 1-177, and figure 1-28 for information on dimpling.



GIVEN
2024-W CLAD
GAGE=0.125
A=1.00
B=3.00
C=1.50

R =0.25 (SEE NOTE 2)
BR_1 =0.05 (FROM SHEET 2)
BR_2 =0.90 (FROM SHEET 2)
SB_1 =0.14 (FROM SHEET 3)
SB_2 =0.80 (FROM SHEET 3)

$$\text{ACTUAL FLAT LENGTH} = A - BR_1 + B - BR_2 + C$$

$$= 1.00 - 0.05 + 3.00 - 0.90 + 1.50$$

$$= 4.55 \text{ INCHES}$$

FLAT PATTERN LAYOUT

a. Take the dimensions and angles from the damaged or opposite hand part with an accurate hundredths scale and make a cross-section sketch of the part. A piece of wire solder may be bent around the damaged part to obtain shape.
b. Determine bend reduction (BR) value from sheet 2 of this figure for each angle. Bend reduction is subtracted from the sum of the flange lengths to determine the actual flat length. Record values for use in next step.
c. Mark outline of part on a flat sheet of repair material with a sharp pencil. Do not use a metal scribe. Determine actual flat length by subtracting the sum of the BR values from the sum of the side dimensions of the damaged part sketched in step "a."
d. Layout marks for "A" and "C" flanges, using the dimensions shown in sketch of damaged part.
e. Layout marks for "BR_1" and "BR_2" dimensions at ends of pattern as shown. The "B" side dimension is determined by the location of these marks.
f. Determine set-back (SB) value from sheet 3 of this figure for each angle. Bend lines are located by set-back dimensions. Record for use in next step.
g. Layout marks for end lines by marking off "SB_1" and "SB_2" dimensions as shown.
h. Trim part to marked outline and hand form or bend angles on brake.

NOTES:
1. BEND CENTER LINES MAY BE LOCATED BY SUBTRACTING ½ BR FROM EACH BENT-UP FLANGE LENGTH.
2. REFER TO TABLE 1-XIX FOR STANDARD BEND RADIUS.

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Figure 1-29. Flat Pattern Layout (Sheet 1 of 3)

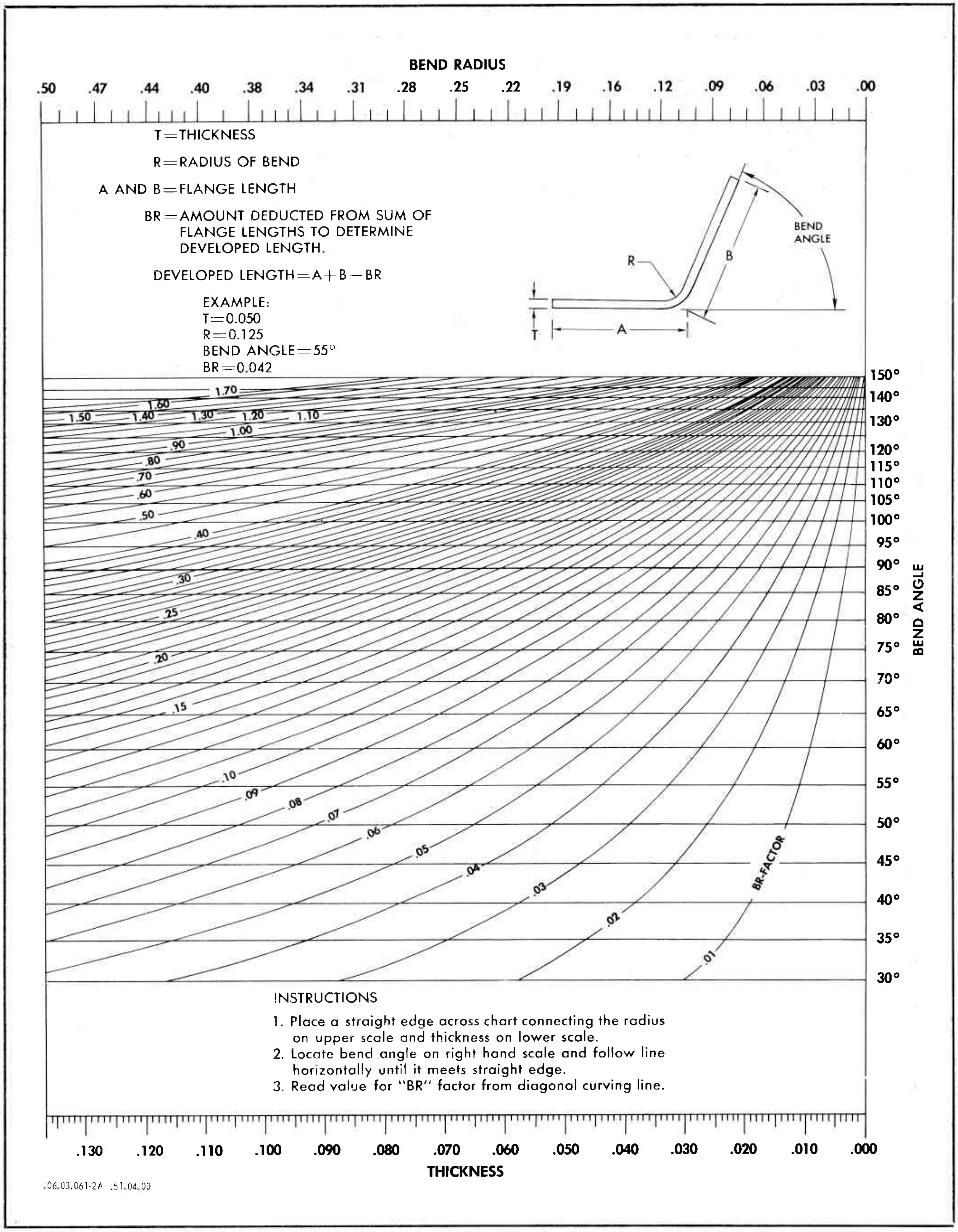

Figure 1-29. Flat Pattern Layout (Sheet 2 of 3)

Table of Bent-Formed Set-Back Values

A°	K	A°	K	A°	K	A°	K
1°	0.00873	46°	0.42447	91°	1.0176	136°	2.4751
2	0.01745	47	0.43481	92	1.0355	137	2.5386
3	0.02618	48	0.44523	93	1.0538	138	2.6051
4	0.03493	49	0.45573	94	1.0724	139	2.6746
5	0.04366	50	0.46631	95	1.0913	140	2.7475
6	0.05241	51	0.47697	96	1.1106	141	2.8239
7	0.06116	52	0.48773	97	1.1303	142	2.9042
8	0.06993	53	0.49858	98	1.1504	143	2.9887
9	0.07870	54	0.50952	99	1.1708	144	3.0777
10	0.08749	55	0.52057	100	1.1917	145	3.1716
11	0.09629	56	0.53171	101	1.2131	146	3.2708
12	0.10510	57	0.54295	102	1.2349	147	3.3759
13	0.11393	58	0.55431	103	1.2572	148	3.4874
14	0.12278	59	0.56577	104	1.2799	149	3.6059
15	0.13165	60	0.57735	105	1.3032	150	3.7320
16	0.14054	61	0.58904	106	1.3270	151	3.8667
17	0.14945	62	0.60086	107	1.3514	152	4.0108
18	0.15838	63	0.61208	108	1.3764	153	4.1653
19	0.16734	64	0.62487	109	1.4019	154	4.3315
20	0.17633	65	0.63707	110	1.4281	155	4.5107
21	0.18534	66	0.64941	111	1.4550	156	4.7046
22	0.19438	67	0.66188	112	1.4826	157	4.9151
23	0.20345	68	0.67451	113	1.5108	158	5.1455
24	0.21256	69	0.68728	114	1.5399	159	5.3995
25	0.22169	70	0.70021	115	1.5697	160	5.6713
26	0.23087	71	0.71329	116	1.6003	161	5.9758
27	0.24008	72	0.72654	117	1.6318	162	6.3137
28	0.24933	73	0.73996	118	1.6643	163	6.6911
29	0.25862	74	0.75355	119	1.6977	164	7.1154
30	0.26795	75	0.76733	120	1.7320	165	7.5957
31	0.27732	76	0.78128	121	1.7675	166	8.1443
32	0.28674	77	0.79543	122	1.8040	167	8.7769
33	0.29621	78	0.80978	123	1.8418	168	9.5144
34	0.30573	79	0.82434	124	1.8807	169	10.385
35	0.31530	80	0.83910	125	1.9210	170	11.430
36	0.32492	81	0.85408	126	1.9626	171	12.706
37	0.33459	82	0.86929	127	2.0057	172	14.301
38	0.34433	83	0.88472	128	2.0503	173	16.350
39	0.35412	84	0.90040	129	2.0965	174	19.081
40	0.36397	85	0.91633	130	2.1445	175	22.904
41	0.37388	86	0.93251	131	2.1943	176	26.636
42	0.38386	87	0.94896	132	2.2460	177	38.188
43	0.39391	88	0.96569	133	2.2998	178	57.290
44	0.40403	89	0.98270	134	2.3558	179	114.590
45	0.41421	90	1.00000	135	2.4142	180	Infinite

Example:

Degrees in Bend (A) = 120°
Gage (T) = 0.032
Radius (R) = 0.125
K (From Table) = 1.732

Set-Back = K (T+R)
Set-Back = 1.732 (0.032+125)
= 1.732 x 0.157
= 0.272

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Figure 1-29. Flat Pattern Layout (Sheet 3 of 3)

e. 7178-T bare sheet stock is employed extensively in the fuselage structure and has a temper designation of "-T6" only. Extrusions, bar stock, and bare roll-formed shapes carry the same temper designation; however, they possess slightly more strength than 7178-T6 clad material, as shown in Table 1-I. When 7178-T6 material is used for repairs and a moderate amount of forming is required, the material should be reheat treated and formed in the "W" or temporarily unstable condition due to the hardness of the material. Following this operation the material must be artificially aged to regain its "-T6" rating or condition. Forming of 7178-T6 material should be confined to large radii of 5T or larger. 7178-T6 material is about 10 percent stronger than 7075-T6 stock and should be carefully inspected for cracks after each forming operation.

1-97. Special Cutting Tool for Aluminum Alloys.

1-98. If standard cutting tools for aluminum alloys are not available, or if the required cutting operation cannot be readily adapted to the standard cutting tools that are available, a special cutting tool similar to the one shown on figure 1-30 may be manufactured locally.

1-99. Forming Characteristics of Titanium.

1-100. Special care in the handling of titanium material to avoid surface and edge imperfections is essential to its successful forming. Higher quality and polish in tools, more effective control of metal flow, higher pressure and slower action in most operations, and the discriminative use of heat for severe forming will aid in overcoming forming difficulties encountered with titanium. Forming operations at room temperatures indicate that this material behaves more like ¼-hard and ½-hard stainless steel than any other metal. In general, parts that can be formed from ¼-hard stainless steel can also be formed from commercially pure titanium, although titanium requires hot forming to accomplish the more severe operations. Titanium alloy will form in straight or very slightly contoured flanges and in curved sections at room temperature.

1-101. Brake Forming of Titanium.

1-102. Titanium sheet and strips may be bent successfully in the power brake, using the same methods and techniques as employed for stainless steel. Bend properties are more favorable if the axis of the bend is perpendicular to the rolling direction of the material. Cold bend tests have determined that a minimum bend radius of 3T may be obtained with all gages of commercially pure titanium. Hot bend tests, heating titanium parts and forming dies to approximately 260°C (500°F), have indicated that a minimum bend radius of 2T may be obtained. Similar results may be obtained with titanium alloy at higher temperatures in gages up to 0.070 inch. Alloys with 8 percent manganese are formed at temperatures up to 565°C (1050°F). Alloys with 5 percent aluminum and 2.5 percent tin are formed at temperatures up to 621°C (1150°F).

CAUTION

Do not apply temperature indicating paint or crayon marks directly on titanium surfaces. Cracks in the titanium may develop from contaminants in these materials. Use a hand pyrometer for temperature measurements.

For gages over 0.070 inch, titanium alloy may be cold formed to bend radii of 3½T and hot formed to 2½T. Refer to Table 1-XIX for bend radii.

1-103. Titanium Acid-Etch Treatment.

1-104. Acid-etch treatment of titanium will remove surface oxides to prevent crack development during forming operations. If a test sample of the material to be used can be formed without cracking at the bend, etching will not be necessary. Mixing of the acids in the correct proportions is important and parts should never be immersed for periods exceeding the time limits given in the following procedure:

a. Prepare etch solution by mixing 1½ to 2½ percent hydrofluoric acid, Specification O-H-795, and 25 percent to 35 percent nitric acid, Specification O-N-350, into tap water at room temperature.

b. Inspect parts for cleanliness before immersion. If parts are oily or soiled, they may be cleaned with an alkaline cleaner solution or vapor degreased.

c. Immerse parts in solution.

WARNING

Operator must wear rubber gloves and face shield during this operation.

Immersing clean, oxide-free titanium in the etch solution for five minutes will remove 0.0005-inch from each surface of the part. Etch only as long as necessary to remove surface scratches. The blue oxide coating, formed by heating part 371° to 426°C (700° to 800°F), will be removed in one to five minutes. The purple oxide coating, formed by heating part 426° to 510°C (800° to 950°F), will be removed in five to ten minutes.

d. Remove part from etch solution and rinse thoroughly in cold running water.

e. Dry part with oil-free air blast or in an oven at 82° to 115°C (180° to 250°F).

1-105. Titanium Drawing Operations.

1-106. Drawn parts of commercially pure titanium may be formed hot or cold, depending on the severity of

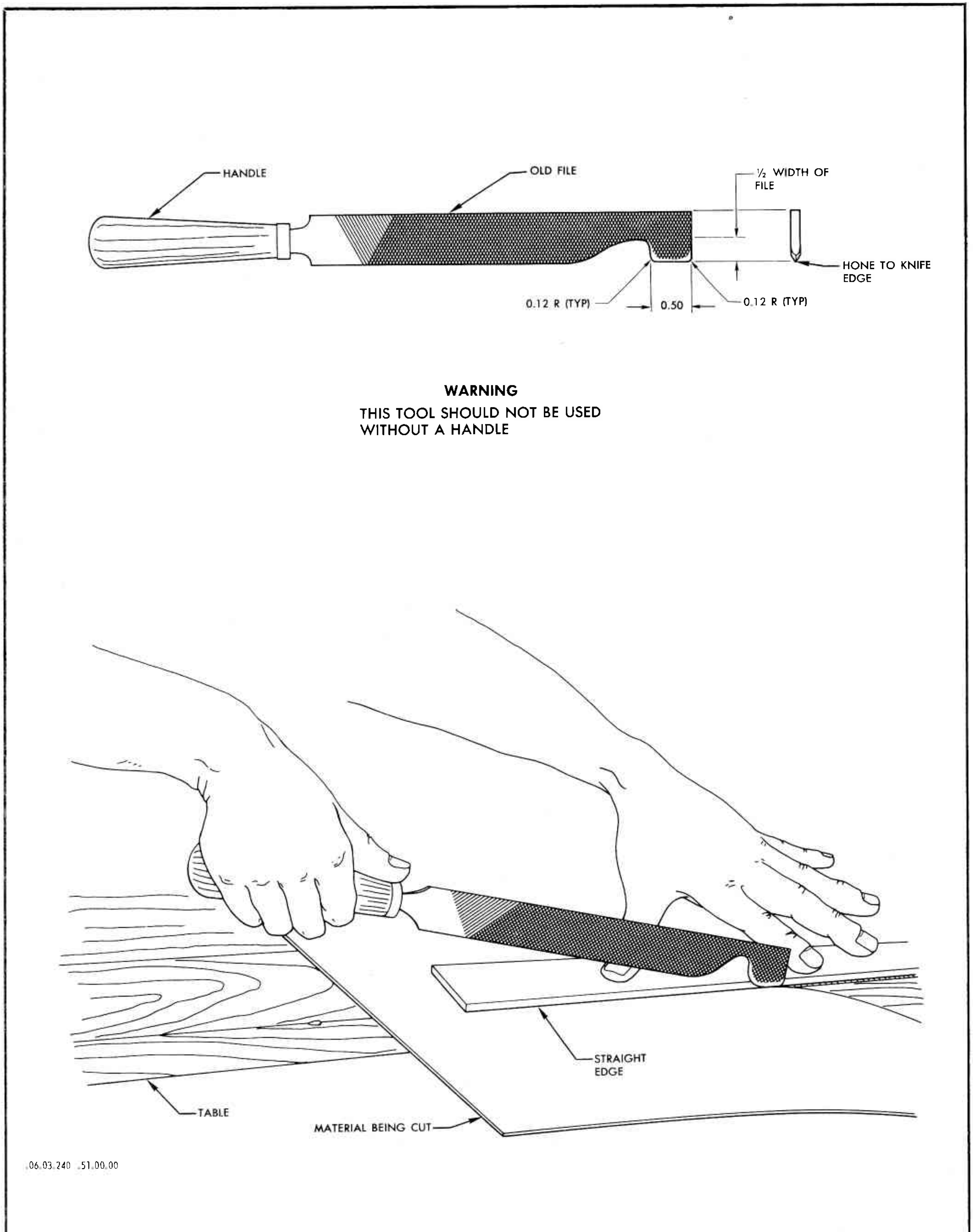


Figure 1-30. Metal Cutting Tool

the operation. In general, most parts can be formed at 260°C (500°F) with dies heated to 135°C (275°F) or higher. Alloy titanium should have an acid-etch treatment prior to the operation and be formed at the temperatures specified for brake forming in paragraph 1-101. Colloidal graphite is a satisfactory lubricant for either hot or cold forming. Tool surfaces require frequent polishing to prevent seizing. Drawing in two stages with an intermediate anneal also gives good results.

1-107. Sawing Titanium.

1-108. Sawing titanium in gages up to 0.250 inch is readily accomplished on a conventional band saw or a friction saw, using sufficient force to keep the saw cutting continuously.

WARNING

The fire precautions used for machining magnesium must be observed when machining titanium. Dry, powder-type fire extinguishing material should be kept available in titanium machining areas. Do not attempt to extinguish titanium fires with carbon dioxide (CO_2). Fine particles of titanium are explosive when suspended in air.

1-109. Blanking and Piercing Titanium.

1-110. Blanking or piercing with a punch press produces results similar to those attained with stainless steel. The force required is greater and the tool life is less when blanking or piercing titanium. Punches and dies should be constructed of the best steel available and should be kept in the best possible condition. Hole punching should not be attempted if the ratio of the hole size to the material thickness is less than one to one.

1-111. Shearing Titanium.

1-112. Commercially pure and alloy titanium in gages of 0.016 to 0.080 inch may be sheared on all types of shears or nibblers. Sharp, closely adjusted steel blades of high quality must be used on all tools and considerably more power is needed. Gages of titanium thicker than 0.080 inch should be sawed, blanked, or cut by square shears.

1-113. Grinding Titanium.

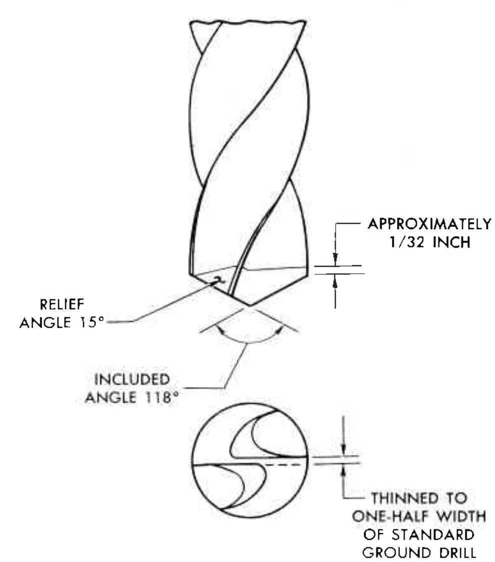
1-114. Grind methods ~~for titanium~~ are similar to those used for stainless steel. An aluminum oxide or silicon carbide abrasive wheel, with open coarse grit, gives the best results. No coolant is necessary for light intermittent grinding. If the grinding wheel is allowed to ride and light pressures are applied, the ground surface will burr heavily and work harden. A water-base, sodium nitrate, amine-type coolant directed on the cut from the bottom side will help prevent formation of heavy burrs. A feed pressure of 3 to 5 pounds against the abrasive wheel will lessen work hardening of the ground surface. The abrasive wheel should be checked periodically and replaced when the peripheral speed becomes appreciably less than 2800 surface feet per minute.

1-115. Routing Titanium.

1-116. Routing of one or two gage thicknesses of 0.016-inch commercially pure titanium is readily accomplished. Stack routing of up to eight thicknesses becomes increasingly difficult due to formation of heavy burrs which may be forced between the stacked sheets. A water-base coolant should be used when routing material with a combined thickness exceeding 0.040 inch. Router rate should be slower than 14,700 rpm when routing heavy gage or stacked material.

1-117. Drilling Titanium.

1-118. Drilling commercially pure titanium is best accomplished with an NAS 907, type C drill or a modified high-speed drill as shown on figure 1-31. Use low drill speeds and maintain a positive feed during entire drilling operation. A conventional chisel-point drill has a short work life and leaves heavy burrs around the edge of the hole. Sufficient pressure must be applied



CAUTION

THINNING DRILL TOO MUCH WILL RESULT IN SHORT DRILL LIFE.

NOTE:
THIS DESIGN FOR 0.250 AND SMALLER HOLES. FOR LARGER HOLE, INCREASE INCLUDED ANGLE FROM 118° TO 135° SPEED SHOULD BE FROM 15 TO 20 SFPM, AND FEED 0.002 TO 0.005 INCH PER REVOLUTION FOR BOTH VERSIONS.

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Figure 1-31. Modified High-Speed Drill for Titanium to keep the drill cutting until the hole is completely drilled. Do not allow the drill to ride in a partially drilled hole. Work hardening will result and further cutting will be extremely difficult. In cases where the titanium will be dimpled after drilling, the work piece should be securely clamped to prevent drill chatter marks which will cause titanium to crack under very small stress. If drill bushings are used, ample chip clearance should be allowed between the drill bushing and the work to reduce the amount of chips carried up through the bushing. If chip clearance is not maintained, choking of the bushing with chips will result in a high rate of drill breakage and excessive marginal wear on the drill.

1-119. FASTENERS.

1-120. All repairs to the outer surfaces of the airplane will require flush rivet installations. Countersinking, machine dimpling, and coin dimpling standards are shown on figure 1-28. Universal-head rivets used on internal structure should have the manufactured head in contact with the thinnest sheet in the joint wherever possible. Hole sizes for protruding head rivets are the same as specified for flush rivets on figure 1-28. Shop-formed rivet heads may be installed at an angle not to exceed 10 degrees from the perpendicular when riveting tapered parts. Protruding-type manufactured rivet heads may also be installed with a 10-degree maximum angular variation. No angular variations are permitted for flush-head rivets. Whenever possible, pick up existing rivet patterns with the same size and type of rivets used in the original construction of the parts involved. If the existing holes in the structure are enlarged past the limits given on figure 1-28, rivets of the next larger size may be installed, provided requirements for minimum rivet edge distance and spacing, as shown on figure 1-32, are met. Oversize fastener holes which indicate a need for a fastener two sizes larger than the original, or create short pitch and edge distance problems, must be submitted to an aeronautical structures engineer for evaluation. Higher strength fasteners may be substituted in the order listed in Table 1-XXII. Strengths of bolts, screws, and rivets are given in Tables 1-XXIV through 1-XXXV for the exclusive use of personnel assigned to design of repairs. Refer to Tables 1-II through 1-XVII for complete data regarding rivets required per inch of seam in various types of materials, and see figure 1-32 for rivet application data. Rivet basic code and numbering is defined in Table 1-XXIII.

1-121. Rivet Patterns.

1-122. Figure 1-32 shows rivet layout patterns. When determining the rivet pattern for a repair, the following terms will be considered:

a. Row: A line of rivets parallel to a seam or splice.

b. Row Spacing: The distance between rows when more than one row of rivets are installed.

c. Stagger: The eccentric or offset location of rivets in adjacent rows.

d. Edge Distance: The distance from the edge of the riveted sheet to the center line of the rivet hole.

e. Pitch: The center-to-center distance between rivets in any row of rivets.

1-123. Rivet Shank Diameter.

1-124. Should the extent of the damage or the required method of repair present some doubt regarding the diameter of a rivet to be used, the accepted rule is to use a rivet diameter approximately three times the thickness of the material to be repaired. See figure 1-32 for rivet pitch and edge distance information.

1-125. Rivet Length.

1-126. The rivet length must provide sufficient material to produce a satisfactory shop-formed head. The shop-formed head should be approximately 1½ times shank diameter in width and one-half times shank diameter in height. To obtain these limits the rivet shank should protrude through the material approximately 1½ times its diameter before driving. See figure 1-33 for rivet nomograph illustration.

1-127. Rivet Pitch.

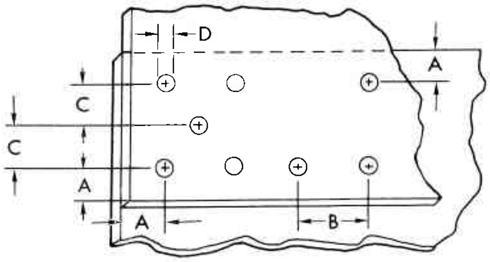
1-128. A minimum rivet pitch of four rivet diameters has been established; however, when a repair requires the next heavier gage of material or where three or more rows of rivets are required, the pitch may be reduced but should never be less than three rivet diameters. Since the pitch of four rivet diameters will reduce the strength of the material to approximately 75 percent along the center line of a row of rivets, care must be exercised at all times to avoid weakening the structure below its original strength. When more than one row of rivets are used they should, whenever possible, be staggered to retain the maximum strength of the repair. See figure 1-32 for rivet pattern layout.

1-129. Aluminum Alloy Rivets.

1-130. Standard solid aluminum alloy rivets are described below. These rivets are manufactured with AN470 universal heads or MS20426, 100-degree countersunk heads. Refer to Table 1-XXII for rivet substitution data and to Table 1-XLIV for standard drill sizes and hole diameter limits.

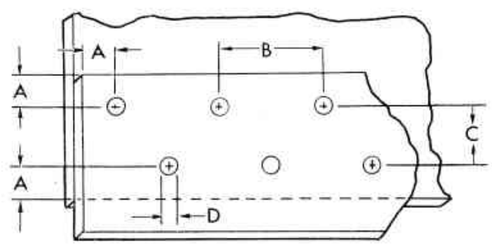
a. "B" Rivets: 5056-F material. Relatively soft, non-heat treatable rivets. These rivets are used only in nonstressed areas or for the purpose of attaching magnesium skins. Refer to Table 1-XV for the number of rivets required per inch of seam.

b. "AD" Rivets: 2117-T3 material. Most commonly used rivet, particularly where flush application is required. Refer to Tables 1-III and 1-IV for rivets required per inch of seam.



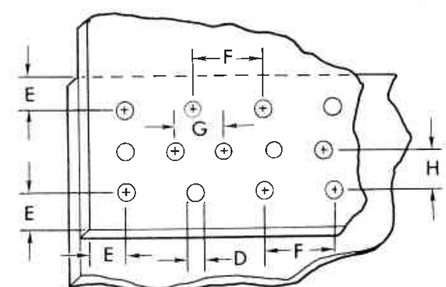
PROTRUDING HEAD OR DIMPLED SKIN

A. Edge distance (2D minimum).
B. Pitch or spacing (4D minimum).
C. Row spacing (2.5D minimum).
D. Rivet diameter.



COUNTERSUNK SKIN

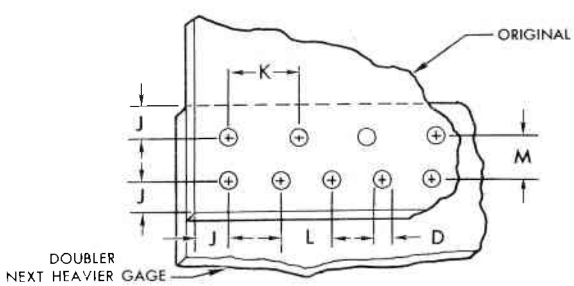
A. Edge distance (2D minimum).
B. Pitch or spacing (6D minimum).
C. Row spacing (3.5D minimum).
D. Rivet diameter.



PROTRUDING HEAD OR DIMPLED SKIN REPAIR

Patch or doubler same gage as original. When 3 or more rows of rivets are used the pitch of the center row may be 3D if the row spacing is 3.5D.

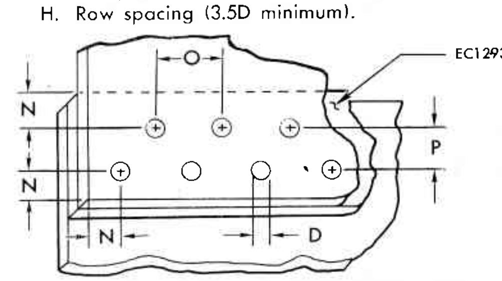
D. Rivet diameter.
E. Edge distance (2D minimum).
F. Pitch, outside rows (4D minimum).
G. Pitch, center rows (3D minimum).
H. Row spacing (3.5D minimum).



PROTRUDING HEAD OR DIMPLED SKIN REPAIR

Patch or doubler one gage heavier than original. When 2 or more rows of rivets are used the pitch of the inner row may be 3D.

D. Rivet diameter.
J. Edge distance (2D minimum).
K. Pitch, outer row (4D minimum).
L. Pitch, inner row (3D minimum).
M. Row spacing (2.5D minimum).



REPAIR OF FUSELAGE SKIN IN PRESSURIZED AREA USING COUNTERSUNK HEAD RIVETS IN DIMPLED SHEET

Repair to be standard except as noted:

D. Rivet diameter.
N. Edge distance (2D minimum).
O. Pitch (4D minimum +10 per cent maximum).
P. Row spacing (2.5D minimum).

Pattern must consist of at least 2 rows with rivets staggered. EC1293 must be applied between all faying surfaces.

NOTE:
THESE MINIMUM VALUES ARE APPLICABLE TO ALL RIVETED REPAIRS UNLESS OTHERWISE NOTED. WIDER SPACING IS GENERALLY DESIRABLE.

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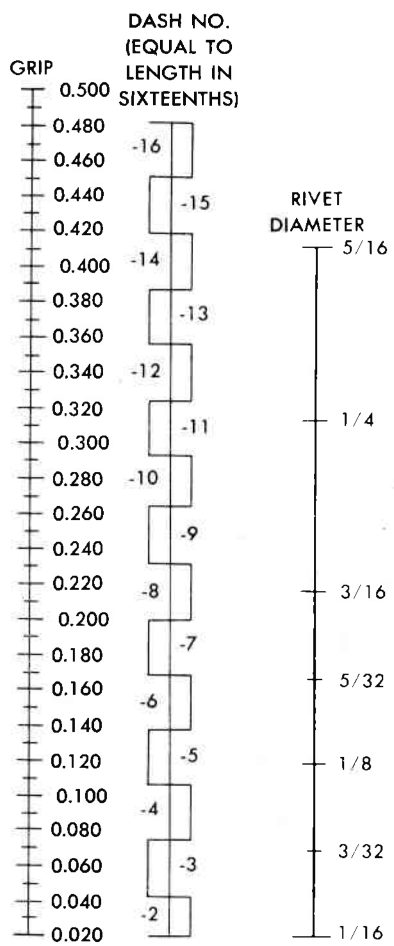
Figure 1-32. Standard Minimum Rivet Spacing

RIVET NOMOGRAPH



Instructions: Place a straight edge on the Nomograph from the point on the rivet diameter scale indicating the diameter of the rivet to be used, to the point on the grip scale indicating the required grip. The dash number for the correct rivet length will be found at the intersection of the straight edge and the dash number scale. In cases where the straight edge intersects the point between the two dash numbers, use the higher of the two numbers.

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Figure 1-33. Rivet Nomograph — Ratio of Length to Grip

c. "DD" Rivets: 2024-T31 material. Strongest solid aluminum alloy rivet. These rivets harden rapidly after heat treatment; however, the hardening process may be suspended by storing the rivets in a refrigerator at −23°C (−10°F). "DD" rivets must be driven within 30 minutes after removal from refrigerator. Rivets kept out of the refrigerator for more than 30 minutes MUST be reheat treated. DO NOT RETURN HARDENED RIVETS TO REFRIGERATOR. Since fresh, soft rivets form better than reprocessed rivets, do not remove large quantities of rivets from the refrigerator. Rivets shall be moisture-free upon installation. Refer to Tables 1-III and 1-IV for rivets required per inch of seam.

1-131. Straylor Rivets.

1-132. The Straylor rivet is a high-bearing, sealing-head rivet designed primarily to insure a liquid-tight seal at the riveted joint. The manufactured head of the rivet is designed to fit nearly flush within a counter-bored hole in the outer surface of fuel-tight skins and is milled after driving. A soft aluminum foil washer is installed under the head of each rivet to aid in producing a liquid-tight seal. These rivets are made of 2024-T31 aluminum alloy that hardens rapidly after heat treatment; however, the hardening process may be suspended by storing the rivets in a refrigerator at −23°C (−10°F). Straylor rivets must be driven within 30 minutes after removal from refrigerator. Rivets kept out of the refrigerator for more than 30 minutes MUST NOT be reheat treated. DO NOT RETURN HARDENED RIVETS TO REFRIGERATOR. Since fresh, soft rivets seal best, do not remove large quantities of rivets from the refrigerator. Rivets shall be moisture-free upon installation. The manufactured heads of these rivets shall be milled after driving to within the tolernance shown on figure 1-25 for aerodynamic smoothness. Mill with a standard ½-inch mill equipped with a guide and stop. Each rivet head contains a special dimple that is partly cut away after milling, thus indicating the depth of the counterbore in the skin. Specification MIL-C-5541 chemical film or equivalent shall be applied to rivet heads after milling. A piece of scrap should be used to test and adjust the counterbore tool for correct depth before counterboring repair parts. Refer to Tables 1-XVI and 1-XVII for rivets required per inch of seam and see figure 1-34 for installation data.

1-133. Removal of Straylor Rivets.

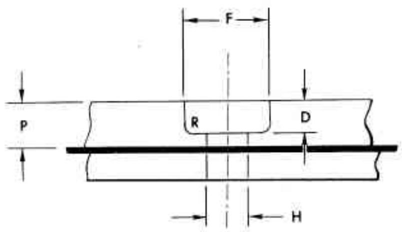
1-134. Use remaining portion of inspection dimple to locate center of manufactured head and proceed as shown on figure 1-35.

1-135. Monel Rivets.

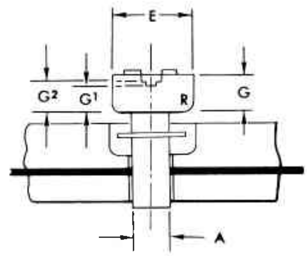
1-136. Monel rivets have excellent corrosion and heat resistant properties. They may be used in high-stressed areas of the structure and are particularly suitable for attaching titanium in areas subjected to high temperatures. Refer to Tables 1-VIII through 1-XIV for rivets required per inch of seam.

1-137. Corrosion Resistant Steel Rivets.

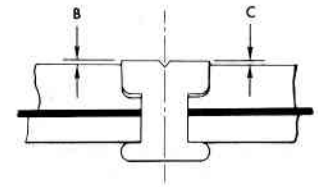
1-138. Corrosion resistant steel rivets are very high-strength rivets with a hollow end in the solid shank to aid in forming the shop head. These rivets are used in areas subjected to high temperatures and are



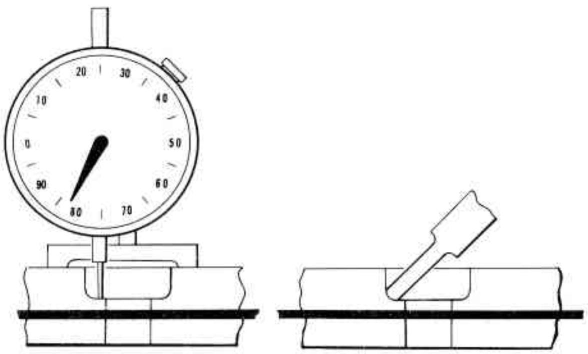
RIVET HOLE DIMENSION
SEE TABLE I



RIVET DIMENSIONS
SEE TABLE II

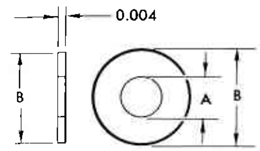


RIVET DIMENSION AFTER DRIVING
SEE TABLE I

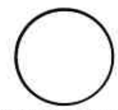


DIAL DEPTH GAGE RADIUS GAGE

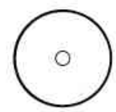
SEE TABLE IV



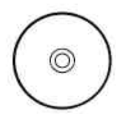
WASHER DIMENSIONS
SEE TABLE III



RIVET NOT SEATED
OR NOT COUNTERBORED
DEEP ENOUGH



RIVET PROPERLY
INSTALLED



COUNTERBORE
TOO DEEP

NOTES:
1. BOTH THE DRILLED HOLE AND THE COUNTERBORE MUST HAVE THE SAME CENTER.
2. USE DIAL DEPTH GAGE AND RADIUS GAGE TO DETERMINE ACCURACY OF COUNTERBORE.
3. USE SPECIAL COUNTERBORE TOOLS LISTED IN TABLE IV.
4. PROTRUSION OF RIVET MUST NOT BE OVER 0.004 AFTER MILLING.
5. TRACE OF DIMPLE TO 0.020 SHOULD BE VISIBLE AFTER MILLING.
6. THE ALUMINUM WASHER IS A MANDATORY SEAL TO INSURE A FUEL TIGHT SEAM.

TABLE I

RIVET DASH NO.		−6	−8
D COUNTERBORE DEPTH	+0.003 −0.003	0.072	0.080
F COUNTERBORE HOLE DIAMETER	+0.005 −0.000	0.330	0.440
H HOLE DIAMETER	+0.005 −0.000	0.190	0.253
P MINIMUM TOP SKIN THICKNESS		0.114	0.125
R COUNTERBORE RADIUS	+0.005 −0.005	0.030	0.040
B MINIMUM HEIGHT AFTER DRIVING		0.015	0.015
BEFORE MILLING		0.015	0.015
C AFTER MILLING		0.004	0.004

TABLE II

RIVET DASH NO.		−6	−8
A SHANK DIAMETER	+0.0025 −0.0005	0.187	0.250
E HEAD DIAMETER	+0.000 −0.005	0.330	0.440
G HEAD DEPTH	+0.011 −0.000	0.105	0.115
R RADIUS	+0.005 −0.005	0.030	0.040
G^1 RIVET DIMPLE	+0.003 −0.003	0.057	0.065
G^2 RIVET DIMPLE	+0.003 −0.003	0.082	0.090

TABLE III

RIVET DIAMETER	WASHER NUMBER	A +0.002 −0.000	B +0.002 −0.000
Q4326-6	Q7106-6	0.189	0.275
Q4326-8	Q7106-8	0.252	0.365

TABLE IV

RIVET DIAMETER	COUNTERBORE TOOL
Q4326-6	NO. CBCUM8J-752
Q4326-8	NO. CBCUM8J-753

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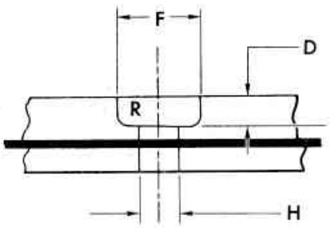
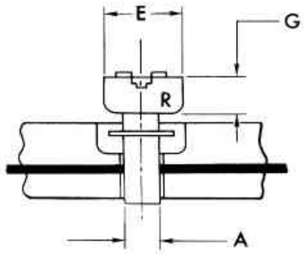
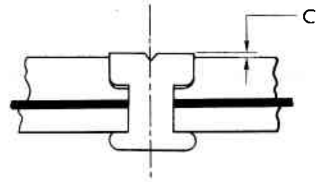
Figure 1-34. Installation of Straylor Rivets (Sheet 1 of 2)



RIVET HOLE DIMENSION
SEE TABLE I

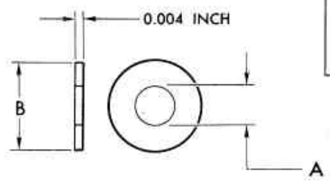


RIVET DIMENSIONS
SEE TABLES II AND III

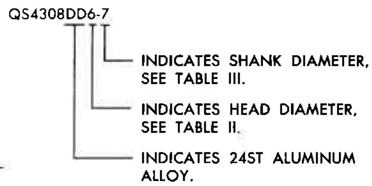


RIVET DIMENSION AFTER DRIVING
SEE TABLE I

NOTES:
1. REPAIR USAGE MUST BE APPROVED BY STRUCTURAL ENGINEERING AUTHORITIES.
2. BOTH DRILLED HOLE AND COUNTERBORE MUST HAVE SAME CENTER.
3. USE DIAL DEPTH GAGE AND RADIUS GAGE TO DETERMINE ACCURACY OF COUNTERBORE, REFER TO SHEET 1.
4. PROTRUSION OF RIVET MUST NOT BE OVER 0.004 INCH AFTER MILLING.
5. TRACE OF DIMPLE TO 0.020 INCH SHOULD BE VISIBLE AFTER MILLING, REFER TO SHEET 1 FOR RIVET DIMPLE IDENTIFICATION.
6. THE ALUMINUM WASHER IS A MANDATORY SEAL TO INSURE A FUEL TIGHT SEAM.



WASHER DIMENSIONS
SEE TABLE IV



RIVET CODE EXAMPLE

CAUTION

QS4308 SERIES, STRAYLOR REPAIR SEAL HEAD RIVETS DEFINITELY SHALL NOT BE USED EXCEPT WITH SPECIFIC APPROVAL OF STRUCTURAL ENGINEERING AUTHORITIES. REFER TO SHEET 1 FOR STANDARD STRAYLOR SEAL HEAD RIVET (Q4326 SERIES) DIMENSIONS AND INSTALLATIONS.

TABLE II

HEAD SIZE DASH NUMBER		DD-6	DD-8
E HEAD DIAMETER	+0.000 -0.005	0.330	0.440
R CORNER RADIUS	+0.005 -0.005	0.030	0.040
G HEAD THICKNESS	+0.003 -0.003	0.111	0.121

TABLE III

SHANK SIZE DASH NUMBER		-7	-9
A SHANK DIAMETER	+0.0025 -0.0005	0.219	0.281

TABLE I
RECOMMENDED DIMENSIONS FOR REPAIR

QS4308 DASH NUMBERS		DD6-7	DD8-9
D COUNTERBORE DEPTH	+0.003 -0.003	0.072	0.080
F COUNTERBORE HOLE DIAMETER	+0.005 -0.000	0.330	0.440
H HOLE DIAMETER	+0.005 -0.000	0.222	0.284
R COUNTERBORE RADIUS	+0.005 -0.005	0.030	0.040
C AFTER MILLING	+0.000 -0.004	0.004	0.004

TABLE IV

RIVET DIAMETER	WASHER NUMBER	A +0.002 -0.000	B +0.002 -0.000
QS4308DD6-7	Q7106-6-7	0.221	0.271
QS4308DD8-9	Q7106-8-9	0.283	0.365

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Figure 1-34. Installation of Straylor Rivets (Sheet 2 of 2)

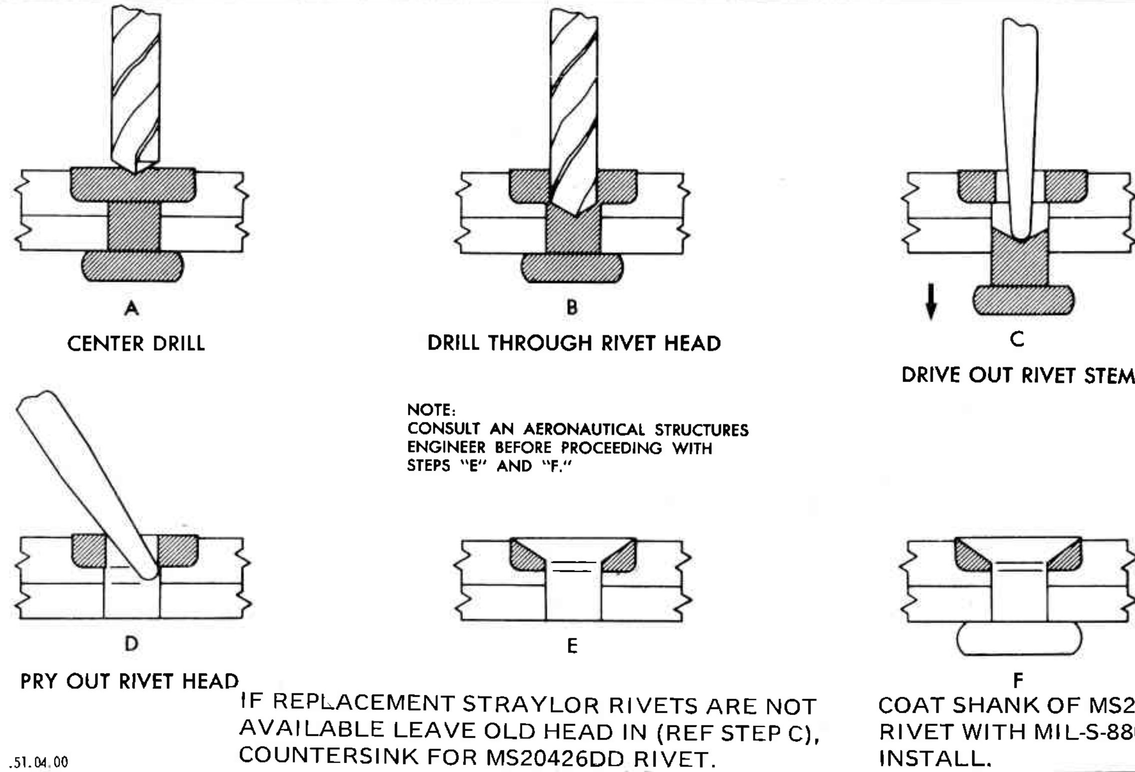


Figure 1-35. Removal of Straylor Rivets

particularly suitable for joining highly stressed parts of the structure. They are driven with ordinary riveting equipment (see figure 1-36). Refer to Tables 1-VII through 1-XIV for rivets required per inch of seam.

1-139. Hi-Shear Rivets.

1-140. Hi-Shear rivets are made of cadmium plated, heat-treated steel. These rivets are used in highly stressed joints in the structure. See figure 1-37 for an illustration of Hi-Shear rivet installation standards. Refer to Tables 1-XXXIV and 1-XXXVII for hole size, countersink, and spotface dimensions. Consult structural engineering authorities before spotfacing material around Hi-Shear rivet holes.

1-141. Huck Lockbolts.

1-141A. Huck Blind Bolts.

1-141B. Huck blind bolts are used in high-stressed areas where the use of a blind fastener is required. The type used is a pull-type, 100-degree countersunk head: serrated steel pin, steel sleeve and steel mechanical lock all of which are cadmium plated. This type of fastener requires the use of a special pulling tool for installation. See figure 1-39A for Huck blind bolt data.

1-142. Huck Lockbolts are used extensively in high-stressed areas where heavier gages of material occur. These lockbolts are notable for their high shear strengths and rigid tightening and locking characteristics. The two types of Huck lockbolts most commonly used are as follows:

a. Pull-type, protruding pan head or 100-degree countersunk head: Cadmium plated steel pin and 2024-T4 aluminum alloy collar. Requires pulling with special lockbolt pulling tool.

b. Stump-type, protruding pan head or 100-degree countersunk head: Cadmium plated steel pin and 2024-T4 aluminum alloy collar. This type is driven with a standard rivet gun and a special set for swaging the collar. It does not have the extended stem with the pull grooves. See figures 1-38 and 1-39 for installation and removal procedures for Huck lockbolts.

1-143. Blind Rivets.

1-144. The use of blind rivets is generally restricted to areas where lack of accessibility prohibits the use of the conventional type of rivet. In most instances, their use is confined to low or nonstressed areas of the structure and, when making repairs, it is recommended that prior structural engineering approval be obtained should their application be contemplated. In all cases of installation of blind rivets, the following rules are most important:

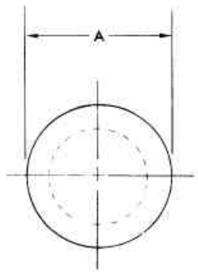
a. Holes must be held to prescribed limits.

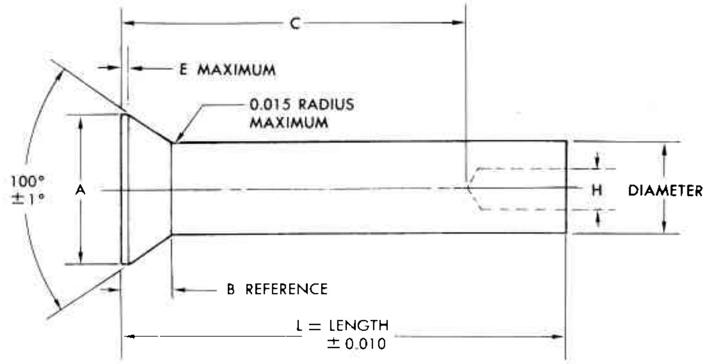
b. Burrs and chips must be cleaned from holes.

c. Material to be riveted must be clamped tightly together.

d. When rivet replacement is required, the next larger size shall be used.

e. Proper rivet grip lengths must be selected.





RIVET CODING EXAMPLE

Q4304-C7-12
- 0.75-INCH LONG
- 0.219-INCH DIAMETER
- CORROSION RESISTING STEEL RIVET, 100° COUNTERSUNK HEAD

DIMENSIONS								
DIAMETER $^{+0.003}_{-0.001}$	0.094	0.125	0.156	0.187	0.219	0.250	0.312	0.375
A	0.170 $^{+0.002}_{-0.006}$	0.216 $^{+0.002}_{-0.006}$	0.278 $^{+0.002}_{-0.006}$	0.344 $^{+0.003}_{-0.007}$	0.403 $^{+0.003}_{-0.007}$	0.467 $^{+0.003}_{-0.007}$	0.555 $^{+0.003}_{-0.007}$	0.685 $^{+0.003}_{-0.007}$
B $^{+0.001}_{-0.003}$	0.036	0.042	0.055	0.070	0.075	0.095	0.106	0.134
C	—	—	—	(L−0.135)	(L−0.175)	(L−0.205)	(L−0.255)	(L−0.305)
H	—	—	—	0.098 0.083	0.114 0.099	0.130 0.115	0.161 0.146	0.193 0.178
E	0.008	0.008	0.010	0.010	0.010	0.012	0.012	0.012

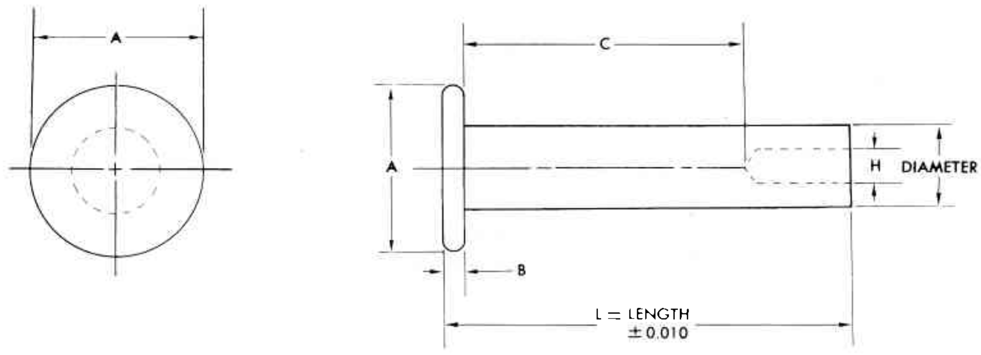
LENGTH	DASH NUMBERS AND DIAMETERS							
	0.094	0.125	0.156	0.187	0.219	0.250	0.312	0.375
0.188	3-3	4-3						
0.250	3-4	4-4	5-4	6-4				
0.312	3-5	4-5	5-5	6-5	7-5			
0.375	3-6	4-6	5-6	6-6	7-6	8-6		
0.438	3-7	4-7	5-7	6-7	7-7	8-7		
0.500	3-8	4-8	5-8	6-8	7-8	8-8	10-8	
0.625	3-10	4-10	5-10	6-10	7-10	8-10	10-10	12-10
0.750	3-12	4-12	5-12	6-12	7-12	8-12	10-12	12-12
0.875	3-14	4-14	5-14	6-14	7-14	8-14	10-14	12-14
1.000	3-16	4-16	5-16	6-16	7-16	8-16	10-16	12-16
1.125	3-18	4-18	5-18	6-18	7-18	8-18	10-18	12-18
1.250	3-20	4-20	5-20	6-20	7-20	8-20	10-20	12-20
1.375	3-22	4-22	5-22	6-22	7-22	8-22	10-22	12-22
1.500	3-24	4-24	5-24	6-24	7-24	8-24	10-24	12-24
1.625	3-26	4-26	5-26	6-26	7-26	8-26	10-26	12-26

NOTES:

1. USE OF CORROSION RESISTING STEEL RIVETS IS RESTRICTED TO SPECIAL CASES IN CORROSION RESISTANT STEEL WHERE RESISTANCE TO CORROSION IS OF PRIME IMPORTANCE AND IN ALUMINUM ALLOY WHERE CLEARANCES AND OTHER DESIGN CONSIDERATIONS DO NOT PERMIT THE USE OF A SUFFICIENT NUMBER OF ALUMINUM ALLOY RIVETS TO OBTAIN THE DESIRED SHEAR VALUES.
2. USE OF THESE RIVETS IS NOT RECOMMENDED IN GRIP LENGTHS EXCEEDING ONE INCH UNLESS THE PROPOSED APPLICATION IS APPROVED BY AN AERONAUTICAL STRUCTURES ENGINEER.
3. THESE RIVETS HAVE BEEN CADMIUM PLATED IN ACCORDANCE WITH SPECIFICATION QQ-P-416, TYPE 1, CLASS A.
4. Q4304 AND Q4310 RIVETS ARE TO BE DRIVEN IN THE SAME MANNER AS ORDINARY SOLID SHANK RIVETS.

06.03.314-1

Figure 1-36. Q4304 and Q4310 Rivet Installation (Sheet 1 of 2)



RIVET CODING EXAMPLE

Q4310-C4-6
- 0.187-INCH LONG
- 0.125-INCH DIAMETER
- CORROSION RESISTING STEEL RIVET, FLAT HEAD

DIMENSIONS								
DIAMETER $^{+0.003}_{-0.001}$	0.094	0.125	0.156	0.187	0.219	0.250	0.312	0.375
A	0.187 ±0.009	0.250 ±0.012	0.312 ±0.016	0.375 ±0.019	0.437 ±0.020	0.500 ±0.025	0.625 ±0.031	0.750 ±0.037
B ±0.005	0.032	0.042	0.052	0.062	0.075	0.083	0.104	0.125
C $^{+0.025}_{-0.000}$	—	—	—	(L−0.135)	(L−0.175)	(L−0.205)	(L−0.255)	(L−0.305)
H	—	—	—	0.098 0.083	0.114 0.099	0.130 0.115	0.161 0.146	0.193 0.178

LENGTH	DASH NUMBERS AND DIAMETERS							
	0.094	0.125	0.156	0.187	0.219	0.250	0.312	0.375
0.188	3-3	4-3						
0.250	3-4	4-4	5-4	6-4				
0.312	3-5	4-5	5-5	6-5	7-5			
0.375	3-6	4-6	5-6	6-6	7-6	8-6		
0.438	3-7	4-7	5-7	6-7	7-7	8-7		
0.500	3-8	4-8	5-8	6-8	7-8	8-8	10-8	
0.625	3-10	4-10	5-10	6-10	7-10	8-10	10-10	12-10
0.750	3-12	4-12	5-12	6-12	7-12	8-12	10-12	12-12
0.875	3-14	4-14	5-14	6-14	7-14	8-14	10-14	12-14
1.000	3-16	4-16	5-16	6-16	7-16	8-16	10-16	12-16
1.125	3-18	4-18	5-18	6-18	7-18	8-18	10-18	12-18
1.250	3-20	4-20	5-20	6-20	7-20	8-20	10-20	12-20
1.375	3-22	4-22	5-22	6-22	7-22	8-22	10-22	12-22
1.500	3-24	4-24	5-24	6-24	7-24	8-24	10-24	12-24
1.625	3-26	4-26	5-26	6-26	7-26	8-26	10-26	12-26

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Figure 1-36. Q4304 and Q4310 Rivet Installation (Sheet 2 of 2)

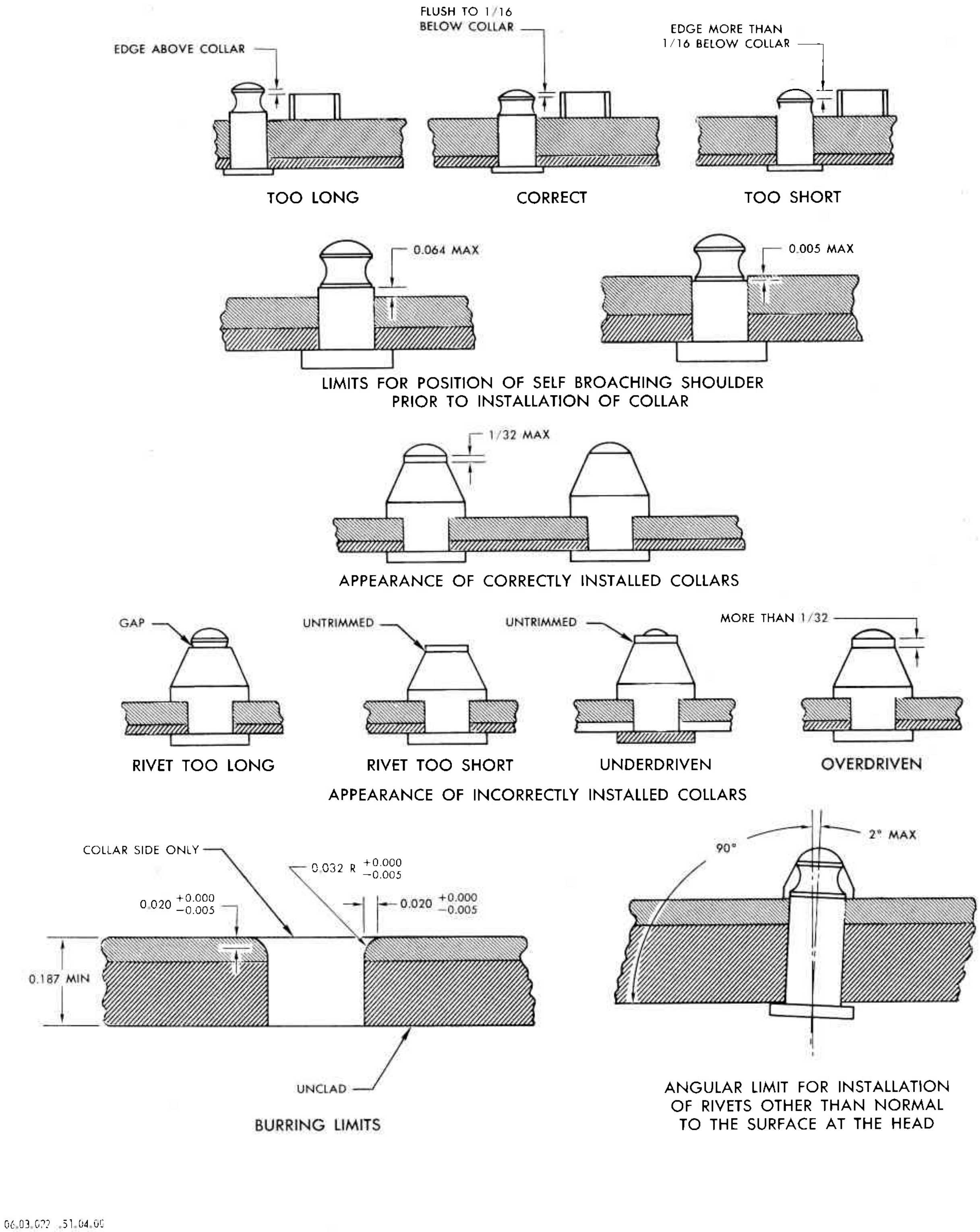
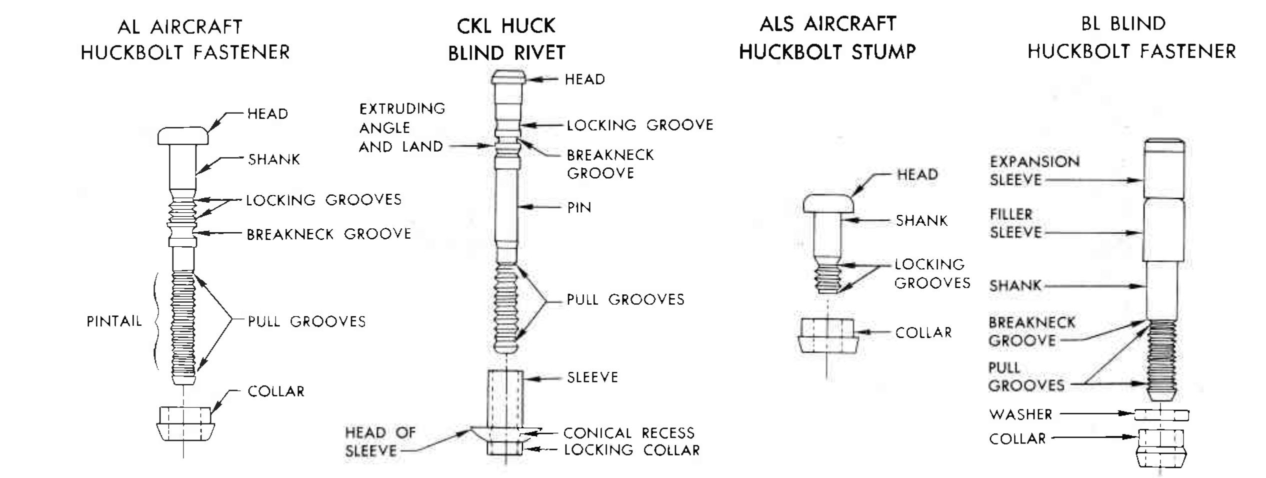
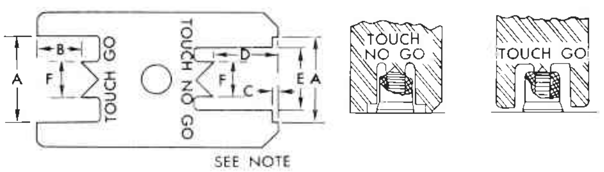
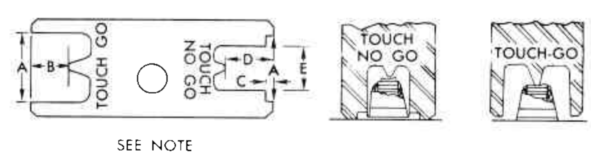


Figure 1-37. Installation of Hi-Shear Rivets







TYPE AL HUCKBOLT INSPECTION GAGE DIMENSIONS							
GAGE	HUCKBOLT	A	B	C	D	E	F
HG85-1	AL-E6 OR T6	3/8	0.190	0.038	0.291	0.303	0.164
HG85-2	AL-E8 OR T8	15/32	0.278	0.038	0.396	0.400	0.224
HG85-3	AL-E10	9/16	0.339	0.083	0.459	0.486	0.268
HG85-4	AL-E12	11/16	0.411	0.062	0.549	0.602	0.339
HG85-5	R1028-29-T10	9/16	0.309	0.136	0.421	0.486	0.268
HG85-6	R1028-29-T12	11/16	0.400	0.124	0.520	0.589	0.339
HG85-7	AL-E5	5/16	0.167	0.030	0.230	0.253	0.136

TYPE ALS HUCKBOLT INSPECTION GAGE DIMENSIONS						
GAGE	HUCKBOLT	A	B	C	D	E
HG34D-1	ALS-E6	3/8	0.238	0.0675	0.322	0.2972
HG34D-2	ALS-T6		0.218	0.0585	0.288	
HG34D-3	ALS-E8	15/32	0.307	0.0735	0.399	0.3942
HG34D-4	ALS-T8		0.301	0.0795	0.371	
HG34D-5	ALS-E10	9/16	0.399	0.0645	0.479	0.4992
HG34D-6	ALS-T10		0.384	0.1065	0.449	0.4862
HG34D-7	ALS-E12	11/32	0.496	0.1195	0.582	0.5892
HG34D-8	ALS-T12		0.480	0.1565	0.545	

TYPE AL HUCKBOLT PART NUMBER IDENTIFICATION

ALP P H – T 6 – 4

- GRIP LENGTH EQUALS SHANK LENGTH IN 16THS OF AN INCH.
- SHANK DIAMETER IN 32NDS OF AN INCH.
- MATERIAL T EQUALS HEAT TREATED ALLOY STEEL CADMIUM PLATED. E EQUALS 7075-T ALUMINUM ALLOY.
- TYPE H EQUALS PINTAIL LONGER FOR AUTOMATIC HOLE FILL. N EQUALS PINTAIL SHORTER FOR DRIVE FIT.
- HEAD P EQUALS PAN HEAD. 426 EQUALS AN426 COUNTERSUNK HEAD. 509 EQUALS CLOSE TOLERANCE HEAD.
- AIRCRAFT HUCKBOLT PIN.

TYPE AL HUCKBOLT IDENTIFICATION			
ALL SIZES	HEAD SHAPE	MATERIAL	IDENTIFICATION
ALPPH-T	PAN	STEEL	⊕ DEPRESSED
ALP426H-T	COUNTERSUNK	STEEL	⊕ RAISED
ALPPH-E	PAN	ALUMINUM	NONE
ALP426H-E	COUNTERSUNK	ALUMINUM	⊙ RAISED
ACT509H-T	COUNTERSUNK	STEEL	⊗ DEPRESSED
ACT509H-E	COUNTERSUNK	ALUMINUM	Ⓥ DEPRESSED

TYPE ALS HUCKBOLT PART NUMBER IDENTIFICATION

ALS F – E 6 – 4

- GRIP LENGTH EQUALS SHANK LENGTH IN 16THS OF AN INCH.
- SHANK DIAMETER IN 32NDS OF AN INCH.
- MATERIAL E EQUALS 7075-T ALUMINUM ALLOY. T EQUALS HEAT TREATED ALLOY STEEL CADMIUM PLATED.
- HEAD F EQUALS FLAT HEAD. ASCT509 EQUALS CLOSE TOLERANCE AN509 HEAD.
- AIRCRAFT HUCKBOLT STUMP.

TYPE ALS HUCKBOLT IDENTIFICATION			
ALL SIZES	HEAD SHAPE	MATERIAL	IDENTIFICATION
ALSF-T	FLAT	STEEL	⊕ DEPRESSED
ALSF-E	FLAT	ALUMINUM	NONE
ASCT509-T	COUNTERSUNK	STEEL	⊗ DEPRESSED
ASCT509-E	COUNTERSUNK	ALUMINUM	Ⓥ DEPRESSED

NOTE:
NORMALLY, A VISUAL INSPECTION IS SUFFICIENT WHEN CHECKING THE INSTALLATION OF HUCK-BOLTS. IF THERE IS ANY DOUBT, CHECK THE HUCK-BOLT WITH AN INSPECTION GAGE AS SHOWN.

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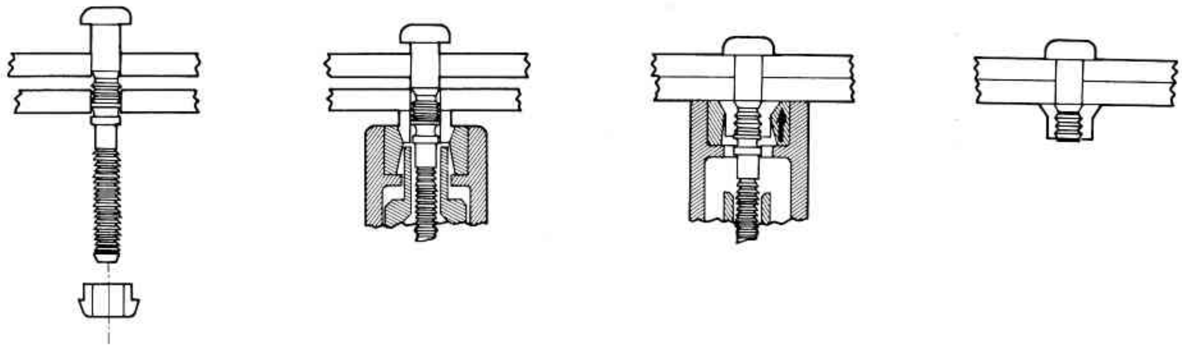
Figure 1-38. Installation of Huck Lockbolts (Sheet 1 of 3)



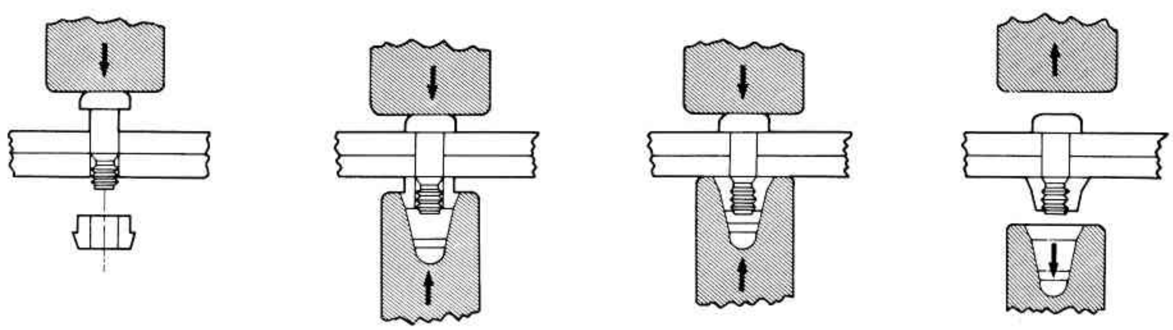
DRILL HOLE ACCORDING TO TABLE BELOW AND INSERT HUCKBOLT.

PLACE COLLAR OVER HUCKBOLT AND PLACE RIVET PULL GUN IN POSITION.

PULL RIVET GUN TRIGGER TO PULL MATERIAL TOGETHER AND TO SQUEEZE COLLAR INTO PLACE.

SQUEEZE TRIGGER UNTIL HUCKBOLT BREAKS. TO INSURE PROPER INSTALLATION, CHECK HUCKBOLT WITH INSPECTION GAGE AS SHOWN ON SHEET 1.

INSTALLATION PROCEDURE FOR TYPE AL HUCKBOLTS



DRILL HOLE ACCORDING TO TABLE BELOW AND INSERT HUCKBOLT.

PLACE COLLAR OVER HUCKBOLT. HOLD A BUCKING BAR AGAINST MANUFACTURED HEAD AND PLACE RIVET GUN NOSE SECTION OVER COLLAR.

HOLDING BUCKING BAR FIRMLY AGAINST MANUFACTURED HEAD, SQUEEZE RIVET GUN TRIGGER UNTIL NOSE SECTION OF RIVET GUN TOUCHES MATERIAL.

CHECK HUCKBOLT FOR PROPER INSTALLATION WITH INSPECTION GAGE AS SHOWN ON SHEET 1.

INSTALLATION FOR TYPE ALS HUCKBOLTS

	SUGGESTED HOLE PREPARATION FOR AL AND ALS HUCKBOLTS				
		DRIVE FIT		LOOSE FIT	
HOLE DIAMETER	PRE DRILL SIZE	DRILL SIZE	HOLE LIMITS	DRILL SIZE	HOLE LIMITS
5/32	NO. 26 (0.147)	NO. 20 (0.161)	0.161 TO 0.164	NO. 19 (0.166)	0.166 TO 0.171
3/16	NO. 18 (0.1695)	NO. 13 (0.185)	0.185 TO 0.188	NO. 11 (0.191)	0.191 TO 0.201
1/4	NO. 1 (0.288)	6.2MM (0.2441)	0.244 TO 0.247	1/4 (0.250)	0.250 TO 0.260
5/16	L (0.290)	7.8MM (0.3071)	0.307 TO 0.372	5/16 (0.3125)	0.3125 TO 0.322
3/8	11/32 (0.343)	U (0.368)	0.369 TO 0.372	3/8 (0.375)	0.375 TO 0.385

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Figure 1-38. Installation of Huck Lockbolts (Sheet 2 of 3)

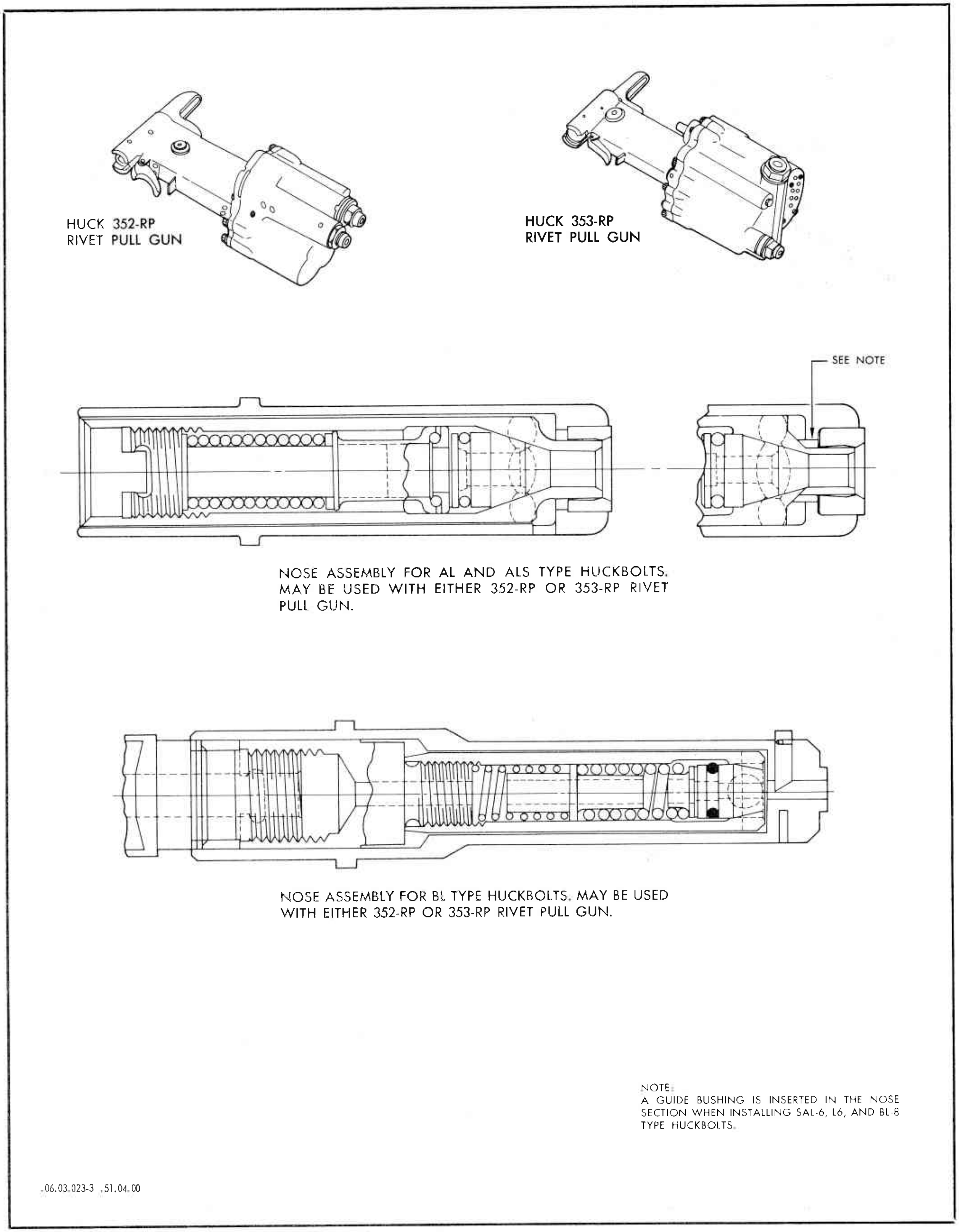

Figure 1-38. Installation of Huck Lockbolts (Sheet 3 of 3)

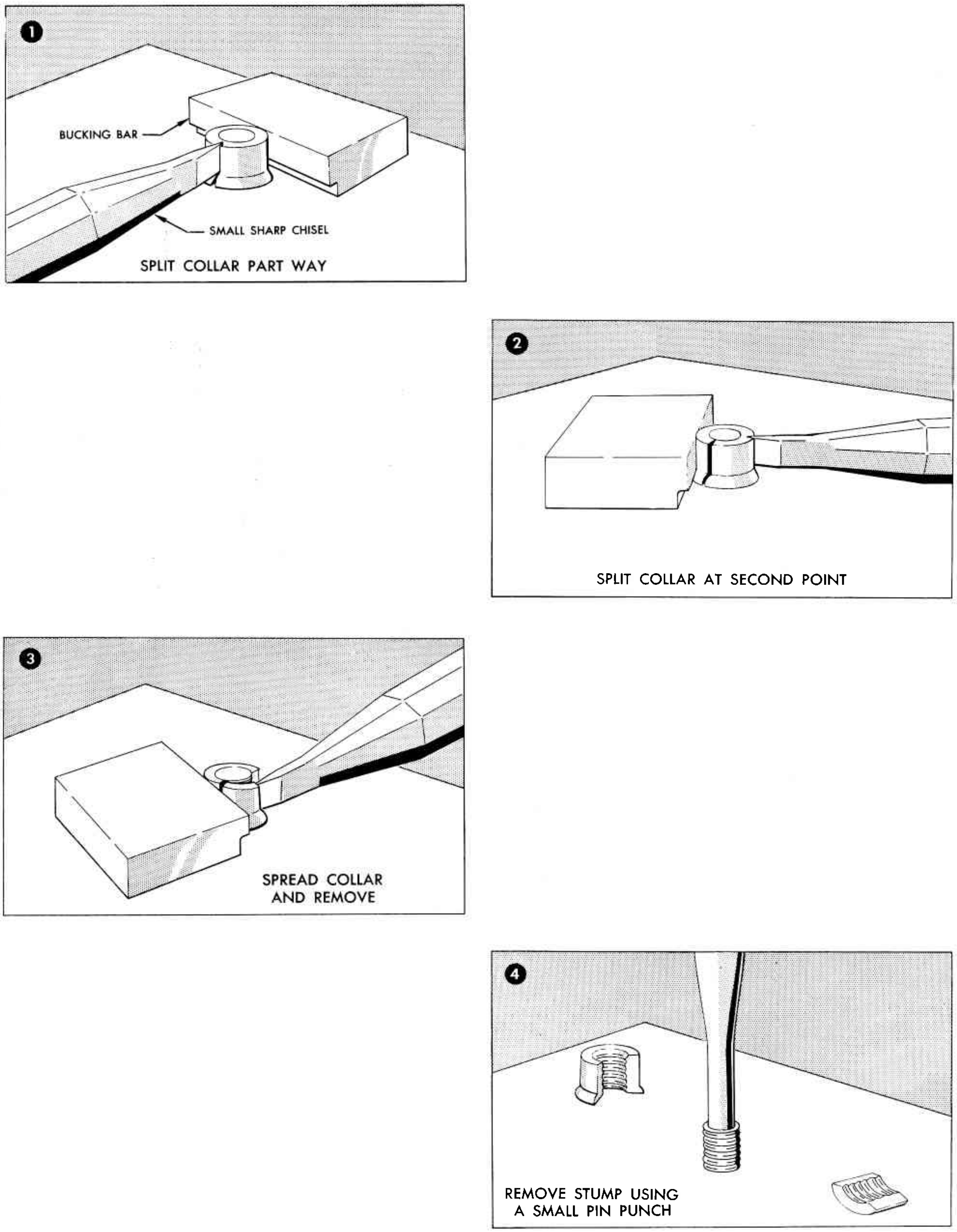

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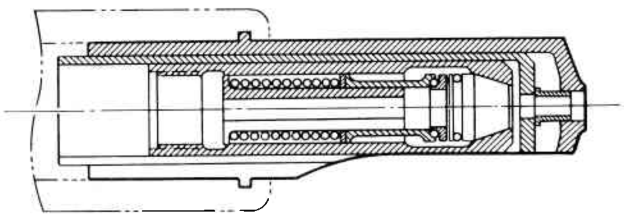
Figure 1-39. Removal of Huck Lockbolts

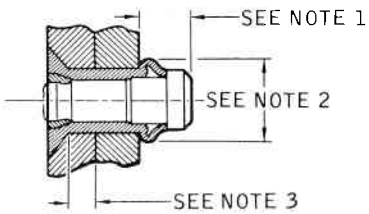


HUCK NOSE ASSEMBLY NO. 99-589 (3/16) OR NO. 99-600 (5/32) USED FOR COUNTERSUNK ALS HUCK BLIND BOLTS. USE WITH MODEL 200 HUCK PULL GUN.

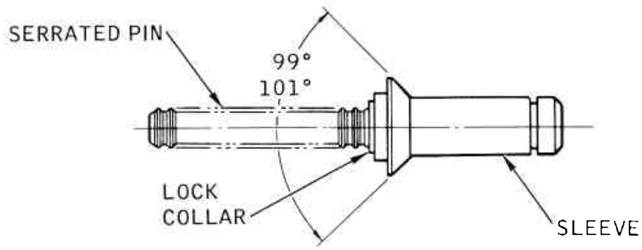
NOTE

HUCK PULL GUN MUST BE ADJUSTED FOR EACH DIAMETER BOLT. FOR 5/32-INCH BOLTS, SET GUN FOR A SHIFT PRESSURE OF 2005 (±100) PSI. FOR 3/16-INCH BOLTS, SET GUN FOR A SHIFT PRESSURE OF 3025 (±150) PSI. USE HUCK PRESSURE SETTING KIT NO. 101300 FOR SETTING SHIFT PRESSURE.

ALS COUNTERSUNK HUCK BLIND BOLT FASTENER



INSTALLED FASTENER



NOTES

1. FOR 5/32-INCH BOLT, 0.120-0.202 INCH. FOR 3/16-INCH BOLT, 0.146-0.231 INCH.
2. MAXIMUM DIAMETER NOT TO EXCEED DIAMETER OF MANUFACTURED HEAD.
3. MINIMUM SHEET BEARING FOR 5/32-INCH BOLT, 0.016 INCH. MINIMUM SHEET BEARING FOR 3/16-INCH BOLT, 0.020 INCH.

MS PART NUMBER:

MS90353-0507 = FASTENER, BLIND, HIGH STRENGTH, PULL TYPE, POSITIVE MECHANICAL LOCK, 100° COUNTERSUNK HEAD, ALLOY STEEL, 112 K.S.I.

- GRIP NUMBER IN 16TH INCHES.
- DIAMETER DASH NUMBER IN 32ND INCHES.
- MS PART NUMBER.

TYPE ALS HUCK BLIND BOLT IDENTIFICATION			
ALL SIZES	HEAD SHAPE	MATERIAL	IDENTIFICATION
MS90353 (B100-T)	COUNTERSUNK	STEEL	DEPRESSED

HUCK PART NUMBER:

B100-T 5-7

- GRIP NUMBER IN 16TH INCHES.
- DIAMETER DASH NUMBER IN 32ND INCHES.
- HUCK FASTENER IDENTIFICATION.

SUGGESTED HOLE PREPARATION FOR HUCK BLIND BOLTS			
HOLE DIAMETER	PRE DRILL SIZE	HOLE FINISH SIZE	HOLE LIMITS
5/32	NO. 26 (0.147)	NO. 19 (0.1660)	0.164 TO 0.167
3/16	NO. 18 (0.1695)	NO. 8 (0.1990)	0.199 TO 0.202

STRENGTH OF HUCK BLIND BOLTS		
PIN MATERIAL	HEAT TREATED ALLOY STEEL	
SLEEVE MATERIAL	HEAT TREATED ALLOY STEEL	
LOCK COLLAR MATERIAL	CRES OR CARBON STEEL	
LOADING / PIN DIAMETER	SHEAR STRENGTH-LBS	TENSILE STRENGTH-LBS
5/32	2340	1350
3/16	3450	2100

65R-986-44

Figure 1-39A. Installation of Huck Blind Bolts

When preparing the holes for rivet installation use a pilot and then a finish drill. Use a sharp drill held at a 90-degree angle to the work. Use prescribed riveting tools, and keep material clamped together at frequent intervals during application. Should a rivet be improperly installed, it should, in all cases, be replaced.

CAUTION

Blind rivets may be used inside of air intake ducts and forward of air induction system only when impossible to use solid rivets. Refer to Section X, paragraph 10-7 for use and substitution of blind rivets/fasteners in these areas.

1-145. DuPont Noiseless Explosive Rivets.

1-146. Two types of DuPont noiseless explosive rivets are used in the structure: the aluminum alloy, 5056 rivet, and the Nickel-L rivet. Refer to Tables 1-VI and 1-VII for rivets required per inch of seam. These rivets are used only where inaccessibility prevents the use of conventional type rivets. They are installed from one side by one operator using a rivet iron or spinning tool on the exposed head to explode the charge of powder in the rivet shank which expands the blind end of the rivet. A DuPont heating iron is recommended, but any iron may be used that is equipped with a temperature control that can be adjusted to cause rivet expansion within 1½ to 6 seconds. See figures 1-41 and 1-42 for operational procedures using heat or spin type explosive rivet tools. Test iron temperature with sample rivets. Holes must be carefully drilled by first using a pilot drill and then redrilling to finish size. Remove burrs and clean chips or shavings from parts. Refer to Table 1-XXXVIII for limits on drill and hole sizes. Rivets are installed as follows:

a. Insert rivet in hole. Be sure it is properly seated. Tap if necessary.

b. Apply heated riveting iron to top of rivet head with sufficient pressure to insure good contact. Keep tip at right angles to, and centered on, the rivet head. Within 1½ to 6 seconds, the charge will fire, expanding the rivet shank to fill the hole and form the upset head.

c. Remove tip of heating iron immediately after rivet expands. Rivet expansion is indicated by the sound of a muffled report.

NOTE

Discard rivets that do not expand within the 6-second time limit. Do not attempt to reheat.

The explosive charge in the DuPont rivet is mixed with a metal powder, and when rivets are driven too close to adjacent structure, particles of the metal powder will be driven into the surface by the explosion and may cause corrosion. In some cases the adjacent material may be protected by the use of a strip of metal or the pitted area can be coated with zinc chromate; however, Table 1-XXXIX, which shows the permissible distances from the ends of the rivets to the adjacent material, should be followed whenever possible. DuPont explosive rivets may be removed by the standard rivet removal procedure outlined in T.O. 1-1A-1, General Manual for Structural Repair.

WARNING

Do not expand explosive rivets in the presence of inflammable vapors, gases, or combustible dusts.

Refer to Tables 1-XL and 1-XLI for grip lengths. Sheets and members must be clamped tightly together while riveting is in process in order to insure a flush contact, as these rivets do not draw the sheets together.

1-147. Storage of Explosive Rivets.

1-148. The following precautions should be exercised in storing explosive rivets:

a. Keep rivets in boxes in which they are received.

b. Store in dry place, preferably in closed cabinet.

c. Avoid temperatures in excess of 48°C (120°F).

d. Keep open fires away from rivets.

1-149. Huck Blind Rivets.

1-150. The Huck blind rivet consists of a sleeve made of 6056 aluminum alloy having a brazier or countersunk head with a conical recess and locking collar at the outer end, and a pin made of 2014-RT aluminum that is pressed into the sleeve. The pin is provided with pull grooves that fit the jaws of the rivet gun. In installation, the pin is pulled through the sleeve to form a head on the blind side. The driven rivet must be expanded to completely fill the hole; therefore, it is important that the drilled holes and grip lengths be held to the limits specified in Tables 1-XLII and 1-XLIII. The installation of Huck blind rivets is illustrated on figure 1-40.

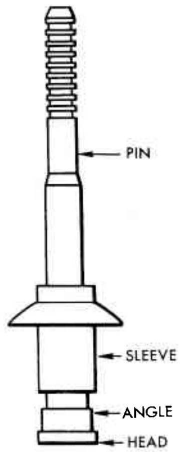
1-151. Cherry Blind Rivets.

1-152. Cherry rivets are the three following types:

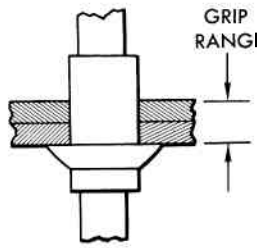
a. The hollow-type rivet in which the stem sets the rivet and is then broken, one portion falling out on the blind side.

b. The self-plugging type rivet, in which the stem remains in the manufactured head of the rivet and must be flush after the stem has been broken.

c. The pull-through type, in which the stem pulls through the rivet and collapses the mandrel on the stem.

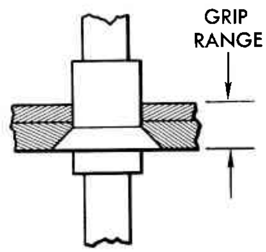


ASSEMBLED
HUCK BLIND RIVET



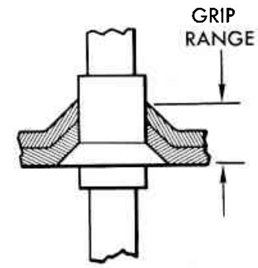
BRAZIER HEAD

Rivets are furnished in various sleeve lengths to accommodate various thicknesses of material. The range of the material thickness is referred to as the grip range.



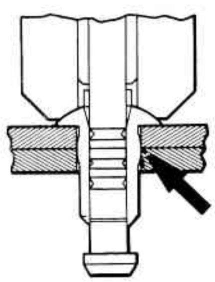
COUNTERSUNK HEAD
(SHEET MACHINE COUNTERSUNK)

To determine the grip range required for rivet application, the total thickness of the material should be measured with a hook scale through the rivet hole. To obtain proper grip range, clamp all parts involved together at point of measurement.



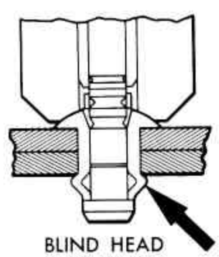
COUNTERSUNK HEAD
(SHEETS DIMPLED)

When rivet holes are to be dimpled the measurement must include the height of the dimple. The type of dimple and degree of bellmouth will affect the grip range.



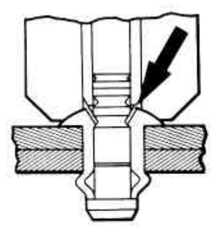
SLEEVE EXPANSION

During the initial part of the driving operation, the rivet gun pulls the extruding angle and land of the pin through the sleeve, expanding the sleeve to fill the hole.



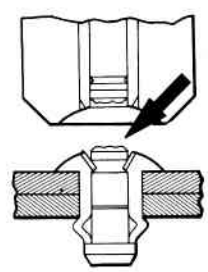
BLIND HEAD
FORMATION

The sleeve is squeezed between the head of the pin and the nose of the rivet gun to form the blind head.



POSITIVE
MECHANICAL LOCK

When blind head has been formed the gun forces the locking collar (at outer end of sleeve) into conical space between the recess in the head and the locking groove in the pin, rigidly locking the parts together.



PIN BREAK

The pin is then broken off in tension at the breakneck groove, flush with the head of the sleeve completing the installation of the rivet.

NOTES:
1. SEE TABLE I-VI FOR RIVETS REQUIRED PER INCH OF SEAM.
2. SEE TABLE I-XLII FOR DRILL SIZES AND HOLE DIAMETER LIMITS.
3. SEE TABLE I-XLIII FOR HUCK BLIND RIVET GRIP LENGTHS.

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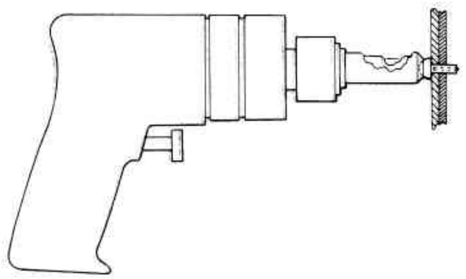
Figure 1-40. Installation of Huck Blind Rivets

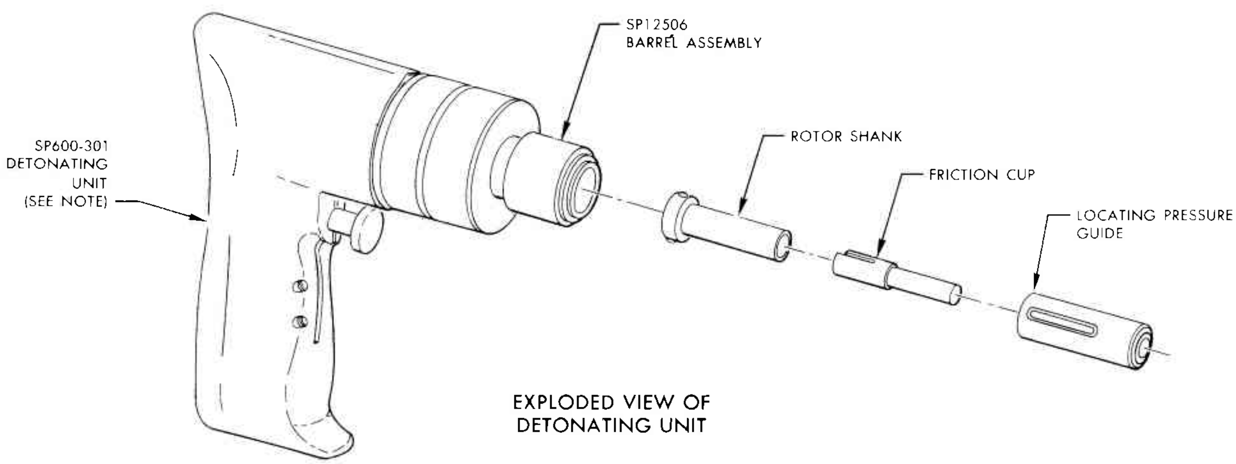


ASSEMBLED VIEW OF DETONATING UNIT

TABLE I

ROTOR SHANK		
PART NO.	LENGTH	LOCATING PRESSURE GUIDES USED
SP601-34	1.828	SP602-1A, -2A, -3A
SP601-35	2.828	SP602-1B, -2B, -3B
SP601-36	3.828	SP602-1C, -2C, -3C
SP601-37	4.828	SP602-1D, -2D, -3D
SP601-38	6.828	SP602-1E, -2E, -3E

NOTE:
IF SP600-301 DETONATING UNIT IS NOT AVAILABLE, ANY AIR MOTOR CAPABLE OF PRODUCING 15,000 TO 18,000 RPMS MAY BE USED BY ATTACHING A DRILL STOP ADAPTER.



EXPLODED VIEW OF DETONATING UNIT

TABLE II

LOCATING PRESSURE GUIDE			
PART NO.	LENGTH	RIVET SIZE	FRICTION CUPS USED
SP602-1A	1.125	1/8	SP603-20
SP602-1B	2.125		
SP602-1C	3.125		
SP602-1D	4.125		
SP602-1E	6.125		
SP602-2A	1.125	5/32	SP603-21
SP602-2B	2.125		
SP602-2C	3.125		
SP602-2D	4.125		
SP602-2E	6.125		
SP602-3A	1.125	3/16	SP603-22
SP602-3B	2.125		
SP602-3C	3.125		
SP602-3D	4.125		
SP602-3E	6.125		

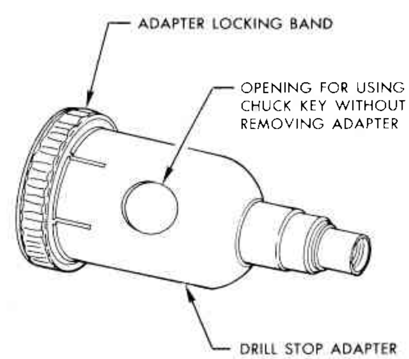


TABLE III

FRICTION CUP	
PART NO.	RIVET SIZE
SP603-20	1/8
SP603-21	5/32
SP603-22	3/16

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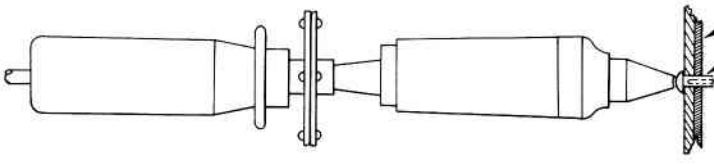
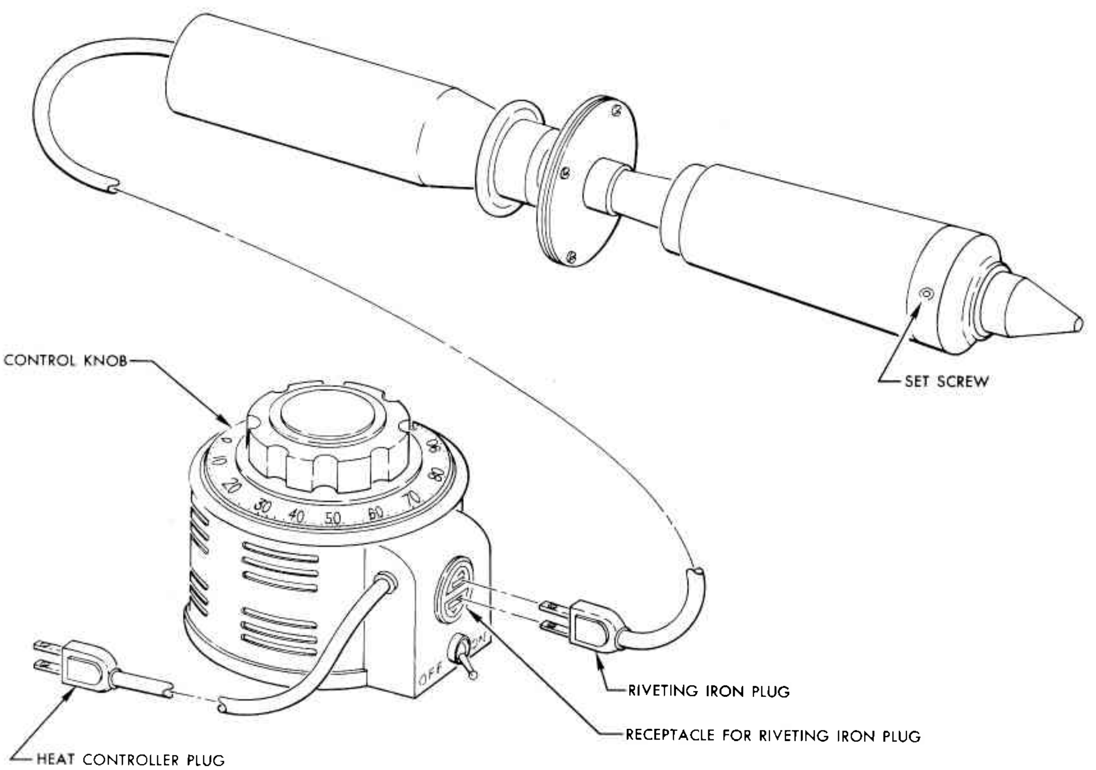
Figure 1-41. Explosive Rivet — Friction Detonator Tool



DUPONT EXPLOSIVE RIVETING IRON



HEAT CONTROLLER
FOR USE WITH
DUPONT RIVETING IRONS

NOTE:
THE HEAT CONTROLLER UNIT MAY BE PLUGGED INTO ANY 110 VOLT AC OUTLET.

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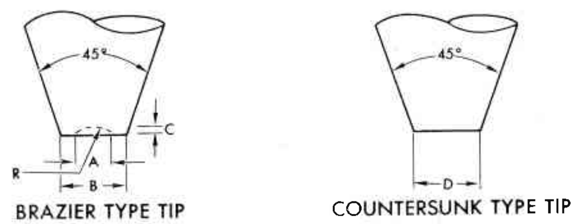
Figure 1-42. Explosive Rivet — Heat Detonator Tool (Sheet 1 of 2)

TABLE I

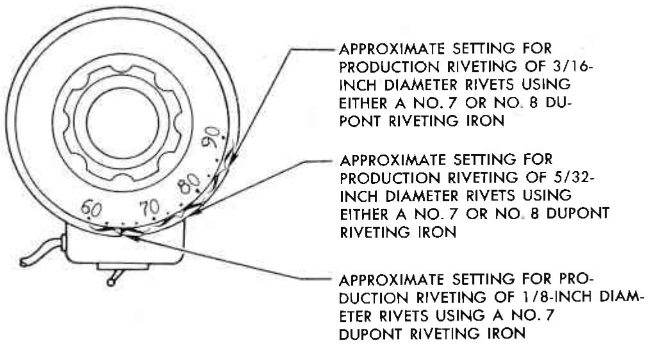
PROPER TIPS AND THEIR POINT DIMENSIONS FOR BRAZIER HEAD RIVETS						
RIVET DIAMETER	TIPS DESIGNATION		POINT DIMENSIONS			
	NO. 7	NO. 8	A	B	C	R(RADIUS)
1/8	7-B4	8-B4	0.170	0.220	0.020	0.181
5/32	7-B5	8-B5	0.210	0.260	0.025	0.226
3/16	7-B6	8-B6	0.250	0.300	0.030	0.272

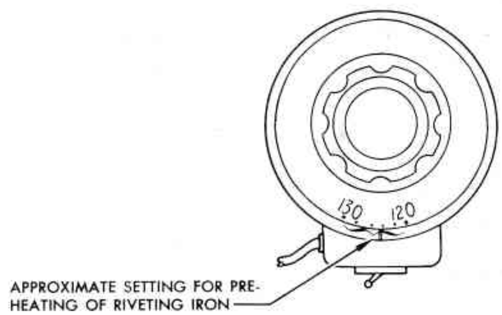
TABLE II

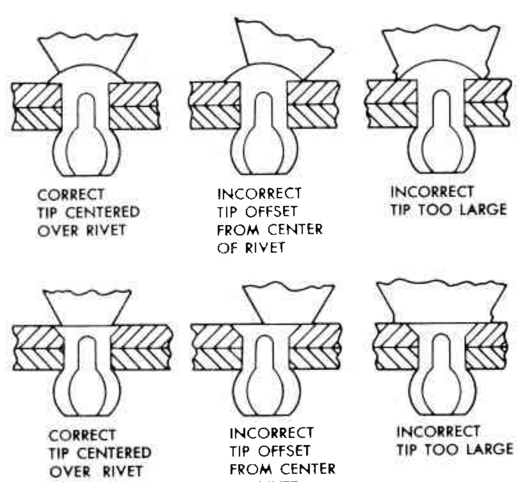
PROPER TIPS AND THEIR POINT DIMENSIONS FOR COUNTERSUNK HEAD RIVETS			
RIVET DIAMETER	TIP DESIGNATION		POINT DIMENSIONS
	NO. 7	NO. 8	D
1/8	7-C4	8-C4	0.135
5/32	7-C5	8-C5	0.175
3/16	7-C6	8-C5	0.200



DUPONT RIVETING IRON TIPS







CORRECT AND INCORRECT METHOD OF APPLYING TIPS TO RIVETS.

TABLE III

APPROXIMATE HEAT CONTROLLER SETTINGS			
RIVET DIAMETER	RIVET DESIGNATION (EITHER DR OR 56S)	HEAT CONTROL SETTINGS FOR	HEAT CONTROL SETTINGS FOR
		NO. 7 IRON	NO. 8 IRON
1/8	-134A -134-100	63 TO 74 78 TO 88	—
5/32	-173A -173-100	63 TO 71 74 TO 80	70 TO 72 78 TO 80
3/16	-204A -204-100	70 TO 75 83 TO 89	75 TO 77 85 TO 87

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Figure 1-42. Explosive Rivet — Heat Detonator Tool (Sheet 2 of 2)

The type most commonly used is the self-plugging rivet which consists of a mandrel or stem and a shank of 2117-T4 alloy or monel. A hand or power rivet tool may be used to pull the mandrel into the sleeve to form a head on the blind side. These rivets shall not be used except in cases of emergency, and then only by structural engineering approval.

CAUTION

Cherry blind rivets shall not be employed under any circumstances in pressurized areas or in the engine air induction system structure.

1-153. Jo-Bolts.

1-154. The three types of Jo-Bolts most commonly used are:

a. Blind, protruding hex head, nonmillable, type P; the bolt, nut, and sleeve are made of cadmium plated steel.

b. Blind, countersunk, millable-head, type FA; bolt and sleeve are made of cadmium plated steel, and the nut is made of 7075-T6.

c. Blind, countersunk, flush-head, type F; the bolt, nut, and sleeve are made of cadmium plated steel.

These Jo-Bolts may be installed by one operator inserting the bolt assembly in the rivet hole and using a power or hand tool fitted with a wrench and nose adapter to complete the application. The nose adapter holds the nut in place as the wrench adapter screws the bolt outward to pull the sleeve over the end of the nut, thus forming a head on the blind side. At the time the blind head is formed, the bolt stem is broken off flush with the outer surface of the nut. Bolt stems that do not break off within the limits shown on figure 1-43 indicate that the bolts are not of the proper grip length and should be removed and replaced with correct grip-length bolts. (See figure 1-44 for Jo-Bolt removal information.) Before milling, measure stems for the millable type FA bolts. The countersunk, millable-head type is milled down to the prescribed limits shown on figure 1-25.

CAUTION

These bolts shall not be employed in locations where the bolt or nut components, in the event they become loose, may be drawn into the engine air induction system.

These bolts should be limited to application in which the loads are primarily shear and where the bolts normally may not be subject to replacement. The break-off end of the bolts should be painted with zinc chromate primer after installation as a preventive against corrosion. This type of fastener shall be employed only when approved by an aeronautical structures engineer.

1-155. Blind Bolts (Hi-Shear Core Bolts).

1-156. Blind bolts (Hi-Shear core bolts) may be used instead of Jo-Bolts. Blind bolts should be used where shear loads exist and where only one side is accessible. See figure 1-46 for blind bolt installation procedure and figure 1-47 for bolt identification and usage.

1-156A. Hi-Lok Fasteners.

1-156B. Hi-Lok fasteners, as shown in Figure 1-47A, are used in areas where high-shear strength is required. The fastener consists of a threaded pin and a threaded collar. Hi-Lok pins are available with protruding heads, 100-degree countersunk heads, and 100-degree countersunk sealing heads (used with wet sealant). The pins are made from the following materials: cadmium-plated corrosion resistant steel (CRES), cadmium-plated or nickel cadmium-plated heat-resistant steel, and titanium alloys. Use of Hi-Lok fasteners ranges from attachment of brackets to joining of structural members. The threaded end of the pin is recessed to accommodate use of an allen hex wrench which is used to hold the pin stationary while the collar is being threaded on or off. The threaded collars are automatically tightened to the proper torque when the wrenching portion of the collar shears off during the tightening process. Tightening of the threaded collars is accomplished using an allen wrench and a box end or open end wrench. Re-use of Hi-Lok pins is permissible providing the pins have not been damaged during removal.

1-156C. Removal of Hi-Lok Fasteners.

1-156D. Remove fasteners as follows:

a. Using the proper size allen wrench to hold the pin, unscrew the threaded collar from the pin using pliers.

b. Remove pin from hole. It may be necessary to tap the pin with a nonmetallic hammer for pin removal.

1-156D. Installation of Hi-Lok fasteners.

1-156E. Install Hi-Lok fasteners as follows:

a. Prepare fastener hole in accordance with Figure 1-47B instructions for flush head or protruding head fastener.

b. Measure thickness of structures to be joined and select Hi-Lok pin of proper length.

Note

When material thickness falls between two standard fastener grip lengths; the next longer grip length fastener shall be used.

c. With structures in place and secured, insert pin into hole.

d. If fastener grip length is too long, one washer may be installed under either the nut or the fastener head (protruding type). To minimize dissimilar metal corrosion and to keep added washer weight to a minimum, washer usage shall be as follows:

1. Use AN960PD( ) anodized aluminum washers against aluminum, magnesium, and nonmetallic surfaces.

2. Use AN960-( ) cadmium-plated steel washers against cadmium-plated or painted steel surfaces.

3. Use AN960C( ) CRES washers against CRES or titanium surfaces.

e. Drive pins, installed in interference fit holes, using a rivet gun; check fastener for proper grip length and add washer if necessary.

f. Thread collar on to pin until finger tight.

g. Using and allen wrench and a box end wrench, tighten collar until collar wrenching portion shears off.

h. Check for a slight protrusion of the pin chamfered end beyond the collar. This indicates that the pin is of the proper length. Threads protruding beyond the collar indicates the need of a washer under the collar, or that the pin is of incorrect length.

1-157. Deutsch Drive Pin Blind Rivets.

Deutsch drive pin blind rivets are not used in production of the F-106. If they are used to make emergency repairs, their use should be restricted to nonstructural parts and they should be replaced as soon as approved types of blind fasteners become available. Figure 1-45 illustrates installation procedures for Deutsch drive pin blind rivets.

1-158. Helical Coil Inserts.

1-159. Helical coil inserts are screw-thread bushings coiled from wire, diamond-shaped in cross section, which accommodate both the internal thread of the tapped hole and the external thread of the bolt or screw. Helical coil inserts are made from 18-8 stainless steel, Specification AMS 7245, helically coiled wire. When installed in a heli-coil tapped hole, the insert has a permanent standard internal thread with a class 3B tolerance. The insert is made with a tang (a portion of the bottom coil that has been offset to provide a means of installation). This tang is left on the insert after installation, except when tang removal is necessary for screw or bolt clearance. A V-shaped notch is provided adjacent to the tang to provide a clean break-off point on the tang after installation. See figure 1-48 for identification and installation of helical coil inserts used on this airplane. See figure 1-49 for helical coil removal procedure.

1-160. Ball-Lok Receptacles.

1-161. Ball-Lok receptacles facilitate rapid installation and removal of ground support equipment. See figure 1-50 for replacement of Ball-Lok receptacles.

1-162. Bolt and Screw Attachments.

1-163. AN bolts may be used as substitutes for Hi-Shear rivets with specific structural engineering approval. Refer to Tables 1-VII through 1-XV for fasteners required per inch of seam. NAS bolts are high-tension bolts used only for attaching structural components where high stress loads are imposed on and transmitted through the bolts. Do not replace these bolts with AN bolts. Where practical, all screws and bolts installed in an up-and-down direction shall be installed with the head upward.

NOTE

Where applicable, all screws and bolts shall be installed with the head on the outside of

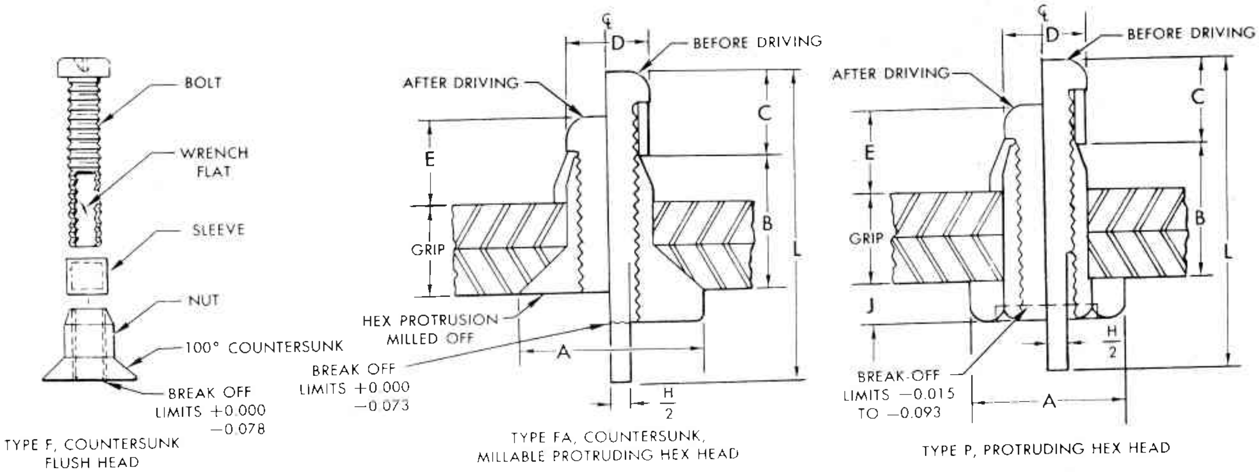

TABLE I

JO-BOLT INSTALLATION DATA										
PART NO.	DRILL SIZE	FINAL DRILL OR REAM DIA	JO-BOLT NOMINAL DIA	A	C	D	E	H	J	100° COUNTERSINK DIAMETER
F200	#15 (0.180)	MAX 0.202	3/16	• 0.385	0.262	0.199	0.209			MAX 0.390
		MIN 0.199		• 0.378	0.242	0.197				MIN 0.375
F260	#D (0.246)	MAX 0.263	1/4	• 0.507	0.315	0.260	0.292			MAX 0.510
		MIN 0.260		• 0.499	0.295	0.258				MIN 0.495
P200	#15 (0.180)	MAX 0.202	3/16	0.312	0.272	0.199	0.235	0.105	0.116	
		MIN 0.199		0.302	0.252	0.197		0.098	0.100	•
FA200	#15 (0.180)	MAX 0.202	3/16	*0.346	0.272	0.199	0.235	0.105		MAX 0.335
		MIN 0.199		*0.332	0.252	0.197		0.098		MIN 0.325
FA260	#D (0.246)	MAX 0.263	1/4	*0.472	0.334	0.260	0.292	0.137		MAX 0.460
		MIN 0.260		*0.458	0.297	0.258		0.129		MIN 0.450

*ACROSS HEX FLATS
• THEORETICALLY SHARP—(MAXIMUM LAND 0.20)

TABLE II

GRIP LENGTHS DATA												
DASH NO.	GRIP RANGE		F200		F260		P200		FA200		FA260	
	MIN	MAX	B ±0.015	L ±0.015	B ±0.015	L ±0.015	B ±0.015	L ±0.015	B ±0.015	L ±0.015	B ±0.015	L ±0.015
2	0.094	0.156	0.228	0.853	0.246	0.880	0.228	0.936	0.308	0.936	0.336	0.960
3	0.156	0.219	0.291	0.916	0.309	0.994	0.291	0.999	0.371	0.999	0.399	1.023
4	0.219	0.281	0.353	0.978	0.371	1.006	0.353	1.061	0.433	1.061	0.461	1.085
5	0.281	0.344	0.416	1.041	0.434	1.069	0.416	1.124	0.496	1.124	0.524	1.148
6	0.344	0.406	0.478	1.103	0.496	1.131	0.478	1.186	0.558	1.186	0.586	1.210
7	0.406	0.469	0.541	1.166	0.559	1.194	0.541	1.249	0.621	1.249	0.649	1.273
8	0.469	0.531	0.603	1.228	0.621	1.256	0.603	1.311	0.683	1.311	0.711	1.335
9	0.531	0.594	0.666	1.291	0.684	1.319	0.666	1.374	0.746	1.374	0.774	1.398
10	0.594	0.656	0.728	1.353	0.745	1.381	0.728	1.436	0.808	1.436	0.836	1.460

TABLE III

JO-BOLT DRIVING TOOL DATA					
JO-BOLT PART NO.	POWER TOOL	ADAPTER ASSEMBLY	NOSE ADAPTER	WRENCH ADAPTER	RATCHET HAND TOOL
F200	LOK-FAST MODEL 30200	PW300	PW3002	PW3001	HWF200
F260	LOK-FAST MODEL 30260	PW360	PW3602	PW3601	HWF260
P200	LOK-FAST MODEL 30200P	PW300P	PW3002P	PW3001	HWP200
FA200	LOK-FAST MODEL 30200A	PW300A	PW3002A	PW3001	HWA200
FA260	LOK-FAST MODEL 30260A	PW360A	PW3602A	PW3601	HWA260

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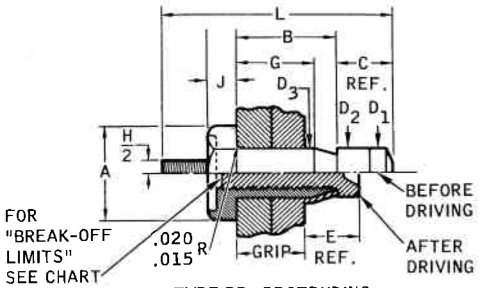
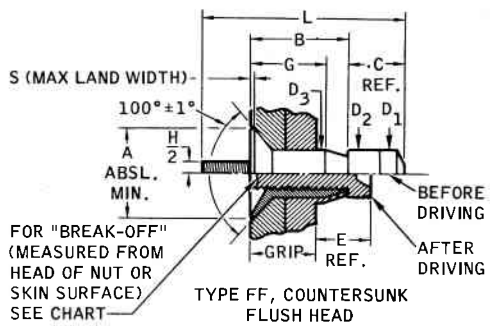
Figure 1-43. Installation of Jo-Bolts (Sheet 1 of 3)



TYPE PP, PROTRUDING HEX HEAD



TYPE FF, COUNTERSUNK FLUSH HEAD

TABLE I

JO-BOLT INSTALLATION DATA

PART NO.	DRILL SIZE	FINAL DRILL OR REAM DATA	JO-BOLT NOMINAL DIA	A	C	D_1	D_2	D_3	E	H	J	100° COUNTERSINK DIAMETER
NAS1669-08 (PP164)	#25	MAX 0.168	0.164	0.250	0.248	0.161	0.163	0.1645	0.215	0.086	0.096	-
		MIN 0.165		0.244	-	0.156	0.158	0.1625	-	0.081	0.086	-
NAS1669-3 (PP200)	#15	MAX 0.202	3/16	0.312	0.276	0.199	0.199	0.1990	0.245	0.104	0.113	-
		MIN 0.199		0.305	-	0.195	0.197	0.1970	-	0.099	0.103	-
NAS1670-3 (FF200)	#15	MAX 0.202	3/16	·0.385	0.276	0.199	0.199	0.1990	0.245	0.104	-	MAX 0.390
		MIN 0.199		·0.378	-	0.195	0.197	0.1970	-	0.099	-	MIN 0.375
NAS1670-6 (FF375)	#S	MAX 0.378	3/8	:0.762	0.488	0.3745	0.3745	0.3745	0.429	0.185	-	MAX 0.765
		MIN 0.375		:0.752	-	0.3660	0.3720	0.3725	-	0.180	-	MIN 0.750

· THEORETICALLY SHARP — (MAXIMUM LAND 0.015)
: THEORETICALLY SHARP — (MAXIMUM LAND 0.023)

TABLE II

GRIP LENGTHS DATA

DASH NO.	GRIP RANGE MIN.	GRIP RANGE MAX(G)	NAS1669-08 (PP164) B ±0.015	NAS1669-08 (PP164) L ±0.015	NAS1669-3 (PP200) B ±0.015	NAS1669-3 (PP200) L ±0.015	NAS1670-3 (FF200) B ±0.015	NAS1670-3 (FF200) L ±0.015	NAS1670-6 (FF375) B ±0.015	NAS1670-6 (FF375) L ±0.015
2	0.094	0.156	0.230	0.795	0.228	0.936	0.228	0.853		
3	0.156	0.219	0.293	0.858	0.291	0.999	0.291	9.916		
4	0.219	0.281	0.355	0.920	0.353	1.061	0.353	0.978	0.479	1.242
5	0.281	0.344	0.418	0.983	0.416	1.124	0.416	1.041	0.542	1.304
6	0.344	0.406	0.480	1.045	0.478	1.186	0.478	1.103	0.604	1.367
7	0.406	0.469	0.543	1.108	0.541	1.249	0.541	1.166	0.667	1.429
8	0.469	0.531	0.605	1.170	0.603	1.311	0.603	1.228	0.729	1.492
9	0.531	0.594	0.668	1.233	0.666	1.374	0.666	1.291	0.792	1.554
10	0.594	0.656	0.730	1.295	0.728	1.436	0.728	1.353	0.854	1.617
11	0.656	0.719	0.793	1.358	0.791	1.499	0.791	1.416	0.917	1.679
12	0.719	0.781	0.855	1.420	0.853	1.561	0.853	1.478	0.979	1.742

TABLE III

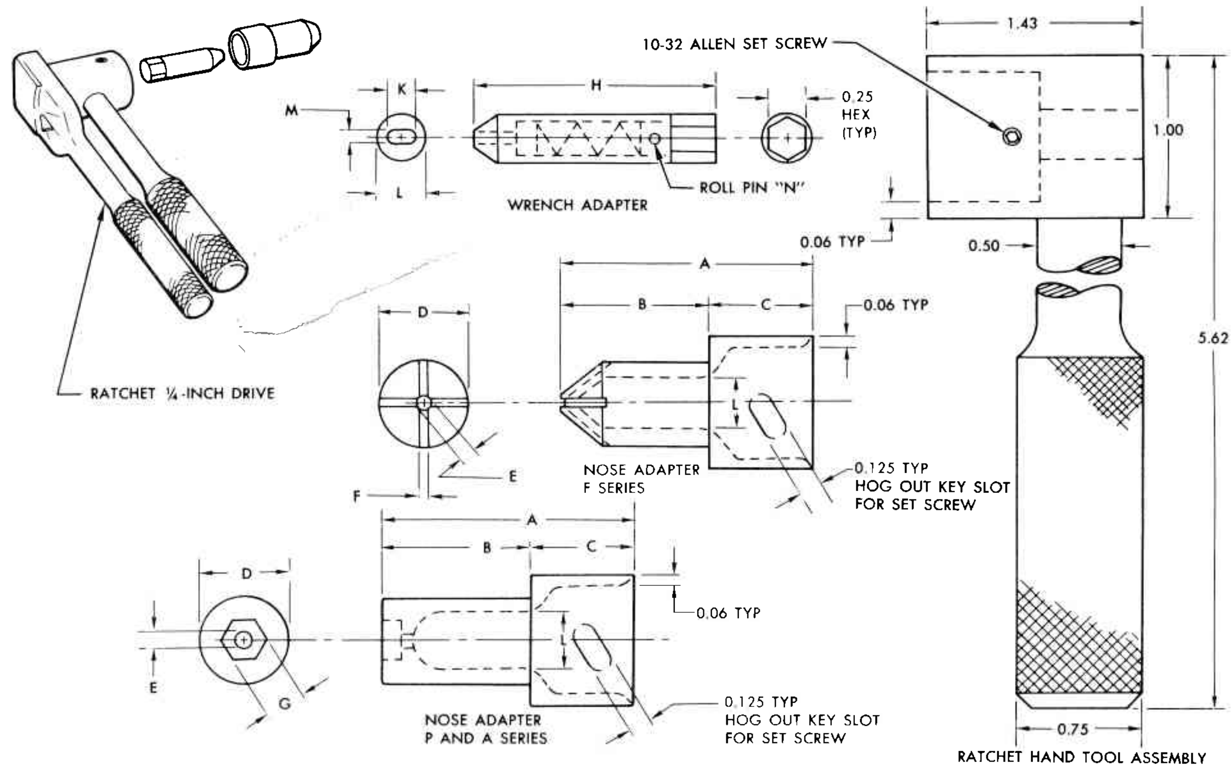
JO-BOLT DRIVING TOOL DATA

JO-BOLT PART NO.	POWER TOOL	ADAPTER ASSEMBLY	WRENCH ADAPTER	RACHET HAND TOOL
NAS1669-08 (PP164)	*LOK-FAST MODEL 302M WITH MASTER TORQUE DRIVER MTD302	TD165P	HW1650P	HW200
NAS1660-3 (PP200)	*LOK-FAST MODEL 302M WITH MASTER TORQUE DRIVER MTD302	TD200P	HW2000P	HW200
NAS1670-3 (FF200)	*LOK-FAST MODEL 302M WITH MASTER TORQUE DRIVER MTD302	TD200	HW2000	HW200
NAS1670-6 (FF375)	*LOK-FAST MODEL 302M WITH MASTER TORQUE DRIVER MTD302	PT375	HW3740	HW300

*LOK-FAST INC.
NEWPORT BEACH, CALIF.

65R-986-45

Figure 1-43. Installation of Jo-Bolts (Sheet 2 of 3)



JO-BOLT ADAPTER ASSEMBLY

JO-BOLT SERIES	RATCHET HAND TOOL ASSEMBLY	ADAPTER ASSEMBLY														
			NOSE ADAPTER	A ±.015	B ±.015	C ±.015	D ±.010	E ±.005	F ±.005	G ±.005	WRENCH ADAPTER	H ±.015	K ±.005	L ±.010	M ±.005	N
F200	HWF200	HW2000	HW2002	1.531	.906	.625	.500	.151	.037		NW2001	1.437	.142	.365	.107	.094
P200	HWP200	HW2000P	HW2002P	1.562	.937	.625	.500	.151		.316						
FA200	HWA200	HW2000A	HW2002A	1.542	.917	.625	.500	.151		.348						
PA200	HWA200	HW2000A	HW2002A	1.542	.917	.625	.500	.151		.348						
F260	HWF260	HW2600	HW2602	1.531	.906	.625	.500	.201	.046		HW2601	1.437	.193	.365	.138	.094
P260	HWP260	HW2600P	HW2602P	1.562	.937	.625	.500	.201		.382						
FA260	HWA260	HW2600A	HW2602A	1.565	.940	.625	.625	.201		.472						
PA260	HWA260	HW2600A	HW2602A	1.565	.940	.625	.625	.201		.472						
F312	HWF312	HW3120	HW3122	1.531	.906	.625	.555	.231	.065		HW3121	1.437	.193	.365	.155	.094
F312	HWP312	HW3120P	HW3122P	1.625	1.000	.625	.575	.231		.439						
F375	HWF375	HW3750	HW3752	1.555	.930	.625	.625	.270	.072 .070		HW3751	2.215	.260	.365	.181	.125
P375	HWP375	HW3750P	HW3752P	1.555	.930	.625	.670	.270	.070	.502						

NOTES:
1. RATCHET HAND TOOL ASSEMBLY AND JO-BOLT ADAPTER ASSEMBLY MAY BE MANUFACTURED LOCALLY OR PURCHASED FROM LOK-FAST INC., 2313 WEST OLIVE, BURBANK, CALIF.
2. TO MANUFACTURE TOOLS LOCALLY MAKE THE RATCHET HAND TOOL ASSEMBLY FROM 7075-T6 BAR STOCK. MAKE THE NOSE AND WRENCH ADAPTERS FROM ALLOY STEEL.

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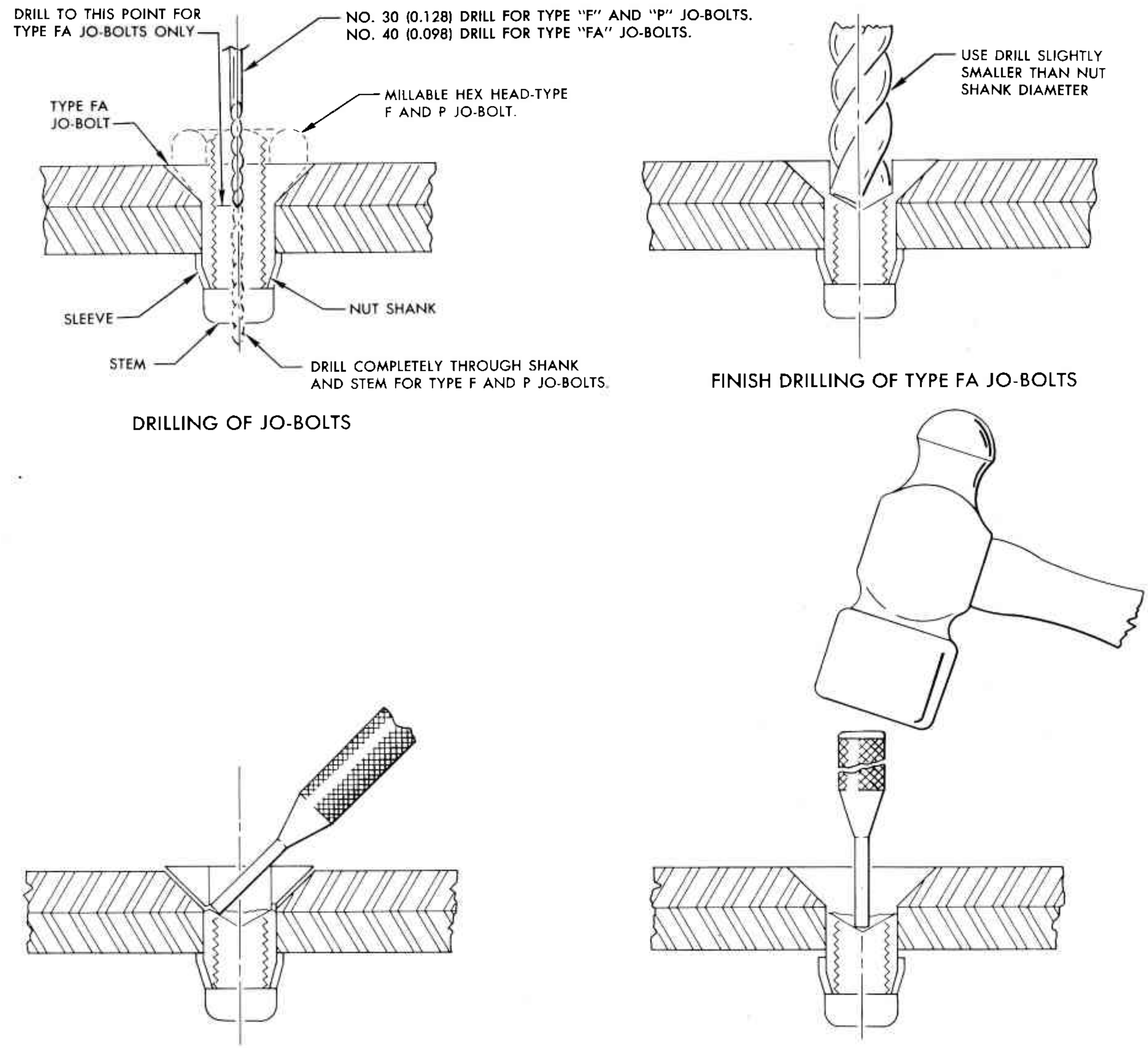
Figure 1-43. Installation of Jo-Bolts (Sheet 3 of 3)



REMOVAL PROCEDURE OF TYPE FA JO-BOLTS

a. Drill center of stem with a No. 40 (0.098) drill to a point slightly below the intersection of the nut head and shank.
b. Using a drill slightly smaller than nut shank, drill to intersection of nut head and shank. Use a No. 8 (0.199) drill or smaller for 3/16-inch diameter jo-bolts, and a G size (0.261) drill or smaller for 1/4-inch diameter jo-bolts.
c. Pry out nut head with punch.
d. Drive shank from hole using a punch and tapping slightly with a hammer.

REMOVAL OF TYPE F AND P JO-BOLTS

a. Drill center of stem, with a No. 30 (0.028) drill for 3/16-inch bolts or a No. 13 (0.185) drill for 1/4-inch bolts, completely through end of bolt shank and stem.
b. Punch sleeve and end of bolt into blind area.
c. Remove nut from hole with punch or hook.

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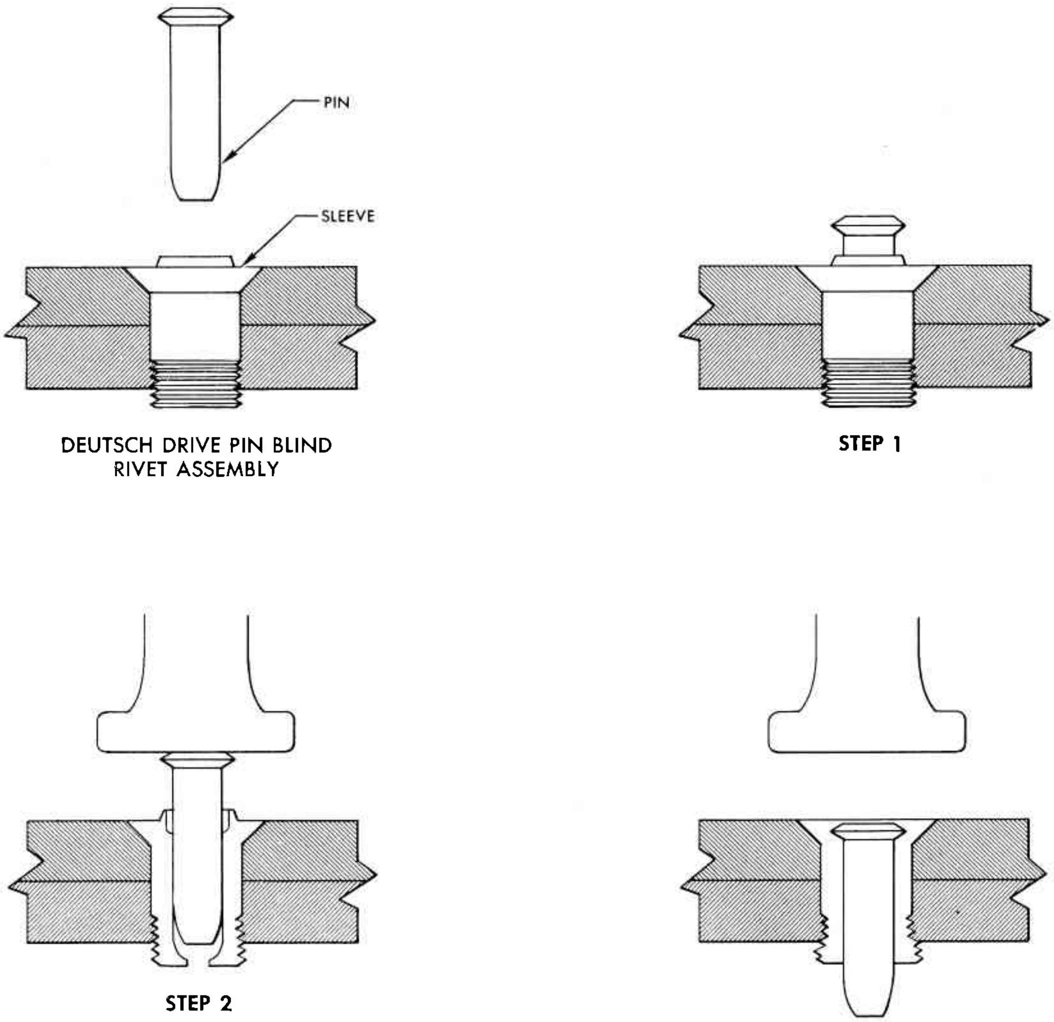
Figure 1-44. Removal of Jo-Bolts



PART NUMBER IDENTIFICATION

6950 – S – C10 – 6 SP

- SP = SHORT PIN — OMIT FOR STANDARD PINS
- 6 — GRIP LENGTH (IN 1/16")
- C10 — DIAMETER C = CLOSE TOL. 10 = 5/16"
- S — MATERIAL S = 303 STAINLESS STEEL, H = 17-4 PH STAINLESS STEEL
- 6950 — TYPE (COUNTERSUNK HEAD)

NOTE:
SEE TABLE 1-XLV AND TABLE 1-XLVI FOR FURTHER IDENTIFICATION OF DEUTSCH DRIVE PIN RIVETS. SEE TABLES 1-XLVII AND 1-XLVIII FOR HOLE SIZES AND GRIP RANGES.

INSTALLATION PROCEDURE

a. Place the proper length rivet for the material thickness into the hole. The rivet must fit firmly and snugly in its seat (**STEP 1**).
b. Use a light air hammer with a flat set or a hand hammer and drive the pin into its seat (**STEP 2**).
c. The pin expands the sleeve to form a blind head. The locking lip is pinned to lock the pin in the sleeve.

CAUTION

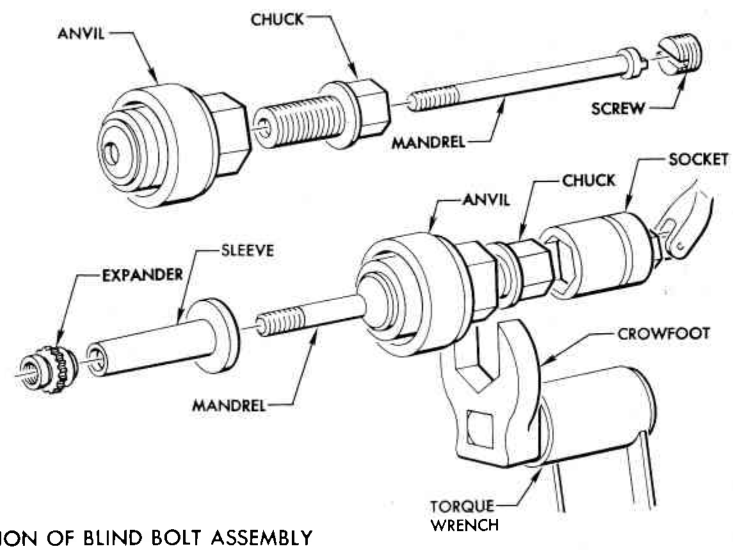
IF THE RIVET IS DRIVEN THROUGH TIGHT HOLES BY THE PIN, PREMATURE EXPANSION OF THE RIVET WILL RESULT. FOR TIGHT INSTALLATIONS, USE A HOLLOW DRIFT WHICH WILL CLEAR THE PIN WHILE SEATING THE SLEEVE.

06.03.279 51.04.00

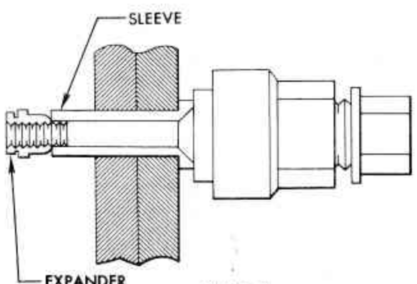
Figure 1-45. Deutsch Drive Pin Blind Rivet — Identification and Installation

INSTALLATION OF BLIND BOLT SLEEVE AND EXPANDER ON TOOL

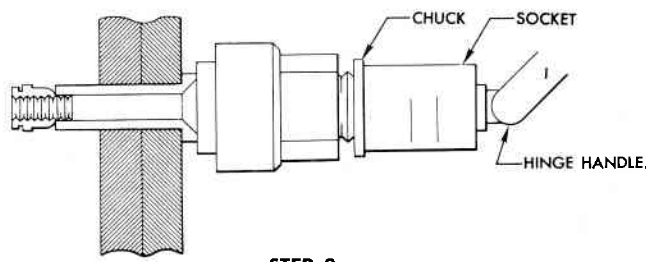
1. Install the mandrel in the chuck.
2. Screw the chuck counterclockwise until it is seated in the anvil.
3. Slip the sleeve over the mandrel.
4. Turn the chuck clockwise until sleeve is firmly seated.



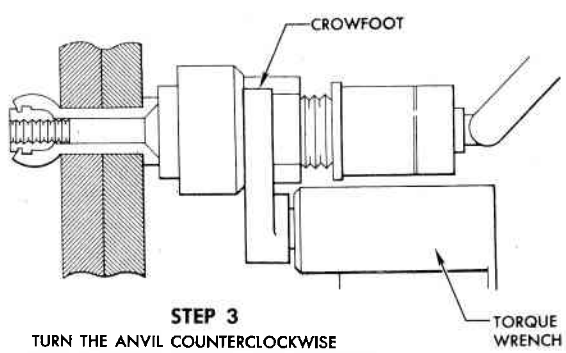
INSTALLATION OF BLIND BOLT ASSEMBLY



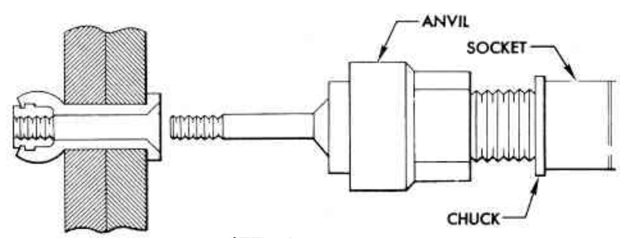
STEP 1
INSERT SLEEVE AND EXPANDER THROUGH WORK.



STEP 2
HOLD CHUCK WITH A SOCKET AND HINGE HANDLE.



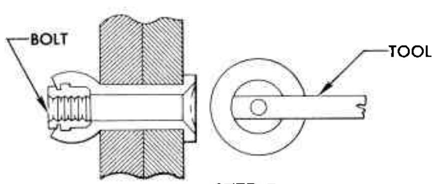
STEP 3
TURN THE ANVIL COUNTERCLOCKWISE WITH A TORQUE WRENCH AND CROWFOOT UNTIL EXPANDER FLARES SLEEVE. TORQUE EXPANDER TO VALUES SHOWN IN TABLE 1.



STEP 4
TO REMOVE TOOL, TURN ANVIL ONE-HALF TURN CLOCKWISE; UNSCREW CHUCK FROM EXPANDER THREADS.

TABLE I

BLIND BOLT SIZE (DASH NO.)	TORQUE INCH POUNDS
-5	65-75
-6	85-95
-8 -832	190-215
-1032	270-295
-10	295-320
-12 -428	465-495



STEP 5
INSTALL BOLT AND TIGHTEN WITH APPLICABLE TOOL.

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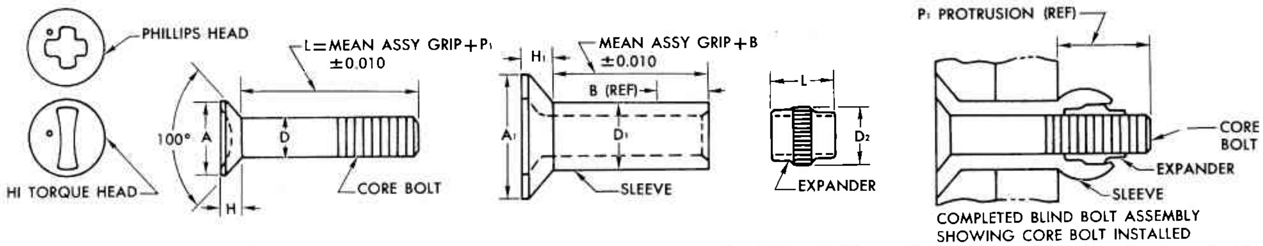
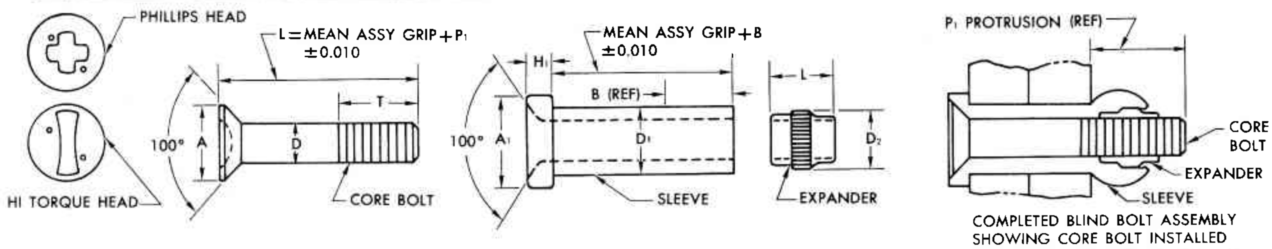
Figure 1-46. Installation of Blind Bolts (Hi-Shear Core Bolts)



BB351 BOLT ASSEMBLY		BB301 CORE BOLT						BB321 SLEEVE				BB341 EXPANDER	
DASH NO.	NOMINAL DIAMETER	HEAD STYLE	A	T	D	H	P_1	A_1	B (GRIP RANGE)	D_1	H_1	D_2	L
— 5	5/32	PHILLIPS	0.227 0.223		0.112 0.110	0.048 0.046	0.208	0.331 0.326	0.210 TO 0.225	0.169 0.166	0.068 0.066	0.167	0.153
— 6	3/16	PHILLIPS	0.251 0.247		0.124 01.22	0.053 0.051	0.239	0.381 0.377	0.250 TO 0.330	0.196 0.193	0.078 0.076	0.194	0.184
— 8	1/4	PHILLIPS	0.330 0.326		0.165 0.163	0.069 0.067	0.316	0.507 0.502	0.315 TO 0.370	0.257 0.254	0.105 0.103	0.255	0.245
—10	5/16	HI TORQUE	0.445 0.441		0.218 0.216	0.095 0.093	0.414	0.634 0.629	0.380 TO 0.420	0.338 0.335	0.124 0.122	0.328	
—12	3/8	HI TORQUE	0.507 0.502		0.249 0.247	0.108 0.106	0.485	0.760 0.756	0.450	0.390 0.387	0.155 0.153	0.383	0.368
—14	7/16	HI TORQUE	0.634 0.629		0.311 0.309	0.135 0.133	0.543	0.888 0.881	0.500	0.468 0.465	0.176 0.173	0.458	
—16	1/2	HI TORQUE	0.760 0.756		0.374 0.372	0.162 0.160	0.600	1.014 1.007	0.550	0.530 0.527	0.203 0.200	0.251	



BB352 BOLT ASSEMBLY		BB302 CORE BOLT						BB322 SLEEVE				BB341 EXPANDER	
DASH NUMBER	NOMINAL DIAMETER	HEAD STYLE	A	T	D	H	P_1	A_1	B (GRIP RANGE)	D_1	H_1	D_2	L
— 5	5/32	PHILLIPS	0.227 0.223	0.274	0.112 0.110	0.048 0.046	0.208	0.294 0.274	0.210 TO 0.225	0.169 0.166	0.055 0.045	0.167	0.153
— 6	3/16	PHILLIPS	0.251 0.247	0.304	0.124 0.122	0.053 0.051	0.239	0.315 0.295	0.250 TO 0.330	0.196 0.193	0.065 0.055	0.194	0.184
— 8	1/4	PHILLIPS	0.330 0.326	0.397	0.165 0.163	0.069 0.067	0.316	0.412 0.387	0.315 TO 0.370	0.257 0.254	0.090 0.080	0.255	0.245
—10	5/16	HI TORQUE	0.445 0.441	0.498	0.218 0.216	0.095 0.093	0.414	0.530 0.500	0.380 TO 0.420	0.338 0.335	0.110 0.100	0.328	0.307
—12	3/8	HI TORQUE	0.507 0.502	0.567	0.249 0.247	0.108 0.106	0.485	0.615 0.580	0.450	0.390 0.387	0.155 0.145	0.383	0.368
—14	7/16	HI TORQUE	0.634 0.629	0.628	0.311 0.309	0.135 0.113	0.543	0.803 0.768	0.500	0.468 0.465	0.160 0.150	0.458	0.429
—16	1/2	HI TORQUE	0.760 0.756	0.685	0.374 0.372	0.162 0.160	0.600	0.896 0.861	0.550	0.530 0.527	0.185 0.175	0.521	0.491

PART NUMBER BREAKDOWN

BB351-8-12

- 2ND DASH NUMBER INDICATES MEAN ASSEMBLY (THICKNESS OF WORK) GRIP LENGTH IN 1/16 INCH= ¾ INCH.
- DIAMETER OF BLIND BOLT IN 1/32 INCH= ¼ INCH
- BLIND BOLT ASSEMBLY 100° COUNTERSUNK WHICH INCLUDES: BB301 CORE BOLT, BB321 SLEEVE, BB341 EXPANDER.

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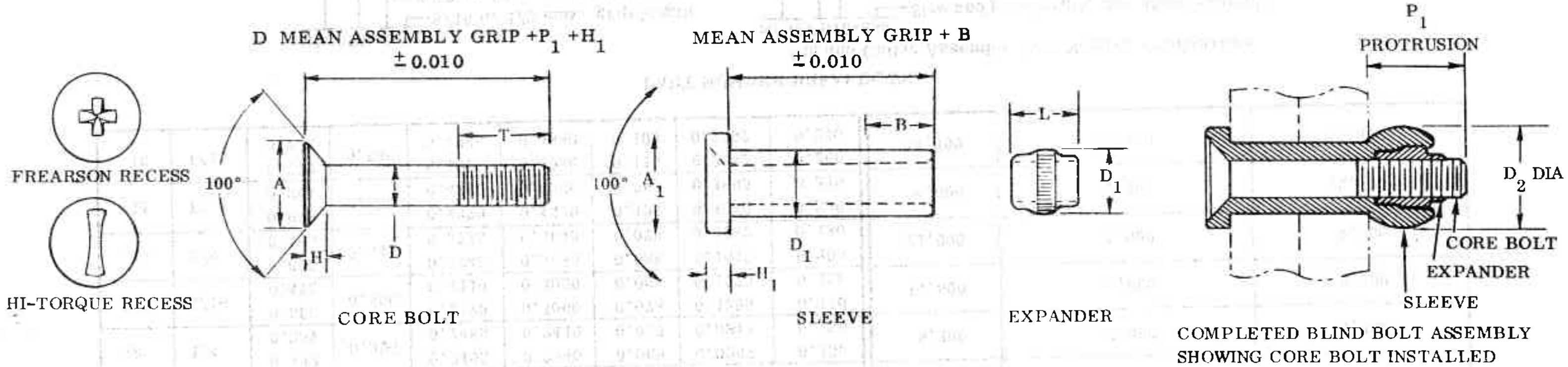
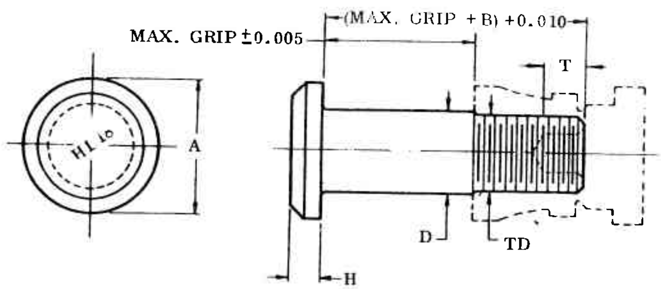
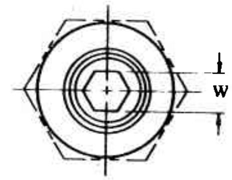
Figure 1-47. Blind Bolt Assembly — Identification (Sheet 1 of 2)

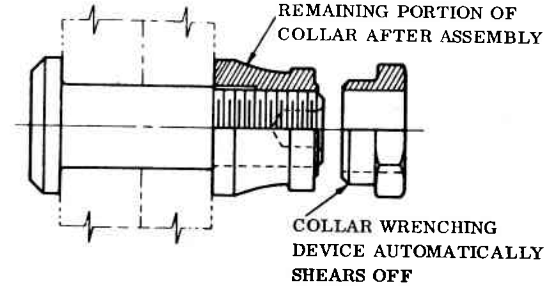


BB366 BOLT ASSEMBLY		BB316 CORE BOLT						BB336 SLEEVE						BB341G EXPANDER	
DASH NUMBER	NOMINAL DIAMETER	RECESS	A	T	D	H	P_1	A_1	B GRIP RANGE 0 - 3/4	B GRIP RANGE 13/16 - 1 1/16	B GRIP RANGE 1 1/8 - 1 5/8	D_1	H_1	D_2	L
-5	5/32	FREARSON	0.2274 0.2226	0.274	0.1120 0.1100	0.048 0.048	0.208	0.294 0.274	0.210	0.225		0.169 0.166	0.055 0.045	0.167	0.153
-6	3/16	FREARSON	0.2514 0.2466	0.304	0.1240 0.1220	0.053 0.051	0.239	0.315 0.295	0.250	0.265	0.330	0.196 0.193	0.065 0.055	0.194	0.184
-8	1/4	FREARSON	0.3304 0.3256	0.397	0.1650 0.1650	0.069 0.067	0.316	0.412 0.387	0.315	0.330	0.370	0.257 0.254	0.090 0.080	0.255	0.245
-10	5/16	HI-TORQUE #3	0.4454 0.4406	0.498	0.2180 0.2160	0.095 0.093	0.414	0.530 0.500	0.380	0.400	0.420	0.338 0.335	0.110 0.100	0.328	0.307
-12	3/8	HI-TORQUE #4	0.5066 0.5018	0.567	0.2490 0.2470	0.108 0.106	0.485	0.615 0.580	0.450	0.450	0.450	0.390 0.387	0.155 0.145	0.383	0.368

Figure 1-47. Blind Bolt Assembly — Identification (Sheet 2 of 2)







PIN AND COLLAR AFTER ASSEMBLY

FIRST DASH NO.	NOM. DIA.	A	B	D	THD	H	SOCKET W HEX	SOCKET T DEPTH
-5	5/32	0.262 0.242	0.312	0.1635 0.1625	0.1595 0.1570	0.047 0.037	0.0801 0.0791	0.135 0.115
-6	3/16	0.315 0.295	0.325	0.1895 0.1835	0.1840 0.1810	0.055 0.045	0.0806 0.791	0.135 0.115
-8	1/4	0.412 0.387	0.395	0.2495 0.2485	0.2440 0.2410	0.069 0.059	0.0967 0.0947	0.150 0.130
-10	5/16	0.505 0.475	0.500	0.3120 0.3110	0.3060 0.3020	0.078 0.068	0.1295 0.1270	0.170 0.150
-12	3/8	0.600 0.565	0.545	0.3745 0.3735	0.3680 0.3640	0.088 0.078	0.1617 0.1582	0.200 0.180
-14	7/16	0.676 0.641	0.635	0.4370 0.4360	0.4310 0.4260	0.105 0.093	0.1930 0.1895	0.230 0.210
-16	1/2	0.770 0.735	0.685	0.4995 0.4985	0.4930 0.4880	0.115 0.103	0.2242 0.2207	0.260 0.240

MECHANICAL PROPERTIES OF PIN AND COLLAR ASSEMBLY		
DOUBLE SHEAR POUNDS MINIMUM	ALUMINUM COLLARS HL70 HL79 HL82 TENSION LBS MIN.	STEEL COLLARS HL94 HL97 HL175 TENSION LBS MIN.
4,010	1,400	1,940
5,380	1,600	2,500
9,300	3,000	4,300
14,600	5,000	6,300
21,000	7,000	8,700
28,600	9,500	12,100
37,300	12,500	15,300

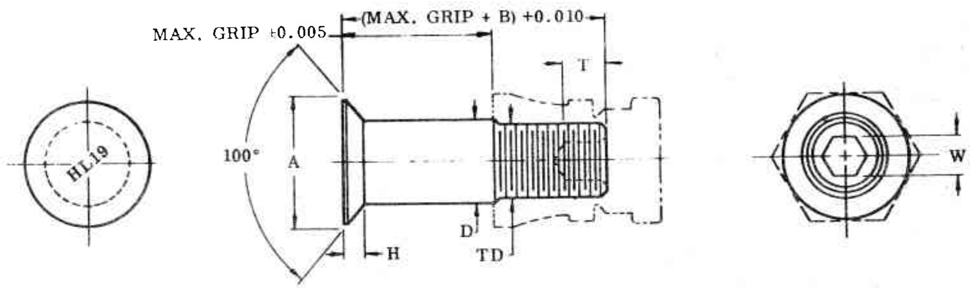
PART NUMBER BREAKDOWN

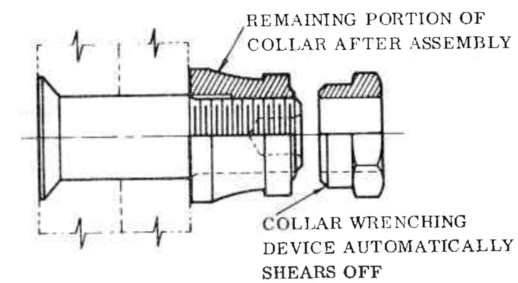
HL18PB-8-8
- 8/16 or 1/2 max. grip length
- 8/32 or 1/4 nominal dia. pin
- Type II cadmium plate, color bronze
- Pin part number

Pin and Collar Assembly Part Number Combination

HL18PB70-8-8
- Size and grip length, see above example
- Collar part number
- Pin finish
- Pin part number

Figure 1-47A. Hi-Lok Fasteners — Identification (Sheet 1 of 4)





PIN AND COLLAR AFTER ASSEMBLY

FIRST DASH NO.	NOM. DIA.	A	B	D	THD	F	H	SOCKET W HEX	SOCKET T DEPTH
-5	5/32	0.2612 0.2564	0.312	0.1635 0.1625	0.1595 0.1570	0.004	0.0410 0.0390	0.0801 0.0791	0.135 0.115
-6	3/16	0.3016 0.2966	0.325	0.1895 0.1885	0.1840 0.1810	0.005	0.0470 0.0450	0.0806 0.0791	0.135 0.115
-8	1/4	0.3948 0.3898	0.395	0.2495 0.2485	0.2440 0.2410	0.006	0.610 0.0590	0.0967 0.0947	0.150 0.130
-10	5/16	0.4739 0.4689	0.500	0.3120 0.3110	0.3060 0.3020	0.007	0.0680 0.0660	0.1295 0.1270	0.170 0.150
-12	3/8	0.5604 0.5554	0.545	0.3745 0.3735	0.3680 0.3640	0.008	0.0780 0.0760	0.1617 0.1582	0.200 0.180
-14	7/16	0.6680 0.6620	0.635	0.4370 0.4360	0.4310 0.4260	0.009	0.0969 0.0944	0.1930 0.1895	0.230 0.210
-16	1/2	0.7540 0.7480	0.685	0.4995 0.4985	0.4930 0.4880	0.010	0.1068 0.1043	0.2242 0.2207	0.260 0.240

MECHANICAL PROPERTIES OF PIN AND COLLAR ASSEMBLY		
DOUBLE SHEAR POUNDS MINIMUM	ALUMINUM COLLARS HL70 HL79 HL82 TENSION LBS MIN.	STEEL COLLARS HL94 HL97 HL175 TENSION LBS MIN.
4,010	1,290	1,290
5,380	1,600	2,000
9,300	3,000	3,700
14,600	5,000	5,000
21,000	7,000	7,200
28,600	9,500	10,000
37,300	12,500	13,500

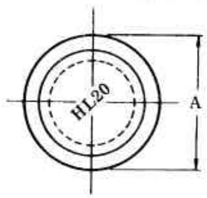
PART NUMBER BREAKDOWN

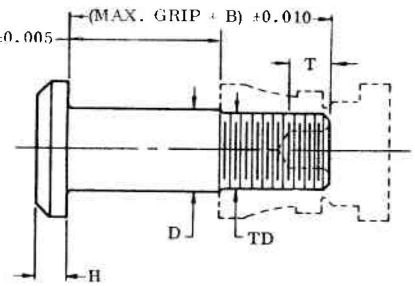
HL19PB-8-8
- 8/16 or 1/2 max. grip length
- 8/32 or 1/4 nominal dia. pin
- Type II cadmium plate, color bronze
- Pin part number

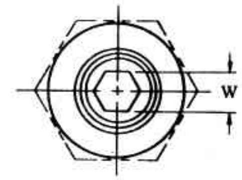
Pin and Collar Assembly Part Number Combination

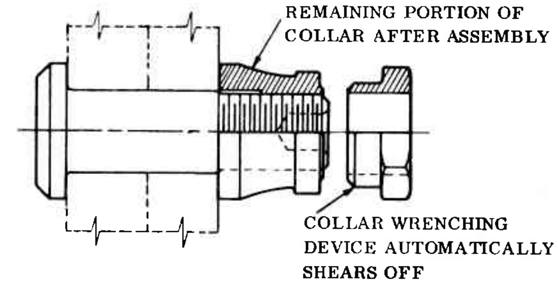
HL19PB70-8-8
- Size and grip length, see above example
- Collar part number
- Pin finish
- Pin part number

Figure 1-47A. Hi-Lok Fasteners — Identification (Sheet 2 of 4)









PIN AND COLLAR AFTER ASSEMBLY

FIRST DASH NO.	NOM. DIA.	A	B	D	THD	H	SOCKET W HEX	SOCKET T DEPTH	MECHANICAL PROPERTIES OF PIN AND COLLAR ASSEMBLY: DOUBLE SHEAR POUNDS MINIMUM	STEEL COLLARS HL75 HL86 HL87 TENSION LBS MIN.
-5	5/32	0.322 0.306	0.312	0.1635 0.1625	0.1595 0.1570	0.060 0.055	0.0801 0.0791	0.135 0.115	4,010	2,180
-6	3/16	0.377 0.357	0.325	0.1895 0.1885	0.1840 0.180	0.074 0.064	0.0806 0.0791	0.135 0.115	5,380	2,750
-8	1/4	0.440 0.415	0.395	0.2459 0.2485	0.2440 0.2410	0.090 0.077	0.0967 0.0947	0.150 0.130	9,300	5,000
-10	5/16	0.502 0.472	0.500	0.3120 0.3110	0.3060 0.3020	0.112 0.098	0.1295 0.1270	0.170 0.150	14,600	8,300
-12	3/8	0.565 0.530	0.545	0.3745 0.3735	0.3680 0.3640	0.140 0.130	0.1617 0.1582	0.200 0.180	21,000	12,700
-14	7/16	0.627 0.592	0.635	0.4370 0.4360	0.4310 0.4260	0.160 0.150	0.1930 0.1895	0.230 0.210	28,600	18,900
-16	1/2	0.752 0.717	0.685	0.4995 0.4985	0.4930 0.4880	0.188 0.178	0.2242 0.2207	0.260 0.240	37,300	25,500

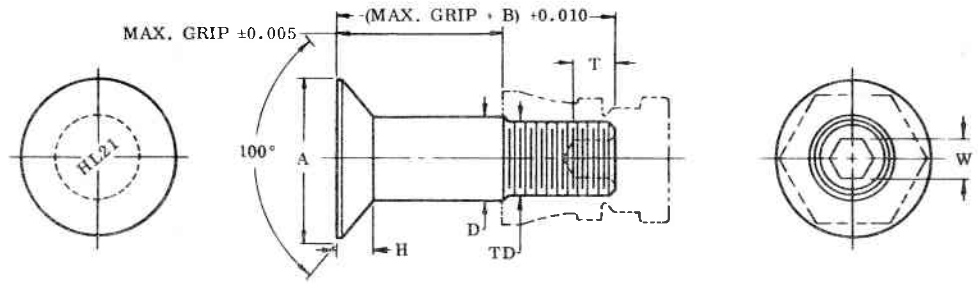
PART NUMBER BREAKDOWN

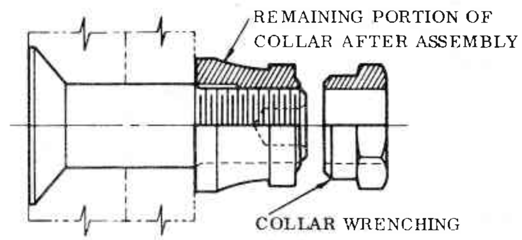
HL20PB-8-8
- 8/16 or 1/2 max. grip length
- 8/32 or 1/4 nominal dia. pin
- Type II cadmium finish
- Pin part number

Pin and Collar Assembly Part Number Combination

HL20PB86-8-8
- Size and grip length, see above example
- Collar part number
- Pin finish
- Pin part number

Figure 1-47A. Hi-Lok Fasteners —Identification (Sheet 3 of 4)





PIN AND COLLAR AFTER ASSEMBLY

FIRST DASH NO.	NOM. DIA.	A	B	D	THD	F	H	SOCKET W HEX	SOCKET T DEPTH	MECHANICAL PROPERTIES OF PIN AND COLLAR ASSEMBLY: DOUBLE SHEAR POUNDS MINIMUM	STEEL COLLARS HL75 HL86 HL87: TENSION LBS MIN.
-5	5/32	0.3304 0.3256	0.312	0.1635 0.1625	0.1595 0.1570	0.004	0.0700 0.0680	0.0801 0.0791	0.135 0.115	4,010	2,180
-6	3/16	0.3813 0.3765	0.325	0.1895 0.1885	0.1840 0.1810	0.005	0.0805 0.0785	0.0806 0.0791	0.135 0.115	5,380	2,750
-8	1/4	0.5066 0.5018	0.395	0.2495 0.2485	0.2440 0.2410	0.006	0.1080 0.1060	0.0967 0.0947	0.150 0.130	9,300	5,000
-10	5/16	0.6335 0.6287	0.500	0.3120 0.3110	0.3060 0.3020	0.007	0.1350 0.1330	0.1295 0.1270	0.170 0.150	14,600	8,300
-12	3/8	0.7604 0.7556	0.545	0.3745 0.3735	0.3680 0.3640	0.008	0.1620 0.1600	0.1617 0.1582	0.200 0.180	21,000	12,700
-14	7/16	0.8884 0.8812	0.635	0.4370 0.4360	0.4310 0.4260	0.009	0.1895 0.1865	0.1930 0.1895	0.230 0.210	28,600	18,900
-16	1/2	1.0139 1.0068	0.685	0.4995 0.4985	0.4930 0.4880	0.010	0.2160 0.2130	0.2242 0.2207	0.260 0.240	37,300	25,500

PART NUMBER BREAKDOWN

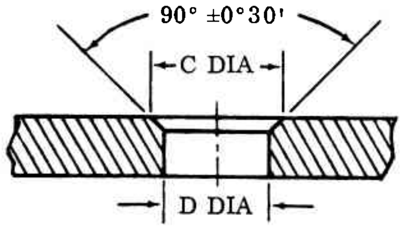
HL21PB-8-8
- 8/16 or 1/2 max. grip length
- 8/32 or 1/4 nominal dia. pin
- Type II cadmium plate
- Pin part number

Pin and Collar Assembly Part Number Combination

HL21PB86-8-8
- Size and grip length, see above example
- Collar part number
- Pin finish
- Pin part number

Figure 1-47A. Hi-Lok Fasteners — Identification (Sheet 4 of 4)

HOLE PREPARATION-PROTRUDING HEAD

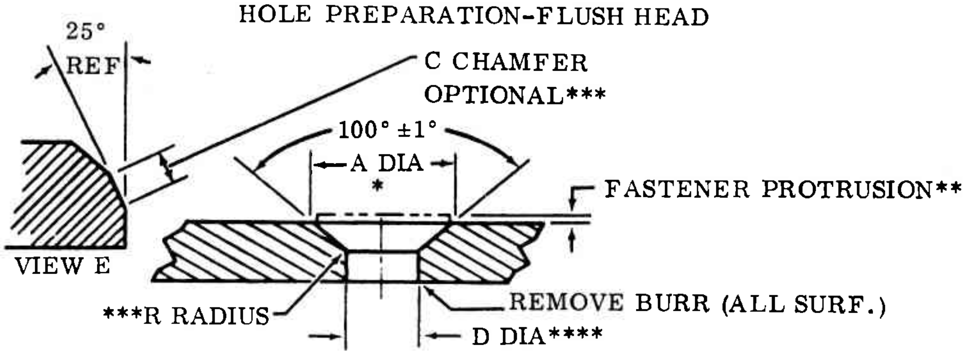


FASTENER (REF)		C ±0.010	HI-LOK PINS**	
SIZE	MAX DIA	DIA*	D MAX	D MIN
5/32	0.1635	0.204	0.1630	0.1615
3/16	0.1895	0.240	0.1890	0.1875
1/4	0.2495	0.300	0.2490	0.2470
5/16	0.3120	0.372	0.3115	0.3095
3/8	0.3745	0.435	0.3740	0.3720
7/16	0.4370	0.497	0.4365	0.4345

*COUNTERSINK IS REQUIRED TO CLEAR THE FILLET RADIUS EXISTING BETWEEN FASTENER HEAD AND SHANK. WHERE WASHERS ARE REQUIRED UNDER FASTENER HEADS, DO NOT COUNTERSINK.

**HOLE SIZES TABULATED RESULT IN INTERFERENCE FITS.

HOLE PREPARATION-FLUSH HEAD



FASTENER (REF)		HI-LOK PINS			R ±0.005	C ±0.010
		A DIA	D DIA****			
SIZE	MAX DIA	*	MAX	MIN	***	***
5/32	0.1635	0.258	0.1630	0.1615	0.025	0.020
3/16	0.1895	0.305	0.1890	0.1875	0.035	0.030
1/4	0.2495	0.399	0.2490	0.2470	0.040	0.035
5/16	0.3120	0.479	0.3115	0.3095	0.045	0.040
3/8	0.3745	0.566	0.3740	0.3720	0.055	0.050
7/16	0.4370	0.674	0.4365	0.4345	0.060	0.050

*COUNTERSINK DIAMETERS ARE STANDARD ONLY AND MUST BE ADJUSTED TO MEET THE FOLLOWING REQUIREMENTS.

**UNLESS OTHERWISE NOTED IN REPAIR OR INSTALLATION PROCEDURES, COUNTERSINKS SHALL ENSURE FASTENER HEAD FLUSHNESS.

***THESE RADII OR CHAMFERS ARE PRIMARILY NEEDED WHEN FASTENERS ARE INSTALLED IN STEEL OR TITANIUM STRUCTURE. OPTIONAL FOR OTHER MATERIALS.

****HOLE SIZE SHOWN IN TABLE RESULT IN INTERFERENCE FITS.

Figure 1-47B. Hi-Lok Fastener Hole Preparation

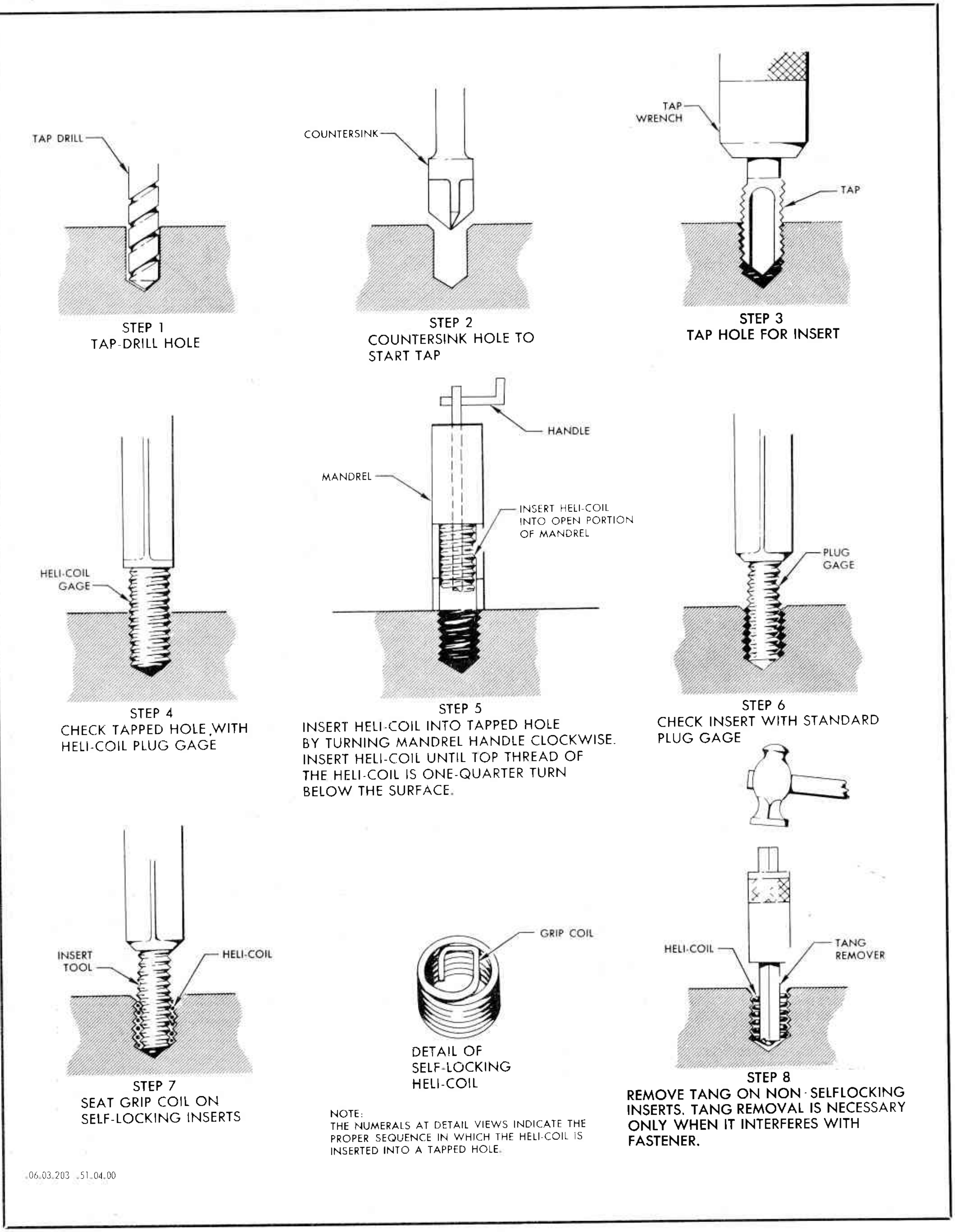
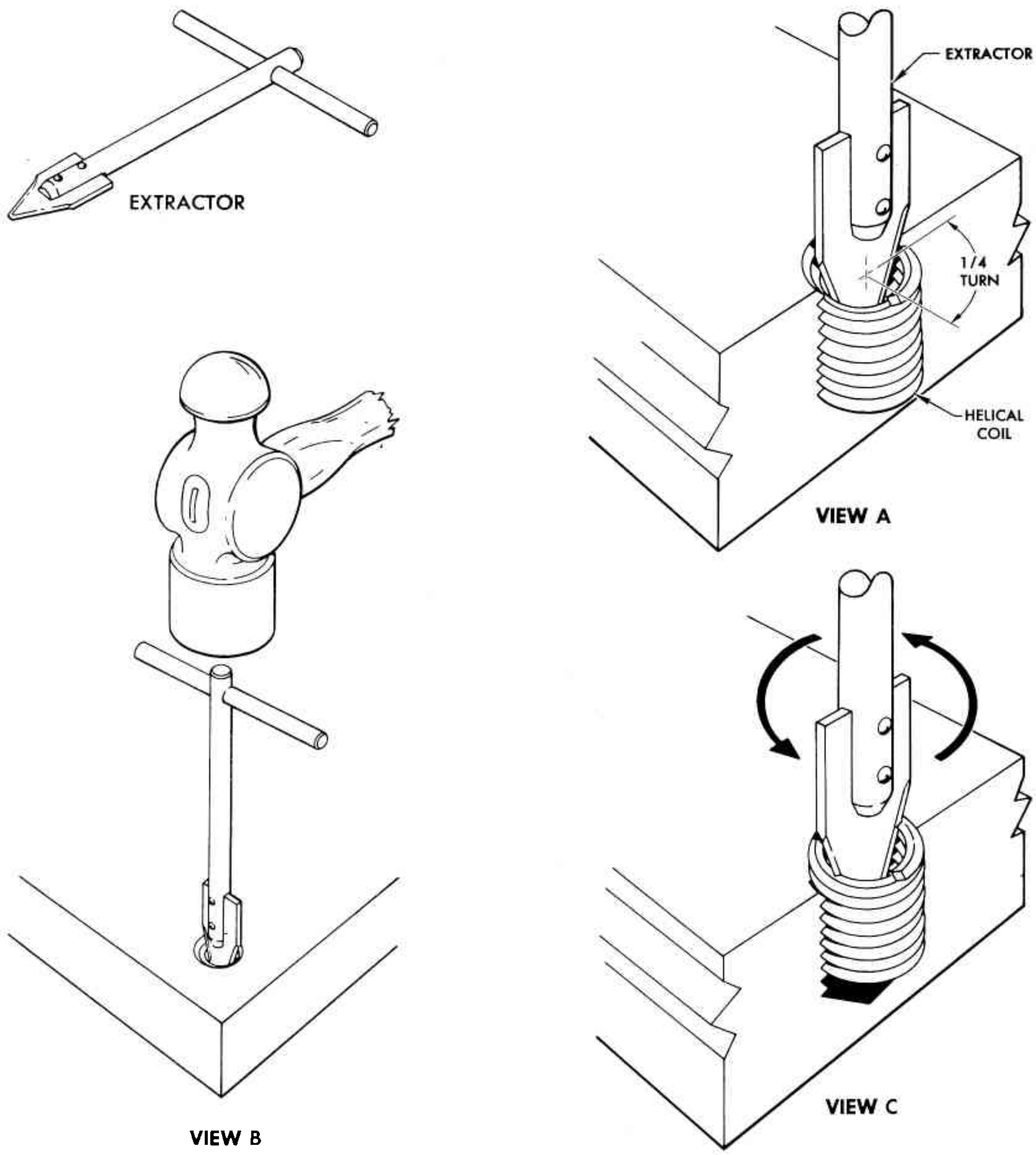


Figure 1-48. Installation of Helical Coils



DAMAGED HELICAL COIL REMOVAL

a. Select proper extractor from Table 1.
b. Place the blade of the extractor on first turn of helical coil insert. The edge of blade should be 1/4 turn away from end of coil. See View A.
c. Rap extractor lightly with hammer to seat blade firmly against coil. See View B.
d. Apply heavy downward pressure and rotate extractor counterclockwise until helical coil is removed. See View C.

TABLE I	
HOLE AND THREAD SIZE	EXTRACTOR PART NO.
10-32	1227-6
¼-28	1227-6
5/16-24	1227-6
3/8-24	1227-16
7/16-20	1227-16
½-20	1227-16
9/16-18	1227-16
¾-16	1227-16
7/8-14	1227-16
1-14	1227-16
1¼-12	1227-24

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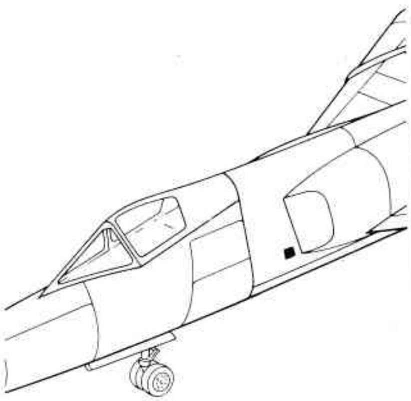
Figure 1-49. Helical Coil Removal



AREA OF ASSUMED DAMAGE

REMOVAL AND
INSTALLATION PROCEDURE

a. Remove access panel between Stations 226.92 and 247.81 inside forward missile bay compartment.
b. Using a No. 30 (0.128) drill, remove two rivets holding receptacle in place.

CAUTION

DO NOT ALLOW DRILL TO PENETRATE SKIN MORE THAN NECESSARY, AS UNDERLYING EQUIPMENT MAY BE DAMAGED.

c. Remove damaged Ball-Lok receptacle.
d. Insert new receptacle through access door and fasten in place with a 1/8-inch cleco.
e. Rivet new receptacle in place with two Q4304-C4 rivets. Refer to Table I-XXII for correct rivet substitution if Q4304 rivets are not available.
f. Replace access door inside forward missile bay compartment.

NOTE
BALL-LOCK RECEPTACLES (P/N 2703) ARE MANUFACTURED BY AVDELL INC., 210 SO. VICTORY BLVD., BURBANK, CALIF.

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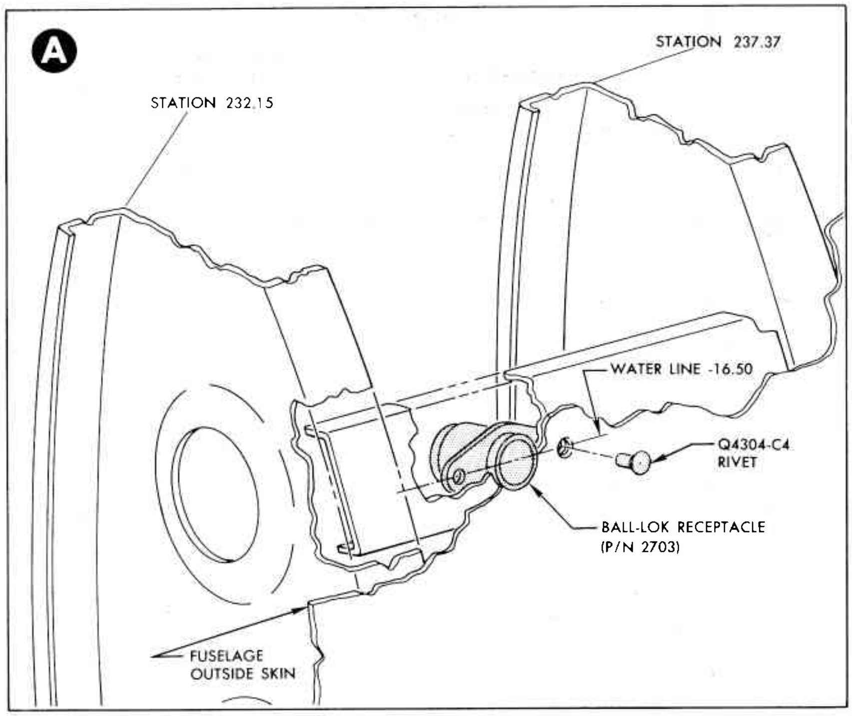
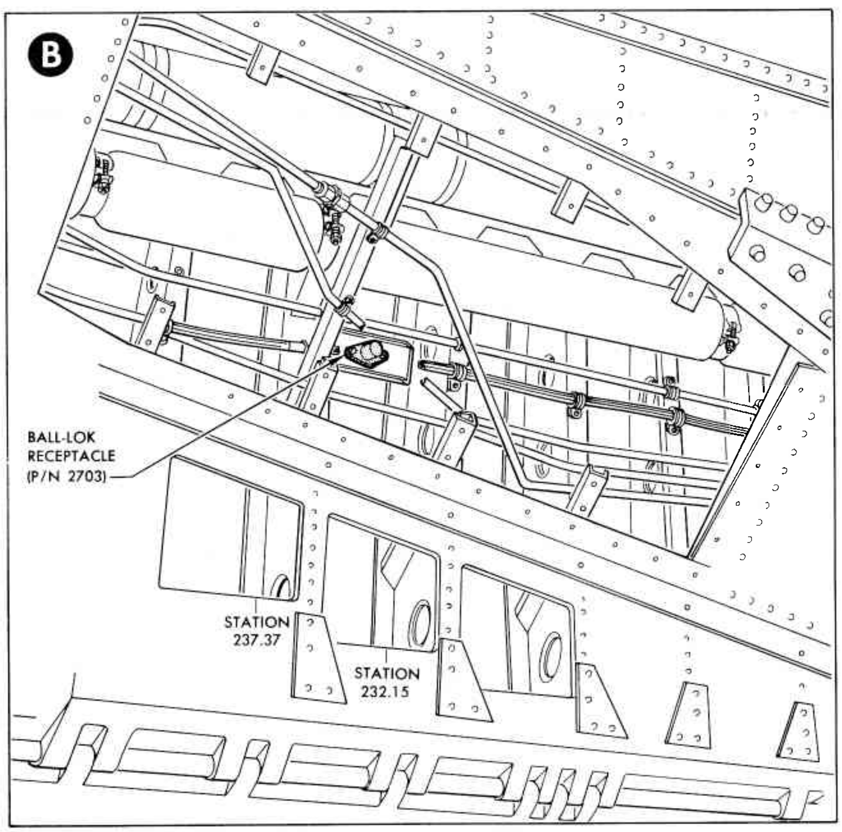


VIEW LOOKING INBOARD (LEFT-HAND SIDE)



VIEW LOOKING OUTBOARD (LEFT-HAND SIDE)

Figure 1-50. Replacement of Ball-Lok Receptacles

BOLT SIZE	STEEL BOLTS				ALUMINUM ALLOY BOLTS (AN365D NUTS)		
	NUT TYPES AN365 AND AN310		NUT TYPES AN364 AND AN320				
	INCH LBS	FOOT LBS	INCH LBS	FOOT LBS	BOLT SIZE	INCH LBS	FOOT LBS
10-32	20-25	—	12-15	—	3/16	10-14	—
1/4-28	50-70	—	30-40	—	1/4	20-35	—
5/16-24	100-140	9-12	60-85	5-7	5/16	50-75	4-6
3/8-24	160-190	13-16	95-110	8-9	3/8	80-110	7-9
7/16-20	450-500	38-42	270-300	23-25	7/16	100-140	8-12
1/2-20	480-690	40-57	290-410	24-34	1/2	170-220	14-18
9/16-18	800-1000	67-83	480-600	40-50	5/8	400-460	34-38
5/8-18	1100-1300	92-108	660-780	55-65			
3/4-16	2300-2500	192-208	1300-1500	109-125			
7/8-14	2500-3000	209-250	1500-1800	125-150			
1-14	3700-5500	308-458	2200-3300	184-275			
1-1/8-12	5000-7000	417-583	3000-4200	250-350			
1-1/4-12	9000-11000	750-916	5400-6600	450-550			

When using torque wrench adapters, if the desired torque is known, the torque wrench dial reading may be found as follows:

T_w = Wrench dial reading.
T_a = Desired torque at end of adapter.
L = Lever length of torque wrench.
A = Length of adapter (center distance).

$$\text{FORMULA}—T_w = \frac{T_a \times L}{L + A}$$

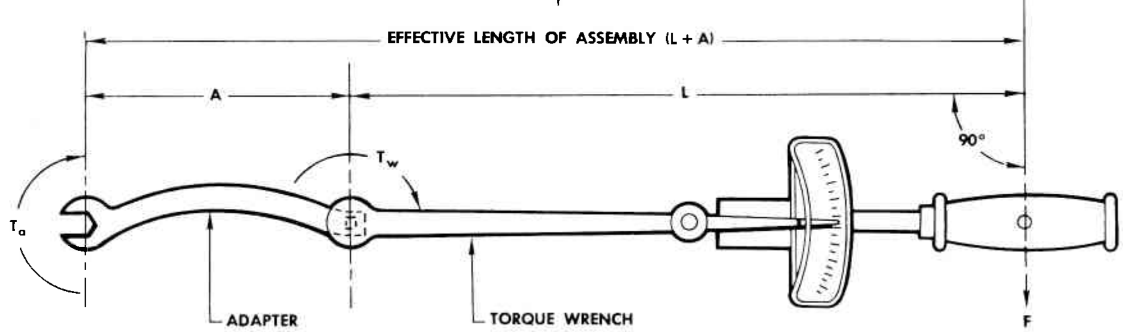


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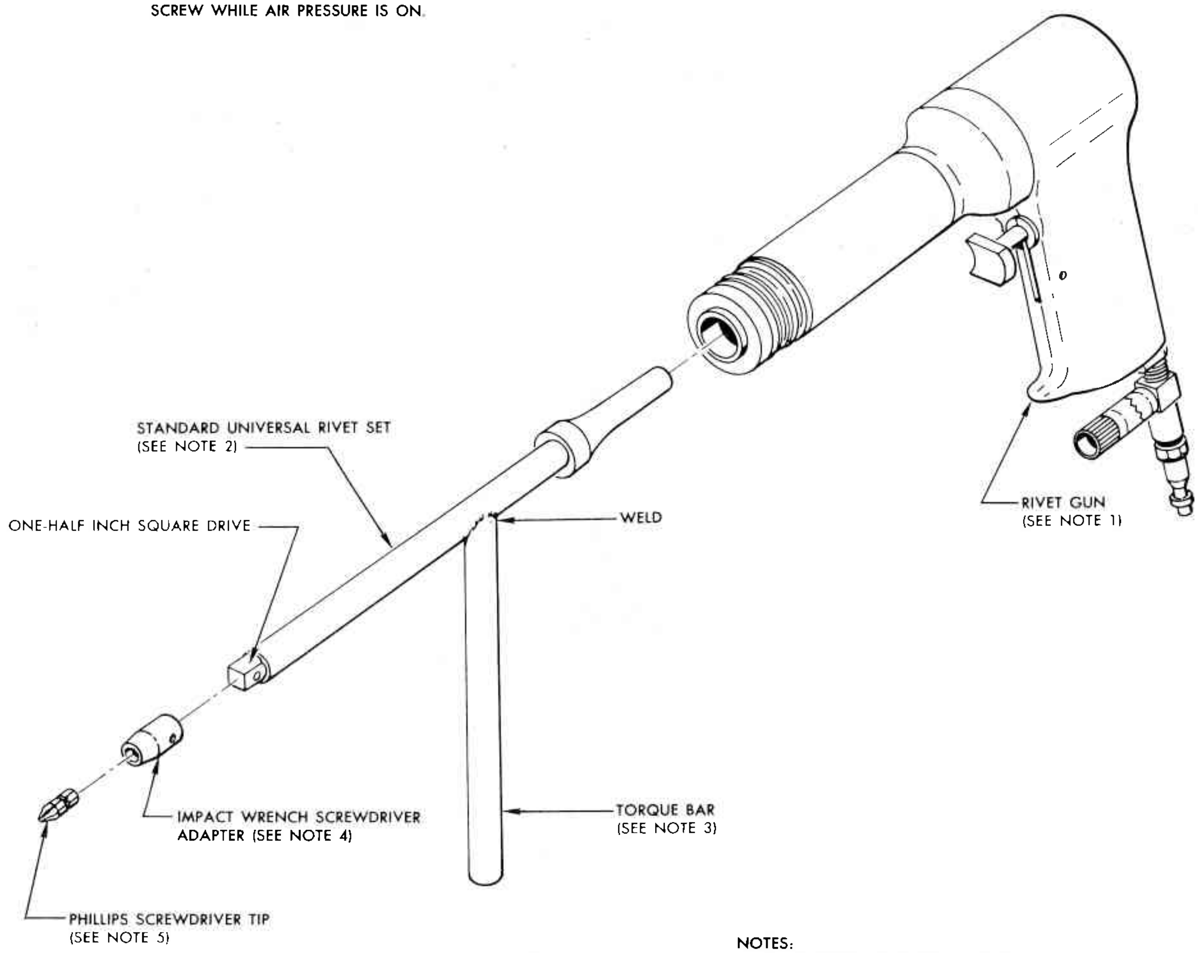
Figure 1-51. Standard Bolt Torque Values

FROZEN SCREW REMOVAL PROCEDURE

a. Insert machined rivet set with attached torque bar into rivet gun.
b. Place screwdriver adapter over machined end of rivet set.
c. Insert Phillips screwdriver tip into screwdriver adapter.
d. Connect air supply.
e. Engage Phillips screwdriver tip with frozen screw.
f. Press trigger on rivet gun and torque bar to remove frozen screw.

NOTE:
KEEP PHILLIPS SCREWDRIVER TIP PRESSED FIRMLY AGAINST SCREW WHILE AIR PRESSURE IS ON.



NOTES:
1. CP-4X RIVET GUN OR A RIVET GUN OF EQUAL STRENGTH.
2. STANDARD UNIVERSAL RIVET SET 7 INCHES LONG. WITH DRIVEN END MACHINED TO ½-INCH SQUARE TO FIT IMPACT WRENCH ADAPTER.
3. TORQUE BAR. PREFERABLY MADE FROM ½-INCH STEEL TUBE FOR WELDING AND STRENGTH PURPOSES.
4. SCREWDRIVER ADAPTER FROM INGERSOL-RAND 504 IMPACT WRENCH.
5. STANDARD PHILLIPS SCREWDRIVER TIP.

FROZEN SCREW REMOVAL

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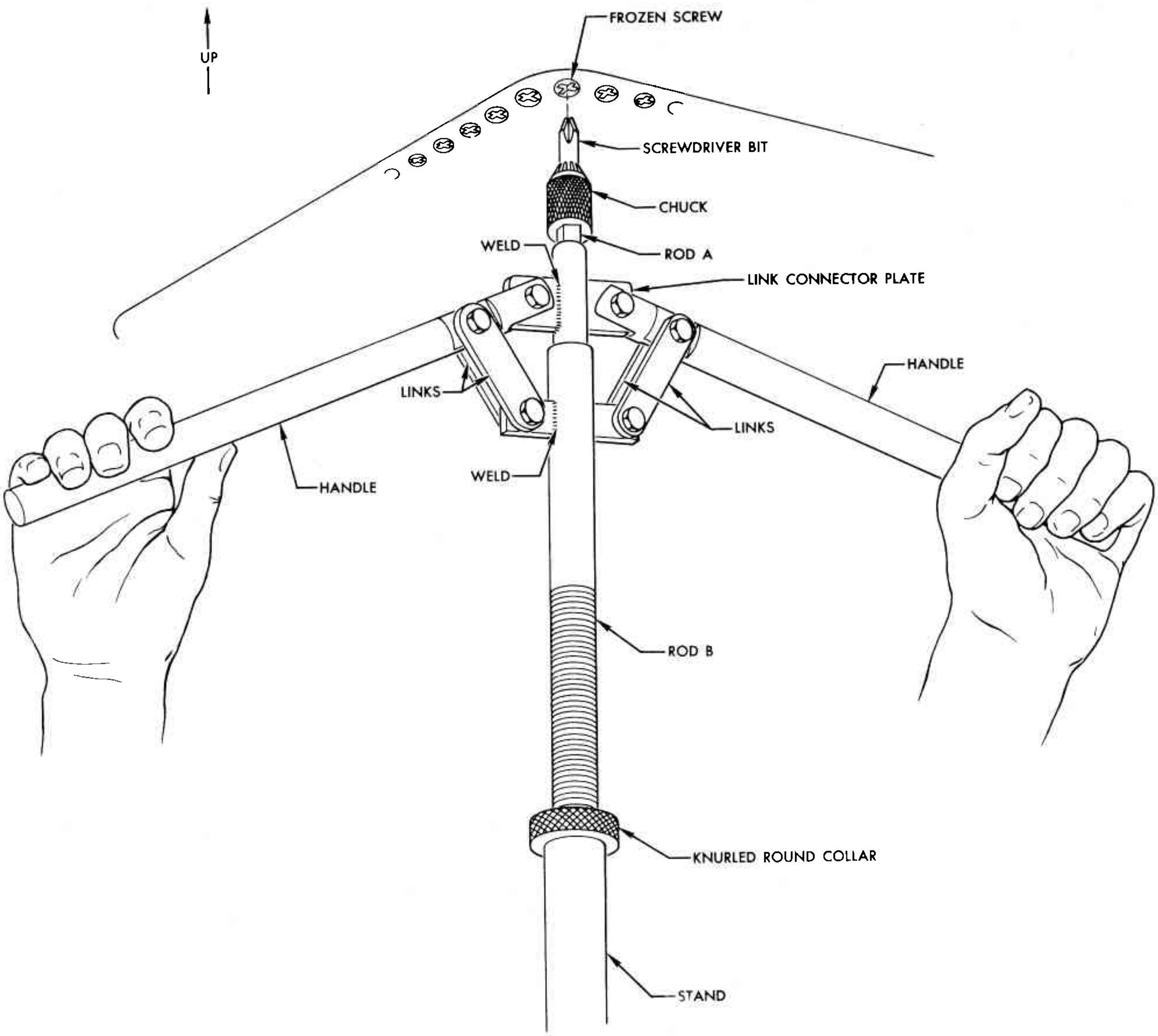
Figure 1-52. Frozen Screw Removal (Sheet 1 of 3)



SCREW REMOVAL PROCEDURE

a. Place stand directly beneath frozen screw.
b. Push handles to up position.
c. Adjust height of stand by turning knurled round collar.

NOTE

ADJUST HEIGHT OF STAND SO THAT TIP OF SCREW-DRIVER BIT IS APPROXIMATELY TWO INCHES FROM HEAD OF FROZEN SCREW.

d. Grasp handles and pull down until screwdriver bit is firmly seated in head of screw.
e. To remove frozen screw, rotate screwdriver bit while pulling down on handles.

NOTE

MAINTAIN CONSTANT DOWNWARD PRESSURE ON HANDLES WHILE TURNING.

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Figure 1-52. Frozen Screw Removal (Sheet 2 of 3)

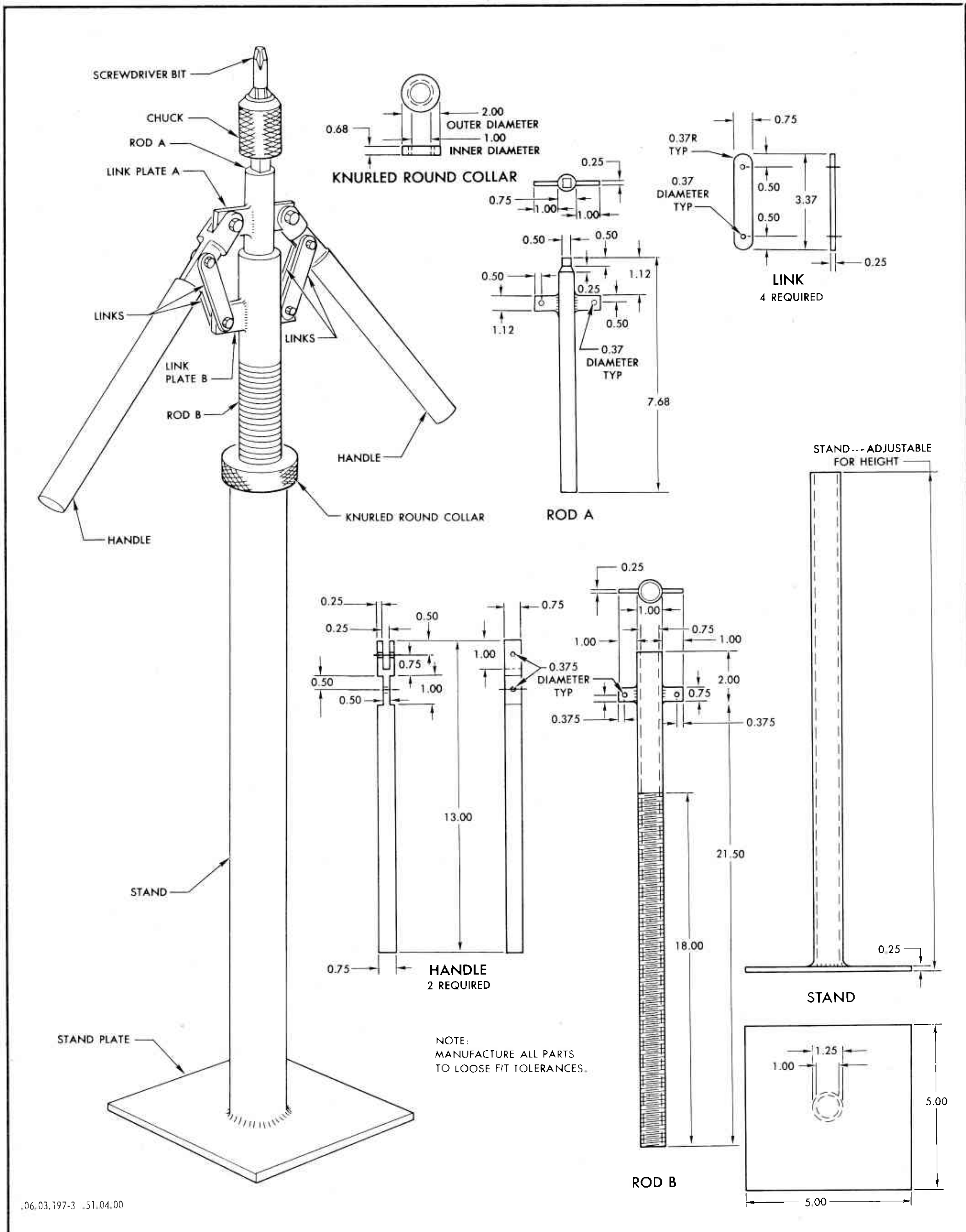


Figure 1-52. Frozen Screw Removal (Sheet 3 of 3)

junction box covers and on the airplane exterior. Where screws or bolts are used to mount junction boxes to the structure, they shall be installed with the head inside the junction box, to eliminate the possibility of loose nuts and washers shorting across electrical terminals.

Where practical, screws and bolts installed in a forward-and-aft direction shall be installed with head forward. Screws and bolts installed in an inboard-and-outboard direction shall be installed with the head inboard. Self-locking nuts, 3/16 and 1/4-inch diameter, shall be used only on fasteners that have not been drilled for cotter pins. Self-locking nuts, 5/16-inch diameter and larger, may be used on fasteners that are drilled for cotter pins if, after the nut is installed, the drilled hole is located beyond the load zone (the flat portion of the nut at the point of wrench contact). Sharp burrs around the cotter pin hole must be removed if fiber insert, self-locking nuts are used. See figure 1-51 for bolt torque values. Methods of removing frozen screws are shown on figure 1-52.

1-164. MS FLARELESS TUBE AND FITTING ASSEMBLY.

1-165. Nylon tubing is used in the airspeed indicating system and the radome anti-ice system. This tubing is installed in the radome section of the airplane. Flareless type MS fittings are used on the airspeed system tubing (see figure 1-53). Standard procedure for installation of MS fittings on aluminum alloy tubing also applies to the installation of MS fittings on nylon tubing, except for torque procedure. Nylon tubing used in the anti-ice system is equipped with flare fittings. The procedure used for flaring and installation of aluminum alloy tubing, as outlined in Air Force Navy Aeronautical Design Standards AND10064, also applies for flaring and installation of nylon tubing, except that the end of the tube must not be chamfered, and for torquing procedures. When installing flared nylon tubing, use the following torque values:

1/4 inch tube — 20 to 30 inch-pounds.

3/8 inch tube — 40 to 60 inch-pounds.

NOTE

The flared ends of nylon tubing have a tendency to return to the tube original shape when the tubing nuts are loosened. Prior to tightening of flared nylon tubing nuts, ascertain that the proper flare shape and size is present at the tube end.

When installing flareless nylon tubing use the following procedure:

a. Ascertain that the tube end is cut smooth and square.

b. Insert tubing into assembled MC fitting, making sure that the tube end seats firmly on shoulder of the fitting.

c. Using thumb and forefinger, tighten nut until the tube cannot be turned in the fitting; then tighten nut 1/3 of a turn, using a wrench; this completes the installation.

1-166. Stressed Panel Camloc Fasteners.

1-167. Electronic and engine access doors are secured with shear-load carrying, stressed-panel Camloc fasteners. Under normal flight conditions these fasteners carry shear loads of considerable magnitude. The airplane should not be flown with any of these fasteners unlocked or with any studs missing. Stud assemblies are of the captive type in all instances, but are retained in the door panel by two different methods. One type of stud is secured in the panel by flaring out the end of the stud bushing, as shown on figure 1-54. The flared end of the bushing serves to hold both the stud and the shear ring in place. The other type of stud is secured in the panel by forcing a spring steel ring with internal nodes over the bushing and engaging the nodes in grooves provided for that purpose in the bushing, as shown on figure 1-55. See figure 1-58 for stud length and grip. See figure 1-59 for installation data on Camloc rings and figure 1-60 for use of shear and buffer plates.

1-168. Standard Camloc Fasteners.

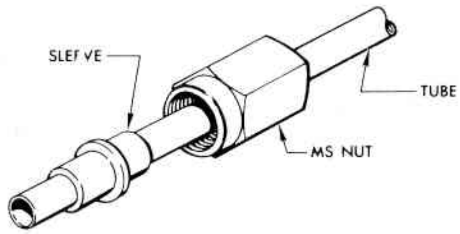
1-169. Standard Camloc panel fasteners are employed to secure the fuselage access doors to the fuselage basic structure. The airplane shall not be flown unless all Camloc fasteners are installed and properly secured. See figure 1-56 for installation procedure, and refer to T.O. 1F-106A-2-2-2-2 for Camloc operational data. See figure 1-57 for stud length and grip.

1-170. Standard Camloc Stud Assembly Removal.

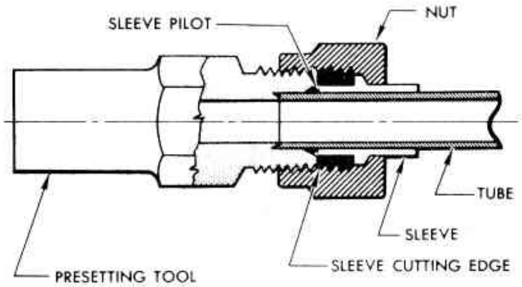
1-171. For quick removal of standard Camloc stud assemblies a tool may be manufactured locally (see figure 1-61). To remove stud assembly, remove lockpin and release stud. Remove grommet as shown on figure 1-61.

1-172. LOCATING BLIND HOLES.

1-173. For many repairs, holes in repair parts or patches must be drilled to match existing holes in the original structure while the repair part or patch is in contact with the original structure. In accessible areas, holes may be properly located by back-drilling through the existing holes into the repair part. In blind areas, the holes must be drilled through the repair part, from the outside, into the existing hole in the underlying structure. Existing holes may be located by using a hole finder similar to that shown on figure 1-62. Burrs and drill chips may be removed from between the faying surfaces without removing the repair parts or patches by using a chip chaser similar to the one shown on figure 1-63. Hole finders can be made locally or purchased in standard sizes for use with common diameter rivet holes. A separate hole finder must be used for each diameter of rivet



STEP 1



STEP 2

ASSEMBLY PROCEDURE

a. Select correct size presetting tool by part number from Table I. If tool is not available, an MS steel or aluminum connector fitting may be used.

CAUTION

IF ALUMINUM CONNECTOR IS USED, USE CONNECTOR ONLY ONCE.

b. Lubricate presetting tool and MS nut threads, sleeve pilot, and tool seat. Refer to Table II for the correct lubricant.
c. Cut tube end square to receive fitting. Burr tube end inside and outside. Chamfer inside and outside edges slightly.
d. Slide MS nut and then sleeve on tube as shown in Step I.
e. Insure that presetting tool is lubricated sufficiently.
f. Bottom tube firmly on presetting tool set and slide sleeve and nut in position. See Step 2.

TABLE I

TUBE O.D.	*PARKER AIRCRAFT CO. PRESETTING TOOL PART NUMBER
1/8"	1149-565702
3/16"	1149-565703
1/4"	1149-565704
5/16"	1149-565705
3/8"	1149-565706
1/2"	1149-565708
5/8"	1149-565710
3/4"	1149-565712
1"	1149-565716
1-1/4"	1149-565720
1-1/2"	1149-565724
1-3/4"	1149-565728
2"	1149-565732

NOTE:
*PARKER AIRCRAFT COMPANY IS LOCATED AT 5827 WEST CENTURY BLVD., LOS ANGELES 45, CALIF.

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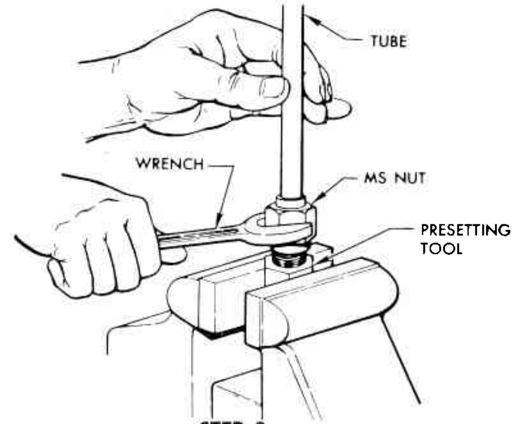
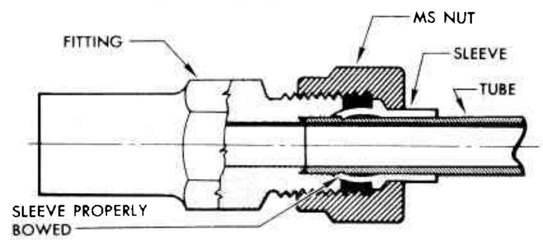


STEP 3

g. Tighten nut until sleeve cutting edge grips tube. Determine proper sleeve cutting edge grip by turning tube slowly but firmly by hand while tightening nut with a wrench. When the tube can no longer be turned between the thumb and fingers, the fitting is ready for presetting or final assembly. See Step 3.
h. Tighten nut an additional 1 to 1⅛ turns for preset or final assembly. Tube material, hardness, wall-thickness and size effect the exact number of presetting turns needed. 1⅛ turns are recommended for ⅛ and ¼-inch hard stainless tubes.



FLARELESS TUBE ASSEMBLY AND FITTING PROPERLY PRESET OR ASSEMBLED

POST ASSEMBLY INSPECTION

a. Remove presetting tool from fitting and tube. Insure that the requirements following in steps "b" through "d" are met.
b. Sleeve pilot shall contact or be within 0.010 inch of tube outside diameter.
c. Sleeve shall be slightly bowed and have a permissible maximum longitudinal 1/64-inch movement. The sleeve may be stationary or have the ability to rotate.
d. Sleeve sealing surface must be smooth, free from scores, nicks and longitudinal cracks or tool marks.

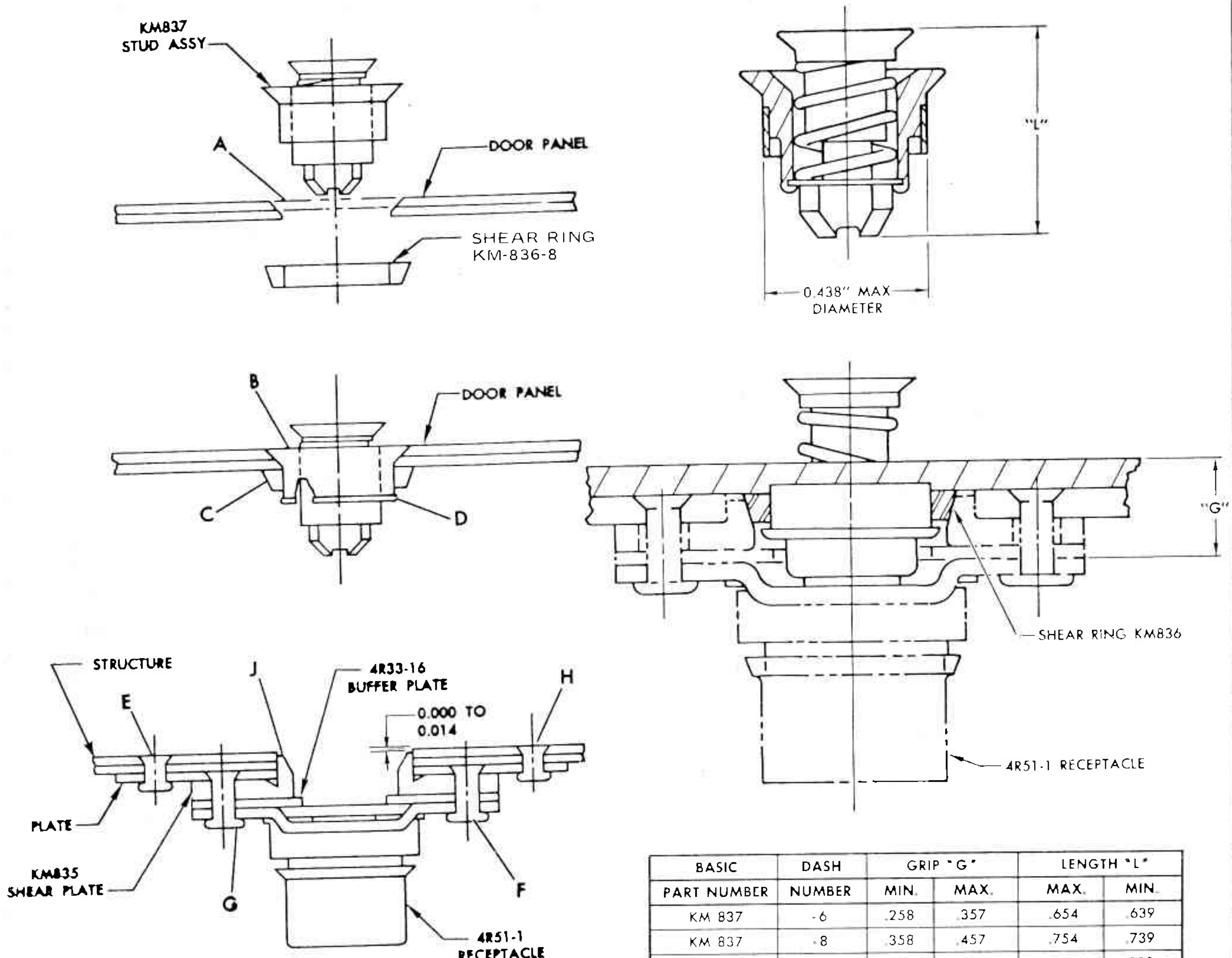
CAUTION

PRESSURE TEST TUBE ASSEMBLY IN ACCORDANCE WITH SPECIFICATION MIL-H-5440 BEFORE AIRPLANE INSTALLATION.

TABLE II

TUBE SYSTEM	LUBRICANT
HYDRAULIC	MIL-H-5606
FUEL	MIL-H-5606
OIL	SYSTEM OIL
PNEUMATIC	MIL-L-4343
OXYGEN	MIL-T-5542

Figure 1-53. MS Flareless Tube and Fitting Assembly



BASIC	DASH	GRIP "G"		LENGTH "L"	
PART NUMBER	NUMBER	MIN.	MAX.	MAX.	MIN.
KM 837	-6	.258	.357	.654	.639
KM 837	-8	.358	.457	.754	.739
KM 837	-10	.458	.557	.854	.839
KM 837	-12	.558	.657	.954	.939

NOTES:

1. SELECTION OF STUD DASH NUMBER DEPENDS ON TOTAL GRIP THICKNESS.
2. GRIP=TOTAL GRIP THICKNESS FROM PANEL SURFACE TO RECEPTACLE SURFACE. (INCLUDES OUTER PANEL, INNER FRAME, SHEAR PLATE, BUFFER PLATE, PAINT, ETC).

A. Drill 0.438 to 0.440 dia. hole (7/16 drill size) and countersink 100° x0.571 max. diameter.
B. Install Camloc stud (KM837) through outer surface panel.
C. Slip Camloc shear ring (KM836-8) over inner end of stud housing.
D. Flare end of stud housing to secure stud assy in panel. Use installation tool shown in figure 1-56. Grind off flare for removal of stud.
E. Remove damaged receptacle assembly by drilling out (4) 1/8-inch diameter rivets (30 drill size) securing receptacle to structure.
F. Drill out (2) 3/32-inch diameter rivets (39 drill size) securing plate to receptacle. Save plate, shear plate, and buffer plate for reinstallation.
G. Assemble and rivet receptacle to buffer plate, shear plate, and plate. Coat all faying surfaces of parts with sealer, Military Specification MIL-S-8802. Secure with MS20426AD3 rivets.
H. Rivet receptacle assembly to structure with same type rivets as removed (either MS20426AD4 or Q4304 rivets).
J. If new shear plate is installed, any lip projection past the outer surface of the of the structure must be milled off flush.

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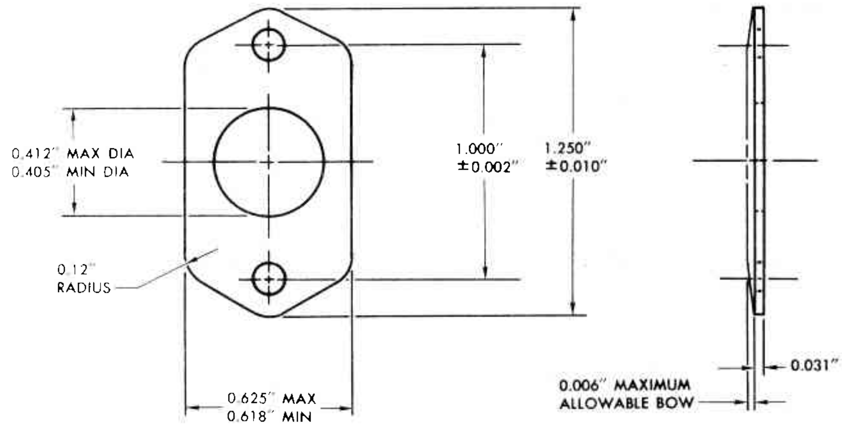
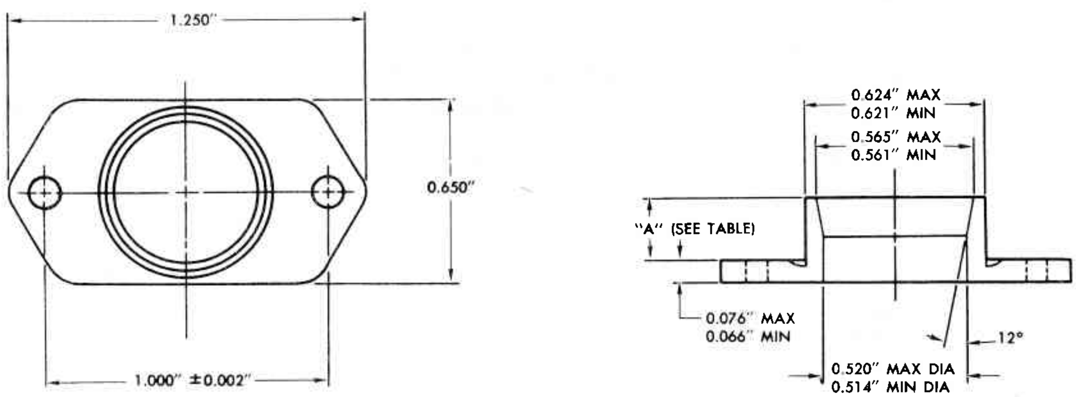
Figure 1–54. Stressed Panel Camloc Fastener–KM837 Stud and 4R51-1 Receptacle



NOTE:
USE WITH KM837 STUD
AND 4R51-1 RECEPTACLE

CAMLOC BUFFER PLATE YR33-16



PART NUMBER	"A" DIMENSION	
	MAXIMUM	MINIMUM
KM 835-8	.080	.075
-12	.120	.115
-13	.130	.125
-15	.150	.145
-16	.160	.155
-17	.170	.165
-18	.180	.175
-20	.200	.195
-21	.210	.205
-25	.250	.245
-27	.270	.265
-29	.290	.285
-42	.420	.415

NOTE:
USE WITH KM837 STUD
AND 4R51-1 RECEPTACLE

CAMLOC SHEAR PLATE KM835

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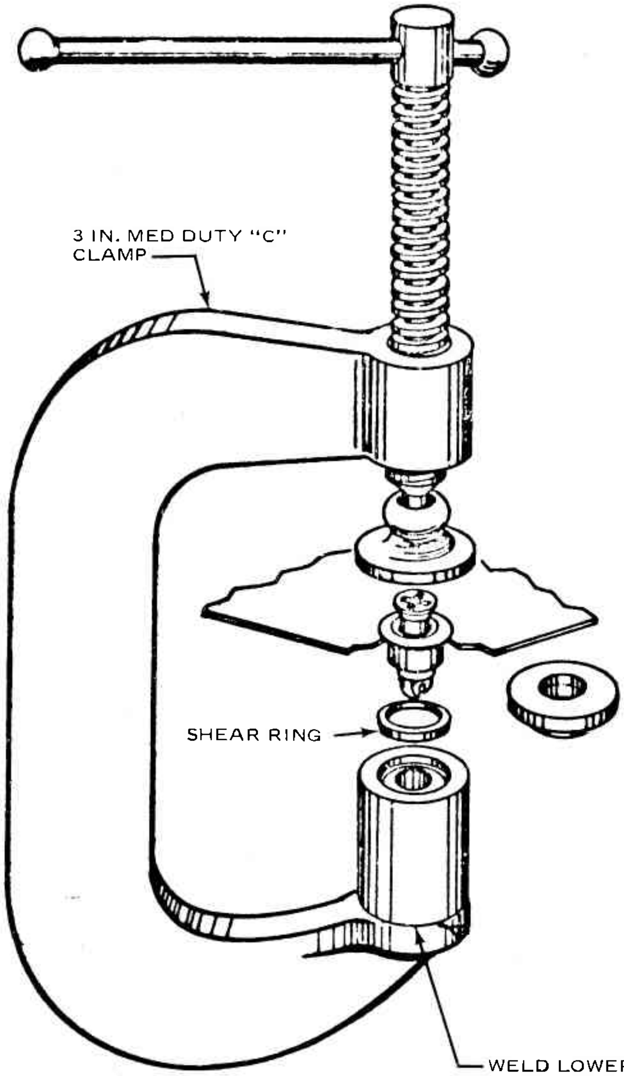
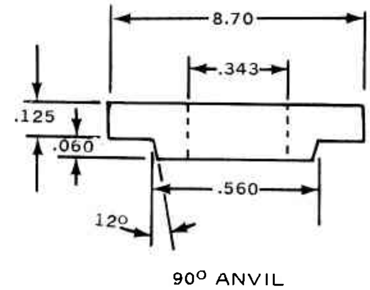
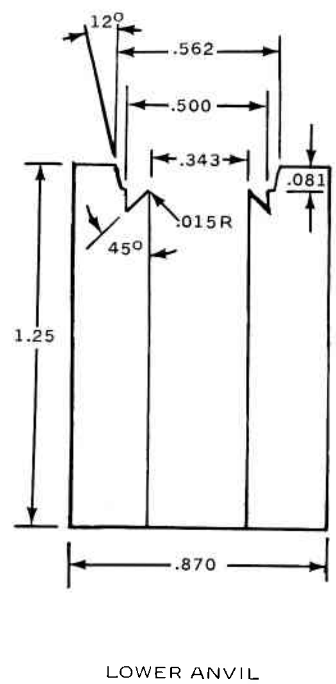
Figure 1-55. Camloc Buffer and Shear Plates





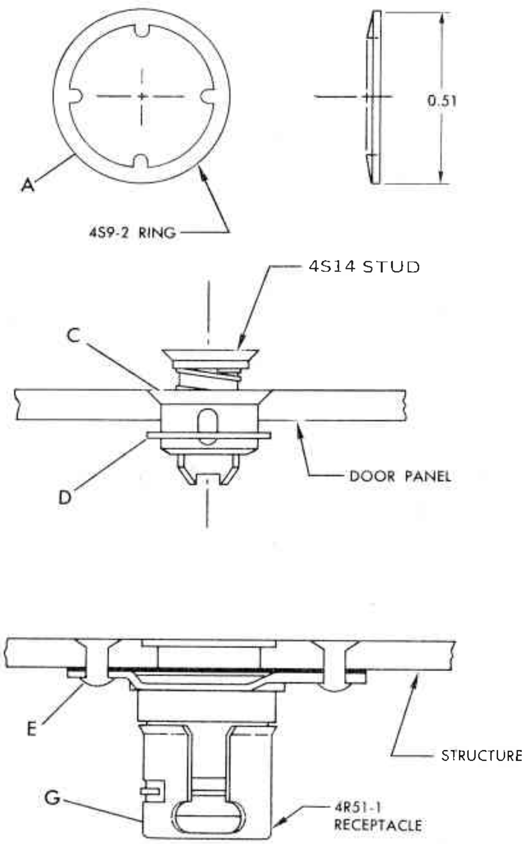


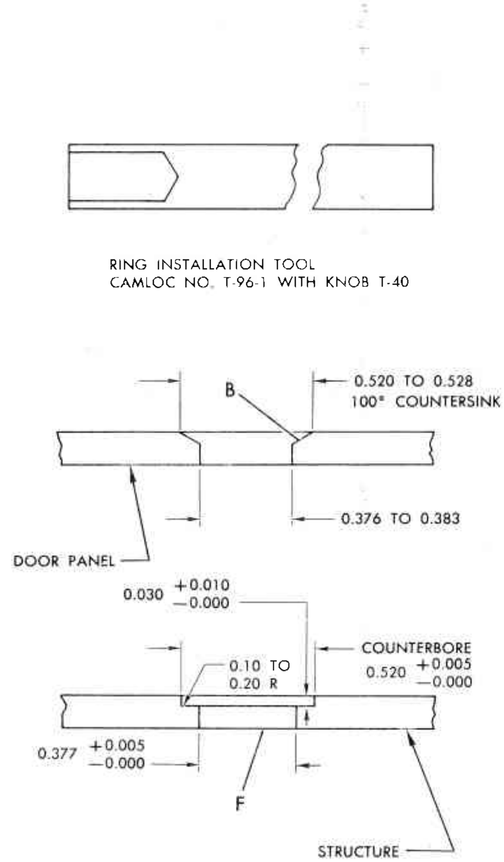
INSTALLATION PROCEDURE:

A. INSERT KM837 STUD IN DOOR PANEL.
B. REMOVE THE 90° FLARE ANVIL FROM "C" CLAMP.
C. PLACE KM836-8 SHEAR RING IN THE 45° FLARE ANVIL.
D. POSITION "C" CLAMP OVER STUD IN DOOR AND TIGHTEN TO FORM 45° FLARE.
E. REMOVE "C" CLAMP AND INSTALL 90° FLARE ANVIL.
F. POSITION "C" CLAMP OVER STUD AND TIGHTEN TO FORM 90° FLARE, SECURING STUD IN DOOR.

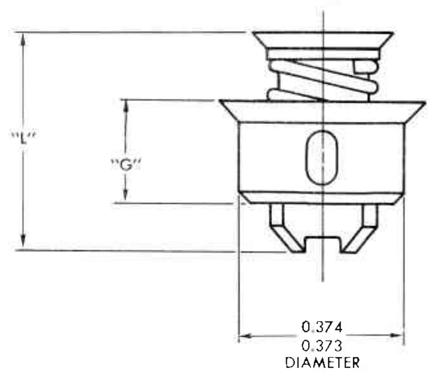
MAKE ANVILS FROM 4130 STEEL
MIL-S-675B
HEAT TREAT TO 125.000 PSI

Figure 1-56. Stressed Panel Camloc Fastener—KM837 Installation Tool





	GRIP "G"		LENGTH "L"	
PART NUMBER	MAXIMUM	MINIMUM	MAXIMUM	MINIMUM
4S14-4	.286	.274	.554	.539
4S14-6	.386	.374	.654	.639
4S14-8	.486	.474	.754	.739
4S14-10	.586	.574	.854	.839
4S14-12	.686	.674	.934	.939



Remove and replace damaged stud assy as follows:

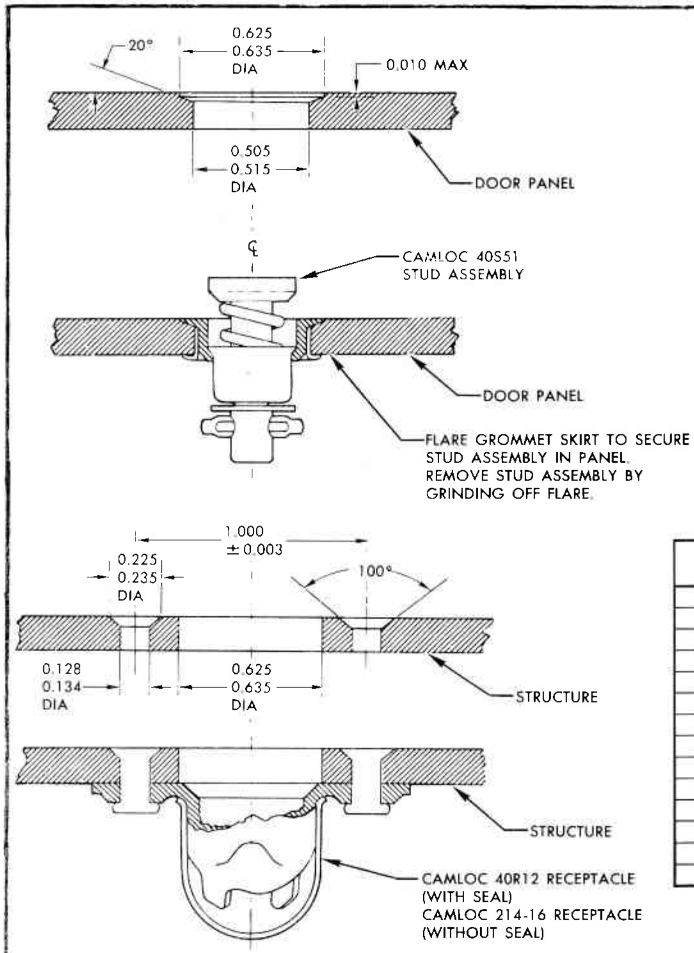
A. Cut or break stud retainer ring and remove stud assembly from panel.
B. Measure hole and countersink dimensions in panel. If panel is damaged, refer to structural engineering authorities.
C. Select camloc stud (4S14) with same grip length as old and insert in door panel.
D. Install spring steel retainer ring (Camloc 4S9-2) over inner end of stud housing. Engage the (4) inner teeth of the ring with (4) grooves in the stud housing, a tool may be manufactured as shown from a length of steel bar. Magnetize bar to hold ring in place during installation.

Remove and replace damaged receptacle as follows:

E. Drill out (2) 3/32-inch diameter rivets (#39 drill size) and remove receptacle.
F. Measure hole and counterbore dimensions in structure. If structure is damaged, refer to structural engineering.
G. Install Camloc receptacle (4R51-1) with applicable sealer applied to faying surfaces. Refer to paragraph on "Repairs to Sealing" in this section. Use **MS20426D3 rivets.**

NOTES:
1. SELECTION OF STUD DASH NUMBER DEPENDS ON TOTAL GRIP THICKNESS.
2. GRIP=TOTAL GRIP THICKNESS FROM PANEL SURFACE TO RECEPTACLE SURFACE.

Figure 1-57. Stressed Panel Camloc Fastener—4S14 Stud and 4R51-1 Receptacle



PART NUMBER BREAKDOWN

40S51 — A 040 — 10 A

"A" SUFFIX = SLOTTED HEAD STUD
NO SUFFIX = #2 PHILLIPS CROSS RECESS

2ND DASH NUMBER INDICATES "L" DIMENSION (STUD GRIP LENGTH).

1ST DASH NUMBER INDICATES GROMMET SKIRT LENGTH.

"A" = FLUSH TYPE GROMMET
"B" = PLUS FLUSH TYPE GROMMET

BASIC PART NUMBER

BASIC PART NUMBER	FIRST DASH NUMBER	"C" DIM.	PANEL THICKNESS
40S51	-A040	.155	.040 - .069
40S51	-A070	.145	.070 - .099
40S51	-A100	.165	.100 - .129
40S51	-A130	.193	.130 - .159
40S51	-A160	.220	.160 - .189
40S51	-A190	.250	.190 - .219

BASIC PART	SECOND DASH	"L" DIMENSION		"S" DIMENSION	
NO.	NO.	MIN.	MAX.	SLOTTED	CROSS
40S51	-2	.413	.423	.69	.68
40S51	-3	.443	.453	.72	.71
40S51	-4	.473	.483	.73	.74
40S51	-5	.503	.513	.78	.77
40S51	-6	.533	.543	.81	.80
40S51	-7	.563	.573	.84	.83
40S51	-8	.593	.603	.87	.86
40S51	-9	.623	.633	.98	.89
40S51	-10	.653	.663	.93	.92
40S51	-11	.683	.693	.96	.95
40S51	-12	.713	.723	.99	.98
40S51	-13	.743	.753	1.02	1.01
40S51	-14	.773	.783	1.05	1.04
40S51	-15 NOTE 1	.803	.813	1.08	1.07

USE OF THE FLARING TOOLS

Remove and replace damaged stud assembly as follows:

a. Remove lock pin and release stud. Refer to figure 1-61 for grommet removal.

CAUTION

DO NOT ATTEMPT USE OF CAMLOC PLIERS

b. Measure hole and countersink in door panel to dimensions shown. If panel is damaged, replace it.

c. Select Camloc 40S51 stud with same grip length as old stud and insert it in panel.

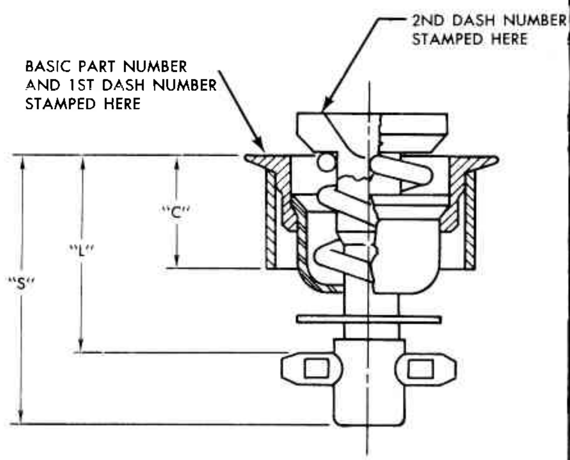
d. Install stud assy using installation tool shown in figure 1-59.

Remove and replace damaged receptacle as follows:

a. Drill out two 1/8-inch diameter rivets (No. 30 drill size) and remove receptacle.

b. Measure hole in structure to dimensions shown. If structure is damaged, refer to an aeronautical structures engineer.

c. Install Camloc receptacle with two MS20426AD4 rivets.

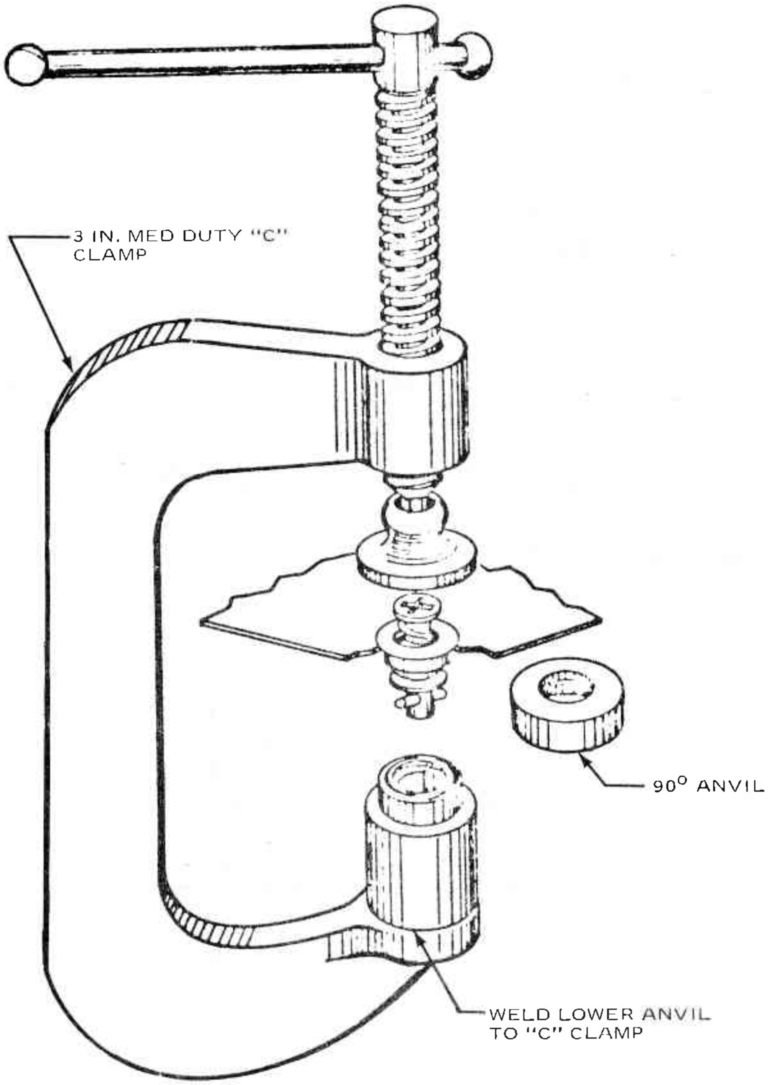


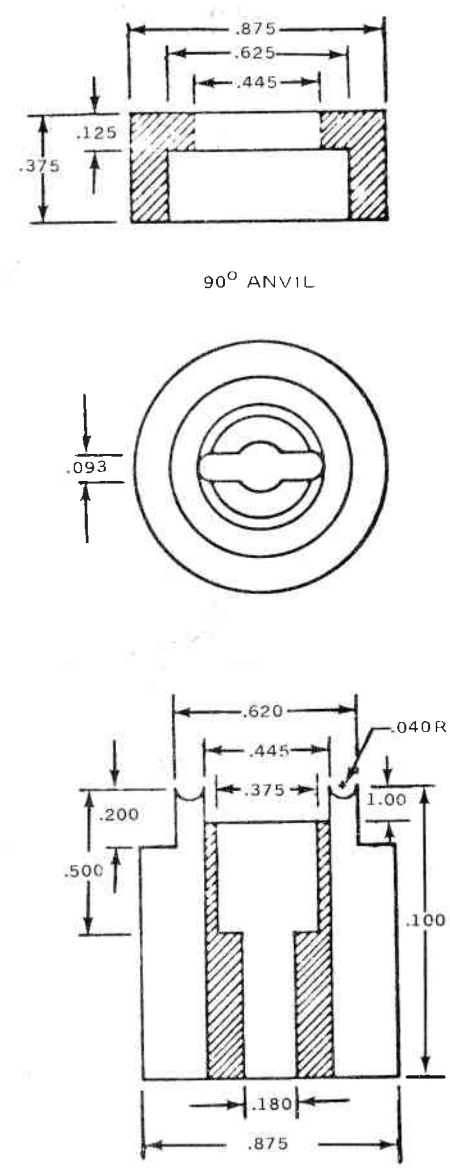
FLUSH GROMMET STUD

NOTES:

1. FOR DASH NUMBER GREATER THAN −15:
   "L" = (0.030" 2ND DASH NO.) + 353.
   "L" MAXIMUM = "L" MINIMUM + 0.010".
   "S" = SLOTTED HEAD — 0.068 + (0.03" × 2ND DASH NO.)
   CROSS HEAD — 0.062" + (0.03" × 2ND DASH NO.)
2. STUD SELECTION DEPENDS ON TOTAL GRIP ("C") THICKNESS.
3. GRIP = TOTAL GRIP THICKNESS FROM THE OUTER PANEL SURFACE TO THE RECEPTACLE MOUNTING SURFACE (INCLUDES OUTER PANEL, INNER FRAME, PAINT, GASKETS, SHIM, ETC.).
4. USE WITH 40R12 RECEPTACLE FOR SEALED INSTALLATION.
5. USE WITH 214-16 RECEPTACLE FOR UNSEALED INSTALLATION.

Figure 1-58. Removal and Replacement of Standard and Sealed Type Camloc Fasteners





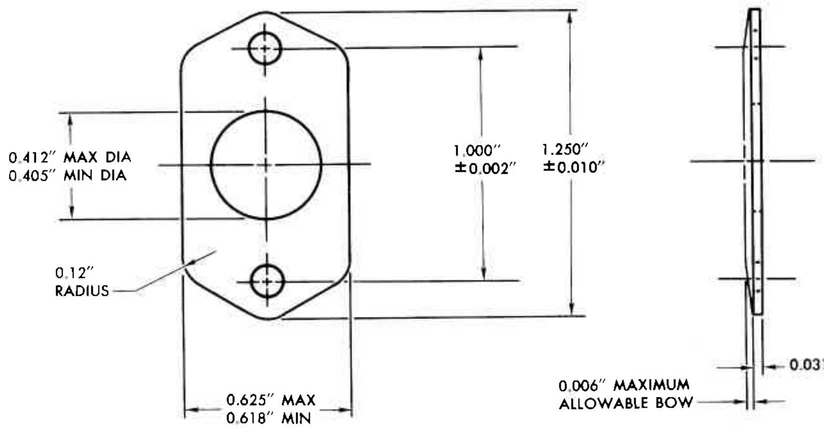
LOWER ANVIL

MAKE ANVILS FROM 4130 STEEL
MIL-S-6758
HEAT TREAT TO 125.000 PSI

INSTALLATION PROCEDURE:

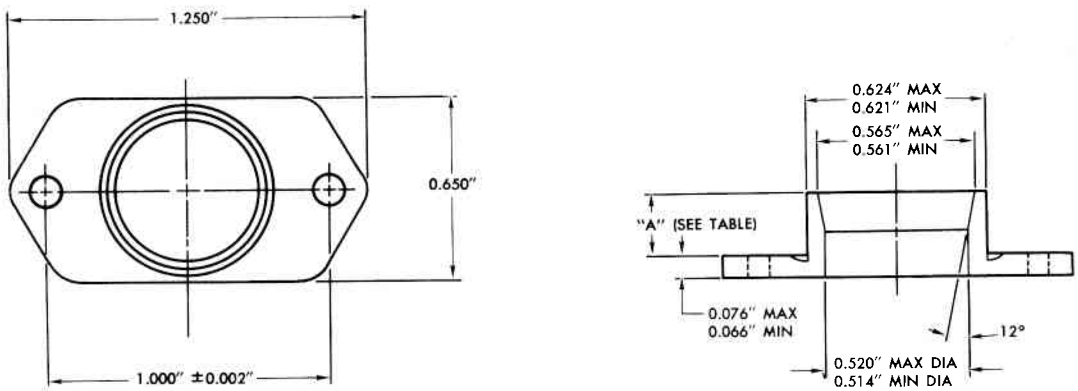
A. INSERT 40S51 STUD IN DOOR PANEL.
B. REMOVE THE 90° FLARE ANVIL FROM "C" CLAMP.
C. POSITION "C" CLAMP OVER STUD, ALIGNING CROSSPIN WITH SLOT IN ANVIL.
D. TIGHTEN "C" CLAMP TO FORM 45° FLARE.
E. REMOVE "C" CLAMP AND INSTALL 90° FLARE ANVIL.
F. POSITION "C" CLAMP OVER STUD AND TIGHTEN TO FORM 90° FLARE, SECURING STUD IN DOOR.]

Figure 1-59. Camloc Stud, 40S51 Installation Tool



NOTE:
USE WITH KM837 STUD
AND 4R51-1 RECEPTACLE

CAMLOC BUFFER PLATE YR33-16



PART NUMBER	"A" DIMENSION MAXIMUM	MINIMUM
KM 835-8	.080	.075
-12	.120	.115
-13	.130	.125
-15	.150	.145
-16	.160	.155
-17	.170	.165
-18	.180	.175
-20	.200	.195
-21	.210	.205
-25	.250	.245
-27	.270	.265
-29	.290	.285
-42	.420	.415

NOTE:
USE WITH KM837 STUD
AND 4R51-1 RECEPTACLE

CAMLOC SHEAR PLATE KM835

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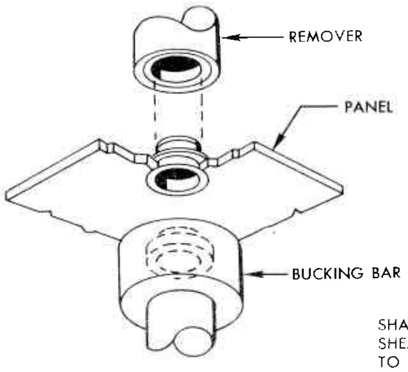
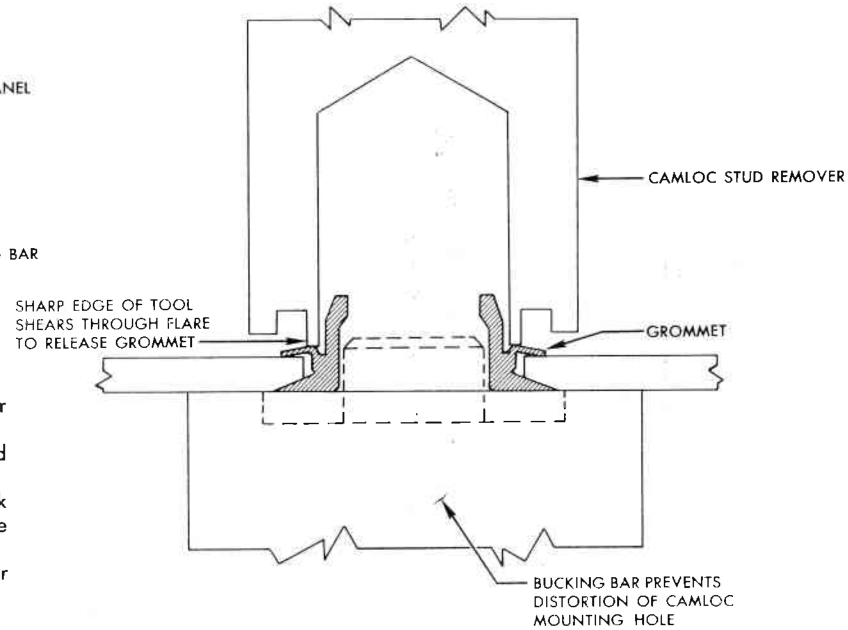
Figure 1-60. Camloc Buffer and Shear Plates



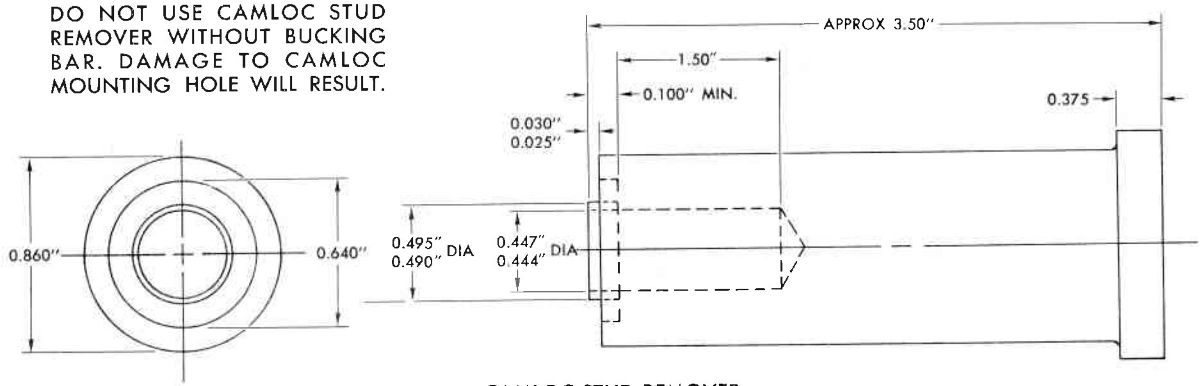


PROCEDURE

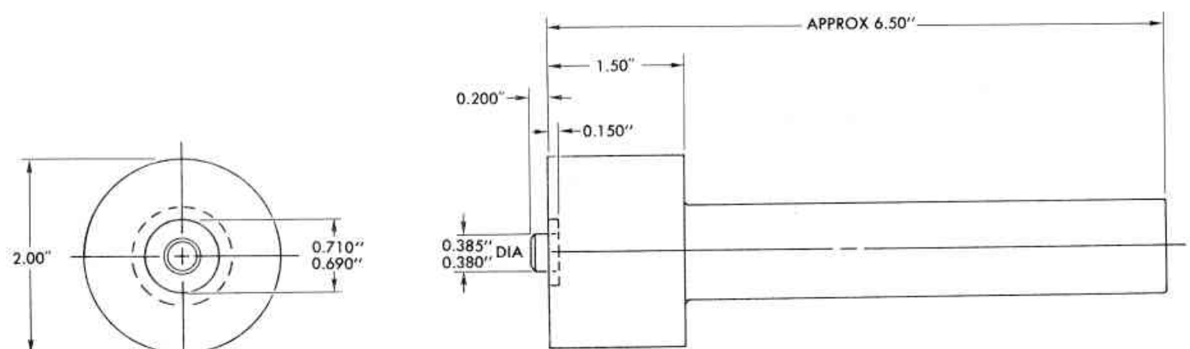
a. Locally manufacture stud remover and bucking bar, as shown.
b. Remove lock pin from camloc stud and remove stud.
c. Insert bucking bar in countersunk side of grommet and remover in flare side.
d. Rap remover with a hammer to shear grommet skirt.

CAUTION

DO NOT USE CAMLOC STUD REMOVER WITHOUT BUCKING BAR. DAMAGE TO CAMLOC MOUNTING HOLE WILL RESULT.



CAMLOC STUD REMOVER



CAMLOC BUCKING BAR

NOTE:
MANUFACTURE CAMLOC STUD REMOVER FROM TOOL STEEL S.A.E. S5 OR EQUAL. HEAT TREAT TO TOUGHEN.

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Figure 1-61. Camloc 40S51 Stud Removal Tool

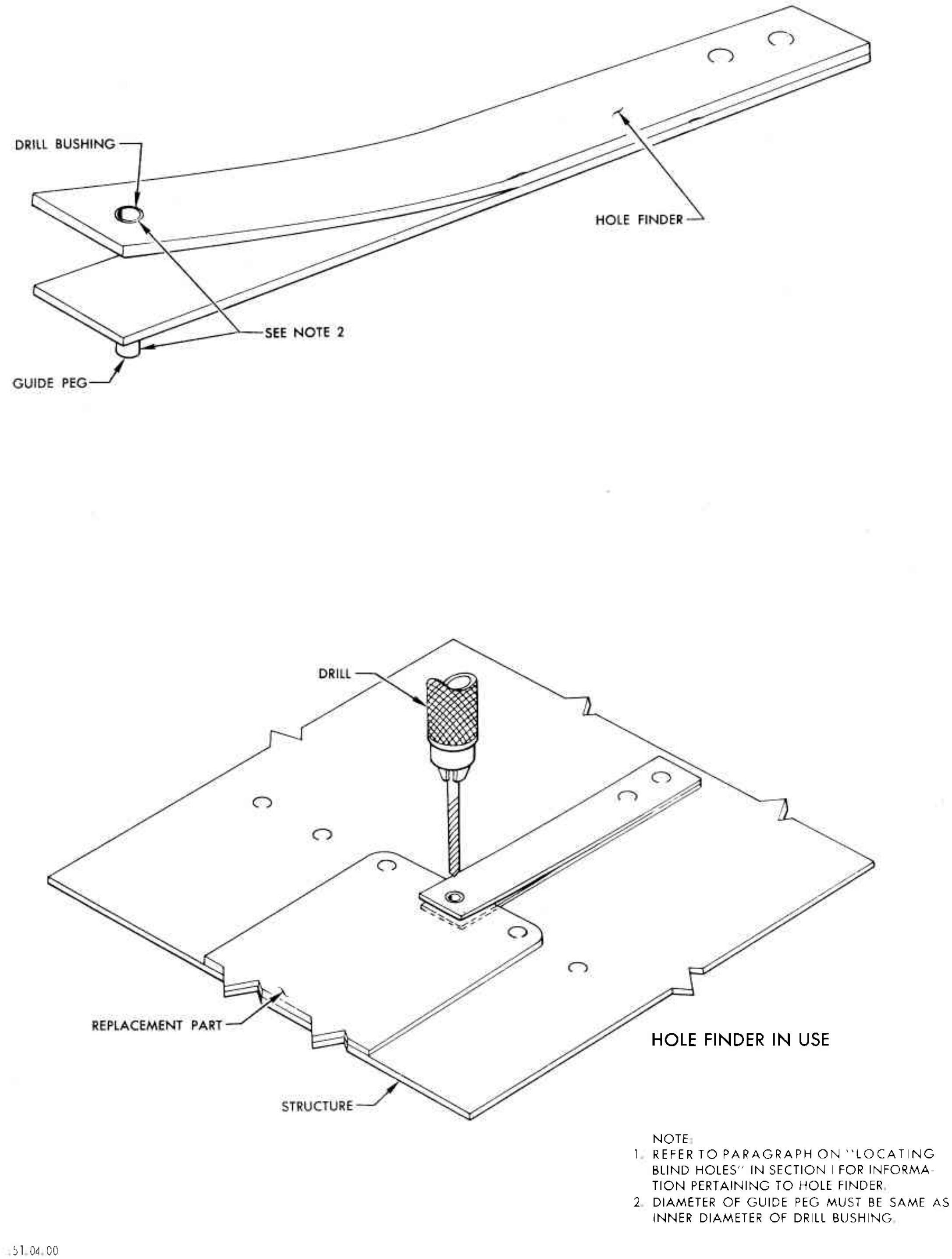
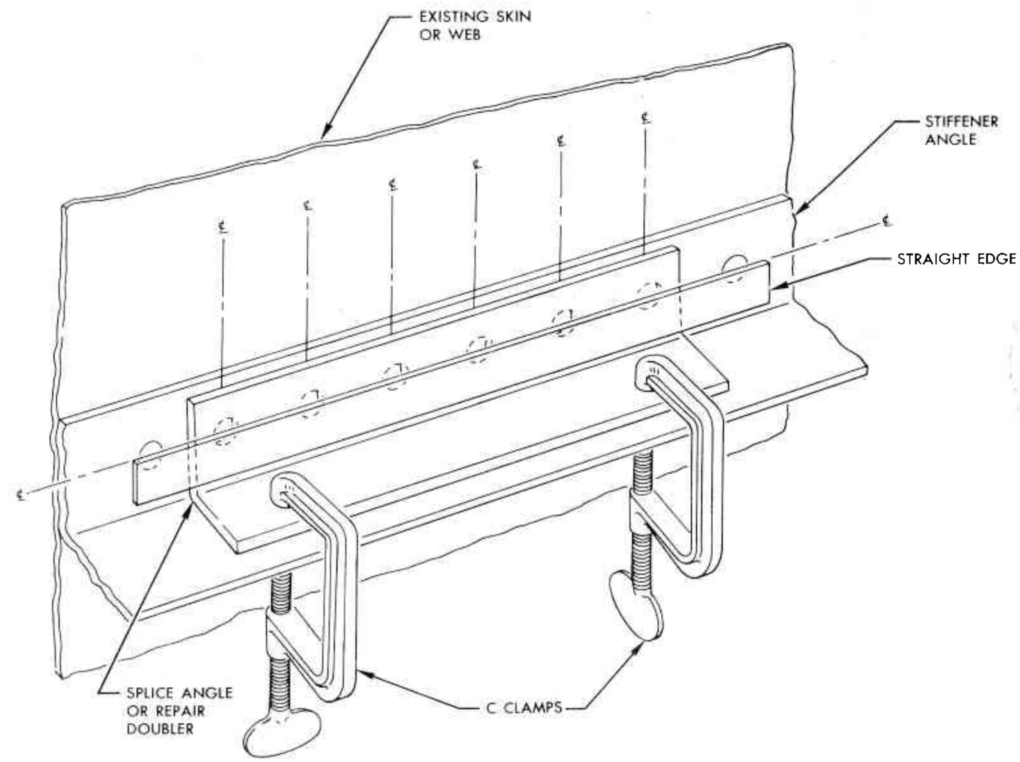


Figure 1-62. Locating Blind Holes (Sheet 1 of 2)



LOCATING BLIND HOLES BY
LINES LAYOUT METHOD

PROCEDURE

a. Remove fasteners from existing structure.
b. Using a straight edge and pencil, draw a vertical and horizontal centerline across each hole.
c. Extend horizontal and vertical centerlines beyond damaged area so that centerlines may be seen when repair part is installed.
d. Remove straight edge.
e. Install repair part.
f. Lay a straight edge across repair part and existing structure and in line with centerlines drawn in step B.
g. Drill holes through intersection of vertical and horizontal centerlines on repair parts.

NOTE

DRILL USED IN STEP G SHOULD BE AT LEAST TWO SIZES SMALLER THAN EXISTING HOLES TO ALLOW FOR ANY ERROR IN LAYING OUT CENTER LINES.

h. Complete drilling operation with final drill size.

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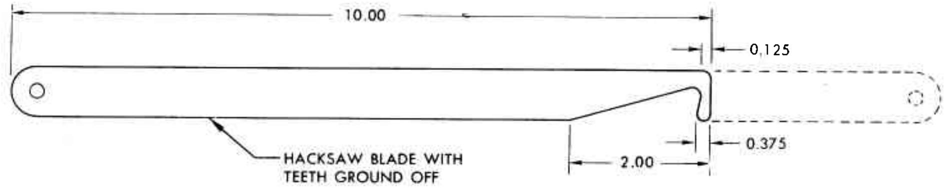
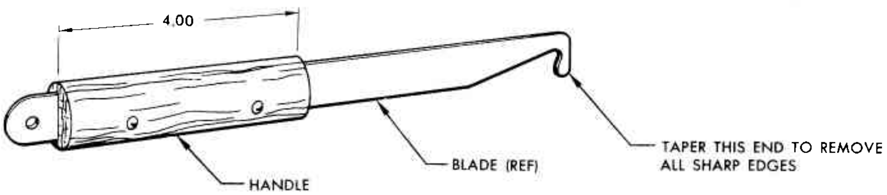
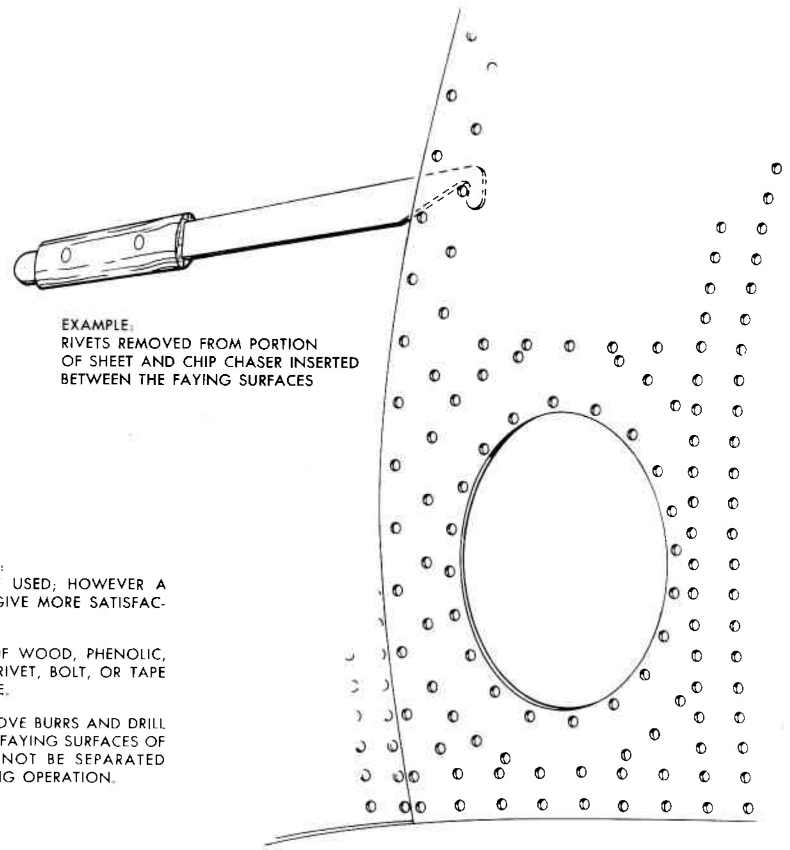
Figure 1-62. Locating Blind Holes (Sheet 2 of 2)







NOTE:

ANY THIN STEEL MAY BE USED; HOWEVER A SPRINGY MATERIAL WILL GIVE MORE SATISFACTORY SERVICE.

HANDLE MAY BE MADE OF WOOD, PHENOLIC, SCRAP PLEXIGLASS, ETC., RIVET, BOLT, OR TAPE THE HANDLE TO THE BLADE.

USE CHIP CHASER TO REMOVE BURRS AND DRILL CHIPS FROM BETWEEN THE FAYING SURFACES OF ASSEMBLIES WHICH CANNOT BE SEPARATED AFTER CUTTING OR DRILLING OPERATION.

Figure 1-63. Chip Chaser

used. The inner diameter of the drill bushing in the top leg and the outer diameter of the guide peg in the lower leg of the hole finder both match the diameter of the existing hole in the structure.

1-174. DIMPLING.

1-175. Hot Dimpling.

1-176. The hot dimpling method consists of locally elevating the temperature of the material to 162.7°C to 176.7°C (325°F to 350°F) by the use of a heated die before dimpling to reduce the possibility of cracks developing in the material. When dimpling light gage 7075-T6 or 2024-T6 is required, this method should be used. The temperature may be determined by the use of Tempilaq applied to the metal adjacent to the hole to be dimpled. Tempilaq is manufactured by Tempil Corporation, 132 W. 22nd Street, New York 11, N.Y. It is a special lacquer that melts at a predetermined temperature, and for this purpose must be rated to melt at 162.7°C (325°F). The temperature of the die should never exceed 332.2°C (650°F), and the dwell time required to melt Tempilaq should never exceed 30 seconds. Tests may be made on scrap material to determine if the die is functioning satisfactorily. See figure 1-28 for dimpling tolerances.

1-177. Equipment for Hot Dimpling.

1-178. The hot dimpling equipment may be either heavy units that are automatically timed to an operating dwell time and controlled by the operator, or simple portable equipment using a 332.2°C (650°F) soldering iron and depending entirely on the use of Tempilaq to determine the correct temperature. With the former, Tempilaq should be applied at intervals to check the accuracy of settings, while with portable equipment it should be applied at each dimple.

NOTE

Emphasis must be placed on the importance of temperature control in hot dimpling operations. Too low temperatures may result in cracked dimples while temperatures that are too high may destroy the hardness of the material.

1-179. Coin Dimpling.

1-180. See figure 1-28 (sheet 3) for the installation of rivets using the coin dimpling procedure.

1-181. PROCEDURES FOR APPLICATION OF STRUCTURAL ADHESIVE BONDS.

1-182. The structural adhesive bonding process is most commonly employed when making repairs to honeycomb structures. This process provides a strong repair without the use of rivets; however, rivets are recommended in addition to the bond when practical. When a repair of this nature is contemplated, all parts to be fabricated and other materials required to accomplish the repair shall be immediately available prior to assembly. Refer to Section X for general honeycomb repairs.

1-183. Procedure for Cleaning Faying Surfaces.

a. All painted surfaces to be repaired shall be stripped.

b. Sandpaper faying surfaces lightly with a very fine sandpaper. Do not use wet-dry type or emery paper which contain oils or abrasives that will contaminate the surfaces to be bonded.

c. Clean area to be bonded with an approved solvent, such as naphtha, toluene, or methyl-ethyl ketone. See figure 10-13 for cleaning procedure. Wipe dry with clean, soft, white, lint-free cloth. Area is clean when drying cloth does not discolor.

1-184. Procedure for Mixing and Applying Bond.

Most adhesives and accelerators are furnished in kit form (under one stock number but in separate containers) in proper proportions for mixing. When using kit form adhesives, always follow the manufacturer's mixing instructions shown on the containers. Standard mixing procedures are explained in Section XI; however, these procedures are to be used only if adhesives and accelerators are not received in kit form. See Table 10-I for honeycomb repair adhesives.

1-185. FINISH REQUIREMENTS.

1-186. Dissimilar Metals.

1-187. Contacts between certain metals tend to create corrosive action; therefore, dissimilar metals shall be defined and grouped as follows:

Group I	Magnesium alloys.
Group II	Aluminum alloys.
Group III	Zinc, cadmium, lead, tin, and steel except corrosion resistant steel.
Group IV	Copper, nickel and their alloys, chromium, corrosion resistant steel, and titanium.

The metals within each group are not dissimilar in reaction, and all metals not grouped together shall be considered dissimilar in reaction with respect to each other. The more widely spaced groups are considered the most dissimilar.

1-188. Corrosion Preventive Measures.

1-189. To reduce the possibility of corrosive action in dissimilar metals, the following precautions should be employed when making repairs:

a. Steel bolts, screws, nuts, washers, steel rivets, and repair parts in contact with dissimilar metals, after surface treatment per item "e," shall receive a coat of zinc chromate primer which may be wet or dry at the time of installation.

b. Aluminum alloy repair parts which are to be treated with primer should be cleaned with methyl, Specification TT-M-261, and treated with chemical film. Do not use iron oxide rouge, steel wool, or other similar abrasives which tend to accelerate corrosion.

c. Close tolerance bolts or press fits may be assembled with an oil bearing material that will not induce corrosion.

d. The faying surfaces or seams of all dissimilar metals, except in fuel tight or pressurized areas where a sealing compound is required, shall be treated with a total of four coats of primer. Any dissimilar metal part may have coats of primer as long as the corresponding faying surface of the dissimilar metal part has enough coats to give a total of four between the faying surfaces. As an alternate method, apply one coat of sealer, Military Specification MIL-S-81733 over prescribed detail finish to a thickness of approximately 15 mils on one of the two faying surfaces. Apply sealer in such a manner as to be squeezed out at all boundaries of joint, leaving a complete fillet all around the boundary when the excess is removed. Contact surfaces of dissimilar metals, where relative motion is involved, shall require insulation only where the end surfaces of bushings do not rotate against the dissimilar metal with which they are in contact. Special precautions shall be taken to insure that in all cases where only one of two adjacent surfaces of dissimilar metal joint requires paint coats, such coats shall be extended past the joint onto unpainted surface by at least one inch wherever possible.

e. All noncorrosion resistant steel repair parts subjected to operating temperatures below 260°C (500°F) shall be cadmium plated in accordance with Specification QQ-P-416, type II, class B. Parts subjected to operating temperatures exceeding 260°C (500°F) shall be nickel plated in accordance with Specification QQ-N-290, class II. After plating, all noncorrosive resistant steel parts shall be painted as required by the applicable portion of paragraph 1-196. Corrosion resistant steel repair parts need not be plated and shall be cleaned (passivated) by a five-minute immersion of the part in a 10 to 15 percent solution of nitric acid, followed by a thorough water rinse. For further information concerning prevention and repair of corrosion refer to T. O. 1-1-2.

1-190. Protective Coatings.

1-191. All exterior surfaces of the airplane require a protective coating, with the exception of the windshield and canopy transparencies, and a portion of the tail cone. When repair work involves the inner and outer surfaces, it is essential that all damaged paint and any existing products of corrosion be throughly removed. For corrosion removal and treatment procedures, refer to T.O. 1-1-2. Prior to and after completion of repairs, refer to T.O. 1F-106A-23 for data concerning application of new protective coating to interior and exterior surfaces, and for repairs to damaged exterior finish.

CAUTION

Aromatic hydrocarbon cleaning solvents shall not be used on this airplane because of possible detrimental effects on paint finishes and some sealant and cement materials. Do not use gasoline, alcohol, kerosene, benzene, xylene, ketones including acetone, carbon tetrachloride, fire extinguisher or de-icing fluids, lacquer thinners, aromatic hydrocarbons, ethers, glass cleaning compounds or any solvent not approved by T.O. 1F-106A-2-2-2-2 for removal of foreign matter from canopy plexiglass panels. During touchup repairs to damaged finish, or cleaning of the airplane with solvents that evaporate at a high rate, paint and solvent fumes must be carried away from the canopy to prevent any possibility of their coming in contact with unprotected plexiglass panels. This is mandatory as acrylic plastic absorbs solvent fumes which destroy its structural strength and optical quality. During repainting of the entire airplane or a major portion of the airplane, such as the wing, fin or fuselage, the canopy should be removed for storage in a cool, dry location away from solvent fumes such as may exist near paint spray or paint storage areas.

1-192. If it is impractical to remove the canopy, refer to paragraph 1-193 for method of protecting plexiglass by masking panels to prevent paint fumes from contacting acrylic plastic sections of canopy. Refer to T. O. 1-1A-12 for additional information concerning repair and maintenance of transparent plastics. Refer to T.O. 1F-106A-2-2-2-2 for approved airframe cleaning materials and procedures.

1-193. Painting Precautions in Area of Canopy Plexiglass Windows.

1-194. When touchup repair painting of a nick or scratch on the airplane finish is to be accomplished within approximately ten feet of the canopy, the transparent area of the canopy window shall be completely masked off as outlined below. If the touchup painting is done in a hangar or enclosed area, the paint fumes must be carried off by forced ventilation. If practical. the airplane should be positioned so that the forced ventilation carries the paint fumes in a direction away from the canopy. During repainting of the entire airplane or major portion of the airplane, the canopy should be removed from the airplane and stored in an area affording protection from all the injurious substances mentioned in paragraph 1-191.

a. Clean both sides of each canopy panel. Refer to T.O. 1F-106A-2-2-2-2 for complete canopy cleaning procedure.

b. Cut one thickness of flannel cloth, Specification CCC-F-456A, or two thicknesses of flannel cloth, Specification CCC-F-466, to fit both the exterior and interior surface of each canopy panel.

c. Secure the flannel cloth in place with tape, Federal Specification PPP-T-60.

NOTE

If flannel cloth is not available, cover both the exterior and interior surfaces of each canopy panel with Protex #20V adhesive paper, manufactured by Maskoff Company, Monrovia, Calif., and proceed with step "d."

d. Cut one sheet of barrier material, Specification MIL-B-131B, to fit both the exterior and interior surfaces of each canopy panel.

e. Secure the barrier material in place with tape, Federal Specification PPP-T-60.

CAUTION

Do not remove the protective covering from the canopy panels until the drying time for the paint has been completed.

f. Remove the protective covering from the canopy panels.

NOTE

If Protext #20V adhesive paper was used in this procedure, peel the adhesive paper slowly from the canopy panels to prevent an excessive buildup of an electrostatic charge in the canopy panels.

g. Upon completion of removing the protective covering from the canopy panels, refer to T.O. 1F-106A-2-2-2-2 for canopy cleaning and polishing procedures.

1-195. and 1-196. Deleted.

All data on pages 1-90, 1-90A and 1-90B, including figure 1-64, deleted.

k. Deleted.

1-197. Repairs to Rain Erosion Protective Coating on Plastic Exterior Surfaces.

a. The plastic laminated exterior surfaces (radome, vertical fin (upper) are coated with rain erosion protective coatings, either MIL-C-7439A, Class I (black) or MIL-C-83231, Class A, Type I, to prevent severe damage to the fiberglass surface when flying through rain.

NOTE

MIL-C-83231 coatings shall be applied, repaired, and removed in accordance with T.O. 1-1-24.

b. These materials will protect the fiberglass surface indefinitely, providing the coating is replaced or repaired when inspection indicates such work is necessary. No other coatings or top coats such as epoxy paint, shoe polish, or wax to enhance the appearance of plastic laminated surfaces is authorized. Polyurethane rain erosion resistant coating, Specification MIL-C-83231, is the preferred coating for general application since it provides more service life than the MIL-C-7439 neoprene. MIL-C-7439 neoprene coating should be used when conditions are such that a proper cure of the polyurethane coating cannot be obtained or when other circumstances warrant its use.

1-198. Repairs to Plastic Exterior Surfaces With MIL-C-7439A.

a. Remove loose, blistered, or erosion damaged sections of coating by sanding with emery cloth (180 grit), or equivalent, to a smooth tapered or feathered edge. Extend feather edge about one-half inch beyond the damaged area. Avoid sanding into the surface of the underlying plastic.

NOTE

The entire coating may be removed, if complete replacement is determined necessary, by application of cloths saturated with toluene to the coating for about two-minute intervals. Between applications of the cloths, vigorously agitate the coating surface with a stiff fiber bristle brush. Continue alternate cloth applications and brushing until coating is entirely removed. Care must be taken to prevent unnecessary exposure of the underlying plastic to the solvent effect of the toluene. If entire coat is removed delete steps "a" and "j" and continue with the remaining steps of the procedure.

b. Wipe surface to be coated until dust free. Use a cloth moistened with toluene.

c. Mask around area to be coated to protect adjacent areas from overspray.

d. Prepare Bostick primer, No. 1007, for spray application by thinning with three volumes of methyl ethyl ketone.

e. Spray on first coat of Bostick primer, No. 1007. Allow to dry for five minues and continue spraying until approximately four coats of primer have been applied with a five-minute drying period between each coat. Total thickness buildup of primer shall be 0.001 to 0.002 inch.

f. Allow final coat of Bostik primer to dry for 20 to 30 minutes.

NOTE

Application of the neoprene protective coating may be applied by brush when repairing an area of four square inches or less. Applications should be made by spraying when repairing an area larger than four square inches.

g. Prepare neoprene coating (Goodyear kit No. 23-56-S) for spraying by mixing Goodyear accelerator, No. 983-C, with Goodyear diluting thinner, No. 1803-C, and adding Goodyear cement, No. 1801-C, in the proportions recommended by the manufacturer. Stir thoroughly. Mix only the amount to be used within the following eight hours.

CAUTION

If 1801-C neoprene coating is not available, use any protective coating which conforms to Specification MIL-C-7439A, Class 1. In any event, the use of a protective coating which contains an anti-static compound will not be allowed for use on the radome.

h. Spray on a 0.0007 to 0.001-inch thick coating of neoprene and allow to dry for 10 to 15 minutes. Continue spraying with a 10 to 15-minute drying period between coats until a sufficient number of layers (approximately 12) have been applied to produce a total primer and coating buildup thickness of 0.007 to 0.012 inch.

i. Remove masking tape and feather edges of the applied coating by sanding with emery cloth (180 grit).

j Spray one additional coating over entire area of repair and the original coating.

k. Allow to cure for 72 hours at 21°C (70°F) or for two to three hours at 66°C (150°F) before use.

1-199. STRUCTURAL FIBERGLASS LAMINATES — MINOR SURFACE REPAIRS.

1-200. Scratch and abrasion damage to the outer surface of fiberglass laminate skins may be repaired at the advanced base level, as directed in the following procedure, if the damage limits are not exceeded for the particular laminate. Refer to applicable component section in this handbook for damage limits.

1-201. Procedure for Minor Surface Repairs.

1-202. Minor surface repairs shall consist of filling in the damaged area with a mixture of catalyzed resin and chopped glass fibers or the laying in of several layers of catalyzed resin-impregnated glass cloth. If the damage has occurred in an area covered by rain-erosion coating, refer to paragraph 1-197 for procedure on removal and replacement of the rain-erosion coating. Generally, a wide abrasion will require patching with sections of glass cloth, while narrow scratches may be loaded with a thick mixture of chopped glass fibers and resin.

a. Remove rough edges of laminate at edges of scratch or abraded area by hand sanding with number 40 to 60 abrasive grit.

b. Remove abrasive particles and dirt from repair area with a cloth moistened with toluene.

c. Catalyze required quantity of resin for filling damaged area by mixing the following weights of resin and catalyst:

100 parts of resin, Epon 815 or 828, Shell Chemical Corp.	and	25 parts of catalyst, Curing Agent T-1, Shell Chemical Corp.

For an alternate, use:

100 parts of resin, Epon 828, Shell Chemical Corp.	and	10 parts of catalyst, Hardener 951, Furane Plastics, Inc.

Measure by weight and mix thoroughly in a clean container. Avoid unnecessary breathing of gases emanating from mixture. Leave container open to prevent possible mild explosion of container due to gas formation.

NOTE

For narrow scratches, proceed with step "d" and delete step "e." For wide abrasions, delete step "d" and continue with step "e."

d. Prepare a thick paste of catalyzed resin mixed with glass fibers chopped from glass cloth. Work paste into scratch with a flat blade or spatula until repair extends slightly beyond flush.

e. Carefully sand the skin in a roughly circular area around the abrasion to produce a neatly feathered or tapered edge. Do not deepen sanded area below first three layers. Taper edge about 0.5 inch wide around damage. Cut filler patches from a section of clean number 181 glass cloth conforming to Specification MIL-F-9084. Cut patches large enough to overlap edges of sanded area. Coat damaged area and one filler patch with brush application of catalyzed resin. Apply resin-impregnated patch to damaged area and brush out air bubbles and excess resin. Continue application of resin-impregnated filler patches until repair is built up slightly above the surface of surrounding area. Remove excess resin that may have been worked out around edges of repair with a cloth moistened with toluene.

f. Cure repair at 150° to 200°F for two to four hours. Use portable infrared heat lamps or heating pads for heat application.

g. After curing, remove excess material and smooth surface of the repair by hand sanding to proper contour. Use number 80 to 120 abrasive grit for finish cut.

h. Mix a small amount of resin and catalyst as specified in step "c." Brush on a thin finish coating of resin over entire area of repair. Extend finish coat a minimum of 0.5 inch beyond edges of repair. Allow to dry at room temperature. Do not sand after cure is completed.

1-202A. DRAG CHUTE CANISTER REPAIR.

1-202B. General canister repair will be accomplished in accordance with existing instructions for non-structural fiberglass. Repair procedure for cracks and damaged rivet holes in canister lower aft body will be accomplished as follows (refer to figure 1-64A):

NOTE

Steps preceded by an asterisk shall apply only when a repair is required for damaged rivet holes.

a. Remove rough edges of laminate at edges of crack by hand-sanding with No. 40 to 60 abrasive grit.

*b. If spacing permits, relocate rivet holes a minimum of 1/4 inch from the damaged holes. Repair damaged holes by tapering from the inner surface and filling the tapered holes with as many plies of No. 181 cloth as required.

*c. If step "b" cannot be accomplished, sand the inner surface aft edge 0.030 inch deep, 0.6 inch wide, and 2.2 inches long, as shown in figure 1-64C, Views B-B and D-D. Note the 0.7 inch taper in View B-B. Sand the outer surface to smooth out all rough edges.

d. Remove abrasive particles and dirt from repair area with a cloth moistened with toluene.

e. Cut repair patches from a section of clean No. 181 cloth, conforming to Specification MIL-F-9084, large enough to extend 1/2 inch beyond the crack in all directions. Cut these patches to have two legs, as shown in figure 1-64B, View A-A, to serve as fillers for the reinforcement patches.

*f. Cut inner and extended reinforcement patches from clean No. 181 cloth to the shapes shown in figure 1-64C, Views B-B and D-D.

g. Coat damaged area and one repair patch with brush application of resin, conforming to Specification MIL-R-9499A, Type 1, or MIL-R-7575B, Grade A or B.

h. Apply resin impregnated repair patch to damaged area, as shown in figure 1-64B, View A-A, and brush out air bubbles and excess resin.

i. Apply at least two more resin impregnated repair patches, in accordance with steps "g" and "h" *j. Coat sanded area on inner patch and on extended reinforcement patch with brush application of resin.

*k. Apply the resin impregnated inner and extended reinforcement patches, as shown in figure 1-64C, Views B-B and D-D.

*l. Apply resin impregnated inner patches, as required, to build up the repaired area, flush with the inner surface.

*m. Apply a second resin impregnated extended reinforcement patch on the outer surface.

n. Apply three reinforcement patches, 0.5 x 4.8 inches, cut from clean No. 181 cloth, on each side of enclosure centerline, as shown in figure 1-64B, Views A-A and C-C. (Only one such patch will be applied when step "m" has been accomplished.)

o. Fill all crack voids on inside of enclosure with resin. Small strips of No. 181 cloth may be used, as required, provided the original surface smoothness and flushness can be restored after curing.

p. Cure the repaired area(s) as specified in the resin vendor's instructions.

q. After curing, remove excess material and smooth surface of the repair(s) by hand-sanding to proper contour. Use No. 80 to 120 abrasive grit to accomplish final sanding.

r. Brush on a thin finish coating of resin over entire area(s) of repair. Extend finish coat a minimum of 0.5 inch beyond edges of repair. Allow to dry at room temperature. Do not sand after cure is complete.

1-203. CRASH HANDLING AND SHIPPING.

1-204. Crash handling and shipping covers general information on the handling of the airplane and various removable components after an incident involving damage to the airplane.

1-205. CRASH LANDING.

1-206. If at all possible, standard maintenance procedures should be followed; however, firm procedures cannot be established since each situation must be evaluated individually. Factors such as extent of damage, damage to primary structure, repair allowances, topography, accessibility of crash site, distance to repair site, road clearances, and weight limitations of available roads must be considered in determining practicality of reclamation and salvage operations. Refer to T.O. 1F-106A-2-2-2-2 for standard ground handling procedures. A successful touchdown on level terrain with all gear retracted and the airplane landed in a level attitude will result in major damage to the following aircraft structural components: pitot static boom and fiberglass radome; nose and main landing gear and their attach point supporting structure; lower longerons; lower portion of fuselage bulkheads; beltframes and skin panels; missile bay doors; and the wings and control surfaces. The engine may incur damage from flying debris. Refer to Section IX for a complete fuselage damage survey applicable to nose and main landing gear failure.

1-207. Hoisting Equipment.

1-208. Special equipment required for hoisting the airframe and various removable components is illustrated on figures 1-65 and 1-66.

1-209. Removal of Airplane Components.

1-210. All electronic equipment should be removed from the airplane prior to any hoisting or lifting operation.

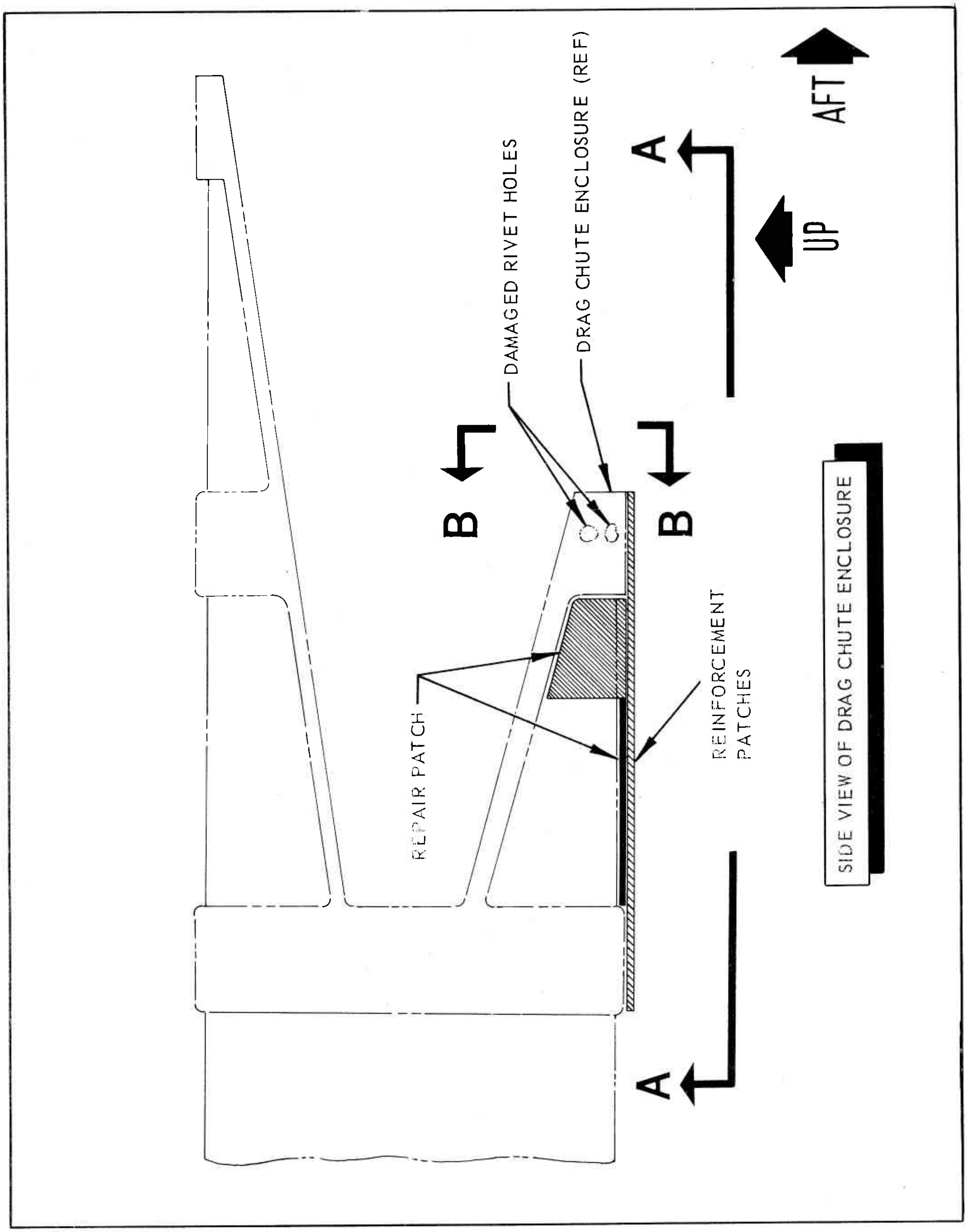


Figure 1-64A. Canister Repair Procedure

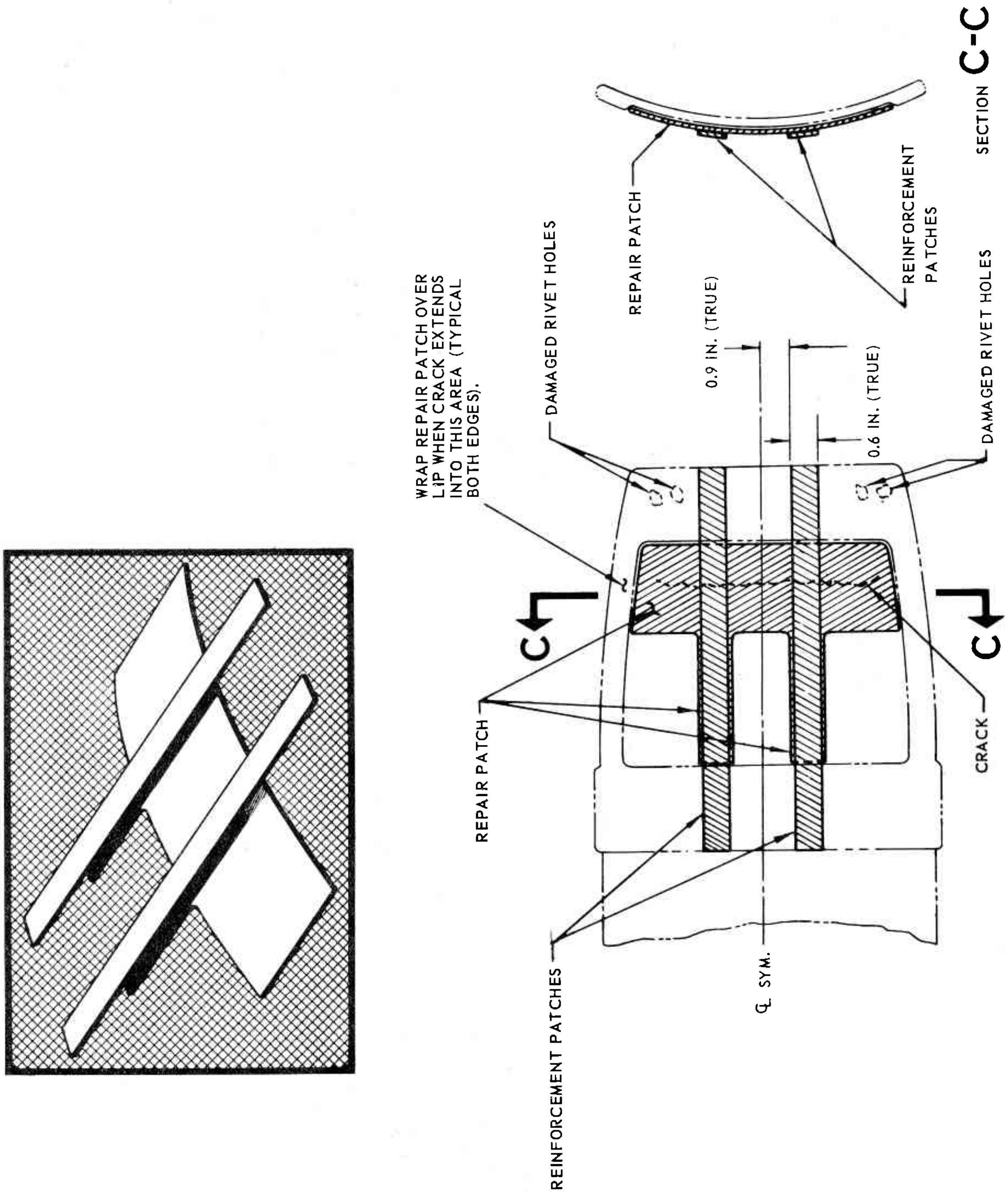


Figure 1-64B. Canister Repair Procedure

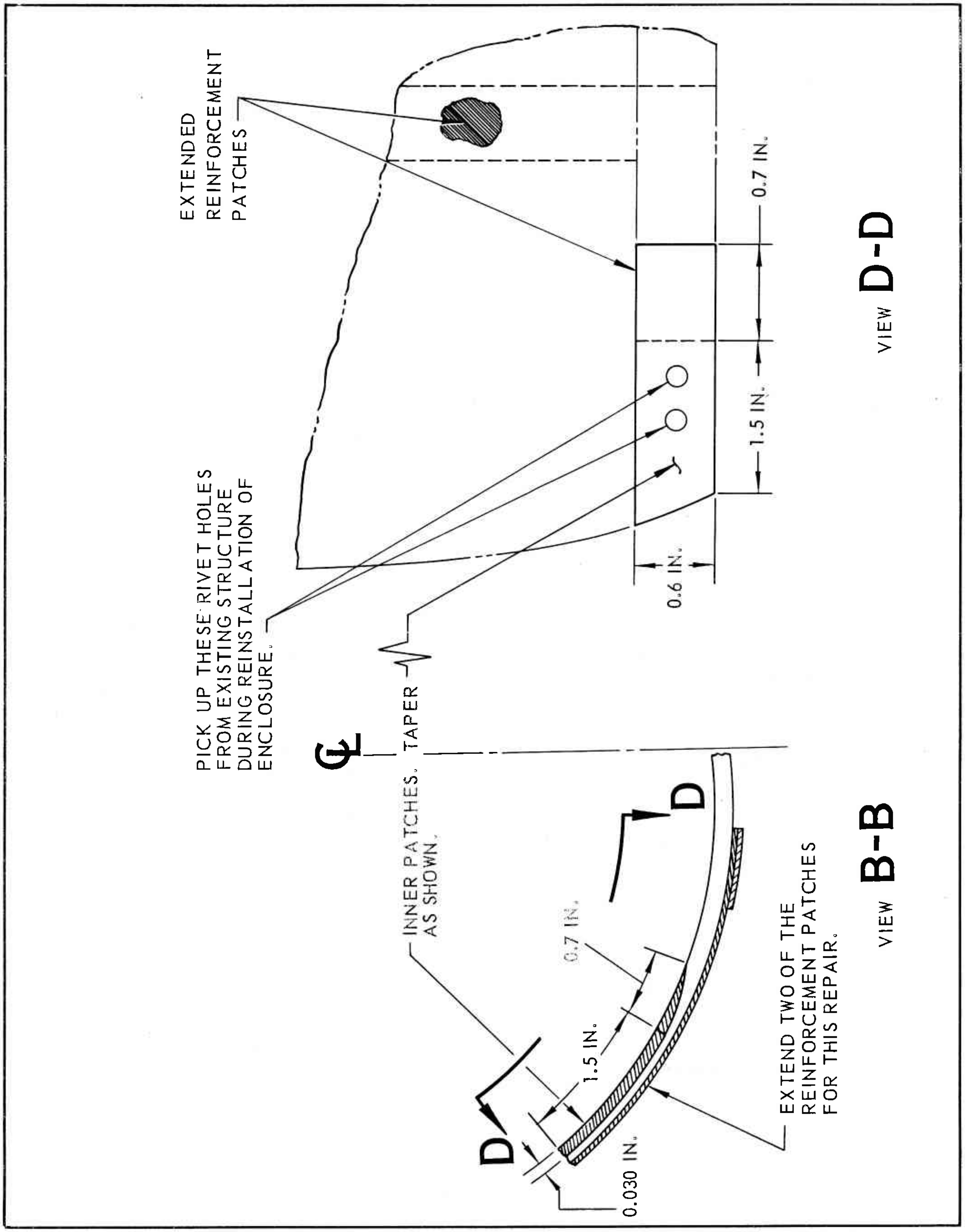
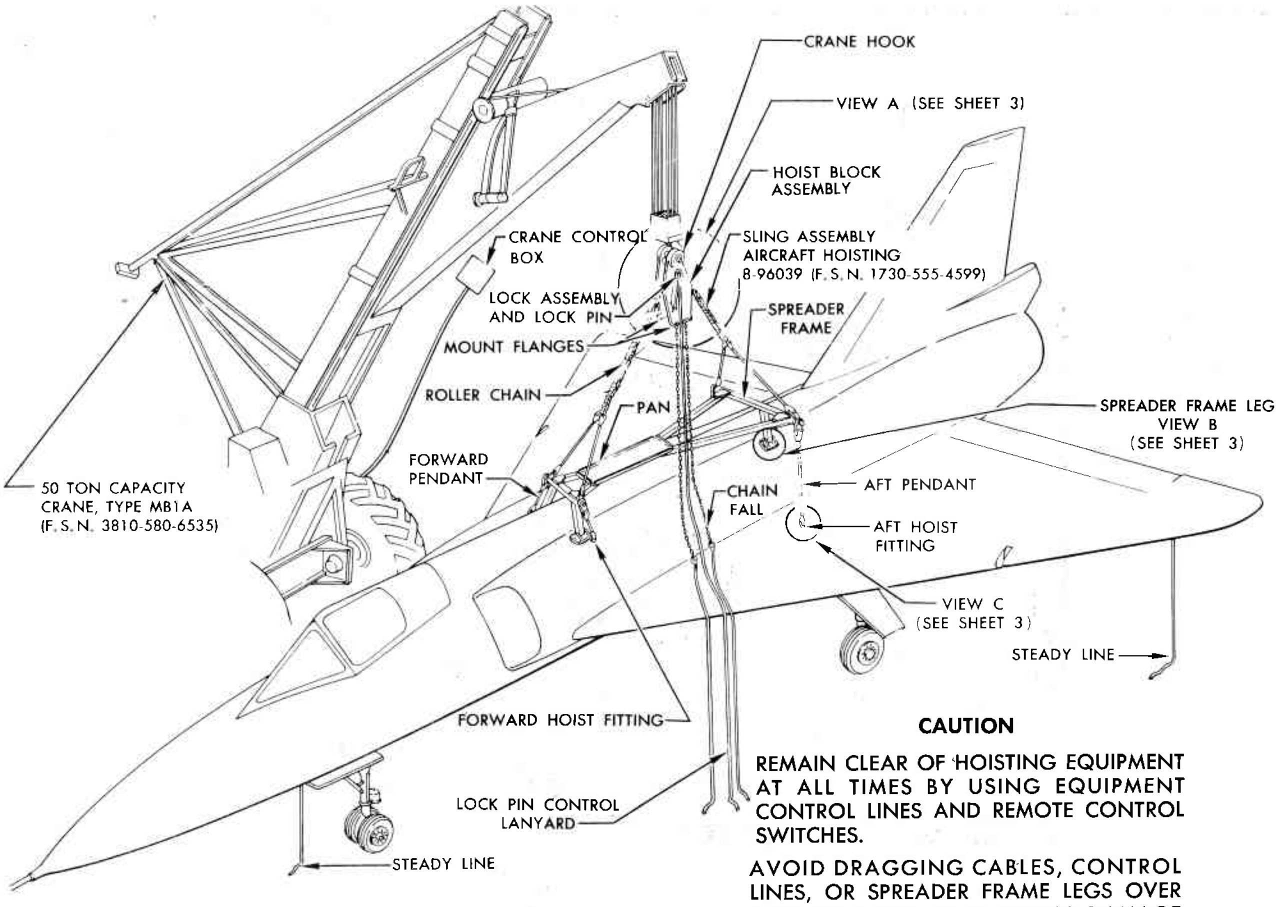


Figure 1-64C. Canister Repair Procedure



CAUTION

REMAIN CLEAR OF HOISTING EQUIPMENT AT ALL TIMES BY USING EQUIPMENT CONTROL LINES AND REMOTE CONTROL SWITCHES.

AVOID DRAGGING CABLES, CONTROL LINES, OR SPREADER FRAME LEGS OVER AIRPLANE SURFACES TO PREVENT DAMAGE TO AIRPLANE SKIN.

HOISTING ENTIRE AIRPLANE

a. Disconnect hoist block from 8-96039 (F. S. N. 1730-555-4599) hoisting sling assembly spreader frame mount pan by removing four attachment bolts from mount flanges. Arrange all pendant cables and fittings on spreader frame to function freely.
b. Pickup hoist sling assembly with 50-ton capacity crane, Type MB1A (F. S. N. 3810-580-6535).
c. Slowly raise hoist sling assembly until roller chain is properly seated over sprocket wheel.
d. Pull lock pin release lanyard and continue raising hoist sling assembly, allowing sprocket wheel to turn freely and hunt balanced position as weight of spreader frame is imposed. Utilize chainfall control as required, to assist in balancing spreader frame.
e. When proper balance has been attained, release lock pin control lanyard to allow engagement of lock pin through sprocket wheel; then raise sling assembly to desired level.

CAUTION

BE SURE THAT LOCK PIN IS IN FULLY LOCKED POSITION (NO RED VISIBLE ON LOCK PIN) PRIOR TO BEGINNING HOISTING OPERATION.

f. Position sling assembly over airplane with crane and carefully lower it into position for attaching hoist fittings, then seat padded legs on fuselage.
g. Remove screw plugs at forward and aft airplane hoist points.
h. Detach hoist fittings from pendant cables of sling assembly by removing shackle pins, then screw hoist fitting firmly into threaded hoist pins on the airplane.
i. Attach pendant cables to airplane hoist fittings and insert shackle pins. Safety shackle pins with cotter key.
j. Ascertain that airplane parking brakes are released.
k. Attach a steady line to either wing tie down fitting and a second line to nose gear. Station a man at each line to hold taut to prevent airplane from turning while it is being hoisted.
l. Station a man to hold tension on lock pin control lanyard, one to operate crane lift control box and one in crane cab. Raise crane lift to apply sufficient tension on hoist sling assembly to observe airplane balance. The angle at which the airplane tips will indicate which way to move crane boom and/or to operate chain fall control to obtain desired balance. Engage lock (by releasing lock pin control lanyard) and make necessary trial lifts and readjustments until true balance is obtained. Fully engage lock pin (no red visible) for each trial lift.
m. When true balance point has been reached, raise airplane to desired height.

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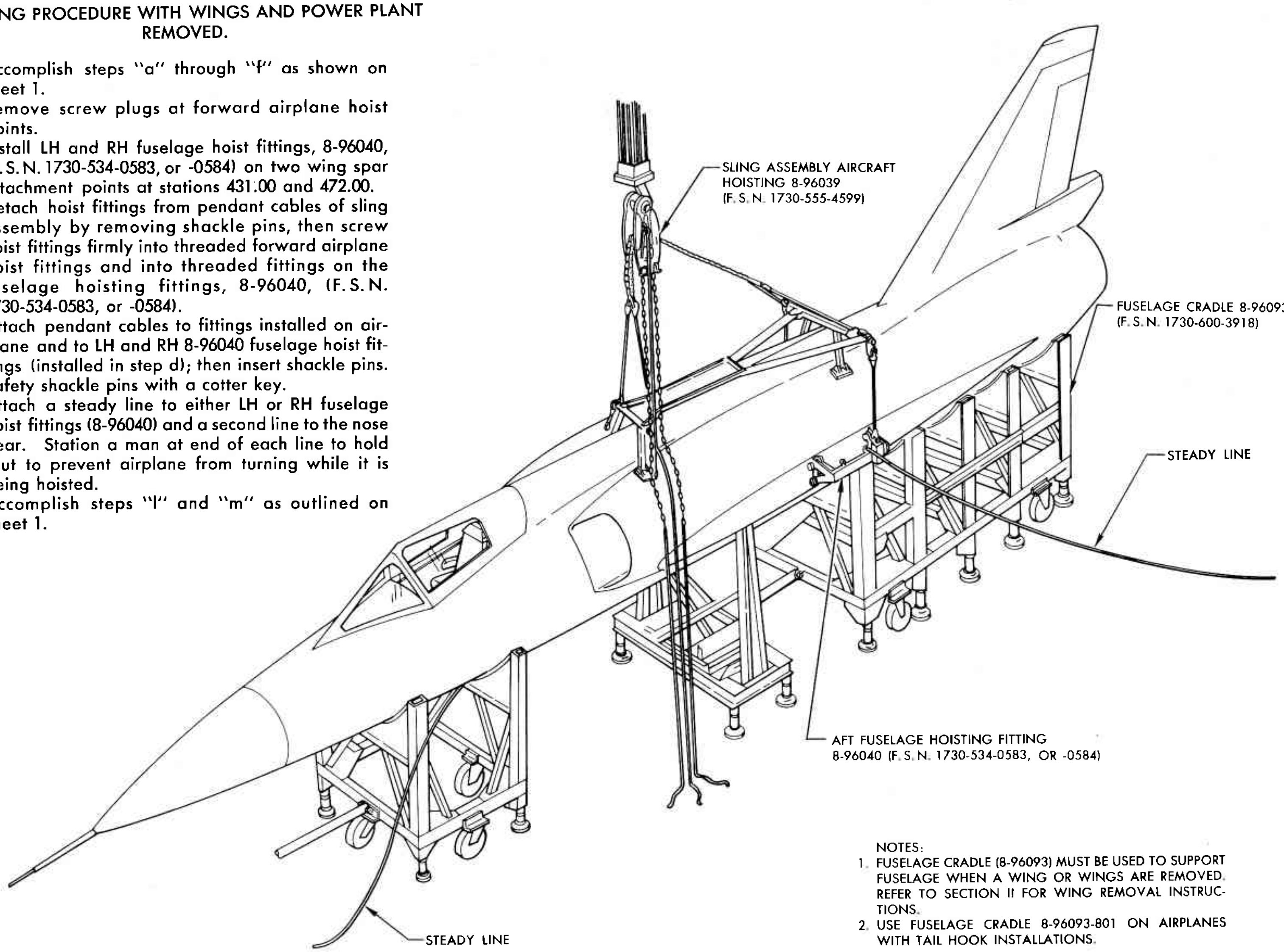
Figure 1-65. Airplane Hoisting (Figure 1 of 3)

HOISTING PROCEDURE WITH WINGS AND POWER PLANT REMOVED.

a. Accomplish steps "a" through "f" as shown on sheet 1.
b. Remove screw plugs at forward airplane hoist points.
c. Install LH and RH fuselage hoist fittings, 8-96040, (F. S. N. 1730-534-0583, or -0584) on two wing spar attachment points at stations 431.00 and 472.00.
d. Detach hoist fittings from pendant cables of sling assembly by removing shackle pins, then screw hoist fittings firmly into threaded forward airplane hoist fittings and into threaded fittings on the fuselage hoisting fittings, 8-96040, (F. S. N. 1730-534-0583, or -0584).
e. Attach pendant cables to fittings installed on airplane and to LH and RH 8-96040 fuselage hoist fittings (installed in step d); then insert shackle pins. Safety shackle pins with a cotter key.
f. Attach a steady line to either LH or RH fuselage hoist fittings (8-96040) and a second line to the nose gear. Station a man at end of each line to hold taut to prevent airplane from turning while it is being hoisted.
g. Accomplish steps "l" and "m" as outlined on sheet 1.



NOTES:
1. FUSELAGE CRADLE (8-96093) MUST BE USED TO SUPPORT FUSELAGE WHEN A WING OR WINGS ARE REMOVED. REFER TO SECTION II FOR WING REMOVAL INSTRUCTIONS.
2. USE FUSELAGE CRADLE 8-96093-801 ON AIRPLANES WITH TAIL HOOK INSTALLATIONS.

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Figure 1-65. Airplane Hoisting (Sheet 2 of 3)

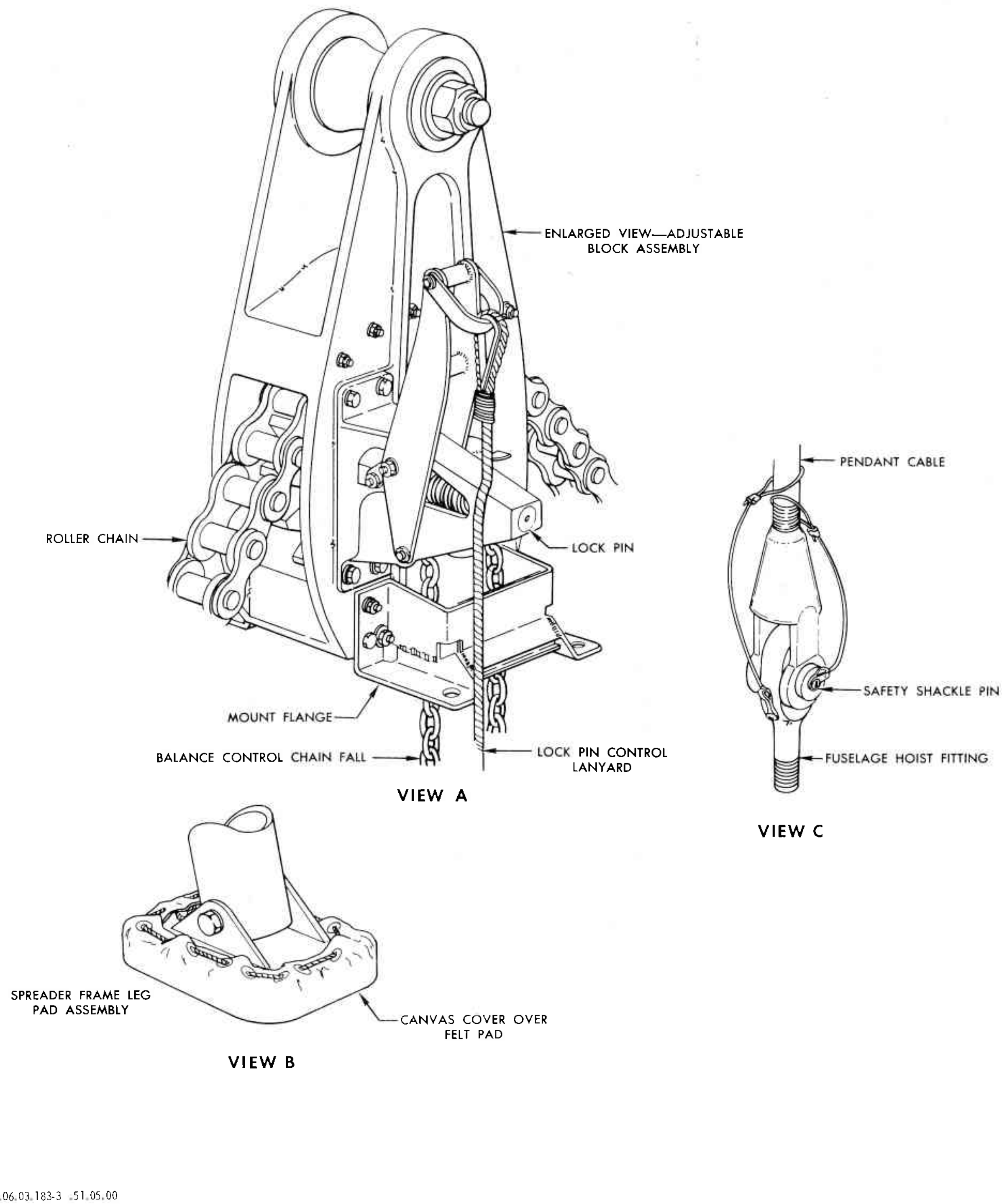


Figure 1-65. Airplane Hoisting (Sheet 3 of 3)

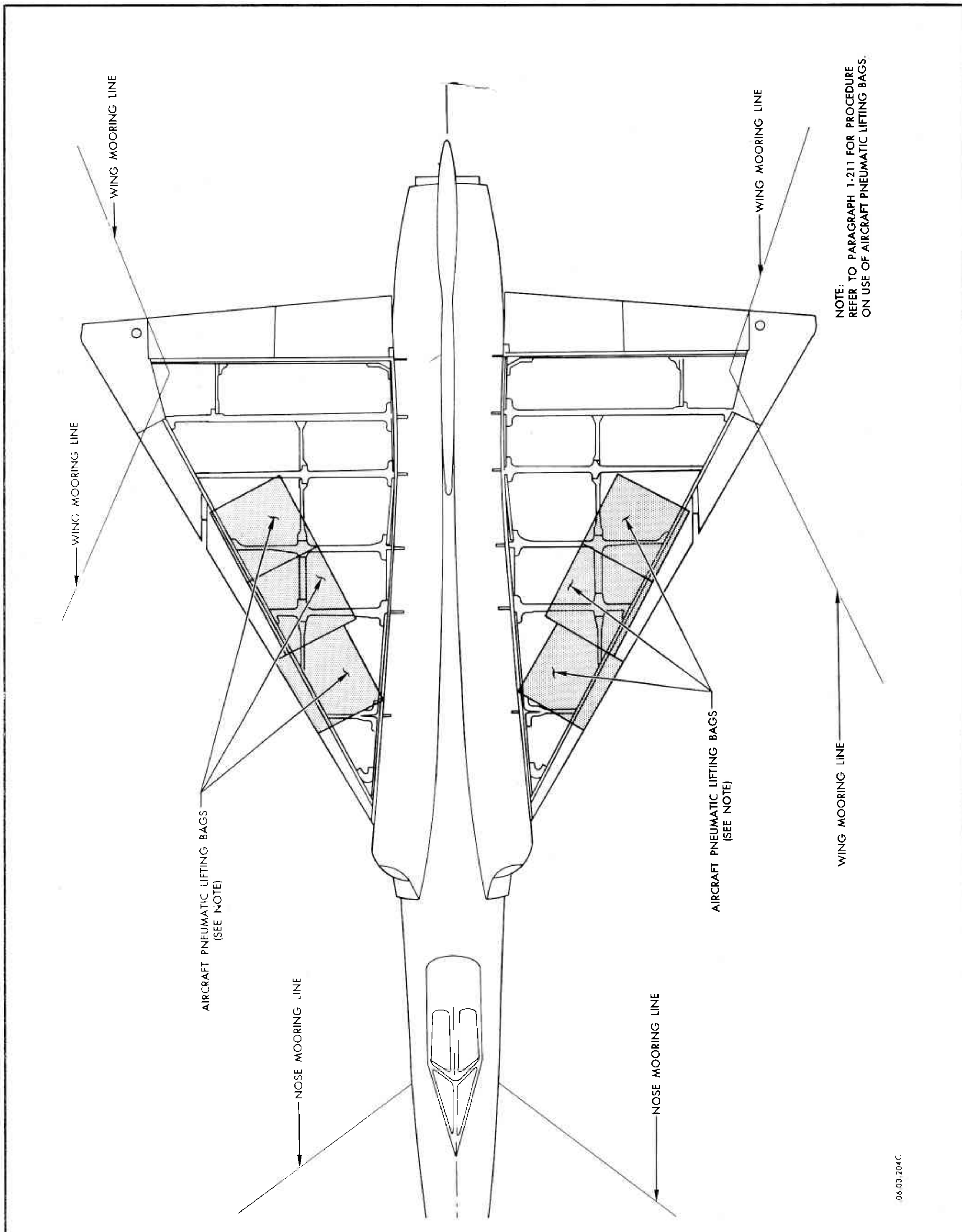


Figure 1-66. Emergency Airplane Lifting

1-211. Lifting Airframe.

1-212. Airplane pneumatic lifting bags (1730-263-2962), type F2, provide a means of raising the airplane from a gear-up landing position to a position which will allow jacks to be inserted when other methods of lifting are impractical. Refer to paragraph 1-215 for jacking provisions. Each lifting bag is designed to lift a total of 24,000 pounds to a maximum height of six feet. Lacing bands are provided to permit the lacing together of two bags when more than six feet of lift is required. Bags are not to be used as supports when other methods of support are available. Each bag is provided with a tarpaulin cover which serves as protection for the bag in shipping or while in storage, and also serves as a ground cover on which to place the bag during the lifting operation. The procedure for lifting an airplane using pneumatic lifting bags is as follows:

a. Moor or tie down the airplane to restrict its side-to-side and forward or backward movement. Mooring tie down points are shown on figure 1-66.

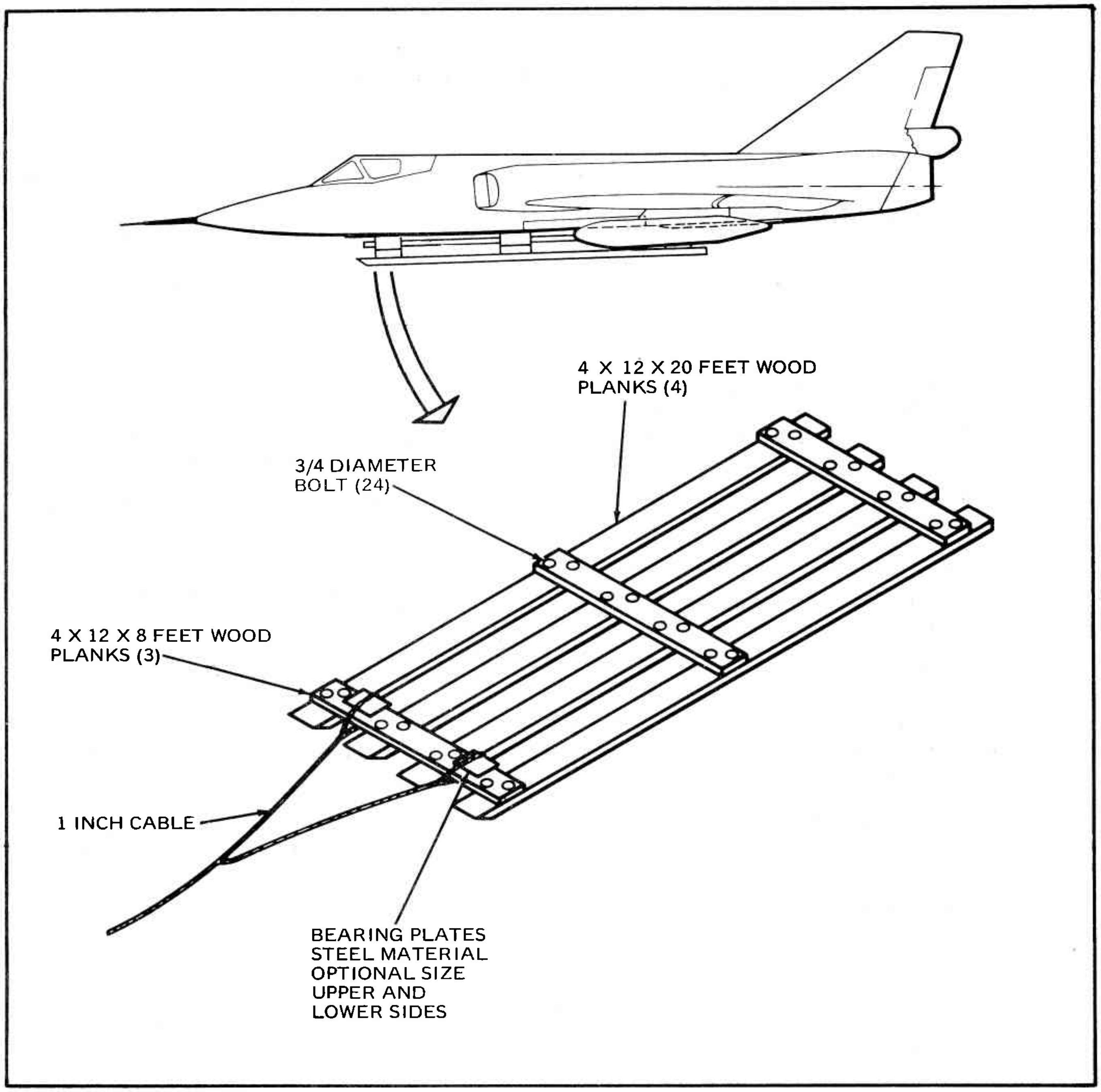
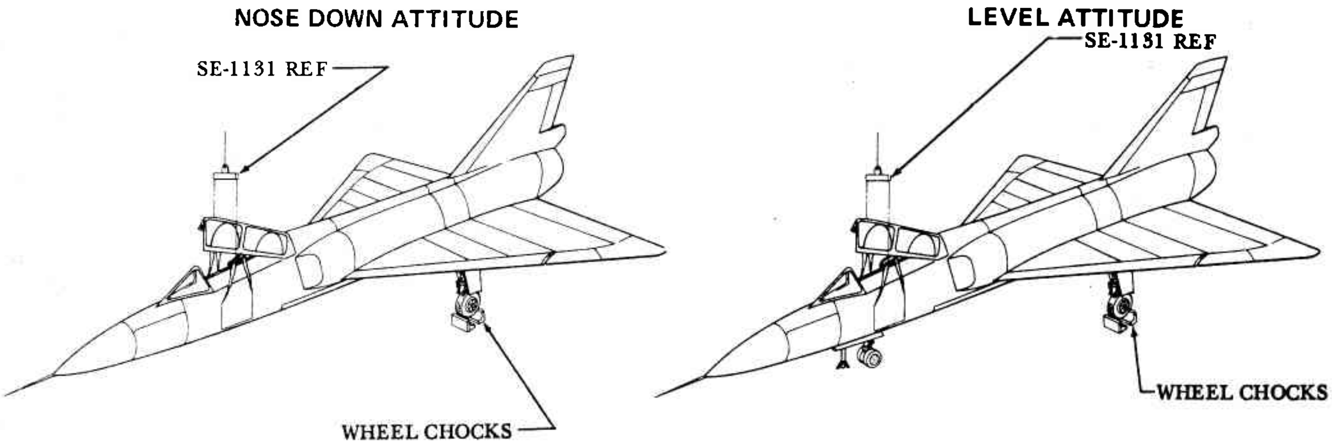


Figure 1-66A. Sled Type Transport Rig-crash Damaged Aircraft Removal



A/F-B-6 Nose Jack Ref used for support after raising aircraft to level attitude

A. DESCRIPTION

The tool is a sling which is made up of spreader bar and two cable assemblies, the ends of which are attached to a 36.0 inch wide belt assembly.

B. FUNCTION

The sling is a tool, used in conjunction with a lift crane, with which the nose of the airplane can be raised to a level attitude in the event of a nose landing gear failure.

C. ADDITIONAL EQUIPMENT REQUIRED

1. Canopy Hold Open Support Assembly Part No. 7140102 or equivalent
2. A lift crane with a minimum capacity of five tons (10,000 pounds)
3. One A/F-B-6 Nose Jack
4. Wheel chocks for main landing gear.

D. INSTALLATION AND OPERATING INSTRUCTIONS:

1. Aircraft Preparation:

   a. Use wheel chocks at main landing gear to prevent fore and aft movement of the aircraft during the lift operation.

   b. Release the main landing gear brakes.

   c. Sling can be used with the canopy open or closed. If open, use upoort assy part no. 7140102.

2. SLING PREPARATION:

   a. Attach 8-96446 (2) cable assemblies to 8-96445 spreader assembly.

   b. Attach lifting eye of 8-96445 spreader assy to crane hook and raise spreader assy (with cables attached) above the cockpit.

   c. Lower spreader assy until the ends of the cable assys can be reached by a man standing on the ground.

   d. Slide 8-96444 belt assy under the airplane and attach one end to each cable assy.

   e. Slowly raise sling until the black stripe on the belt assy matches the centerline of the A/P on the lower surface and the belt lies between the two sta lines indicated on the fuselage. Sta 163.0 and sta 199.0

   f. Raise nose of the A/P to a level attitude and use an A/F-B-6 jack to support the nose while working on the nose landing gear.

Figure 1-66B. Sling Assembly Emergency Nose Lifting

b. Check the under side of the airplane for sharp or rough projections before locating bags under the wings. Cover the top of the lifting bags with felt pads to prevent puncture or damage to the bag while it is being inflated. Place the lifting bag protective tarpaulin covers on the ground under the bags for protection against sharp objects on the ground.

c. Position the inflatable bags to take advantage of the structural strength of the airplane. See figure 1-66 for points that will provide sufficient strength to sustain the weight of the airplane during lifting operations.

d. Connect the blower unit (1730-506-8986) to the inflatable bags and inflate all bags simultaneously.

CAUTION

Do not exceed 2 psi maximum bag pressure. Higher pressure may result in damage to the lifting bag and to the airplane.

e. When the plane has been raised to a sufficient height, install USAF type B-6 jacks (1730-516-2019) with SE-0580-7 jack pads (1730-640-7155), as shown on figure 1-67; then deflate and remove lifting bags as soon as possible.

CAUTION

Airplane jacks must rest on level ground with the jack ram in a vertical position to avoid imposing side loads on the jack and jack pad. If necessary, use steel or wood plates of sufficient thickness and bearing area to distribute high loads over a large area so that the jacks will not sink into the supporting surface or allow the airplane to slip from side to side.

f. If the airplane has been damaged sufficiently to necessitate the removal of major components for packing and shipping, place the airplane on an appropriate cradle, as shown in figure 2-7.

1-212A. REMOVAL-DAMAGED AIRCRAFT FROM RUNWAY.

1-212B. A sled type transport rig, reference figure 1-66A, provides a means to rapidly remove crash damaged aircraft from the runway on a one runway base. Use of this method as outlined below will keep the additional recovery damage to an absolute minimum. This method is effective and can be fabricated locally and made readily available at all bases. Lifting the aircraft to a height to clear sled can be accomplished using pneumatic lifting bags or hydraulic jacks, reference paragraph 1-211. Mooring the aircraft during lifting can be done using heavy vehicles. After cribbing the aircraft with sand bags or cotton matresses, the aircraft can then be moved off the runway using a bed of foam provided by a crash truck. The foam supplies a wetting and smothering agent to minimize the possibility of fire and to reduce sled friction. This method should be used to clear runway for emergency flight operations only. Subsequent moving should be accomplished by conventional methods, reference paragraphs 1-212, 1-214, and 1-216.

1-213. Preparation for Towing.

1-214. If it is possible to manually lower the landing gear, the following procedure may be used for removing the airplane from the crash site:

a. Survey the available routes from the crash site to the repair area for road weight limitations and clearances. See figure 1-1 for aircraft dimensions.

b. Emergency towing should be performed in accordance with procedures outlined in T.O. 1F-106A-2-2-2-2.

c. When towing over earthen roadways, steel landing mats or pierced planking may be used to distribute the weight of the airplane over a larger area.

CAUTION

Defuel airplane before attempting any removal operations. Refer to T.O. 1F-106A-2-5-2-1 for defueling procedure.

1-215. Jacking Airplane.

1-216. Three jack pads, SE-0580-7 (1730-640-7155), provide a bearing surface for the USAF B-6 type jacks (1730-516-2019) used in jacking the airplane. Provisions to receive the jack pads are incorporated in each wing at the spar immediately aft of the main landing gear and at the center line of the fuselage forward of the nose wheel well. Jacking provisions are illustrated on figure 1-67. When jacking at any gear, the landing gear and external tank ground lock safety pins must be installed. The steer-damp unit ground lock pin must also be installed when jacking at the nose gear. Landing gear strut restraining clamps are used in conjunction with the respective nose and/or wing jacks for all jacking operations that do not require gear strut extension. Restraining clamp 8-96175 (1730-631-6467) is for the nose gear and restraining clamps 8-96296 (1730-657-9638) are for the main gears. These clamps prevent the gear struts from extending, thus the main landing gear safety switches are effective and a minimum jacking of the airplane will provide wheel clearance.

NOTE

In special cases, such as electronic weighing, the airplane may be jacked using only the strut pads. The airplane should be raised only enough to clear main wheels from ground, with nose wheel at the minimum height necessary for ground clearance and leveling.

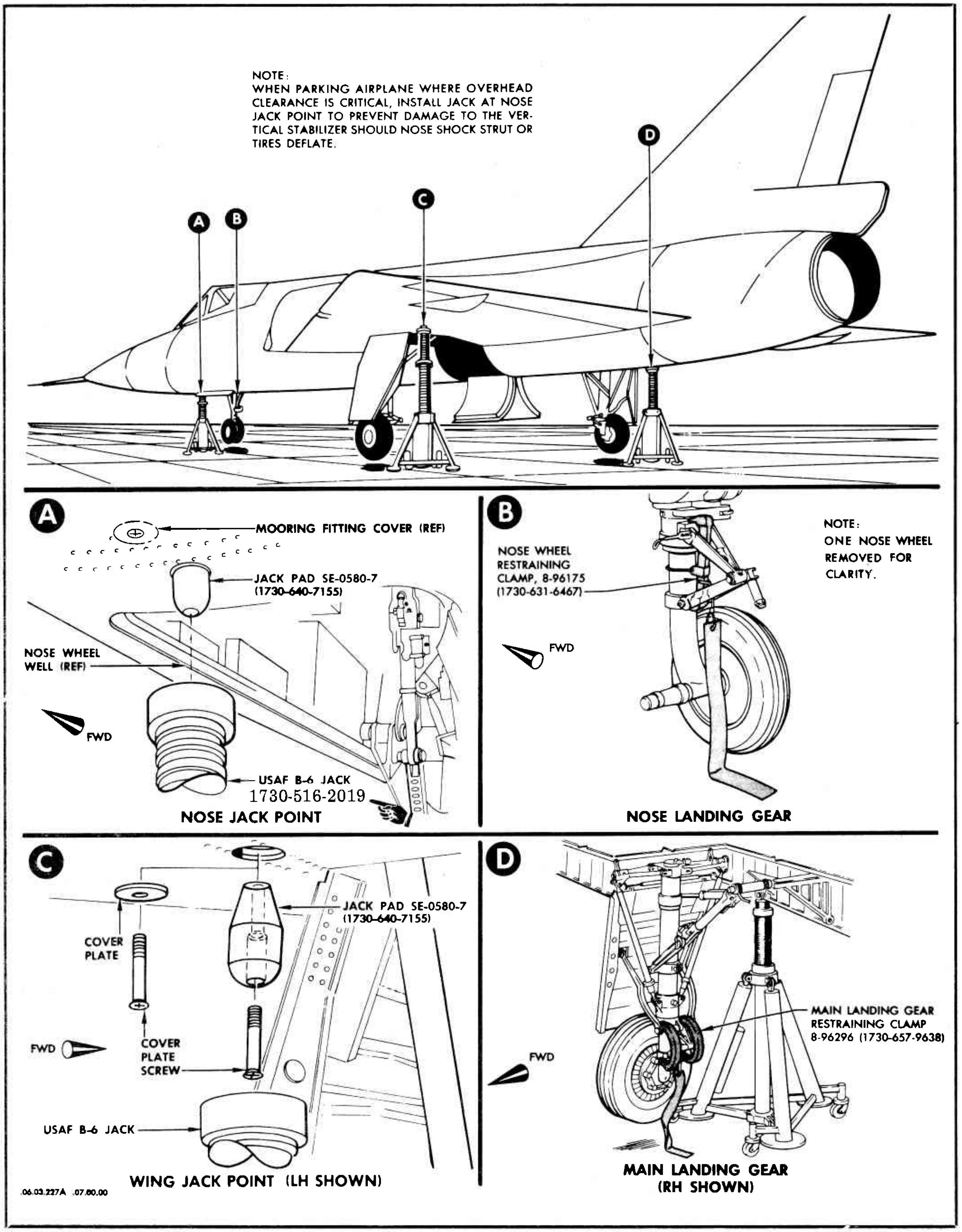


Figure 1-67. Jacking Provisions

All normal precautions, such as removal of unncessary and obstructing equipment and checking of overhead clearances, should be observed prior to performing jacking operations. Jacking operations should be performed in a hangar whenever possible. If operation in a sheltered area or hangar is impractical, jacking may be accomplished outside except during gusty or high wind conditions. The maximum wind velocity in which jacking may be performed safely, in an emergency, is considered to be 30 knots. For specific jacking instructions refer to T.O. 1F-106A-2-2-2-2.

WARNING

Jacking the airplane with the main landing gear struts extended deactuates the main landing gear ground safety switches. This readies the landing-gear-up circuit and other circuits for airborne type operation; exercise care to prevent inadvertent operation.

1-217. Leveling Airplane.

1-218. Leveling lugs are incorporated in the F-106A and F-106B airplanes and are shown on figures 1-68 through 1-70. To level the airplane perform the following operations:

a. Install landing gear and external tank ground safety pins.

b. Jack the airplane; refer to paragraph 1-215.

c. Place a spirit level on the lateral leveling lugs.

d. Place a spirit level on the longitudinal leveling lugs.

e. Adjust jacks to level the airplane laterally and longitudinally.

1-219. Hoisting Airplane.

1-220. Hoisting provisions are incorporated on the airplane to accommodate a four-point hoisting sling as shown in figure 1-65. This sling, 8-96039 (1730-555-4599), permits the hoisting of the airplane in a level attitude, with wings installed or removed. With the wings installed, the sling is attached to fittings on each side of the fuselage and in the upper surface of each wing. With the wings removed the sling is attached to fittings on each side of the fuselage, and to the hoisting adapter which is attached to the wing fittings on each side of the fuselage. Fuselage cradle assembly, 8-96093 (1730-600-3918), is used to support the fuselage when the wings are removed.

NOTE

It is desirable, but not essential, that the airplane be defueled and the engine removed when hoisting the airplane with the wings installed.

1-221. DAMAGE EVALUATION AND INSPECTION CRITERIA.

1-222. Preliminary External Preparation.

1-223. The entire external surface shall be thoroughly examined after an airplane has been damaged. Look for buckled or wrinkled skin, holes, dents, and scratches in the outer surface. Skin wrinkles are a common indication of damage to the internal structural members. Whenever buckles and wrinkles appear, the rows of rivets within the surrounding wrinkled area should be closely inspected. Wrinkles that extend across the substructure indicate failure of attachment. Look for rivets that are sheared, loose, or have tilted heads. Check closely for skin cracks radiating from rivet holes to skin edges. Small skin cracks of this nature frequently occur around door openings. If several discrepancies are found, mark each item with a short strip of masking tape. Using a grease pencil, briefly flag the type of damage by writing on the tape. Do not apply the grease pencil marks directly to the skin surface. When the exterior surface inspection has been completed, enter the airplane and inspect the substructure as indicated in the following paragraphs.

1-224. Access Provisions.

1-225. See figures 1-2 through 1-5 for access to the internal structure for inspection and repair.

1-226. Detailed Visual Inspection of Substructure.

1-227. When a component of the airplane has been damaged, a complete investigation must be made both of the damaged structure and the adjacent or attaching structure to determine if the damaging force has been transmitted over the intervening structure to cause secondary damage. To determine the full extent of the damage, proceed as follows:

a. Remove all dirt and grease so that the surface of each part, rivet, bolt, or weld may be inspected.

b. Look for web or skin wrinkles which indicate distortion. A flashlight and long-handled mirror are useful inspection tools.

c. Look for dents, abrasions, scratches, cracks, tears, and holes. Examine suspected cracks closely with a flashlight and pocket magnifying glass.

d. Look for rivets that are sheared, loose, or have tilted heads. The rivets still may be in position, but may be loose or sheared. An 0.003-inch feeler gage may be used to probe between faying surfaces and under rivet heads to reveal loose or sheared rivets. If bolt heads are found tilted or loose, remove and check for a bent bolt or elongated bolt hole.

e. Look for cracks in paint coatings which may reveal loose rivets or buckled skins or webs.

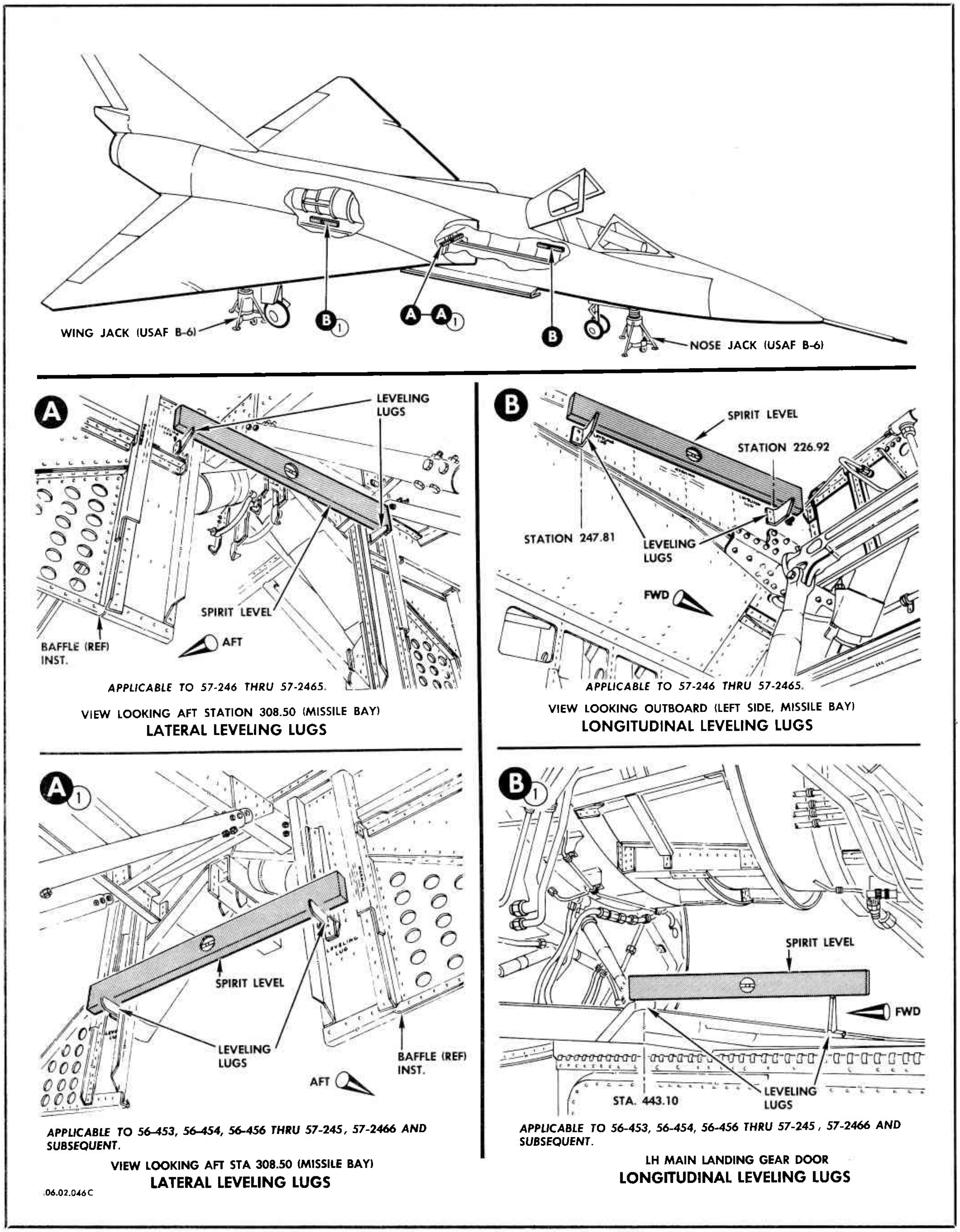


Figure 1-68. Leveling Provisions, F-106A

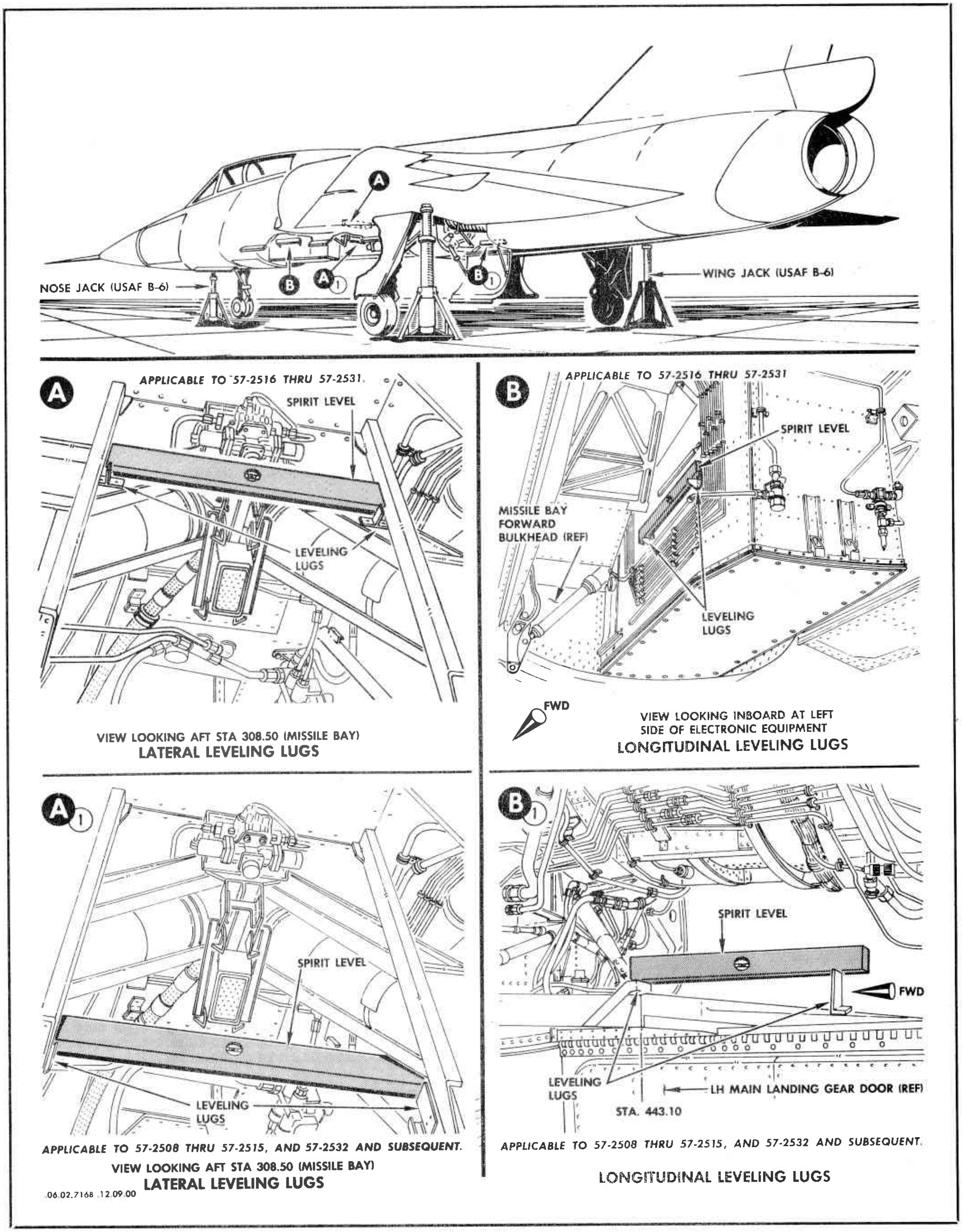
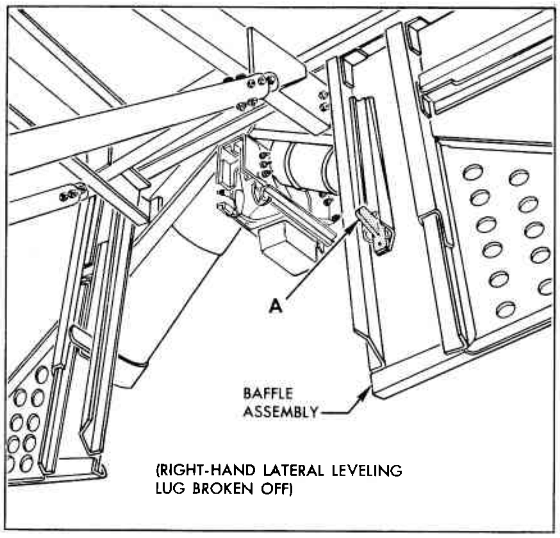
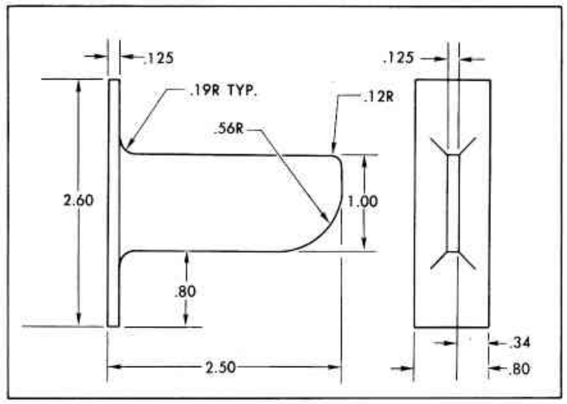
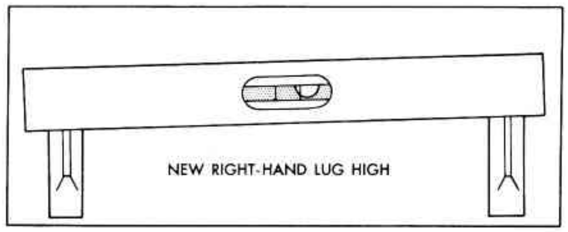


Figure 1-69. Leveling Provisions, F-106B

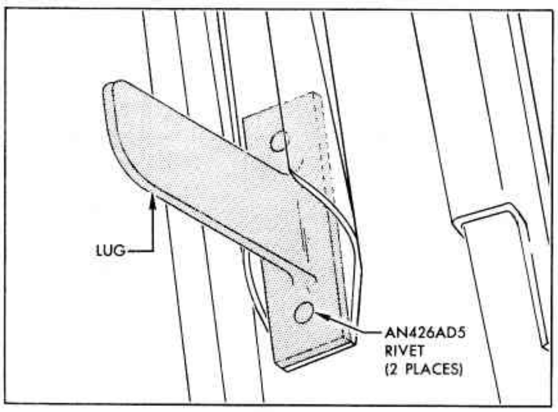




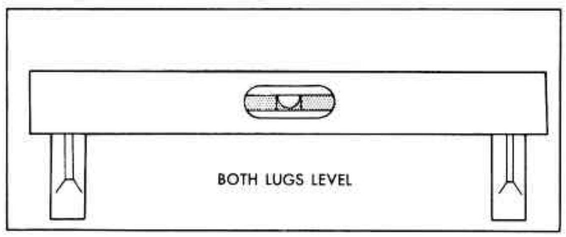
VIEW C



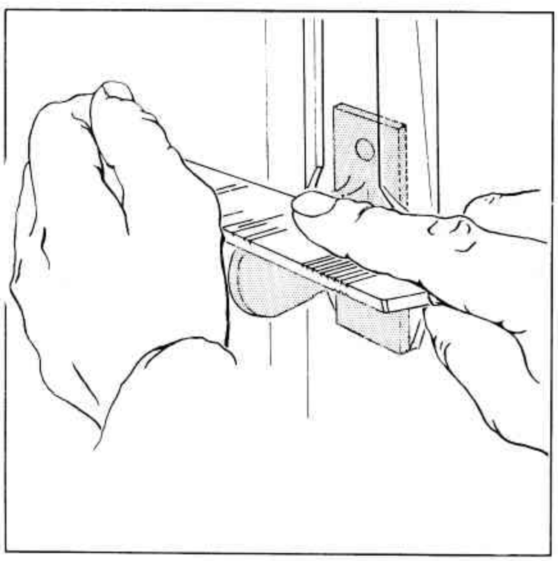
VIEW D



VIEW A



VIEW E



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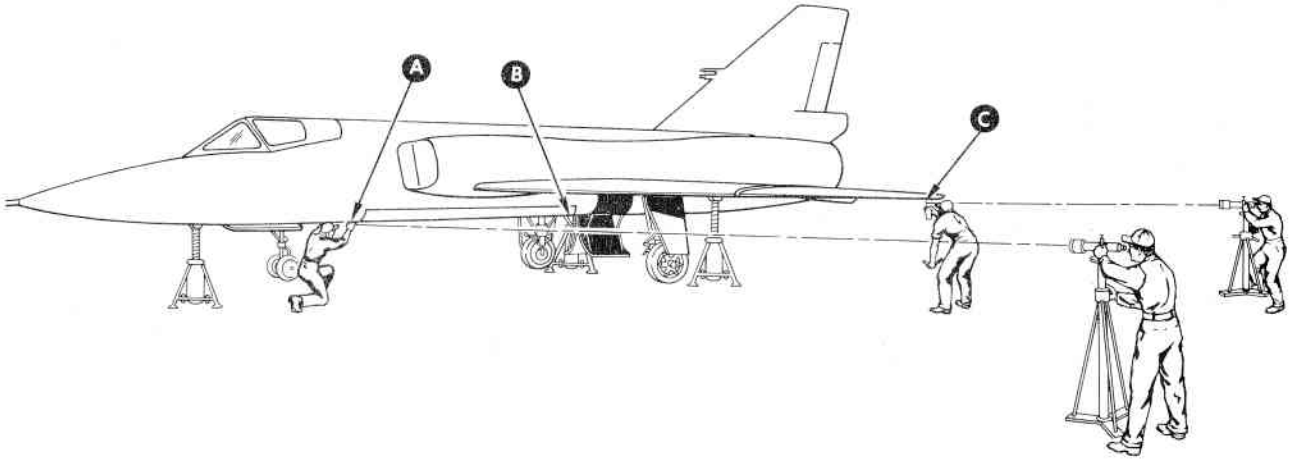
VIEW B

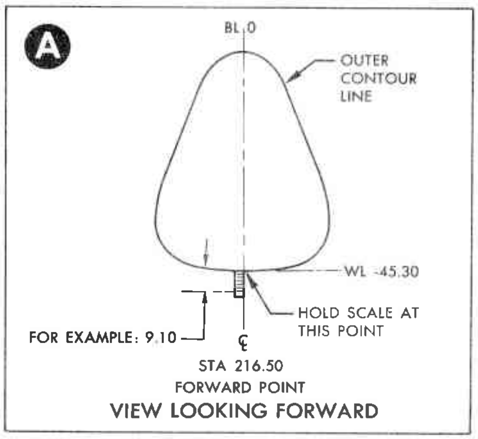
REPLACEMENT PROCEDURE

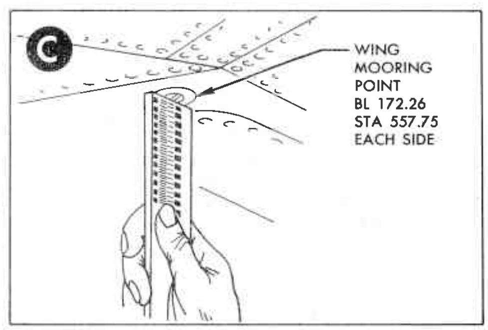
a. Using a No. 20 (0.161) drill, remove two rivets securing broken leveling lug. Remove and discard broken lug.
b. Fabricate new lug to dimensions shown in view C. Make lug from 2024-T4 extrusion, Alcoa Die No. 13630 or equivalent. If extrusion stock is not available, make temporary lug from 0.125-inch thick aluminum sheet. Bend sheet to provide 0.80-inch flange for attachment and trim lug as shown in side view of view C.
c. Install new lug in approximate location of old lug with two AN426AD5 rivets, as shown in view A.
d. Establish lateral level of airplane with sight level or water hose as shown on sheet 2 of this illustration.
e. Place spirit level across lateral lugs as shown in view D.
f. Remove spirit level and file top of high lug as shown in view B. File evenly across top of lug, removing small amounts of metal and checking at periods with level until a level attitude is observed as shown in view E.

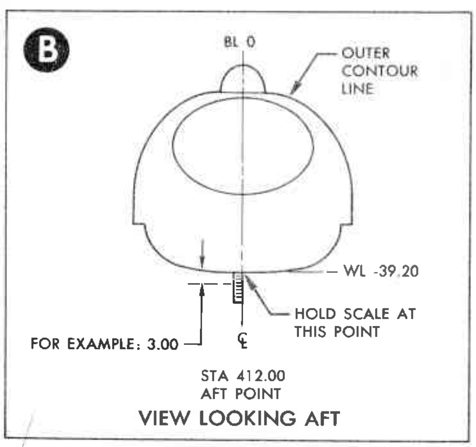
NOTE:
USE REPLACEMENT PROCEDURE GIVEN ON THIS SHEET FOR REPLACEMENT OF LONGITUDINAL LEVELING LUGS.

Figure 1-70. Replacement of Damaged Leveling Lugs (Sheet 1 of 3)









LEVELING PROCEDURE WITH SIGHT LEVEL

a. With airplane suitably jacked, level airplane longitudinally by using a sight level to sight on a scale held alternately at points indicated in details A and B.
b. Adjust nose jack until forward point is 6.10 inches below aft point as indicated on scale.
c. Level airplane laterally, by sighting on scale held alternately at point indicated in detail C and same point under opposite wing.
d. Adjust wing jack until left- and right-hand scale readings are identical.

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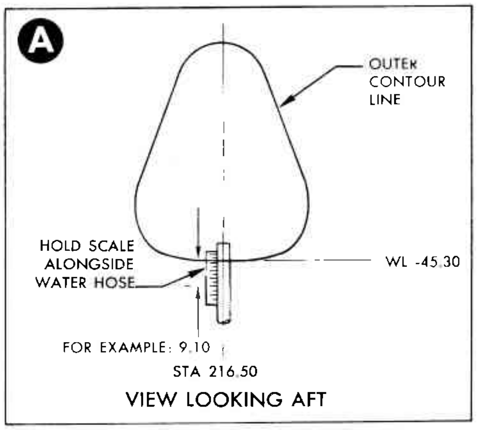
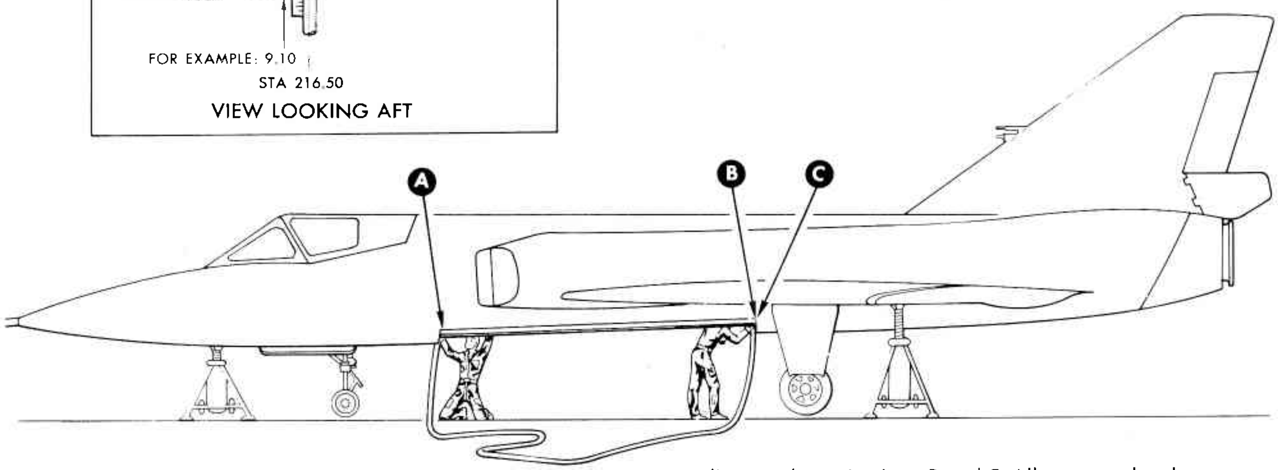
Figure 1-70. Replacement of Damaged Leveling Lugs (Sheet 2 of 3)





LEVELING PROCEDURE WITH WATER LEVEL

An emergency level may be improvised by filling a section of transparent water hose or large diameter transparent plastic electrical sheathing with water. The hose should be 20 feet long or longer. Eliminate all air bubbles from the entire length of the hose by forcing the water through under pressure until bubbles are not visible.

a. With airplane suitably jacked, hold both ends of tubing upright and tape one end of the tubing in a vertical position, as shown in detail A, in such a manner that the water level line viewed through the tubing nearly coincides with the outer surface or skin contour line of the fuselage skin at bottom center line.

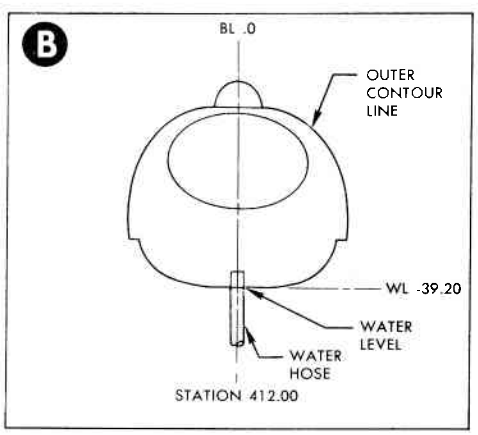
b. Move forward, holding the free end of the hose at about the same height as the opposite end, laying out the center section of the hose on the ground alongside the airplane. Attach the forward end of the tubing inside the missile bay at the forward end.

c. As water will seek its own level, raise or lower the nose of the airplane at the nose jack to establish a longitudinal level condition when the aft hose water level matches with the fuselage skin at bottom center line, and forward water level is 6.10 inches below fuselage skin at bottom center line, as shown in views B and C. Allow water level to stabilize in hose for a few minutes after each adjustment of the hose or change in airplane attitude.

NOTE

TO ELIMINATE UNNECESSARY ADJUSTMENT OF HOSE, WATER LEVELS MAY BE MEASURED WITH SCALES TO ESTABLISH RELATIONSHIP FROM OUTER CONTOUR LINE AT EACH END OF AND BETWEEN ENDS OF MISSILE BAY.

d. The lateral level may be established by repeating the procedure with the vertical ends of the hose attached inside the missile door hinge longerons on each side of the airplane at the same fuselage station.



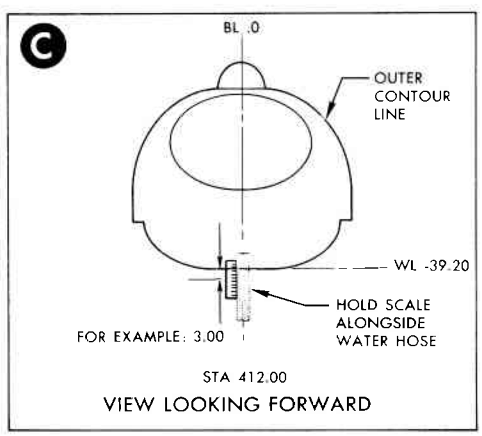


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Figure 1-70. Replacement of Damaged Leveling Lugs (Sheet 3 of 3)

f. Look for stress damage transmitted through structural members to the surrounding area and check all parts for possible misalignment.

g. Operate all functional parts such as hinges and door locks to determine if the damage has caused binding or distortion.

1-228. FLUORESCENT PENETRANT AND X-RAY INSPECTIONS.

1-229. Accomplish fluorescent penetrant and x-ray inspections in accordance with procedures/instructions outlined in T.O. 1F-106A-36 and T.O. 33B-1-1.

Paragraphs 1-230. thru 1-236. Deleted.

All data on page 1-106, including figure 1-71 deleted.

Paragraphs 1-237. thru 1-241. Deleted.

1-242. Description of Damage Causes.

1-243. The causes and degree of damage are variable and may be divided into the following categories:

a. Collision or Gunfire Damage. This type of damage may consist of such minor damage as a small hole or dent, up to very severe damage such as torn, crushed, or burned skin and structural numbers, plus misalignment of the airplane. Inspect the area immediately surrounding the obvious damage to which loads may have been transmitted.

b. Stress Damage. This type of damage usually consists of tilted or loosened rivets, wrinkled skin or webs, and cracked or distorted structural members incurred by resistance to abnormal stresses imposed on the main structural members. Stress damage is usually the result of extra hard landings or violent air maneuvers.

c. Fatigue Damage. This type of damage usually consists of small cracks slowly developed by vibration, oil canning, and repeated application of loads on the structure. Fatigue damage appears more frequently as the operation time on the airplane accumulates. Fatigue cracks appear most frequently on skins in pressurized areas and around door openings, on fittings, and on load-bearing members adjacent to fitting attachments.

d. Corrosion Damage. Corrosion damage may be observed by the presence of white or reddish powdery deposits on the metal accompanied by a roughened, pitted surface. Blistering or scaling of the paint coating when pressed is also evidence of underlying corroded metal. This type of damage may develop in areas where dissimilar metals are near each other, the paint coating is broken, or areas are exposed to moisture. Inspections for corrosion damage should be made at regular intervals and steps taken to arrest further action of any corrosion found before serious damage, which may require replacement of the part, has developed.

1-244. Evaluation of Damage.

1-245. When the full extent of the damage has been resolved, prepare an analysis of the materials, work, time, and cost factors required to complete the repair. To prepare this analysis the following requirements shall be observed:

a. The repaired structure must perform its special function, such as restoring pressurization and fuel-tightness.

b. External structure must conform to the original aerodynamic contour unless otherwise approved by structural engineering authorities.

c. Repair must not restrict the functioning of any moving parts.

d. Precautions against corrosion must be observed, such as priming faying surfaces and insulating dissimilar metals.

e. In cases where cost of repair is prohibitive, the damaged member should be replaced.

1-246. DELETED.

1-247. DELETED.

All data on pages 1-108 thru 1-110, including figure 1-72 deleted.

1-254. IDENTIFICATION OF MATERIALS.

1-255. Steel, aluminum, and magnesium are readily identified by the difference in weight and color of the metal. The various aluminum and magnesium alloys are identified by chemical tests. Chemical test methods for aluminum and magnesium alloys are described in paragraphs 1-256 through 1-259. Methods for identifying titanium alloys are described in paragraph 1-50.

1-256. Chemical Test, Aluminum Alloy Identification.

1-257. The primary airframe structure contains strong heat treatable aluminum alloys, such as 2024, 6061, 7075, and 7178. The nonheat treatable aluminum alloys, such as 1100, 3003, 5052, and 5056, are used in nonstructural components of the airplane.

CAUTION

Nonheat treatable alloys shall not be used for repair or replacement of parts made of heat treatable alloys.

Nonheat treatable and heat treatable alloys may be identified by immersing a sample of the material in a 10 percent solution of sodium hydroxide (caustic soda) for a few minutes. The nonheat treatable alloys will remain brightly unaffected and will not darken, but the heat treatable alloys will turn black due to their copper content. Clad surfaces of heat treatable alloys will remain bright, but may be identified by a dark core area when viewed from the edge. Heat treatable 2024 may be further distinguished from other heat treatable alloys, such as 7075 or 7178, by treating a clean area of the material with a 10 percent solution of cadmium chloride or cadmium sulfate. A dark gray deposit will appear on 7075 or 7178 within two minutes, but will not appear for 15 to 20 minutes on 2024 material.

1-258. Chemical Test, Magnesium Alloy Identification.

1-259. Magnesium alloy parts may be separated from those made of aluminum alloy by using the following spot-test procedure: Remove paint, if present, from an area about the size of a dime. Wipe the area clean with solvent. If methyl ethyl ketone is not available, lacquer thinner may be substituted. Dissolve a few silver nitrate crystals in a small amount of distilled water. Place one or two drops of the silver nitrate solution on the test area. If the material is magnesium, the clear liquid will immediately turn black. If the material is an aluminum alloy there will be no reaction. If the material tested is an aluminum alloy, refinish as directed on figure 1-64. If material is a magnesium alloy:

a. Wash the test area thoroughly with clean water.

b. Remove the black stain with Bon Ami or fine abrasive emery cloth.

c. Swab with a 20 percent solution of chromic acid. Allow to stand two minutes.

d. Rinse with clean water and dry thoroughly.

e. Apply protective coating and paint as directed on figure 1-64.

1-260. FIRE OR HEAT DAMAGE EVALUATION.

1-261. Primer and Paint Discoloration.

1-262. Discoloration or blistering of primer or paint on structural parts indicates fire or heat damage which may have affected the temper of the metals and reduced the strength of the parts below safe levels. To determine fire or heat damage to structural parts, use the sample color plate comparison method as directed in paragraph 1-263. Do not use standard hardness testers to evaluate heat damage to structural parts.

1-263. Hardness Testing of Fire or Heat Damaged Parts.

1-264. Most aluminum alloys used in F-106A and F-106B airplanes are heat-treated and age-hardened under controlled conditions. When these materials are later subjected to temperature exceeding 260°C (500°F), the hardness may increase and indicate acceptable readings if standard hardness testers are used. Upon reheating to normal operating temperatures, however, the strength of the material will fall off rapidly to the point where structural integrity is lost. To evaluate heat damage to "T" condition aluminum alloy structural parts, use the following color comparison method with sample color plates:

a. Make up six sample plates of aluminum alloy sheet stock. The plates should be approximately 1″ x 1″ x 0.032″.

b. Paint all plates with wash primer, Specification MIL-C-8514, and zinc chromate primer, Specification MIL-P-8585, of the color used in the airplane interior. Finish three of the plates with exterior gray acrylic lacquer or polyurethane. Applicable steps of paragraph 1-196 may be used for painting procedures. Air-dry the sample plates for at least 48 hours at normal room temperatures.

c. Place one zinc chromate plate and one gray plate in an oven, and heat to a temperature of 176.7°C (350°F) for 15 minutes. The sample plates will just start to discolor at this temperature. Remove the plates from the oven and mark "350°F."

d. Place one zinc chromate plate and one gray plate in the oven and heat to a temperature of 260°C (500°F) for 15 minutes. Both plates will turn brown at this temperature. Remove plates from the oven and mark "500°F."

e. Place one zinc chromate plate and one gray plate in the oven and heat to a temperature of 315°C (600°F) for 15 minutes. The plates will turn black-green and dark brown, respectively, at this temperature. Remove plates from the oven and mark "600°F."

f. Use the marked sample plates to compare with colors in interior and exterior damaged areas on the airplane.

NOTE

Any heat-damaged member, which indicates by color that 260°C (500°F) has been reached, must be replaced or repaired.

GENERAL COLOR INDICATIONS DUE TO TEMPERATURE

Finish Color (approximate)	Temperature	Action
Zinc Chromate (Aircraft Interior Color)		
Starting to discolor.	148.9°C (300°F) to 204.4°C (400°F)	Continue in service.
Brown.	260.0°C (500°F)	Replace/Repair
Black-green.	315.6°C (600°F)	Replace/Repair
Exterior Gray acrylic lacquer or polyurethane		
Light brown.	148.9°C (300°F) to 204.4°C (400°F)	Continue in service.
Brown.	260.0°C (500°F)	Replace/Repair
Dark brown to black.	315.6°C (600°F)	Replace/Repair

1-265. HARDNESS TESTS.

CAUTION

Do not use the Barcol or Ernst portable hardness testers to evaluate heat damage to structural parts.

1-266. Hardness tests may be made on structural parts with either a Barcol portable hardness tester, Barber-Coleman Co., Rockford, Ill., or an Ernst portable hardness tester, Models RAR and RBR, Newage International, Inc., 235 East 42 Street, New York 17, N.Y., which reads directly for Rockwell A and B scales, to aid in identifying unknown materials. Material hardness is determined by measuring the depth of penetration of a spring-loaded diamond indenter on the Ernst tester and a steel ball indenter on the Barcol tester. The Ernst Model RAR is used for testing hardened steel and hard alloys. The Ernst Model RBR and the Barcol testers are used for checking unhardened steel and most nonferrous metals. The hardness values obtained from clad aluminum alloys will vary considerably because of the coating of pure aluminum on the core alloy. This coating may be removed from areas to be tested and the hardness of the core material determined. The parts tested must be thick enough to prevent bulging on the reverse side. The following procedure describes the use of the Barcol and Ernst testers (see figure 1-73):

a. Remove paint coat from surface of test area. Select a smooth, scratch-free area on an accessible surface of the part.

b. Check calibration of tester on test plate provided with instrument. If the reading varies more than one point from the nominal reading marked on the test plate, the instrument should be recalibrated in accordance with manufacturer's instructions.

c. The instrument must be held so that the penetrator is at a right angle to the surface being tested.

1-267. Structural parts may be accurately tested for hardness with a Riehle portable hardness tester as follows (see figure 1-74):

CAUTION

Do not use the Riehle portable hardness tester to evaluate heat damage to structural parts.

a. Place the part to be tested on a work bench or table.

b. Remove the paint and/or primer coating from surface of test area. Select a smooth, scratch-free area or an accessible surface of the part.

c. Remove the Riehle tester from the carrying case provided and calibrate according to the manufacturer's instructions.

d. Select proper anvil to suit the shape of the material being tested.

e. Select penetrator to be used. Use a diamond penetrator for hardened or heat-treated steel. Use a ball penetrator for nonferrous material, or materials harder than C-20 on the Rockwell scale.

f. Make sure loading screw is backed off sufficiently so that penetrator does not project beyond upper clamp.

g. Clamp material being tested firmly between upper clamp and anvil by means of adjusting knob on lower clamp.

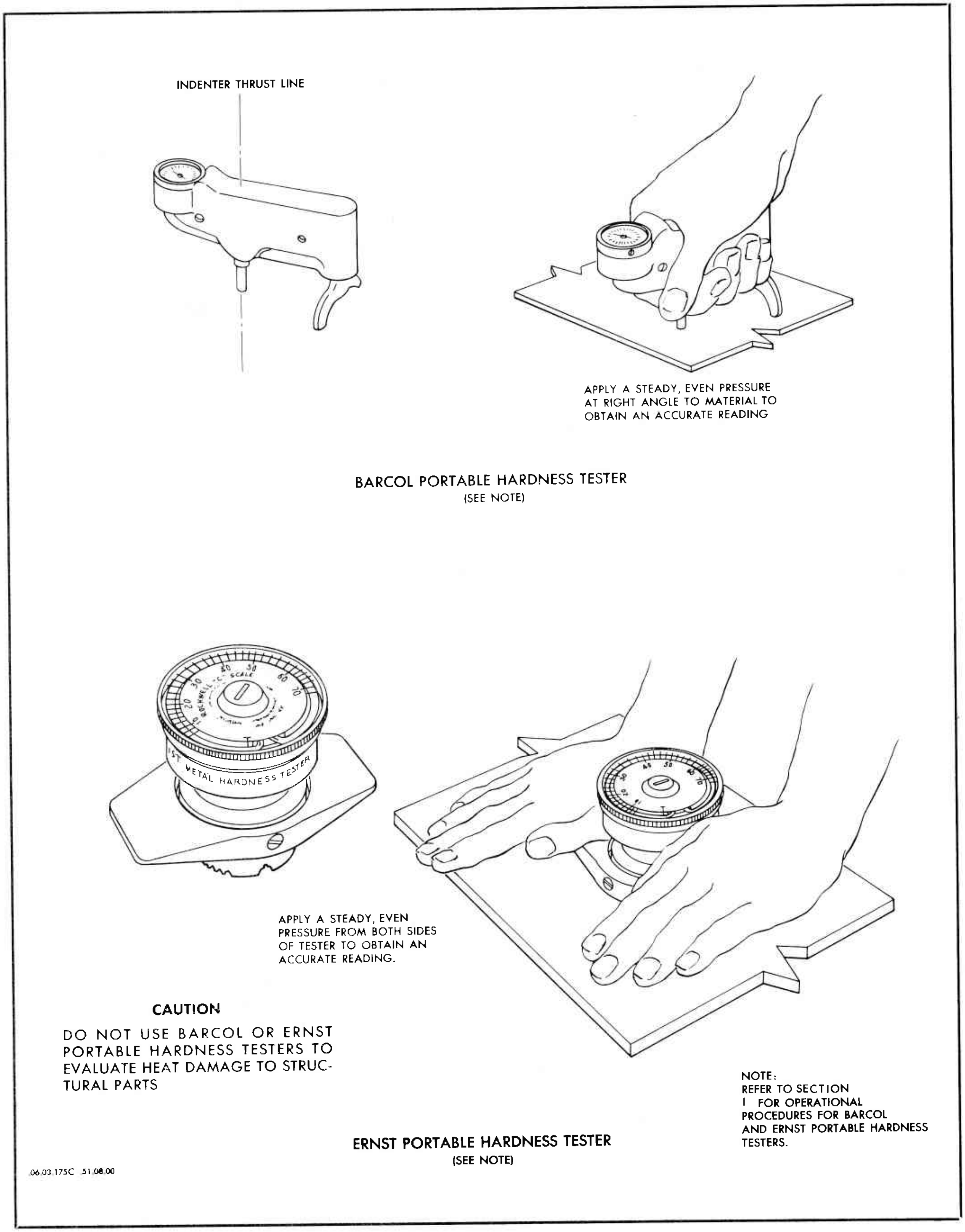
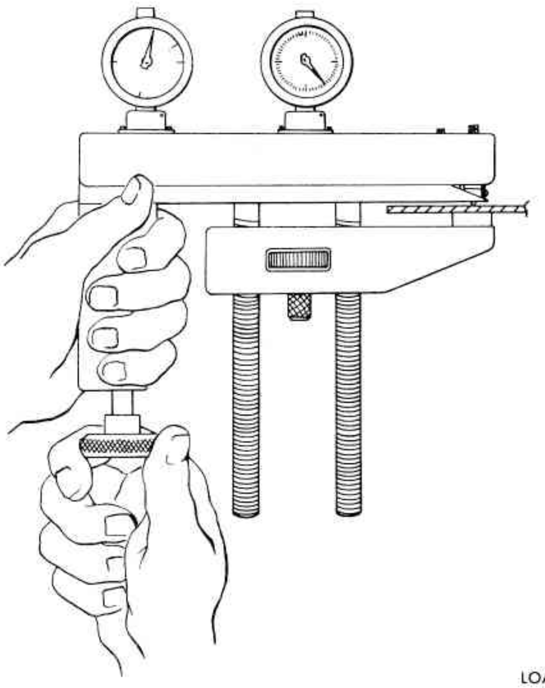
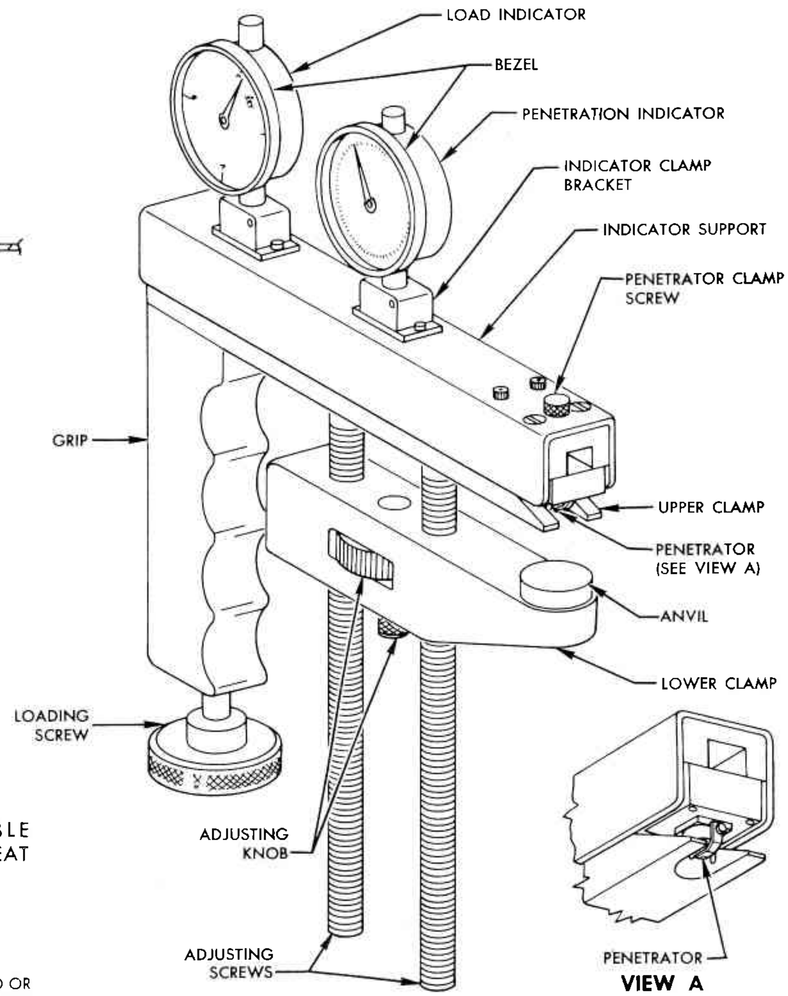


Figure 1-73. Barcol and Ernst Portable Hardness Testers





OPERATING PROCEDURE

CAUTION

DO NOT USE RIEHLE PORTABLE HARDNESS TESTER TO EVALUATE HEAT DAMAGE TO STRUCTURAL PARTS

a. Select penetrator to be used.

NOTE

USE A DIAMOND PENETRATOR FOR HARDENED OR HEAT TREATED STEEL. USE A BALL PENETRATOR FOR NONFERROUS OR MATERIALS SOFTER THAN A ROCKWELL READING OF C-20.

b. Select proper anvil to suit shape of part to be tested.
c. Turn loading screw counterclockwise until penetrator does not project beyond upper clamp.
d. Place material to be tested between upper clamp and anvil.

NOTE

PLACE THE MATERIAL TO BE TESTED BETWEEN UPPER CLAMP AND ANVIL SO THAT TEST MAY BE MADE AS FAR FROM THE EDGE AS POSSIBLE.

e. Clamp material to be tested between upper clamp and anvil by turning adjusting knob on lower clamp clockwise.

CAUTION

DO NOT FORCE

f. Set load indicator dial for zero setting by rotating bezel until pointer is over small black dot.
g. Apply a 10 kg minor load by turning loading screw clockwise until pointer of load indicator dial is over SET.

CAUTION

INSTRUMENT MUST BE HELD STEADY TO OBTAIN AN ACCURATE READING

h. Rotate bezel on penetration indicator dial until pointer is on "0" of the black scale.
i. Apply major load by turning screw clockwise.

NOTE

NORMAL SETTINGS FOR MAJOR LOADS AS INDICATED ON LOAD INDICATOR DIAL ARE AS FOLLOWS: "C" SCALE (150 kg) OR "A" SCALE (60 kg) FOR DIAMOND PENETRATOR AND "B" SCALE (100 kg) FOR BALL PENETRATOR.

j. Remove major load and reduce to minor load by turning loading screw counterclockwise until pointer of load indicator dial moves back to SET.
k. Hardness of material being tested will be indicated on penetration indicator dial.

NOTE

READ BLACK FIGURES WHEN USING DIAMOND PENETRATOR. READ RED FIGURES WHEN USING BALL PENETRATOR.

l. Remove instrument by turning adjusting knob counterclockwise.

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Figure 1-74. Riehle Portable Hardness Tester

h. Press the material to be tested through the throat of the Riehle tester.

i. Check zero setting of load indicator. Rotate bezel (setting ring) to bring pointer over small black dot for zero load.

j. Apply a 10 kg minor load by turning loading screw clockwise until pointer of load indicator dial is directly over the word SET.

CAUTION

To obtain an accurate reading, the instrument must be held steady to avoid twisting motion of penetrator on work.

k. Check zero setting of penetration indicator. Rotate bezel to bring pointer to zero on the black scale.

l. Apply major load, normally "C" scale (150 kg) or "A" scale (60 kg), for diamond penetrator, or "B" scale (100 kg) for ball penetrator.

m. Take off major load by turning loading screw counterclockwise until pointer on load indicator returns to SET.

n. Hardness is indicated on penetration indicator. Read black figures when using diamond penetrator and red figures when using ball penetrator.

o. Release load completely by backing off loading screw.

p. Turn adjusting knob to left to loosen lower clamp.

1-268. ALIGNMENT CHECK.

1-269. An alignment check of various components of the airplane may be made to determine the cause of reported unsatisfactory operation, to check for conformance to original dimensions after major repairs, as an aid in evaluation of damage, or to position components correctly during repair. The airplane alignment may be checked by direct measurement. This is done with the airplane in the "flight-line" condition with a full load of fuel and oil and with the shock struts inflated. An alignment check by direct measurements, as shown in figures 1-75 and 1-76, is an approximate method which permits a rapid appraisal after an accident or after a major repair. For more precise measurements, a jig transit should be used as shown in figure 1-77. Because of the critical structure and close tolerance built into the F-106A and F-106B airplanes, special optical tooling and equipment are required to check alignment of components and to effect satisfactory major structural repairs.

1-270. Landing Gear Alignment.

1-271. The alignment check for the nose and main landing gear shall consist of an operational check as specified in T.O. 1F-106A-2-8-2-1. During this check, all gear assemblies shall be checked for proper operation throughout a complete extension and retraction cycle. If the operational check is not satisfactory or all clearances are not as specified in T.O. 1F-106A-2-8-2-1, carefully inspect the landing gear assembly and its attachment to the structure for damage, and for further disposition consult an aeronautical structures engineer.

1-272. Wing and Fin Alignment Check Procedure.

1-273. By using the following procedure, alignment of the wing and fin may be checked in the field (see figure 1-77):

a. Place the airplane on jacks, and level airplane by means of the leveling lugs in the armament bay. Refer to paragraph 1-215 for jacking provisions and to paragraph 1-217 for leveling provisions.

b. Position an instrument stand at a point approximately in line with the centerline of the airplane and about 25 feet aft of the tail cone.

c. Mount a sight level on the instrument stand. Crank up or adjust height of stand to position the sight level slightly below the lower surface of the wing. Carefully level the instrument to maintain a level line of sight when the scope is rotated to left or right. Refer to paragraph 1-280 for the procedure of leveling the sight level.

d. Raise and hold up the left elevon to provide access through the lower openings in the elevon leading edge at the hinge fittings.

e. Position and hold an accurate scale (graduated in hundredths of an inch or less) against the shoulder of the flanged bushing in the rear spar elevon hinge fittings at BL 175.44. Do not hold end of scale against the tapered cone in the hinge assembly. Refer to paragraph 1-280 for technique of holding scale in vertical position. See figure 2-2 for BL locations.

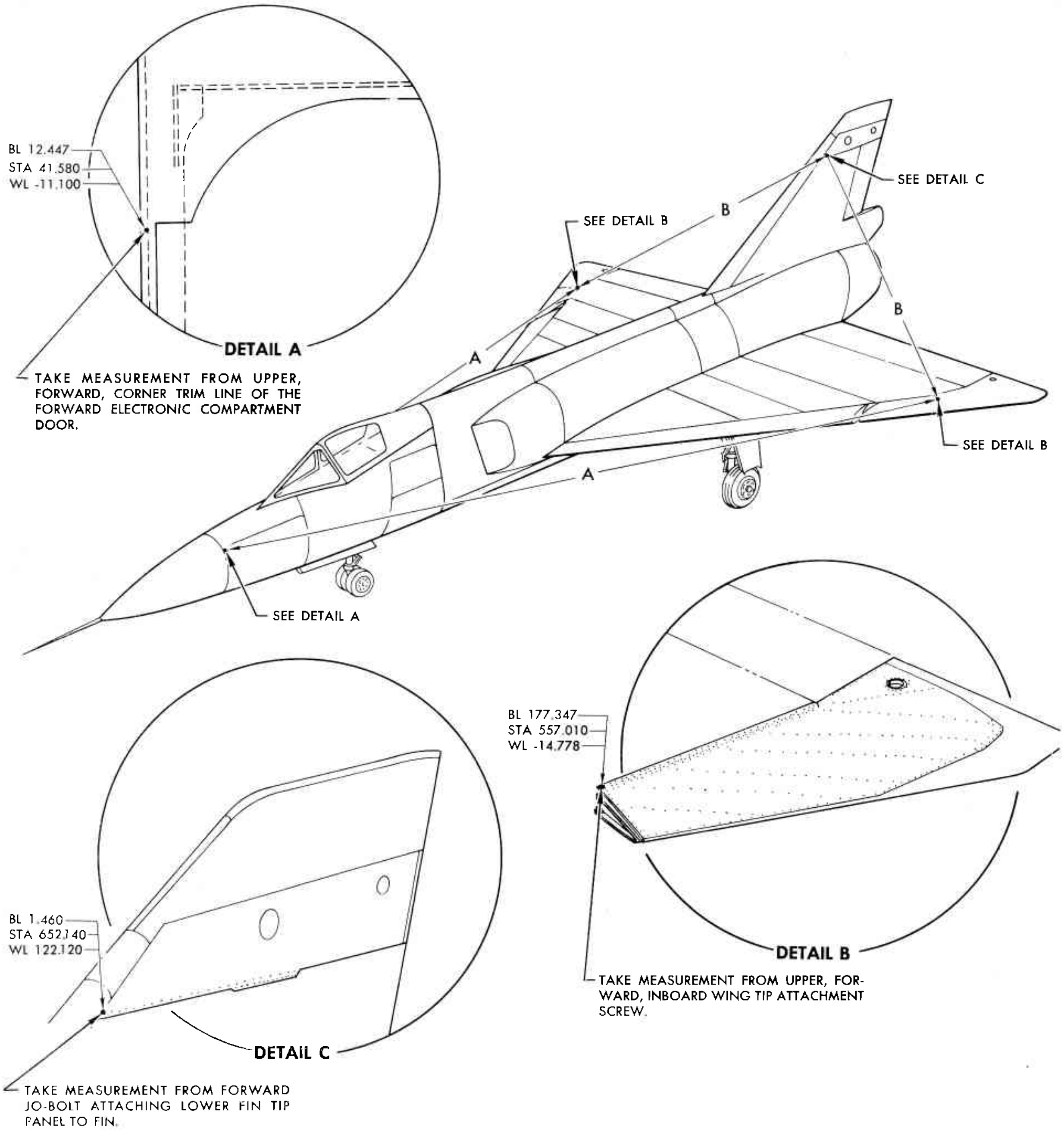
f. Aim and focus instrument on scale. Record scale graduation viewed at intersection by the scope horizontal cross hair. This location is the fix or reference point on the scale.

g. Move to opposite wing, raise the right elevon, and repeat step "e."

h. Swing scope; aim and focus on scale. Record scale graduation viewed at intersection by the scope horizontal cross hair. If this reading is identical with the reading recorded in step "f," the wing is in lateral level. If the readings differ, adjust either wing jack until the same reading is observed at BL 175.44 as called out in steps "e" and "g."

i. Verify alignment of wings by checking all remaining (7) elevon hinge fittings inboard of BL 175.44 on both wing rear spars as follows:

1. Check reading at left-hand elevon hinge fitting at BL 130.44 and right-hand elevon hinge fitting at BL 130.44; the readings should agree.



NOTES:
1. USE STEEL TAPE MEASURE FOR MEASURING BETWEEN WING TIP POINT AND POINTS ON THE FIN AND NOSE. TAKE MEASUREMENT WITH AIRPLANE RESTING ON LANDING GEAR AND IN SHADED AREA PROTECTED FROM WIND GUSTS.
2. THE ABOVE MEASUREMENTS ARE FOR A FIELD ALIGNMENT CHECK. IF THE TOLERANCES ARE EXCEEDED, USE THE WING AND FIN ALIGNMENT CHECK PROCEDURE IN THIS SECTION FOR A MORE PRECISE OPTICAL MEASURING SYSTEM.

REFERENCE LETTER	CONFIGURATION	NOMINAL DIMENSION	TOLERANCE EACH SIDE	TOLERANCE BETWEEN SIDES
A	FUSELAGE NOSE TO WING TIP	45′ 1¼″	±1″	±½″
B	WING TIP TO FIN TIP	20′ 1½″	±½″	±½″

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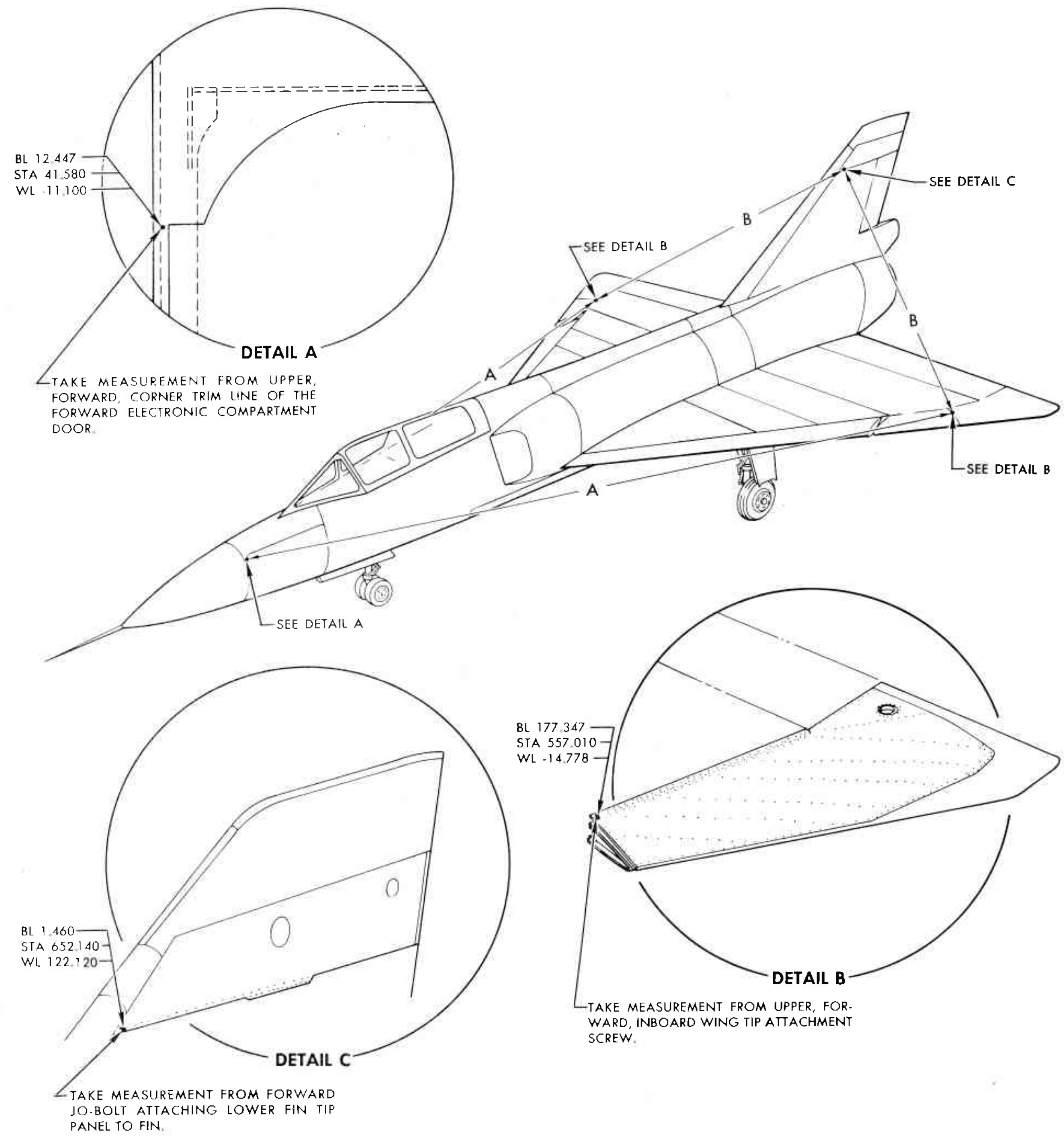
Figure 1-75. Rigging Check Diagram — F-106A



NOTES:
1. USE STEEL TAPE MEASURE FOR MEASURING BETWEEN WING TIP POINT AND POINTS ON THE FIN AND NOSE. TAKE MEASUREMENT WITH AIRPLANE RESTING ON LANDING GEAR AND IN SHADED AREA PROTECTED FROM WIND GUSTS.
2. THE ABOVE MEASUREMENTS ARE FOR A FIELD ALIGNMENT CHECK. IF THE TOLERANCES ARE EXCEEDED, USE THE WING AND FIN ALIGNMENT CHECK PROCEDURE IN THIS SECTION FOR A MORE PRECISE OPTICAL MEASURING SYSTEM.

REFERENCE LETTER	CONFIGURATION	NOMINAL DIMENSION	TOLERANCE EACH SIDE	TOLERANCE BETWEEN SIDES
A	FUSELAGE NOSE TO WING TIP	45′ 1 ¼″	±1″	± ½″
B	WING TIP TO FIN TIP	20′ 1 ½″	± ½″	± ½″

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Figure 1-76. Rigging Check Diagram — F-106B

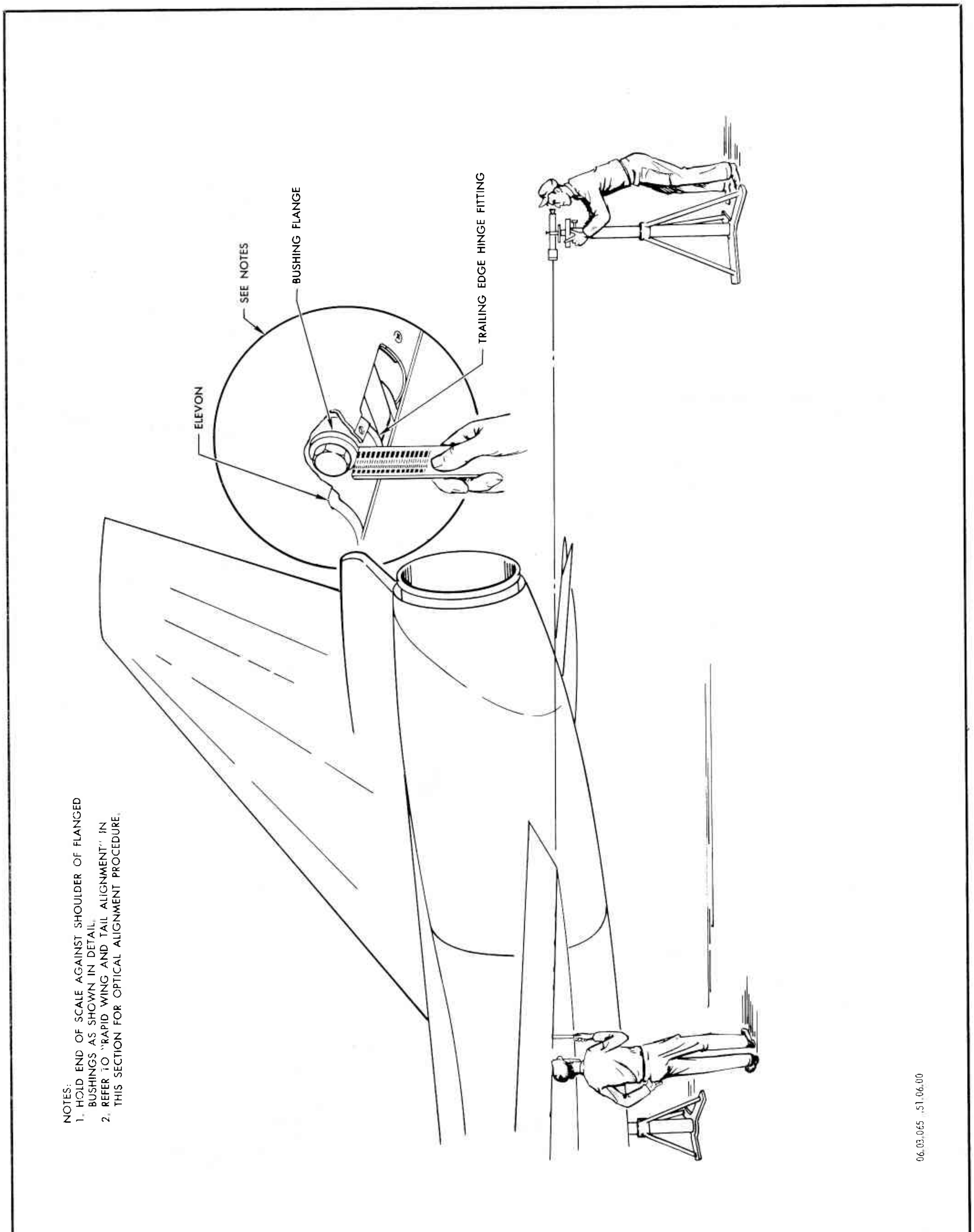


Figure 1-77. Wing and Fin Alignment Method

2. Repeat measurements at each elevon hinge fitting as indicated in step 1. If reading between like BL fittings indicate a misalignment in excess of ± 0.20 inch, check the wing structure for damage and refer to structural engineering authorities.

j. Remove sight level and mount a jig transit on the instrument stand. Carefully level the jig transit by using the procedure outlined in steps "l" through "q" in paragraph 1-293.

k. Adjust rudder to streamline position by aligning rudder trailing edge as shown in T.O. 1F-106A-2-7-2-1.

l. Loosen transit vertical and horizontal axis clamps, and swing the instrument around and up to aim on the centerline of the rudder trailing edge at the lowest point visible. Tighten the vertical axis clamp on the transit snug enough to prevent horizontal motion.

m. Elevate transit line-of-sight to tip of fin. Centerline of fin tip trailing edge should be in line with centerline point observed in step "l." Sweep transit up and down along fin and rudder trailing edge and measure any deviations with a scale held horizontally across trailing edge. If the misalignment exceeds ± 0.25 inch at top of fin, check for damage to fin structure and refer to structural engineering authorities.

1-274. ARMAMENT BORESIGHT PLATE ALIGNMENT.

1-275. Boresight mounting plates are provided for alignment of missile launchers as described in T.O. 1F-106A-2-12. These boresight plates are located inside the missile bay at BL 8.15 on the right-hand side. See figure 1-78 for an illustration of the boresight plate locations and alignment data. The forward plates are secured to the aft side of the bulkhead at station 216.50 and the aft plates are secured to the forward side of the bulkhead at station 412.00. The boresight mounting plates are adjustable to provide for readjustment in the field. They are jig located at time of manufacture and fixed in place with two MS 20600B4-4 blind rivets. If the boresight is readjusted in the field, the blind rivets must be drilled out to free the mounting plate and two new holes must be drilled for two AN173-5A close tolerance bolts to permanently lock the mounting in place. Broken or damaged boresight plates may be realigned after replacement by the following procedure:

a. Place the airplane on jacks, and level airplane by means of the leveling lugs in the armament bay. Refer to paragraphs 1-215 and 1-217 for procedures.

b. Drop two plumb lines, with fairly heavy bobs or weights attached, from the lower surface of the fuselage at flush screws installed through skin. These screws are located at stations 102.00 and 431.00 and BL 8.15 on the right-hand side. Secure plumb line by replacing existing screws with like screws which have been drilled through recess in head, creating a hollow shank for plumb line.

c. Position an instrument stand, equipped with cross slide or lateral adjuster, at a point approximately in line with the plumb lines and about 25 feet forward of the nose of the airplane. Refer to paragraph 1-292 for tools and accessories required.

d. Carefully remove the jig transit from its carrying case and mount it on the instrument stand.

e. Rough level the instrument with the bullseye level in transit frame by adjusting the four leveling screws at the bottom plate of the instrument.

f. Set optical micrometer at "0" reading and focus the eyepiece until the telescope cross hairs are seen clearly. Adjust height of stand to allow sighting under airplane.

g. With near plumb line held to one side, aim the telescope at the far plumb line (station 431.00) and adjust the telescope focusing knob until the line appears sharp and clear. See figure 1-79 for an illustration of the telescope focusing procedure.

h. Tighten clamp on the instrument vertical axis snug enough to prevent horizontal movement and adjust the horizontal motion tangent screw until the telescope vertical cross hair is superimposed on the estimated center of the plumb line.

i. Readjust the telescope focusing knob until the near plumb line (station 102.00) is clearly visible.

j. Adjust the instrument stand cross slide or lateral adjuster to move the transit left or right until the vertical cross hair is superimposed on the estimated center of the near plumb line. The transit telescope line-of-sight will be aligned with both plumb lines when estimated centers of the lines are visible, each in turn, without moving the telescope.

NOTE

Do not readjust the horizontal motion tangent screw during this step.

k. Repeat steps "g" through "j" until telescope vertical cross hair remains superimposed on estimated centers when focused to each plumb line.

l. At forward boresight mounting plate inside missile bay at station 216.50, move upper mounting plate until center of 5/16-inch bolt hole in upper plate is aligned with two 0.250-inch diameter tooling holes in web of bulkhead at WL −32.00 and BL 20.00 (one each side, left-hand and right-hand), as determined by an accurate straight edge. Tighten adjustment bolts.

m. Suspend third plumb line with heavy weight attached from a 5/16-inch bolt installed in upper mounting plate.

n. Sight through transit on third plumb line and move slightly loosened mounting plate to left or right until estimated center of plumb lines established in steps "g" through "k" is in line with transit vertical cross hair. Lock plate in place with two AN3 bolts in mounting plate assembly.

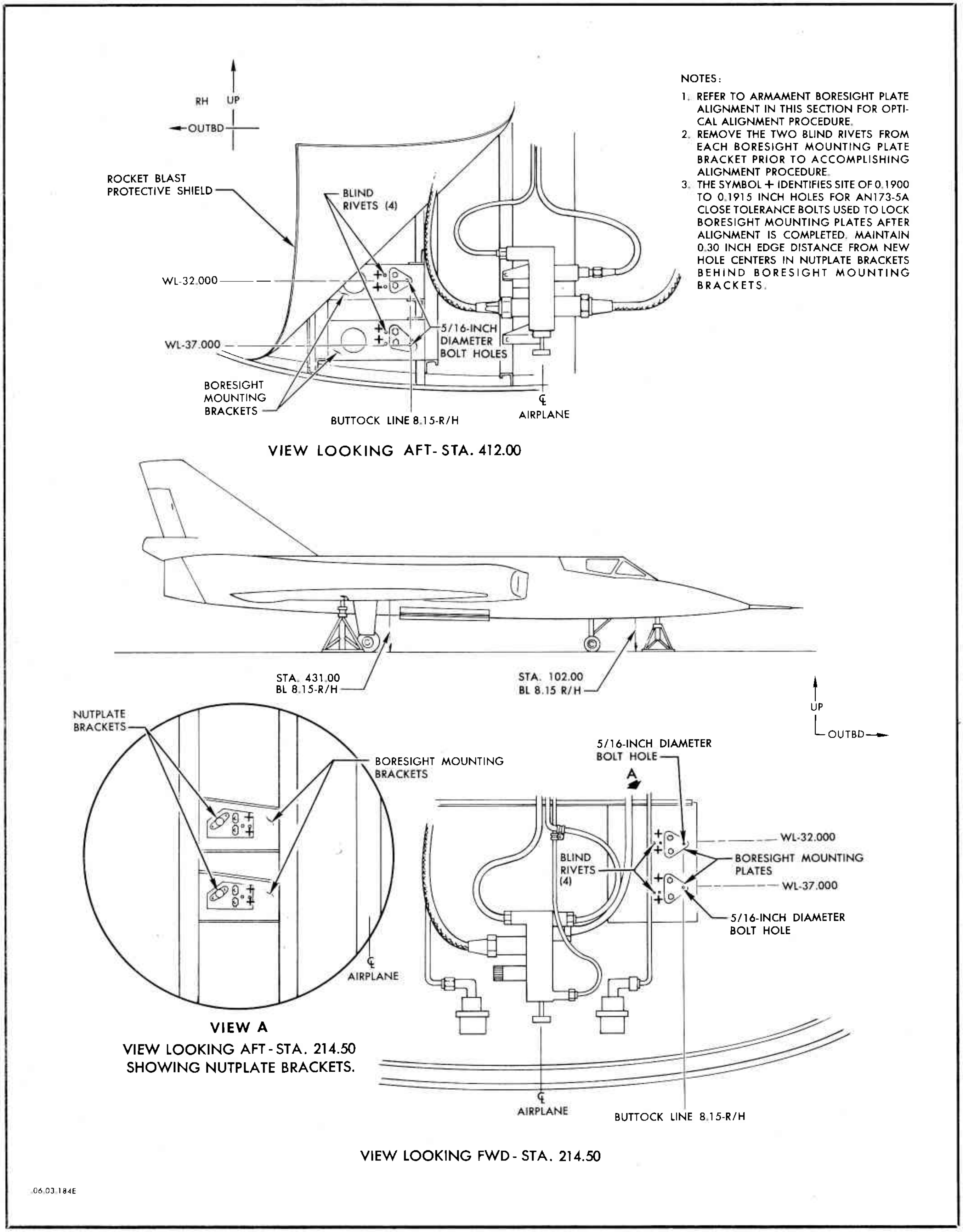
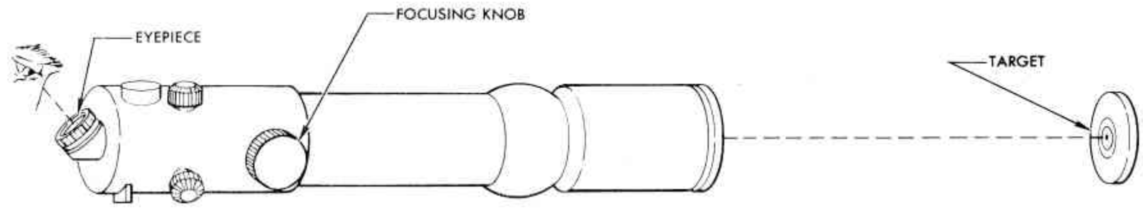
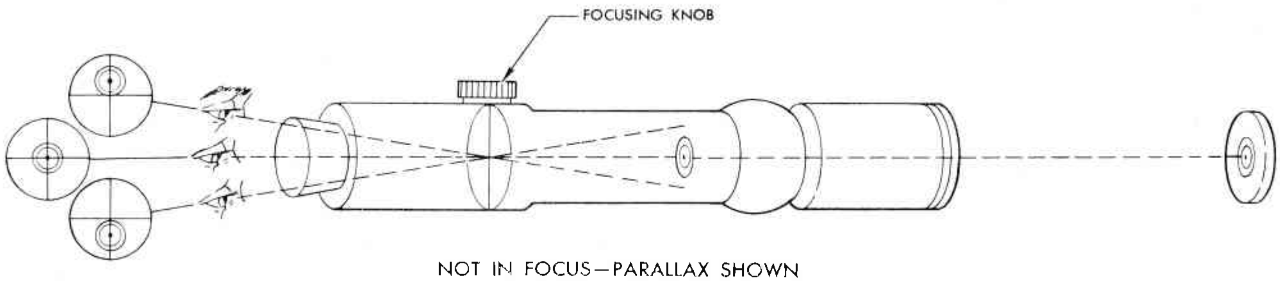


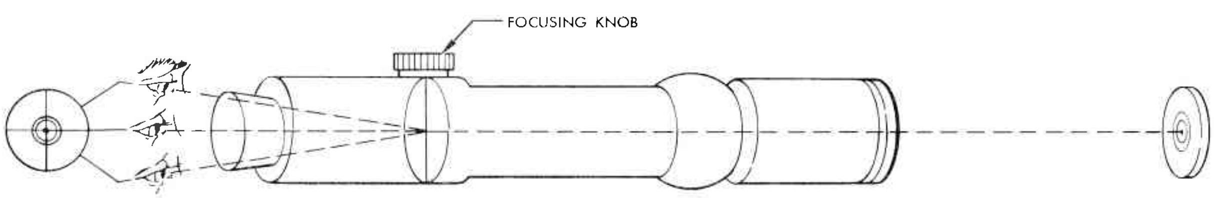
Figure 1-78. Boresight Mounting and Plumbline Locations



A. Focus eyepiece on telescope cross hairs by turning eyepiece until cross hairs appear sharp and clear.
B. Focus telescope on target by turning telescope focusing knob until target image appears to be superimposed on the telescope cross hairs.
C. Adjust focusing knob slightly, until target image remains stationary when head is moved.



NOT IN FOCUS—PARALLAX SHOWN



FOCUSED—PARALLAX ELIMINATED

NOTE:
LOOK THROUGH TELESCOPE WITH BOTH EYES OPEN AND ASSUME A RELAXED POSITION AS WHEN VIEWING A DISTANT OBJECT. IF CORRECTIVE LENS GLASSES ARE NORMALLY WORN BY THE OPERATOR THEY SHOULD NOT BE REMOVED.

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Figure 1-79. Telescope Focusing Procedure

o. Drill two 0.1900 to 0.1915-inch holes for two AN173-5A close tolerance bolts in each plate near old blind rivet holes. Maintain 0.30-inch edge distance from hole centers in nutplate brackets behind boresight mounting brackets.

p. Install two AN173-5A close tolerance bolts in plate. Permanently lock in place with two AN365-1032 nuts.

q. Align lower mounting plate at station 216.50 in same manner as in preceding steps, except locate 5/16-inch bolt hole in lower plate at a distance of 5.00 inches below 5/16-inch bolt hole in upper plate. Swing plumb line from lower plate to align and lock as in steps "m" through "p."

r. Align aft mounting plates at station 412.00 in same manner as in preceding steps, except locate 5/16-inch bolt hole in upper plate at a distance of 7.00 inches below two 0.25-inch tooling holes in bulkhead web at BL 10.00 and WL −25.00, left and right sides. Locate lower mounting plate at a distance of 5.00 inches below upper plate. Align and lock both plates as in steps "m" through "q."

1-276. Optical Tooling.

1-277. Optical tooling applies the principles of surveying and precision optics to the task of erecting or rechecking assembly jigs and fixtures required in aircraft construction. Aircraft component alignment and reference points are established by using line-of-sight instruments. Line-of-sight instruments are also used to establish true lines and planes in space, to determine precise right angles, and to measure horizontal and vertical distances. Heavy mechanical tools, used in previous methods, are lightened and simplified by use of the optical tooling system, since lines-of-sight are established as reference lines instead of mechanical devices. Use of optical instruments also eliminates some errors that are commonly found in the larger mechanical measuring devices. Errors in mechanical measurements may be caused by wear, dirt, corrosion, obstructions, distance, sag of long scales or bars, inaccessibility of points to be located, and distortion from unequal coefficients of expansion. The following four instruments are the basic optical tools used in erecting F-106A and F-106B jigs and fixtures:

1. The sight level.
2. The jig transit.
3. The alignment telescope.
4. The optical square.

These four optical instruments and complementary accessories cut jig and fixture construction time by as much as 20 percent and, at the same time, attain equal if not greater accuracy than previous methods. The instruments and accessories listed in the procedures that follow are typical of the optical tools used in the construction of the F-106A and the F-106B. Other instruments and accessories may be used in these procedures by making allowances for the variations in design of the instruments. A glossary of optical tooling terms is listed in paragraph 1-333.

1-278. THE SIGHT LEVEL.

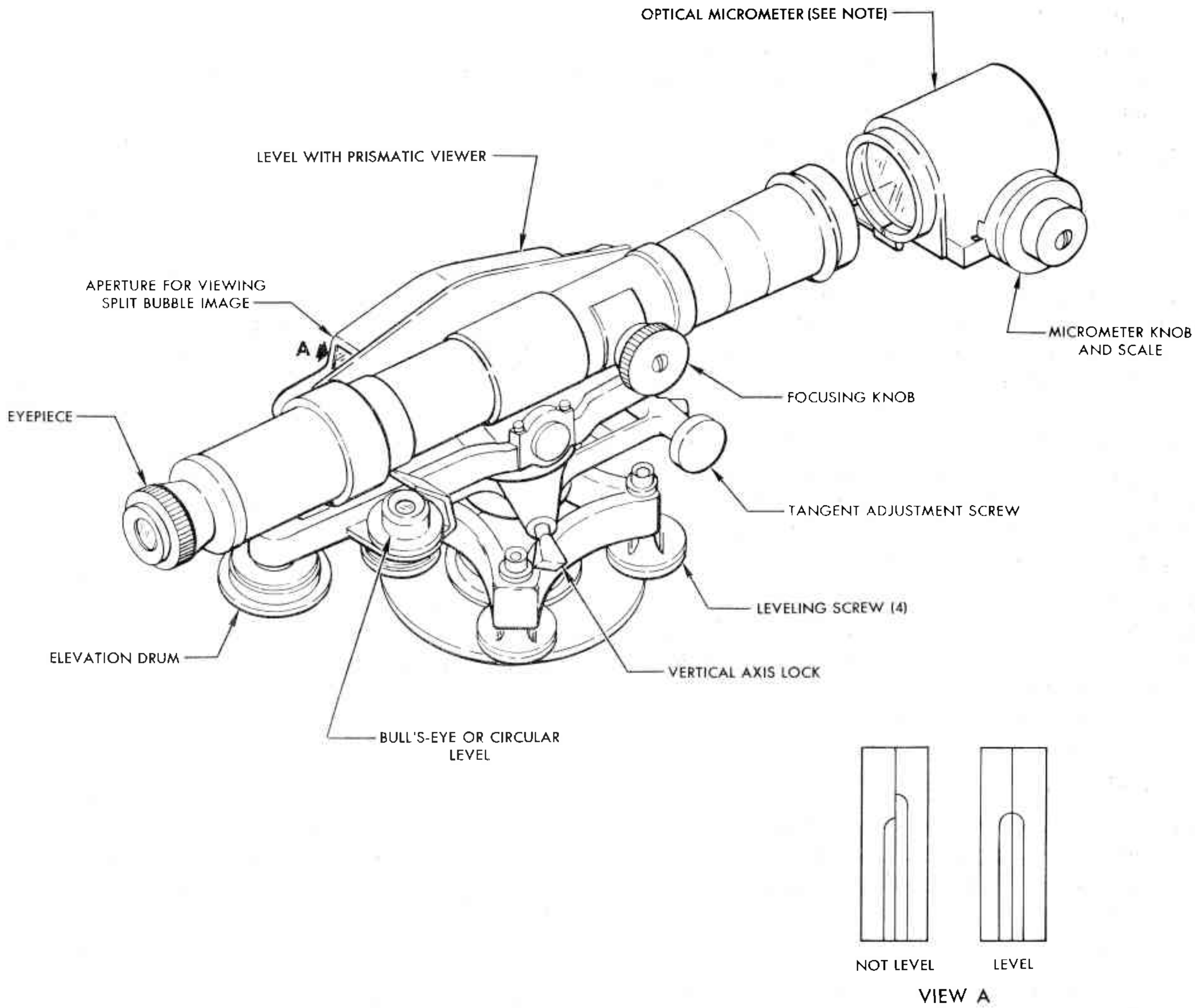
1-279. The sight level is the optical counterpart of the master or precision level. A sensitive prismatic viewing type spirit level is attached parallel to the optical axis of a telescope tube. The prismatic level provides a split image of a bubble. By matching the halves of each end of the split bubble, an accurate level can be established. Most sight levels are equipped with a bullseye bubble or circular level for roughing in the level of the instrument. The sight level is illustrated on figure 1-80. When the level of the scope is established, any point sighted through the telescope can be adjusted to the same horizontal plane with every other point.

1-280. Setting Level Points on a Horizontal Plane.

1-281. By using the following procedure, two or more points may be established on a surface in a horizontal plane to very close tolerances by means of the sight level.

1-282. Tools and Accessories Required.

FIGURE	QUANTITY	NAME	PART NUMBER
1-90	1	Instrument stand, standard height.	Keuffel & Esser No. 9092-20 or Brunson Model 230-2 or equivalent
1-90	1	Wyteface scale, 10.0 inches or more.	Keuffel & Esser No. 9099-30 or equivalent
1-80	1	Tilting (Dumpy) sight level, equipped with	Keuffel & Esser No. P-5022 or equivalent
	1	Prismatic viewing level, and	Keuffel & Esser No. 5097-46A or equivalent
	1	Optical micrometer.	Keuffel & Esser No. 9092-7 or equivalent



NOTE:
OPTICAL MICROMETER IS CLAMPED ON OBJECTIVE END OF TELESCOPE AND POSITIONED TO MEASURE DISPLACEMENT IN THE VERTICAL PLANE. SHOWN DETACHED FOR CLARITY.

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Figure 1-80. The Sight Level

1-283. Procedure.

a. Position stand near a point midway between ends of object to be leveled, taking care that a minimum distance of about six feet is maintained between the stand and the object.

b. Crank up or adjust height of stand to allow the sight level to be slightly above the surface to be leveled.

c. Carefully remove the sight level from the carrying case and mount it on the stand.

d. Level the instrument roughly by adjusting the four leveling screws until the bullseye or circular level indicates a leveled setting.

e. Swing telescope barrel in a horizontal plane until it is in line with a scale held vertically on any point of the surface.

NOTE

Move the scale forward and aft and observe the minimum scale dimension reading. Hold scale at the position where the minimum reading intersects the line-of-sight.

f. Observe the split bubble through the prismatic level viewing aperture to the left of the eyepiece. Adjust elevating drum under telescope eyepiece until the two halves of the split bubble are aligned. See figure 1-80 for an illustration of the split bubble.

g. With micrometer dial on sight level set at zero, focus on scale. See figure 1-79 for an illustration of telescope focusing procedure. Then rotate micrometer dial until the horizontal cross hair is located on an even decimal point on the scale. This location becomes the fix or reference point on the scale.

h. Record micrometer dial setting.

i. Swing instrument to aim at second point to be set. Reset split bubble by turning elevation drum under eyepiece to bring telescope portion of the instrument back on a level plane.

j. Focus telescope on the scale.

k. Rotate micrometer dial up or down until horizontal cross hair is located at same point on the scale as in step "g."

l. Compute the difference between the two micrometer readings. The resulting figure is the difference or error in level between the two points checked. To check a number of points repeat steps "i" thru "l," remembering to realign the split bubble by adjusting the elevating drum at each position before adjusting the micrometer.

1-284. Measuring or Setting Vertical Distance Between Two Horizontal Planes or Points With a Sight Level.

1-285. By using the following procedures, the vertical distance between two horizontal planes may be accurately measured with any reliable sight level.

1-286. Tools and Accessories Required.

Refer to paragraph 1-282 for list of equipment required:

1-287. Procedure.

a. Adjust stand height until sight level is located at any elevation between the two surfaces on which measurements are to be taken.

b. Adjust the leveling screws until the spirit level or split bubble level indicates a level setting.

c. Suspend a scale from upper point or horizontal plane so that a portion of the scale intersects the telescope line-of-sight.

d. Read and record the dimension viewed on the scale at the telescope horizontal cross hair.

e. Stand a scale from lower point or horizontal plane. Telescope may be swung to the right or left to allow scale to intersect the line-of-sight.

NOTE

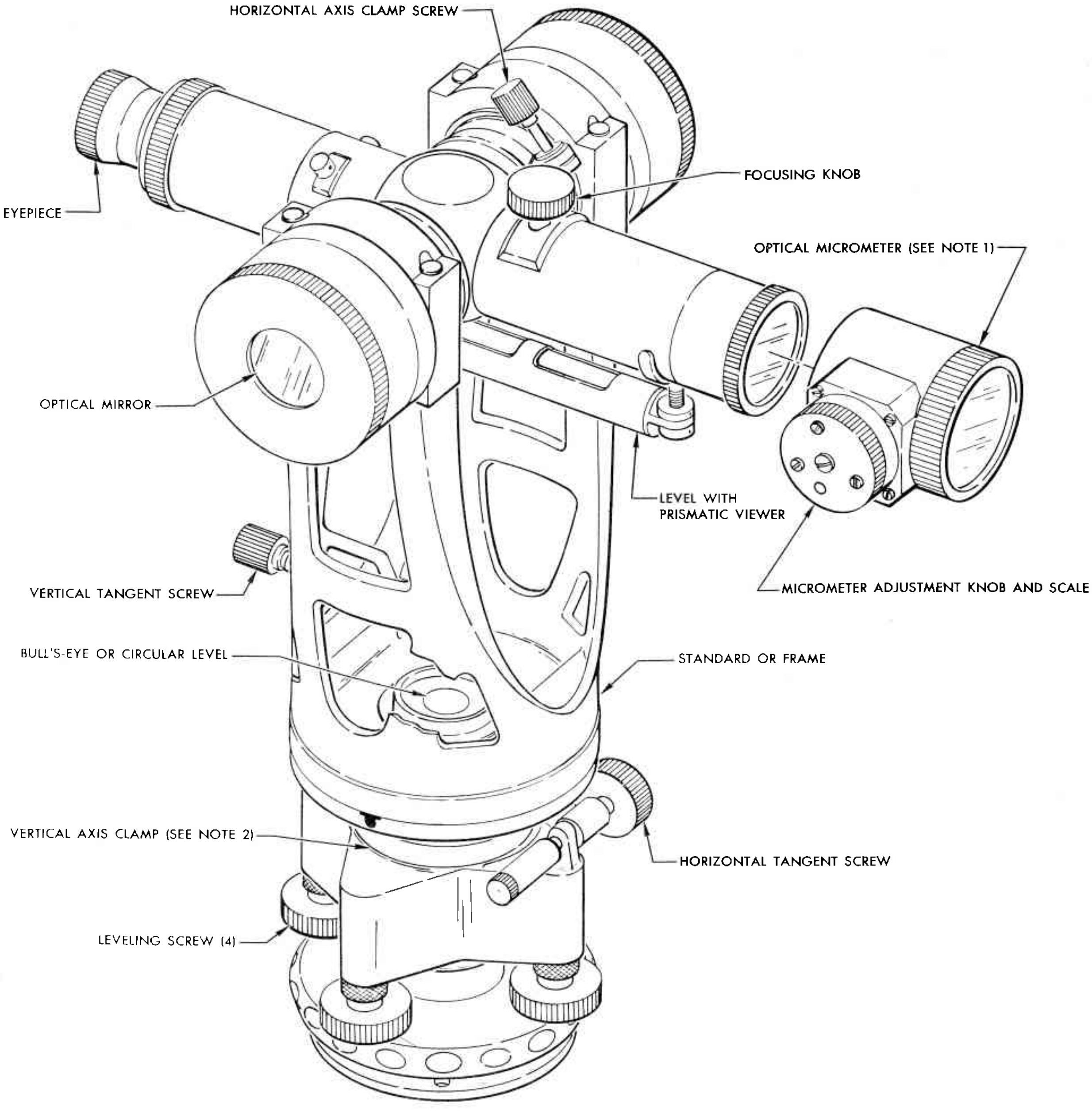
If the telescope is moved in a horizontal plane, the level indication must be checked and reset if any change has occurred.

f. Read and record the dimension viewed on the scale at the telescope horizontal cross hair.

g. Add the dimensions recorded in steps "d" and "f." The resulting total is the vertical distance between the points measured.

1-288. THE JIG TRANSIT.

1-289. The jig transit is similar to the surveyor's transit, an optical instrument commonly used for measuring land, except that the jig transit does not incorporate the 360-degree horizontal and vertical circle plates used for angular measurements. The major components of a jig transit are: a telescope, a heavy frame or standard, a spider, and a bottom plate. The telescope is mounted near its center of balance in the frame or standard to allow the scope to be easily rotated or plunged through a 360-degree vertical arc. The frame in turn is mounted on a spider and bottom plate with bushings or bearings which allow the telescope to be swung through 360 degrees in the horizontal plane. See figure 1-81 for an illustration of the jig transit. A bullseye, or circular level is mounted in the base of the standard or frame for rough leveling of the instrument. A more precise level, with a prismatic viewer which provides a split bubble image (see figure 1-80), is mounted on the tube of the telescope. An optical micrometer is mounted on the objective end of the telescope. Both horizontal axes of the telescope mount an axis mirror parallel to the sighting axis of the telescope. These mirrors are ground optically flat to within 0.000004-inch tolerance. Precise right angles to a line-of-sight can be determined by autoreflection or autocollimation with the optical axis mirror. This instrument has a lock clamp on the vertical



NOTES:

1. THE OPTICAL MICROMETER IS CLAMP MOUNTED ON THE OBJECTIVE END OF THE TELESCOPE AND MAY BE ROTATED TO MEASURE DISPLACEMENT IN THE VERTICAL OR HORIZONTAL PLANE, WHICHEVER APPLIES. IN THIS ILLUSTRATION THE OPTICAL MICROMETER IS SHOWN DETACHED FROM THE TELESCOPE FOR CLARITY.
2. TIGHTENING THE VERTICAL AXIS CLAMP SCREW, NOT VISIBLE IN THIS ILLUSTRATION, PREVENTS TRAVERSING OR HORIZONTAL MOVEMENT OF THE TRANSIT.

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Figure 1-81. The Jig Transit

and horizontal axes and fine adjustment screws for both these movements. These are known as tangent screws. The tangent screws permit rapid close tolerance settings when aligning or "bucking in" the jig transit. The primary use of the jig transit is to erect vertical planes or to establish lines perpendicular to a horizontal plane.

1-290. Erection of a Vertical Plane from a Horizontal Plane Using the Jig Transit.

1-291. A vertical plane or line may be erected from a horizontal plane or level surface by utilizing the following procedure. Refer to paragraph 1-280 for the procedure for leveling a horizontal plane.

1-292. Tools and Accessories Required.

FIGURE	QUANTITY	NAME	PART NUMBER		
1-90	1	Instrument stand, standard height, with cross slide.	Brunson Mod. 230-2	or	Keuffel & Esser No. 9092-20 or equivalent
	1	Lateral adjuster, for Keuffel & Esser stand.	Keuffel & Esser		No. 9099-71
1-81	1	Jig transit, equipped with	Brunson Model 71	or	equivalent
	1	Optical micrometer	Brunson No. 190	or	equivalent
	2	Mirrors, horizontal axis, auto-reflecting, and	Brunson No. 189	or	equivalent
	1	Prismatic viewing level.	Brunson No. 194	or	equivalent
1-90	1	Wyteface scale, 10.0-inches long.	Keuffel & Esser No. 9099-30	or	equivalent

1-293. Procedure.

a. Lay out a line or two points on the leveled horizontal surface of the fixture or object from which the vertical plane will be erected. This line will represent the lower edge of the vertical plane to be erected. If this procedure is used to check an existing vertical member, locate the line or points at an arbitrary distance of 1.00 to 3.00 inches to one side. Use a scale held against the vertical surface for a reference dimension from the vertical plane. This procedure prevents reflected light from interfering with the telescope line-of-sight.

b. Set up the instrument stand in line with the reference points or the line on the horizontal surface. Maintain a minimum distance of four feet from the nearest point on the reference line.

c. Carefully remove the jig transit from its carrying case and mount it on the instrument stand.

d. Rough level the instrument with the bullseye level by adjusting the four leveling screws at the bottom plate of the instrument.

e. Set optical micrometer at "0" reading and focus the eyepiece until the telescope cross hairs are seen clearly.

f. Aim the telescope at the far end of the reference line and adjust the telescope focusing knob until the line appears sharp and clear. See figure 1-79 for illustration of telescope focusing procedure.

g. Tighten clamp on the instrument vertical axis snug enough to prevent horizontal movement and adjust the horizontal motion tangent screw until the telescope vertical cross hair is superimposed on the far end of the line.

h. Rotate the telescope downward on its horizontal axis, aiming for the near end of the reference line. Readjust the telescope focusing knob until the line is clearly visible.

i. Adjust the instrument stand cross slide or lateral adjuster to move the telescope left or right until the vertical cross hair is again superimposed on the line or point.

NOTE

Do not readjust the horizontal motion tangent screw during this step.

j. Repeat steps "f" through "i" until telescope vertical cross hair remains superimposed on the line when swung to either end point.

k. Rotate transit on vertical axis to 90 degrees from the sighting position called out in step "g."

l. Plunge the telescope so that the prismatic leveling device is on the top of the scope tube. Plunging is a rotation of the telescope on its horizontal axis by an up-and-over movement of the eyepiece through 180 degrees of arc in a vertical plane.

m. With the sighting axis of the telescope in a horizontal position, tighten clamp on instrument horizontal axis snug enough to prevent vertical movement and adjust the vertical motion tangent screw until the split bubble (viewed through aperture in prismatic level) indicates a precise level position. See figure 1-80 for an illustration of the split bubble.

n. Loosen the vertical axis clamp and swing the telescope to left or right through 180 degrees of arc in a horizontal plane and note error indicated by split bubble. Remove one-half of the error by adjusting the two base leveling screws that are in line with the telescope sighting axis. The remaining error is removed by adjusting the vertical motion tangent screw.

o. Swing telescope back 180 degrees to original position and repeat the process of removing one-half the level error with the two base leveling screws and the remaining error with the vertical motion tangent screw as done in step "m."

p. Repeat steps "m" and "n" until the split bubble indicates a precise level at both positions.

q. Rotate telescope to left or right through 90 degrees of arc in a horizontal plane and remove *all* the remaining level error with the two base leveling screws that are in line with the telescope sighting axis.

r. Recheck level at all positions when swinging telescope in a horizontal plane. If a precise level indication is not maintained, repeat steps "m" through "p."

s. Swing telescope until the telescope vertical cross hair is superimposed once more on the near end of the reference line on the level surface or horizontal plane.

t. Tighten the vertical axis clamp snug enough to prevent horizontal motion and loosen the horizontal axis clamp.

u. Rotate telescope by depressing eyepiece end until the far end of the reference line is visible. If the telescope vertical cross hair does not remain superimposed on the reference line when moved to either end point, correct by repeating steps "f" through "i."

v. Elevate the telescope line-of-sight to the desired height for locating the upper points of the vertical plane above the horizontal surface. An existing vertical member, as mentioned in step "a," may be checked for deviations from the lower reference line by sighting the 1.00 to 3.00-inch arbitrary dimension point on a scale held against the vertical surface. Measure the deviation by turning the optical micrometer adjustment knob until the vertical cross hair has moved into line with the reference points, and reading the error on the micrometer scale.

1-294. THE ALIGNMENT TELESCOPE.

1-295. The modern optical tooling system for aircraft is founded upon the alignment telescope. This is the primary tool for the construction and checking of jigs and fixtures by establishment of optical reference lines in relationship to basic aircraft reference lines. The use of bulky master gages is falling rapidly into discard as tool engineers find that new optical methods bring previously unattainable precision and simplification into their tasks. An alignment telescope is an optical instrument of sturdy, rigid construction, containing cross hairs in the optical axis. The telescope optical axis is identical with the mechanical axis. The telescope is capable of providing and maintaining an accurate optical reference line or line-of-sight. The outer surface of the telescope tube is ground concentrically to the center line-of-sight of the telescope to provide for mounting the telescope within a standard spherical mount. See figure 1-82 for an illustration of the alignment telescope. Telescope magnification usually ranges from 30 to 60 power, depending on the make or model. Vertical or horizontal displacement from the telescope line-of-sight may be measured by integral or built-in optical micrometers. See figure 1-83 for an illustration of the principle of operation of the optical micrometer. Micrometer measurements, in a range of plus or minus 0.100 inch, are accomplished by a system of internal glass optical flats, mechanically linked to external adjustment knobs. Increments of 0.001 inch are read on a graduated scale at the external micrometer adjustment knob. An exception, the Farrand alignment telescope, provides a view of internal scales through the telescope eyepiece. See figure 1-84 for an illustration of the conditions to be avoided in establishment of the telescope line-of-sight.

1-296. Establishment of an Optical Reference Line or Line-of-Sight on the Fixture Using the Alignment Telescope.

1-297. The following procedure will provide one or more stable optical reference lines from which contour plates or facility gages may be set or fixed. Where more than one optical reference line is established, one line is designated as the basic line-of-sight and the other lines-of-sight are known as auxiliary or secondary lines-of-sight. This procedure employs an alignment telescope equipped with an adjustable focus eyepiece, adjustable objective focus, integral optical micrometers for measuring in two planes, and a spherical mount or adapter for mounting the telescope in the cup mounting base. The spherical mount is a steel sphere bored through the center to a diameter large enough to permit mounting a target or alignment telescope on the axis of the sphere. The cup mounting bases are permanently fixed by screws and dowel pins to a steel plate welded to the fixture. The spherical mount is not required for the Farrand alignment telescope which incorporates an integral spherical mount.

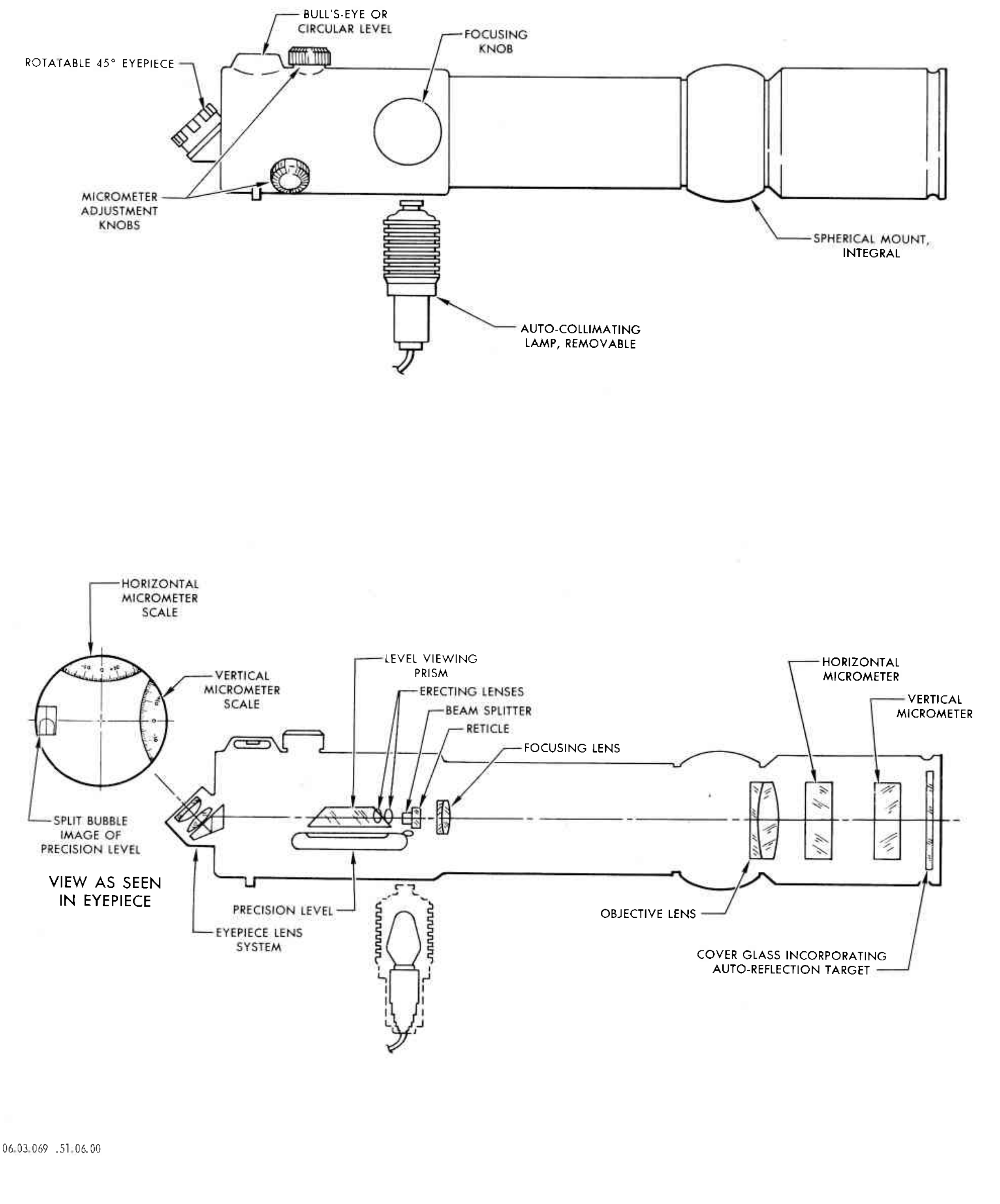


Figure 1-82. The Alignment Telescope

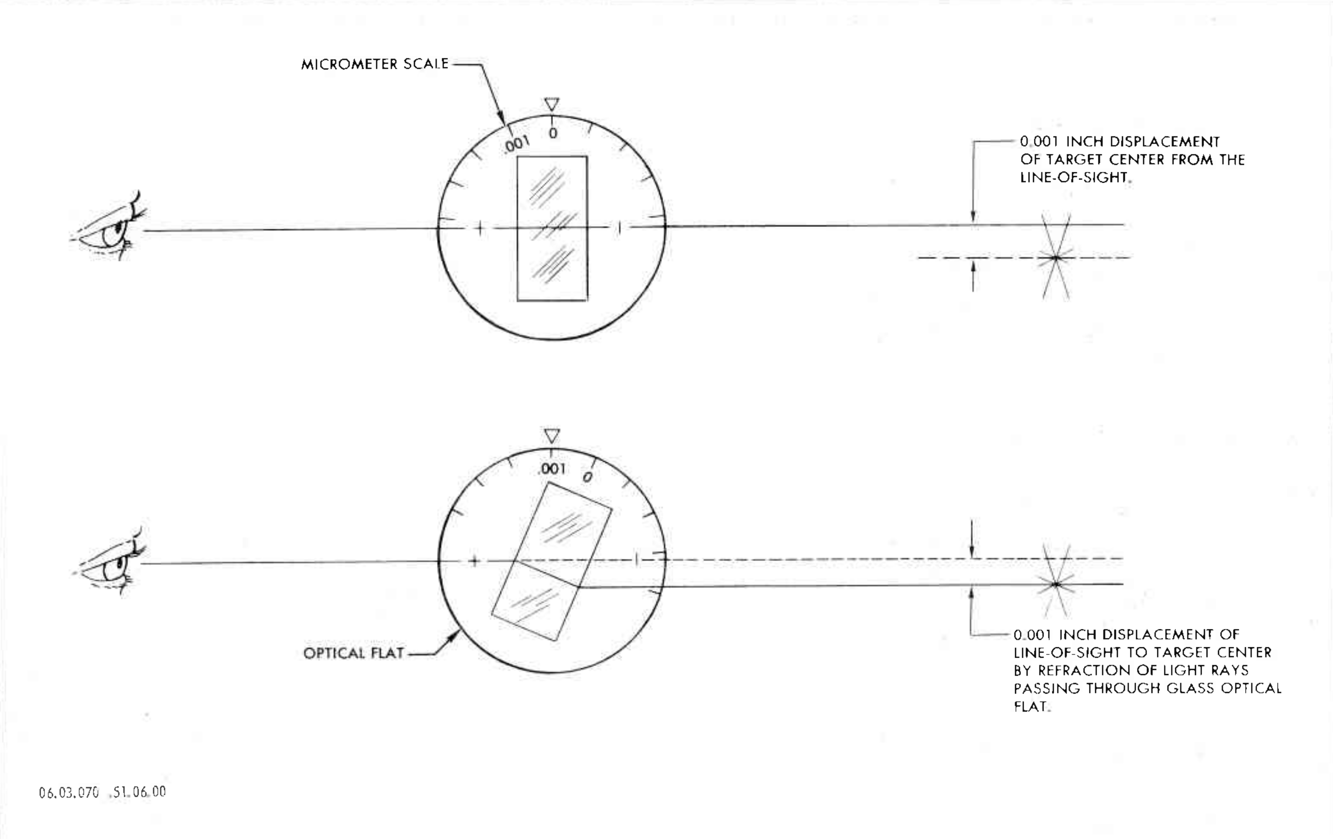


Figure 1-83. Principle of the Optical Micrometer

1-298. Tools and Accessories Required.

FIGURE	QUANTITY	NAME	PART NUMBER
1-85	2	Adjustable cup mounting base.	Farrand No. 87955 or Keuffel & Esser No. 9099-54, or equivalent
1-85	2	Cup mount clamp.	Farrand No. 31354 or Keuffel & Esser No. 9099-54½, or equivalent
1-85	2	Spherical mount or adapter.	Farrand No. 88291 or Keuffel & Esser No. 9099-53
1-85	1	Adjustable telescope mounting bracket.	Farrand No. 91671 or Keuffel & Esser 9099-57
1-85	1	Plastic target, 2¼-inch diameter.	Farrand No. 88342 or equivalent
1-90	1	Plastic target positioning tool.	Farrand No. 24391 or equivalent
1-85	1	Target illuminator.	Farrand No. 88292 or Any portable drop light
1-82	1	Alignment telescope.	Farrand No. 95360 or Taylor-Hobson No. TTH 112/365 or equivalent

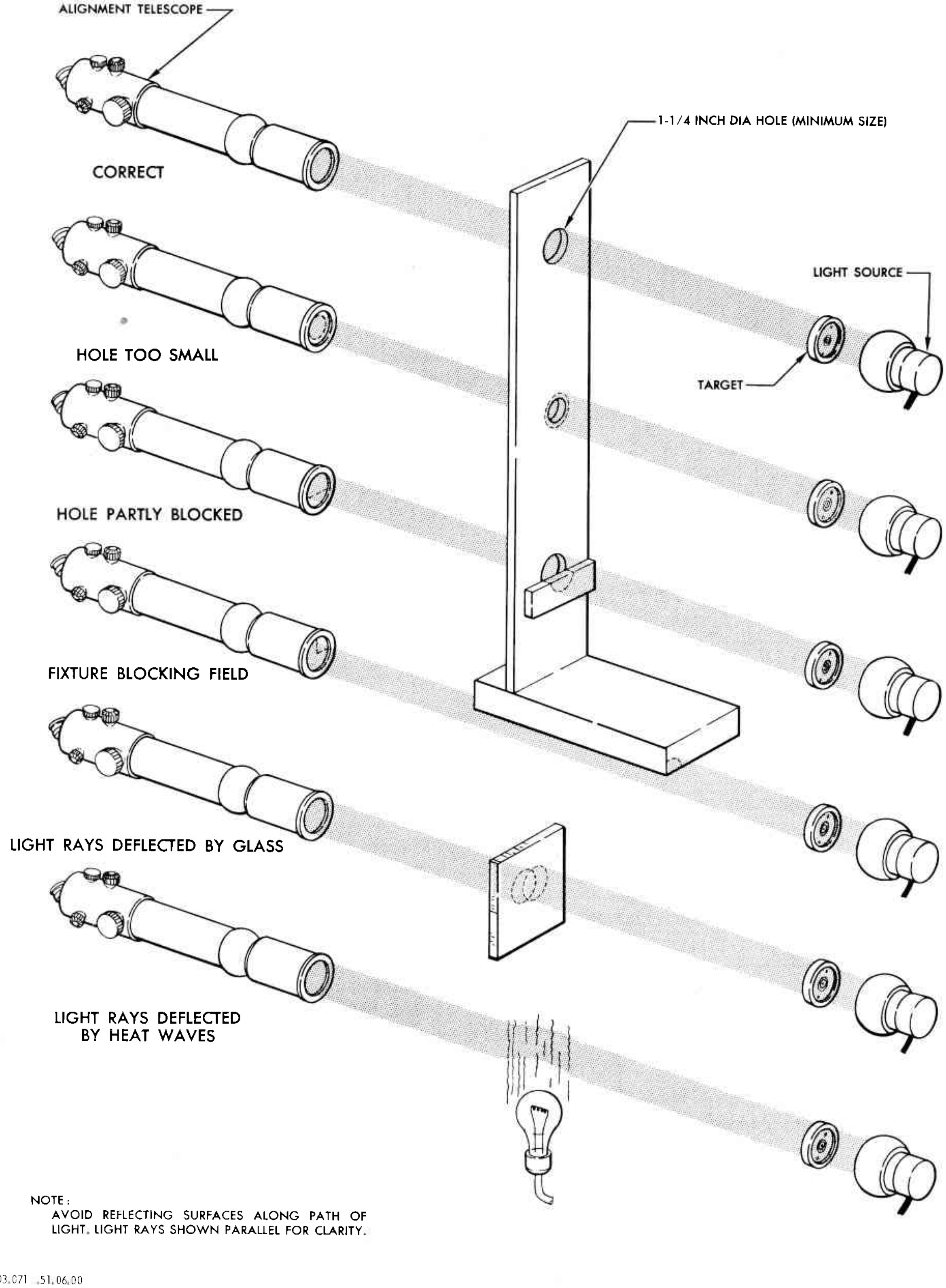


Figure 1-84. Telescope Line of Sight

1-299. Procedure.

a. In accordance with applicable tool design specifications, locate on the fixture and attach one adjustable cup mounting base at each end of the proposed line-of-sight. The cup mounts are secured by screws and dowel pins to a steel plate welded to the fixture. See figure 1-85 for attachment details.

b. If target is not already mounted in spherical mount, push a 2¼-inch diameter target into the bore of a spherical mount with the target positioning tool. The target center will coincide with the c/l of the spherical mount, thus allowing sphere to be tilted slightly without displacing the line-of-sight. Targets should remain installed in spheres at all times.

c. Install sphere containing target on cup mount base and secure with cup mount clamp. Tighten clamp snugly, but do not overtighten. Adjust sphere to face target towards the opposite cup mount.

d. Install the telescope bracket on the cup mount base at the opposite end of the fixture from the target mounting.

e. Carefully remove the telescope from its carrying case and set it on the cup mount with the spherical mount resting in the cup and the eyepiece end of the telescope resting in the adjustable bracket. Place cup mount clamp over the spherical mount on the telescope and tighten it just enough to assure a firm seat between the sphere and the cup.

f. Aim telescope at center of target and focus eyepiece until the image of the telescope cross hair appears sharp and clear. With both micrometers set at "0," adjust telescope focusing knob until target image is sharply defined and appears to be superimposed on telescope cross hairs. See figure 1-79 for an illustration of telescope focusing procedure and elimination of parallax.

g. Adjust the aim of the telescope by turning the adjusting screws on the telescope bracket until the telescope cross hair center point coincides with the target center.

1-300. Establishment of the Remote Line-of-Sight Using the Alignment Telescope.

1-301. The following procedure will reduce the number of optical instruments needed and will permit the fabrication of several fixtures at the same time. The procedure utilizes the remote line-of-sight which is established by mounting a target and an alignment telescope on stands separate from the fixture. Vertical planes at 90-degree angles to the fixture basic line-of-sight may then be rapidly established from the remote line-of-sight.

1-302. Tools and Accessories Required.

FIGURE	QUANTITY	NAME	PART NUMBER
1-90	3	Instrument stand, standard height.	Brunson Mod. 230-2 or Keuffel & Esser No. 9092-20 or equivalent
	2	Adapter, female with 3½-inch diameter x 8 threads per inch, for mounting cup mounting base on instrument stand.	Manufacture locally
1-85	2	Cup mounting base, adjustable.	Farrand No. 87955 or Keuffel & Esser No. 9099-54 or equivalent
1-85	2	Cup mount clamp.	Farrand No. 31354 or Keuffel & Esser No. 9099-54½ or equivalent
	2	Spherical mount, with	Farrand No. 88291 or Keuffel & Esser No. 9099-53 or equivalent
1-85	2	Target plastic, 2¼-inch diameter.	Farrand No. 88342 or equivalent
	1	Tilting (Dumpy) sight level,	Keuffel & Esser No. P-5022 or equivalent
1-85	1	Prismatic viewing level.	Keuffel & Esser No. 5097-46A or equivalent

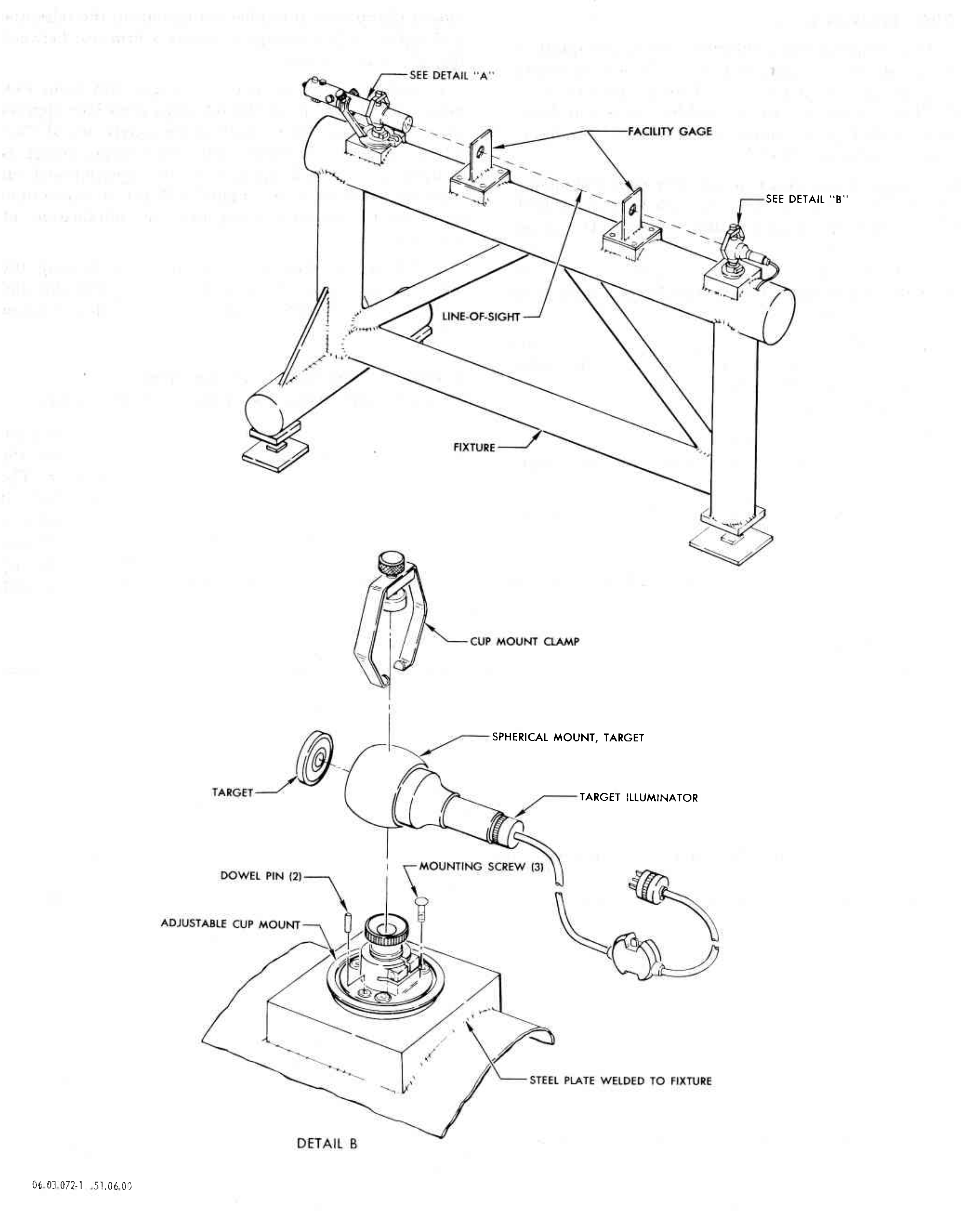


Figure 1-85. Mounting of Alignment Telescope and Target to Fixture (Sheet 1 of 2)

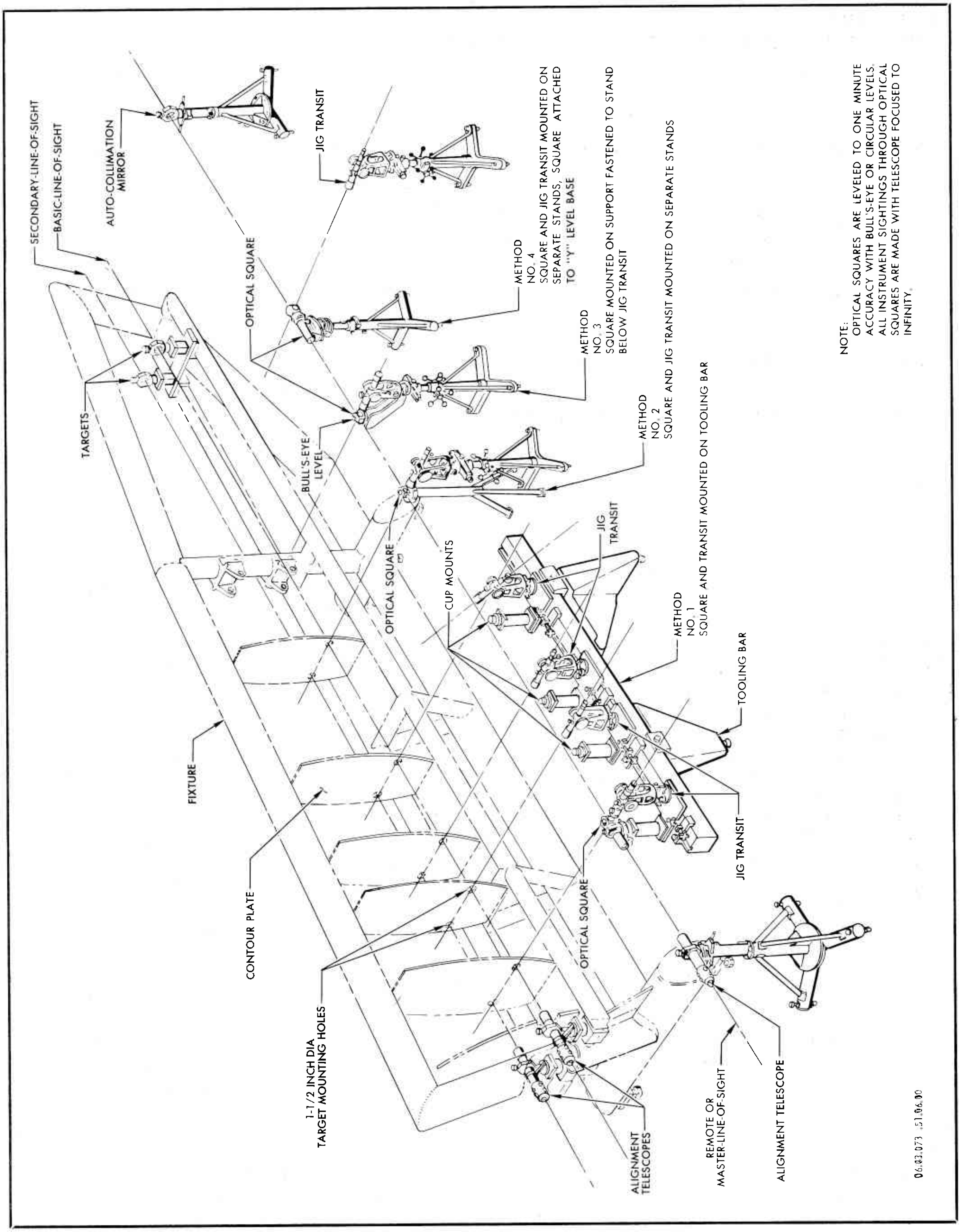


Figure 1-86. Methods of Establishing Right Angle Planes to the Line-of-Sight

1-306. Tools and Accessories Required (Cont).

FIGURE	QUANTITY	NAME	PART NUMBER
	1	Tilting (Dumpy) sight level equipped with	Keuffel & Esser No. P-5022 or equivalent
1-80	1	Prismatic viewing level.	Keuffel & Esser No. 5097-46A or equivalent
1-85	2	Cup mounting base, adjustable.	Farrand No. 87955 or Keuffel & Esser No. 9099-54 or equivalent
	2	Adapter, for mounting cup mounting base on instrument stand, female with 3½-inch diameter x 8 threads per inch.	Manufacture locally
	2	Spherical mount with	Farrand No. 88291 or Keuffel & Esser No. 9099-53 or equivalent
1-85	2	Plastic target, 2¼-inch diameter.	Farrand No. 88342 or equivalent
1-85	1	Bracket, alignment telescope mounting.	Farrand No. 91671 or Keuffel & Esser No. 9099-57 or equivalent
1-82	1	Alignment telescope.	Farrand No. 95360 or Taylor-Hobson TTH 112/365 or equivalent
1-81	1	Jig transit, equipped with	Brunson Mod. 71 or equivalent
	2	Mirror, horizontal axis, autoreflecting.	Brunson No. 189 or equivalent

1-307. Procedure.

a. Establish remote line-of-sight on two instrument stands, employing a target on one stand and an alignment telescope on the opposite stand. Refer to paragraph 1-300 for procedure. Position stands beyond ends of fixture on a line five or six feet to either side and parallel to the level fixture basic line-of-sight. Parallel of the lines-of-sight is established by measurement at each end with a steel tape or other mechanical measuring device.

b. Position instrument stand, equipped with lateral slide or adjuster, between the two stands along the remote line-of-sight and at an approximate right angle to the station or point on the fixture where the vertical plane will be erected.

c. Carefully remove the jig transit from its carrying case and mount it on the instrument stand.

d. Rough level the instrument by means of the bullseye or circular level in the base of the jig transit.

e. Adjust the height of the instrument stand until the jig transit axis mirror is intersecting the remote line-of-sight with the jig transit telescope aimed at the station or point on the fixture. See figure 1-81 for an illustration of the jig transit.

f. Insert the autoreflecting light source in the remote line-of-sight alignment telescope and turn it on. See figure 1-82 for an illustration of the light source.

g. Focus alignment telescope until the image of the autoreflection target in the telescope front cover glass is reflected from the jig transit axis mirror and appears sharp and clear. See figure 1-79 for an illustration of telescope focusing procedure.

h. Adjust the two leveling screws in the base of the jig transit that are in line with the remote line-of-sight until the telescope horizontal cross hair is superimposed on the center of the reflected target image. See figure 1-87 for an illustration of the principle of autoreflection.

i. Tighten vertical axis clamp screw (see figure 1-81) just enough to prevent movement of the transit in a horizontal plane and adjust the horizontal motion tangent screw until the telescope vertical cross hair is superimposed on the center of the reflected target image.

j. Sight through the jig transit telescope and focus on the station line or point established on the fixture. Adjust the lateral slide or adjuster on the jig transit instrument stand until the transit vertical cross hair is superimposed on the station line or point on the fixture.

k. Recheck autoreflection by sighting through the remote line-of-sight alignment telescope. Correct any error by repeating steps "g" through "i."

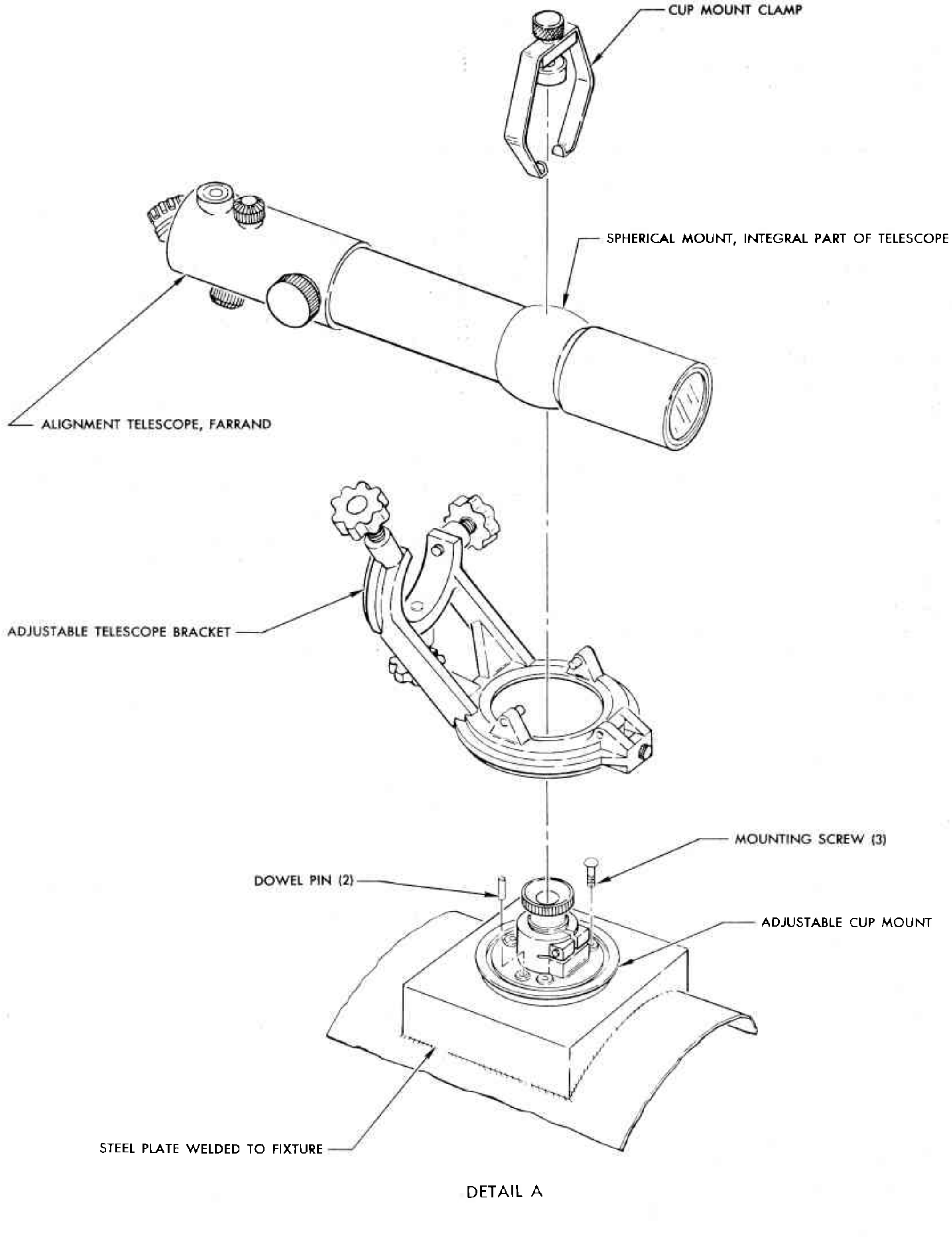


Figure 1-85. Mounting of Alignment Telescope and Target to Fixture (Sheet 2 of 2)

1-302. Tools and Accessories Required (Cont).

FIGURE	QUANTITY	NAME	PART NUMBER		
1-85	1	Bracket, adjustable, alignment telescope mounting.	Farrand No. 91671	or	Keuffel & Esser No. 9099-57 or equivalent
1-82	1	Alignment telescope.	Farrand No. 95360	or	Taylor-Hobson TTH 112/365 or equivalent

1-303. Procedure.

a. Position two of the instrument stands (designate as stands No. 1 and No. 2) as far apart as space will allow. The stands should be placed to the length of the area where the fixtures will be constructed, and to bisect the area. To eliminate the possibility of accidental movement, plaster the bases or feet of the instrument stands to the floor.

b. Install an adapter for cup mount base, a cup mount base, and a spherical mount containing a 2¼-inch diameter plastic target on each of stands No. 1 and No. 2. Secure the spherical mounts with cup mount clamps. See figure 1-85 for an illustration of mounting sequence.

c. Position the third stand (designate as stand No. 3) between stands No. 1 and No. 2 and five to six feet to either side of the line from stand No. 1 to stand No. 2.

d. Carefully remove the sight level from its carrying case and mount it on stand No. 3.

e. Adjust sight level to a rough level as indicated by the bullseye or circular level on the base of the instrument.

f. Crank up stand No. 3 until the sight level line-of-sight is 60 to 62 inches above the floor level.

g. Adjust the height of both targets on stands No. 1 and No. 2 to a level horizontal plane. Follow the sight level operating procedure outlined in paragraph 1-287.

h. Remove one target and spherical mount from either stand No. 1 or stand No. 2.

i. Install the telescope bracket on the cup mount base vacated by the target removed in step "h."

j. Carefully remove the telescope from its carrying case and set it on the cup mount with its spherical mount resting in the cup and the eyepiece end of the telescope resting in the adjustable bracket. Place cup mount clamp over the spherical mount on the telescope and tighten it just enough to assure a firm seat between the sphere and the cup.

k. Aim telescope at center of target and focus eyepiece until the image of the telescope cross hair appears sharp and clear. With both micrometers set at "0," adjust telescope focusing knob until target image is sharply defined and appears to be superimposed on telescope cross hairs. See figure 1-79 for an illustration of telescope focusing procedure and elimination of parallax.

l. Adjust the aim of the telescope by turning the adjusting screws on the telescope bracket until the telescope cross hair center point coincides with the target center. If the remote line-of-sight is 80 to 100 feet long and several fixtures are to be constructed at the same time, it is recommended that the target at the opposite end from the telescope be replaced with an alignment telescope. Both alignment telescopes will then be collimated from either end of the remote line-of-sight.

1-304. Establishment of a Vertical Plane 90 Degrees to the Line-of-Sight with the Jig Transit Being Autoreflected by Using the Jig Transit Axis Mirror.

1-305. The following procedure will establish a vertical plane at right angles to the line-of-sight. See figure 1-86 for an illustration of the method described in this procedure.

1-306. Tools and Accessories Required.

FIGURE	QUANTITY	NAME	PART NUMBER		
1-90	3	Instrument stand, standard height, with cross-slide.	Brunson Mod. 230-2	or	Keuffel & Esser No. 9092-20 or equivalent
	1	Lateral adjuster, for Kueffel & Esser instrument stand if used.	Keuffel & Esser No. 9099-71		

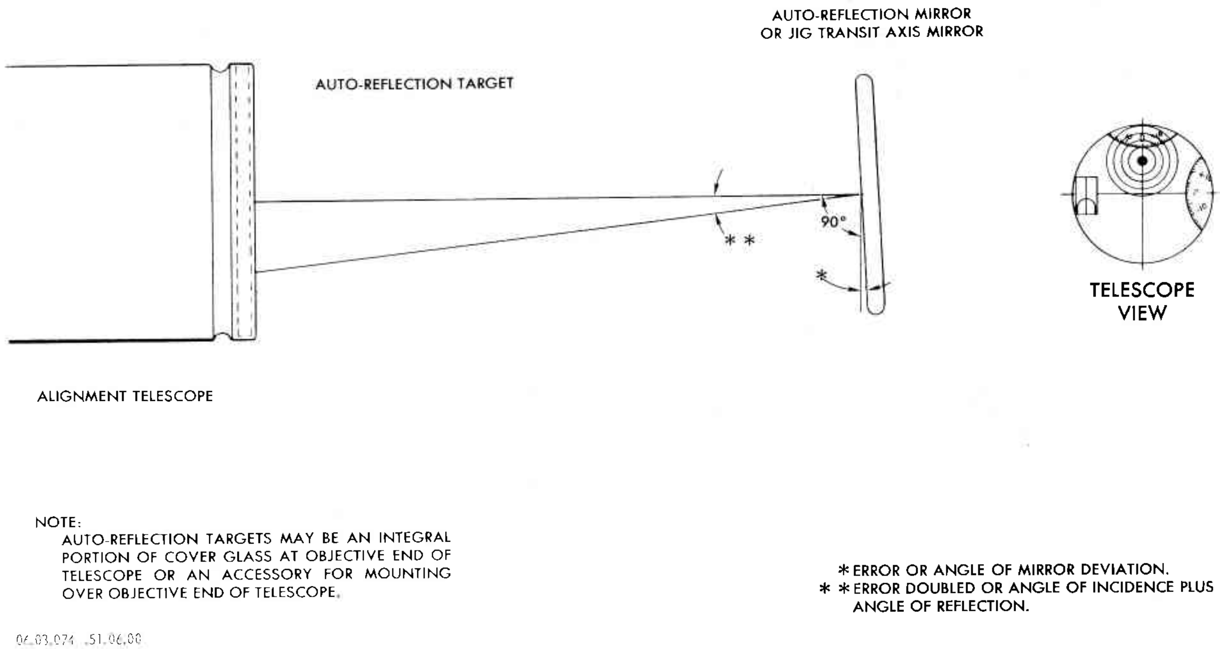


Figure 1-87. Principle of Autoreflection

1-308. THE OPTICAL SQUARE.

1-309. An important operation in the construction of aircraft jigs and fixtures is the establishment of station planes perpendicular to the lines-of-sight. One widely used method employs the optical square. The optical square provides a means of sighting targets at 90 degrees to the basic line-of-sight. The optical square consists of two reflecting mirrors located in a penta-prism relationship and mounted within a rigid metal frame. The frame is provided with a large bore for mounting the square over the objective end of the alignment telescope. The mounting surfaces for the mirrors are precision machined and lapped to hold the deviation from 90-degree angle to a tolerance of plus or minus two seconds. The reflecting mirror, located in line with the telescope sighting axis, is a partial mirror which permits viewing of the target in the line-of-sight without removing the square from the telescope. The optical square may be attached to an adapter and mounted on a separate instrument stand or support at a distance from the telescope. Separation of the optical square from the telescope will allow the square to be moved to any point along the line-of-sight and applied to a control system of linear measurement with the tooling bar. The jig transit may be fixed at right angles to the line-of-sight at any linear station location by mounting the optical square in the sighting axis of the alignment telescope and collimating by sighting through the jig transit and the optical square to the infinity-focused telescope. Two optical squares may be combined to determine parallel lines-of-sight. See figure 1-88 for an illustration of the optical square.

1-310. Establishment of a Vertical Plane 90 Degrees to the Basic Line-of-Sight by Collimation of the Jig Transit Through the Optical Square to the Alignment Telescope.

1-311. The following procedure will establish a vertical plane at right angles to the basic line-of-sight on the fixture. See figure 1-86 for an illustration of the methods described in this procedure.

1-312. Tools and Accessories Required.

FIGURE	QUANTITY	NAME	PART NUMBER
1-90	2	Instrument stand, standard height.	Brunson Mod. 230-2 or Keuffel & Esser No. 9092-20 or equivalent

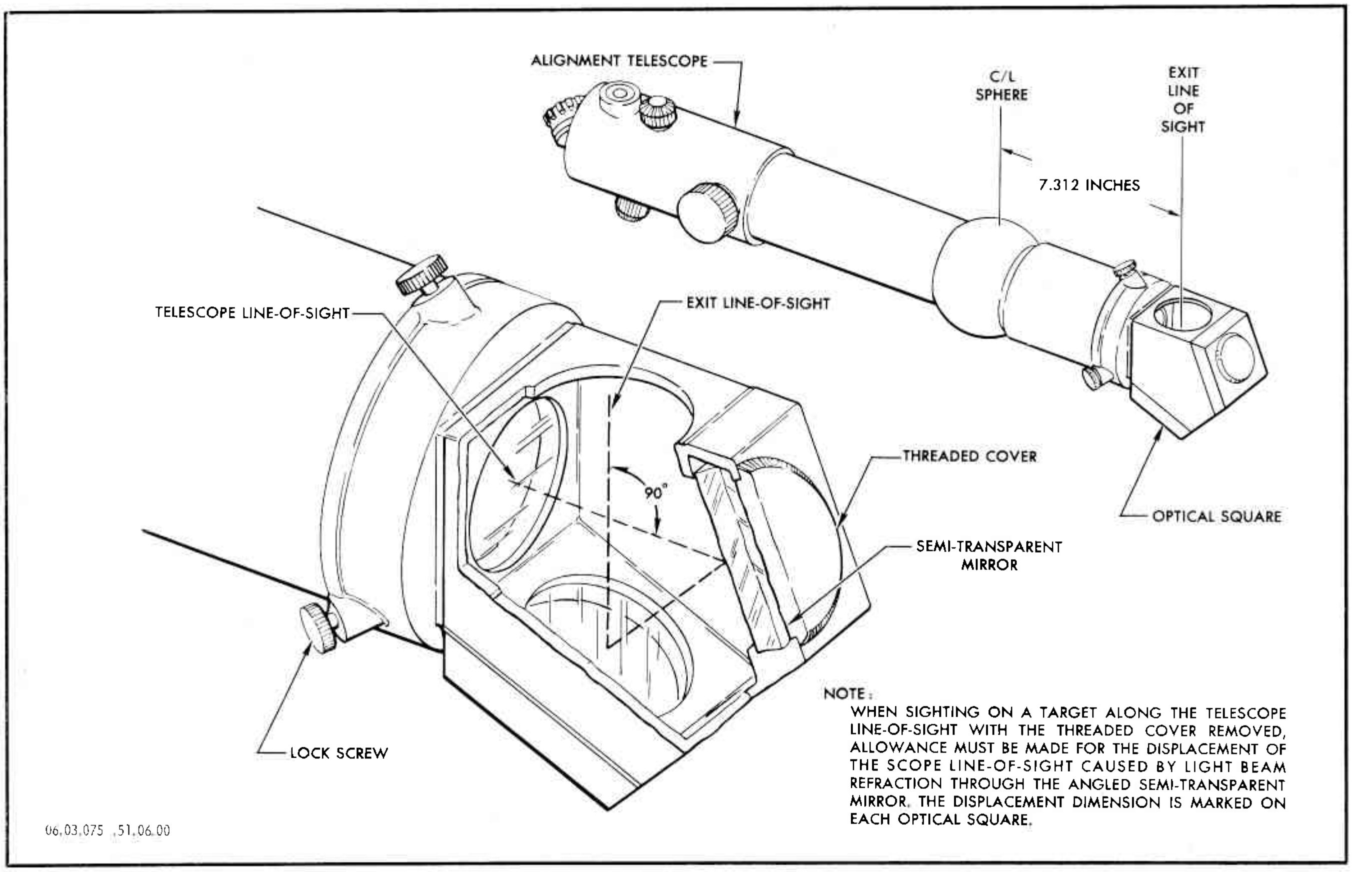


Figure 1-88. The Optical Square

1-312. Tools and Accessories Required (Cont).

FIGURE	QUANTITY	NAME	PART NUMBER
	1	Lateral adjuster, if Keuffel & Esser stand is used.	Keuffel & Esser No. 9099-71 or equivalent
1-81	1	Jig transit.	Brunson Mod. 71 or equivalent
1-88	1	Optical square.	Farrand No. 88305 Farrand No. 88183 or equivalent
	1	Adapter, optical square mounting.	Farrand No. 88308 or equivalent
1-85	1	Cup mount base.	Farrand No. 87955 or Keuffel & Esser No. 9099-54 or equivalent
1-78	1	Spherical mount.	Farrand No. 88291 or Keuffel & Esser No. 9099-53 or equivalent
1-78	1	Cup mount clamp.	Farrand No. 31354 or Keuffel & Esser No. 9099-54½ or equivalent

1-313. Procedure.

a. Establish remote line-of-sight on two instrument stands, employing a target on one stand and an alignment telescope on the opposite stand. Position stands beyond ends of fixture on a line five or six feet to either side and parallel to the level fixture basic line-of-sight. The lines-of-sight are made parallel by measuring between them at each end with a steel tape or other mechanical measuring device. Refer to paragraph 1-300 for the remote line-of-sight setting procedure and tool requirements.

b. Position instrument stand, equipped with lateral slide or adjuster, between the two stands along the remote line-of-sight and about 13 inches from the remote line-of-sight in a direction away from the fixture. Locate instrument stand at an approximate right angle to the station line or point on the fixture where the vertical plane will be erected.

c. Carefully remove the jig transit from its carrying case and mount it on the instrument stand.

d. Rough level the jig transit by means of the bullseye or circular level in the base of the instrument.

e. Mount the optical square with one opening intersecting the remote line-of-sight and the other opening facing the jig transit. Four methods of mounting the optical square are shown on figure 1-86. Method No. 1 is effected by securing the optical square to an adapter, Farrand No. 88308, which is an accurately ground tube that is, in turn, secured within a spherical mount. Farrand No. 88291. The spherical mount is then clamped in a cup mount base. Methods No. 2 and No. 3 utilize a flat metal plate which is mounted on an instrument stand or a support attached to the instrument stand below the base of the jig transit. The metal plate and the support arms are fabricated locally. Method No. 4 requires an adapter, Farrand No. 88308, to which the optical square is secured, and a "Y" support base. The tube of the adapter rests in a horizontal plane in the vees of the "Y" support which is in turn mounted on an instrument stand. The "Y" support is fabricated locally and incorporates a bullseye or circular level in its base. Regardless of the type of mounting employed, it is recommended that the optical square be leveled to one-minute accuracy by a bullseye or circular level mounted on the side of the square housing in order that the square may be set in the same plane each time it is moved.

f. Level jig transit telescope with the prismatic viewing level on the tube of the telescope. Refer to paragraph 1-293 for leveling procedure.

g. Aim the jig transit at the opening in the optical square and focus until the target pattern on the front glass of the alignment telescope is sighted. Adjust the transit horizontal motion tangent screw until the transit vertical cross hair is centered on the target bullseye. It is not necessary to center the transit horizontal cross hair on the target center.

h. Focus both the alignment telescope and the jig transit to infinity. Collimate by superimposing the transit vertical cross hair over the alignment telescope cross hair which will be faintly visible. Make minor adjustment of transit horizontal motion screw until the vertical cross hairs are aligned.

i. Aim and focus jig transit on the station line or point on the fixture where the vertical plane will be erected. Remove square if necessary. Adjust lateral slide or adjuster on the instrument stand until the jig transit vertical cross hair is superimposed on the station point on the fixture. Do not adjust the horizontal motion tangent screw.

j. Recheck square relationship of jig transit to the remote line-of-sight by repeating step "h."

1-314. Establishment of a Vertical Plane 90 Degrees to the Basic Line-of-Sight with the Optical Square and the Tooling Bar.

1-315. Linear station control and squaring of the station planes along the fixture may be achieved by employment of tooling bars, alignment telescopes, and optical squares. The tooling bar may be any form of dimensionally stable bar or beam that is fixed in a parallel relationship to the lines-of-sight. The bar is provided to support one or more instruments to be used for squaring purposes. The tooling bar may be designed as a separate portable unit or it may be attached as a permanent and integral part of the fixture to provide a permanent linear dimension control. The bar is usually constructed with a base or slide unit attached to mount an optical instrument. The slide unit may be moved along the bar to any station plane and is designed to index into a series of precise, jig-bored holes spaced evenly at 10.000-inch intervals along the length of the bar. The following procedure will fix a station vertical plane at right angles to the fixture basic line-of-sight and at the proper linear dimension point along the fixture, assuming the basic line-of-sight on the fixture has been leveled and cup mounts are installed on each end of the portable tooling bar. See figures 1-86 and 1-89 for an illustration of the tooling bar.

1-316. Tools and Accessories Required.

FIGURE	QUANTITY	NAME	PART NUMBER
1-89	1	Tooling bar, portable, with index and slide unit and cup mounts at each end of the bar.	Fabricate locally

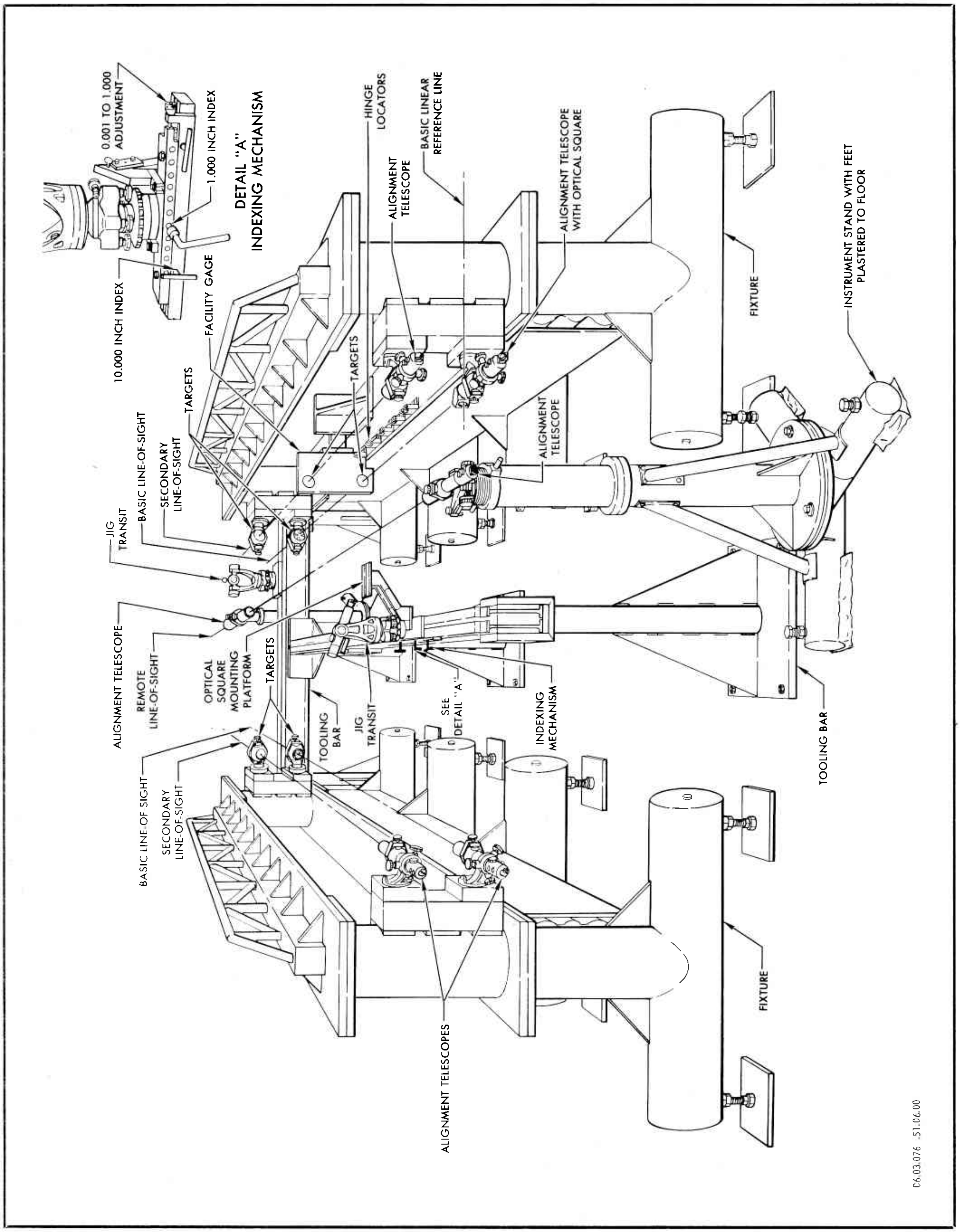


Figure 1-89. Optical Tooling Index Bars Used in "T" Formation

1-316. Tools and Accessories Required (Cont).

FIGURE	QUANTITY	NAME	PART NUMBER		
1-85	1	Cup mount base, adjustable.	Farrand No. 87955	or	Keuffel & Esser No. 9099-54 or equivalent
1-85	2	Spherical mount.	Farrand No. 88291	or	Keuffel & Esser No. 9099-53 or equivalent
1-85	2	Target, plastic, 2¼-inch diameter.	Farrand No. 88242	or	equivalent
1-85	3	Cup mount clamp.	Farrand No. 31354		Keuffel & Esser No. 9099-54½ or equivalent
1-90	1	Instrument stand, standard height.	Keuffel & Esser No. 9092-20	or	Brunson Mod. 230-2 or equivalent
1-85	1	Tilting (Dumpy) sight level equipped with	Keuffel & Esser No. P-5022	or	equivalent
	1	Prismatic level and	Keuffel & Esser No. 5097-46A	or	equivalent
	1	Optical micrometer.	Keuffel & Esser No. 9092-7	or	equivalent
1-82	2	Alignment telescope.	Farrand No. 95360	or	equivalent
1-88	3	Optical square.	Farrand No. 88305	or	Farrand No. 88183 or equivalent

1-317. Procedure.

a. Move the tooling bar alongside the level fixture and parallel to the fixture basic line-of-sight. The basic line-of-sight setting procedure is outlined in paragraph 1-296. Locate the tooling bar at a minimum distance of five to six feet from the fixture basic line-of-sight. Position the tooling bar so that one of its end cup mounts is located at an approximate right angle to the alignment telescope cup mount on the fixture. (The tooling bar will be moved to a more precise relationship with the fixture in the steps that follow.) Check for an approximate parallel with as much accuracy as may be determined by direct measurements between the long axis of the tooling bar and the fixture basic line-of-sight, with a steel tape or other mechanical measuring device.

b. Set a spherical mount containing a target on each of the two cup mounts on the tooling bar, one at each end. Secure the spherical mounts with cup mount clamps.

c. Position the instrument stand alongside the tooling bar and level targets on tooling bar ends to a horizontal plane by means of the sight level. Refer to paragraph 1-280 for the sight level operating procedure. The tooling bar elevation may be above or below the fixture basic line-of-sight but both of them must be level.

d. Remove the spherical mount and target and mount an alignment telescope on one end of the tooling bar (opposite the alignment telescope mounted on the fixture) and aim for the target on the other end of the tooling bar. Establish a reference line-of-sight on the tooling bar.

e. Secure an optical square on the objective ends of both alignment telescopes, one on the tooling bar and one on the fixture. Rotate both of the squares so that the exit sighting axis of one square is aimed at the exit opening of the opposite square. Both telescopes and both squares must be of the same type in order that the line-of-sight between them will be located at the same distance from the telescope cup mounts. See figure 1-88 for an illustration of the optical square.

f. Insert telescope internal illuminating light (see figure 1-82) in each of the telescopes. Designate the fixture telescope as scope No. 1 and the telescope on the tooling bar as scope No. 2.

g. Turn on the light in scope No. 2.

h. Sight through scope No. 1 and adjust the focusing knob until the target on the front glass of the lighted telescope is visible. The path of the line-of-sight will then be through the fixture telescope, through an optical square, across to the square on the tooling bar, and through it to the front glass target of scope no. 2 Revolve the telescope on its optical axis in the cup mount until the telescope horizontal cross hair is superimposed on the center of the target image.

i. Turn off the light in scope No. 2 and turn on the light in scope No. 1. Sight through scope No. 2 and repeat step "i." Alignment of both of the optical squares in one plane is then established.

j. Move the base of the tooling bar assembly slightly along the long axis of the tooling bar until the telescope vertical cross hair is superimposed on the center of the front glass target image. This will establish the spherical mounts of both telescopes on the same station plane.

k. Collimate by turning the focusing knobs on both telescopes to infinity and sighting through scope No. 2 to scope No. 1. The cross hairs of the tooling bar telescope should be superimposed on the faint image of the cross hair from within the telescope on the fixture. Make final adjustment by slightly moving the tooling bar until both the vertical and horizontal cross hairs are aligned. The parallel alignment of the fixture and the tooling bar lines-of-sight is then assured.

l. Mount a third telescope (designate as scope No. 3) with an optical square attached to its objective end on the index slide unit on the tooling bar. Adjust the telescope until its sighting axis is aimed at the objective end of scope No. 2.

m. Turn on the light in scope No. 2 and remove the screw-type covers from the ends of the optical squares on scopes No. 2 and No. 3. See figure 1-88.

n. Collimate by turning the focusing knobs on scopes No. 2 and No. 3 to infinity and adjusting the slide unit telescope aim until the cross hairs in both telescopes are superimposed on each other. The slide unit telescope sighting axis will then be parallel to the sighting axis of the end telescope on the tooling bar.

o. Index the slide unit to the desired station setting and recheck collimation of the two telescopes on the tooling bar.

p. Replace the screw-type cover on the slide unit optical square and rotate the square by revolving the telescope in the cup mount until the desired station point on the fixture is visible.

q. Recheck collimation by again removing the square cover on scope No. 3 and sighting to the end telescope. The right angle plane to the fixture basic line-of-sight is now established.

1-318. The Contour Plate and the Facility Gage.

1-319. The contour plate is similar to the facility gage in function but is designed to fix a contour outline, whereas the facility gage is designed to locate specific points on the fixture. The contour plate and the facility gage are tailored to the specific requirements of the fixture being constructed. They are designed for the purpose of fixing airplane reference points at the proper location in the fixture and with the proper relationship to the line or lines-of-sight. They serve as a link or mechanical extension from the line-of-sight to locate fixture details at precise airplane reference points. They must be made of metal of sufficient thickness to insure dimensional stability. They may be of any shape or size needed to incorporate a target or targets in one or more lines-of-sight and to incorporate one or more reference points of the airplane assembly to be constructed. These tools may be set or fixed in the line-of-sight at any point between the target and the telescope, keeping in mind that most telescopes have a minimum focal length of from 10 to 20 inches forward of the objective end of the telescope. Therefore, the nearest detail to the telescope must be located at a distance slightly more than the minimum focal length. See figure 1-89 for an illustration of a typical facility gage and figure 1-86 for an illustration of a typical contour plate.

1-320. Preparation of the Contour Plate or Facility Gage.

1-321. Dimensions, materials, hole locations and other pertinent details for fabrication of contour plates and facility gages are generally called out on the tool design specification for the fixture under construction. Frequently the jig builder will be required to design and build supplementary facility gages or contour plates to accomplish the job in hand. A contour plate is indicated if, for example, a contour outline is to be established at a fixture station point. The contour plate is prepared by selecting a thick sheet of metal, preferably aluminum, and shaping the edge of the sheet to the desired contour outline. The edge of the contour plate is shaped to the skin or bulkhead contour by inscribing the contour outline on the sheet and trimming away the portion of the plate outside the skin. Target mounting holes are jig-bored to 1½-inch diameter at a point or points where the target center will intersect one or more lines-of-sight. A facility gage is indicated when it is desirable to fix a specific point on a fixture detail such as a hinge point, which will be installed on the fixture at a point not intersecting the line-of-sight. The facility gage must be designed to bridge the gap between a point on the detail and the line-of-sight. These gages are usually a flat metal plate having one or more 1½-inch diameter target mounting holes to intersect the lines-of-sight and one or more dowel pin holes to match identical holes in the detail. Both the facility gage and the contour plate may be mounted to the detail or fixture by dowel pins or by a surface that coincides with a related surface on the detail or fixture.

1-322. Establishment of the Contour Plate or Facility Gage in the Fixture.

1-323. The following procedure will locate and fix a contour plate or facility gage in the fixture in the proper relationship to the line-of-sight:

1-324. Tools and Accessories Required.

FIGURE	QUANTITY	NAME	PART NUMBER
1-90	As required	Target, plastic, 1½-inch diameter.	Farrand No. 88383 or Keuffel & Esser No. 8013956 or equivalent

1-325. Procedure.

a. Establish the line-of-sight by mounting an alignment telescope at one end of the fixture and a target at the opposite end. Refer to paragraph 1-297 for procedure.

b. Mount the small plastic target by pushing it into the 1½-inch diameter hole in the contour plate or facility gage. If more than one line-of-sight is employed, additional targets are mounted as indicated.

c. At a point beyond the telescope minimum focal length between the telescope and the target secure the contour plate or facility gage to the detail or fixture with C-clamps.

d. Adjust the contour plate or adjust the facility gage and the detail by movement in vertical and horizontal directions until the telescope cross hairs are centered on the bullseye of the target mounted in step "b." Two men are required for this operation.

e. Bolt and dowel pin the detail in place.

f. Recheck telescope sighting on contour plate or facility gage target.

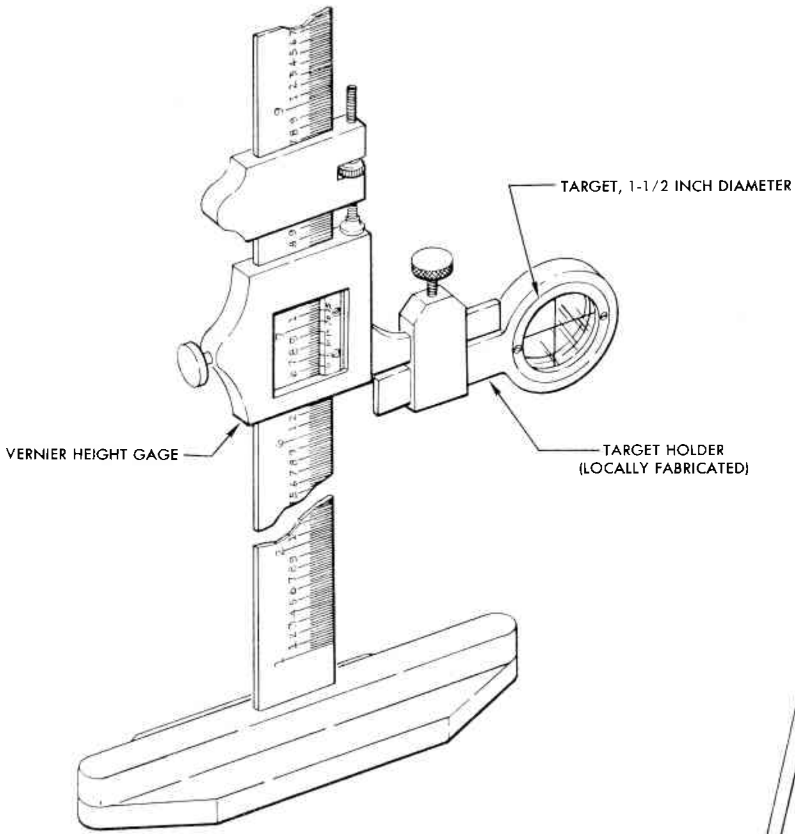
g. Remove the 1½-inch diameter target from the facility gage or contour plate and recheck the telescope sighting on the target at the end of the fixture.

1-326. Establishment of Vertical Planes Either Parallel or 90 Degrees to the Basic Line-of-Sight.

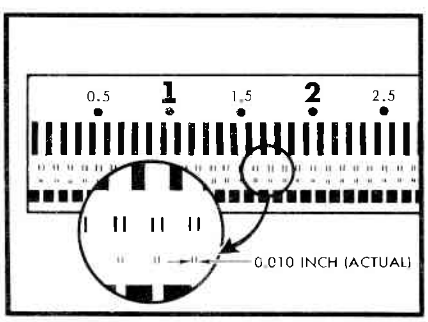
1-327. Fabrication of several fixtures simultaneously may be expedited by utilizing two tooling bars and a remote line-of-sight. The remote line-of-sight established in this procedure is separate from the indexed tooling bars to allow the tooling bar to serve as a mounting stand for more than one jig transit. The tooling bars are located to form a "T," one bar parallel to the remote and the fixture basic lines-of-sight and the other at right angles to the lines-of-sight. This setup permits rapid changes from one vertical plane either parallel or at right angles to the lines-of-sight, and maintaining precise linear dimension control by use of the indexing mechanism on the tooling bars. An understanding of the previous paragraphs describing the use of the four basic instruments, i.e., sight level, jig transit, alignment telescope, and optical square, is a prerequisite to setting up this optical system. See figure 1-89 for an illustration of this system.

1-328. Tools and Accessories Required.

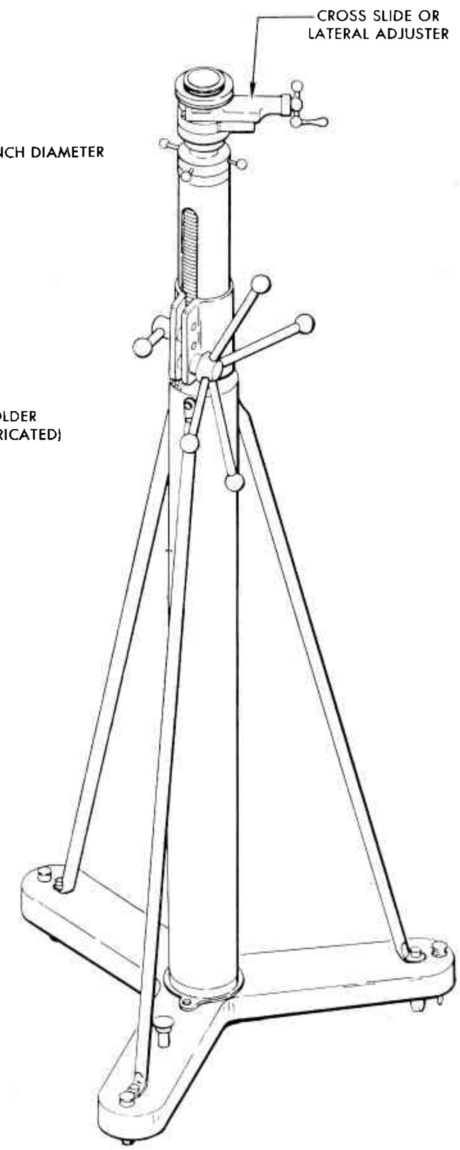
FIGURE	QUANTITY	NAME	PART NUMBER
1-89		Tooling bar, with indexed slide unit.	Fabricate locally
	1	Jig transit, equipped with	Brunson Mod. 71 or equivalent
1-81		Prismatic level, and	Brunson No. 174 or equivalent
	2	Mirror, horizontal axis, autoreflection.	Brunson No. 189 or equivalent



VERNIER HEIGHT GAGE, TARGET, AND HOLDER COMBINATION



WYTEFACE SCALE



INSTRUMENT STAND, WITH CROSS SLIDE

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Figure 1-90. Optical Tooling Accessories (Sheet 1 of 2)

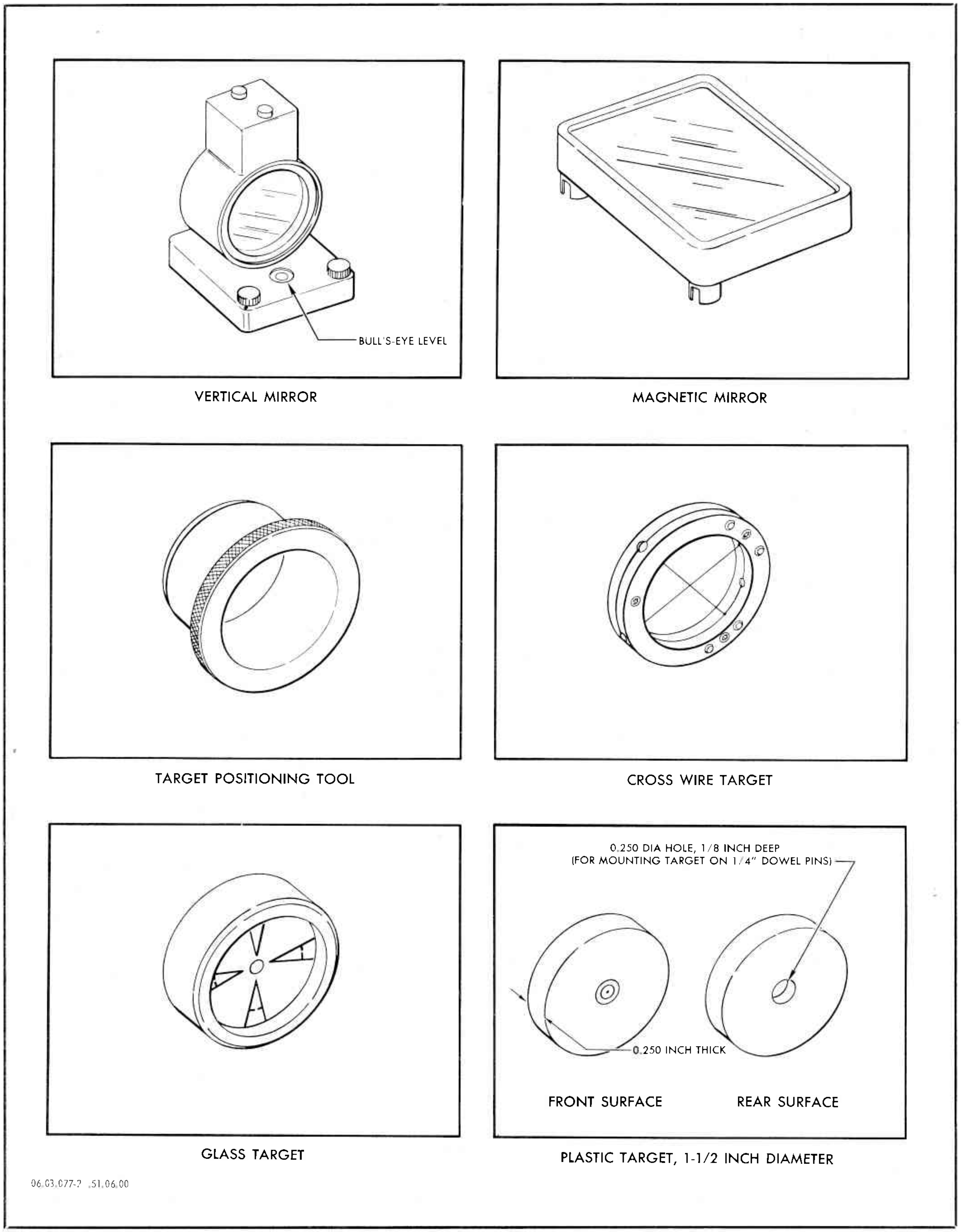


Figure 1-90. Optical Tooling Accessories (Sheet 2 of 2)

1-329. Procedure.

a. Establish a remote line-of-sight in the most practical location available in the work area to allow fixture construction on both sides along the line-of-sight. Refer to paragraph 1-300 for tools required and a step-by-step procedure. Plaster stand bases to floor.

b. Position one of the tooling bars on either side parallel to the remote line-of-sight. Accomplish parallel alignment by scale measurement. Locate the center line of the tooling bar about 13 inches from the remote line-of-sight.

c. Adjust the height of the tooling bar top surface to a position about 15.625 inches below the remote line-of-sight. This dimension will compensate for the height of the slide unit, mounting adapter and the jig transit. Level bar with a sight level. Refer to paragraph 1-280 for tools required and the procedure for leveling tooling bar. Plaster bases of bar to floor.

d. Position the second tooling bar at right angles to the first tooling bar so as to form a letter "T." Adjust the height of the top surface of the bar and the level to the same horizontal plane as established for the first tooling bar in the previous step.

e. Mount a jig transit equipped with axis mirrors on the second tooling bar with the slide unit moved near a point midway between the bar ends and so the axis mirrors intersect the remote line-of-sight. Align the sighting axis of the jig transit with the center line of the second tooling bar by "plunging" transit telescope to sight on first one end and then the opposite end of the center line. Compensate until center line of transit is on center line of bar. Refer to paragraph 1-293 for operational procedure of the jig transit.

f. Sight through the remote line-of-sight alignment telescope and square the second tooling bar by autoreflection from the jig-transit axis mirror. Move tooling bar slightly until the alignment telescope cross hairs are superimposed on the reflected target image of its front glass target. Refer to paragraph 1-304 for a detailed procedure employing autoreflection from the jig transit axis mirror. Plaster bases of bar to floor.

g. Move fixtures into position parallel to the remote line-of-sight. Parallel alignment is established by direct measurement with steel tape. Locate the fixtures at a minimum distance of five or six feet from remote line-of-sight, allowing a two-foot clearance between the end of the fixture and the second tooling bar. Establish a basic line-of-sight on the fixture and level fixture with the sight level. Refer to paragraph 1-297 for tools required and procedure for setting the basic line-of-sight.

h. Establish vertical planes on the fixture by mounting optical squares in the remote line-of-sight and collimating to jig transits mounted on the tooling bar. Refer to paragraph 1-312 for procedure and tools required for this operation.

1-330. Procurement of Instruments and Accessories.

1-331. Optical instruments and their accessories which have been mentioned in the foregoing procedures may be obtained by contacting the following manufacturers:

Brunson Instrument Co.
1405 Walnut St., Kansas City, Missouri

Farrand Optical Co., Inc.
Bronx Blvd & East 238 St., New York 70, N.Y.

Keuffel & Esser Co.
Hoboken, New Jersey

Taylor, Taylor & Hobson Ltd.
Stoughton St., Leicester, England

1-332. GLOSSARY OF OPTICAL TOOLING TERMS.

ABERRATION—The failure of rays of light to converge to a focal point passing through a lens.

ALIGNING BRACKET—A clamping and/or adjusting device which holds an optical instrument on its mounting base and permits alignment of this instrument with a predetermined line-of-sight.

ALIGNMENT TELESCOPE—A telescope containing cross hairs with its optical axis coinciding to the center of the bore of the tube. Provisions for mounting the telescope on a standard spherical mount are also incorporated on the scope.

AUTOCOLLIMATION—The process of establishing 90-degree angles by means of a mirror which reflects the telescope reticle. The angular deviation is measured by the amount of displacement of the reflected cross hairs from the original reticle.

AUTOREFLECTION—The process of checking 90-degree angles to the line-of-sight by using a mirror to reflect the image of a target placed at the objective lens of the telescope. The image is observed through the telescope.

BUCKING IN—To bring the telescope optical axis into alignment with two or more predetermined points.

CLAMP, SPHERICAL MOUNT—A clamping device which holds a spherical mount on the mounting base.

COLLIMATE—To bring into line; to make parallel.

COLLIMATION LINE—The correct line-of-sight or the optical axis of a telescope. Refer to line-of-sight definition.

COLLIMATOR, JIG—A transit type instrument used to establish vertical planes, also referred to as a jig transit.

COLLIMATOR, OPTICAL—A ground tubular instrument containing a displacement graticule, a diffusing lens, a colored filter, and a light source, used for determining deviations from the line-of-sight.

ERECTION LENS—A lens placed between the eye piece and objective lens of a telescope to give normal appearance of the object viewed. Without the erection lens the image would appear to be upside-down.

FOCUS—The point at which light rays converge after being refracted by a lens; also the act of adjusting an optical instrument so that the eye may see an image clearly.

GRATICULE, DISPLACEMENT—The graduated pattern of an optical target which is used to detect or measure horizontal and vertical displacement.

GRATICULE, TILT—The graduated pattern of an optical target which is used to detect and measure angular deviation.

INCIDENCE, ANGLE OF—The angle formed between a ray of light which strikes a surface and an imaginary perpendicular to the surface from that point.

LEVEL, OPTICAL (Sight Level)—A telescope with spirit level attached. The level is parallel to the optical axis.

LEVEL, PRISMATIC—A spirit level with a prism and mirror device for viewing the level bubble to give the appearance of two half-bubbles, which, when brought into coincidence, assures a level position.

LEVEL STRIDE—A spirit level mounted on vee-shaped legs; used for leveling tubular objects.

LINE-OF-SIGHT, BASIC—An optical reference line established by means of an alignment telescope and target mounted in a permanent location on the jig; the line-of-sight from which secondary or auxiliary lines-of-sight are determined.

LINE-OF-SIGHT, REMOTE—An optical reference line established by means of an alignment telescope and target mounted in a separate temporary location on instrument stands or on a tooling bar near the jig.

MONOCHROMATIC LIGHT—A light which gives off a single color light, all the rays of which are of approximately the same wave length.

MOUNTING BASE, ADJUSTABLE—A pedestal type base with a female cone for supporting spherical mounts. This base is normally mounted permanently to the tool being set optically.

MOUNTING BASE, FIXED—A pedestal-type base with a female cone for supporting spherical mounts. This base is nonadjustable and is normally mounted permanently to the tool being set optically.

OBJECTIVE LENS—The lens at the end of a telescope which is normally presented toward the object viewed; the lens which receives the light from the object.

OPTICAL FLAT—A section of fused quartz or glass, ground and polished flat. Opposite faces are parallel within close tolerances. It can be mounted in an optical instrument to displace the line-of-sight. It can be used to measure the flatness of smooth surfaces using the principal of interference of visible light.

OPTICAL MICROMETER—A calibrated device used to provide and measure refracted rays of light within certain limits.

OPTICAL MIRROR—A front surface mirror which is flat within extremely fine tolerances.

OPTICAL REFERENCE PLANE—A plane determined by any two lines-of-sight.

OPTICAL SQUARE—Any optical instrument which will turn the line-of-sight 90 degrees from its original path.

OPTICAL TOOLING—A tooling system using precision optical instruments to establish and maintain lines-of-sight as reference lines.

PARALLAX—The apparent movement of the cross hairs of an instrument across the image of the object being viewed when the eye is moved; caused by cross hairs and image not being in same focus.

PENTA-PRISM—A five-sided prism which bends a ray of light at a 90-degree angle.

PHYSI-OPTICAL—A mensuration or measurement system combining features of both optical and physical methods of measurements.

PLANIZE—To erect a perpendicular to a line-of-sight.

PLUNGE – To rotate jig transit telescope 180 degrees in vertical plane.

PRISM—A transparent body bounded in part by two plane surfaces which are not parallel.

REFLECTION, ANGLE OF—The angle between the line-of-sight and a line perpendicular to the surface of the reflecting medium.

REFRACTION—The bending of light rays passing obliquely through materials of different optical densities, i.e., air and water, air and glass.

RESOLUTION—The ability of a lens or an optical system to distinguish between two adjacent points. Resolution is often expressed in terms of the minimum angle between two points that can definitely be resolved or separated.

RETICLE OR RETICULE—The pattern of an optical target or cross hair used to establish alignment.

SPHERICAL MOUNT—A steel sphere bored through the center to permit the mounting of optical instruments on the axis of the sphere. Refer to Specification N.A.S. 900.

TARGET, OPTICAL TOOLING—A mounted graticule, reticle or other device for determining accurate positions when viewed by a telescope or other optical instrument.

THEODOLITE—An instrument identical in function to a transit.

TRANSIT—An instrument which permits the establishment of horizontal, vertical, or inclined lines-of-sight and capable of measuring angles between lines-of-sight in the horizontal and vertical planes. Normally used in surveying.

TRANSIT, JIG—A transit type instrument used to establish vertical planes.

1-333. SUPPORT OF STRUCTURE DURING REPAIR.

1-334. When planning a major repair requiring removal or replacement of a structural component, two important factors must be considered:

a. Support of the part to be repaired or replaced during working operations.

b. Support of adjacent structure subject to static loads when damaged component is removed.

The airplane can be supported adequately by the landing gear and/or jacks in cases where repair work is to be accomplished on removal components, except the wings. To remove or replace wings, use a cradle designed for support of fuselage structure as shown on figure 1-91. Removal components may be supported on sawbucks padded with small sandbags. The sandbags should measure about 8 by 12 inches with parallel seams spaced every 2 inches so the sand will readily conform to the contour of the component placed upon it. An adequate number of padded sawbucks and sandbags spaced from 3 to 5 feet apart to avoid possible distortion of the component involved should be employed under spars, bulkheads, or other main structural members of the component. For built-up repair support fixtures, refer to Sections II and IV.

1-335. MAJOR COMPONENTS — F-106A AND F-106B.

1-336. The F-106A and F-106B airplanes are broken into 29 major comoponents as shown on figures 1-92 and 1-93. Each of the major components is designated by name and by a Convair engineering blueprint number.

1-337. IN-SERVICE USE CRITERIA.

1-338. Refer to paragraphs 1-78 through 1-82.

1-339. WEIGHT AND CENTER OF GRAVITY.

1-340. General.

1-341. Gross weight and balance must be maintained within stipulated limits. Therefore, the weight of materials used to repair the airplane structure must be held at the minimum consistent with design requirements. Refer to Table 1-L for airplane and airplane component weights. It is important that accurate records of all repairs which affect weight and balance be maintained in accordance with T.O. 1-1B-40.

1-342. Basic Weight.

1-343. The basic weight of an airplane is that weight which includes all fixed operating equipment and trapped fuel and oil, to which it is necessary to add only the "variable" or "expendable" load items. Basic weight changes with structural modifications and may vary widely between airplanes of the same model. Basic weight plus the variable items which remain constant for a certain mission, including oil, crew and emergency equipment, is referred to as the operating weight.

1-344. Gross Weight.

1-345. The gross weight of an airplane is the total weight of the airplane including all variable items and modifications. Takeoff gross weight is the operating weight plus the expendable load items. Landing gross weight is takeoff gross weight less the expendable load items.

1-346. Moment.

1-347. Moment is the weight of an item multiplied by its arm. Arm of an item is the horizontal distance in inches from the reference datum and center of gravity of the item. Average arm is that distance obtained by adding the weights and moments of a number of items and by dividing the total moment by the total weight. Basic moment, when using data from an actual weighing of the airplane, is the total moment of the basic airplane in respect to the reference datum.

1-348. Reference Datum.

1-349. Reference datum is an imaginary vertical plane, at or forward of the nose of an airplane, from which all horizontal distances are measured for balancing purposes. The horizontal reference datum for F-106A and F-106B airplanes is located at station 0.0 which is 103.23 inches aft of the tip of the nose boom. The vertical reference datum for F-106A and F-106B airplanes is 100.00 inches below waterline 0.0.

1-350. Center of Gravity.

1-351. The center of gravity of an airplane is that point about which it would balance if suspended. The center of gravity distance from the reference datum is obtained by dividing the total moment by the gross weight of the airplane. Limits are the extremes of movement which the center of gravity may have without resulting in the airplane becoming unsafe to fly. The center of gravity of a loaded airplane must be within these limits at takeoff. Refer to Table 1-LI for airplane center of gravity information.

1-352. PACKING AND CRATING.

1-353. High-value components, such as the radome, fin tip, rudder, elevons, etc., must be packed and crated to prevent further damage when shipped to repair facilities. The crate or container in which the new part is received should be used for this purpose. Care must be exercised when serviceable components are removed from shipping crates. The correct disassembly sequence must be followed to prevent unnecessary damage to the crate, and all

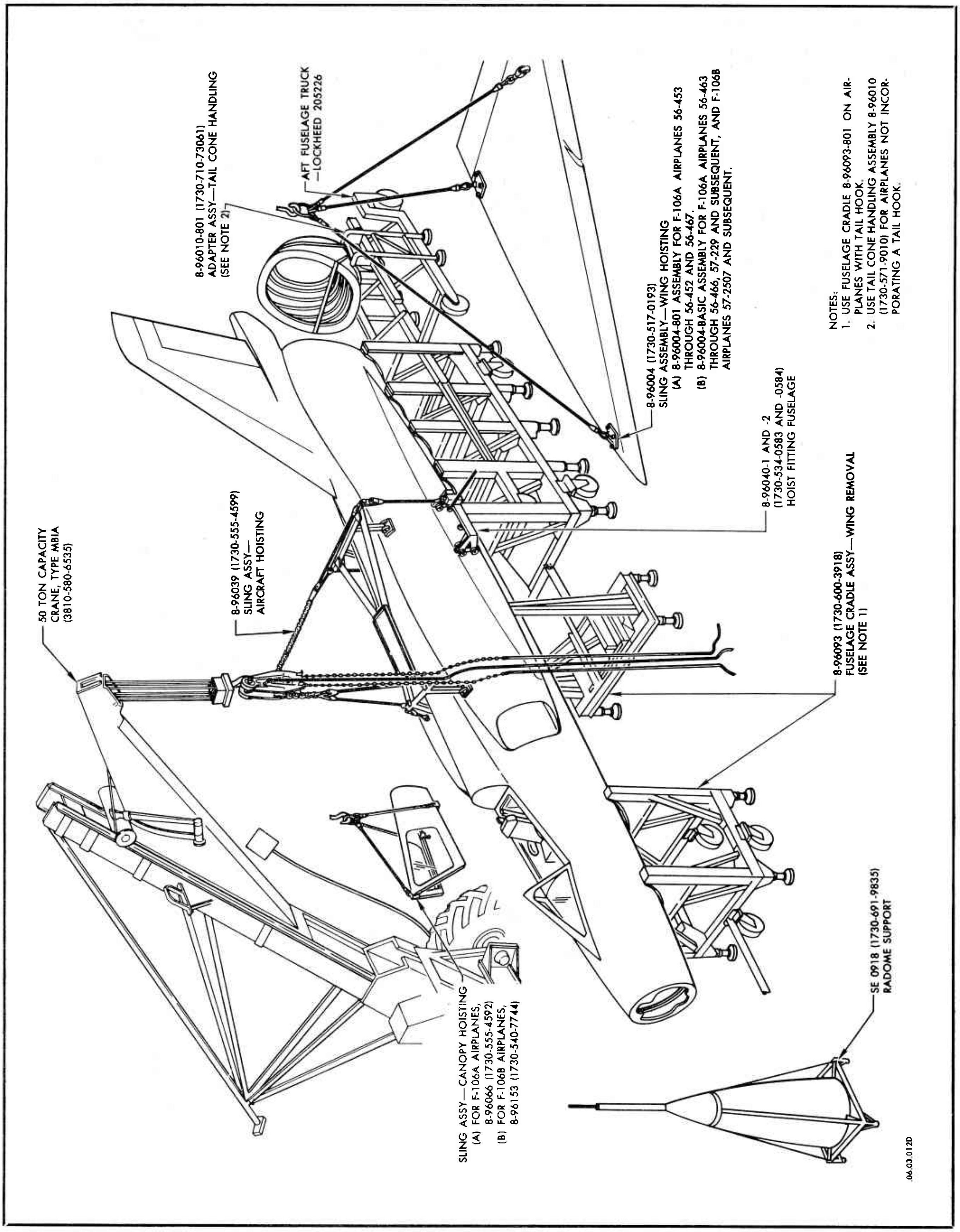


Figure 1-91. Airplane Handling Equipment

KEY	FIGURE NUMBER	NOMENCLATURE	ENGINEERING DRAWING NO.
1.	2-1	WING LEADING EDGE	8-18230
2.	2-1	WING FORWARD TANK	8-18030
3.	2-1	WING TIP	8-18910
4.	2-1	WING AFT TANK	8-18032
5.	2-1	WING TRANSFER TANK	8-18033
6.	2-1	ELEVON	8-13020
7.	3-1	FIN LEADING EDGE	8-14230
8.	3-1	FIN STRUCTURE	8-14035
9.	3-1	FIN TIP	8-14540
10.	3-1	RUDDER	8-15041
11.	4-1	SPEED BRAKE AND DRAG CHUTE HOUSING	8-74870
12.	4-1	TAIL CONE STRUCTURE	8-74490
13.	6-1	ENGINF SHROUD	8-22679
14.	4-1	FUSELAGE INSTALLATION	8-74470
15.	4-1	DORSAL FAIRING	8-74498
16.	4-1	DORSAL FAIRING	8-74497
			66J40244 (SEE NOTE 1)
			66J40245 (SEE NOTE 1)
17.	4-1	FUSELAGE INSTALLATION	8-74477
18.	4-1	VARIABLE RAMP INLET DUCT	8-74566
19.	5-1	MAIN LANDING GEAR FUSELAGE DOOR	8-44545
20.	4-1	FUSELAGE INTEGRAL FUEL TANK	8-74430
21.	4-1	FUSELAGE MISSILE BAY DOORS (SEE NOTE 2)	8-74463
21A.	4-1	FUSELAGE MISSILE BAY DOORS, GUN NOT INSTALLED (SEE NOTE 3)	7144550
21B.	4-1	FUSELAGE MISSILE BAY DOORS, GUN INSTALLED (SEE NOTE 3)	7144550
21C.	4-1	M61A1 GUN ENCLOSURE ASSEMBLY (SEE NOTE 3)	7144521
21D.	4-1	M61A1 GUN FAIRING/ASSEMBLY (SEE NOTE 3)	7144522
22.	4-1	FUSELAGE INSTALLATION	8-74405
23.	5-1	NOSE LANDING GEAR DOOR	8-74626
24.	5-1	NOSE LANDING GEAR	8-52003
25.	4-1	FORWARD RADAR ELECTRONICS COMPARTMENT	8-74600
26.	4-1	RADOME	8-70410
27.	4-1	WINDSHIELD AND CANOPY STRUCTURE	8-74800
28.	5-1	MAIN LANDING GEAR WING FAIRING	8-17630
29.	5-1	MAIN LANDING GEAR	8-51013

NOTES:

1. APPLICABLE AFTER INCORPORATION OF TCTO 1F-106-986
2. APPLICABLE PRIOR TO INCORPORATION OF TCTO 1F-106A-558
3. APPLICABLE AFTER INCORPORATION OF TCTO 1F-106A-558
4. TCTO 1F-106A-558 IS APPLICABLE TO F-106A VERTICAL INSTRUMENTED AIRCRAFT

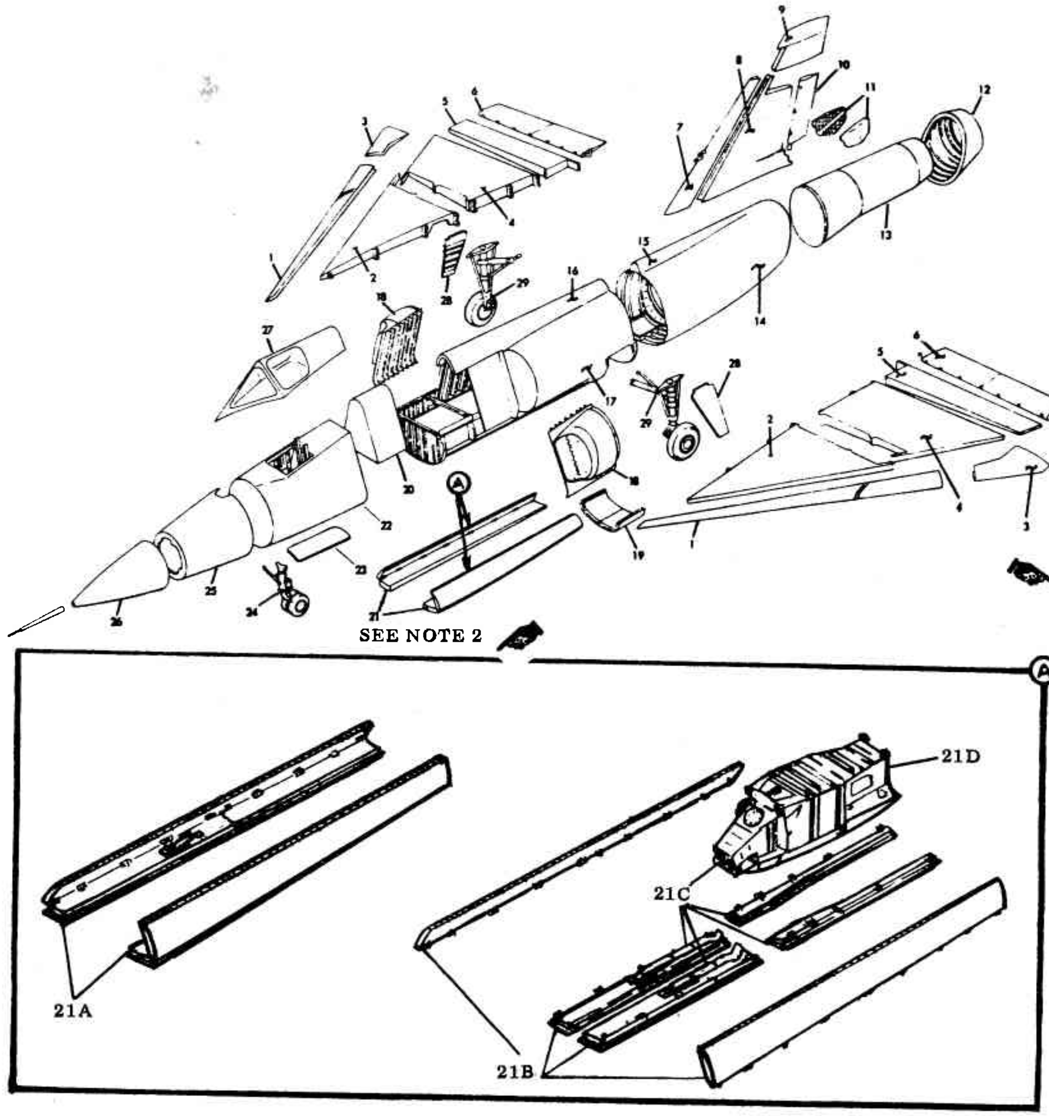
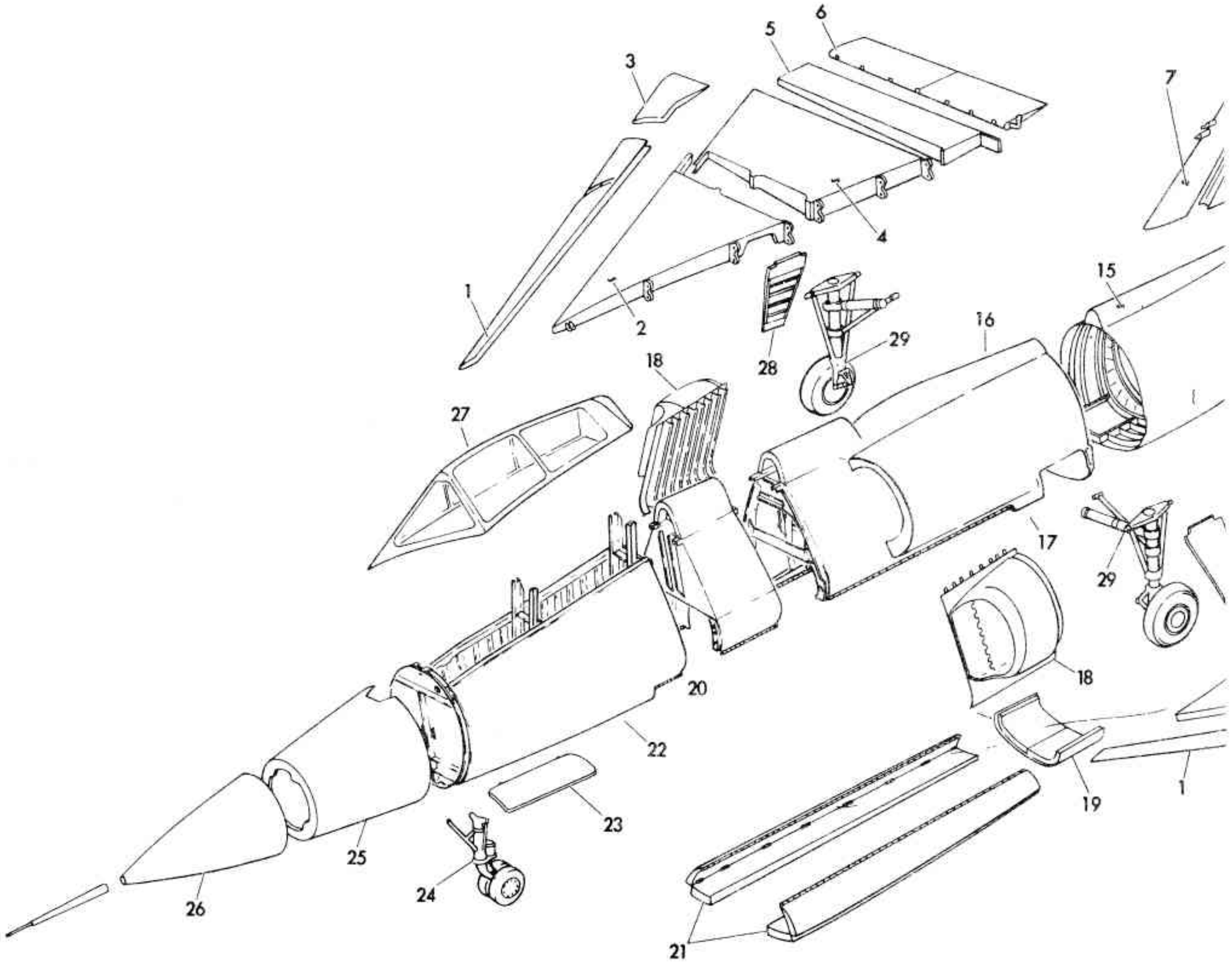


Figure 1-92. Major Components — F-106A



NOTE:
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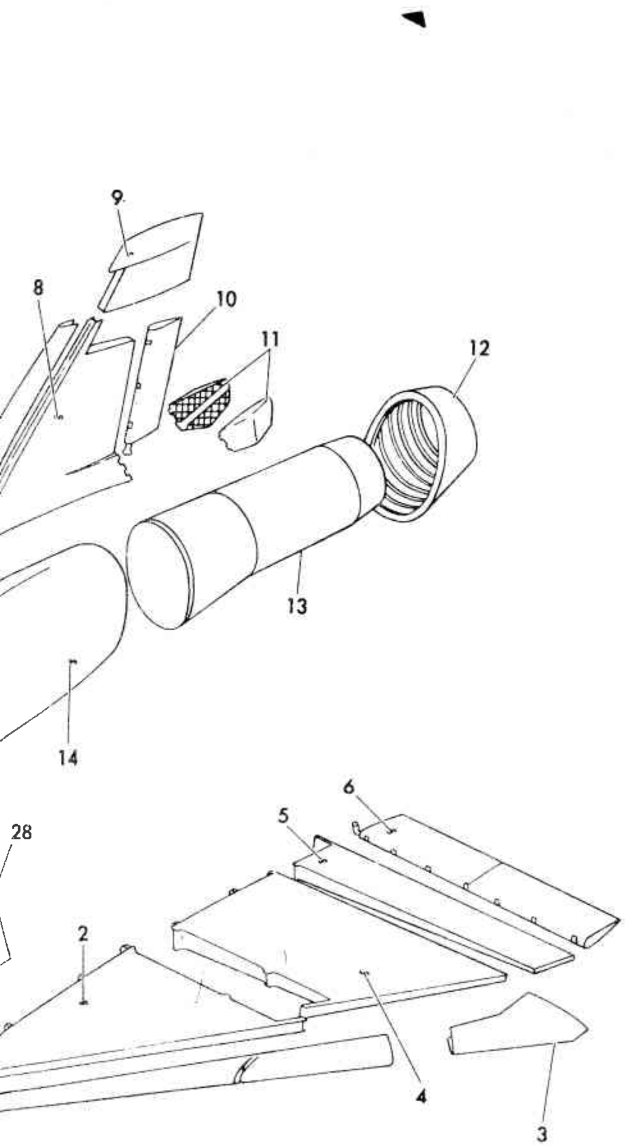


BLE AFTER INCORPORATION OF TCTO 1F-106-986.

KEY	FIGURE NUMBER	NOMENCLATURE	ENGINEERING DRAWING NO.
1.	2-1	WING LEADING EDGE	8-18230
2.	2-1	WING FORWARD TANK	8-18030
3.	2-1	WING TIP	8-18910
4.	2-1	WING AFT TANK	8-18032
5.	2-1	WING TRANSFER TANK	8-18033
6.	2-1	ELEVON	8-13020
7.	3-1	FIN LEADING EDGE	8-14230
8.	3-1	FIN STRUCTURE	8-14035
9.	3-1	FIN TIP	8-14540
10.	3-1	RUDDER	8-15041
11.	4-1	SPEED BRAKE AND DRAG CHUTE HOUSING	8-74870
12.	4-1	TAIL CONE STRUCTURE	8-74490
13.	6-1	ENGINE SHROUD	8-22679
14.	4-1	FUSELAGE INSTALLATION	8-74470
15.	4-1	DORSAL FAIRING	8-76501
16.	4-1	DORSAL FAIRING	8-76498 66J40240 (SEE NOTE)
17.	4-1	FUSELAGE INSTALLATION	8-76002
18.	4-1	VARIABLE RAMP INLET DUCT	8-74568
19.	5-1	MAIN LANDING GEAR FUSELAGE DOOR	8-44545
20.	4-1	FUSELAGE INTEGRAL FUEL TANK	8-76467
21.	4-1	FUSELAGE MISSILE BAY DOORS	8-74663
22.	4-1	FUSELAGE INSTALLATION	8-76003
23.	5-1	NOSE LANDING GEAR DOOR	8-74626
24.	5-1	NOSE LANDING GEAR	8-52003
25.	4-1	FORWARD RADAR ELECTRONICS COMPARTMENT	8-74600
26.	4-1	RADOME	8-70410
27.	4-1	WINDSHIELD AND CANOPY STRUCTURE	8-74800
28.	5-1	MAIN LANDING GEAR WING FAIRING	8-17630
29.	5-1	MAIN LANDING GEAR	8-51013

Figure 1-93. Major Components — F-106B

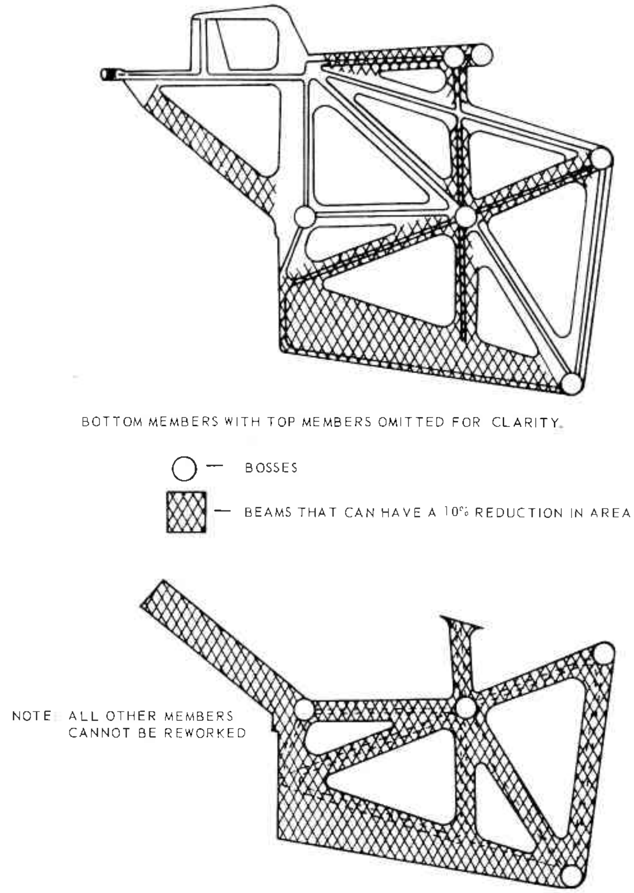
attaching parts and hardware held to recrate the old part. Unless otherwise specified, all shipping crates are marked in accordance with Military Standard MIL-STD-129. The words "Removable End—Reversible Crate" are stenciled on both ends. Heavy crates have the words "Sling Here," and "Centef of Balance" stenciled in their proper places. All bearings and bearing surfaces of components must be cleaned. Use a clean cloth and a solvent conforming to Federal Specification P-S-661 to remove dirt, metal chips, or other foreign substances. After cleaning, the lubricant normally used in service should be applied and the bearing or bearing surface covered with a greaseproof barrier material conforming to Military Specification MIL-B-121. Unless authorized to the contrary, the repairable component should always be packed and crated to the same configuration in which the serviceable component was received. Should repair or manufacture of crates be necessary, instructions and procedures provided in this manuel should be followed. These instructions are explained and illustrated at the end of the section in which the component is discussed. Should airlift of components be necessary, Table 1-LII provides dimensions, weights, and center of balance data for detached, uncrated components.

1-354. REWORK OF REPLACEMENT PARTS.

1-355. Some replacement parts and assemblies drawn from stock require a small amount of rework in order to fit. This is due to manufacturer's tolerances or engineering changes to the parts. Table 1-XLIX lists the part number, nomenclature and brief instructions for reworking or fitting these parts and assemblies. Parts are listed by part number numerical sequence.

1-356. Repair of Bellcrank Part Numbers 8-42694-5 and 8-42694-6.

1-357. Loose bushings in flight control bellcranks may be reworked by handwelding to reduce the bushing hole and remachining the hole to the original size. Heat treatment will be required per MIL-H-6857. This procedure may be used several times to accomplish repairs and will permit reuse of standard size bushings.



THE MEMBERS THAT MAY BE GROUND DOWN TO REMOVE CRACKS AND NICKS ARE INDICATED ON THE ABOVE SKETCH. NO MEMBER MAY BE GROUND DOWN MORE THAN TEN PERCENT OF ITS MINIMUM CROSS-SECTIONAL AREA. THE GRINDING MUST BE DONE IN SUCH A MANNER THAT THE REDUCTION IN CROSS-SECTIONAL AREA WILL BE GRADUAL ALONG THE LENGTH OF THE BEAM. ALL BOSSES IN THE MIXER FRAME MAY HAVE A REDUCTION IN CROSS-SECTIONAL AREA OF FIFTEEN PERCENT.

1-358. WELDING PROCESS CONTROLS.

1-359. Accepted welding practice shall be in accordance with T.O. 34W4-1-5, Welding Theory and Practice and/or T.O. 1-1A-9, Aerospace Metals - General Data and Usage Factors. If additional process details are needed, these are to be furnished to the welding shop in Process Orders in accordance with applicable AFLC directives.

1-360. Weld repair F-106 elevon (mixer) casting.

a. PROBLEM:

1. Surface defects are found on mixer castings after fluorescent penetrant inspection (zyglo).

2. Internal defects may also be present.

b. AUTHORIZE:

Rejected and condemned castings tagged as such will be processed and weld-repaired per instructions contained herein.

c. RESTRICTIONS:

1. The maximum number of weld repair areas per casting shall be four or less.
2. Only one repair area is allowed at each intersection, boss area, or leg strut.

d. MATERIAL IDENTIFICATION:

1. The minor casting is manufactured from magnesium alloy AZ91-HTA, QQ-M-56.

2. The casting classification is ZA, MIL-C-6021.

e. PROCESS:

1. Remove zyglo indication by hand filing or grinding. Periodically zyglo inspect to determine if the defect has been removed.

2. Metal removal shall be performed only on one side of an area. The maximum metal removal shall be one-half the thickness of the member.

CAUTION

Observe precautions as specified in T.O. 1-1A-9, Section IV, for grinding and subsequent welding for magnesium.

3. Clean area to be welded by stainless steel wire brush and vapor degreasing.

4. Locally preheat area at 400° -450° F with a neutral acetylene flame.

5. Weld repair the defective area, using TIG fusion process (heliarc).

(a) Use AZ92 filler rod per MIL-R-6944, and build-up sufficiently for clean up.

6. Stress releive immediately upon completion of one repaired area at 400° -450° F with torch.

7. Hand file or grind repair area to original configuration.

8. Apply a local "brush on" chromic acid treatment to the repaired areas, T.O. 42C2-1-7, Section XIV.

f. QUALITY STANDARD:

1. Radiographic inspection of the welded areas shall show no cracking.

2. The repaired casting shall have no warpage or distortion that would prevent proper installation.

1-361. Stylus plating elevon troque tube plug.

a. PROBLEM:

Some elevon torque tubes (part No. 8-42365 and part No. 8-48348) are unserviceable due to wear and gouges in the bearing surface of the plugs (part No. 8-23467). The defects in the aluminum base bearing surface can be eliminated by stylus plating as a means of restoring the plug to a serviceable condition. See paragraph g.3.

b. AUTHORIZE:

The following materials equipment and process details for stylus plating bearing surfaces of plugs (Part No. 8-42367).

c. MATERIALS.

1. Anodes, special high purity, high density graphite, (6810-201-7030).

2. Cotton batting, U.S.P. long fiber, lint free, surgical grade 1, (6510-201-7030).

3. Gauze, surgical tube, (6510-200-7010, 7015- and 7020).

4. Tape, electroplaters', (8135-833-7311).

5. "Electro Coated" abrasive paper, grit 120 through 600, (6350). No substitute. Available at the following compainies.

(a) 3M, Dri-Lube (Electro Coated) paper.

(b) Armour, Electrocoated paper.

(c) Bear Manning, no-load (Electro-Coated) paper.

6. Paper, filter, Whatman No. 41 and 41H, (6640 N/L).

7. Trichloroethylene, (0810--223-2731).

8. Freon TF solvent, (6830-082-2411).

9. Electrocleaning solution, (6810 N/L).

10. Activating solution No. 2, (6850 N/L).

11. Nickel acid stylus plating solution, (6810 N/L).

d. EQUIPMENT:

1. Stylus plating power unit and accessories, (3426-759-0187.)

2. Glass beakers, 100 to 400 cc, (6640 N/L).

3. Porcelain evaporating dishes, 30 to 200 cc capacity, (6640 N/L).

e. OPERATOR:

The stylus plating operator shall be trained in techniques of stylus plating in an approved course, and be certified by a Materials and Process Laboratory.

f. PROCESS:

1. Degrease in a trichloroethylene vapor degreaser.

2. Open up the affected area so that it is 3 to 5 times wider than it is deep, and remove anodize from the surface to be plated by some suitable mechanical means. The S.S. White abrasive blaster is an excellent tool for removing old anodic coatings, and for opening up surface defects. Measure minimum diameter of plug in opened up area. If minimum diameter is less than 2.00 inches, plug should be condemned.

3. Choose styli of suitable size and configuration for cleaning, activating and plating the defective surface. Wrap the styli appropriately in cotton batting and gauze.

4. Clean the surface electrolytically with electrocleaning solution. Use 12 volts direct current. Move the stylus at a medium rate during the electrocleaning step. Rinse thoroughly with cold tap water.

5. Treat with activating solution No. 2. Use 8 volts reverse current and move the stylus at a moderate rate until a gray color just forms on the surface being treated. Rinse thoroughly with cold tap water.

6. Swab with a stylus soaked in electroclean solution. Do not apply current. Rinse.

7. Swab thoroughly with a stylus soaked in nickel acid solution. Move the stylus at a moderate rate. Do not apply current.

8. Now apply current to the stylus used for swabbing in Step f.7 above and plate immediately with nickel acid solution. Use 10 volts direct current and moderate anode-cathode movement. Plate just until the full color of nickel is evident over the surface. Rinse thoroughly with cold tap water.

9. Use no load electrocoated abrasive paper or other suitable abrasive to abrade the nickel deposit for deposition of subsequent layers of nickel and to blend the nickel with the surrounding unplated area. If additional nickel is required for filling a defect, proceed at step f.11 thru f.12.

10. Clean electrolytically. See Step f.4. Rinse thoroughly with tap water.

11. Plate with nickel acid solution (See Step f.9) so that the diameter of the finished bearing surface is within the specification of $2.0630 \pm^{.0000}_{.0005}$. This will give a minimum of 0.0015 interference fit of plug-bearing and a maximum of 0.0010 loose fit.

12. Use electrocoated abrasive paper or other suitable abrasive to blend and finish the nickel plated plug to print specification.

g. INSPECTION:

1. Each stylus plated electrodeposit shall comply with the requirements of the respective specification listed in paragraph h.

2. The stylus plated electrodeposits shall not pull away when electroplaters' tape is applied with heavy hand pressure and pulled away suddenly at right angles to the plated surface. Any plate adhering to the tape shall be cause for rejection.

3. Defects, which penetrate to a depth exceeding 10% of the original part thickness, shall not be filled by stylus plating unless otherwise authorized.

h. REFERENCES:

1. Drawing No. 8-42367.

2. T.O. -42C2-1-7, Metal Treatments.

3. Federal Specification QQ-N-290, Nickel Plating (Electrodeposited).

TABLE 1-I

Material Substitution Table

Material to be Replaced	Ultimate Tensile Strength	SUBSTITUTE MATERIAL FOR SHEET STOCK AND EXTRUSIONS																		
		2024						7075			7178			Titanium				Stainless Steel		
								T6			T6			AMS						
		T3 Clad	T4 Clad	T4 Ex-truded	T6 Bare	T81 Clad	T86 Clad	Clad	Ex-truded	Bare	Clad	Ex-truded	Bare	4900	4901	4908	4925	¼ hard 301	½ hard 301	321 & 347 Cres
6061-T6 Extruded	38,000 PSI	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00							
AZ31A-H Magnesium	39,000 PSI	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00							
6061-T6 Clad	42,000 PSI	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00							
2024-T4 Extruded	57,000 PSI	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00							
2024-T4 Clad	58,000 PSI	1.00	1.00	1.02	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00							
2024-T3 Clad	60,000 PSI	1.00	1.04	1.05	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00							
2024-T6 Bare	62,000 PSI	1.03	1.07	1.09	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00							
2024-T81 Clad	64,000 PSI	1.07	1.10	1.12	1.03	1.00	1.00	1.00	1.00	1.00	1.00	1.00	1.00							
AMS 4900 Titanium														1.00	1.00	1.00	1.00	1.00	1.00	1.00
2024-T86 Clad	70,000 PSI	1.17	1.21	1.23	1.13	1.09	1.00	1.00	1.00	1.00	1.00	1.00	1.00							
7075-T6 Clad	72,000 PSI	1.20	1.24	1.27	1.16	1.13	1.03	1.00	1.00	1.00	1.00	1.00	1.00		1.00					
7075-T6 Bare	78,000 PSI	1.30	1.35	1.37	1.26	1.22	1.11	1.09	1.00	1.00	1.00	1.00	1.00		1.00					
7075-T6 Extruded	78,000 PSI	1.30	1.35	1.37	1.26	1.22	1.11	1.09	1.00	1.00	1.03	1.00	1.00		1.00					
7178-T6 Bare	84,000 PSI	1.40	1.45	1.48	1.36	1.32	1.20	1.17	1.08	1.08	1.10	1.00	1.00		1.05					
7178-T6 Extruded	84,000 PSI	1.40	1.45	1.48	1.36	1.32	1.20	1.17	1.08	1.08	1.10	1.00	1.00		1.05					
AMS 4901 Titanium	80,000 PSI													1.23	1.00	1.00	1.00	1.00	1.00	1.00
Type 321 and 347 CRES	100,000 PSI																	1.00	1.00	1.00
AMS 4908 Titanium	120,000 PSI													1.85	1.50	1.00	1.00	1.00	1.00	1.20
Type 301 Stainless steel, ¼ hard	125,000 PSI														1.56	1.04	1.00	1.00	1.00	1.25
AMS 4925 Titanium	140,000 PSI														1.75	1.17	1.00	1.12	1.00	1.40
Type 301 Stainless steel, ½ hard	150,000 PSI																1.07	1.20	1.00	1.50

a. Locate the material to be replaced on the line in the left hand column.

b. Locate the substitute material in the vertical columns.

c. To obtain the minimum thickness of the substitute material, multiply the thickness of the material to be replaced by the factor shown at the intersection of the line and column found in steps "a" and "b." Substitute standard gage equal to this thickness or nearest standard gage.

TABLE 1-IA

Sheet Metal Gages

Standard Gage	Superseded Standard Gage		
	Aluminum	Corrosion Resistant Steel	Steel
0.006 0.008 0.010 0.012			
0.016		0.015	
0.018 0.020 0.022 0.025 0.028			
0.032		0.030, 0.031	0.030
0.036		0.035	0.035
0.040		0.042	0.042
0.045		0.044	
0.050	0.051		
0.056	0.057		
0.063	0.064		0.062
0.071	0.072		0.072
0.080	0.081		0.078, 0.083
0.090	0.091	0.093	0.093
0.100	0.102		
0.112	0.114		0.109
0.125			
0.140	0.144		
0.160	0.156		0.156
0.180			
0.190	0.188		0.187
0.200			
0.224	0.218		

TABLE 1-II

Rivets Required Per Inch of Seam — Protruding Head Rivets or 100 Deg Countersunk Rivets in Dimpled Sheet

*2024-T3 Sheet Material or 2024-T4 Extrusions						
Type	AN426 and AN470 Rivets					
Rivets	AD				DD	
Diameter	3	4	5	6	6	8
Gage						
0.025	6.0	4.5				
0.032		4.5	3.5			
0.040		5.2	3.5	3.0	3.0	
0.051		6.6	4.3	3.0	3.0	2.3
0.064			5.4	3.0	2.9	2.1
0.072			6.1	4.2	3.1	2.1
0.081				4.8	3.5	2.1
0.091				5.3	3.9	2.3
0.102				6.0	4.4	2.6
0.125				7.3	5.4	3.2
0.156					6.7	4.0
0.188						4.6

*Also applicable to 2024-T6 and 6061-T6 materials.

NOTE: This table is computed by using a design factor of 80 percent of 2024-T3 alclad ultimate tensile strength.

*7075-T6 Sheet Material or 7075-T6 Extrusions						
Type	AN426 and AN470 Rivets					
Material	AD				DD	
Diameter	3	4	5	6	6	8
Gage						
0.025	7.2	4.6	3.8			
0.032		5.2	3.8			
0.040		6.9	4.1	3.1	3.1	2.3
0.051		8.2	5.4	3.7	3.1	2.3
0.064			6.7	4.6	3.4	2.3
0.072			7.6	5.2	3.8	2.3
0.081				6.9	4.3	2.5
0.091				6.6	4.8	2.8
0.102				7.4	6.6	3.2
0.125						3.8
0.156						4.8
0.188						5.5

*Also applicable to 2024-T81, 2024-T86 materials and AMS 4901 annealed titanium.

NOTE: This table is computed by using a design factor of 80 percent of 7075-T6 extrusion ultimate tensile strength.

TABLE 1-III

Rivets Required Per Inch of Seam — Protruding Head or 100 Deg Countersunk Rivets for 7178-T6 Material

7178-T6 bare sheet or 7178-T6 extrusions						
Type	AN470 Rivets					
Material	AD				DD	
Diameter	3	4	5	6	6	8
Gage						
0.025	8.0					
0.032	9.9	5.8				
0.040		7.0	4.7			
0.050		8.7	5.7	4.0	3.2	
0.063			7.1	4.9	3.6	2.1
0.071			8.0	5.6	4.0	2.3
0.080				6.2	4.6	2.6
0.090				7.0	5.2	2.9
0.100				7.8	5.7	3.2
0.125				9.7	7.1	4.0
0.160					9.1	5.1
0.190						6.0

7178-T6 bare sheet or 7178-T6 extrusions						
Type	AN426 Rivets					
Material	AD				DD	
Diameter	3	4	5	6	6	8
Gage						
0.025	10.8					
0.032	12.1	7.9				
0.040	14.0	8.8	6.5			
0.050		9.9	7.1	4.6	4.4	
0.063			8.1	6.0	4.8	3.3
0.071			8.8	6.5	5.1	3.4
0.080			9.6	7.0	5.4	3.5
0.090				7.7	5.9	3.7
0.100				8.2	6.3	3.9
0.125				9.9	7.4	4.5
0.160						5.4
0.190						6.1

NOTE: These tables are computed by using a design factor of 80 percent of 7178-T6 ultimate tensile strength.

TABLE 1-IV

Rivets Required Per Inch of Seam — 100 Deg Countersunk Head Rivets

***2024-T3 Sheet Material or 2024-T4 Extrusions**

Type	AN426 Rivets					
Rivets	AD				DD	
Diameter	3	4	5	6	6	8
Gage						
0.032	9.3	6.0				
0.040	10.5	6.5	4.8			
0.051		7.6	5.4	4.5	3.4	
0.064		8.9	6.6	4.9	3.7	2.5
0.072			6.7	5.3	3.9	2.6
0.081				5.7	4.1	2.7
0.091				6.2	4.6	2.8
0.102				6.3	4.8	3.0
0.125						3.5
0.156						3.9

*Also applicable to 2024-T6 and 6061-T6 materials.

NOTE: This table is computed by using the design factor of 80 percent of 2024-T3 alclad ultimate tensile strength.

***7075-T6 Sheet Material or 7075-T6 Extrusions**

Type	AN426 Rivets					
Material	AD				DD	
Diameter	3	4	5	6	6	8
Gage						
0.032	11.3	7.5				
0.040	13.1	8.3	6.1			
0.051		9.6	6.8	5.2	4.3	
0.064		11.6	7.5	5.1	4.5	3.1
0.072		12.8	8.6	6.3	4.9	3.3
0.081			9.1	6.7	5.2	3.4
0.091				7.3	5.5	3.5
0.102				7.8	6.1	3.8
0.125					7.0	4.3
0.156						4.8

*Also applicable to 2024-T81, 2024-T86 materials and AMS 4901 annealed titanium.

NOTE: This table is computed by using a design factor of 80 percent of 7075-T6 extrusion ultimate tensile strength.

TABLE 1-V

Rivets Required Per Inch of Seam — Protruding Head Blind Rivets

***2024-T3 Sheet Material or 2024-T4 Extrusions**

Type Rivets	Cherry Blind Rivets		
Diameter	4	5	6
Gage			
0.020	6.2		
0.025	7.1	4.5	
0.032	7.9	5.1	3.8
0.040	9.2	6.0	4.3
0.051		6.8	5.1
0.064		7.3	5.8
0.072			6.3
0.081			6.8

*Also applicable to 2024-T4 and 6061-T6 materials.

NOTE: This table is computed by using a design factor of 80 percent of 2024-T3 alclad ultimate tensile strength.

***7075-T6 Sheet Material or 7075-T6 Extrusions**

Type Rivets	Cherry Blind Rivets		
Diameter	4	5	6
Gage			
0.020			
0.025	8.3	5.7	
0.032	9.7	6.3	4.7
0.040	11.0	7.1	5.1
0.051	12.3	8.1	6.1
0.064		8.9	7.0
0.072			7.5
0.081			8.1

*Also applicable to 2024-T81, 2024-T86 materials and AMS 4901 annealed titanium.

NOTE: This table is computed by using a design factor of 80 percent of 7075-T6 extrusion ultimate tensile strength.

TABLE 1-VI

Rivets Required Per Inch of Seam — DuPont Aluminum Noiseless or Huck Blind Rivets

*2024-T3 Sheet Material or 2024-T4 Extrusions							
Type Rivets	Brazier Head		Dimple		100° Countersunk		
Diameter	4	5	4	5	4	5	6
Gage							
0.032	6.4	5.0	6.1	4.1			
0.040	7.2	5.6	5.9	3.9	10.0		
0.051	8.6	6.7			9.8	13.1	
0.064						9.3	8.4
0.072							7.0

*Also applicable to 2024-T6 and 6061-T6 materials.

NOTE: This table is computed by using a design factor of 80 percent of 2024-T3 alclad ultimate tensile strength.

*7075-T6 Sheet Materials or 7075-T6 Extrusions							
Type Rivets	Brazier Head		Dimple		100° Countersunk		
Diameter	4	5	4	5	4	5	6
Gage							
0.032	8.0	6.2	7.6	5.0			
0.040	9.0	7.0	7.4	4.9	12.4		
0.051	10.7	8.4			12.1	16.1	
0.064						11.2	10.0
0.072							8.4

*Also applicable to 2024-T81, 2024-T86 materials, and AMS 4901 annealed titanium.

NOTE: This table is computed by using a design factor of 80 percent of 7075-T6 extrusion ultimate tensile strength.

TABLE 1-VII

Rivets Required Per Inch of Seam — DuPont Nickel Noiseless Explosive Blind Rivets

*2024-T3 Sheet Material or 2024-T4 Extrusions							
Type Rivets	Brazier Head		Dimple		100° Countersunk		
Diameter	4	5	4	5	4	5	6
Gage							
0.032	5.1	4.0	3.8	2.6			
0.040	5.8	4.5	4.5	3.0	8.0		
0.051	6.9	5.4			7.8	10.0	
0.064						7.4	6.7
0.072							5.6

*Also applicable to 2024-T6 and 6061-T6 materials.

NOTE: This table is computed by using a design factor of 80 percent of 2024-T3 alclad ultimate tensile strength.

*7075-T6 Sheet Material or 7075-T6 Extrusions							
Type Rivets	Brazier Head		Dimple		100° Countersunk		
Diameter	4	5	4	5	4	5	6
Gage							
0.032	6.4	5.0	4.8	3.2			
0.040	7.2	5.6	5.6	3.7	9.9		
0.051	8.6	6.7			9.7	12.8	
0.064						9.0	8.0
0.072							6.7

*Also applicable to 2024-T81, 2024-T86 materials, and AMS 4901 annealed titanium.

NOTE: This table is computed by using a design factor of 80 percent of 7075-T6 extrusion ultimate tensile strength.

TABLE 1-VIII

Rivets Required Per Inch of Seam — Countersunk Head High-Strength Fasteners for 2024-T3 and 2024-T4 Materials

Type Rivets	Corrosion Resistant Steel			Monel			Huck Lockbolts Steel		Hi-Shear Rivets AN Bolts	
Diameter	5	6	8	5	6	8	6	8	6	8
Gage										
0.051	5.6	4.8		5.6	4.8		4.8		4.8	
0.064	5.7	4.8		5.7	4.8		4.8		4.8	
0.072	5.6	4.5		8.5	4.5		4.5		4.5	
0.081	5.6	4.5		8.4	6.7		4.5		4.5	
0.091	5.6	4.7		8.4	6.9		4.7		4.7	
0.102	5.8	5.1		8.7	7.6		5.1		5.1	
0.125	6.6	4.8	4.0	9.9	7.2	6.0	4.8	4.0	4.8	4.0
0.156	8.0	5.4	3.6	12.0	8.1	5.4	5.4	3.6	5.4	3.6
0.188	9.8	6.3	4.4		9.4	6.6	6.3	4.4	6.3	4.4

NOTES: A. This table also applicable to 2024-T6 and 6061-T6 materials.

B. This table is computed by using a design factor of 80 percent of 2024-T3 alclad ultimate tensile strength.

TABLE 1-IX

Rivets Required Per Square Inch of Seam — Protruding Head or 100 Deg Countersunk Head High-Strength Fasteners in Dimpled Sheet for 2024-T3 and 2024-T4 Materials

Type Rivets	Corrosion Resistant Steel				Monel				Huck Lockbolts Steel		Hi-Shear Rivets AN Bolts	
Diameter	4	5	6	8	4	5	6	8	6	8	6	8
Gage												
0.040	3.5	2.9			4.5	3.0						
0.051	3.5	2.9	2.4		4.4	2.9	2.4		2.4		2.4	
0.064	3.5	2.9	2.5	1.9	4.7	3.1	2.5	1.9	2.5	1.9	2.5	1.9
0.072		3.0	2.5	1.9		3.5	2.5	1.9	2.5	1.9	2.5	1.9
0.081		3.0	2.5	1.9		3.9	2.7	1.9	2.5	1.9	2.5	1.9
0.091		3.2	2.5	1.9		4.4	3.1	1.9	2.5	1.9	2.5	1.9
0.102		3.6	2.5	1.9		5.0	3.5	1.9	2.5	1.9	2.6	1.9
0.125			3.0	1.9			4.2	2.3	2.6	1.9	3.2	1.9
0.156				2.1				2.4	3.1	1.9	4.0	2.2
0.188				2.4				3.5		2.0		2.6

NOTES: A. This table also applicable to 2024-T4 and 6061-T6 materials.

B. This table is computed by using a design factor of 80 percent of 2024-T3 alclad ultimate tensile strength.

TABLE 1-X

Rivets Required Per Inch of Seam — Countersunk Head High-Strength Fasteners for 7075-T6 Material and AMS 4901 Titanium

Type Rivets	Corrosion Resistant Steel			Monel			Huck Lockbolts Steel		Hi-Shear Rivets AN Bolts	
Diameter	5	6	8	5	6	8	6	8	6	8
Gage										
0.051	4.6	4.0		4.6	4.0		4.0		4.0	
0.064	4.7	4.0		4.7	4.0		4.0		4.0	
0.072	4.6	3.7		4.6	3.7		3.7		3.7	
0.081	4.4	3.7		6.6	3.7		3.7		3.7	
0.091	4.5	3.9		6.7	3.9		3.9		3.9	
0.102	4.7	4.2		6.8	6.3		4.2		4.2	
0.125	5.4	4.0	3.3	7.1	6.0	3.3	4.0	3.3	4.0	3.3
0.156	6.7	4.4	2.9	10.1	6.6	4.4	4.4	2.9	4.4	2.9
0.188	8.0	5.2	3.6	12.0	7.8	5.4	5.2	3.6	5.2	3.6

NOTES: A. This table also applicable to 2024-T6, 2024-T81, and 2024-T86 materials.

B. This table is computed by using a design factor of 80 percent of 7075-T6 extrusion ultimate tensile strength.

TABLE 1-XI

Rivets Required Per Inch of Seam — Protruding Head or 100 Deg Countersunk Head High-Strength Fasteners in Dimpled Sheet for 7075-T6 Material and AMS 4901 Titanium

Type Rivets	Corrosion Resistant Steel				Monel				Huck Lockbolts Steel			Hi-Shear Rivets AN Bolts		
Diameter	4	5	6	8	4	5	6	8	6	8	10	6	8	10
Gage														
0.051	3.5	2.9	2.4		3.5	2.9	2.4		2.5			2.5		
0.064	4.1	2.9	2.4	1.8	4.1	2.9	2.4	1.8	2.5	1.9		2.5	1.9	
0.072	4.7	2.9	2.4	1.8	4.7	2.9	2.4	1.8	2.5	1.9	1.5	2.7	1.9	
0.081		3.4	2.4	1.8		5.1	2.4	1.8	2.5	1.9	1.5	3.6	1.9	1.5
0.091		3.8	2.4	1.8		5.7	4.5	1.8	2.5	1.9	1.5	4.0	1.9	1.5
0.102			3.0	1.8			5.5	1.8	2.5	1.9	1.5	4.5	1.9	1.5
0.125			3.7	2.1			6.9	3.2	3.0	1.9	1.5	5.4	1.9	1.5
0.156			4.6	2.6				3.9	3.8	2.1	1.7	6.8	2.2	1.5
0.188				3.1				4.5		2.6	2.0		2.7	1.7
0.210				3.4				5.1		2.9	2.2		3.2	1.8

NOTES: A. This table also applicable to 2024-T6, 2024-T81, and 2024-T86 materials.

B. This table is computed by using a design factor of 80 percent of 7075-T6 extrusion ultimate tensile strength.

TABLE 1-XII

Rivets Required Per Inch of Seam — Countersunk Head High-Strength Fasteners for 7178-T6 Material

Type Rivets	Corrosion Resistant Steel			Monel			Huck Lockbolts Steel		Hi-Shear Rivets AN Bolts	
Diameter	5	6	8	5	6	8	6	8	6	8
Gage										
0.051	5.0	4.3		5.0	4.3		4.3		4.3	
0.064	5.1	4.3		5.1	4.3		4.3		4.3	
0.072	5.0	4.0		5.0	4.0		4.0		4.0	
0.081	4.8	4.0		7.1	4.0		4.0		4.0	
0.091	4.9	4.2		7.2	4.2		4.2		4.2	
0.102	5.1	4.5		7.3	6.8		4.6		4.6	
0.125	5.8	4.3	3.6	7.7	6.5	3.6	4.3	3.6	4.3	3.6
0.156	7.2	4.8	3.1	10.9	7.1	4.8	4.8	3.1	4.8	3.1
0.188	8.6	5.6	3.9	12.9	8.4	5.9	5.6	3.9	5.6	3.9

NOTE: This table is computed by using a design factor of 80 percent of 7178-T6 ultimate tensile strength.

TABLE 1-XIII

Rivets Required Per Inch of Seam — Protruding Head High-Strength Fasteners in Titanium and Steel

Applicable to AMS 4901 and AMS 4908 ½-Hard Titanium and ¼-Hard Corrosion Resistant Steel

Type Rivets	Corrosion Resistant Steel				Monel				Huck Lockbolts Steel		Hi-Shear Rivets AN Bolts	
Diameter	4	5	6	8	4	5	6	8	6	8	6	8
Gage												
0.016	5.2	4.2			5.2	4.2						
0.020	5.2	4.2			5.2	4.2						
0.025	5.2	4.2			5.2	4.2						
0.032	5.2	4.2			5.2	4.2						
0.040	5.2	4.2	3.5		6.2	4.2	3.5		3.6		3.6	
0.051	5.2	4.2	3.6	2.7	7.9	5.2	3.6	2.7	3.7	2.8	3.7	2.8
0.064	6.6	4.3	3.6	2.7	9.9	6.5	4.5	2.7	3.7	2.8	3.7	2.8
0.072		4.9	3.6	2.7		7.3	5.1	2.8	3.7	2.8	3.7	2.8
0.081		5.5	3.8	2.8		8.2	5.7	3.2	3.7	2.8	4.0	2.8
0.091			4.2	2.8			6.4	3.5	3.7	2.8	4.4	2.8

NOTES: A. Also applicable to 100 Deg countersunk head high-strength fasteners in dimpled sheet.

B. This table is computed by using a design factor of 80 percent of 301 corrosion resistant steel.

TABLE 1-XIV

Rivets Required Per Inch of Seam — 100 Deg Countersunk Head High-Strength Fasteners in Titanium and Steel

Applicable to AMS 4901 and AMS 4908 ½-Hard Titanium and ¼-Hard Corrosion Resistant Steel

Type Rivets	Corrosion Resistant Steel			Monel			Huck Lockbolts Steel		Hi-Shear Rivets AN Bolts	
Diameter	5	6	8	5	6	8	6	8	6	8
Gage										
0.040	7.5			7.5						
0.051	7.5	6.3		7.5	6.3		6.3		6.3	
0.064	7.5	6.3		11.2	6.3		6.3		6.3	
0.072	7.2	6.2		10.8	6.2		6.2		6.2	
0.081	7.1	6.0		10.6	8.0		6.0		6.0	
0.091	7.1	6.1		10.6	8.1		6.1		6.1	
0.102	7.5	6.5	4.8	11.2	9.7	7.2	6.5	4.8	6.5	4.8
0.125	8.6	6.5	5.2	12.9	9.7	7.9	6.5	5.2	6.5	5.2
0.156	9.4	7.1	5.5	14.1	10.6	8.2	7.1	5.5	7.1	5.5

NOTE: This table is computed by using a design factor of 80 percent of 301 corrosion resistant steel.

TABLE 1-XV

Rivets Required Per Inch of Seam — 5056 Rivets in Magnesium

Protruding Head Rivets in AZ31A—Magnesium

Type Rivets	5056			
Diameter	4	5	6	8
Gage				
0.016	4.2	3.4	2.8	2.1
0.020	4.2	3.4	2.8	2.1
0.025	4.2	3.4	2.8	2.1
0.032	4.2	3.4	2.8	2.1
0.040	4.2	3.4	2.8	2.1
0.051	4.6	3.4	2.8	2.1
0.064	5.7	3.7	2.8	2.1
0.072	6.4	4.2	2.9	2.1
0.081	7.2	4.7	3.3	2.1
0.091	8.1	5.3	3.7	2.1
0.102	9.1	6.0	4.1	2.3

Countersunk Head Rivets in AZ31A — Magnesium

Type Rivets	5056		
Diameter	4	5	6
Gage			
0.016			
0.020			
0.025			
0.032			
0.040	5.0	3.5	3.4
0.051	5.1	3.7	4.0
0.064	6.2	4.2	3.9
0.072	6.9	5.2	3.9
0.081			
0.091			
0.102			

NOTE: This table is computed by using a design factor of 80 percent of AZ31A magnesium sheet ultimate tensile strength.

TABLE 1-XVI.

Rivets Required Per Inch of Seam—Straylor Fuel-Tight Rivets in 2024-T86 Machined Skins

Type Rivets	Counterbore	Seal Head
Diameter	6	8
Gage		
0.114	5.2	
0.125	5.7	
0.168	7.7	
0.200		5.1
0.250		6.4
0.312		8.0

NOTE

When a repair is required in an area where Straylor rivets are used, the existing rivet pattern should be picked up. Should the extent of the damage be of such a serious nature that a new rivet pattern is required, the above values may be used. The rivet pitch for staggered rows should be 4D and row spacing 2½D. For even or unstaggered rows the rivet pitch and row spacing should be 3D. For extensive repairs, engineering should be consulted. This table is computed by using a design factor of 80% of 2024-T86 ultimate tensile strength.

TABLE 1-XVII.

Rivets Required Per Inch of Seam—Straylor Fuel-Tight Rivets in 7075-T6 Machined Skins

Type Rivets	Counterbore	Seal Head
Diameter	6	8
Gage		
0.114	6.1	
0.125	6.6	
0.156	8.2	
0.218		6.8
0.250		7.8
0.312		9.8

NOTE

When a repair is required in an area where Straylor rivets are used, the existing rivet pattern should be picked up. Should the extent of the damage be of such a serious nature that a new rivet pattern is required, the above values may be used. The rivet pitch for staggered rows should be 4D and row spacing 2½D. For even or unstaggered rows the rivet pitch and row spacing should be 3D. For extensive repairs, engineering should be consulted. This table is computed by using a design factor of 80% of 7075-T6 ultimate tensile strength.

TABLE 1-XVIII.

Minimum Rockwell Hardness Acceptance Values for Metal

Material	Type	Rockwell Scale "A" Min	Rockwell Scale "B" Min	Rockwell Scale "E" Min
Steel	17-4PH	70		
	17-7PH	70		
	4130, H.T. 90,000 PSI	61.5	99	
	4130, H.T. 125,000 PSI	64	104	
	4130, H.T. 150,000 PSI	70		
	301		81	
	302	64	104	
	321	70		
	347	74.1		
Aluminum	2024-T4		74	95
	7075-T6		82	103
	7079-T6		81	106
	2014-T6		74	103
	7178-T6		86	109
	6061-T6		46	85
	2024-T86		81	
	2024-T81		73	
	2024-T3		74	95
	356-T6		29	
	5154-H34		31	
	2024-T80		71	
Magnesium	AZ 91-T6			72
	AZ 92-T6			72
	ZK 51-T5			60
Titanium	Commer. Pure	57		
	A 110-AT	67		
	C 130-AM	65.8		

TABLE 1-XIX

Standard Bend Radii for Flat Sheet

	MATERIAL	MATERIAL CONDITION	FORMING TEMP.	GAUGE																
				.008	.012	.016	.020	.025	.032	.040	.050	.063	.071	.080	.090	.100	.125	.160	.190	.250
ALUMINUM	3003-0, 5052-0, 6061-0	NOTED	ROOM		.03	.03	.03	.03	.03	.06	.06	.06	.09	.09	.12	.12	.16	.16	.19	.25
ALUMINUM	2014-0, 2024-0, 5052-H34, 6061-T4, 7075-0, 7178-0	NOTED	ROOM		.03	.03	.03	.03	.06	.06	.06	.09	.09	.12	.16	.16	.19	.25	.31	.44
ALUMINUM	2014-T4, 2024-T3, 2024-T4 6061-T6, 7075-W, 7178-W	NOTED	ROOM		.03	.03	.03	.06	.09	.09	.12	.16	.19	.22	.25	.31	.38	.44	.69	1.00
ALUMINUM	2014-T6, 2024-T36, 7075-T6, 7178-T6	NOTED	ROOM		.06	.06	.08	.09	.12	.19	.25	.31	.38	.44	.50	.56	.69	.88	1.00	1.50
ALUMINUM	2024-W	NOTED	ROOM		.03	.03	.03	.03	.06	.06	.09	.12	.12	.16	.18	.22	.25	.31	.44	.69
ALUMINUM	7075-T6	NOTED	HOT 300°F ± 25°		.06	.06	.06	.06	.06	.06	.09	.14	.15	.15	.18	.22	.31	.38	.50	.69
ALUMINUM	7178-T6	NOTED	HOT 275°F ± 25°		.06	.06	.06	.06	.06	.06	.09	.14	.15	.15	.18	.22	.31	.38	.50	.69
MAGNESIUM	AZ31A(SPEC. QQ-M-44)	A	HOT 400°F - 625°F			.03	.06	.06	.06	.09	.09	.12	.16	.19	.19	.22	.25	.31	.38	.50
MAGNESIUM	AZ31A(SPEC. QQ-M-44)	A	ROOM			.09	.09	.12	.16	.19	.25	.31	.38	.44	.44	.50	.62	.75	1.00	1.25
MAGNESIUM	AZ31A(SPEC. QQ-M-44)	H	HOT 325°F ± 25°			.06	.09	.12	.12	.22	.25	.38	.44	.44	.44	.44	.50	.62	.81	1.00
MAGNESIUM	AZ31A(SPEC. QQ-M-44)	H	ROOM			.19	.19	.25	.31	.38	.50	.62	.81	.81	.88	1.00	1.25	1.50	2.00	2.50

	MATERIAL	MATERIAL CONDITION	FORMING TEMP.	GAUGE																
				.008	.012	.016	.020	.025	.032	.036	.040	.050	.063	.080	.090	.112	.125	.160	.190	.250
STEEL	1025		ROOM	.03	.03	.03	.06	.06	.06	.06	.09	.09	.12	.16	.19	.22	.25	.31	.38	.50
STEEL	4130	65,000 TO 90,000 PSI	ROOM	.03	.03	.03	.06	.06	.06	.06	.09	.09	.12	.16	.19	.22	.25	.31	.38	.50
STEEL	4130	90,000 TO 125,000 PSI	ROOM	.03	.03	.06	.06	.09	.09	.12	.12	.16	.19	.25	.31	.38	.44	.50	.62	.75
STEEL	CORROSION RESISTANT	ANNEALED	ROOM	.03	.03	.03	.03	.03	.03	.03	.03	.06	.06	.09	.12	.12	.16	.16	.19	.25
STEEL	CORROSION RESISTANT	1/4 HARD	ROOM	.03	.03	.03	.03	.06	.06	.06	.06	.09	.09	.12	.16	.19	.22	.25	.31	.38
STEEL	CORROSION RESISTANT	1/2 HARD	ROOM	.03	.03	.06	.06	.09	.09	.12	.12	.16	.16	.19	.25	.31	.38	.50	.62	.75
STEEL	CORROSION RESISTANT	FULL HARD	ROOM	.06	.09	.09	.12	.16	.16	.16	.19	.19	.25	.31	.38	.44	.50	.62	.75	1.00
TITANIUM	COMMERCIALLY PURE AMS 4901	ANNEALED	ROOM	.03	.03	.06	.06	.09	.09	.12	.12	.16	.19	.25	.31	.38	.38	.44	.56	.75
TITANIUM	ALLOY AMS 4908	ANNEALED	ROOM	.03	.03	.06	.06	.09	.09	.12	.12	.16	.19	.25	.31	.38	.38	.44	.56	.75
TITANIUM	ALLOY CONVAIR SPEC. 0-01014	ANNEALED	ROOM			.09	.09	.12	.16	.19	.19	.25	.31	.38	.44	.56	.62	.78	.94	1.25

TABLE 1-XX

Standard Bend Radii for Aluminum Alloy Tubing

TUBE O. D.	5052-0		6061-T6 & 2024-T4
	STANDARD	*MINIMUM	STANDARD
1/8	0.38	0.38	0.50
3/16	0.75	0.44	0.75
1/4	0.75	0.56	1.00
5/16	1.00	0.69	1.25
3/8	1.25	0.94	1.50
7/16	1.50	1.25	1.75
1/2	1.50	1.25	2.00
5/8	2.00	1.50	2.50
3/4	2.25	1.75	3.00
7/8	2.50	2.00	3.50
1	3.00	3.00	4.00
1 1/8	3.50	3.50	4.50
1 1/4	3.75	3.75	5.00
1 3/8	5.00	5.00	6.00
1 1/2	5.00	5.00	7.00
1 5/8	6.00	6.00	8.00
1 3/4	7.00	7.00	9.00
2	8.00	8.00	10.00
2 1/4	9.00	9.00	11.00
2 1/2	10.00	10.00	13.00
2 3/4	11.00	11.00	14.00
3	12.00	12.00	15.00

*The limits listed for 5052-0 aluminum are also applicable to annealed or ¼ hard corrosion resistant steel.

TABLE 1-XXI

Temper Designations—Old and New for Aluminum Alloys

TEMPER DESIGNATION Old	New	ALLOY Sheet	Extruded	TREATMENT AND CONDITIONS
-F	-F			As fabricated: applies to various aluminum alloy products which acquire some temper qualities in the shaping or manufacturing processes.
-O	-O			Annealed, recrystallized: applies to the softest temper of wrought products.
-W	-W			Solution heat treated: unstable temper.
-T	-T			Treated to produce stable tempers other than -F and -O: applies to products treated to produce stable tempers with or without strain hardening. -T followed by numerals, designates a specific combination of basic operations.
24S-T3	2024-T3	2024-T3		Solution heat treated and then strain hardened.
24S-T4	2024-T4	2024-T4	2024-T4	Solution heat treated and naturally aged to a substantially stable condition.
24ST-80	2024-T6	2024-T6		Solution heat treated and then artificially aged.
24S-RT	2024-T36	2024-T36		Strain hardened by manufacturer.
24S-T81	2024-T81	2024-T81		2024-T3 plus artificial aging.
24S-T86	2024-T86	2024-T86		2024-T36 plus artificial aging.
61S-T6	6061-T6	6061-T6	6061-T6	Solution heat treated and then artificially aged.
75S-T6	7075-T6	7075-T6	7075-T6	Solution heat treated and then artificially aged.
XA78S-T6	7178-T6	7178-T6	7178-T6	Solution heat treated and then artificially aged.

NOTE: Aluminum alloys are designated by numbers and letters rather than names. The number preceding the letter indicates the composition of the alloy. For heat treated materials, the basic temper designation consists of a letter-T followed by one or more numbers which specifically define the type of treatment and conditioning used. See Table 1-I for material strength comparisons.

TABLE 1-XXII
Rivet Substitution

Nominal Fastener Diameter (Inches)	Head Config-uration	Order of Rivets By Increasing Strength	Rivets (See Notes)							
			Blind Rivets (See Note 4)	Standard Aluminum Rivets	Blind Bolt	7075-T6 Huck Lock Rivets	Monel	Cres. 302	Hi-Shear (Steel)	Steel Huck Lock Bolt
3/32	Protruding	1.0		MS20470A						
		2.0		(AN470AD						
		2.1		(MS20470AD						
		3.0	6951							
		4.0	06951							
		5.0	7951							
	Flush	1.0		MS20246AD						
		2.0	6950							
		3.0	06950							
		4.0	7950							
1/8	Protruding	1.0	(MS20600B	AN470B						
		1.1	(NAS1389B							
		1.2	(MS20602B							
		2.0	(MS20600AD	(AN470AD						
		2.1	(MS20602AD	(MS20470AD						
		2.2	(NAS1398D							
		3.0	(MS20600MP				(NAS508M			
		3.1	(NAS1398MW				(MS20615M			
		3.2	(MS20602MP				(MS20615MP			
		4.0	6951				(Q4310			
		4.1					(MS20613C			
		5.0	7951						NAS529	

TABLE 1-XXII
Rivet Substitution (Cont)

Nominal Fastener Diameter (Inches)	Head Config-uration	Order of Rivets By Increasing Strength	Rivets (See Notes)							
			Blind Rivets (See Note 4)	Standard Aluminum Rivets	Blind Bolt	7075-T6 Huck Lock Rivets	Monel	Cres. 302	Hi-Shear (Steel)	Steel Huck Lock Bolt
	Flush	1.0	MS20601B	(AN426B						
		1.1	(NAS1399B	(MS20426B						
		1.2	(MS20603B							
		2.0	(MS20601AD	(AN426AD						
		2.1	(MS20603AD	(MS20426AD						
		2.2	(NAS1399D							
		3.0	(MS2060MP				MS20427M			
		3.1	(NAS1399MP							
		3.2	(MS20603MP							
		4.0	6950					(Q4304 (MS20427F		
		5.0	7950					NAS525		
	Protruding	1.0	(MS20600B	AN470B						
		1.1	(NAS1398B							
		1.2	(MS20602B							
		2.0	(MS20600AD	(AN470AD		(ALSF-E				
		2.1	(MS20602AD	(MS20470AD		(ALPPH-E				
		2.2	(NAS1398D							
		3.0	(MS20600MP				(NAS508M			
		3.1	(NAS1398MW				(MS20615M			
		3.2	(MS20602MP				(MS20615MP			
		4.0	6951					(Q4310		
		4.1						(MS20613C		

TABLE 1-XXII
Rivet Substitution (Cont)

Nominal Fastener Diameter (Inches)	Head Configuration	Order of Rivets By Increasing Strength	Rivets (See Notes)							
			Blind Rivets (See Note 4)	Standard Aluminum Rivets	Blind Bolt	7075-T6 Huck Lock Rivets	Monel	Cres. 302	Hi-Shear (Steel)	Steel Huck Lock Bolt
5/32 thru 1/4	Protruding	5.0			NAS1669					
		6.0	7951		BB-352					(ALPPH-T
		6.1								(ALSF-T
5/32 thru 1/4	Flush	1.0	(MS20601B	(AN426B						
		1.1	(NAS1399B	(MS40426B						
		1.2	(MS20603B							
		2.0	(MS20601AD	(AN426AD		(ACT509-E				
		2.1	(MS20603AD	(MS20426AD		(ALP426H-E				
		2.2	(NAS1399D							
		3.0	(MS20601MP		NAS1674		(MS20427M			
		3.1	(NAS1399MW				(AN427MC			
		3.2	(MS20603MP							
		4.0	6950					(Q4304		
		4.1						(MS40427F		
		5.0			NAS1670					
		6.0	7950		BB-351				NAS525	SAL-100
		6.1			MS90353					ASCT509-T
		6.2								ALP426HT
5/16	Protruding	1.0				(ALSF-E				
		1.1				(ALPPH-E				
		2.0	6951							

TABLE I-XXII
Rivet Substitution (Cont)

Nominal Fastener Diameter (Inches)	Head Config-uration	Order Rivets By Increasing Strength	Rivets (See Notes)							
			Blind Rivets (See Note 4)	Standard Aluminum Rivets	Blind Bolt	7075-T6 Huck Lock Rivets	Monel	Cres. 302	Hi-Shear (Steel)	Steel Huck Lock Bolt
5/16	Protruding	3.0			P-312					
		4.0	7951		BB-352				NAS529	(ALSF-T
		4.1								(ALPPH-T
	Flush	1.0				(ACT509-E				
		1.1				(ALP426H-E				
		2.0			FA-312					
		3.0	6950							
		4.0			F-312					
		5.0	7950		BB-351				NAS525	SAL-100
		5.1								(ALPH426HT
		5.2								(ACT509H-T

NOTES:
1. Rivets are listed in the above Table by their basic part number. For a more complete description of the above rivets, refer to Table I-XXIII.
2. To select a substitute rivet, use the following procedure:
   a. Find the rivet in the above Table that was used original assembly of the airplane.
   b. Any rivet that is listed in the same size group and is either below or to the right of the original rivet may be used as a substitute. Fasteners below but to the left are not acceptable substitutes.
   c. Brackets indicate fasteners of approximately equal strength.
3. For rivet substitution purposes, rivets may be increased one diameter, except for the following restrictions:
   a. Where the increase of rivet diameter would necessitate decreasing quantity or size of adjacent rivets.
   b. Where increase of rivet size would entail the rivet head riding in a radius, short edge distance, or interference with adjacent parts and/or subsequent installation.
   c. In web splices, door frame splices, skin splices, forged bulkheads and through longerons or outside skins.
   d. When rivet spacing is less than five rivet diameters, no more than one rivet hole is to be enlarged in a group of five nearest adjacent rivet holes.
   e. When rivet spacing is five diameters or more, no more than two adjacent rivet holes may be enlarged.

TABLE 1-XXII
Rivet Substitution (Cont)

NOTES: (Cont'd)

4. Refer to paragraph 1-144 for restricted use of blind rivets.
5. Refer to the appropriate sections of this manual for other rivet substitution restrictions.
6. If a steel rivet is to be substituted for an aluminum rivet, the steel rivet must be treated for corrosion prevention as outlined in paragraph 1-189.
7. Aluminum alloy rivets used in magnesium require corrosion prevention treatment.

TABLE 1-XXIII

Rivet Basic Code

Basic Code	Description	Shank Material	Collar No.	Collar Material	Basic Part No.
AB	Blind, Protruding Head	5056-F			MS20600B
AE	Blind, Flush Head	5056-F			MS20601B
AL	Blind, Protruding Head, Cherry	Monel			CR563
AX	Blind, Flush Head, Cherry	2117-T4			CR116
AY	Blind, Protruding Head, Cherry	2117-T4			CR117
BA	Solid, Flush Head	1100-F			AN426A
BB	Solid, Flush Head	2117-T3			AN426AD
BC	Solid, Flush Head	5056-F			AN426B
BF	Solid, Flush Head	Monel			AN427M
BH	Solid, Protruding Head	1100-F			AN470A
BJ	Solid, Protruding Head	5056-F			AN470AD
BK	Solid, Protruding Head	5056-F			AN470B
BO	Solid, Protruding Head	Monel			NAS508M
BR	Blind, Flush Head, Cherry	Monel			CR562
BS	Hi-Shear Rivet, Flush Head	Steel	NAS528	2117-T4	NAS525
BT	Hi-Shear Rivet, Protruding Head	Steel	NAS528	2117-T4	NAS529
CC	Lockbolt, Blind, Huck	Steel			BL
CS	Lockbolt, Protruding Head, Huck	7075-T6	LC-F	6061-T6	ALPPH-E5
CX	Solid, Protruding Head	2024-T31			AN470DD
CY	Solid, Flush Head	2024-T31			AN426DD
DA	Lockbolt, Protruding Head, Huck	Steel	LC-C	2024-T4	ALSF-T
DB	Lockbolt, Flush Head, Huck	Steel	LC-C	2024-T4	ASCT509-T
DH	Rivet - 100° CSK Head	CRES			MS20427F
DJ	Lockbolt, Flush Head, Huck	7075-T6	LC-F	6061-T6	ALP426-E5
DK	Bolt, Blind, Flush Head Jo-Bolt	Steel			F
DL	Bolt, Blind, Flush Head, Jo-Bolt	7075-T6			FA
DR	Bolt, Blind, Protruding Head, Jo-Bolt	Steel			P
DT	Lockbolt, Protruding Head, Huck	Steel	LC-C	2024-T4	R-3001-T6 & 8
DW	Lockbolt, Flush Head, Huck	Steel	LC-C	2024-T4	R-3002-T10 & 12
DX	Lockbolt, Protruding Head, Huck	Steel	6LC-C	2024-T4	SALP
DY	Lockbolt, Protruding Head, Huck	Steel	6LC-C	2024-T4	SLSP
DZ	Lockbolt, Flush Head, Huck	Steel	6LC-C	2024-T4	SAL100
EA	Lockbolt, Flush Head, Huck	Steel	6LC-C	2024-T4	SLS100

TABLE 1-XXIII

Rivet Basic Code (Cont)

Basic Code	Description	Shank Material	Collar No.	Collar Material	Basic Part No.
FA	Rivet—Blind Explosive 100 deg Csunk (DuPont)	5056-F			56S( )100
FK	Blind, Flush Head, Expl. DuPont	5056-F			P56S( )100
FL	Blind, Protruding Head, Expl. DuPont	5056-F			P56S( )A
HB	Blind, Flush Head Expl. DuPont	Nickel			PN( )100
HC	Blind, Protruding Head, Expl. DuPont	Nickel			PN( )A
HW	Solid, Flush Head, Cad-plated	Monel			AN427MC
HX	Solid, Protruding Head, Cad-plated	Monel			AN435MC
HY	Blind, Flush Head, Expl. DuPont, Cad-plated	Nickel			CPPN( )100
HZ	Blind, Protruding Head, Expl. DuPont, Cad-plated	Nickel			CPPN( )A
JE	Lockbolt, Protruding Head, Huck	Steel	7LC-C	2024-T4	R-3001-T10 & 12
JF	Lockbolt, Flush Head, Huck	Steel	7LC-C	2024-T4	R-3002-T10 & 12
LM	Rivet - Universal Head	CRES			MS20631C
LN	Rivet - Universal Head	Monel			MS20615M
RK	Rivet - Blind, Protruding Head, Locked Spindle	5056			NAS1398B
RL	Rivet - Blind, Protruding Head, Locked Spindle	2017			NAS1398D
RO	Rivet - Blind, 100° CSK Head, Locked Spindle	5056			NAS1399B
RP	Rivet - Blind, 100° CSK Head, Locked Spindle	2017			NAS1399D
SH	Rivet - Universal Head	Monel CAD PLTD			MS20615MP
TK	Rivet - Blind, Protruding Head, Locked Spindle	Monel CAD PLTD			NAS1398MW
TO	Rivet - Blind, 100° CSK Head, Locked Spindle	Monel CAD PLTD			NAS1399MW
VN	Bolt - Blind, Hex Head (Jo-Bolt)	Steel			P312
XA	Solid, Flush Head, CONVAIR Q4304	Cres. 302			97-67001
XB	Solid, Flat Head, CONVAIR Q4310	Cres. 302			97-67002
XC	Solid, Straylor Fuel-Tight, with sealing washer, CONVAIR Q4326	2024-T31			97-67000

TABLE 1-XXIII
Rivet Basic Code (Cont)

Basic Code	Description	Shank Material	Collar No.	Collar Material	Basic Part No.
XD	Solid, Straylor Fuel-Tight, without sealing washer, CONVAIR Q4326	2024-T31			97-67000
XE	Solid, 82 deg Flush Head. Fuel-Tight, CONVAIR Special Rivet	2024-T31			97-67007
	Standard Rivet, Blind, 100 deg Csunk. Made by Deutsch Rivet Co.	Steel		Cres. 303	6950
	Oversize Rivet, Blind, 100 deg Csunk. Made by Deutsch Rivet Co.	Steel		Cres. 303	06950
	Hi-Strength Rivet, Blind, 100 deg Csunk. Made by Deutsch Rivet Co.	Steel		Cres. 303	7950
	Blind, Protruding Head. Made by Deutsch Rivet Co.	Steel		Cres. 303	6951
	Oversize,Blind, Protruding Head. Made by Deutsch Rivet Co.	Steel		Cres. 303	06951
	Hi-Strength,Blind, Protruding Head. Made by Deutsch Rivet Co.	Steel		Cres. 303	7951

NOTE

Rivet-type fasteners which are not included in this table are listed in specification NAS 523.

TABLE 1-XXIV

Strength of AN Bolts and MS2000 Bolts

BOLT SIZE	Area of Solid Section	Moment of Inertia of Solid Section	HEAT TREATED STEEL BOLTS HT 125000 psi min			HEAT TREATED ALLOY STEEL BOLTS HT 160000 psi min		
			Tension	*Single Shear	**Bending	Tension	Single Shear	Bending
	sq. in.	in.^{4}	lbs.	lbs.	in.-lbs.	lbs.	lbs.	in.-lbs.
10-32	0.02835	0.00006399	2210	2126	84			
1/4-28	0.04908	0.0001918	4080	3680	192	5000	4650	245
5/16-24	0.07669	0.0004682	6500	5750	374	8200	7300	479
3/8-24	0.1105	0.0009710	10100	8280	647	12700	10500	830
7/16-20	0.1503	0.001797	13600	11250	1025	17100	14300	1310
1/2-20	0.1963	0.003069	18500	14700	1534	23400	18650	1965
9/16-18	0.2485	0.004914	23600	18700	2184	29800	23600	2795
5/8-18	0.3068	0.007492	30100	23000	2996	38000	29150	3830
3/4-16	0.4418	0.01553	44000	33150	5177	55600	41950	6640
7/8-14	0.6013	0.02878	60000	45050	8221	76200	57100	10510
1-14	0.7854	0.04908	80700	58900	12272	102500	74600	15700
1 1/8-12	0.9940	0.07863	101800	73750	17470	128800	94150	22400
1 1/4-12	1.2272	0.1198	130200	91050	23970	162600	116600	30620

* Based on $F_{su} = 75000$ psi

** Based on $F_{bu} = 125000$ psi

TABLE 1-XXV

Strength of 100 Deg Flush Head Screws (AN 509)

Bolt Type	100 Deg Countersunk Screws (AN 509)							
Bolt Material	Steel HT 125000 psi							
Fastened Material	Clad 2024-T $F_{br} = 92000$ psi				Clad 7075-T $F_{br} = 108000$ psi			
Bolt Diameter	3/16	1/4	5/16	3/8	3/16	1/4	5/16	3/8
Bolt Sgl. Shear	2125	3680	5750	8280	2125	3680	5750	8280
Thickness	Strength — Pounds				Strength — Pounds			
0.032	493				569			
0.040	657	761			791	905		
0.051	903	1074	1224		1065	1277	1454	
0.064	1115	1439	1690	1887	1228	1748	1995	2211
0.072	1227	1683	1955	2235	1326	1903	2242	2608
0.081	1354	1848	2268	2600	1447	2053	2415	3067
0.091	1483	2031	2449	3022	1594	2218	2596	3328
0.102	1626	2235	2647	3366	1768	2400	2785	3601
0.125	1929	2622	3001	3838	2126	2842	3147	4048
0.156	2126	3174	3501	4408	2126	3544	3751	4632
0.188		3680	4053	5041		3680	4527	5401
0.250			5106	6295			5750	7302
0.312				7638				8280

TABLE 1-XXVI

Strength of Huck Lockbolts

Pin Material	Heat Treated Alloy Steel		7075-T6	
Collar Material	2024-T4		6061-T6	
Loading	Single Shear	Tension	Single Shear	Tension
Pin Diameter	Strength – Pounds		Strength – Pounds	
5/32	1770	1530		
3/16	2620	2210	1330	1375
1/4	4650	4080	2280	2535
5/16	7300	6500	3620	4025
3/8	10500	10100	5270	6275

TABLE 1-XXVII

Strength of Jo-Bolts

Jo-Bolt Type	100 Deg Flush Head FF200, FF375		Hex Protruding Head PP164, PP200	
Jo-Bolt Material	Bolt – 4130 Steel Nut – 4130 Steel Sleeve – Type 303 CRES		Bolt – 4130 Steel Nut – 4130 Steel Sleeve – Type 303 CRES	
Bolt Diameter	Single Shear	Tension	Single Shear	Tension
0.164 3/16 3/8	2620	1400	1677 2620 9750	900 1400 5600

TABLE 1-XXVIII

Strength of B Rivets

Rivet Type	Universal Head					100 Deg Head in Countersunk Hole		
Rivet Material	5056 (B)					5056 (B)		
	$F_{su} = 27000$ psi							
Rivet Diameter	3/32	1/8	5/32	3/16	1/4	1/8	5/32	3/16
Single Shear	195	350	536	774	1400	350	536	774
Riveted Material	Strength — Pounds					Strength — Pounds		
0.016	89							
Magnesium 0.020	111	149						
FS—1h 0.025	139	186	231					
F_{br}=58000 psi 0.032	178	239	295	355				
0.040	195	298	369	443	596	291	415	426
0.045	195	335	415	498	670	316	454	441
0.051		350	471	565	760	312	479	459
0.057		350	526	631	850	318	481	522
0.064			536	707	950	325	483	595
0.072			536	774	1070	326	434	633
0.081				774	1210			
0.091					1360			
0.102					1400			
0.125					1400			

TABLE 1-XXIX

Strength of Blind Rivets

Rivet Type	CHERRY CR156 CR157	DU PONT (Blast-free) DR, DR100	DU PONT (Blast-free) DR, DR100	HUCK (Keystone) P, 100V
Rivet Material	5056	5056	Nickel	5056
Rivet Size	SINGLE SHEAR STRENGTH — POUNDS			
1/8	395	387	442	445
5/32	605	624	701	684
3/16	875	800	1094	986
1/4	1580	1360		

TABLE 1-XXX

Strength of DuPont Blast-Free Blind Rivets

Rivet Type	Brazier Head	100 Deg Head in Dimpled Hole	100 Deg Head in Countersunk Hole			100 Deg Head in Countersunk Hole
Rivet Material	5056	5056	5056			Nickel
Riveted Material	7075-T Clad	7075-T Clad	7075-T Clad			1/4 H CRES
Rivet Diameter	1/8	1/8	1/8	5/32	3/16	1/8
Single Shear	387		387	605	800	387
Material Thickness	Strength—Pounds	Strength—Pounds	Strength—Pounds			Strength—Pounds
0.032	254	336	59			186
0.040	278	360	201			
0.051	296		262	196		
0.064				355	398	
0.072					534	

TABLE 1-XXXI

Strength of AD and DD Rivets (Sheet 1 of 2)

Rivet Type		Universal Head							100 Deg Head in Dimpled Holes				
Rivet Material		2117 — T3 (AD) $F_{su} = 30000$ psi				2024 — T31 (DD) $F_{su} = 41000$ psi			2117 — T3 (AD) $F_{su} = 30000$ psi			2024—T31 (DD) $F_{su} = 41000$ psi	
Rivet Diameter		3/32	1/8	5/32	3/16	5/32	3/16	1/4	1/8	5/32	3/16	3/16	1/4
Single Shear		217	388	596	862	815	1180	2120	388	596	862	1180	2120
Riveted Material		Strength — Pounds							Strength — Pounds				
	0.020	177	236						299				
Clad 2024-T3,	0.025	211	295						360	462			
Clad 2024-T4,	0.032	217	374	468					413	568	722	744	
Bare 2024-T3,	0.040	217	386	574		586			451	635	839	941	879
Bare 2024-T4,	0.045		388	584	791	659	791		466	661	885	1020	1100
Bare 2024-T6,	0.051		388	593	837	746	896	1210	484	693	940	1110	1360
Bare 2024-T81,	0.057			596	851	815	1000	1350		713	973	1170	1530
Bare 2024-T86,	0.064			596	862	815	1120	1510		736	1010	1240	1730
	0.072				862		1180	1700		755	1050	1290	1880
F_{br}=92000 psi min	0.081						1180	1910			1070	1340	2030
	0.091							2120			1090	1380	2150
	0.102							2120					2260
	0.125												
	0.020	206	278						327	452	572		650
	0.025	211	347						413	462	652		812
Clad 7075-T6,	0.032	217	374	550					535	599	725	786	1040
Bare 7075-T6	0.040	217	386	574		687			555	695	891	982	1300
	0.045		388	584	823	774	929		564	733	957	1060	1480
F_{br}=108000 psi min	0.051		388	593	837	811	1050	1420	575	778	1040	1150	1710
	0.057			596	851	815	1168	1580	576	807	1090	1210	1850
	0.064			596	862	815	1180	1770	576	840	1140	1280	2010
	0.072				862		1180	2000		867	1190	1330	2150
	0.081							2112			1230	1380	2260
	0.091							2120			1267	1420	2370
	0.102							2120					2460
	0.125												

TABLE 1-XXXI

Strength of AD and DD Rivets (Sheet 2 of 2)

Rivet Type		100 Deg Head in Countersunk Hole					
Rivet Material		2117-T3 (AD) $F_{su} = 30000$ psi			2024-T31 (DD) $F_{su} = 41000$ psi		
Rivet Diameter		1/8	5/32	3/16	5/32	3/16	1/4
Single Shear		388	596	862	815	1180	2120
Riveted Material		Strength — Pounds			Strength — Pounds		
Bare 2024-T3, Bare 2024-T6, Bare 2024-T81, Bare 2024-T86	0.020						
	0.025						
	0.032	272	306	391	477	553	708
	0.040	309	397	504	489	628	897
	0.045	323	435	534	504	687	951
	0.051	340	479	583	522	758	982
	0.057	351	500	639	634	817	1100
	0.064	363	523	705	651	886	1220
	0.072	373	542	739	726	942	1350
	0.081	388	560	769	735	992	1470
	0.091		575	795	747	1030	1580
	0.102			818		1070	1670
	0.125			853		1130	1877
Clad 7075-T6, Bare 7075-T6	0.020						
	0.025						
	0.032	272	306	391	477	553	708
	0.040	309	397	504	489	628	897
	0.045	323	435	534	504	687	951
	0.051	340	479	583	522	758	982
	0.057	351	500	639	634	817	1100
	0.064	363	523	705	651	886	1220
	0.072	373	542	739	726	942	1350
	0.081	388	560	769	735	992	1470
	0.091		575	795	747	1030	1580
	0.102			818		1070	1670
	0.125			853		1130	1877

TABLE 1-XXXII

Strength of Corrosion Resistant Steel Rivets

Rivet Type	Universal Head				Universal Head				100 Deg Head in Dimpled Hole		100 Deg Head in C sunk hole	
Rivet Material	CRES $F_{su} = 75000$ psi				CRES $F_{su} = 75000$ psi				CRES $F_{su} = 75000$ psi		CRES $F_{su} = 75000$ psi	
Riveted Material	2024-T				7075-T6				7075-T6		7075-T6	
Rivet Diameter	1/8	5/32	3/16	1/4	1/8	5/32	3/16	1/4	5/32	3/16	5/32	3/16
Single Shear	973	1490	2150	3880	973	1490	2150	3880	1490	2150	1490	2150
Material Thickness	Strength — Pounds				Strength — Pounds				Strength — Pounds			
0.020	237				278				296	416		
0.025	296	366	439		347	430			370	520		
0.032	379	468	563		444	550			474	666		
0.040	473	526	704	920	555	687			686	832		
0.045	532	659	791	1035	624	774	929		791	962		
0.051	603	746	896	1172	707	876	1050	1376	918	1118	683	807
0.057	673	834	1002	1311	790	979	1180	1539			764	902
0.064	757	938	1124	1472	887	1100	1320	1728			858	1012
0.072	851	1050	1265	1655	973	1230	1480	1943			1010	1193
0.081	958	1190	1423	1863		1390	1670	2186			1182	1396
0.091	973	1330	1599	2093		1490	1880	2455			1280	1485
0.102		1490	1793	2345			2060	2755			1387	1533
0.125			2150	2875			2150	3375			1451	1947
0.156				3588				3880				2200

TABLE 1-XXXIII

Strength of Steel and Monel Rivets

Rivet Type	100 Deg Head Flat Head	100 Deg Head Universal Head
Rivet Material	CRES $F_{su} = 75000$ psi	Monel $F_{su} = 50000$ psi
Rivet Size	Single Shear	Single Shear
1/8	973	649
5/32	1490	993
3/16	2150	1430
1/4	3880	2590

TABLE 1-XXXIV

Strength of Hi-Shear Rivets

Rivet Size	Heat Treated Steel HT 160,000 psi
	Single Shear
1/8	920
5/32	1438
3/16	2070
1/4	3680
5/16	5750
3/8	8280

NOTE: Use sheet bearing strength same as bolts.

TABLE 1-XXXV

Strength of Straylor Rivets

Rivet Type	Straylòr	
Rivet Material	2024 — T31 $F_{su} = 41000$ psi	
Rivet Diameter	3/16	1/4
Single Shear	1180	2120
Riveted Material	Clad 7075 — T6	
0.102	1136	
0.125	1170	1970
0.156	1175	1970
0.188	1175	1970
0.200	1175	1970
0.250		2083

TABLE 1-XXXVI

Hole Sizes for Hi-Shear Rivets

Rivet Shank Diameter	Convair Modified Hi-Shear Rivets Q4317 and Q4318	Alternate Installation NAS177 or NAS178 Rivets
	Hole Size	Hole Size
3/16	0.1850 + 0.0030	0.1890 + 0.0020
1/4	0.2450 + 0.0030	0.2490 + 0.0020
5/16	0.3065 + 0.0040	0.3115 + 0.0020
3/8	0.3690 + 0.0040	0.3740 + 0.0020

TABLE 1-XXXVII

Countersink Dimensions for Hi-Shear Rivets

Rivet Diameter	3/16	1/4	5/16	3/8
Pilot Hole Diameter	1/8	5/32	5/32	5/32
Nominal Countersink Diameter	0.298	0.391	0.475	0.568
Head Spotface Diameter	3/8	1/2	5/8	3/4
Collar Spotface Diameter	9/16	5/8	3/4	7/8

TABLE 1-XXXVIII

Drill Sizes and Hole Diameter Limits for DuPont Explosive Rivets

Rivet Diameter	Pilot Drill Size	Finish Drill Size	Hole Dia. Limits	
			Minimum	Maximum
1/8	No. 30 (0.1285)	No. 29 (0.136)	0.136	0.140
5/32	No. 21 (0.159)	No. 17 (0.173)	0.173	0.177
3/16	No. 11 (0.191)	No. 6 (0.204)	0.204	0.208

TABLE 1-XXXIX

Minimum Distance from Explosive Rivet to Adjacent Material

Rivet Used		Adjacent Material	
Type	Diameter	Aluminum Alloy	Magnesium
56S	1/8 inch	1/4 inch	1/2 inch
56S	5/32 inch	1/2 inch	1 inch
56S	3/16 inch	3/4 inch	1 1/2 inch

TABLE 1-XL

Grip Length—Aluminum Explosive Noiseless Rivets

Rivet Dia.	Hole Size	Grip Range	Basic Rivet Code "FL" Protruding Head Du Pont No.	Basic Rivet Code "FK" 100 deg Countersunk Head Du Pont No.	Color Code
1/8	0.134	0.005-0.045	P56S-134A-4		Yellow
	(±0.001)	0.045-0.085	P56S-134A-8	P56S-134-100-8	Red
		0.085-0.125	P56S-134A-12	P56S-134-100-12	Brown
		0.125-0.165	P56S-134A-16	P56S-134-100-16	Black
		0.165-0.205	P56S-134A-20	P56S-134-100-20	Blue
		0.205-0.245	P56S-134A-24	P56S-134-100-24	Yellow
		0.245-0.285	P56S-134A-28	P56S-134-100-28	Red
		0.285-0.325	P56S-134A-32	P56S-134-100-32	Brown
		0.325-0.365	P56S-134A-36	P56S-134-100-36	Black
5/32	0.171	0.025-0.085	P56S-173A-8	P56S-173-100-8	Red
	(±0.001)	0.085-0.145	P56S-173A-14	P56S-173-100-14	Yellow
		0.145-0.205	P56S-173A-20	P56S-173-100-20	Blue
		0.205-0.265	P56S-173A-26	P56S-173-100-26	Black
		0.265-0.325	P56S-173A-32	P56S-173-100-32	Brown
		0.325-0.385	P56S-173A-38	P56S-173-100-38	Red
3/16	0.202	0.025-0.105	P56S-204A-10	P56S-204-100-10	Blue
	(±0.001)	0.085-0.165	P56S-204A-16	P56S-204-100-16	Black
		0.165-0.245	P56S-204A-24	P56S-204-100-24	Yellow
		0.245-0.325	P56S-204A-32	P56S-204-100-32	Brown
		0.325-0.405	P56S-204A-40	P56S-204-100-40	Blue
		0.405-0.485	P56S-204A-48	P56S-204-100-48	Red
1/4	0.265	0.025-0.105	P56S-263A-10		
	(±0.001)	0.085-0.165	P56S-263A-16	P56S-263-100-16	Black
		0.165-0.245	P56S-263A-24	P56S-263-100-24	Yellow
		0.245-0.325	P56S-263A-32	P56S-263-100-32	Brown
		0.325-0.405	P56S-263A-40	P56S-263-100-40	Blue
		0.405-0.485		P56S-263-100-48	Red

TABLE 1-XLI

Grip Range—Explosive Blind Rivet—Nickel Alloy

Rivet Nominal Diameter	Grip Range	Basic Rivet Code "HC" Protruding Head Du Pont No.	Basic Rivet Code "HB" 100 deg Countersunk Head Du Pont No.
1/8	0.005-0.073		PN-134-100-X 1/16
	0.010-0.072	PN-134A-X 1/16	
	0.073-0.134	PN-134A-X 1/8	PN-134-100-X 1/8
	0.135-0.197	PN-134A-X 3/16	PN-134-100-X 3/16
	0.198-0.259	PN-134A-X 1/4	PN-134-100-X 1/4
5/32	0.010-0.072	PN-173A-X 1/16	
	0.010-0.134	PN-173A-X 1/8	PN-173-100-X 1/8
	0.073-0.197	PN-173A-X 3/16	PN-173-100-X 3/16
	0.135-0.259	PN-173A-X 1/4	PN-173-100-X 1/4
3/16	0.010-0.072	PN-204A-X 1/16	
	0.010-0.134	PN-204A-X 1/8	PN-204-100-X 1/8
	0.073-0.197	PN-204A-X 3/16	PN-204-100-X 3/16
	0.135-0.259	PN-204A-X 1/4	PN-204-100-X 1/4

TABLE 1-XLII

Drill Sizes and Hole Diameter Limits—Huck Blind Rivets

Rivet Diameter	Pilot Drill Size	Finish Drill Size	Hole Diameter Limits	
			Minimum	Maximum
1/8	No. 32 (0.116)	No. 30 (0.128)	0.129	0.132
5/32	No. 26 (0.147)	No. 20 (0.161)	0.159	0.1635
3/16	No. 16 (0.177)	No. 10 (0.1935)	0.191	0.196
1/4	F (0.257)	I (0.272)	0.272	0.276
5/16	O (0.316)	R (0.339)	0.339	0.343
3/8	V (0.377)	Y (0.404)	0.404	0.408

TABLE 1-XLIII

Grip Range—Huck Blind Rivets

Rivet Diameter	"P" Type Brazier Head		100 deg Countersunk Head	
	Grip Range	Part No.	Grip Range	Part No.
1/8	0.020—0.036	P4A	0.062—0.078	100V4A
	0.037—0.061	P4B	0.079—0.103	100V4B
	0.062—0.086	P4C	0.104—0.128	100V4C
	0.087—0.111	P4D	0.129—0.153	100V4D
	0.112—0.136	P4E	0.154—0.178	100V4E
	0.137—0.161	P4F	0.179—0.203	100V4F
	0.162—0.186	P4G	0.204—0.228	100V4G
	0.187—0.211	P4H	0.229—0.253	100V4H
	0.212—0.236	P4J	0.254—0.278	100V4J
5/32	0.025—0.045	P5A	0.080—0.100	100V5A
	0.046—0.076	P5B	0.101—0.131	100V5B
	0.077—0.107	P5C	0.132—0.162	100V5C
	0.108—0.138	P5D	0.163—0.193	100V5D
	0.139—0.169	P5E	0.194—0.224	100V5E
	0.170—0.200	P5F	0.225—0.255	100V5F
	0.201—0.231	P5G	0.256—0.286	100V5G
	0.232—0.262	P5H	0.287—0.317	100V5H
	0.263—0.293	P5J	0.318—0.348	100V5J
	0.294—0.324	P5K	0.349—0.379	100V5K
	0.325—0.355	P5L	0.380—0.410	100V5L
	0.356—0.386	P5M	0.411—0.441	100V5M
	0.387—0.417	P5N	0.442—0.472	100V5N
3/16	0.030—0.054	P6A	0.100—0.124	100V6A
	0.055—0.091	P6B	0.125—0.161	100V6B
	0.092—0.128	P6C	0.162—0.198	100V6C
	0.129—0.165	P6D	0.199—0.235	100V6D
	0.166—0.202	P6E	0.236—0.272	100V6E
	0.203—0.239	P6F	0.273—0.309	100V6F
	0.240—0.276	P6G	0.310—0.346	100V6G
	0.277—0.313	P6H	0.347—0.383	100V6H
	0.314—0.350	P6J	0.384—0.420	100V6J
	0.351—0.387	P6K	0.421—0.457	100V6K
	0.388—0.424	P6L	0.458—0.494	100V6L
	0.425—0.461	P6M	0.495—0.531	100V6M
	0.462—0.498	P6N	0.532—0.568	100V6N
	0.499—0.535	P6P	0.569—0.605	100V6P

TABLE 1-XLIV.

Drill Sizes and Hole Diameter Limits—Standard AN Rivets

Rivet Diameter	Hole Diameter Limits	Drill Size and Decimal Diameter
1/16	0.067 to 0.073	No. 50 (0.0700)
3/32	0.099 to 0.105	No. 39 (0.0995)
1/8	0.128 to 0.134	No. 30 (0.1285)
5/32	0.161 to 0.167	No. 20 (0.1610)
3/16	0.191 to 0.199	No. 10 (0.1935)
7/32	0.221 to 0.229	No. 1 (0.2880)
1/4	0.257 to 0.265	F (0.2570)
5/16	0.323 to 0.331	P (0.3230)
3/8	0.386 to 0.394	W (0.3860)
1/2	0.515 to 0.523	33/64 (0.5160)

TABLE 1-XLV

Identification of Deutsch Drive Pin Blind Rivets

Rivet Type	Part Number	Pin Color	Sleeve Color
Standard Diameter	6950-6951-69509	Electrofilm Black	Electrofilm Black
Short Pin	6950-6951-SP	Cadmium Plated	Electrofilm Black
Close Tolerance Shank	6950-C 6951-C	Electrofilm Black	Cadmium Plated
Close Tolerance Shank	6950-C 6951-C-SP	Cadmium Plated	Cadmium Plated
1/32 Inch Oversize	06950-06951	Electrofilm Black	Green Dye
1/32 Inch Oversize	06950-06951-SP	Cadmium Plated	Green Dye
High Temperature	7950-7951	Red Dye	Silver
High Temperature	7950-7951-SP	Silver	Silver
1/32 Inch Oversize, High Temperature	07950-07951	Red Dye	Red Dye
1/32 Inch Oversize, High Temperature	07950-07951-SP	Silver	Red Dye

TABLE 1-XLVI

Part Number Identification — Deutsch Drive Pin Blind Rivets
(Sample Part Number 6950-S-C10-6SP)

6950	S	C10	6SP
Head Type	Material	Diameter	Grip Length
6950 — Countersunk Head. 6951 — Protruding Head. 69509 — AN509 Countersunk Head, (Special order only). 06950 — Countersunk Head — 1/32 Inch Oversize Diameter. 09651 — Protruding Head — 1/32 Inch Oversize Diameter. 7950 — High Temperature — High Strength, Countersunk Head. 7951 — High Temperature — High Strength, Protruding Head. 07950 — High Temperature — High Strength, Countersunk Head — 1/32 Inch Oversize Diameter. 07951 — High Temperature — High Strength, Protruding Head — 1/32 Inch Oversize Diameter.	Stainless Steel	04=1/8 inch 05=5/32 inch 06=3/16 inch 08=1/4 inch 10=5/16 inch 12=3/8 inch 14=7/16 inch 16=1/2 inch NOTE: A "C" before the diameter number indicates close tolerance shank — omit for standard tolerance shank.	(Refer to grip range table.) An "SP" is added to part number for short pin rivets — omit for standard pin rivets.

TABLE 1-XLVII

Drill Sizes and Hole Diameter Limits — Standard Drive Pin Deutsch Blind Rivets

Rivet Diameter	Pilot Drill Size	Finish Drill Size	Hole Diameter Limits Minimum	Maximum
1/8 inch	7/64	1/8	0.125	0.126
5/32 inch	9/64	5/32	0.156	0.157
3/16 inch	#16 (0.177)	#11 (0.191)	0.190	0.194
1/4 inch	A (0.234)	G (0.261)	0.260	0.265
5/16 inch	M (0.295)	O (0.316)	0.315	0.320
3/8 inch	T (0.358)	W (0.386)	0.385	0.390

NOTE: Close tolerance shank rivets should be installed in holes reamed to size to suit the class of fit required by the application in which it is intended.

TABLE 1-XLVIII

Grip Range — Deutsch Drive Pin Blind Rivets

Grip No.	Grip Range		Grip No.	Grip Range		Minimum Available Grip No.	
	Minimum	Maximum		Minimum	Maximum	Rivet Diameter	Grip No.
2	0.103	0.166	16	0.978	1.040	1/8 inch	2
3	0.166	0.227	17	1.040	1.103	5/32 inch	2
4	0.227	0.290	18	1.103	1.165	3/16 inch	3
5	0.290	0.353	19	1.165	1.228	1/4 inch	4
6	0.353	0.415	20	1.228	1.290	5/16 inch	4
7	0.415	0.478	21	1.290	1.353	3/8 inch	6
8	0.478	0.540	22	1.353	1.415	7/16 inch	7
9	0.540	0.603	23	1.415	1.478	1/2 inch	8
10	0.603	0.665	24	1.478	1.540		
11	0.665	0.728	25	1.540	1.603		
12	0.728	0.790	26	1.603	1.665		
13	0.790	0.853	27	1.665	1.728		
14	0.853	0.915	28	1.728	1.790		
15	0.915	0.978					

TABLE 1-XLIX

Replacement Parts Rework

Part Number	Nomenclature	Rework
8-13601-13 -16	Inboard elevon door	Trim edge 0.030" max.
8-13603-7 thru -36	Elevon nose door	Chamfer and trim edge 0.030" max. Realign by bumping.
8-14116	Fin bay lower aft door	Trim edge 0.030" max.
8-14139-7	Fin bleed air door	Trim edge 0.030" max.
8-14140-1	Fin equipment door panel	Trim edge 0.030" max.
8-14141-1	Waveguide access door panel	Trim edge 0.030" max.
8-14142-3	Fin electrical equipment door panel	Trim edge 0.030" max.
8-14144-3	Fin rudder controls access panel	Trim edge 0.030" max.
8-14228	Fin tip leading edge	Trim edge 0.010" max.
8-14232	Fin lower leading edge	Trim edge 0.010" max.
8-14234-1	Fin upper leading edge	Trim edge 0.010" max.
8-14518-1	Fin tip, lower	Remove and reinstall jo-bolt attachment. Trim 0.030" max.
8-14521-5	Fin tip, upper	Trim 0.030" max.
8-14650-7	Rudder hinge access door	Trim edge 0.030" max.
8-14655-9	Fin hydraulic and control access door	Trim edge 0.030" max.
8-14825-11	Fin tip waveguide access door	Trim edge 0.020" max.
8-17630-81 -82	Main landing gear wing fairing	Trim edge 0.250" max.
8-18004-7 -8	Wing fuselage intersection angle	Match drill for fasteners for fuselage and wing.
8-18004-9	Wing fuselage intersection angle	Match drill for fasteners for fuselage and wing. Replace jo-bolts through fairing to wing.

**Applicable to F-106A airplanes only.*

***Applicable to F-106B airplanes only.*

TABLE 1-XLIX

Replacement Parts Rework (Cont)

Part Number	Nomenclature	Rework
8-18040-15 -16 -67 -68 -69 -70 -87 -88 8-18041-7 -8 -9 -10 8-18060-9 thru -14 -77* & -83*		Same rework requirements as 8-18004-9 for all listed dash no's.
8-18201-1 -2	Stub leading edge assy	Match drill and rivet at attachment to wing-fuselage attach angles. Trim 0.010" max.
8-18202-1 -2	Sta 144 to 216 wing leading edge	Trim 0.010" max.
8-18203-1 -2 8-18204-1 -2 8-18204-3 -4	Wing leading edge Sta 221 to 296 Sta 301 to 334 Sta 338 to 384 Sta 301 to 334	Trim 0.010" max.
8-18231-1 thru 8-18231-803	Wing leading edge panels	Trim 0.030" max.
8-18231-805 -806	Wing leading edge panels	Match drill outboard attachment holes. Trim 0.030" max.
8-18252-3 thru -6	Sta 144 to 216 leading edge	Trim 0.010" max.
8-18254-1 thru -4	Sta 301 to 334 wing leading edge	Trim 0.010" max.
8-18255-3 -4	Sta 338 to 384 wing leading edge	Trim 0.010" max.
8-18256-1 -2	Sta 301 to 344 wing slotted leading edge	Trim 0.010" max.
8-18281-1 thru -14	Wing leading edge panel	Trim 0.030" max.

**Applicable to F-106A airplanes only.*

***Applicable to F-106B airplanes only.*

TABLE 1-XLIX

Replacement Parts Rework (Cont)

Part Number	Nomenclature	Rework
8-18911-1 -2	Wing tip	Trim 0.020" max.
8-18913-7 -8	Wing fuselage intersection angle	Match drill fasteners from wing and fuselage. Replace jo-bolt through fairing to wing.
8-18989-1 -2	Spar I access attachment angle (fairing)	Match drill for fasteners. Replace jo-bolts through fairing to wing.
8-44719*	External power access door	Trim edge 0.030" max. Adjust latch. Drill and lockwire latch screws.
8-44545-7* -8	Main landing gear door	Drill door attachment holes from hinge. Locate and rivet wing fuselage closure angle. Trim 0.080" max.
8-45958-21*	Ram air turbine door	Trim 0.120" max., aft and lower edge. Trim 0.030" max., forward edge.
8-73830-65	Retainer windshield glass	Trim 0.090" max., forward edge. Trim 0.060" max., other edges. Chamfer if necessary.
8-73830-69	Retainer windshield glass	Trim 0.060" max. Chamfer edge where necessary.
8-74017-1 * -3 -5 -805	Canopy assy	Match drill shear plate after aligning. Trim edge 0.060" max.
8-74406-99	Radar compartment lower panel access door.	Trim edge 0.030" max.
8-74497-7 * -8	Fixed dorsal access door.	Trim edge 0.030" max.
8-74498-7 thru -17	Removable dorsal assy	Trim edge 0.030" max.
8-74600-220 -295	Forward radar compartment door	Trim edge 0.250" max. Crimp lower edge if required. Install rivets to retain hinge pin.
8-74602-23	Engine oil cooler door	Trim edge 0.030" max.
8-74606-7 * -9	Fuselage fuel tank door	Trim edge 0.120" max.
8-74609-63	Hydraulic access door	Trim forward and aft edge 0.030" max. Trim inboard edge 0.200" max.

**Applicable to F-106A airplanes only.*

***Applicable to F-106B airplanes only.*

TABLE 1-XLIX

Replacement Parts Rework (Cont)

Part Number	Nomenclature	Rework
8-74614-43 -44	Forward engine mount access door	Trim edge 0.030" max.
8-74626-187	Nose landing gear door	Trim edge 0.060" max.
8-74652-13 * -15	Emergency canopy release door	Trim 0.030" max.
8-74655-25 *	Liquid oxygen access door	Trim 0.030" max.
8-74659-13	Generator oil cooler door	Trim 0.030" max.
8-74663-1 -2 -3 -4 -801	Lower inboard missile bay door	Trim edge 0.090" max.
8-74664-1 -2 -3 -4	Upper outboard missile bay door	Trim edge 0.090" max.
8-74669-3 * -4	Electronics access door	Trim edge 0.120" max.
8-74845-7 *	Canopy access door	Trim edge 0.030" max.
8-74854-65 *	Pilot's canopy aft fairing	Trim edge 0.030" max.
8-74860-23	Drag chute mechanism door	Trim edge 0.030" max.
8-74863-13	Speed brake mechanism door	Trim edge 0.040" max.
8-74870-7 -8	Speed brake mechanism door	Trim edge 0.060" max.
8-77611-7 * -8	Tank inspection door	Trim edge 0.030" max.
8-77620-31	Liquid oxygen access door	Trim edge 0.030" max.
8-77628-31 * -55	Pneumatic starter door	Locate and drill hinge attachment holes from fuselage structure. Trim edge 0.030" max.
8-77629-7	Sta 145 LH access door	Trim edge 0.030" max.
8-77727-7 *	Center refrigeration door beam	Chamfer ends 0.020" max. Trim hinge nodes 0.010" max.
8-77804 *	Windshield deflector	Trim edge 0.030" max.
8-92703	Fuel nozzle ground door	Trim edge 0.030" max.

**Applicable to F-106A airplanes only.*

***Applicable to F-106B airplanes only.*

TABLE 1-XLIX

Replacement Parts Rework (Cont)

Part Number	Nomenclature	Rework
8-76017-1 **	Canopy assy	Match drill phase plate after aligning. Trim forward end 0.075" max. Trim lower edge 0.060" max.
8-76017-3 **	Canopy assy	Same as -1 except no trim to forward end.
8-76204-7 **	Cover, hydraulic access, sta 329 to 357	Trim edge 0.030" max.
8-76198-3 **	Cover plate, access hole, sta 384.50 to 405.20	Trim edge 0.060" max.
8-76315-5 **	Canopy, fuel tank vapor seal	Trim edge 0.020" max.
8-76361-1 ** -2 ** 8-76362-7 ** -9 ** 8-76372-7 ** -8 **	Door, maintenance, missile bay, sta 222.40 to 236.88	Trim edge 0.020" max. NOTE Refer to Air Force drawing number 7545908 for longeron repair.
8-76498-7 ** -9 ** -11 ** -13 **	Removable dorsal assy	Trim edge 0.150" max.
8-76498-105** -107** -109** -111**	Removable dorsal assy	Trim edge 0.030" max.
8-76501-35 ** -43 ** -49 **	Dorsal panel instl, sta 308.50 to 318.44	Trim 0.120" max.
8-76567-65 **	Duct leading edge access door	Trim 0.020" max.
8-76611-51 **	Mid bay electronics access door	Trim edge 0.060" max.
8-76646-29 ** 8-76653-7 ** 8-76870-13 **	Canopy island	Trim lower and aft edge 0.120" max.
8-76652-11 ** -13 ** -15 **	Door assy, emergency canopy release	Trim edge 0.030" max.
8-76802-801** -802**	Canopy, forward, transparent panel	Trim forward edge 0.075" max.
8-76856 -1 **	Aft canopy fairing	Trim edge 0.030" max.

**Applicable to F-106A airplanes only.*

***Applicable to F-106B airplanes only.*

TABLE 1-XLIX

Replacement Parts Rework (Cont)

Part Number	Nomenclature	Rework
8-76873 **	Fairing, canopy assy, telescoping	Trim forward and aft edge 0.060" max.
8-76942-5 **	Dorsal refrigeration compartment door	Trim edge 0.030" max.
8-76948-105** -109** -123** -124**	Heat exchanger exhaust	Countersink attachment holes. Trim 0.030" max.
8-99954-1 ** -3 ** -803** -805** -807** -809**	Cover assy, splash, nose wheel well	Trim to clear cut outs for tubing and harness 0.500" max.

***Applicable to F-106B airplanes only.*

TABLE 1-L

Weight of Airplane and Components

Component	Condition	Weight in Pounds	
		F-106A	F-106B
Airplane.	Basic Weight.	25,227	26,200
Wing.	Basic (Including Main Landing Gear Attach Fittings, Elevon Support Fittings, and Access Doors).	1,661	1,661
Wing Tip.	Basic.	57	57
Elevon.	Basic (Without Support Fittings).	128	128
Fin Tip.	Basic.	71	71
Rudder.	Basic.	71	71
Main Landing Gear.	Wheel. Brake. Actuator (Dry). Side Brace. Oleo Strut (Including Drag Brace, Pivot Beam, and Retract Mechanism). Door.	48 95 15 43 208 57	48 95 15 43 208 57

TABLE 1-L

Weight of Airplane and Components (Cont)

Component	Condition	Weight in Pounds	
		F-106A	F-106B
Nose Landing Gear.	Wheel.	11	11
	Actuator (Dry).	4	4
	Oleo Strut (Including Drag Brace, Steer Unit, Retract Mechanism).	119	119
	Door.	23	23
Missile Bay Door.	Upper Panel.	91	91
	Lower Panel.	53	53
Canopy.	Basic.	131	380
Radome.	Basic.	156	156
Gun (F-106A).	Basic.	870	———

TABLE 1-LI

Airplane Center of Gravity

Airplane	Condition (Landing Gear Down)	Center of Gravity Location	
		Vertical	Horizontal
F-106A.	Basic Weight.	7.8 inches below W.L. 0.00.	Station 422.0
F-106B.	Basic Weight.	8.8 inches below W.L. 0.00.	Station 415.0

TABLE 1-LII

Packing and Crating Information — Airplane Components

Component	Weight Lbs	Dimensions (Inches)	Center of Balance (Inches)	Remarks
Wing.	1661	294½ x 16½ x 166¾	100 forward of No. 7 spar and 54 from inboard edge.	Wing tip, elevon and MLG removed.
Wing Tip.	57	77 x 44 x 8	27 forward of trailing edge and 19 from and perpendicular to outboard edge.	Basic.
Elevon.	128	161 x 11	70 from inboard end.	Basic.
Drop Tank.	236	162 x 26 x 38	134 aft of tank nose tip.	
Radome.	156	38½ Dia. x 83½	57 aft of forward end.	Basic.
Canopy, F-106A.	131	72 x 34½ x 29	39 from apex.	Basic.
Canopy, F-106B.	380	130 x 40 x 36	69 from apex.	Basic.
Fin Tip.	71	82 x 21 x 3	45 from top of aft edge.	Basic.
Rudder.	71	68½ x 27 x 7½	59 from top of trailing edge and 19 from bottom of trailing edge.	Complete with control horn.
Missile Bay Door, Upper Panel.	91	196 x 12 x 5½	98 from forward hinge point.	Basic.
Missile Bay Door, Lower Panel.	53	196 x 13 x 3	96 from forward hinge point.	Basic.
Gun (F-106A)	1540	104 x 46 x 60	56.4 inches from forward end	Gun assembly included.

Section II

WING GROUP

2-1. WING GROUP.

2-2. The wing is of full cantilever, stressed skin construction with a delta configuration. This configuration has a 60-degree sweepback of the leading edge and a 5-degree sweep forward of the trailing edge, as shown on figure 2-1. Early F-106 airplanes (F-106A 56-453 thru 56-466 and F-106B 57-2507) have case XIV wings. Later F-106 airplanes have case XXIX wings. The wings are quite similar in construction, the basic difference being that the case XXIX has more camber. The right and left wing panels are attached to the fuselage by means of bolts through the forged fittings on the wing spars and fuselage frames. Drag angles are also used to attach the wing panels to the fuselage. These drag angles are riveted to the inboard edge of the wing and are attached to the outside surface of the fuselage by means of screws. Each wing panel consists of an interspar section with integral fuel tanks, a cambered leading edge, a cambered wing tip, a main landing gear and landing gear fairing, an elevon, outboard elevon actuator fairing, and provisions for the external mounting of droppable fuel tanks. Wing stations are shown on figure 2-2. The F-106A and F-106B airplanes incorporate nine integral fuel tanks. There are four fuel tanks in each wing, identified as No. 1, No. 2, No. 3 and "T" tanks. The F-106A center of gravity control transfer system consists of the "F" tank in the fuselage and the "T" (transfer) tank in the trailing edge structure of each wing. Refer to paragraph 2-4 for the location of the wing fuel tanks Nos. 1, 2 and 3, and the "T" (transfer) tank. See figure 4-1 for the location of the fuselage "F" tank.

2-3. DESCRIPTION OF WING COMPONENTS.

2-4. Interspar Structure.

2-5. The interspar structure is fabricated principally of 7075-T6 aluminum alloy. This structure consists basically of chordwise ribs and spanwise spars to which machined, stressed skin is riveted. The three main integral fuel tanks are contained in this interspar structure between the number 1 and number 6 spars and the wing closing rib; the "T" or transfer tank is located between the number 6 and number 7 spars. All fuel tanks within the interspar structure are designed with machined fuel-tight corner fittings. Small openings in the chordwise ribs permit an even distribution of fuel within each tank. Figure 2-3 illustrates the wing structure and Figure 2-4 shows the location of the fuel tanks.

2-6. Spars and Ribs.

2-7. The wing leading edge spar and the number 7 spar are of built-up construction. Number 2, 3, 4, 5, and 6 spars are constructed of machined forgings. The spanwise spars extend at an angle of 90 degrees from the centerline of the airplane and are interconnected by a series of closely spaced chordwise ribs. The main landing gear attaches to the trunnion mounts on the number 3 and 4 wing spars. See figure 5-4 for the location of the trunnion mounts on the wing spars. The built-up ribs consist of upper and lower extruded rails which are joined together with stiffener reinforced webs. The inboard bulkheads incorporate a series of heavy-gage press-formed brackets and clips to provide attachment for the wing plating and wing to fuselage drag angles.

2-8. Wing Plating.

2-9. The wing plating is made of highly stressed, heavy gage, bare aluminum alloy. Figures 2-5 and 2-6 show the wing plating and the alloy designation for each section of wing plating. Refer to paragraph 2-35 for information concerning repairs to the wing plating. The wing plating is installed with the machined surfaces inside the fuel tank area. Machining provides for the heaviest gages at the points of attachment to the ribs and spars, eliminating doublers and fillers. The requirements for maximum aerodynamic smoothness are partially achieved by machining the large wing skins on the inner surface so that internal ribs, doublers, and fillers are an integral part of the skin. This practice increases rigidity and reduces the tendency of skin buckling, with the attendant effects of disrupted airflow and vibration. The number of outside seams is also reduced, and the existing ones are filled with aerodynamic smoothing compound. Plating above the main landing gear area is of uniform thickness and is attached by means of standard flush-head fasteners. Access doors are located throughout the plating area of the lower wing surface to provide access to all internal areas. The access doors are machined to provide a flush installation and to preserve the aerodynamic characteristics of the wing surfaces.

KEY	STRUCTURAL COMPONENT	FIGURE REFERENCE
1.	Wing	
	Plating	2-5 and 2-6
	Structure	2-3
2.	Elevon plating and structure	
	Inboard	2-13
	Outboard	2-14
3.	Leading edge plating and structure	2-9
		2-10
4.	Wing tip plating and structure	2-11 and 2-12

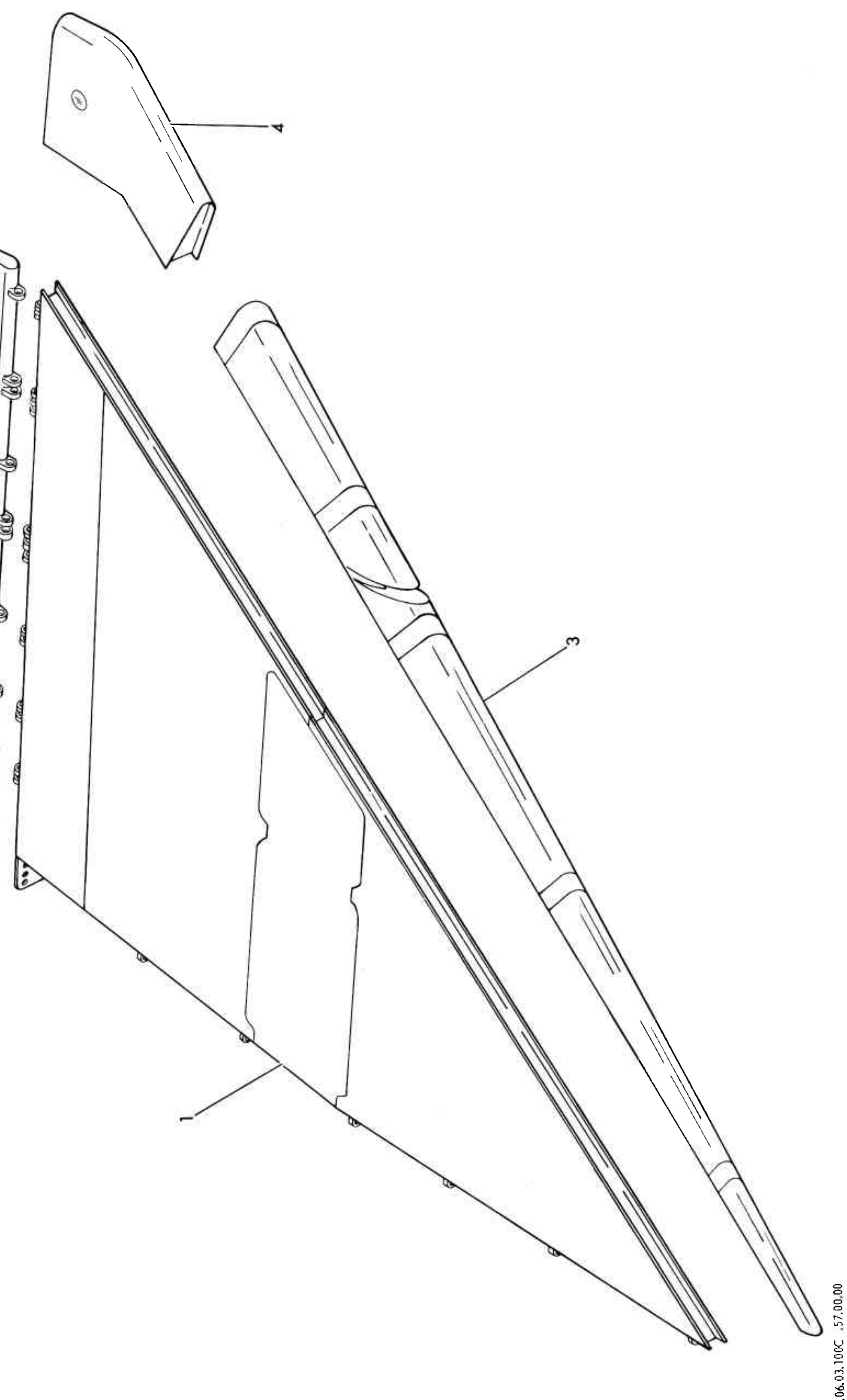


Figure 2-1. Wing Group Components and Index

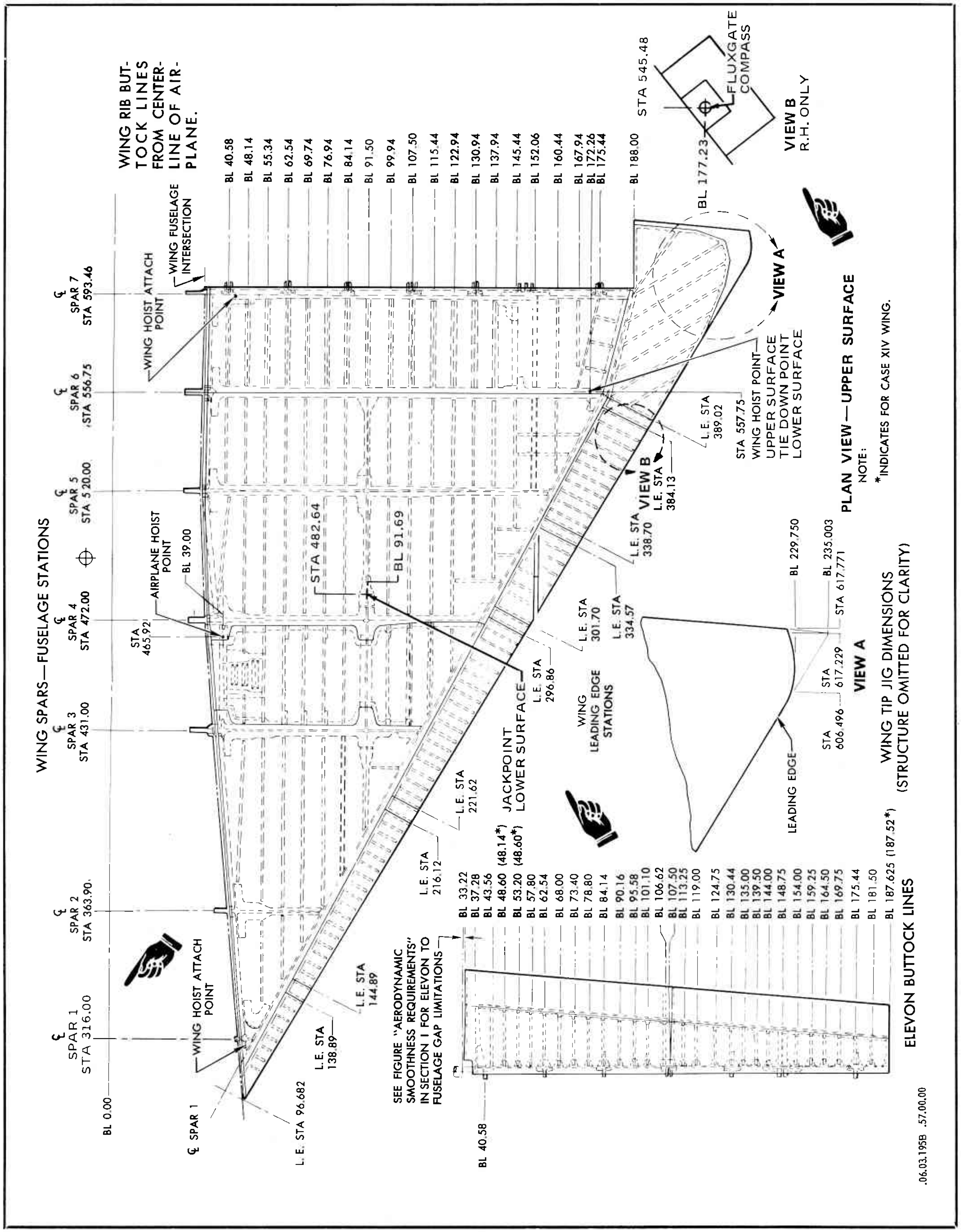


Figure 2-2. Wing Station Diagram

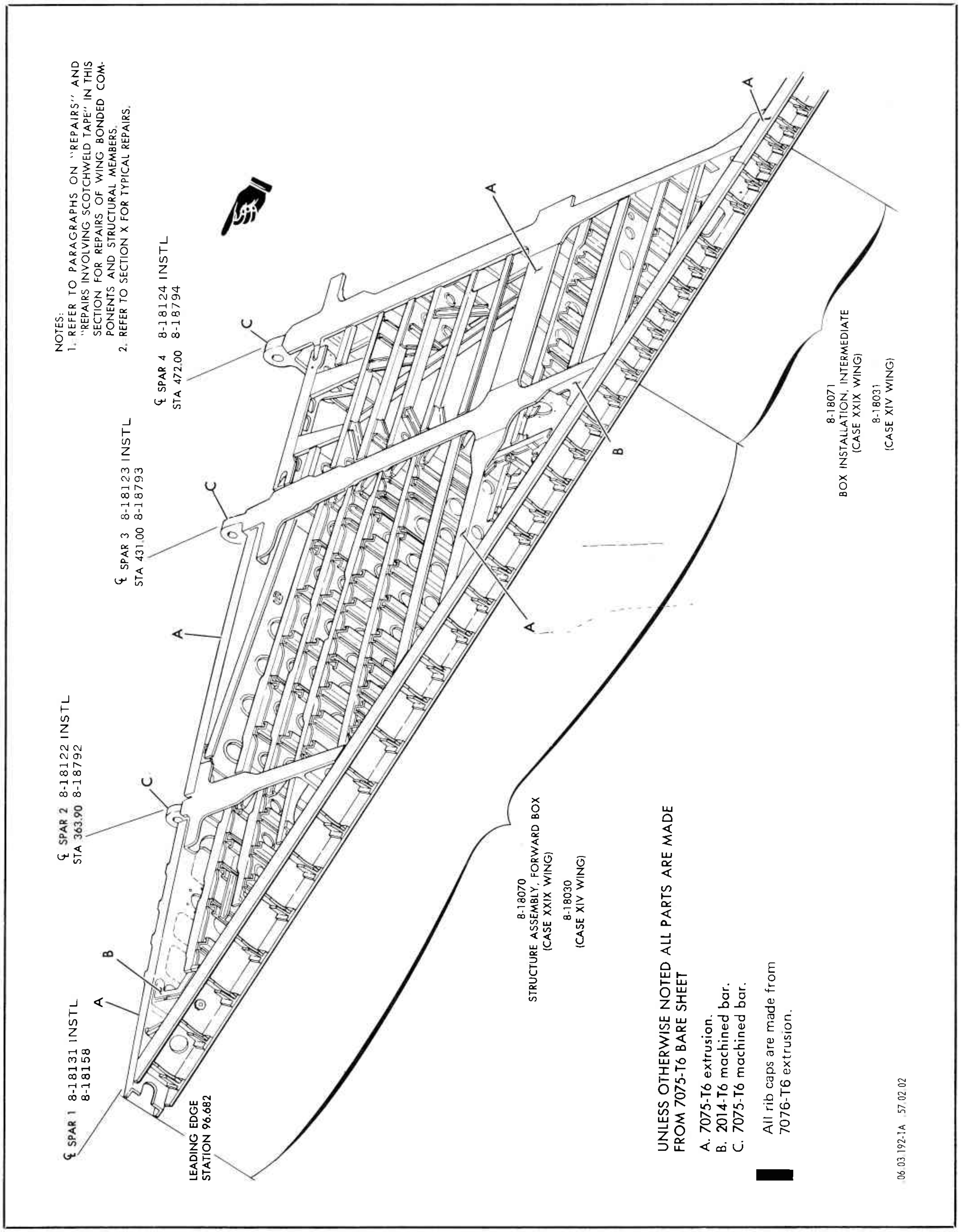


Figure 2-3. Wing Structure (Sheet 1 of 2)

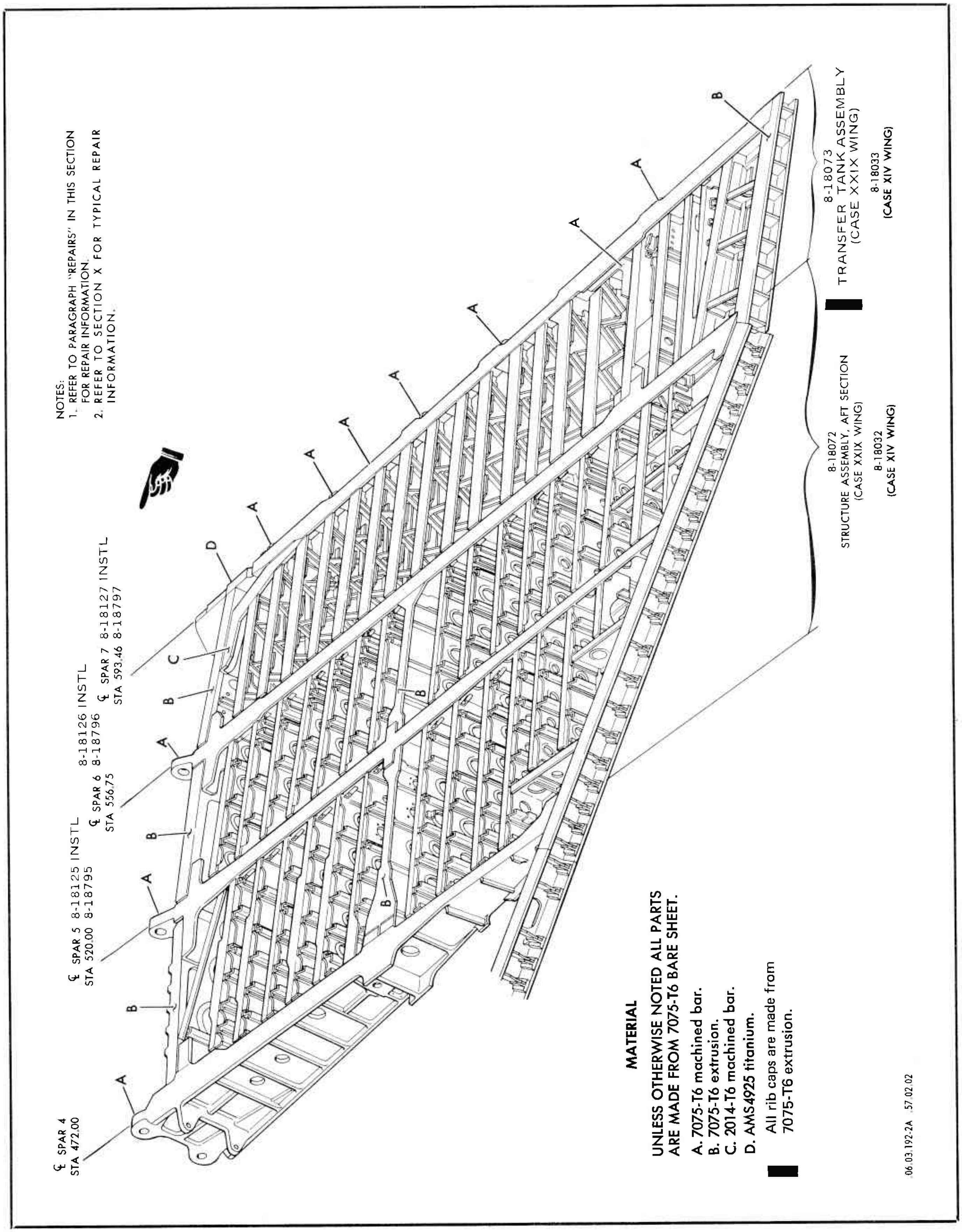


Figure 2-3. Wing Structure (Sheet 2 of 2)

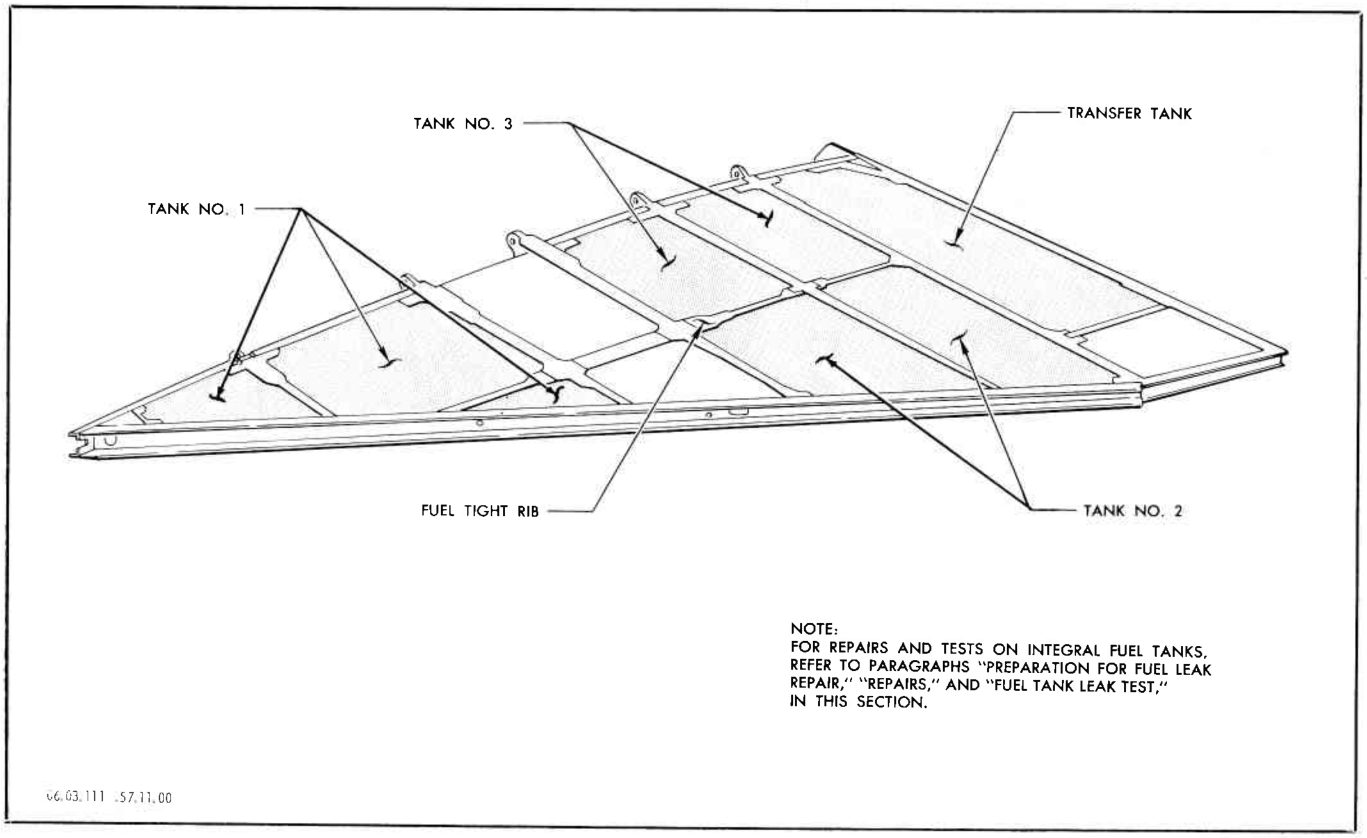


Figure 2-4. Wing Fuel Tank Area

2-10. Wing Attachment.

2-11. The wing is attached to the fuselage with bolts through the wing spars and spar fittings on the fuselage bulkheads, and by drag angles riveted to the inboard edge of the wing and attached to the outside surface of the fuselage with screws. No drilling or reaming is required for the installation of a new wing at the forged fitting attach points. Some match-drilling of screw holes in the drag angles is required. Otherwise the wing is completely interchangeable. Number 1 spar attaches to the fuselage with one bolt; numbers 2, 3, 4, 5, and 6 spars are connected to the fuselage with two bolts, and number 7 spar is attached to the fuselage with a web attachment requiring ten bolts. Detailed instructions giving torque values, type of fittings, special tools, and equipment required for the installation and removal of the wings are given on figure 2-7. Since original installation wing attachment fittings do not require bushings, wing attachment fittings that have become worn or damaged may require bushings to be within tolerances when a new wing is attached. See figure 2-8 for information on bushed fitting repairs.

2-12. Wing Leading Edge.

2-13. The wing leading edge sections are cambered to maintain the airfoil in a given contour specifically for the purpose of improving the aerodynamic characteristics of the wing. See figures 2-9 and 2-10 for details of the wing leading edge construction. The basic construction consists of medium-gage press-formed, aluminum alloy ribs, doublers, and fillers riveted to extruded angles in sections. A short section located between leading edge stations 301.70 and 338.70 provides additional ribs and angles for reinforcement of the leading edge slot. The leading edge slot controls outward air flow over the wing similar to a wing fence but with less drag. Leading edge sections are secured to the wing leading edge spar by means of screws through gang channel nuts. The gang channels are riveted to the inner flange of the wing leading edge spar. The leading edge does not provide for anti-icing.

2-14. Wing Tip.

2-15. The wing tip attached to the Case XIV wing consists basically of spanwise, press-formed channel spars and chordwise ribs as shown on figure 2-11. Two "tee" shaped rails are riveted together to form the main spar at the point of attachment to the wing trailing edge closing rib. Light-weight machined magnesium castings form the leading edge, corner, and trailing edge of the structure. The inner structural parts are riveted together

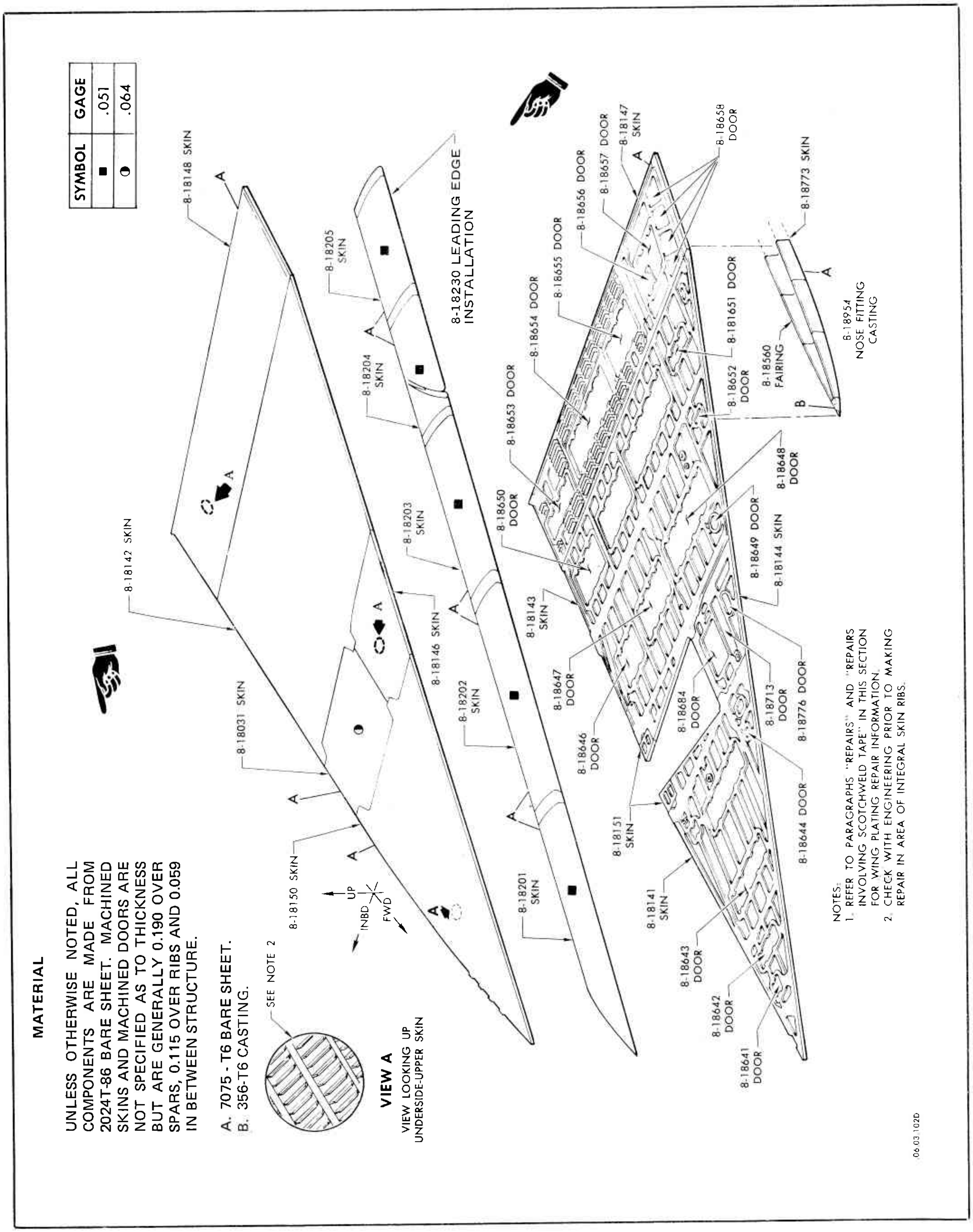


Figure 2-5. Case XIV Wing Plating Diagram
Applicable to F-106A airplanes 56-453 thru 56-466, and F-106B airplane 57-2507

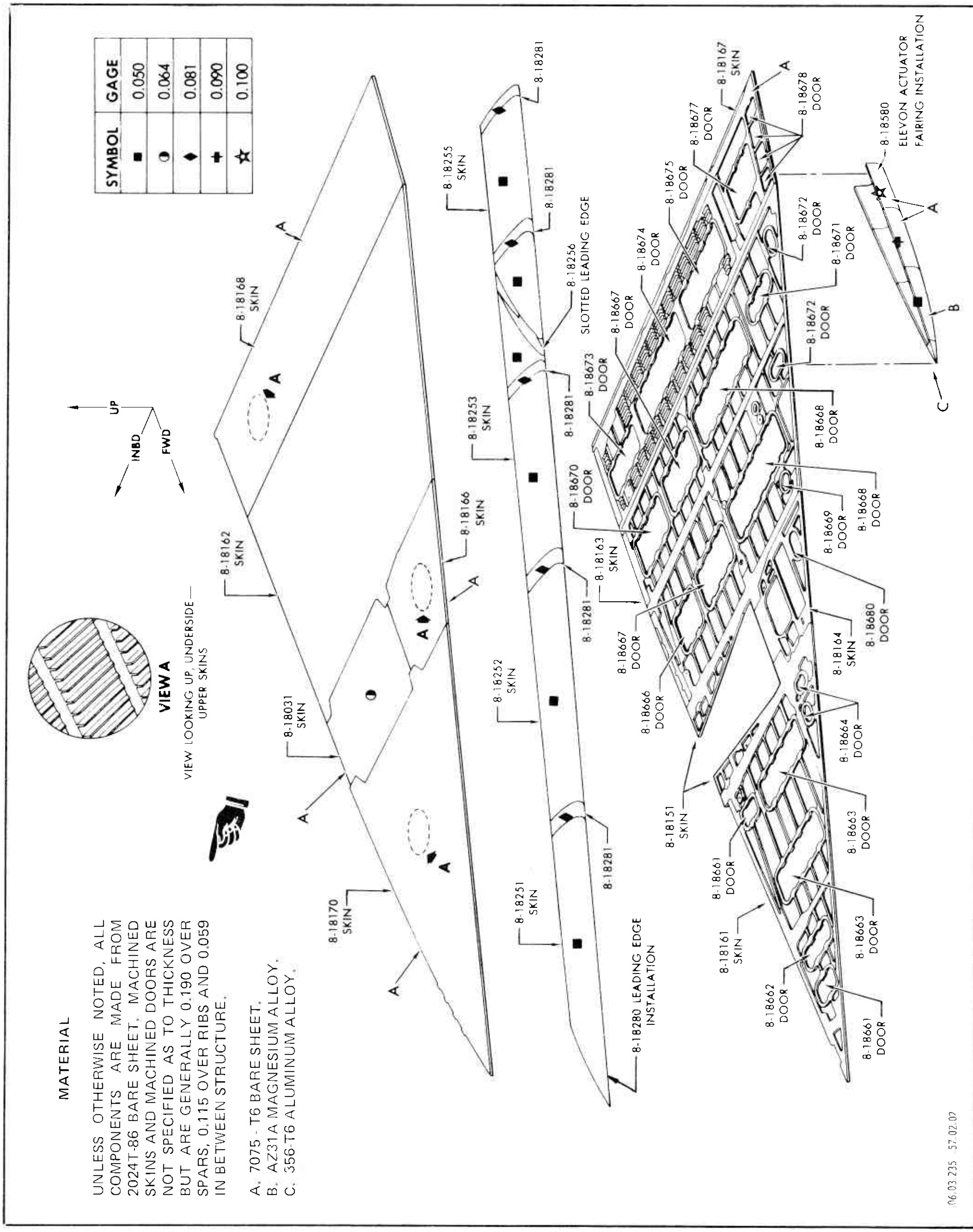
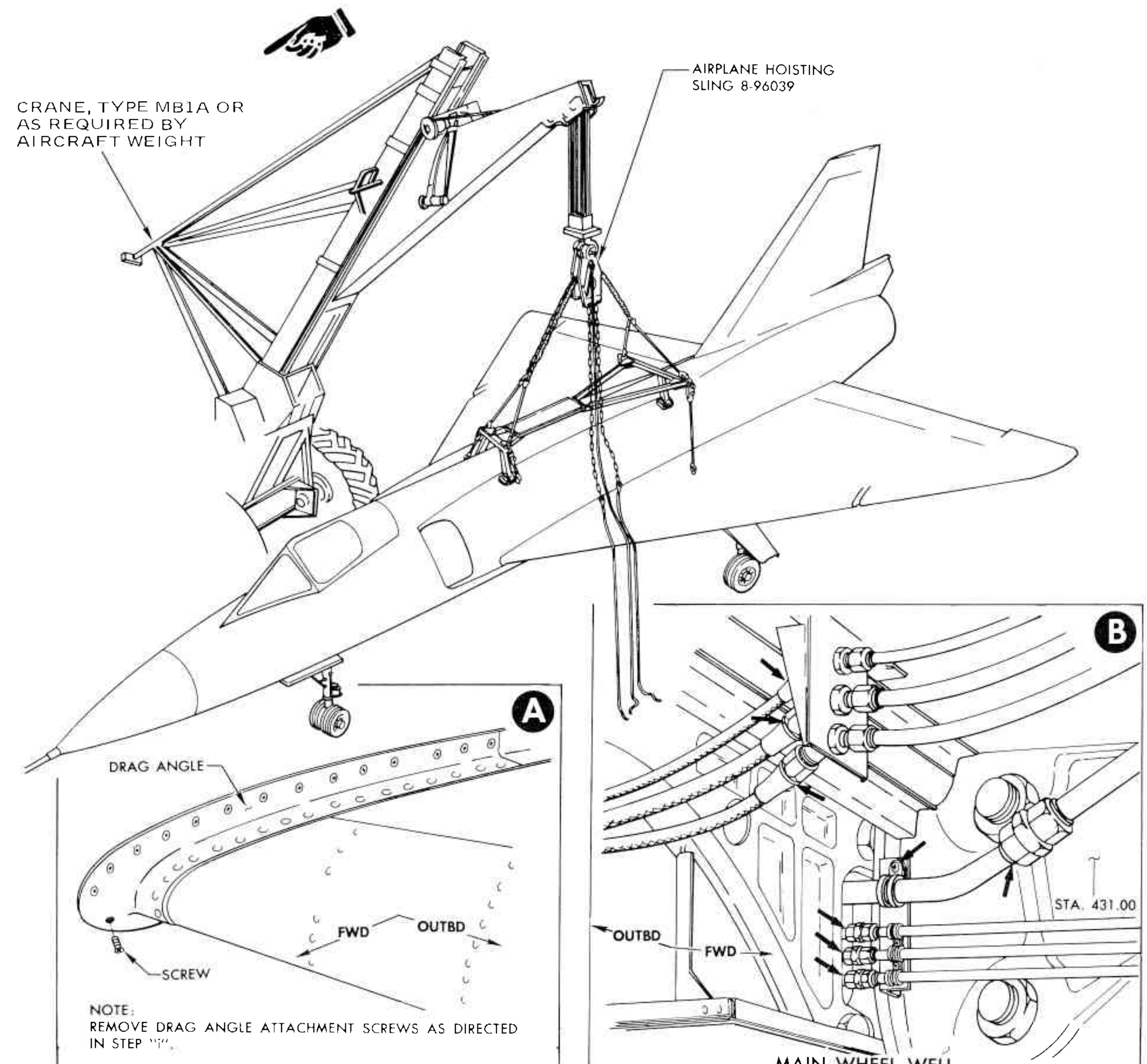


Figure 2-6. Case XXIX Wing Plating Diagram

Applicable to F-106A airplanes 57-230 and subsequent, and F-106B airplanes 57-2508 and subsequent



WING REMOVAL

a. Defuel airplane and remove external fuel tanks if installed. Refer to T.O. 1F-106A-2-5-2-1 for procedure.
b. Jack airplane and remove tailcone and engine. Refer to T.O. 1F-106A-2-4-2-1 for procedure.
c. Remove electrical power from airplane. Remove battery as directed in T.O. 1F-106A-2-10-2-1.
d. Remove the fuselage main landing gear doors and fairings and the ram air turbine doors as directed in T.O. 1F-106A-2-8-2-1. Disconnect the missile bay door actuators as directed in T.O. 1F-106A-2-12-2-1 and allow doors to hang free.
e. Lower aircraft onto landing gear and remove jacks.
f. Attach airplane hoisting sling assembly, 8-96039 to airplane. Refer to illustration titled "Airplane Hoisting" in Section 1 for procedure. Hoist airplane off ground.
g. Retract nose landing gear, using the manual retraction procedure. Disconnect nose gear door actuator and close nose landing gear door manually. Refer to T.O. 1F-106A-2-8-2-1 for procedure.

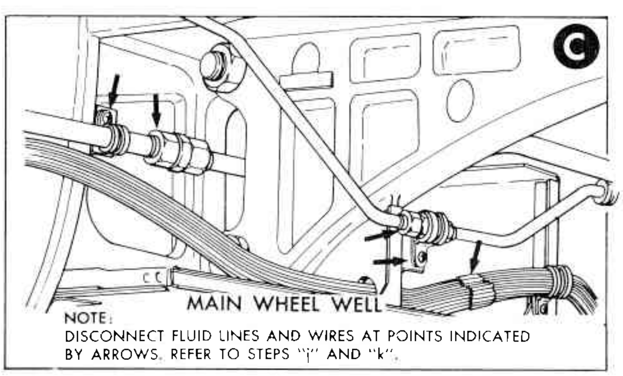
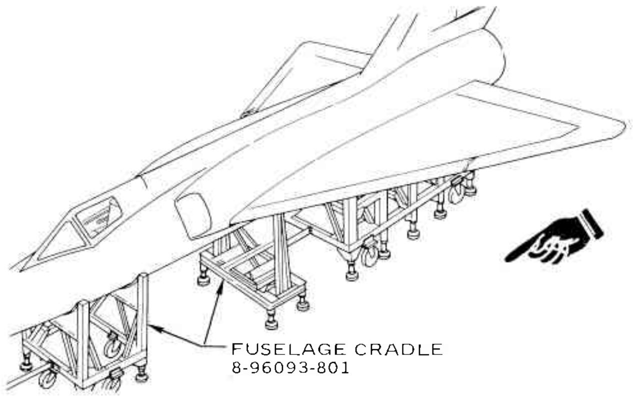
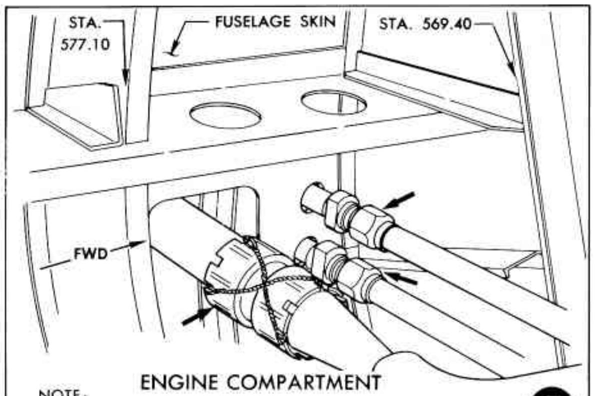
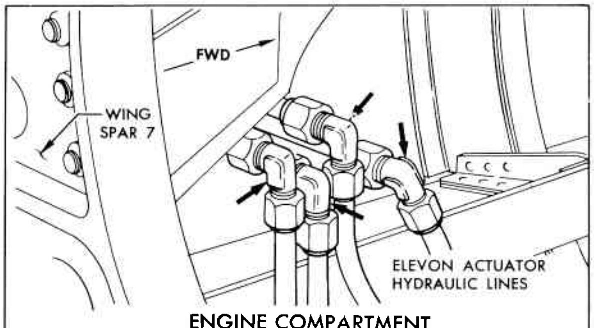


Figure 2-7. Wing Replacement (Sheet 1 of 5)

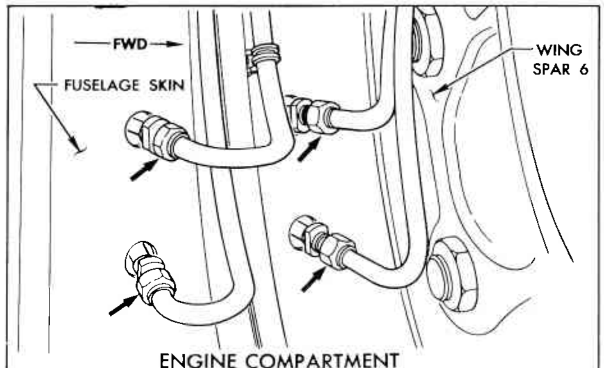




NOTE:
DISCONNECT FUEL SENSE LINES AND WIGGINS COUPLING AT POINTS INDICATED BY ARROWS. REFER TO STEP "n". **F**



NOTE:
DISCONNECT FLUID LINES AT POINTS INDICATED BY ARROWS. REFER TO STEP "l." **D**

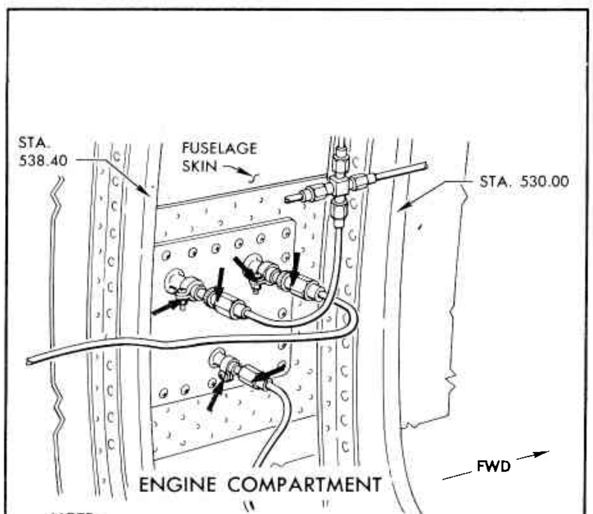


NOTE:
DISCONNECT FUEL SENSING LINES AT POINTS INDICATED BY ARROWS. REFER TO STEP "n." **G**

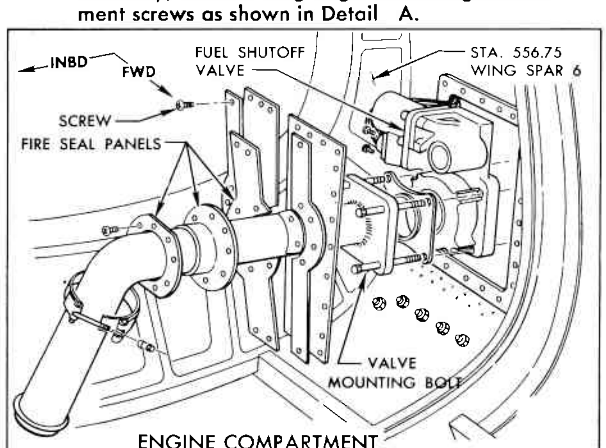
WING REMOVAL (Cont'd)

h. Position fuselage cradle 8-96093-801 and lower airplane onto cradle.

i. After removal of 8-96039 airplane hoisting sling assembly, remove drag angle-to-fuselage attachment screws as shown in Detail A.



NOTE:
DISCONNECT HYDRAULIC HEAT EXCHANGER FLUID LINES AND LOW PRESSURE PNEUMATIC LINE AT POINTS INDICATED BY ARROWS. LOOSEN PLUG CLAMPS AT POINTS INDICATED BY ARROWS. THIS DETAIL IS APPLICABLE ONLY TO AIRPLANES WITH HYDRAULIC HEAT EXCHANGERS. **E**



NOTE:
REMOVE AND REPLACE FUEL SHUTOFF VALVE AS DIRECTED BY INSTRUCTIONS IN T.O. 1F-106A-2-5-2-1. **H**

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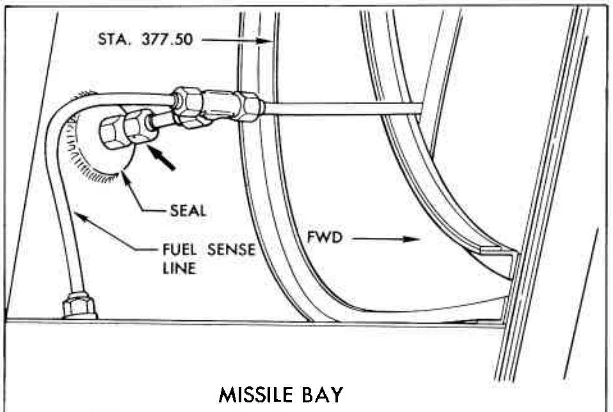
Figure 2-7. Wing Replacement (Sheet 2 of 5)



NOTE:
REMOVE SIDE PANEL FOR ACCESS AND DISCONNECT FUEL SENSE LINE AT POINT INDICATED BY ARROW. REFER TO STEP "n".

I

j. Remove elevon by procedure outlined in T.O. 1F-106A-2-7-2-1. Remove main landing gear from wing; refer to T.O. 1F-106A-2-8-2-1. Disconnect main landing gear hydraulic lines as shown in Detail B and C.

k. Disconnect all wing circuit electrical wires at permanent splices in harness located in main wheel well. Refer to T.O. 1F-106A-2-10-2-1 for wiring diagrams. Identify each wire to facilitate reinstallation. See Detail C.

l. Disconnect elevon actuator hydraulic lines as shown in Detail D.

m. Disconnect hydraulic heat exchanger lines and low pressure pneumatic line as shown in Detail E. This step applies only to airplanes with hydraulic heat exchangers in the wings.

n. Disconnect fuel lines as shown in Details F through K. Refer to T.O. 1F-106A-2-5-2-1 for detailed information.

o. Remove rivets as required (approximately 12 each) from AC exciter regulator and DC control support assembly, part No. 8-65078, located at approximately station 465.00 in right hand main wheel well.

p. Remove nuts from bolts in all wing-to-fuselage attach fittings as shown in Details L and M.

q. Place a suitable padded prop under opposite wing to maintain airplane balance when wing is removed.

r. Attach wing hoisting sling assembly, 8-96004 as shown below.

s. Support wing with hoist and remove bolts from all wing-to-fuselage attach fittings.

t. Pull wing outboard until all lines and fittings are free: remove wing.

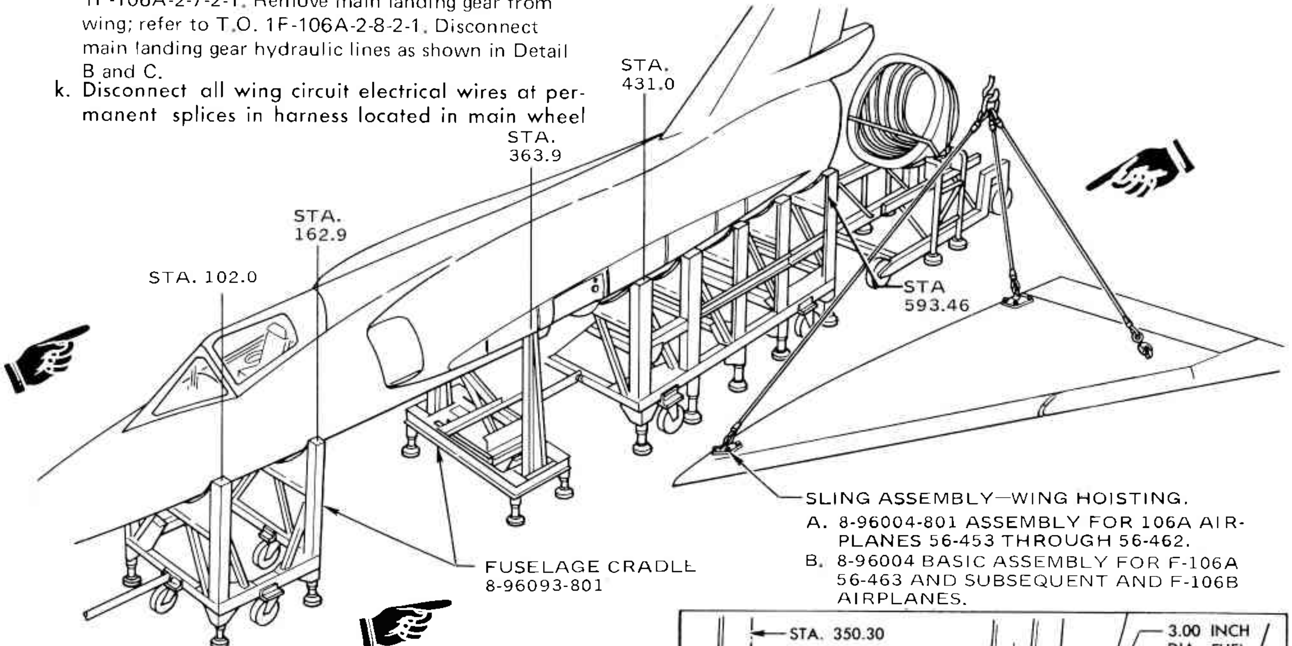
INSTALLATION:

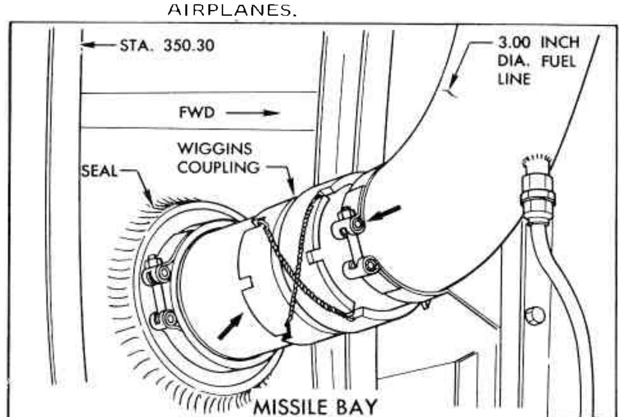
To install wing, perform removal steps in reverse order and observe the following:

a. Prior to installing wing attach bolts, insert one aligning pin, 8-96096 in each wing attach fitting except at spar No. 7. Chill pins in refrigerator to ease installation.

b. Install and torque wing attach bolts according to the table on sheet 4.

c. When assembling Wiggins couplings, torque size -12D to 8-10 foot-pounds and size -16D to 10-13 foot-pounds.





NOTE:
REMOVE SIDE PANEL FOR ACCESS AND DISCONNECT FUEL LINE AND SEAL CLAMP AS INDICATED BY ARROWS. REFER TO STEP "n".

J

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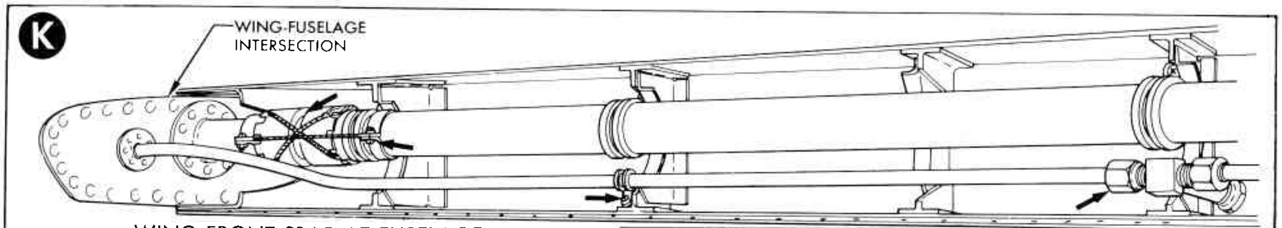
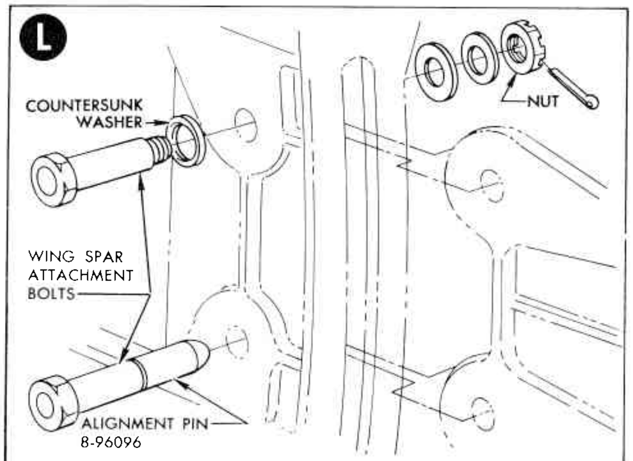
Figure 2-7. Wing Replacement (Sheet 3 of 5)



WING FRONT SPAR AT FUSELAGE

NOTE:
REMOVE INBOARD LEADING EDGE SECTION AND DISCONNECT FUEL VENT LINE WIGGINS COUPLING AND FUEL SENSE LINE AT POINTS INDICATED BY ARROWS. REFER TO STEP "n"



WING SPAR-FUSELAGE ATTACHMENT

NOTE:
USE 3/8 INCH ALLEN WRENCH TO SCREW ALIGNMENT PINS ON ENDS OF WING BOLTS. TYPICAL FOR WING SPARS 2, 5 AND 6.

INSTALLATION (Cont'd)

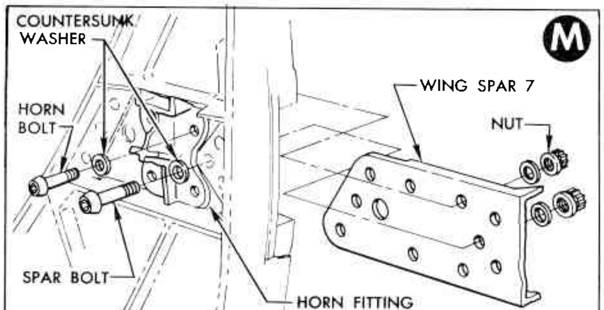
d. After assembling operations are completed, bleed air from hydraulic system. Refer to T.O. 1F-106A-2-3-2-1 for procedure.

e. Perform elevon operation check. Refer to T.O. 1F-106A-2-7-2-1 for procedure.

f. Perform fuel system operation checks. Refer to T.O. 1F-106A-2-5-2-1 for procedures.

g. Perform landing gear operation check. Refer to T.O. 1F-106A-2-8-2-1 for procedure.

h. Check operation of wing tip and landing lights. Refer to T.O. 1F-106A-2-10-2-1.



WING SPAR 7-FUSELAGE ATTACHMENT

NOTE:
TORQUE SIX SPAR BOLTS AND FOUR HORN BOLTS WITHIN LIMITS GIVEN IN TABLE BELOW.

SPAR NO.	BOLT DIRECTION	BOLT AND WASHER	NUT AND WASHER	TORQUE (INCH-POUNDS)
1	NUT AFT	(1)NAS464-8-22 (1)MS20002-C8	(1)AN320-8 (1)AN960-816	40 TO 80
2	NUTS AFT	(2)8-18941-15 (2)MS20002-C16	(2)AN320-14 (2)8-18930-11 (2)AN960-1416L	100 TO 200
3	NUTS FWD	(2)8-18941-13 (2)8-17975-7	(2)AN320-20 (2)8-17975-13 (2)AN960-2016L	100 TO 200
4	NUTS AFT (SEE NOTE 2)	(2)8-18941-11 (2)8-17975-7	(2)AN320-20 (2)8-17975-13 (2)AN960-2016L	100 TO 200
5	NUTS AFT	(2)8-18941-9 (2)MS20002-C20	(2)AN320-16 (2)8-18930-9 (2)AN960-1616L	100 TO 200
6	NUTS AFT	(2)8-18941-7 (2)8-18930-7	(2)AN320-20 (2)8-18930-13 (2)AN960-2016L	100 TO 200
7	NUTS AFT	(3)MS20012-18 (3)MS20012-22 (6)MS20002-C12	(6)42FW-1216 (6)MS20002-12	2300 TO 2500
7 (HORN FITTING)	NUTS AFT	(2)MS20010-28 (2)MS20010-30 (4)MS20002-C10	(4)42FW-1018 (4)MS20002-10	1100 TO 1300

NOTE:
1. PROCEDURE 42FW-1018 AND -1216 NUTS FROM STANDARD PRESSED STEEL COMPANY, JENKINTOWN, PA.
2. UPPER LEFT ATTACH BOLT IS INSTALLED WITH NUT FORWARD.

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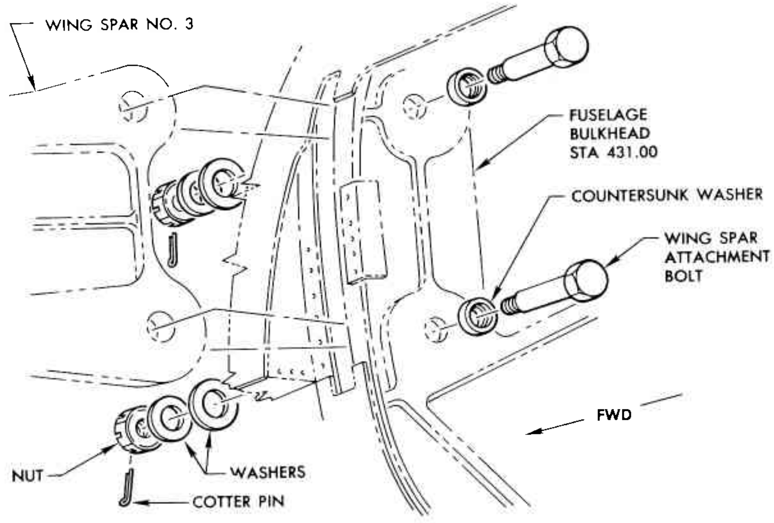
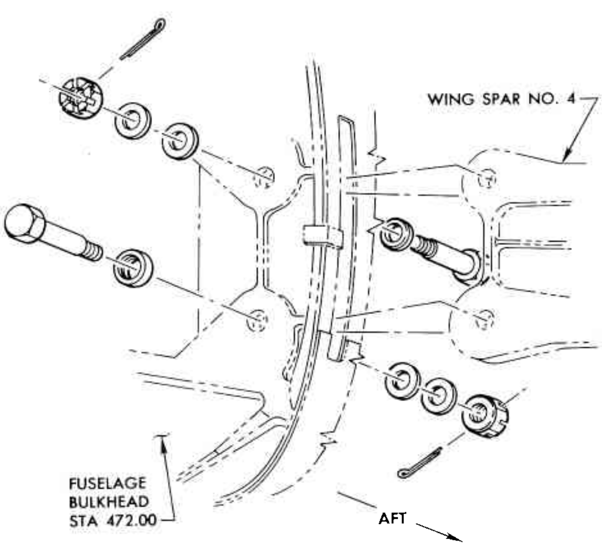
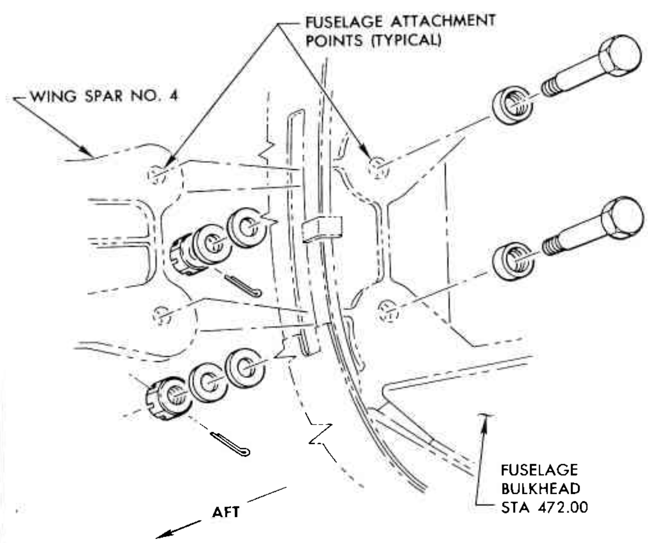
Figure 2-7. Wing Replacement (Sheet 4 of 5)



LEFT NO. 3 WING SPAR—FUSELAGE ATTACHMENT



LEFT NO. 4 WING SPAR—FUSELAGE ATTACHMENT



RIGHT NO. 4 WING SPAR—FUSELAGE ATTACHMENT

NOTE:
SEE SHEET 4 FOR WING ATTACHMENT HARDWARE PART NUMBERS.

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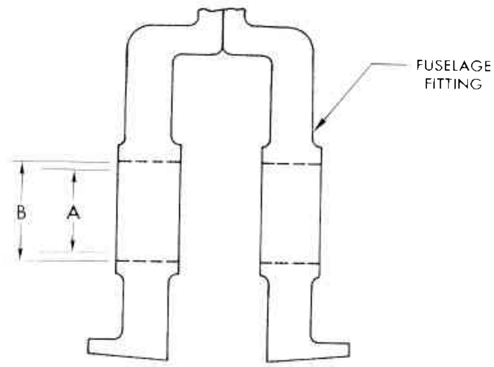
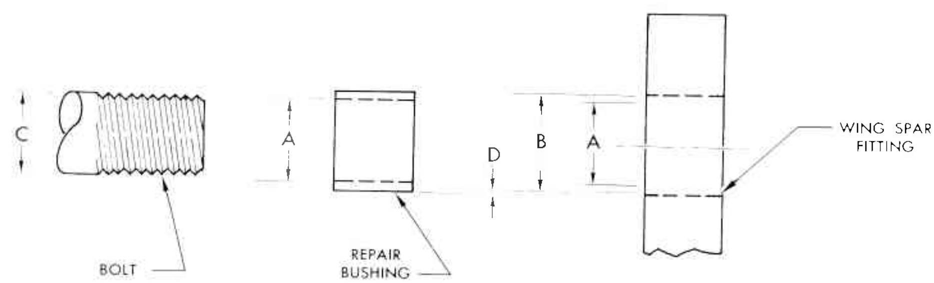
Figure 2-7. Wing Replacement (Sheet 5 of 5)

NOTES:
1. MAKE BUSHINGS FROM SAE4130 OR SAE8630 STEEL. HEAT TREAT TO 125,000 TO 150,000 PSI TENSILE STRENGTH PER SPECIFICATION MIL-H-6875.
2. CADMIUM PLATE ALL BUSHINGS SURFACES PER SPECIFICATION QQ-P-416. DIMENSIONS TO BE MET AFTER PLATING.
3. BUSHINGS SIDES TO BE PARALLEL AND CONCENTRIC WITHIN 0.003 INCH TOTAL INDICATOR READING.
4. BUSHINGS TO RECEIVE MAGNETIC INSPECTION PER SPECIFICATION MIL-I-6868.
5. MAINTAIN 0.06 INCH MINIMUM BUSHING WALL THICKNESS.





	A		B		C	D
SPAR NUMBER	ORIGINAL FITTING HOLE SIZE	MAXIMUM ALLOWABLE WEAR TOLERANCE	MAXIMUM ALLOWABLE HOLE OVERSIZE		BOLT DIAMETER	BUSHING CLEARANCE
1.	0.5000 +0.0010 −0.0005	0.5000 +0.0040 −0.0005	0.6250 +0.0010 −0.0005		0.4991 +0.0000 -0.0010	±0.0004
			WING	FUSELAGE		
2.	1.0000 +0.0010 −0.0005	1.0000 +0.0040 −0.0005	1.2500 +0.0010 −0.0005	1.1250 +0.0010 −0.0005	0.9990 +0.0000 -0.0020	±0.0005
3.	1.6250 +0.0010 −0.0005	1.6250 +0.0040 −0.0005	1.8750 +0.0010 −0.0005	1.7500 +0.0010 −0.0005	1.6240 +0.0000 -0.0020	±0.0005
4.	1.6250 +0.0010 −0.0005	1.6250 +0.0040 −0.0005	1.8750 +0.0010 −0.0005	1.7500 +0.0010 −0.0005	1.6240 +0.0000 -0.0020	±0.0005
5.	1.2500 +0.0010 −0.0005	1.2500 +0.0040 −0.0005	1.5000 +0.0010 −0.0005		1.2490 +0.0000 -0.0020	±0.0005
6.	1.6875 +0.0010 −0.0005	1.6875 +0.0025 −0.0005	1.9375 +0.0010 −0.0005		1.6865 +0.0000 -0.0020	±0.0005
7.	0.7500 +0.0010 −0.0005	0.7500 +0.0025 −0.0005	0.8750 +0.0010 −0.0005		0.7488 +0.0000 -0.0030	±0.0005
	0.6250 +0.0010 −0.0005	0.6250 +0.0025 −0.0005	0.7500 +0.0010 −0.0005		0.6240 +0.0000 -0.0030	±0.0005

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Figure 2-8. Wing Attachment — Bushed Fitting Repair

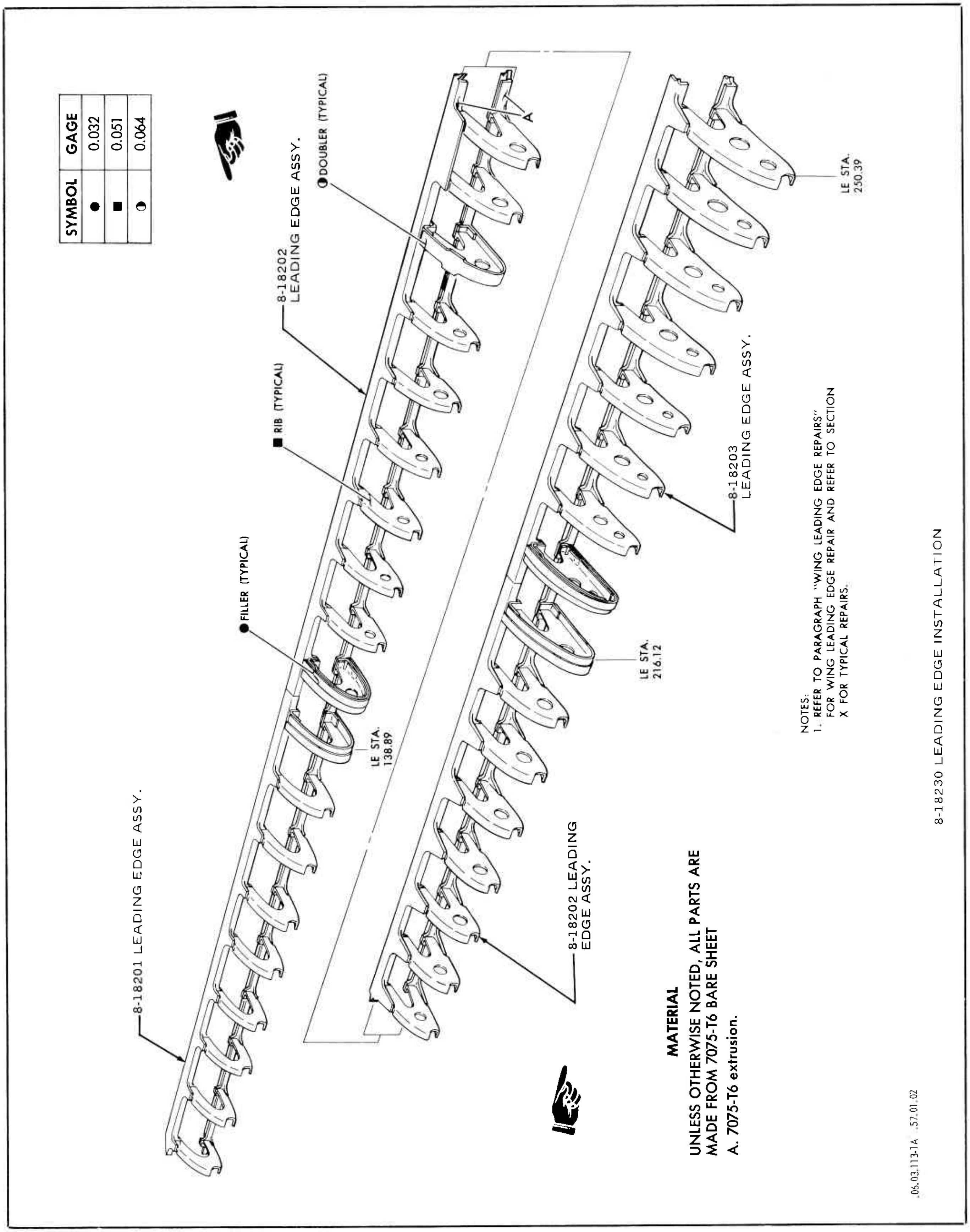


Figure 2-9. Case XIV Wing Leading Edge Structure (Sheet 1 of 2)
Applicable to F-106A airplanes 56-453 thru 56-466, and F-106B airplane 57-2507

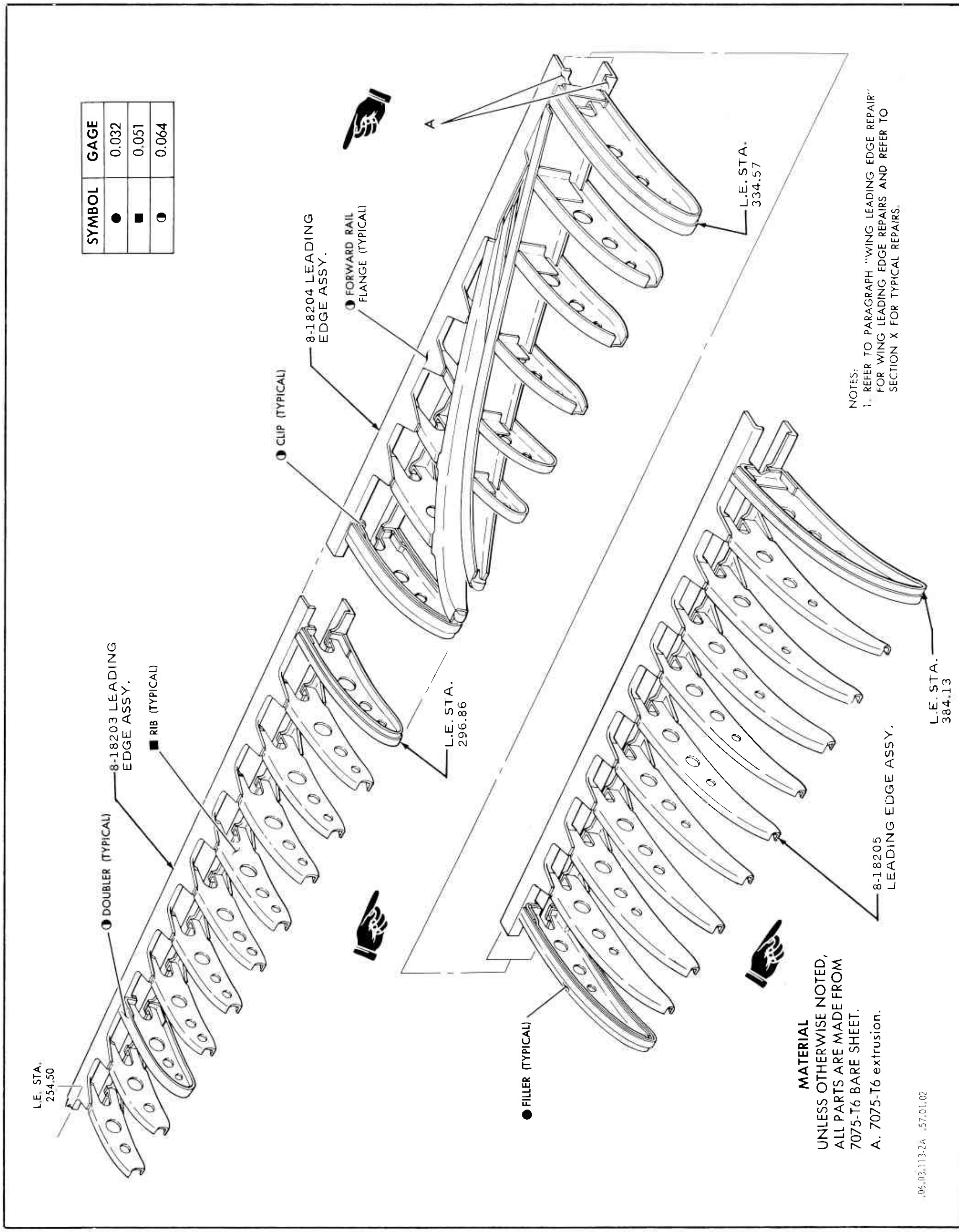


Figure 2-9. Case XIV Wing Leading Edge Structure (Sheet 2 of 2)
Applicable to F-106A airplanes 56-453 thru 56-466, and F-106B airplane 57-2507

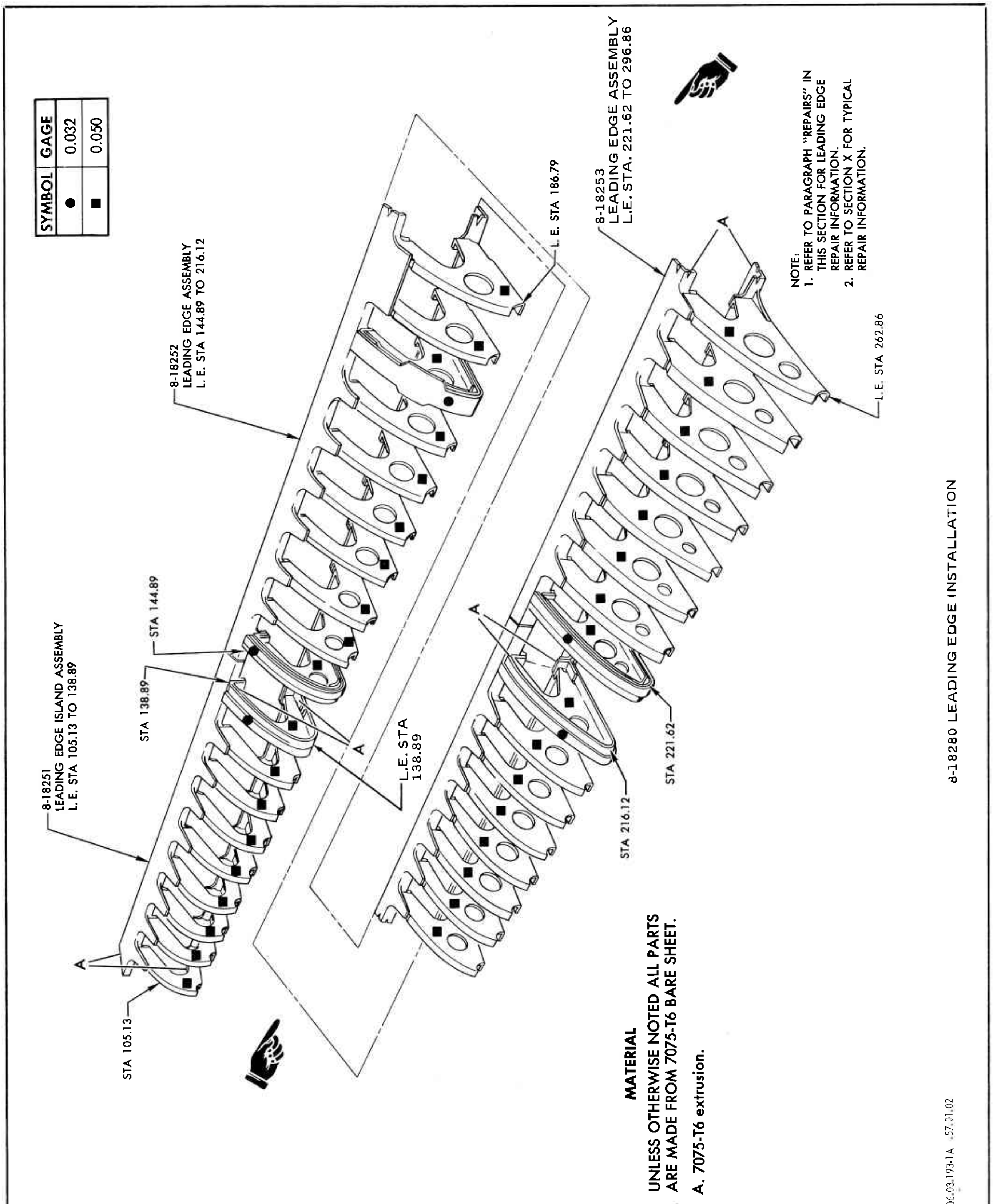


Figure 2-10. Case XXIX Wing Leading Edge Structure (Sheet 1 of 2)
Applicable to F-106A airplanes, 57-230 and subsequent, and F-106B airplanes 57-2508 and subsequent

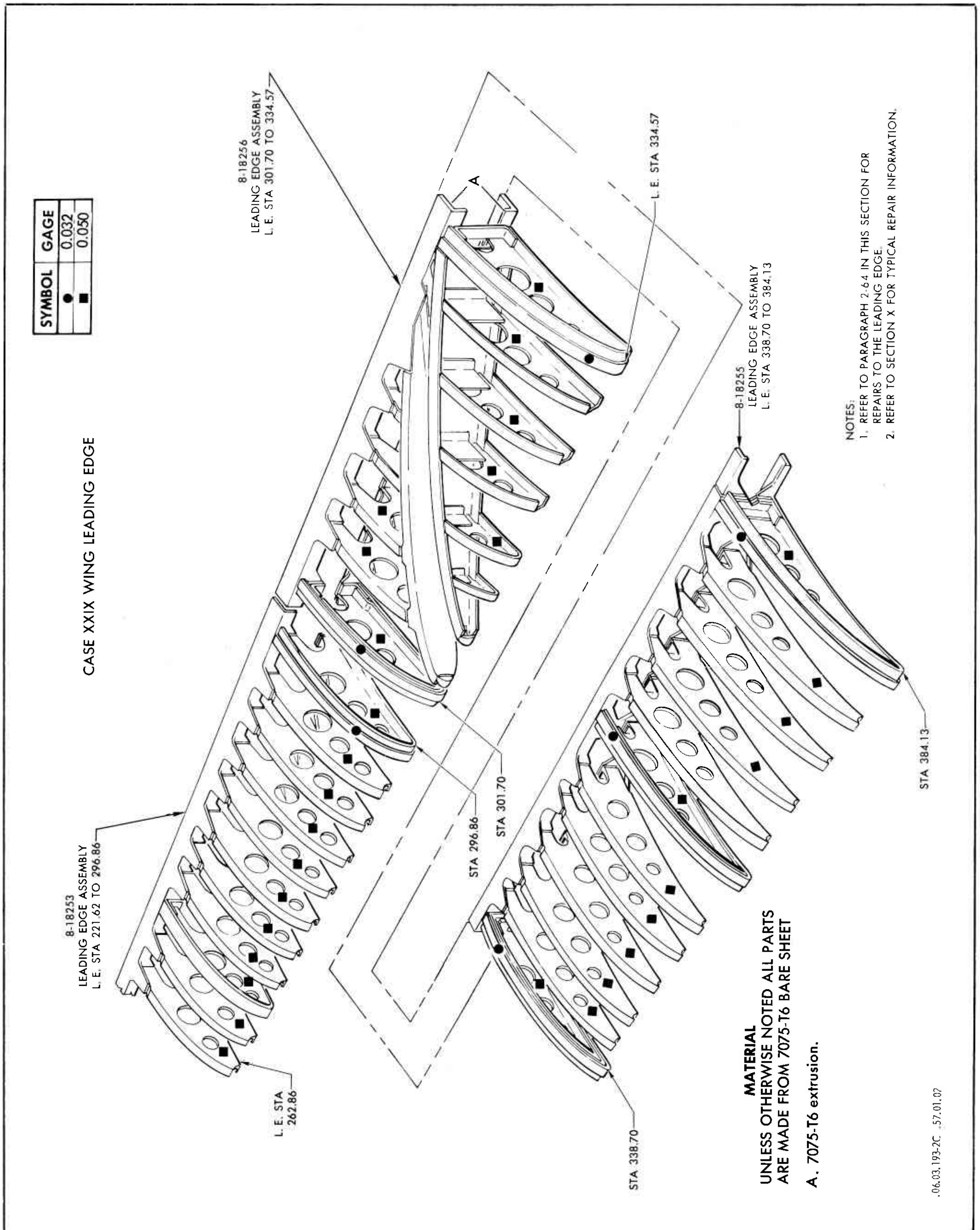


Figure 2-10. Case XXIX Wing Leading Edge Structure (Sheet 2 of 2)

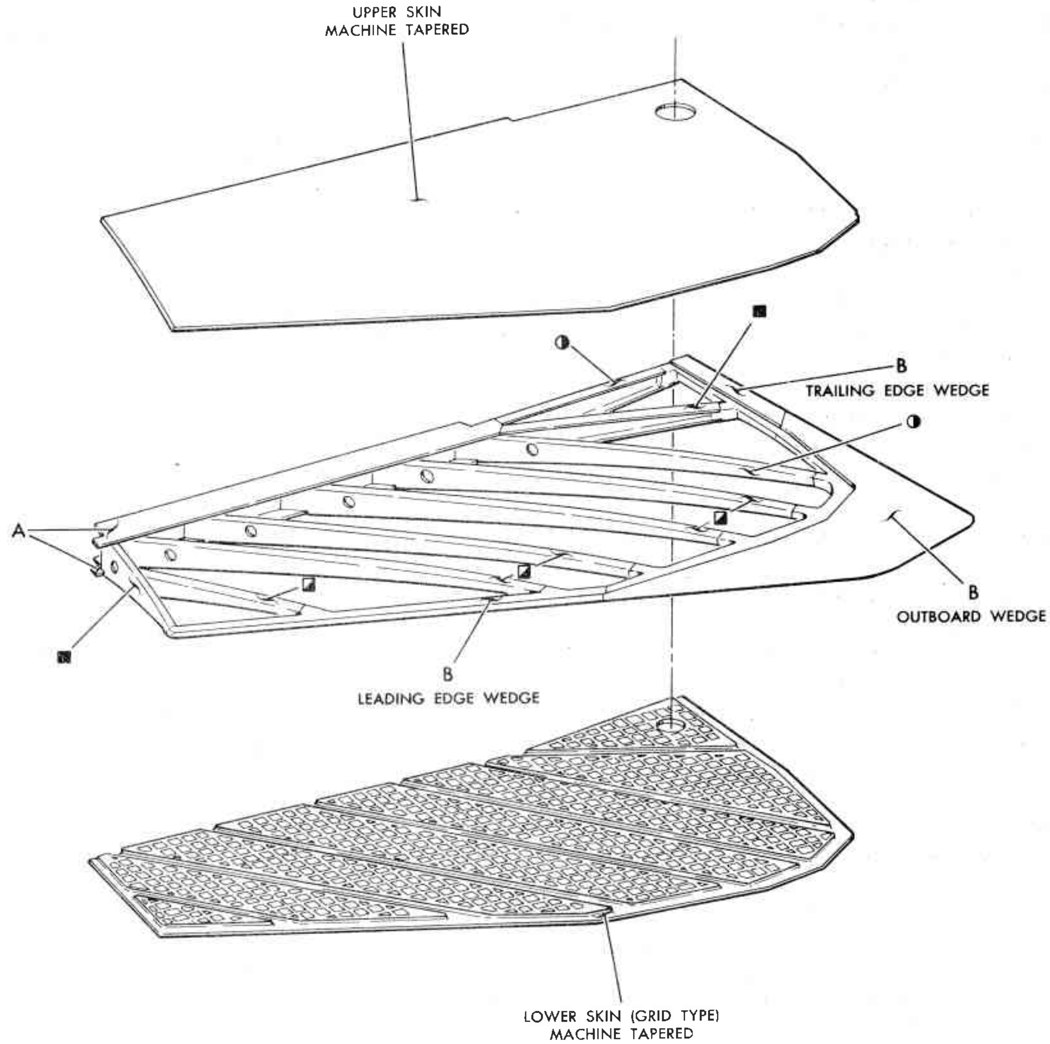
Applicable to F-106A airplanes, 57-230 and subsequent, and F-106B airplanes 57-2508 and subsequent

MATERIAL

UNLESS OTHERWISE NOTED ALL PARTS ARE MADE FROM 7075-T6 BARE SHEET.

A. 7075-T6 extrusion.
B. AZ91 magnesium alloy casting.

SYMBOL	GAGE
■	0.051
◑	0.064
◪	0.072



NOTE:
REFER TO PARAGRAPH "WING TIP REPAIR" FOR REPAIRS.

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Figure 2-11. Case XIV Wing Tip Structure and Plating
Applicable to F-106A airplanes 56-453 thru 56-466, and F-106B airplane 57-2507

with clips. The structure is enveloped by chemically milled, grid type, 7075-T6 aluminum alloy plating. The wing tip assembly is attached by screws through nut plates attached to the inside flange of the wing rib wing tip support. This type of wing tip is used on *F-106A airplanes 56-453 through 56-466 and on F-106B airplane 57-2507.*

2-16. The wing tip attached to the Case XXIX wing, except for an increase in camber and minor structural differences, is similar to the wing tip used on the Case XIV wing. These wing tips are not interchangeable from one type of wing to the other. See figure 2-12 for an illustration of the Case XXIX wing tip. This wing tip is used on *F-106A airplanes 56-467, 57-229 and subsequent, and on F-106B airplanes 57-2508 and subsequent.*

2-17. Elevon.

2-17A. The elevon structure is divided into an inboard and an outboard section, as shown on figures 2-13 and 2-14. The primary structural portion of each section consists of an aluminum alloy torque box covered with machine tapered plating. Each section also incorporates a metal honeycomb trailing edge. The two elevon sections are attached to number 7 spar by means of forged aluminum hinge fittings. Each section attaches at four locations, although there are only seven hinge fittings. The middle hinge fitting is common to both sections. The elevon chordwise, press-formed ribs are reinforced by a heavy gage spar to provide added rigidity at points of attachment to the wing trailing edge spar assembly. The two sections are bolted together through a steel link between aluminum forged fittings riveted to the elevon ribs at elevon buttock line 107.50. The link permits a slight horizontal flexing to take place between the two sections back of the common hinge fitting. The gap between the two sections is filled by a teflon plastic chafing strip which provides aerodynamic smoothness not only in a nonflexed condition but also during flexing.

2-18. INSPECTION AND REPAIR OF SURFACES CONTAINING HONEYCOMB CORE.

a. Inspect, dehydration, repair, and seal surfaces containing honeycomb core, applicable to trailing edge wedge.

(1) X-ray elevon assembly in accordance with T.O. 1F-106A-36 to determine if moisture is present, core is damaged or corroded.

NOTE

Non-perforated cores do not require dehydration.

(2) If moisture is present, remove elevon from aircraft in accordance with T.O. 1F-106A-2-2-2-2 and T.O. 1F-106A-2-7-2-1.

(3) Utilizing a "Q" drill bit (0.3320 diameter) drill hole in the core assembly inboard end rib on the centerline of symmetry at least 3 inches aft of spar.

CAUTION

Use drill stop to assure that bit does not penetrate more than 1/8 inch into honeycomb core.

(4) Tap hole with 3/8–24 tap and install union fitting AN815-3 (FSN 4730-187-0483) in tapped hole using sealant MIL-S-8802 class B.

(5) Utilizing low temperature boil procedure, dehydrate assembly using set moisture removal FSN 4020-000062F Vacuum shall be between 10 PSIG to 12 PSIG to assure boiling of water at as low as 160°F and temperature shall not exceed 180°F. 4 to 8 hours may be required for dehydration of core assembly, dependent on the amount of water in the core.

NOTE

A vacuum chamber capable of safely maintaining 12.3 PSIG vacuum at 160° to 180°F with adequate safety provisions to assure preventing excess temperature or collapsing from excess vacuum may be used as a suitable substitute.

(6) Re-X-ray assembly to determine that all moisture has been removed.

(7) When all moisture has been removed from assembly or x-ray reveals no moisture present, repair any damage in accordance with applicable procedures outlined in this technical order.

(8) Test and seal all assemblies as follows:

(a) Connect air line to fitting (Part No. AN815-3)

(b) Pressureize assembly between 10 to 12 PSI with dry oil free air and submerge in water to locate leaks.

(c) Mark leaks with black grease pencil or suitable substitute.

(d) Seal all leaks with sealant MIL-S-8802 class B 1/2. Surfaces to be sealed shall be cleaned with a clean cloth and methylethyl-ketone prior to applying sealant.

NOTE

To assure that sealant penetrates into all openings, apply sealant while assembly is subjected to a slight vacuum (approximately 7 PSIG), after sealant has cured, recheck assembly (submerged) and repeat sealing process until all leaks are sealed.

(e) If leaks are indicated along the core assembly front spar, remove skin that is closed out with blind fasteners (upper or lower) and repeat steps (a) thru (d).

(9) Remove union fitting (Part No. AN815-3) and install plug, machine thread FSN 4730-529-4871 or suitable substitute with sealant MIL-S-8802 class B.

(10) Reinstall panel assembly skin if removed, using solid rivets except the last seven rivet on each rib and the double row on aft edge of skin which are blind fasteners. (Refer to drawings 8-13031 and 8-13062).

(11) Identify sealed assemblies by stenciling "SEALED" and date on inboard rib of elevon.

(12) Reinstall surfaces on aircraft in reverse order of removal.

2-18A. Removal and Installation of Elevon Trailing Edge Wedges.

a. Removal of elevon wedges:

(1) Remove skin closed out with blind fasteners from elevon assembly by drilling out all attaching rivets.

NOTE

Do not remove fairing from elevon trailing edge (wedge) unless a new trailing edge replacement is required.

(2) Drill out rivets (double row) attaching lower skin to trailing edge.

(3) Drill out rivets attaching actuator fairing to elevon skin.

(4) Remove chromate primer from ribs to facilitate locating all rivets heads.

(5) Remove rivets attaching trailing edge to elevon ribs.

(6) Remove inboard and outboard elevon trailing edges. If trailing edges do not slip out freely, inspect for additional rivets.

b. Test and seal all elevon trailing edge wedges in accordance with paragraph 2-18.

(1) Replace wedges in reverse order or removal.

(2) To identify sealed elevon assemblies, paint or stencil "SEALED" and the date on inboard rib of elevon assembly.

c. Install new trailing edges as follows:

(1) Place inboard and outboard trailing edges in position on elevon.

(2) Align the trailing edges.

(3) Flush trailing edge ends with the elevon ends.

(4) Maximum permissible gap between trailing edge and elevon upper and lower skins is 0.200 inch.

(5) Match inboard and outboard trailing edges.

(6) Match drill holes in lower skin through new trailing edges.

(7) Match drill holes in ribs through clips of new trailing edges.

(8) Replace trailing edge rivets in reverse order of removal, except use blind rivets, FA 200 in lieu of rivets removed along the aft surface of the trailing edge.

d. Reinstall surfaces on aircraft in reverse order of removal.

2-19. Elevon Fairing.

2-20. The elevon actuator wing fairing is composed of a forward and an aft section. The forward section is attached to the wing and the aft section is attached to the elevon. The forward section consists of an end assembly attached with screws to the wing structure, two adjacent extruded angles fitted with plate nuts attached to wing structure, ribs attached to wing structure, and three removable sections of skin which are attached to the contour ribs and angles with screws. The aft section is composed of ribs and angles attached to elevon structure and covered by one piece of skin which is riveted to the angles and ribs.

2-21. External Fuel Tank.

2-22. The external fuel tank is a welded assembly equipped with lugs for attachment to a fixed pylon. The fixed pylon, with installed tank ejection rack, provided attachment for the pylon fairings. See figure 2-15A for the different types of material used in the external tank and pylon.

2-23. INDEXING.

2-24. Figure 2-1 indexes the various wing components by reference to figure number. Refer to applicable figure number for information concerning repairs to the wing components.

2-25. REPAIRS.

2-26. Repair information is given on figures 2-4 and 2-16 through 2-31. Most repairs to the wing structure which do not involve the Scotchweld tape or wing plating integral rib areas may be made in the conventional manner. All repairs in the fuel tank area will require sealing to provide fuel-tight seams. Figure 2-31 and paragraphs 2-35 through 2-50 describe the method for repairing wing plating areas which are bonded with Scotchweld tape. See figures 2-23, 10-1, 10-5, and 10-7 for repairs applicable to the components of the intermediate box area and outer trailing edge structure. To remove moisture from the honeycomb sections of the elevon, follow the moisture removal procedure given on figure 2-22. One-time flight or temporary repairs may be made in case of emergency, and are described on figure 2-29. Due to the restrictions which may be imposed on the strength factors of the components involved and the flight characteristics of the airplane, one-time flight or temporary repairs should not be employed except by prior structural engineering approval. Although the delta-form wing is much less critical aerodynamically than conventional types, maximum performance and range cannot be realized with outer surface discontinuities. The range of an airplane restricted to low speeds because of a considerable number of outside repairs could easily be reduced to less than half of the maximum obtainable under the most favorable conditions. Prescribed limits for negligible damage to the various components of the wing structure are shown in Table 2-I.

2-27. Negligible Damage Limits — Wing Group.

2-28. Negligible damage is damage allowed to remain "as is," after minor rework such as stop-drilling cracks and fairing nicks or scratches. Table 2-I indicates the maximum allowable classification of five types of negligible damage. The maximum allowable damage classification will be found to the right of the component name in the vertical column under the "Type of Damage" heading. After classification is determined, see figures 1-17 through 1-19 for the damage limits allowed for each class: I, II, or III. The limits given on figures 1-17 through 1-19 apply only for a damaged area after rework, as shown on figures 1-20 and 1-21. An aeronautical engineer must be consulted for damage exceeding the limits given on figures 1-17 through 1-19, and for damage to components not listed in this table.

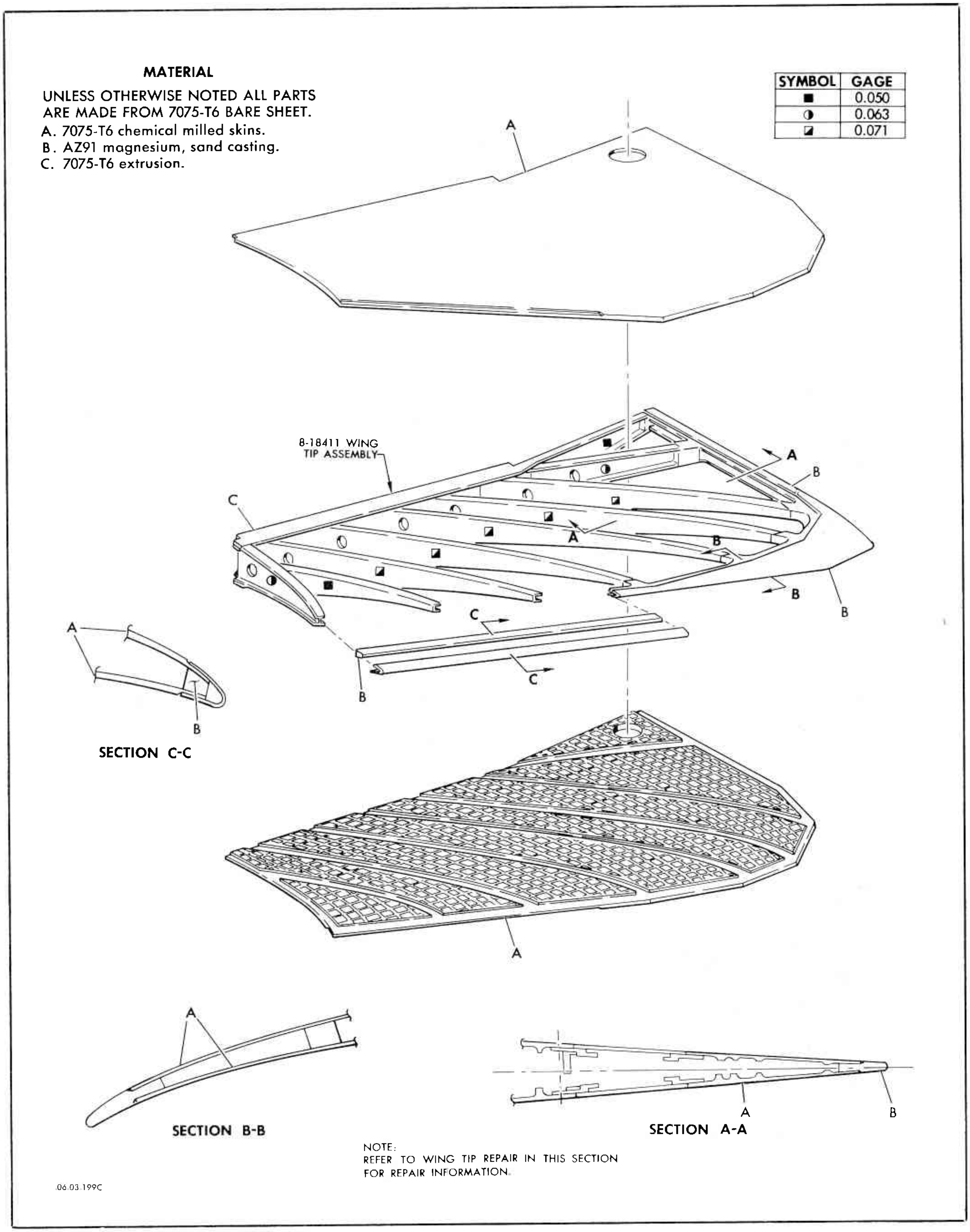
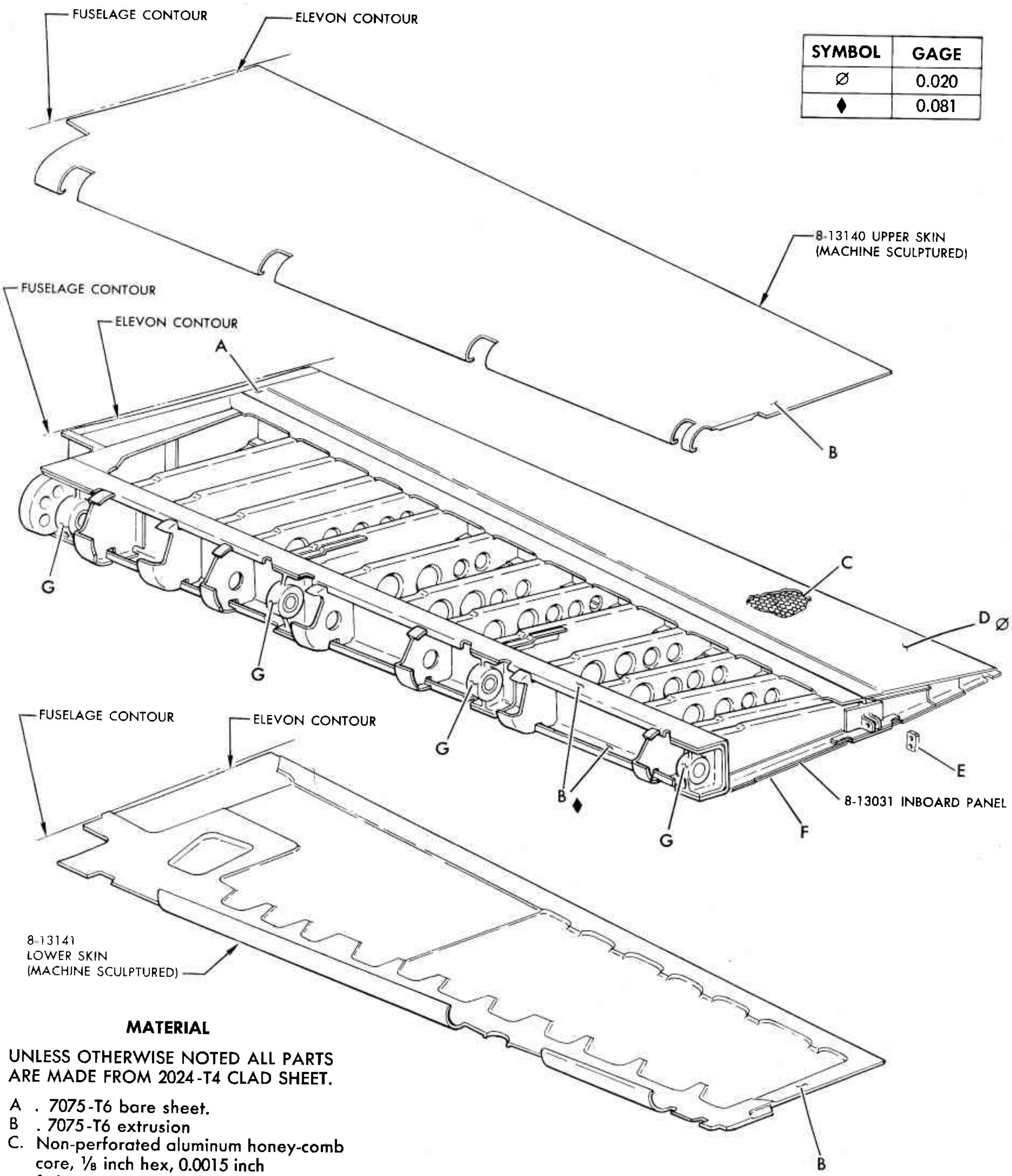


Figure 2-12. Case XXIX Wing Tip Structure and Plating

Applicable to F-106A airplanes 56-467, 57-229 and subsequent, and F-106B airplanes 57-2508 and subsequent



SYMBOL	GAGE
∅	0.020
♦	0.081

MATERIAL

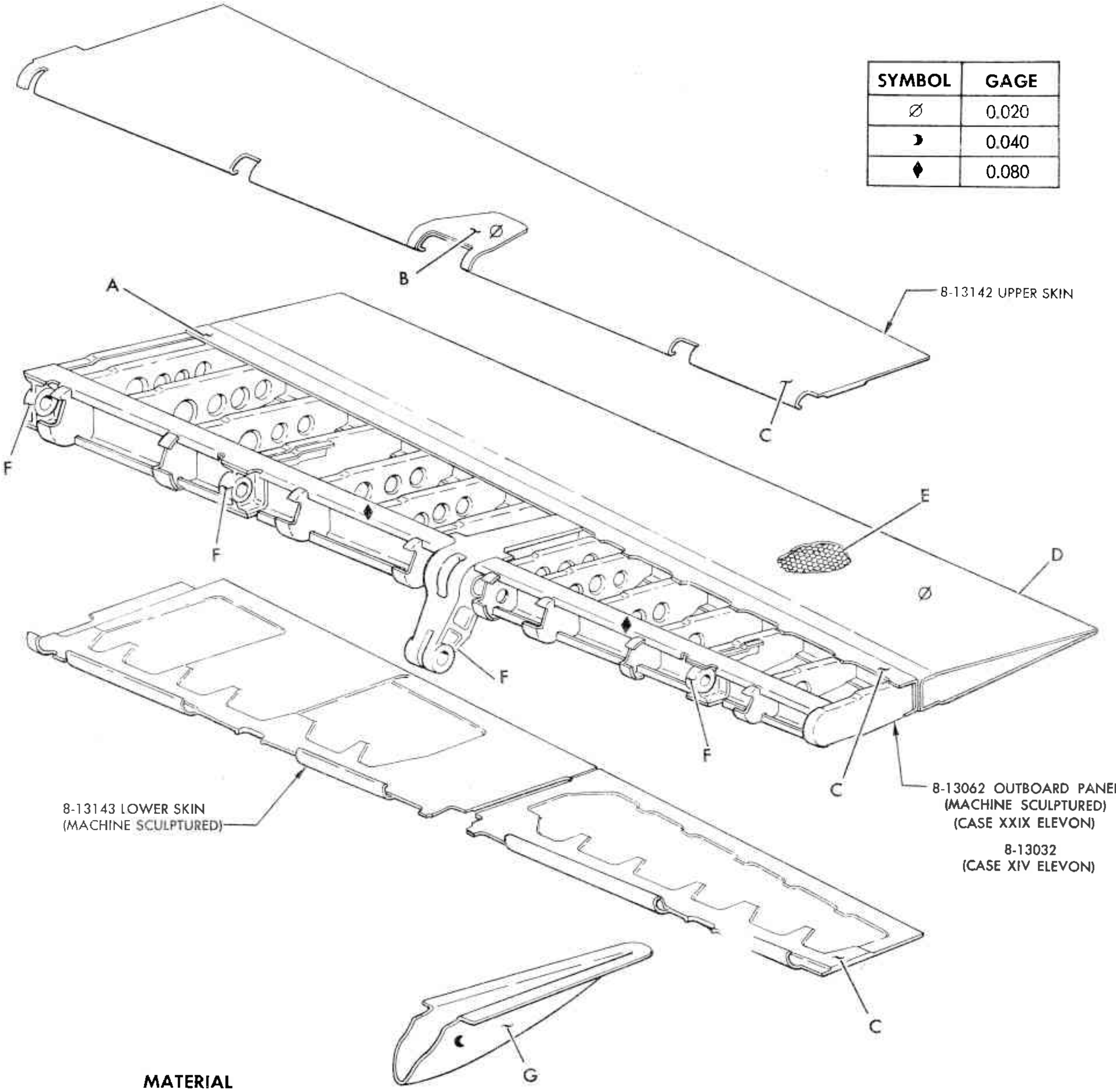
UNLESS OTHERWISE NOTED ALL PARTS ARE MADE FROM 2024-T4 CLAD SHEET.

A . 7075-T6 bare sheet.
B . 7075-T6 extrusion
C. Non-perforated aluminum honey-comb core, 1/8 inch hex, 0.0015 inch foil.
D. 2024-T6 clad sheet.
E. SAE 4130 steel.
F. AMS3150 Teflon.
G. 7075-T6 machined forging.

NOTE:
REFER TO PARAGRAPH "ELEVON REPAIRS" IN THIS SECTION FOR REPAIR INFORMATION.

06.03.2018

Figure 2-13. Elevon Structure and Plating — Inboard Panel



SYMBOL	GAGE
∅	0.020
❩	0.040
♦	0.080

MATERIAL

UNLESS OTHERWISE NOTED ALL PARTS ARE MADE FROM 2024-T4 CLAD SHEET.

A. 7075-T6 bare sheet.
B. 302 corrosion resistant steel.
C. 2024-T6 clad sheet.
D. 7075-T6 bare sheet.
E. Non-perforated aluminum honey-comb core, 1/8 inch hex, 0.0015 inch foil.
F. 7075-T6 machined forging.
G. 7075-T6 clad sheet.

NOTE:
REFER TO PARAGRAPH "ELEVON REPAIRS" IN THIS SECTION FOR REPAIR INFORMATION.

06.03.202C

Figure 2-14. Elevon Structure and Plating—Outboard Panel

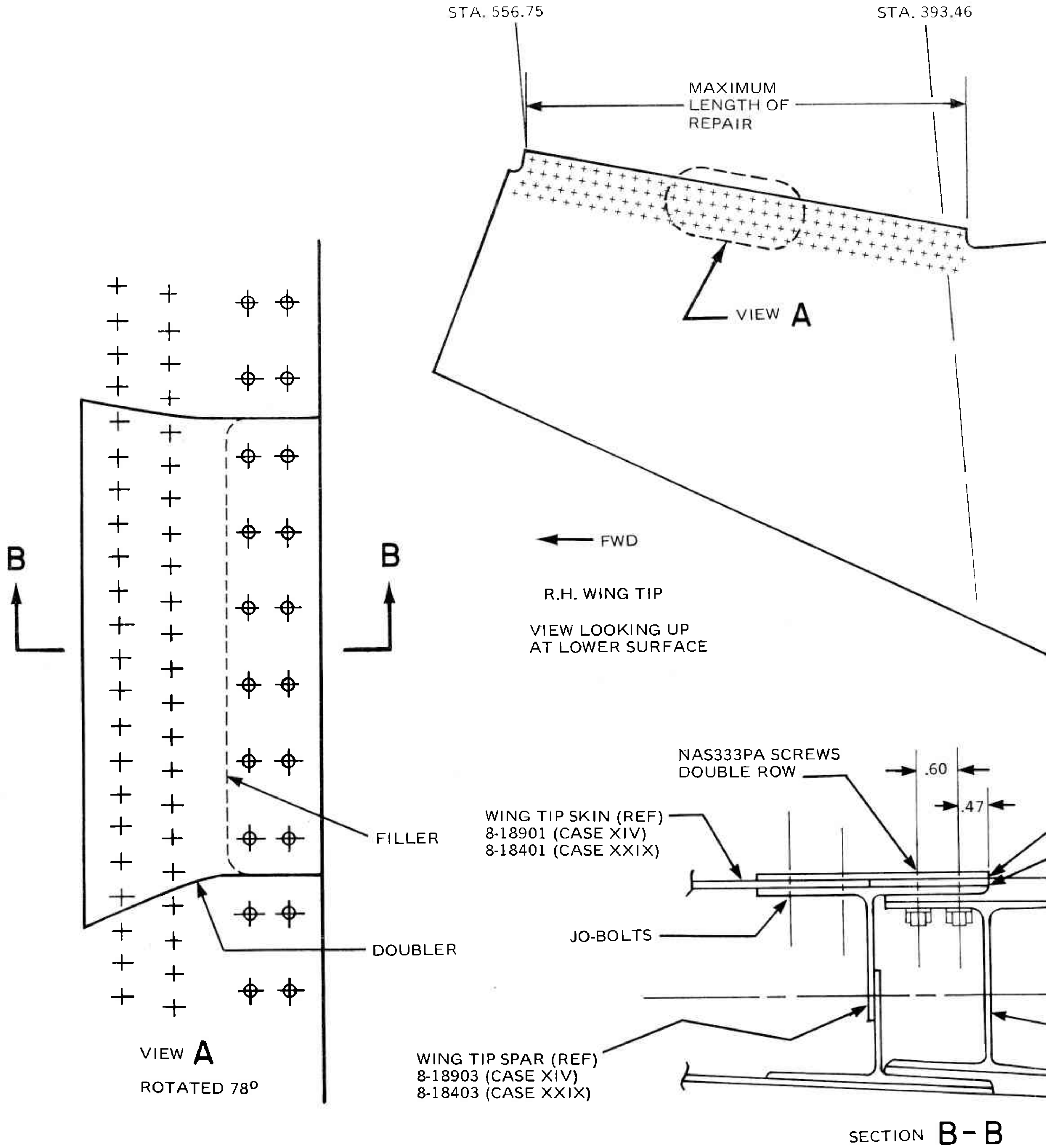
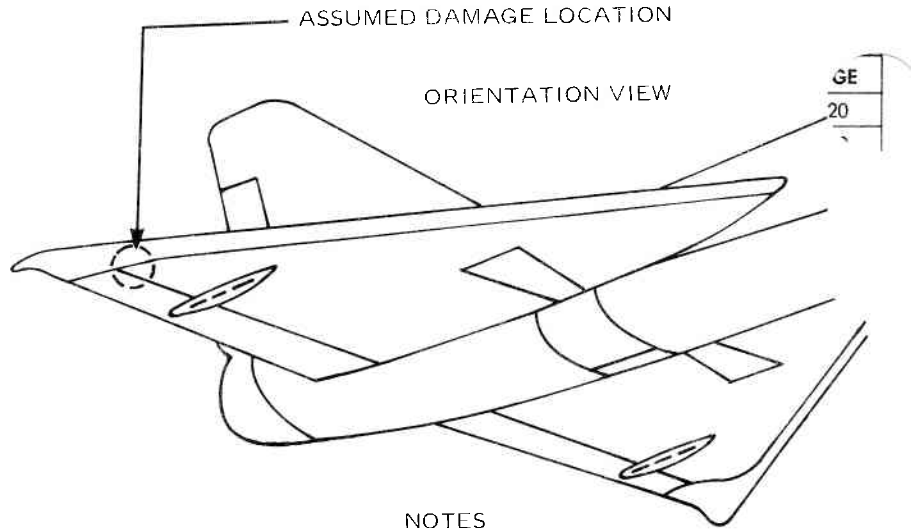
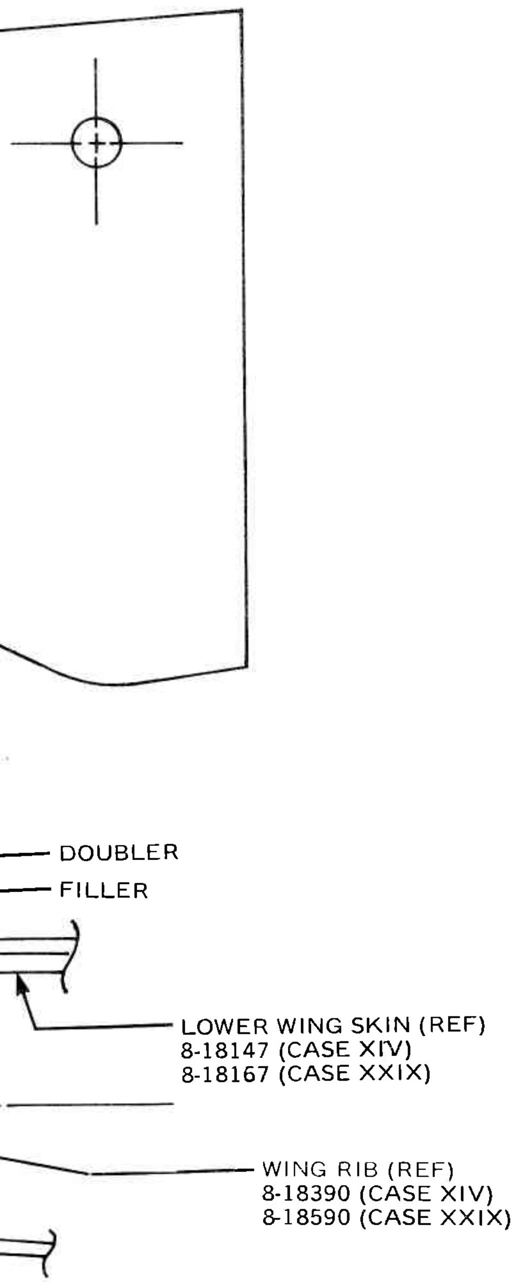


Figure 2-15. Wing Tip Repair





NOTES

A. THIS REPAIR IS A TEMPORARY REPAIR FOR TIPS HAVING ILLEGAL REPAIRS, CORROSION AND DEEP GRINDOUTS ON THE LOWER SKIN. REPAIRED TIPS WILL BE REPLACED AS SOON AS SPARE TIPS ARE AVAILABLE.

B. REMOVE THE FILLER STRAP, OR CUTOUT DAMAGED AREA SUPPORTED BY COUNTERSUNK SCREWS. RETAIN STRAP FOR TRANSFERING SCREW HOLES TO NEW FILLER STRAP AND DOUBLER.

C. FABRICATE DOUBLER FROM 7075 T6 BARE SHEET 0.080 THICK AND WIDE ENOUGH TO PICK UP FOUR ROWS OF FASTENERS. MAXIMUM LENGTH OF DOUBLER IS FULL LENGTH OF WING TIP.

D. FABRICATE NEW FILLER FROM 7075-T6 BARE SHEET SAME SIZE AS OLD FILLER OR CUT OUT.

E. INSTALL NEW FILLER AND DOUBLER AND DRILL HOLES TO MATCH EXISTING HOLES IN WING TIP. MACHINE COUNTERSINK HOLES IN DOUBLER ONLY.

F. FILL IN COUNTERSINKS IN SKIN USING THE COUNTERSUNK HEADS FROM REMOVED JO-BOLTS OR COUNTERSUNK RIVET HEADS. BOND TO COUNTERSINKS WITH ADHESIVE TO PREVENT SPINNING DURING DRILLING.

G. REFER TO T.O. 1F-106A-23 FOR FINISH REQUIREMENTS.

H. INSTALL DOUBLER AND FILLER USING MIL-S-81733 FAYING SURFACE SEALANT AND MILLABLE COUNTERSUNK JO-BOLTS.

I. FILLET ALL EDGES WITH MIL-S-38228 FAIRING COMPOUND AFTER WING TIP IS INSTALLED ON AIRCRAFT. SEE FIGURE 1-27.

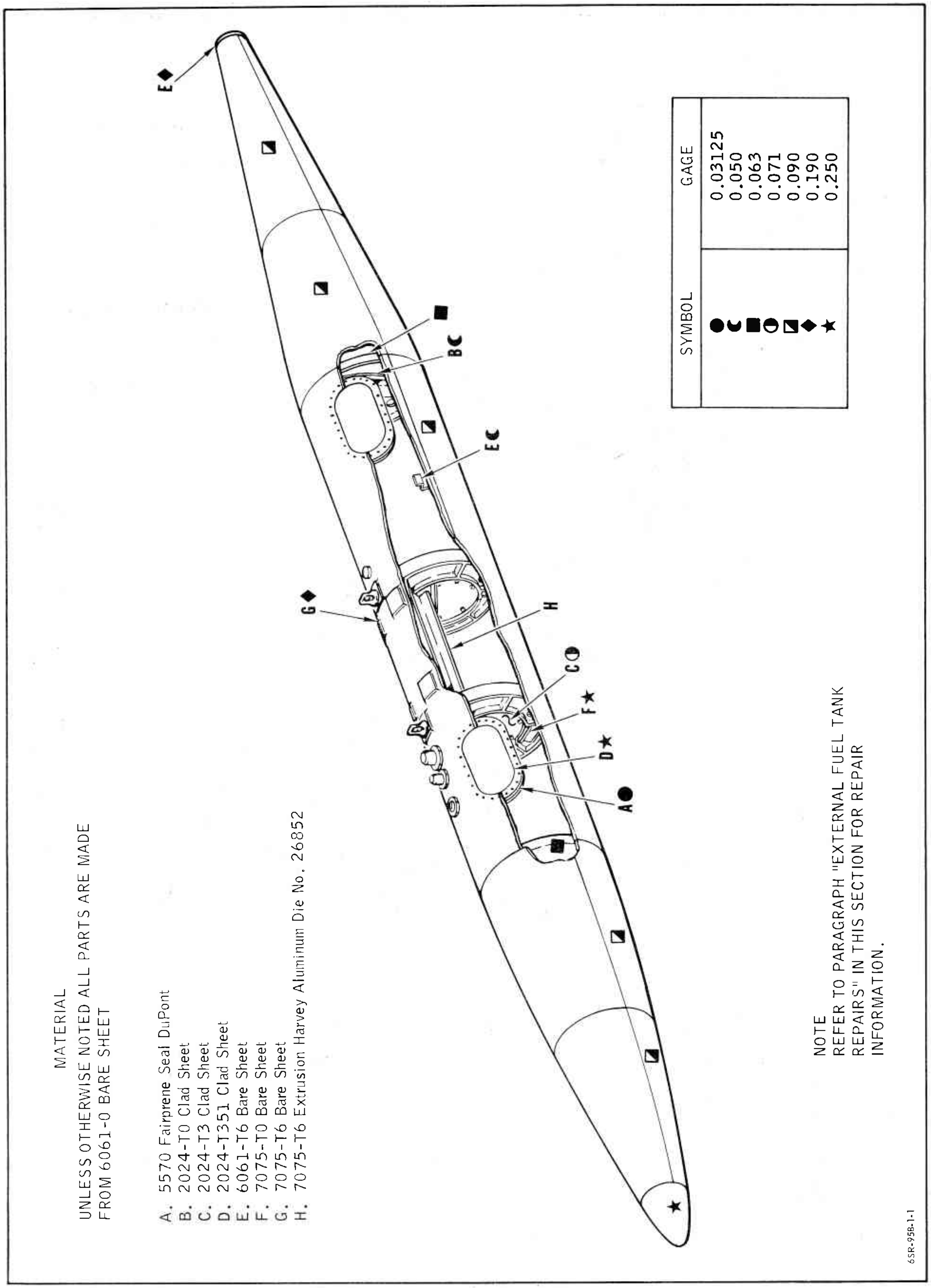


Figure 2-15A. External Fuel Tank and Pylon (Sheet 1 of 2)
Applicable after incorporation of TCTO 1F-106-958

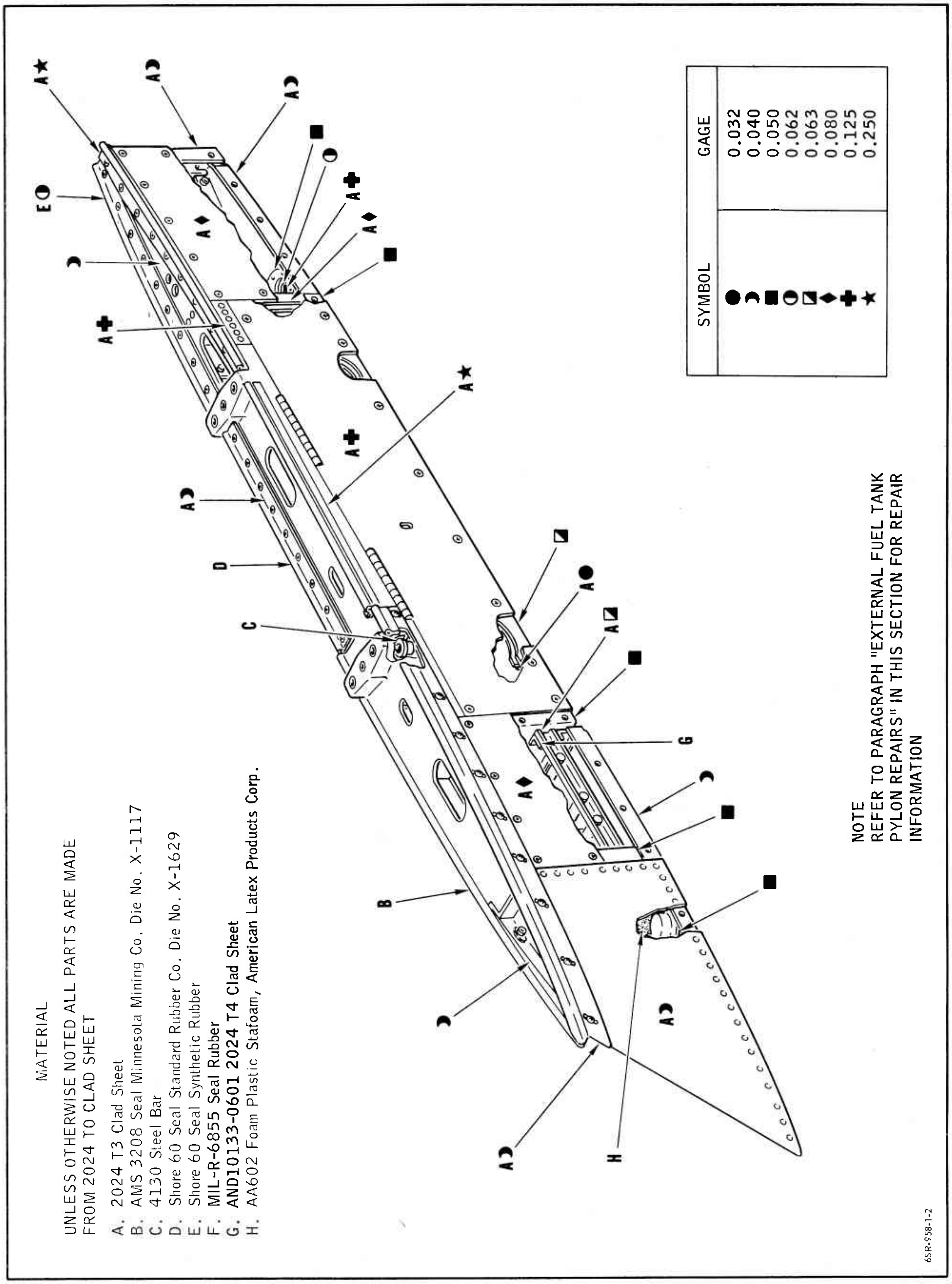
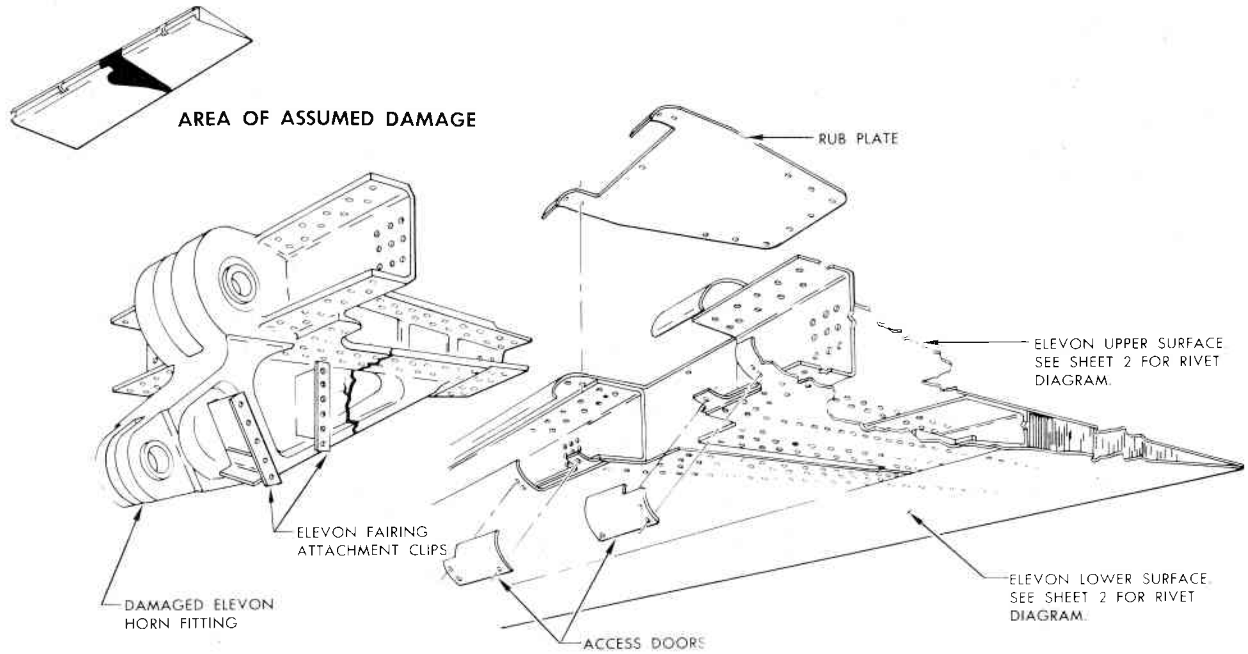


Figure 2-15A. External Fuel Tank and Pylon (Sheet 2 of 2)
Applicable after incorporation of TCTO 1F-106-958



REMOVAL OF DAMAGED HORN FITTING

a. Remove damaged elevon from wing in accordance with procedures outlined in T.O. 1F-106A-2-7-2-1.
b. Place damaged elevon on work stand or bench which has been suitably padded to protect elevon skins.
c. Remove all rivets connecting horn fitting to upper surface.
d. Remove all rivets connecting rub plate to upper surface. Remove rub plate and hold for reinstallation.
e. Remove access doors (2) from lower surface of elevon by extracting NAS333 PA4 screws (3 each door). Hold doors and screws for reinstallation.
f. Remove all rivets connecting elevon fairing to support angles on side of horn fittings.
g. Remove all rivets connecting elevon fairing to horn fitting and to lower surface. Remove elevon fairing and hold for reinstallation.
h. Remove enough rivets each side of elevon horn fitting from lower surface to allow insertion of a micarta wedge.
j. Insert micarta wedge between elevon lower surface plating and internal structure. Micarta wedge should be of sufficient thickness to hold elevon plating away from internal structure so that mechanic may work within the structure.
k. Remove two NAS464-4A-5 bolts and two AN365-428 nuts which connect horn fitting to support tee in elevon aft spar.
l. Remove micarta wedges and remove horn fitting.

INSERTION OF REPLACEMENT HORN FITTING

a. Locate and drill two (2), 0.2500 (−0.0000, +0.0020) inch diameter bolt holes in replacement horn fitting as shown in view A.
b. Insert replacement fitting in elevon.
c. Insert micarta wedges between lower surface elevon plating and internal structure.
d. Install two (2) NAS464-4A-5 bolts and two (2) AN365-428 nuts through support tee in elevon aft spar and through replacement horn fitting.
e. Locate replacement horn fitting buttock line wise by using inside micrometer. See figure 2-44 for horn fitting location and use of inside micrometer.
f. Remove micarta wedge and clamp replacement fitting in place.
g. Drill rivet holes in replacement horn fitting using holes in elevon plating as a guide.
h. Remove clamps and insert micarta wedge between lower surface elevon plating and internal structure. Remove two (2) NAS464-4A-5 bolts and two (2) AN365-428 nuts and remove replacement horn fitting.
i. Remove all burrs and chips from replacement horn fitting and elevon.

NOTE
REFER TO FINISH REQUIREMENTS IN SECTION I FOR MATERIAL FINISHES PRIOR TO FINAL INSTALLATION OF REPLACEMENT HORN FITTING.

j. Rivet elevon fairing support clips to horn fitting. (Refer to engineering blueprint No. 8-18560 for support clip location and method of attachment.)
k. Install replacement horn fitting. Reinstall two (2), NAS464-4A-5 bolts and two (2) AN365-428 nuts. (Install bolt head on outboard side).

NOTE
REFER TO SECTION I FOR STANDARD BOLT TORQUE VALUES.

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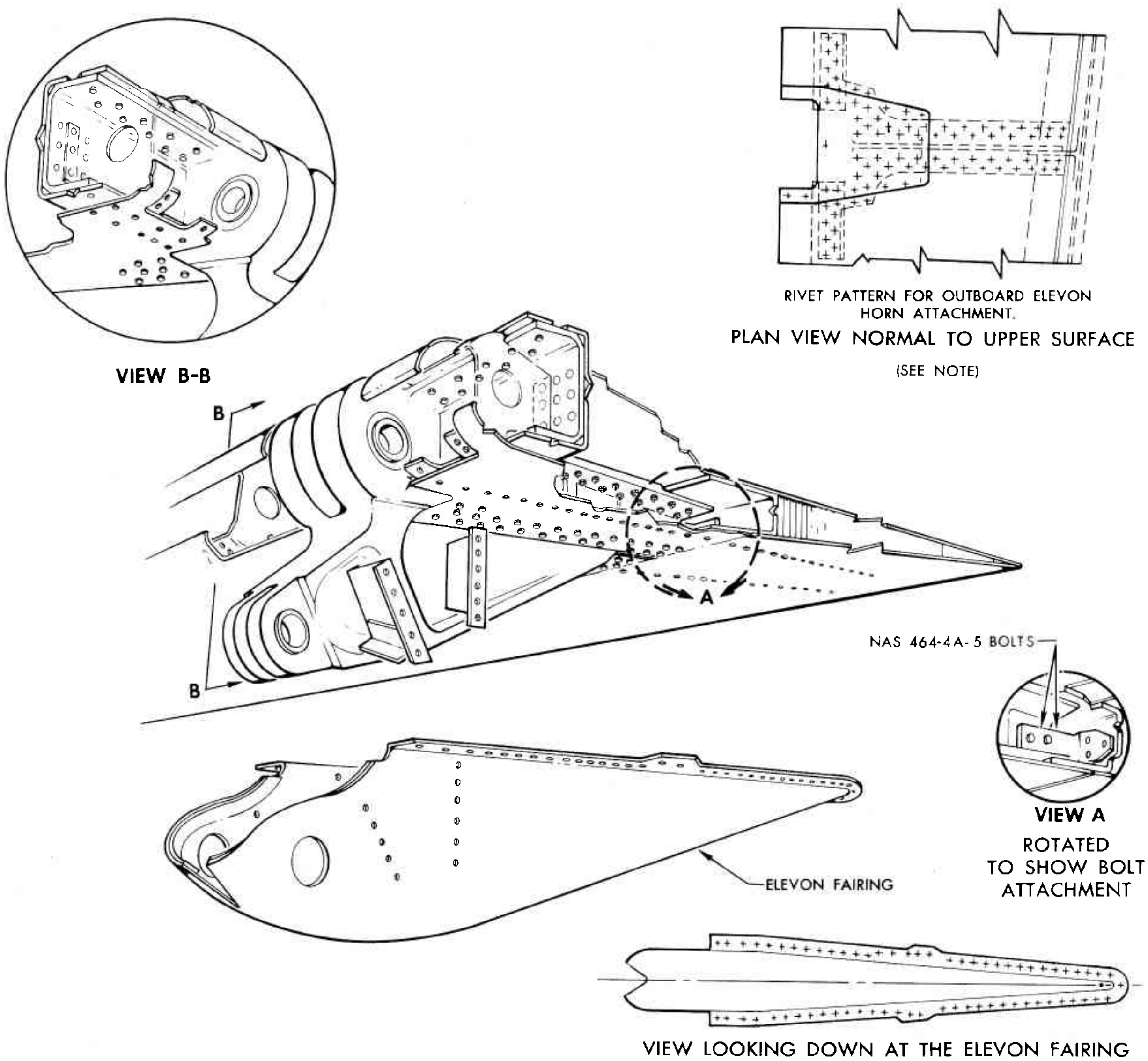
Figure 2-16. Replacement of Outboard Elevon Actuator Horn (Sheet 1 of 2)



l. Remove micarta wedge; then align drilled holes in horn fitting with holes in elevon plating and clamp in place.
m. Spot rivet five (5) places in upper and lower surfaces to prevent horn fitting from slipping out of place during riveting process.
n. Remove clamps and finish riveting on upper and lower surface.
o. Reposition nut plate on upper surface and install rivets.
p. Reposition elevon fairing on lower surface and hold in place with Clecos.
q. Use existing holes in elevon fairing as a guide to drill holes in elevon fairing support clips.
r. Rivet elevon fairing to support angles on horn fitting.
s. Rivet elevon fairing to lower elevon surface.
t. Replace elevon nose doors with NAS333-PA4 screws.
u. Reinstall elevon to wing in accordance with procedures outlined in T.O. 1F-106A-2-7-2-1.

NOTE:
REFER TO ENGINEERING DRAWING NO. 8-13032 FOR LOCATION AND TYPE OF RIVETS USED IN THE ELEVON.

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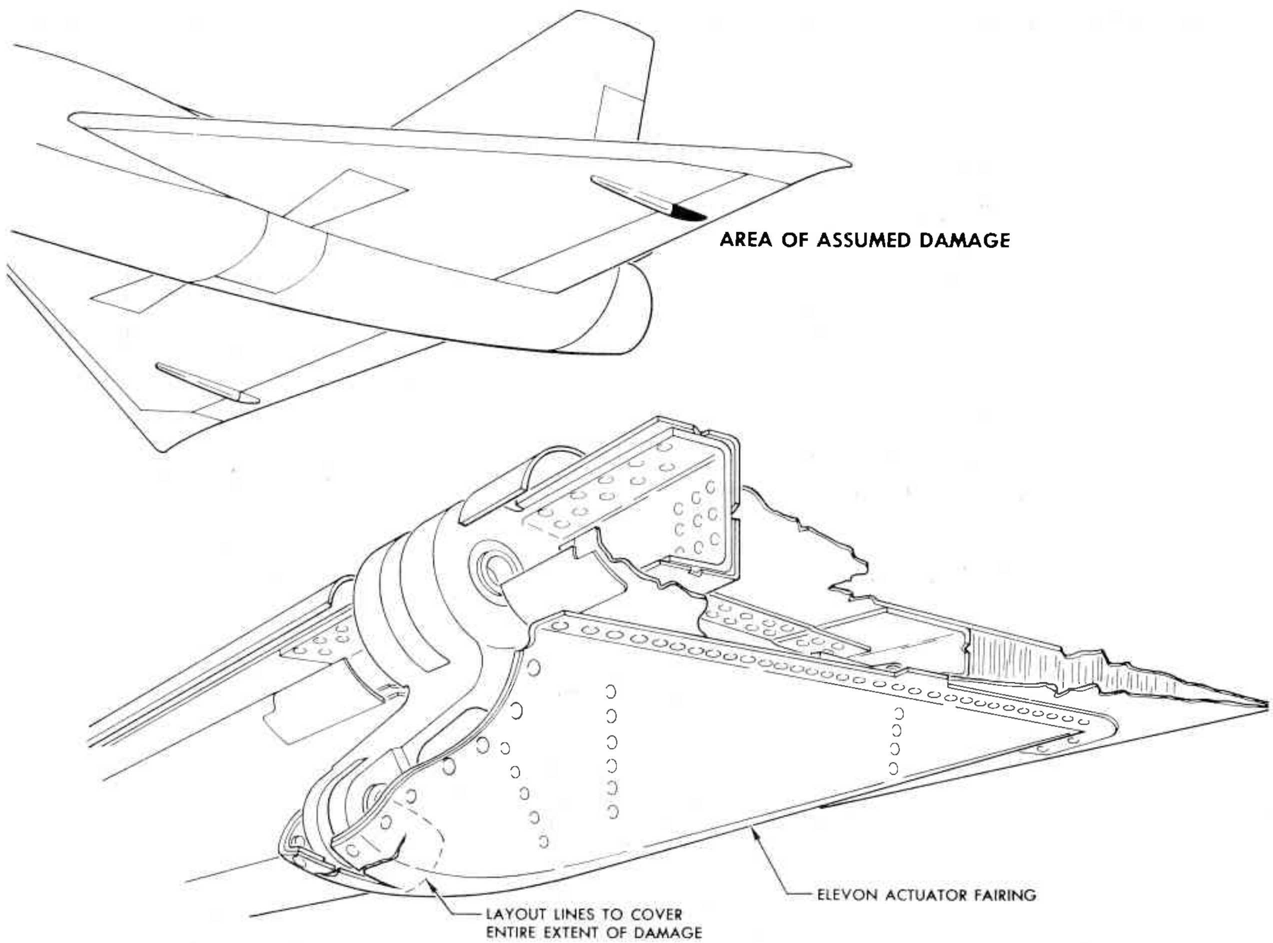
Figure 2-16. Replacement of Outboard Elevon Actuator Horn (Sheet 2 of 2)



REPAIR PROCEDURE

a. Move the elevon to extreme up position and place a padded support beneath the elevon.
b. Remove rivets attaching actuator fairing to elevon skin.
c. Remove rivets attaching actuator fairing to actuator clips.
d. Remove actuator fairing.
e. Remove all paint and primer from damaged area of actuator fairing. Use Methyl Ethyl Ketone, Specification TT-M-261, to remove paint and primer.
f. Perform Fluorescent penetrant inspection on damaged actuator fairing. Refer to Section I for Fluorescent penetrant materials and inspection procedures.
g. Cut out damaged portion of actuator fairing.
h. Fabricate a doubler from 0.050, 7075-T6 bare sheet to dimensions shown in Detail A.
i. Fabricate a flush skin patch from 0.040, 7075-T6 bare sheet to dimension shown in Detail B.
j. Treat repair parts and bare portion of actuator fairing with primer. See Primer and Paint Coating in Section I.
k. Predrill the doubler and flush skin patch as shown using a No. 30 (0.1285) drill.
l. Install the doubler as shown and hold in place with clamps.
m. Using the predrilled holes as a guide, drill holes through doubler and actuator fairing. Use a No. 30 (0.1285) drill.
n. Remove the doubler and remove all drill chips and burrs from doubler and actuator fairing.
o. Reinstall doubler and hold in place with clecos.
p. Ream holes in doubler and actuator fairing with a No. 20 (0.161) drill.
q. Heat dimple holes in actuator fairing and doubler for 5/32 countersunk rivets. Refer to Section I for flush rivet installation procedure.
r. Rivet doubler to actuator flaring with BB rivets. Refer to Table 1-XXIII for basic rivet code.
s. Install flush skin patch and hold in place with clamps.
t. Using predrilled holes in flush skin patch as a guide, drill holes through patch and repair doubler. Pick up holes from existing doubler. Use a No. 30 (0.1285) drill.

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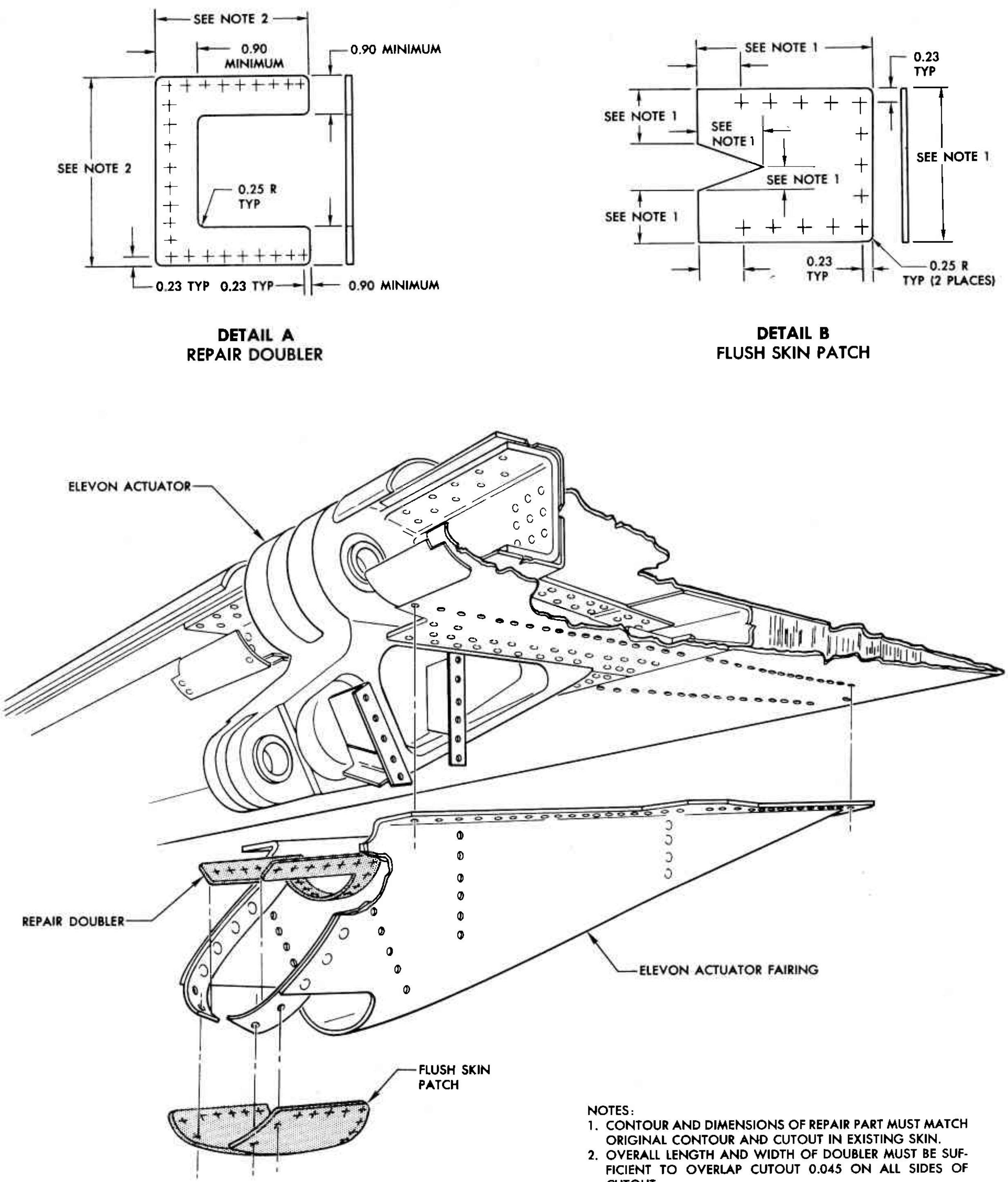
Figure 2-17. Elevon Actuator Aft Fairing Repair (Sheet 1 of 2)



NOTES:
1. CONTOUR AND DIMENSIONS OF REPAIR PART MUST MATCH ORIGINAL CONTOUR AND CUTOUT IN EXISTING SKIN.
2. OVERALL LENGTH AND WIDTH OF DOUBLER MUST BE SUFFICIENT TO OVERLAP CUTOUT 0.045 ON ALL SIDES OF CUTOUT.

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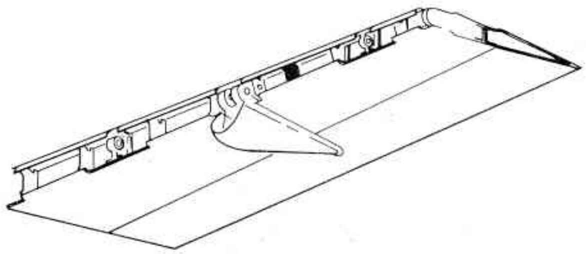
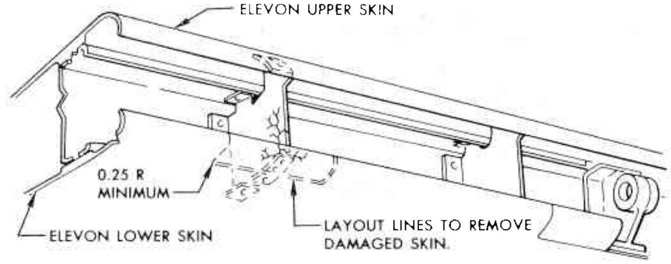
Figure 2-17. Elevon Actuator Aft Fairing Repair (Sheet 2 of 2)

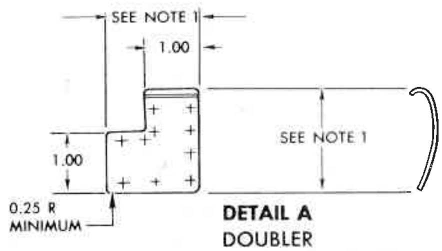


AREA OF ASSUMED DAMAGE

NOTES:
1. OVER-ALL DIMENSIONS OF DOUBLERS MUST BE DETERMINED BY A PHYSICAL MEASUREMENT.
2. DIMENSIONS OF FLUSH SKIN PATCH MUST MATCH CUTOUT IN EXISTING SKIN. REFER TO AERODYNAMIC SMOOTHNESS REQUIREMENTS IN SECTION I FOR ALLOWABLE GAP AND MISMATCH.
3. DIMENSIONS FOR NEW RIB MAY BE OBTAINED FROM DAMAGED RIB AND BY PHYSICAL MEASUREMENT OF AREA VACATED BY REMOVED DAMAGED RIB.



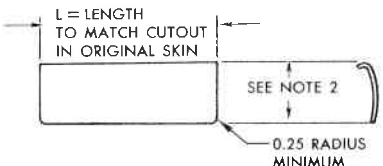
ENLARGED AREA OF ASSUMED DAMAGE



DETAIL A

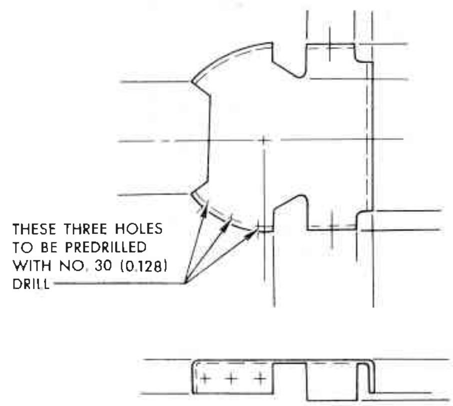
DOUBLER

LEFT HAND SHOWN, RIGHT HAND OPPOSITE



DETAIL B

FLUSH SKIN PATCH



DETAIL C

RIB REPAIR

(SEE NOTE 3)

REPAIR PROCEDURE

a. Remove the elevon. Refer to T.O. 1F-106A-2-7-2-1 for elevon removal procedure.

b. Place the elevon on a suitably padded work bench or table.

c. Remove all paint and primer from damaged area of elevon. Use Methyl Ethyl Ketone, Specification TT-M-261, to remove paint and primer.

d. Perform a dye penetrant inspection to determine extent of damage. Refer to Section I for dye penetrant inspection procedures and materials.

e. Using a No. 20 (0.161) drill, remove rivets attaching damaged elevon nose rib to elevon upper and lower skins and elevon forward spar.

NOTE:
DOUBLERS AND FLUSH SKIN PATCH MUST BE HEAT TREATED TO -T6 CONDITION PER SPECIFICATION MIL-H-6088 AFTER FORMING.

f. Remove damaged elevon nose rib.

g. Cut out damaged portion of elevon skin.

h. Fabricate doublers from 0.050, 7075-0 bare sheet as shown in detail A.

i. Fabricate a flush skin patch from 0.040, 7075-0 bare sheet as shown in detail B.

j. Fabricate a new rib from 0.040, 2024-T4 clad as shown in detail C.

k. Install the new rib and hold in place with clamps.

l. Using the existing holes in elevon skins and elevon spar as a guide, drill holes through flanges of new rib. Use a No. 20 (0.161) drill.

NOTE
USE LINE LAYOUT METHOD OF LOCATING BLIND HOLES AS SHOWN IN SECTION I TO PICK UP EXISTING HOLES FROM ELEVON SPAR.

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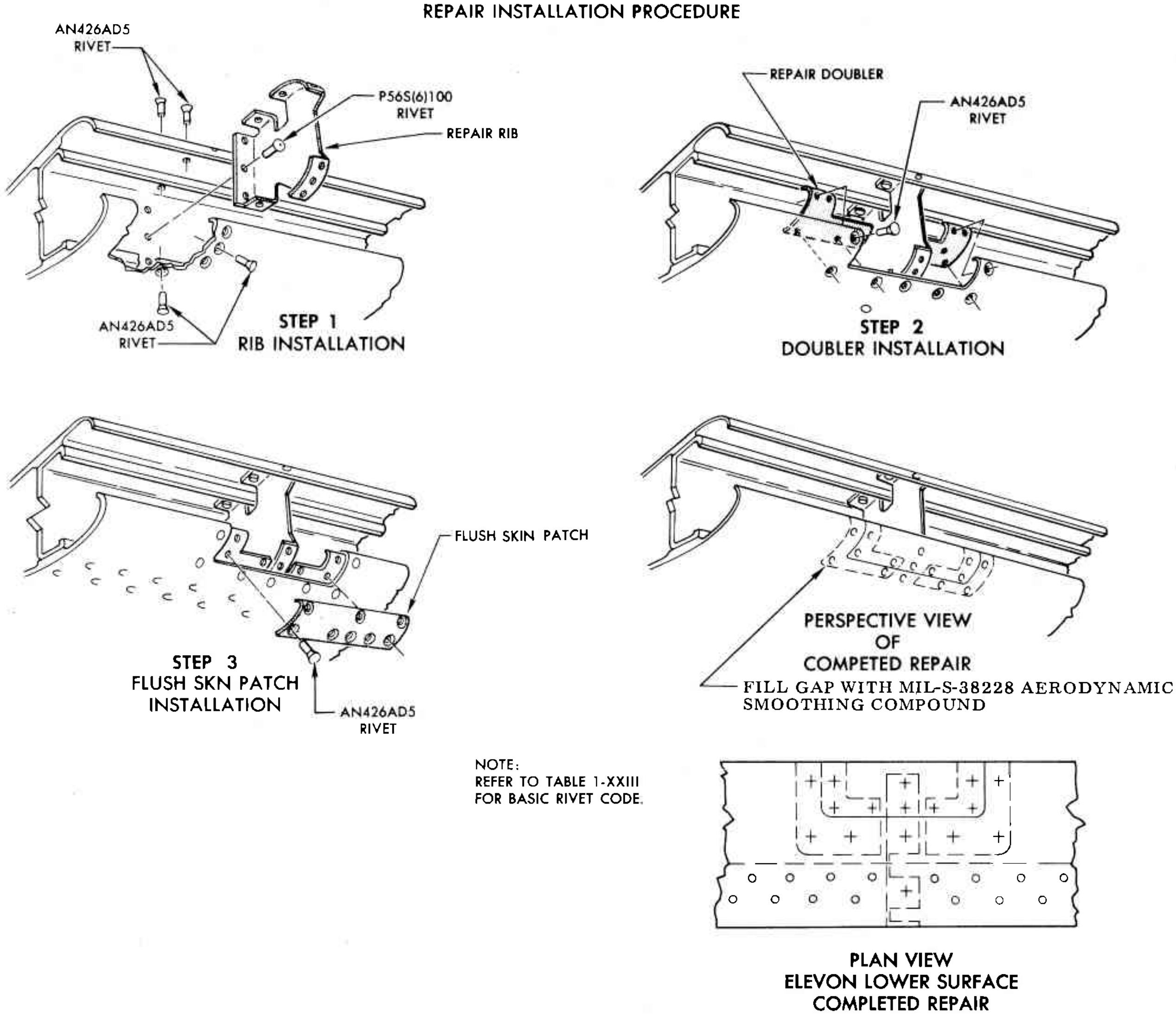
Figure 2-18. Elevon Leading Edge Repair (Sheet 1 of 2)

REPAIR INSTALLATION PROCEDURE



NOTE:
REFER TO TABLE 1-XXIII
FOR BASIC RIVET CODE.

m. Ream holes in vertical flange of rib and elevon spar with a 3/16 (0.187) drill.
n. Remove rib from elevon and remove all burrs from rib and elevon.
o. Predrill holes in undrilled flanges of rib as shown. Use a No. 30 (0.128) drill.
p. Reinstall rib and rivet in place as shown.
q. Using No. 30 (0.128) drill, predrill holes in doublers as shown.
r. Install doublers and hold in place with clamps.
s. Using predrilled holes in doublers as a guide, drill holes through existing elevon skin. Use a No. 30 (0.128) drill.
t. Ream holes drilled in steps with a No. 20 (0.161) drill.
u. Remove doublers and remove all burrs from doublers and elevon.
v. Countersink 100 degrees all holes, in exterior surface, of elevon skin.
w. Reinstall doublers and rivet in place as shown.
x. Install flush skin patch and hold in place with clamps.
y. Using predrilled holes in doublers and rib as a guide, drill holes through flush skin patch. Use a No. 30 (0.128) drill.
z. Ream holes drilled in step x with a No. 20 (0.161) drill.
aa. Remove flush skin patch and remove all burrs from patch and elevon.
ab. Countersink 100 degrees all holes in exterior surface of flush skin patch. Reinstall flush skin patch and rivet in place as shown.
ac. Apply paint to elevon exterior surface in accordance with finish requirements contained in Section I.
ad. Reinstall elevon. Refer to T.O. 1F-106A-2-7-2-1 for elevon installation procedures.

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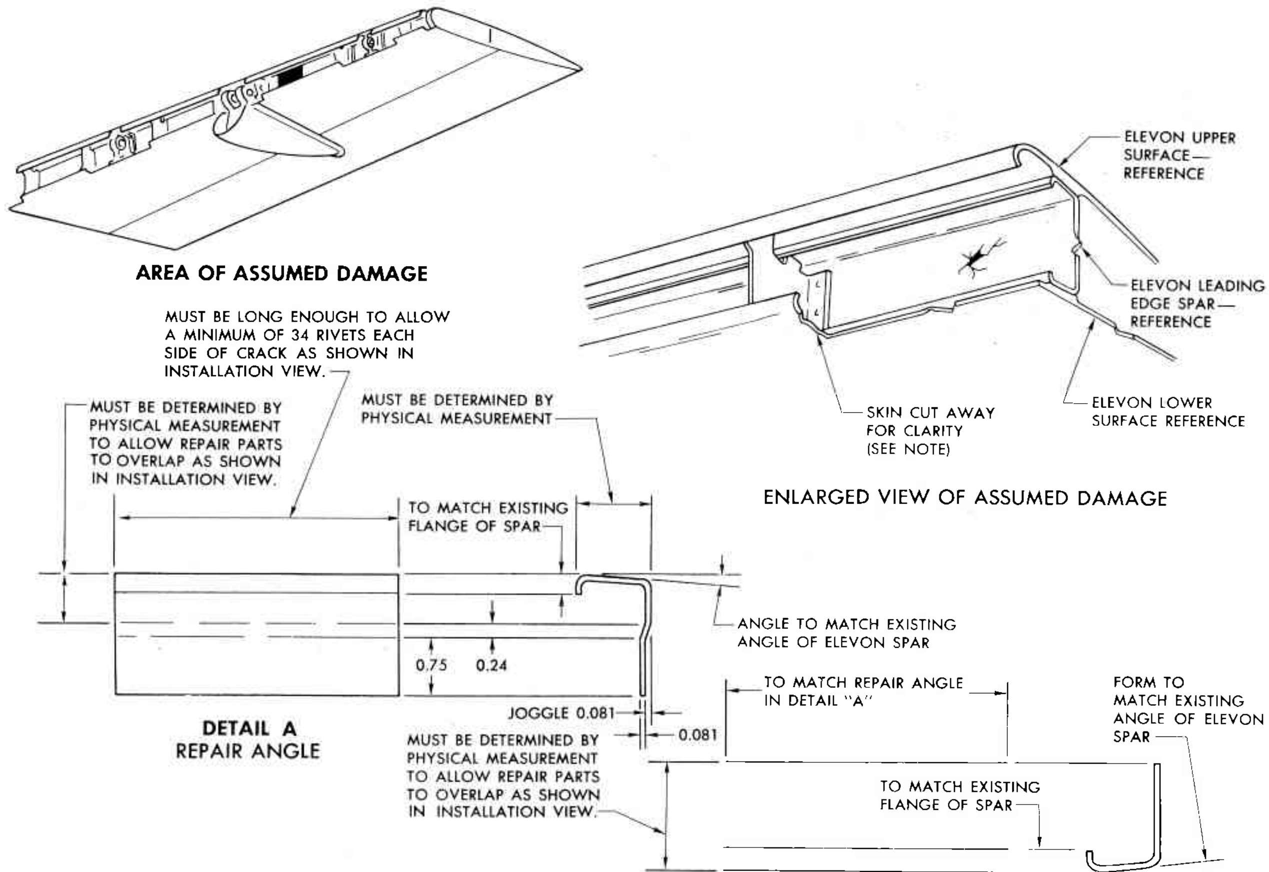
Figure 2-18. Elevon Leading Edge Repair (Sheet 2 of 2)



REPAIR PROCEDURE

a. Remove the elevon. Refer to T.O. 1F-106A-2-7-2-1 for removal procedure.

b. Remove all paint and primer from damaged area. Use Methyl Ethyl Ketone, Federal Specification TT-M-261, to remove paint and primer.

c. Perform a dye penetrant inspection to determine extent of damage. Refer to Section I for dye penetrant inspection procedures and materials.

NOTE

IF SPAR IS BENT AND CANNOT BE STRAIGHTENED IN PLACE OR IF A SECTION OF SPAR MUST BE CUT OUT TO REMOVE DAMAGE, REMOVE AND REPLACE THE ENTIRE SECTION OF ELEVON SPAR WHICH CONTAINS THE DAMAGE.

d. Fabricate repair angles as shown in Details A and B from 7178-0 bare sheet. Repair parts must be heat treated to -T6 condition after forming per Specification MIL-H-6088.

e. Apply one coat of wash primer to repair parts. Apply primer in accordance with Specification MIL-C-8514.

f. Apply one coat of zinc chromate primer to repair parts. Apply zinc chromate primer in accordance with Specification MIL-P-8585 or MIL-P-6808.

g. Install repair parts as shown and hold in place with clamps.

h. Using the existing holes in upper and lower surface of elevon as a guide, drill holes through the repair parts. Use a 3/16 (0.1875) drill.

i. Layout and drill holes through vertical flanges of repair parts and spar as indicated in repair installation view. Use a No. 30 (0.128) drill.

j. Ream holes drilled in step I with a 3/16 (0.1875) drill.

k. Remove repair parts and remove all burrs from repair parts and elevon.

l. Reinstall repair parts and hold in place with clecos.

m. Rivet repair parts in place as indicated in installation view.

n. Refinish elevon exterior surface in accordance with finish requirements contained in Section I.

o. Reinstall elevon. Refer to T.O. 1F-106A-2-7-2-1 for elevon installation procedure.

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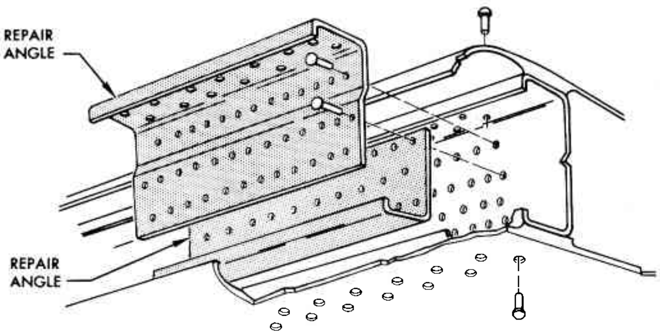
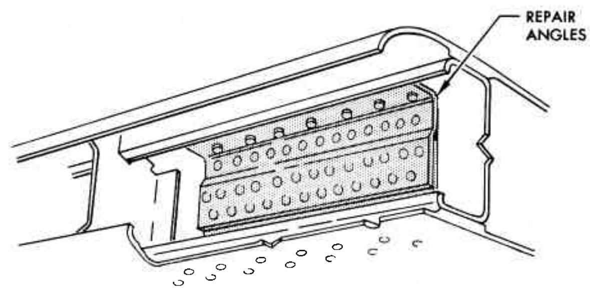
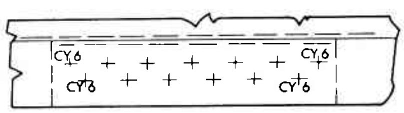
Figure 2-19. Elevon Forward Spar Repair (Sheet 1 of 2)



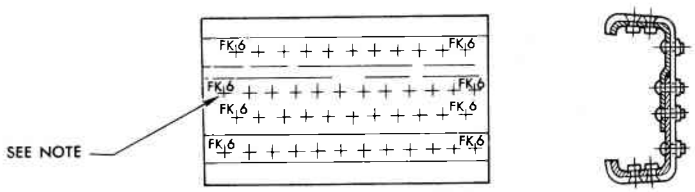
STEP 1
EXPLODED REPAIR



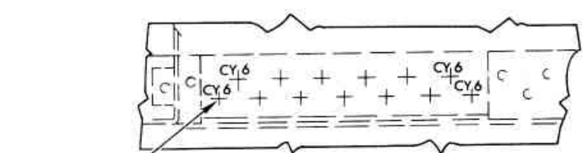
STEP 2
ASSEMBLED REPAIR



TOP VIEW COMPLETED REPAIR



FRONT VIEW COMPLETED REPAIR



BOTTOM VIEW COMPLETED REPAIR

NOTE:
REFER TO TABLE 1-XXIII FOR BASIC RIVET CODE.

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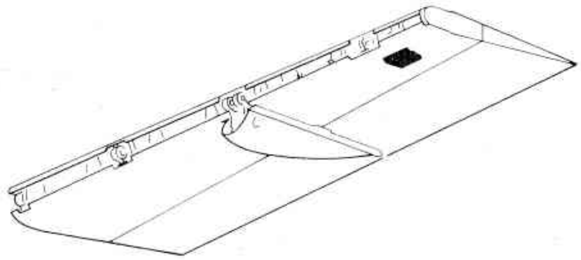
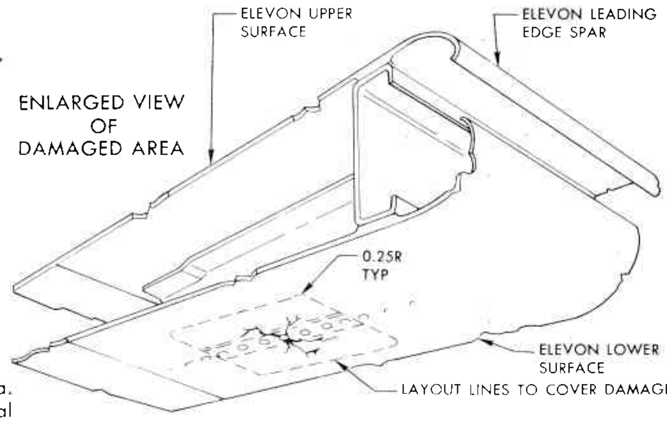
Figure 2-19. Elevon Forward Spar Repair (Sheet 2 of 2)



AREA OF ASSUMED DAMAGE



REPAIR PROCEDURE

a. Remove all paint and primer from the damaged area. Use a cloth dampened with Methyl Ethyl Ketone, Federal Specification TT-M-261, to remove paint and primer.
b. Perform a Fluorescent penetrant inspection to determine extent of skin damage. Refer to Section I for Fluorescent penetrant inspection procedures and materials.
c. Using a No. 30 (0.218) drill, remove the rivets attaching the skin to internal structure in damaged area.
d. Using a 0.375-inch diameter router bit, rout out the damaged skin to a regular shape.

NOTE
ROUT OUT DAMAGED SKIN EACH SIDE OF ELEVON RIB; PLACE A SPACER MADE FROM SOFT ALUMINUM SHEET BETWEEN SKIN AND RIB, THEN CONTINUE ROUTING TO REMOVE REMAINDER OF DAMAGED SKIN.

e. Visually inspect internal rib for damage.

NOTE
IF RIB HAS BEEN DAMAGED, CONTINUE WITH STEP "f." IF RIB HAS NOT BEEN DAMAGED, CONTINUE WITH STEP "u."

f. Remove primer from elevon rib. Use a cloth dampened with Methyl Ethyl Ketone, Federal Specification TT-M-216, to remove primer.
g. Perform a Fluorescent penetrant inspection to determine extent of damage. Refer to Section I for Fluorescent penetrant inspection procedures and materials.
h. Using metal cutting hand shears and hand files, cut out damaged section of rib to a regular shape.
i. Fabricate a filler angle and stiffener angle, as shown in Details B and C, from 7075-0 bare sheet. Rib repair parts must be heat treated to -T6 condition per Specification MIL-H-6088 after forming.
j. Predrill holes in rib filler and stiffener angles as shown. Use a No. 39 (0.099) drill.
k. Apply one coat of wash primer to rib repair parts per Specification MIL-C-8514.
l. Apply one coat of zinc chromate primer to rib repair parts per Specifications MIL-P-8585 or MIL-P-6808.
m. Install stiffener angle, as shown, and hold in place with clamps.
n. Using predrilled holes in stiffener angle as a guide, drill holes in elevon rib. Use a No. 39 (0.099) drill.
o. Using existing holes in elevon rib as a guide, drill holes through stiffener angle. Use a No. 30 (0.128) drill.
p. Install rib filler angle, as shown, and hold in place with clamps.
q. Using predrilled holes in filler angle as a guide, drill holes through stiffener angle. Use a No. 39 (0.099) drill.
r. Ream all No. 39 (0.099) holes with a No. 30 (0.218) drill.

NOTE
DUE TO RESTRICTED WORKING AREA, A RIGHT ANGLE DRILL MUST BE USED TO DRILL HOLES THROUGH VERTICAL FLANGES OF REPAIR PARTS AND ELEVON RIB.

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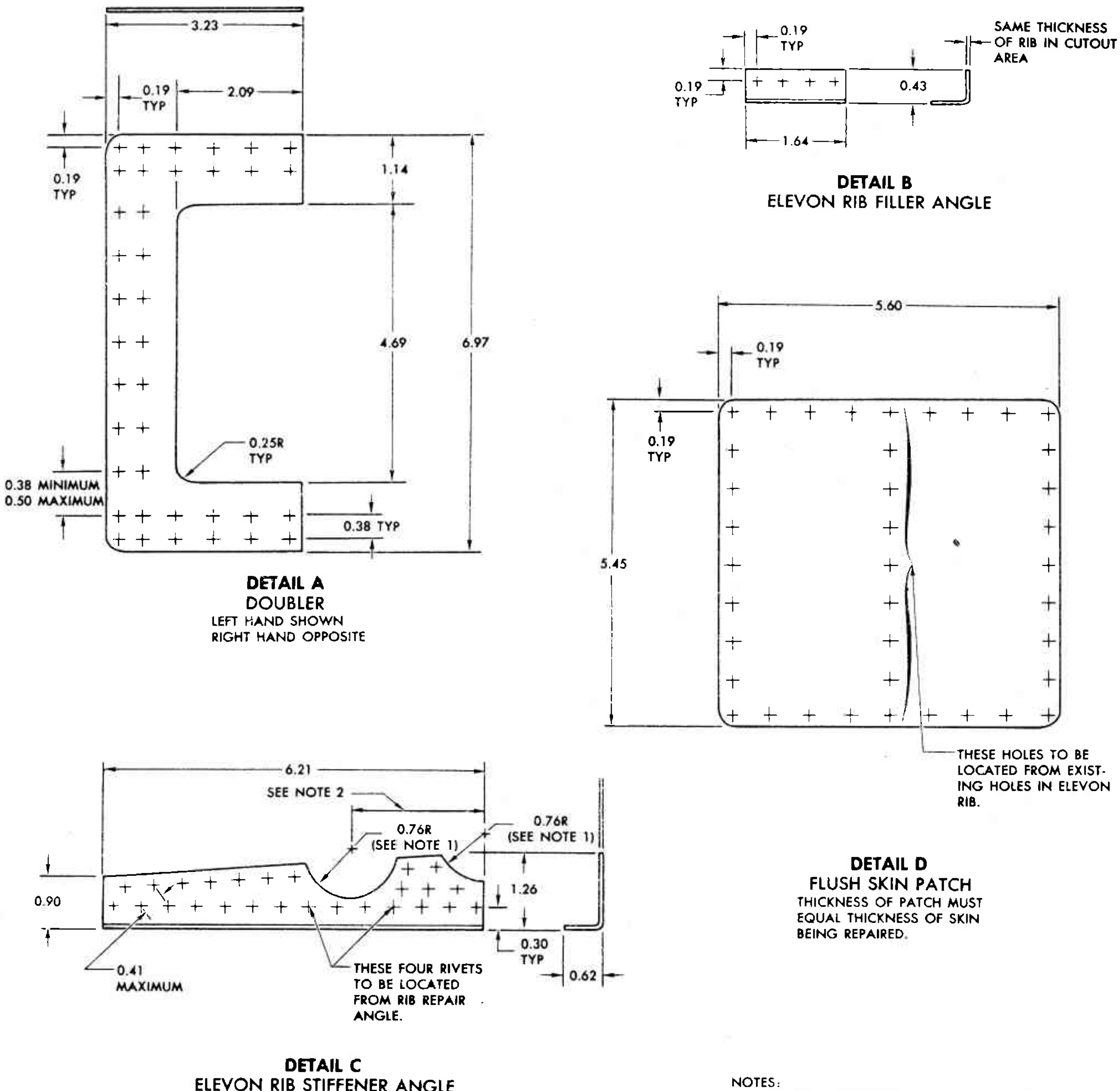
s. Remove rib repair parts and remove all burrs from repair parts and from elevon.
t. Reinstall rib repair parts and rivet in place through vertical flanges only, with AN470AD-4 rivets.

NOTE
DUE TO RESTRICTED WORKING AREA, AN AIR OPERATED RIVET SQUEEZER MUST BE USED TO INSTALL RIVETS THROUGH VERTICAL FLANGES OF REPAIR PARTS AND ELEVON RIB.

u. Fabricate doublers and flush skin patch from 7075-T6 bare sheet as shown in Details A and D.
v. Predrill doublers and flush skin patch as shown. Use a No. 39 (0.099) drill.
w. Apply one coat of wash primer to doublers and flush skin patch per Specification MIL-C-8514.
x. Apply one coat of zinc chromate primer to doublers and flush skin patch per Specifications MIL-P-8585 or MIL-P-6808.
y. Install doublers and hold in place with clamps.
z. Using predrilled holes in doublers as a guide, drill holes through existing skin. Use a No. 39 (0.099) drill.
aa. Ream holes drilled in step "z" with a No. 30 (0.128) drill.
ab. Remove doublers and remove all burrs from doublers and elevon.
ac. Machine countersink 100 degrees all new holes in existing elevon skin.
ad. Reinstall doublers and rivet in place with AN426AD-4 rivets.
ae. Install flush skin patch.
af. Using predrilled holes in flush skin patch as a guide, drill holes through doublers. Use a No. 39 (0.099) drill.
ag. Using lines layout method of locating blind holes contained in Section I, pick up existing holes in elevon rib. Use a No. 39 (0.099) drill.
ah. Ream holes drilled in steps "af" and "ag" with a No. 30 (0.218) drill.
ai. Remove flush skin patch and remove all burrs from patch and from elevon.
aj. Machine countersink 100 degrees all holes in flush skin patch.
ak. Reinstall flush skin patch and rivet in place with 56S-100-4 Dupont blind explosive rivets.
al. Fill gaps between flush skin patch and existing elevon with MIL-S-38228 Aerodynamic Smoothing Compound.
am. Refinish exterior surface of elevon skin in repair area according to applicable finish schedule contained in Section I.

Figure 2-20. Elevon Skin Repair (Sheet 1 of 3)



NOTES:
1. CUTOUTS IN STIFFENER ANGLE ARE PROVIDED FOR CLEARANCE OF Q2001-100 HOLES IN ELEVON RIB.
2. DIMENSIONS BETWEEN CUTOUTS IN RIB STIFFENER ANGLE MUST BE DETERMINED BY A PHYSICAL MEASUREMENT.
3. ALL DIMENSIONS ARE MINIMUM. REFER TO REPAIR LIMITATIONS SHOWN IN THIS REPAIR FOR MAXIMUM ALLOWABLE DIMENSIONS.

REPAIR LIMITATIONS

1. Maximum size of skin cutout = 27.50 square inches.
2. Maximum length of coutout in rib = 5.00 inches.

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Figure 2-20. Elevon Skin Repair (Sheet 2 of 3)

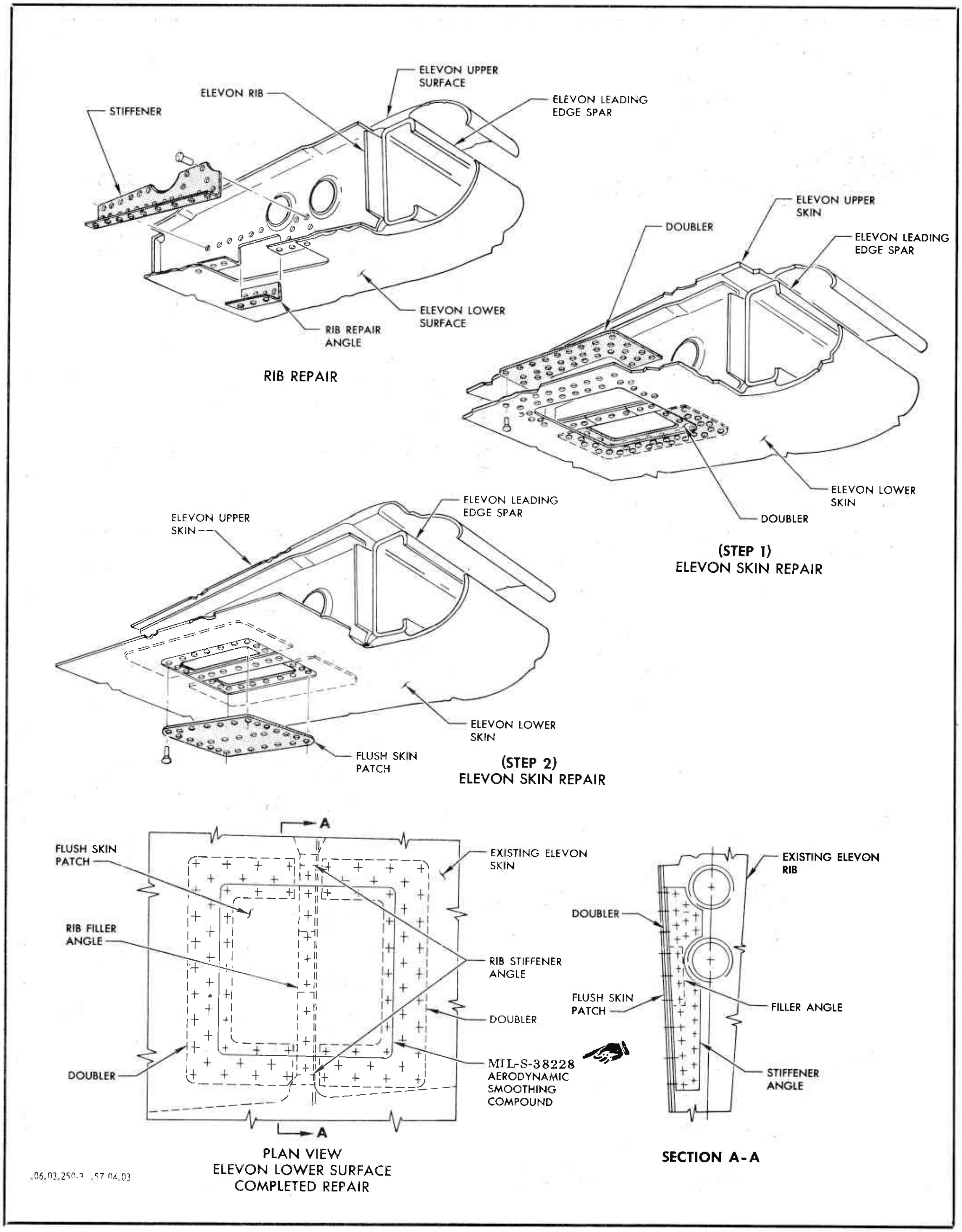
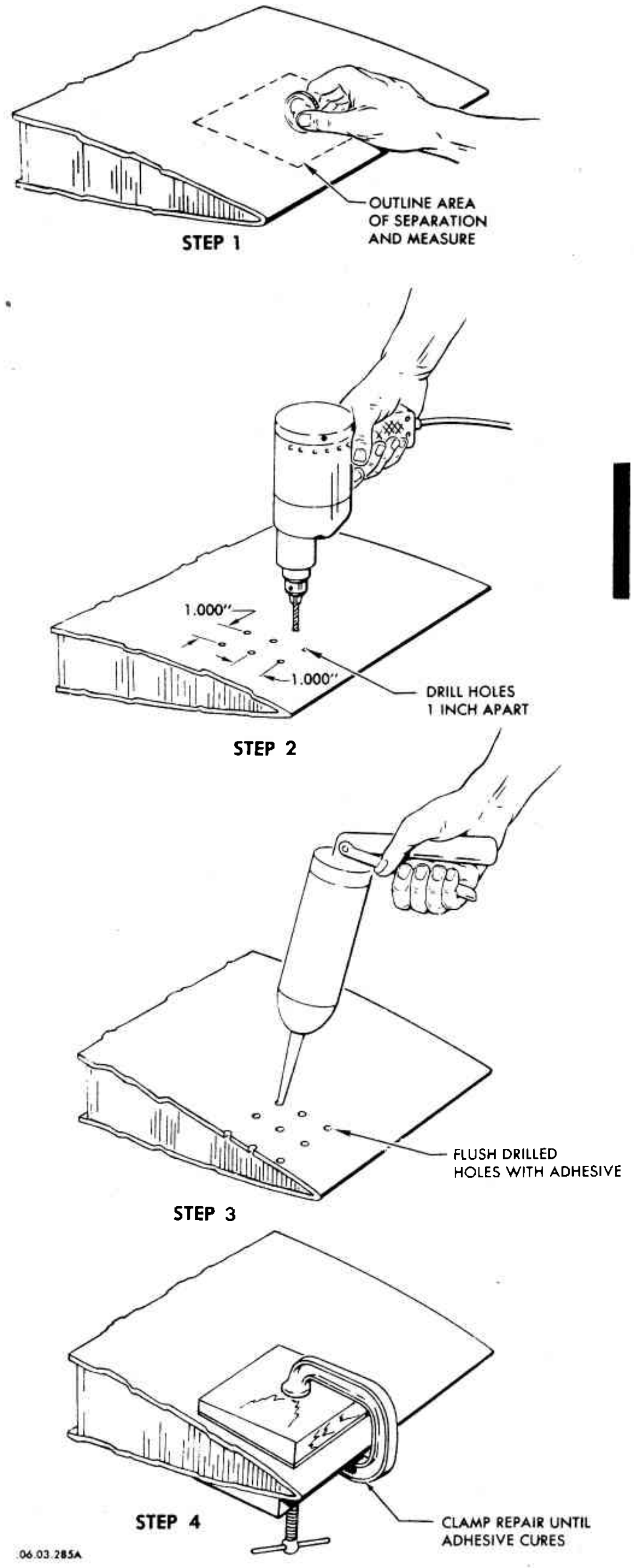


Figure 2-20. Elevon Skin Repair (Sheet 3 of 3)



REPAIR PROCEDURE

a. Determine exact area of separation by rapping the upper and lower surface of the elevon with a large coin. **(STEP 1)**

b. Repair lmitations

(1) Circular shape areas-4inch maximum diameter.

(2) All other shape areas-maximum area of 12 square inches with maximum dimension being no greater than 6 inches.

c. Outline area of separation with a pencil and measure area. If area of separation exceeds 12 square inches, refer to Section X for additional repairs. If area is less than 12 square inches, continue with the following repair procedure.

NOTE

REMOVE AND INVERT ELEVON IF SEPARATION IS ON LOWER SURFACE.

(1) Drill holes through skin in area where separation has occurred. Use a No. 40 (0.098") drill. Holes should be drilled one inch apart throughout area as shown in **STEP 2**.

CAUTION

DO NOT ALLOW DRILL TO PENETRATE MORE THAN 1/16 INCH THROUGH SKIN TO PREVENT DAMAGING HONEYCOMB SEGMENTS.

(2) Mix adhesive. Refer to Table II-VIII for mixing procedure. Use one of the following adhesives: Epon VIII, Epon 3119 or EC1469.

(3) Use a sealant gun with a small tip on a large hypodermic type needle and inject enough mixed adhesive to saturate separated area until all the drilled holes are flushed. **(STEP 3)**

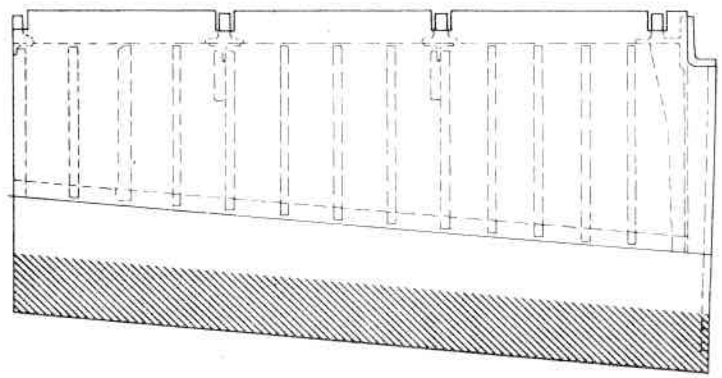
NOTE

IF AREA OF SEPARATION IS SMALL (2 SQUARE INCHES OR LESS) ADHESIVE MAY BE INJECTED FIRST WITH ONE NEEDLE FOLLOWED BY THE ACCELERATOR IN ANOTHER NEEDLE. THIS METHOD WILL ELIMINATE PRE-MIXING. HOWEVER, PROPER MIXTURE RATIOS BETWEEN ADHESIVE AND ACCELERATOR MUST BE OBSERVED. REPAIRS MADE BY THIS METHOD DO NOT REQUIRE CLAMPING.

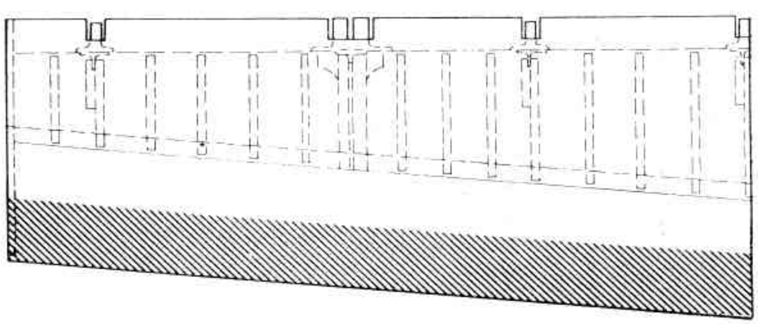
(4) Clamp a block over repair area and allow adhesive to cure for time recommended by the manufacturer. **(STEP 4)**

d. If area of separation exceeds limits specified in step b, refer to figure 2-22 for extended limits, and repair in accordance with figure 10-41.

Figure 2-21. Elevon Honeycomb Separation Repair



ELEVON–INBOARD



ELEVON–OUTBOARD

REPAIR LIMITATIONS

a. Elevon trailing edge repair, see Figure 10–41 or 10–42, is restricted to:
   (1) Shaded area shown.
   (2) Maximum spanwise length of repairs in a section are:
      (a) Single Repair – 20 per cent of the spanwise length of section.
      (b) Multiple Repairs – Total of 35 per cent of spanwise length of section.
   (3) Minimum distance between repair cutouts is 4 inches.

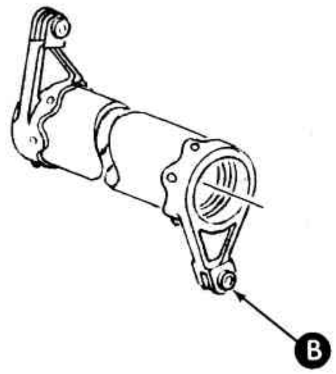
b. Use the typical repair shown in Figures 10–37 or 10–39 to repair damages in elevons having a minimum of 3 inches between cutout and trailing edge, and a diameter of 6 inches or less. Minimum distance between repair cutout is diameter of larger of the two adjacent repairs.

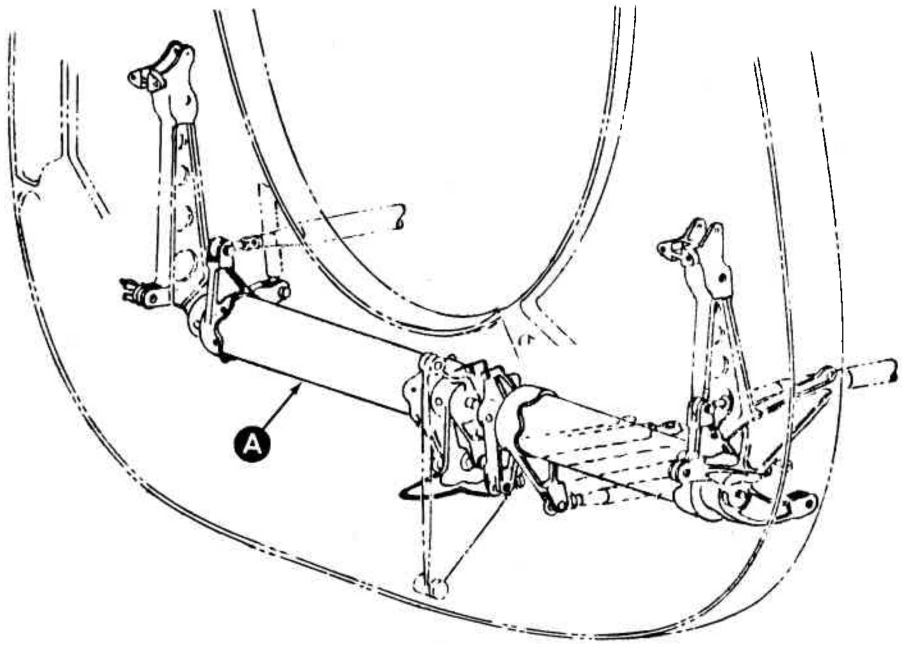
c. See Figure 2–21 for separation repair limits.

d. Mixing of Repairs:
   (1) Trailing edge repair, Figures 10–41 or 10–42, and plating repair, Figure 10–37, must not overlap.
   (2) Trailing edge repair, Figures 10–41 or 10–42, and plating repair, Figure 10–37, may overlap a separation repair, Figure 2–21.

e. Refer damages that exceed above limitations to an aeronautical structures engineer for disposition or replace components.

Figure 2-22. Elevon Honeycomb Repair–Moisture Removal Procedure





VIEW A
ELEVON CROSSOVER

VIEW B

1. Repair of out-of-round holes for bushings as shown in View B above will be accomplished as follows:

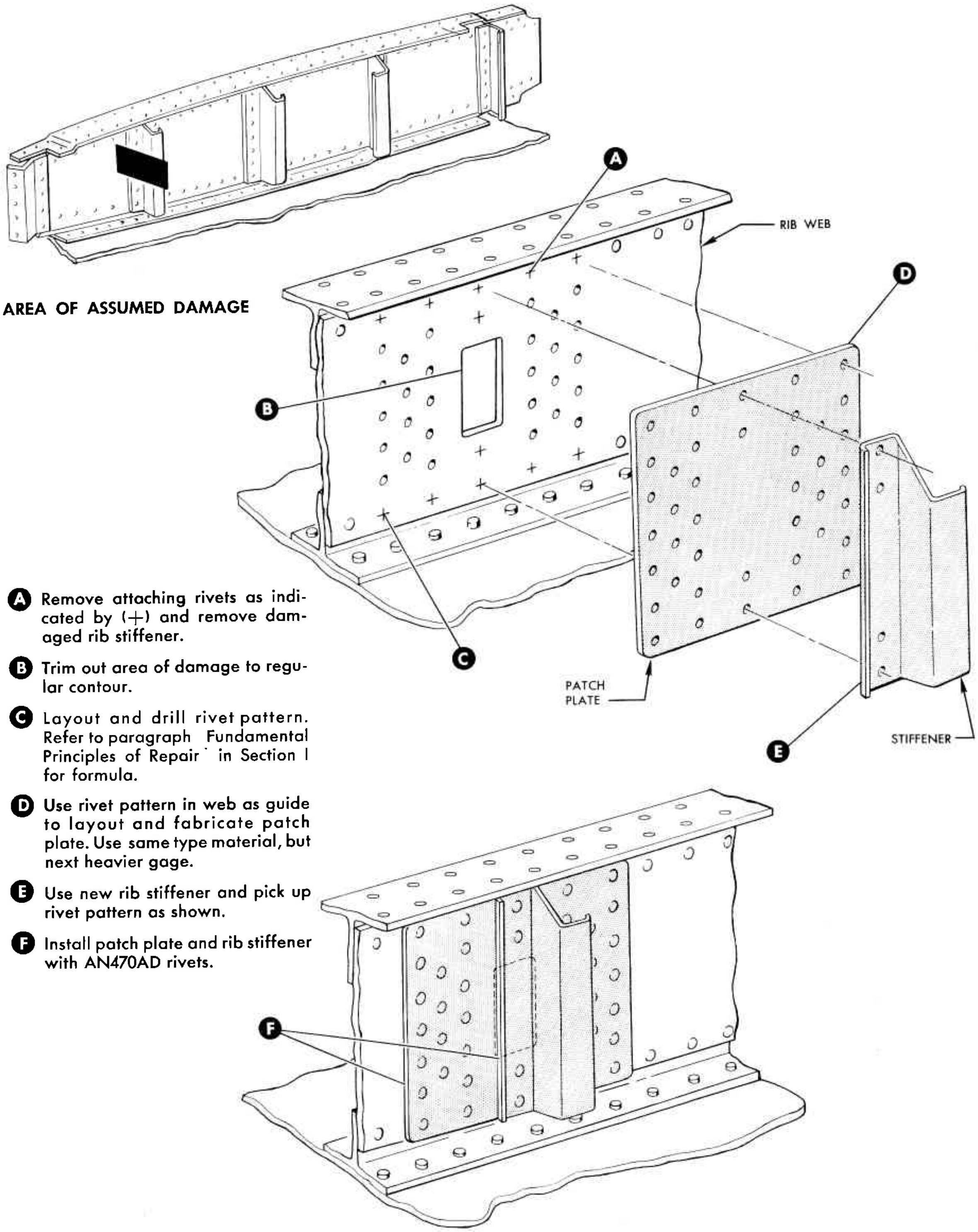
   a. Line ream or line bore holes in end fitting only to a point that will eliminate out-of-round condition. Maximum diameter allowed after line reaming or boring is 0.415 inch.

   b. Manufacture repair bushings in accordance with NAS77-4-19 and NAS77-4-2D but increase the O.D. by 0.001 in. $^{+\,0.0000\text{ in.}}_{-\,0.0005\text{ in.}}$ larger than the diameter of the ream or bored hole.

   c. Freeze bushings in dry ice alcohol solution five minutes prior to installation.

2. The repair procedure applies to left and right hand cross over tube assemblies.

Figure 2-22A. Elevon Crossover Tube, Bushing Hole Repair



A Remove attaching rivets as indicated by (+) and remove damaged rib stiffener.

B Trim out area of damage to regular contour.

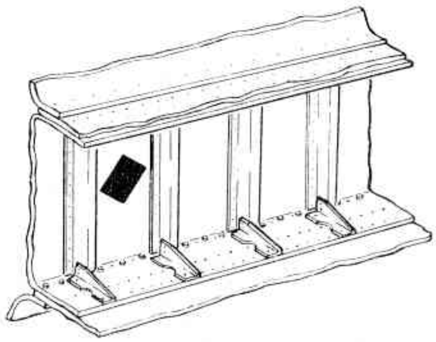
C Layout and drill rivet pattern. Refer to paragraph "Fundamental Principles of Repair" in Section I for formula.

D Use rivet pattern in web as guide to layout and fabricate patch plate. Use same type material, but next heavier gage.

E Use new rib stiffener and pick up rivet pattern as shown.

F Install patch plate and rib stiffener with AN470AD rivets.

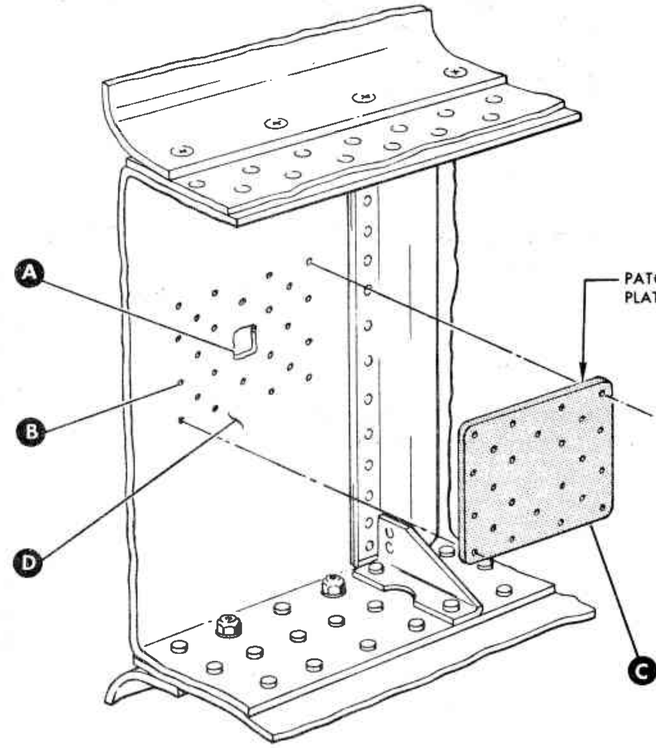
Figure 2-23. Wing Rib Repair—Intermediate Box Area

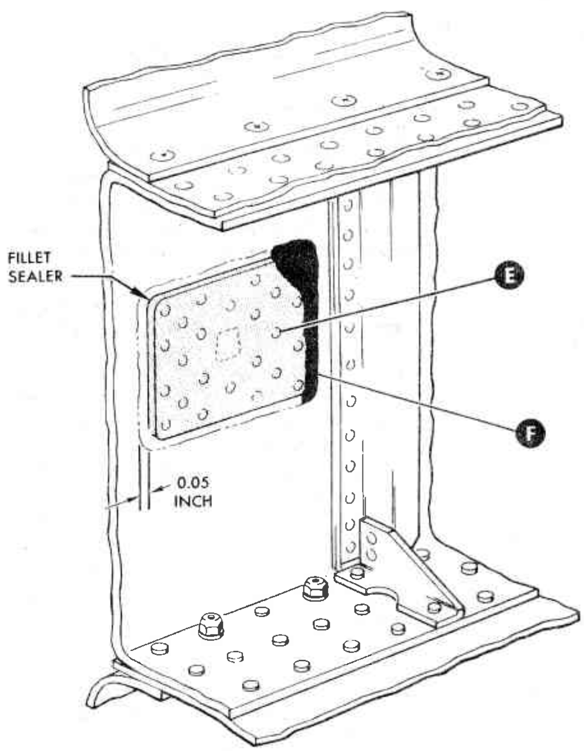


A Trim out area of damage to regular contour.

B Refer to paragraph "Fundamental Principles of Repair" in Section I for rivet pattern layout. Drill No. 11 (0.191) rivet holes as shown.

C Use rivet pattern in web as guide to layout and fabricate patch plate. Use same type material as web, but next heavier gage.





D Apply sealer, Military Specification MIL-S-8802 to faying surfaces of reinforcing plate, rib web member, and to shank of rivets on installation.

E Install patch plate with MS20470AD6 rivets.

F Apply 0.060 inch thick coating of fillet sealer, Military Specification MIL-S-8802 over entire repair. Allow sealer to dry until tack free.

CAUTION

DO NOT SUBSTITUTE SCREWS FOR SOLID RIVETS IN FUEL TIGHT REPAIRS.

NOTES:

1. REFER TO PARAGRAPHS TITLED "PREPARATION OF SURFACE SEALING," "APPLICATION OF CEMENTS AND SEALERS," AND "STRUCTURAL ADHESIVE PRIME" IN THIS SECTION FOR SEALANT.
2. PRESSURE TEST FUEL TANK IN ACCORDANCE WITH INSTRUCTIONS CONTAINED IN THIS SECTION AFTER COMPLETION OF REPAIR.

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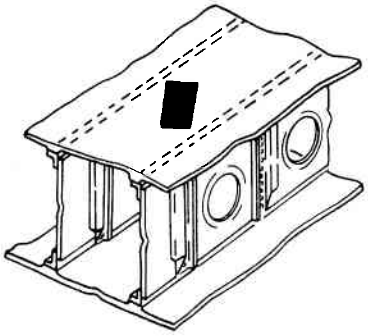
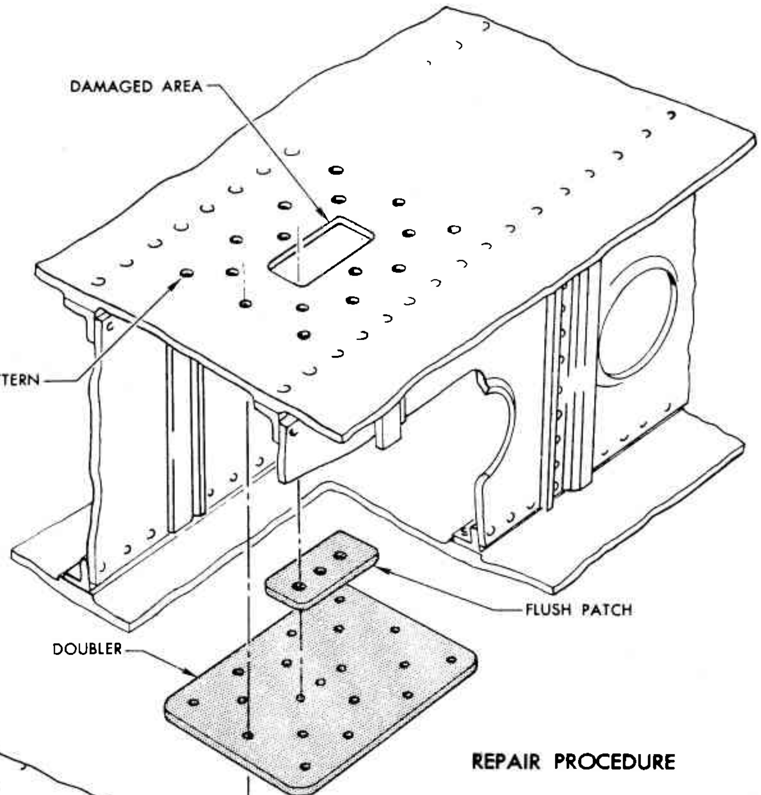
Figure 2-24. Wing Rib Repair—Fuel-Tight Area

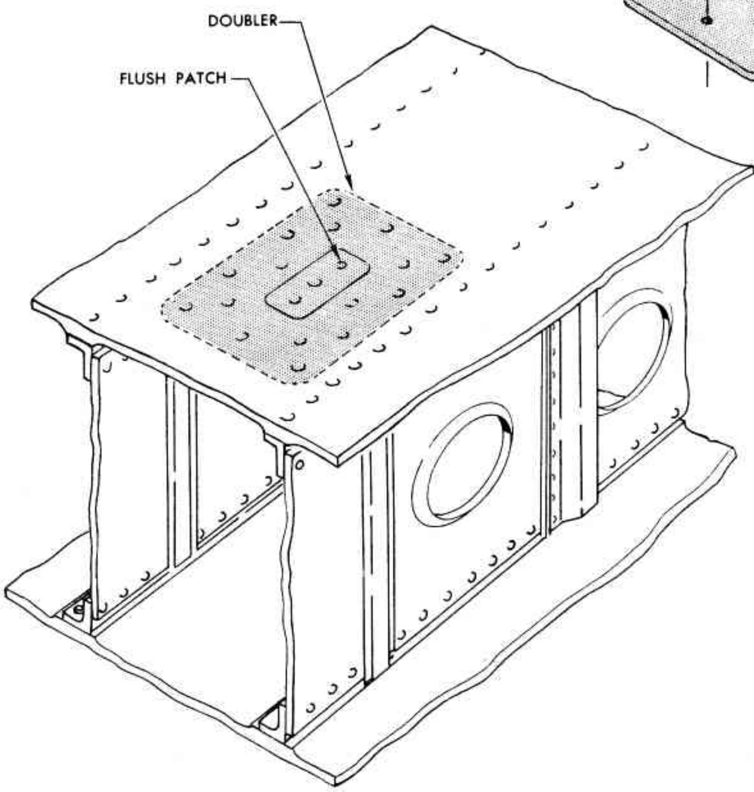


AREA OF ASSUMED DAMAGE
(PLATING THICKNESS UNIFORM AT DAMAGE)



CAUTION

DO NOT SUBSTITUTE SCREWS FOR SOLID RIVETS IN FUEL TIGHT REPAIRS.



REPAIR PROCEDURE

a. **Trim out area of damage to smooth contour.**
b. **Layout and drill rivet pattern in plating. Refer to Section I for rivet pattern.**
c. **Use rivet pattern in plating as guide to layout, and fabricate doubler. Use same type material as original plating, but next heavier gage.**
d. **Fabricate flush patch and drill out rivet holes as shown. Use same type material and gage as original plating.**
e. Apply sealer, Military Specification MIL-S-8802 on faying surfaces of patch plate, doubler and shank of rivets on installation.
f. Attach flush patch to doubler with MS20426AD4 rivets.
g. Install doubler with MS20426AD6 rivets.
h. Apply 0.060 inch thick coating of fillet sealer, Military Specification MIL-S-8802 over entire repair. Allow sealer to dry until tack free.

NOTES:
1. REFER TO PARAGRAPHS TITLED "PREPARATION OF SURFACE SEALING," "APPLICATION OF CEMENTS AND SEALERS," AND "STRUCTURAL ADHESIVE PRIME" IN THIS SECTION FOR SEALANT.
2. PRESSURE TEST FUEL TANK IN ACCORDANCE WITH INSTRUCTIONS CONTAINED IN THIS SECTION AFTER COMPLETION OF REPAIR.

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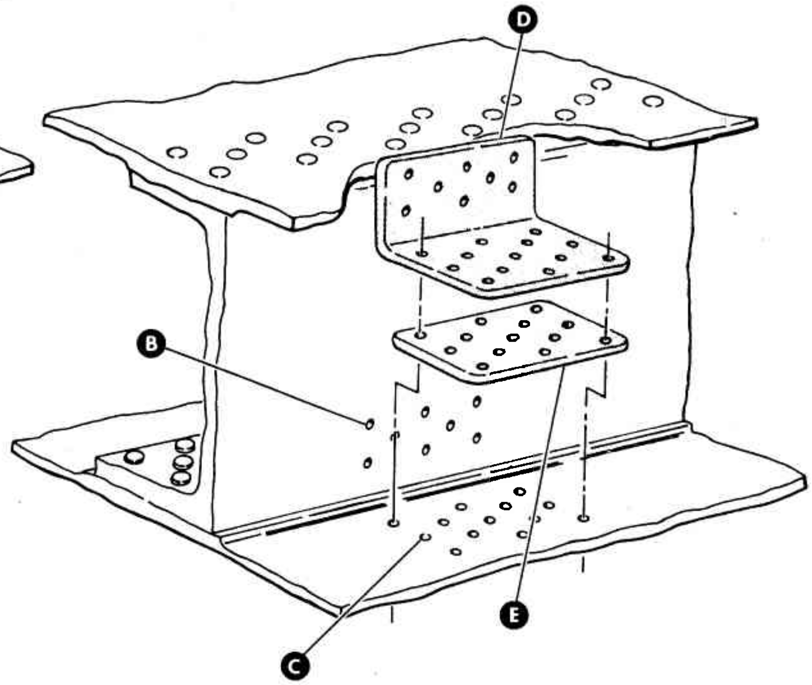
Figure 2–25. Wing Plating Repair–Fuel-Tight Area

All data on pages 2-41 thru 2-49 including figure 2-26 deleted.

AREA OF ASSUMED DAMAGE
(PLATING THICKNESS
UNIFORM AT DAMAGE)



A Tap out dent using a plastic hammer or rawhide mallet. Back opposite side of plating with heavy bucking bar covered with masking tape when tapping out dent.

B Layout and drill rivet pattern. Refer to paragraph "Fundamental Principles of Repair" in Section I.

C Layout and drill rivet pattern same as for hole repair. Countersink holes 100° on far side.

D Use rivet pattern as a guide and fabricate reinforcing angle of same type material as original plating, but next heavier gage.

E Use rivet pattern as a guide and fabricate filler plate of same type material as original plating. Use gage equal to depth of machined portion of plating.

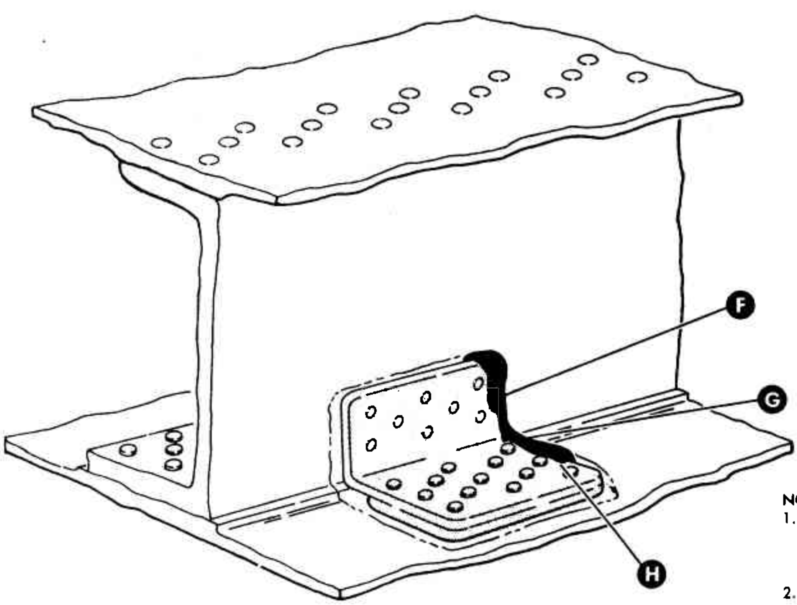
F Apply sealer, Military Specification MIL-S-8802 to all faying surfaces and install filler plate and reinforcing angle. Use MS20470D rivets through reinforcing angle and channel and MS20426AD rivets through reinforcing angle, filler, and plating.

G Plug ends of voids created by assembly of repair with plugs made of 1100 aluminum.

H Apply 0.060 inch thick coating of fillet sealer, Military Specification MIL-S-8802 over entire repair. Allow sealer to dry until tack free.

CAUTION

DO NOT SUBSTITUTE SCREWS FOR SOLID RIVETS IN FUEL TIGHT REPAIRS.



NOTES:
1. REFER TO PARAGRAPHS TITLED, "PREPARATION OF SURFACES FOR SEALING," "APPLICATION OF CEMENTS AND SEALERS," AND "STRUCTURAL ADHESIVE PRIME" FOR CORRECT SEALANT APPLICATION AND PROCEDURE.
2. REFER TO PARAGRAPH TITLED "FUEL TANK LEAK TEST" FOR FUEL TANK PRESSURE TEST AFTER COMPLETION OF REPAIR.

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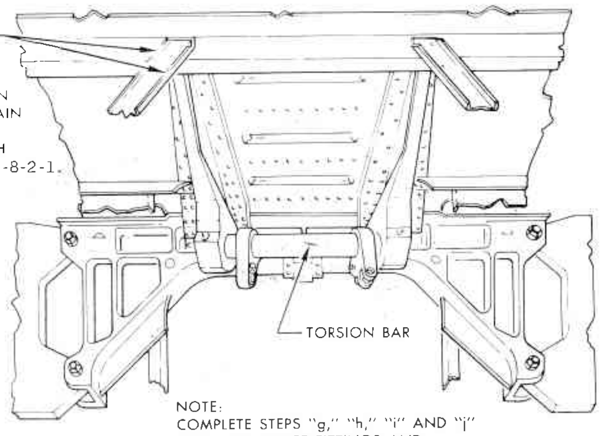
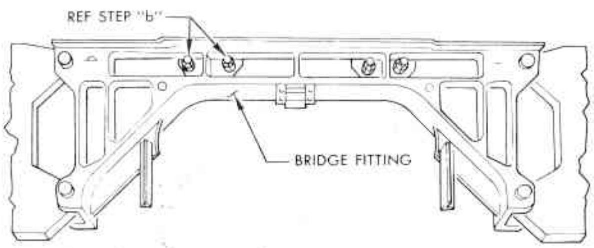
Figure 2–27. Wing Plating Dent Repair–Fuel-Tight Area

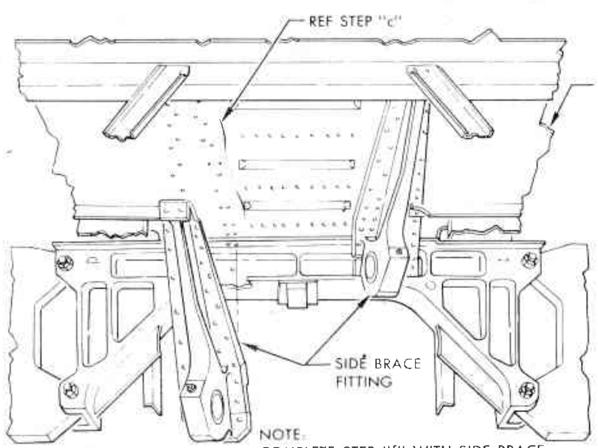
AREA OF ASSUMED DAMAGE



ASSEMBLY VIEW LOOKING UP AND INBOARD



VIEW LOOKING OUTBOARD



EXPLODED VIEW LOOKING UP AND INBOARD WITH SIDE BRACE FITTING REMOVED

a. Using a No. 13 (0.185) drill, remove existing lockbolts in angle and damaged side brace fitting.
b. Remove NAS464-8A10 bolts (2) from side brace fitting and bridge fitting.
c. Working from upper surface of the wing skin, remove MS20426-DD6 rivets (48). Remove side brace fitting.

CAUTION

CARE MUST BE TAKEN NOT TO DAMAGE ADJACENT STRUCTURE WHEN REMOVING SIDE BRACE FITTING.

d. Align side brace fitting replacement part in approximate position and drill No. 40 (0.098) pilot holes through flanges of side brace support. Locate holes from existing holes in angle, wing plating and bridge fitting.
e. Align side brace fitting in permanent position by using torsion bar as an aligning tool. (Torsion bar must turn freely for proper alignment). Clamp and cleco side brace fitting securely and drill 3/16-inch (0.187) holes (48) through flanges. Drill two, ½-inch (0.500) holes through inboard end of side brace fitting. Drill two No. 13 (0.185) holes through outboard end of side brace fitting.
f. Remove torsion bar and side brace fitting, then remove all drill chips and burrs from wing plating and side brace fitting. Holes in side brace fitting must be spotfaced on surface indicated by arrows. Refer to Section I for material finish requirements for repairs.
g. Reinstall and align side brace fitting as outlined in step "e."
h. Rivet side brace fitting to wing plating, using MS20426-DD6 rivets (48).
i. Secure side brace fitting to bridge fitting with two NAS464-8A10 bolts, two AN364-820 nuts and four AN960-816L washers.
j. Secure side brace fitting to angle with two Huck Lockbolts (5320-262-7672) and two collars (5320-489-4823).

NOTE

INSTALL HUCK LOCKBOLTS IN ACCORDANCE WITH INSTALLATION INSTRUCTIONS GIVEN IN SECTION I.

k. Reinstall main landing gear as outlined in T.O. 1F-106A-2-8-2-1.

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Figure 2-28. Replacement of Main Landing Gear Side Brace Fitting

A. Remove screws attaching damaged wing access door and remove door.

B. Hold access door, attaching screws and Fairprene seal for reinstallation. Place seal on clean, flat surface until reinstallation.

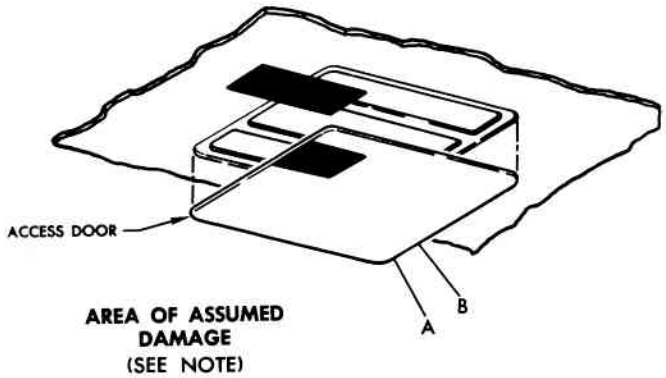
C. Bump out damaged area in wing skin and door to original contour as near as possible. Stop drill all cracks at end with ⅛ (0.125) drill.

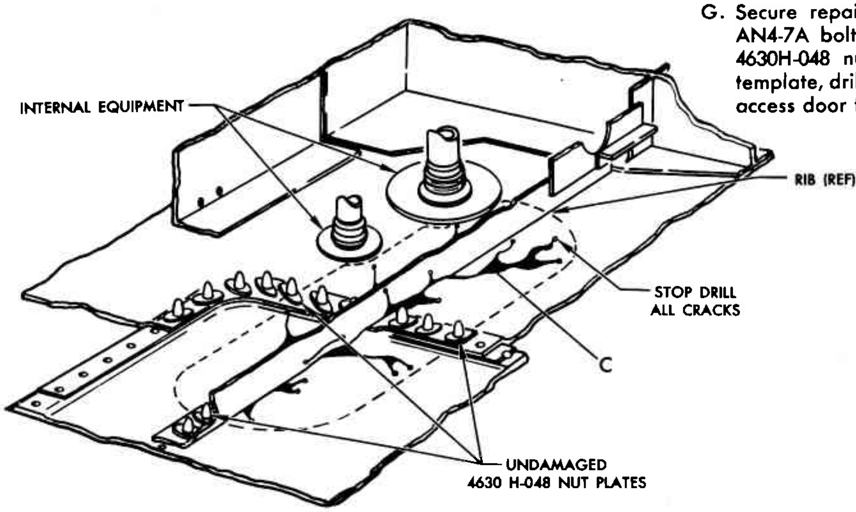
D. Fabricate repair plate from 0.125 gage 7075-T6 clad material. Dimension plate to exceed damaged area approximately 1.500 all around. Determine location of wing internal equipment adjacent to damaged area and contour repair plate accordingly. Edge distance of 1.500 may be reduced where necessary to alter contour to accept internal equipment.

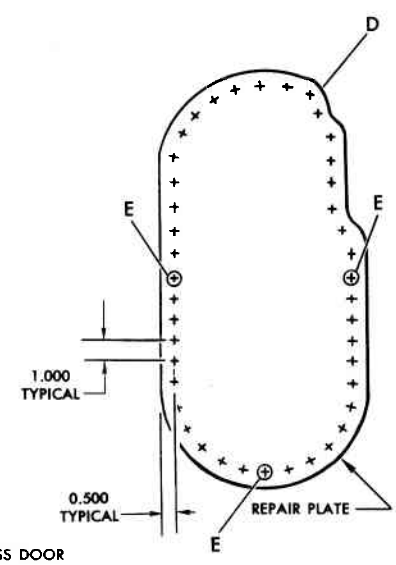
E. Layout and drill No. F (0.257) holes in repair plate for AN4 bolts as dimensioned. Adjust pattern to pick up three undamaged 4630H-048 nut plates in wing as indicated by ⊕.

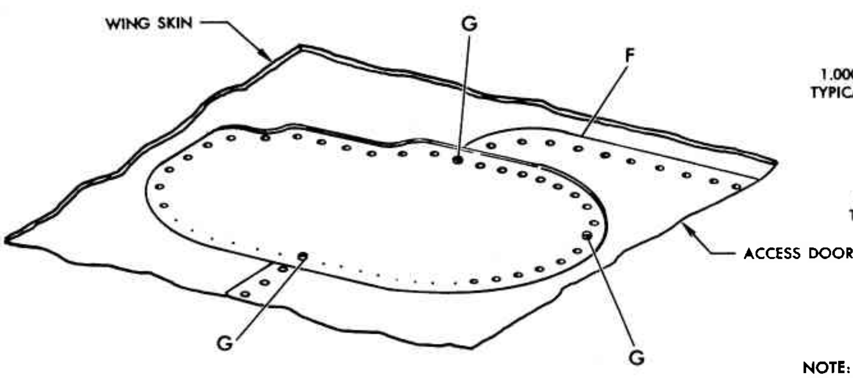
F. Reinstall damaged access door with sufficient screws to hold in place.

G. Secure repair plate over damaged area with AN4-7A bolts picking up the three undamaged 4630H-048 nut plates. Using repair plate as a template, drill No. F (0.257) holes in wing skin and access door for installation of dome nuts.









NOTE:
THIS REPAIR IS RESTRICTED TO FUEL TANKS NO. 1, NO. 2, AND NO. 3.

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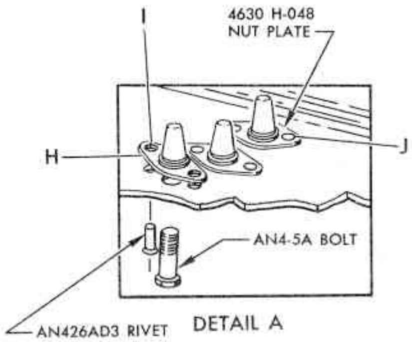
Figure 2-29. Wing Plating Repair — One-Time Flight (Sheet 1 of 2)



H. Position 4630 H-048 nut plates as shown and set
I. up AN4 bolt to hold dome nut during installation process.

J. Drill No. 40 (0.098) holes through nut plate flanges and wing and access door skins. Countersink holes 100°x0.189 far side.

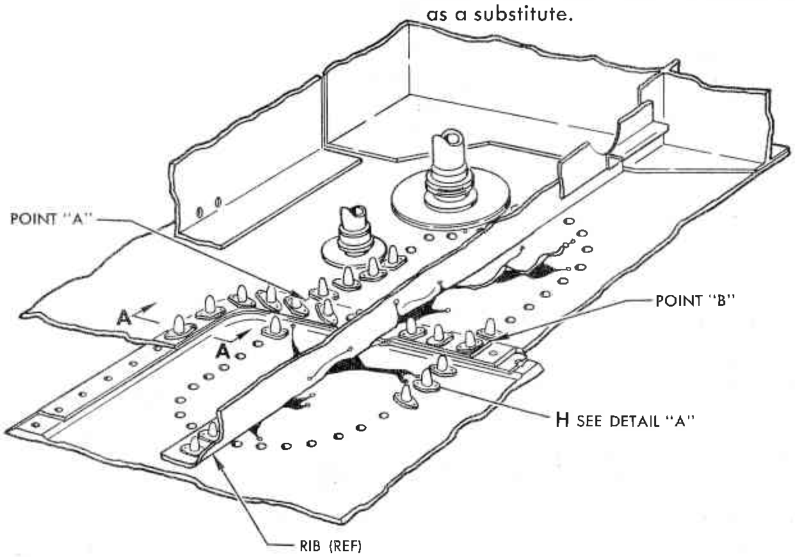
K. Install 4630 H-048 nut plates with AN426AD3 rivets.

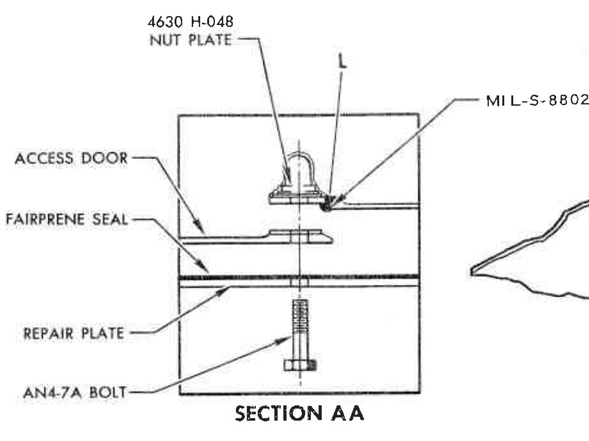
L. Use repair plate as template and fabricate 0.032 seal from Fairprene 5570.

M. Apply bead of sealant, Military Specification MIL-S-8802 to skin recess for access door attachment from point "A" to point "B".

N. Install access door with original attaching bolts.

Install Fairprene seal and repair plate with AN4-5A, AN4-7A bolts. If AN4 bolts are not available, AN525-416-9 and -10 screws may be used as a substitute.





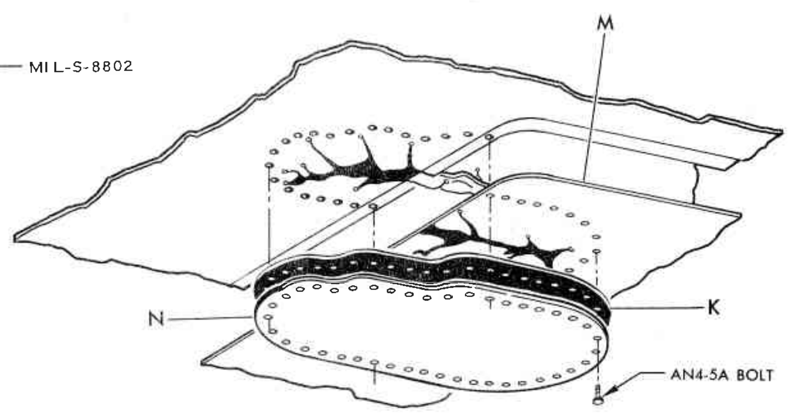


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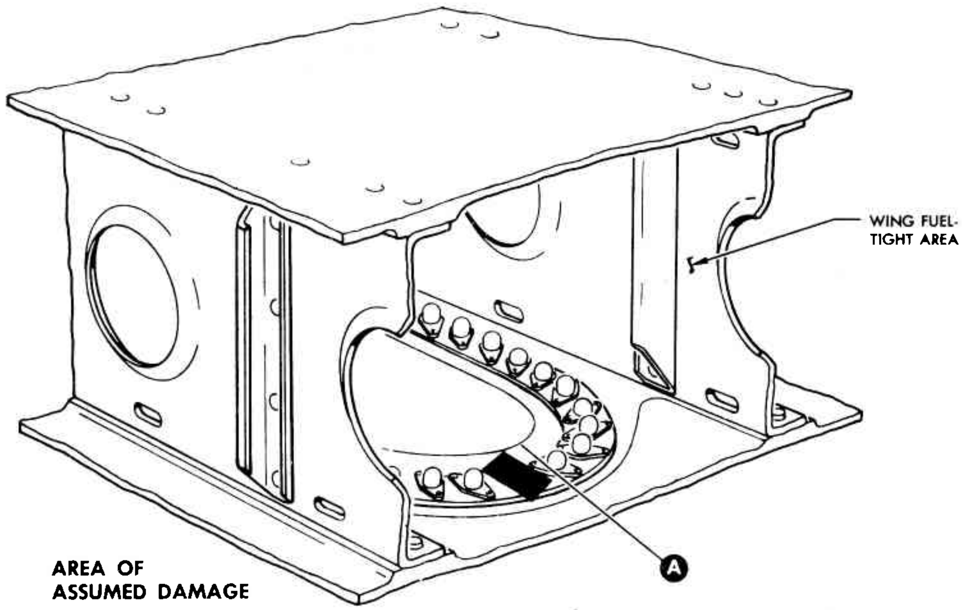
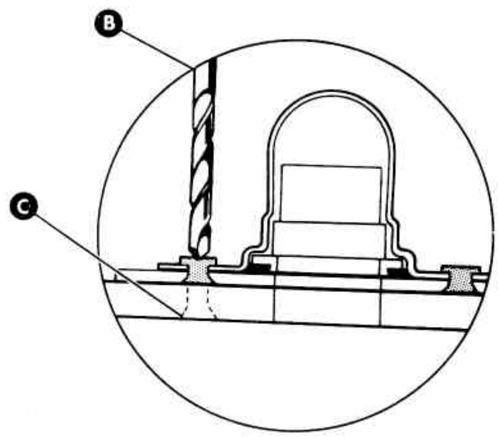
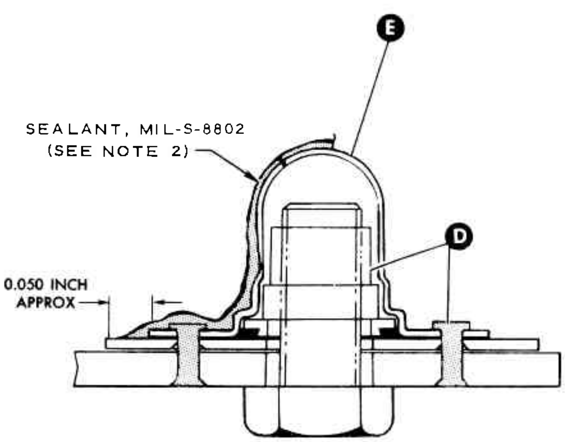
Figure 2–29. Wing Plating Repair–One-Time Flight (Sheet 2 of 2)



- **A** Remove access door.
- **B** Drill through rivet and wing plating using a No. 40 (0.098) drill. Remove dome nut.
- **C** Countersink plating 100° x0.189.
- **D** Install new dome nut with AN426AD3 rivets. Use bolt to hold dome nut during riveting process.
- **E** Cover dome nut installation with a coating of Military Specification MIL-S-8802.

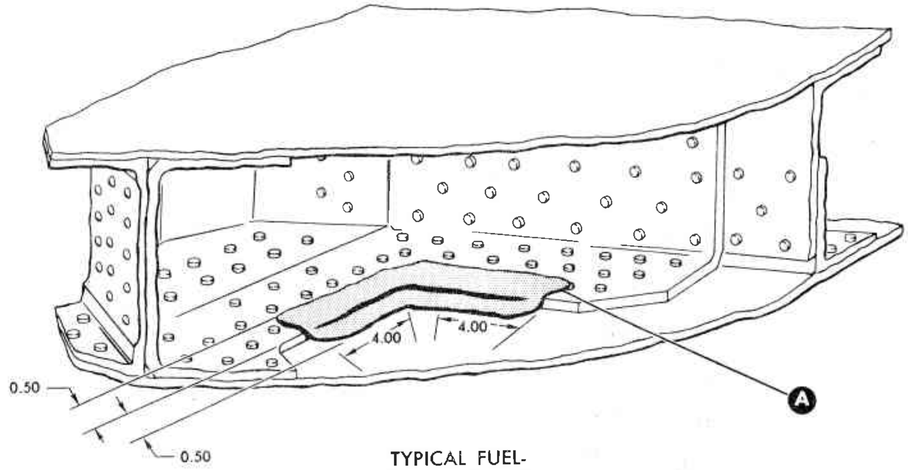




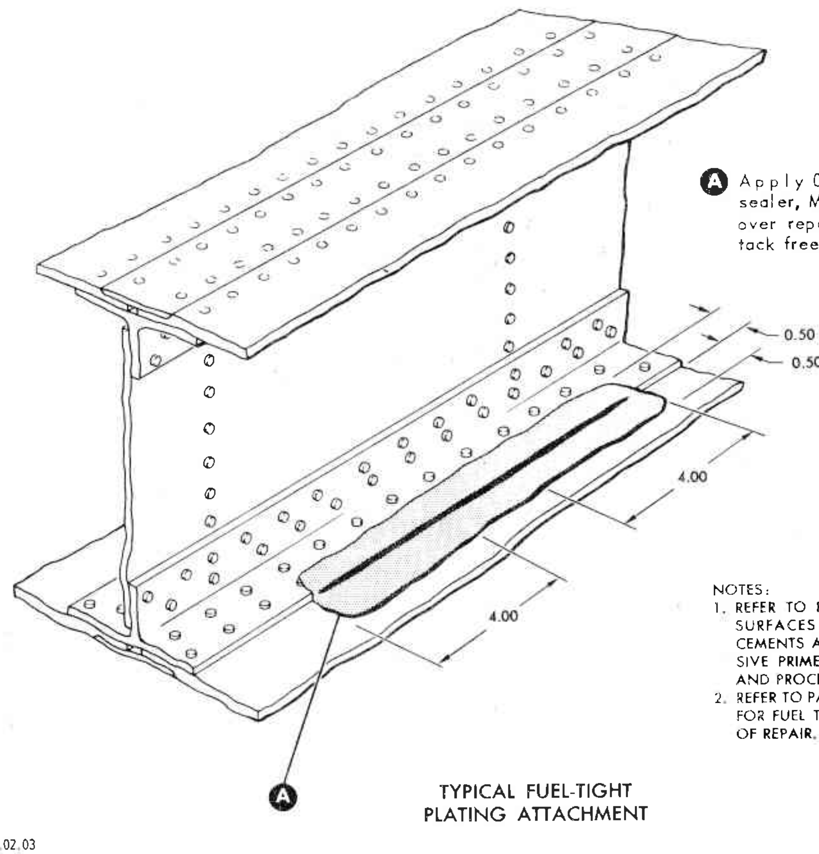
NOTES:

1. REFER TO PARAGRAPHS TITLED, "PREPARATION OF SURFACES FOR SEALING," "APPLICATION OF CEMENTS AND SEALERS," AND "STRUCTURAL ADHESIVE PRIME" FOR CORRECT SEALANT APPLICATION AND PROCEDURE.
2. TEMPORARY REPAIR TO CRACKED DOME NUT MAY BE MADE BY COATING DOME NUT INSTALLATION WITH 0.060 COAT OF SEALER, MILITARY SPECIFICATION MIL-S-8802. ALLOW SEALER TO DRY UNTIL TACK FREE.
3. REFER TO PARAGRAPH TITLED "FUEL TANK LEAK TEST" FOR FUEL TANK PRESSURE TEST AFTER COMPLETION OF REPAIR.

Figure 2–30. Dome Nut Replacement Repair



TYPICAL FUEL-TIGHT CORNER



A Apply 0.060 inch thick coating of fillet sealer, Military Specification MIL-S-8802 over repair area. Allow sealer to dry until tack free.

NOTES:
1. REFER TO PARAGRAPHS TITLED, "PREPARATION OF SURFACES FOR SEALING," "APPLICATION OF CEMENTS AND SEALERS," AND "STRUCTURAL ADHESIVE PRIME" FOR CORRECT SEALANT APPLICATION AND PROCEDURE.
2. REFER TO PARAGRAPH TITLED "FUEL TANK LEAK TEST" FOR FUEL TANK PRESSURE TEST AFTER COMPLETION OF REPAIR.

TYPICAL FUEL-TIGHT PLATING ATTACHMENT

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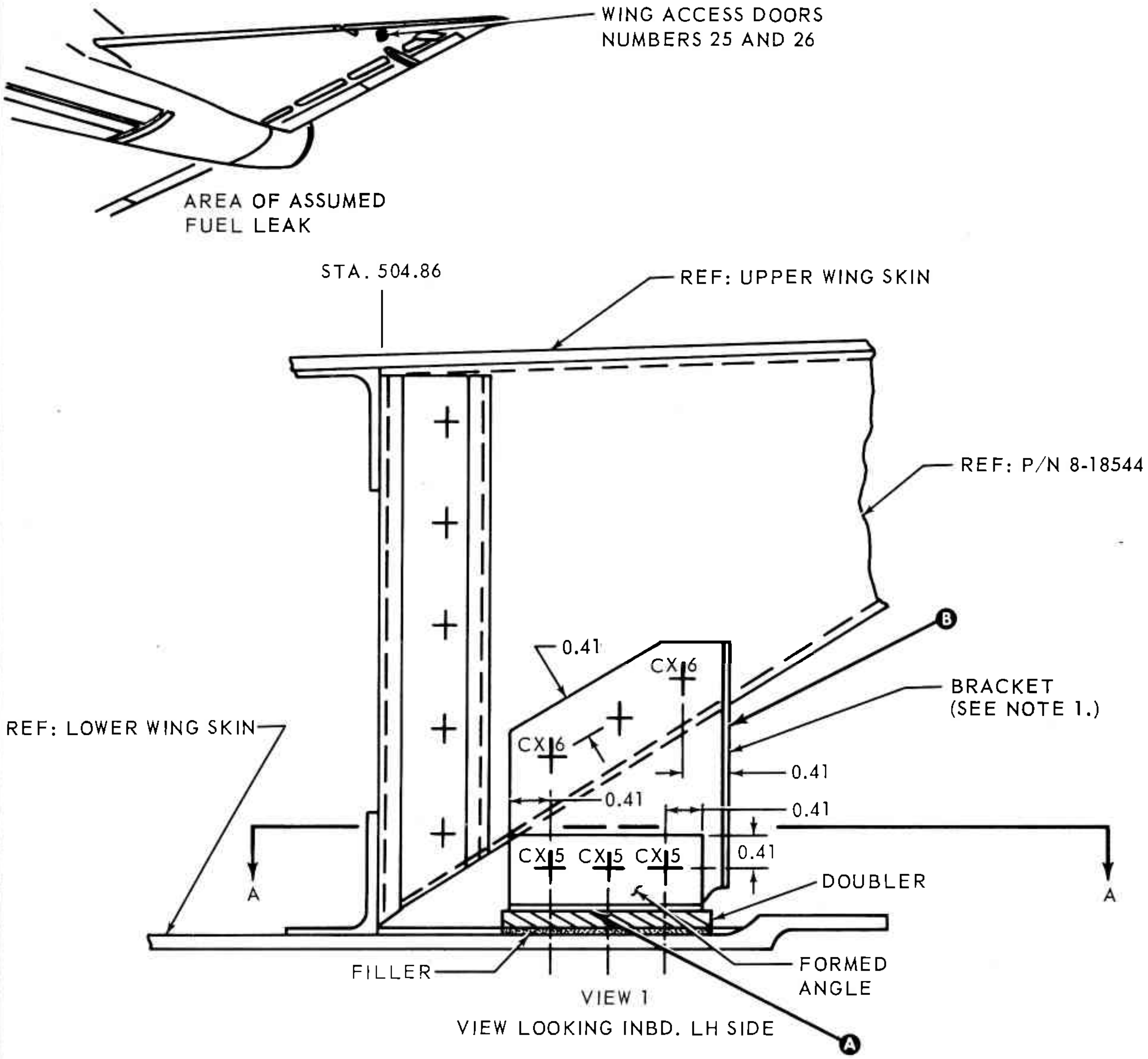
Figure 2-31. Wing Fuel Leak Repair

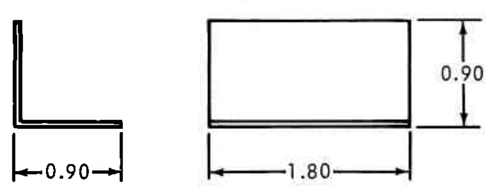


REPAIR FOR STRAYLOR RIVETS PULLING LOOSE FROM THE LOWER WING SKIN AT THE INDICATED LOCATION

1. BRACKET REPLACES P/N 8-18544
2. FILLER FORMED FROM 7075-T6 SHEET 0.051 THICKNESS
3. DOUBLER FORMED FROM 7075-T6 SHEET 0.125 THICKNESS
4. SEE DETAIL A & B FOR DIMENSIONS OF BRACKET AND ANGLE

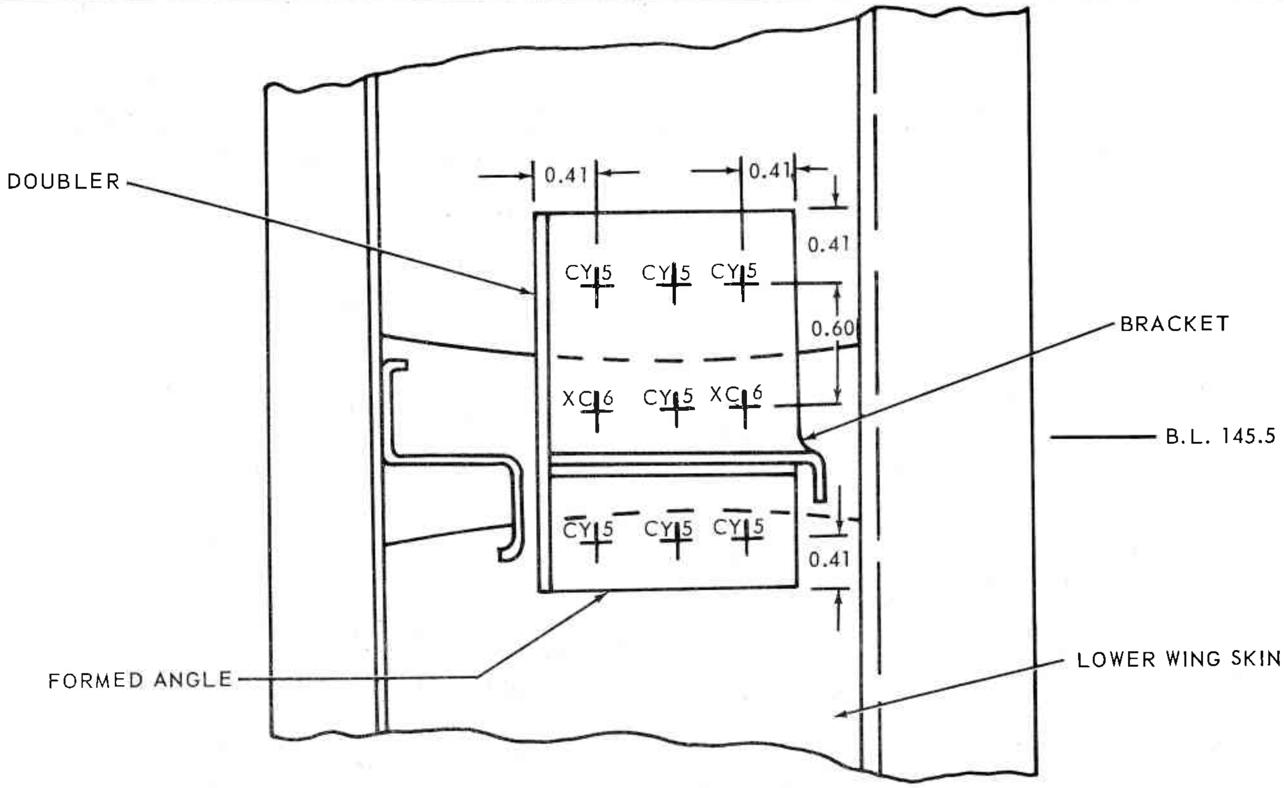
RIVET CODE & BASIC P/N's	CX - P/N MS20470DD XC - P/N 97-67000 CY - P/N MS20426DD

A VIEW 1

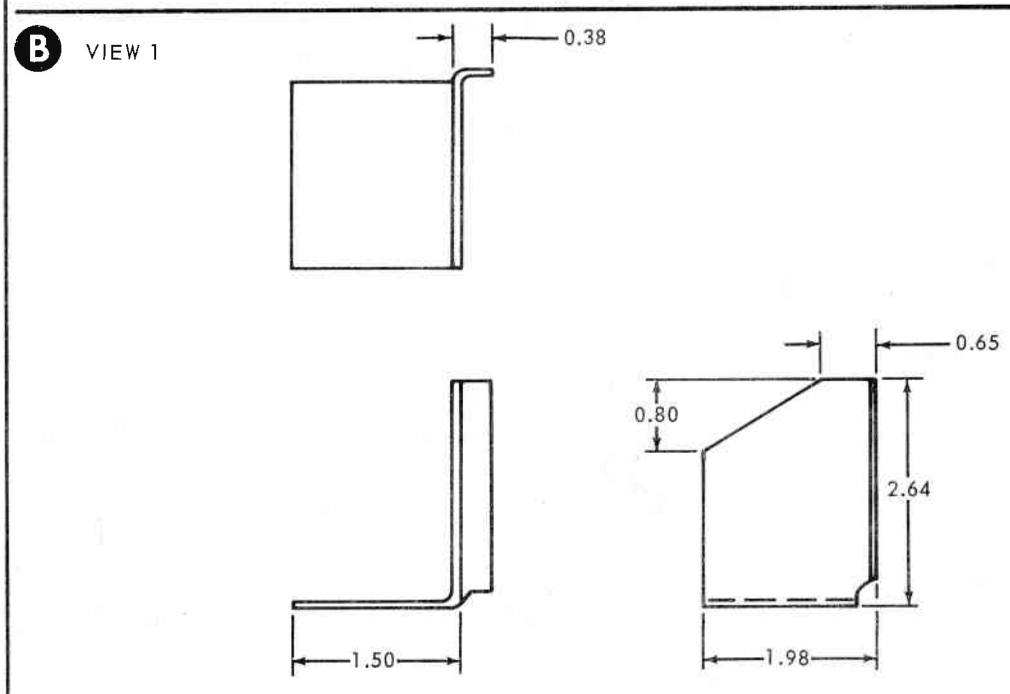


FORM ANGLE FROM 7075-0 COND. SHEET 0.063 THICKNESS

Figure 2-31A Fuel Leak (Straylor Rivets) Wing B.L. 145.5 Repair (Sheet 1 of 5)



SECTION A-A OF VIEW 1

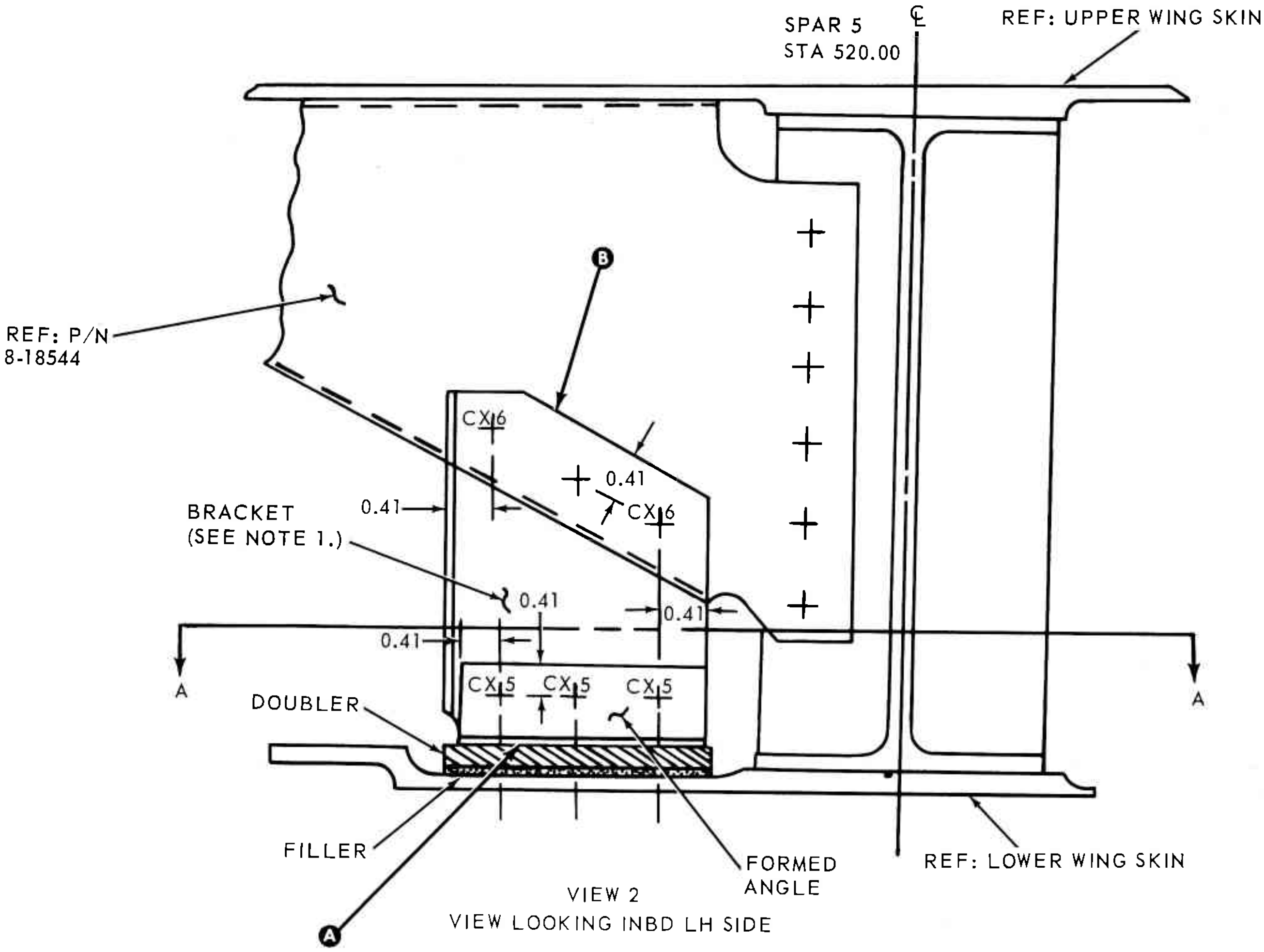


FORM BRACKET FROM 7075-0 COND. SHEET 0.063 THICKNESS

NOTES:

1. ALL BEND RADII 0.09
2. ALL RELIEF RADII 0.19
3. HEAT TREAT PER MIL-H-6088 TO T6 COND.

Figure 2-31A Fuel Leak (Straylor Rivets) Wing B.L. 145.5 Repair (Sheet 2 of 5)

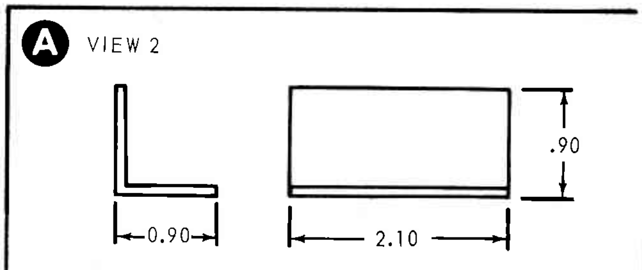


VIEW 2
VIEW LOOKING INBD LH SIDE

REPAIR FOR STRAYLOR RIVETS PULLING LOOSE FROM THE LOWER WING SKIN AT THE INDICATED LOCATION

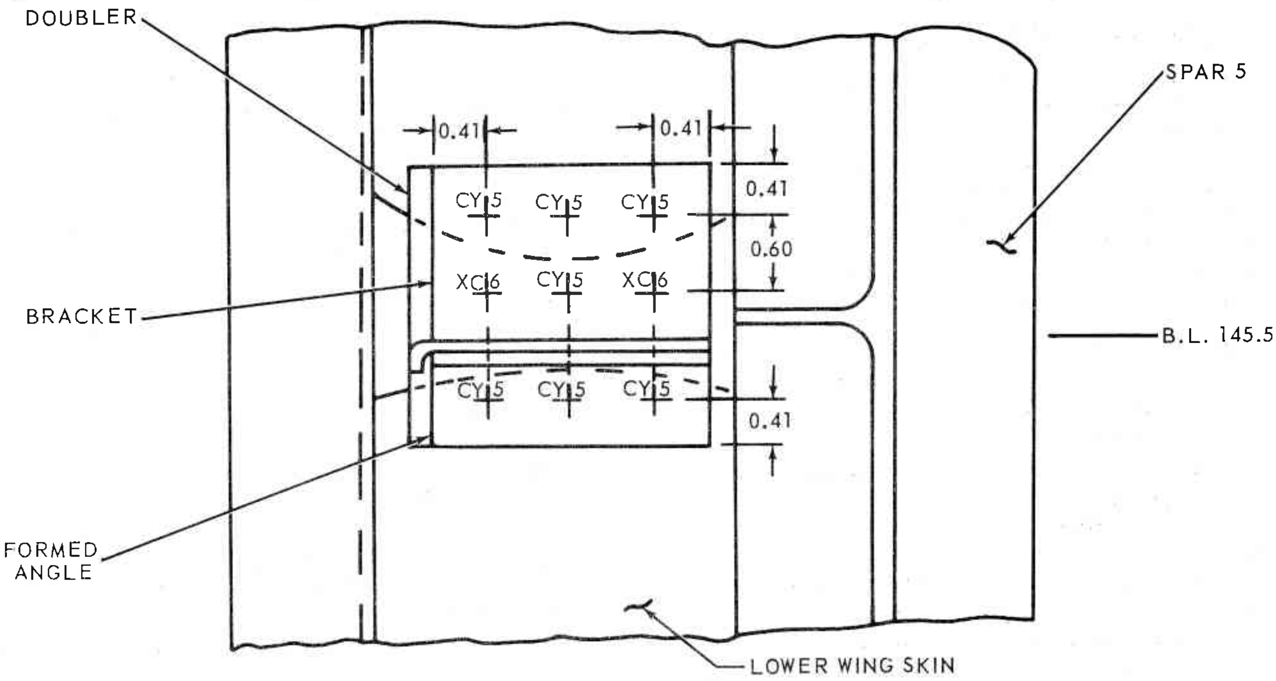
1. BRACKET REPLACES P/N 8-18544
2. FILLER FORMED FROM 7075-T6 SHEET 0.051 THICKNESS
3. DOUBLER FORMED FROM 7075-T6 SHEET 0.125 THICKNESS
4. SEE DETAIL A & B FOR DIMENSIONS OF BRACKET AND ANGLE

RIVET CODE & BASIC P/N's	CX - P/N MS20470DD
	XC - P/N 97-67000
	CY - P/N MS20426DD



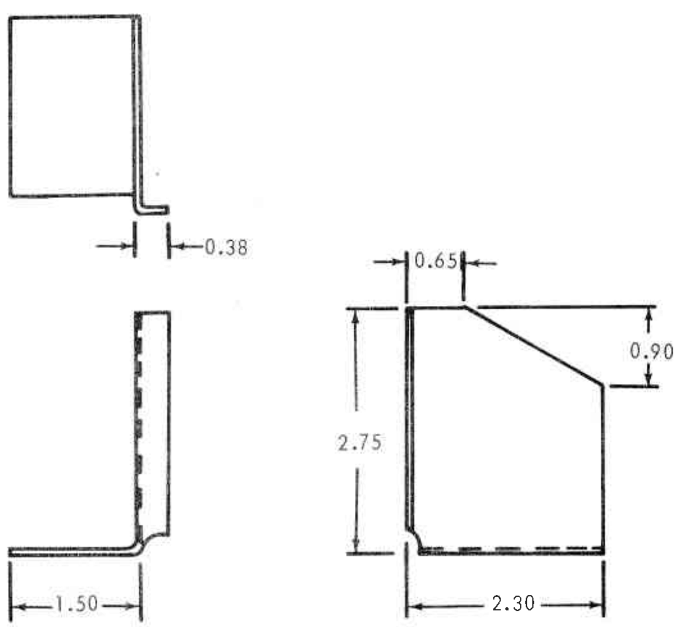
FORM ANGLE FROM 7075-0 COND. SHEET 0.063 THICKNESS

Figure 2-31A Fuel Leak (Straylor Rivets) Wing B.L. 145.5 Repair (Sheet 3 of 5)



SECTION A-A OF VIEW 2

B VIEW 2



FORM BRACKET FROM 7075-0 SHEET 0.063 THICKNESS

NOTES:

1. ALL BEND RADII 0.09
2. ALL RELIEF RADII 0.19
3. HEAT TRANT PER MIL-H-6088 TO T6 COND.

Figure 2-31A Fuel Leak (Straylor Rivets) Wing B.L. 145.5 Repair (Sheet 4 of 5)

FUEL LEAK REPAIR PROCEDURES, B. L. 145.5

1. Remove existing bracket and scotchweld tape in accordance with paragraph 2-35, T.O. 1F-106A-3.
2. Fabricate fillers to match contour of milled wing skin. Fabricate doubler from 7075-T6 0.125 thickness to fit flush over fillers and raised portion of milled wing skin.
3. Manufacture new bracket and angle as shown in Details A and B of Views 1 and 2.
4. Apply sealer MIL-S-8802 to all faying surfaces.
5. Attach fillers, doubler, angle and bracket to wing skin using (7) MS20426DD rivets as shown in Views 1 and 2.
6. Attach doubler and bracket to wing skin with (2) Straylor rivets P/N 97-67000 using existing rivet location.
7. Attach bracket to angle and former P/N 8-18544 using existing rivet holes in former and rivet pattern in Views 1 and 2 as a guide.
8. Apply 0.060 inch thick coating of fillet sealer MIL-S-8802 over entire repair. Allow sealer to dry until tack free.

NOTE

REFER TO PARAGRAPHS TITLED "PREPARATION OF SURFACES FOR SEALING," "APPLICATION OF CEMENTS AND SEALERS," AND "STRUCTURAL ADHESIVE PRIME" FOR CORRECT SEALANT APPLICATION AND PROCEDURE.

REFER TO PARAGRAPH TITLED "FUEL TANK LEAK TEST" FOR FUEL TANK PRESSURE TEST AFTER COMPLETION OF REPAIR.

RIVET HOLE LOCATIONS MAY VARY. LOCATE HOLES AS NECESSARY MAINTAINING GOOD SHOP PRACTICE.

Figure 2-31A Fuel Leak (Straylor Rivets) Wing B.L. 145.5 Repair (Sheet 5 of 5)

2-29. Rivet Substitution.

2-30. The Straylor fuel-tight rivet used to attach the wing plating to the wing internal structure in the wing fuel-tight areas has been especially designed for use on the F-106A/B airplanes and may not be substituted by any other type of rivet. See figures 1-34 and 1-35 for installation and removal of Straylor fuel-tight rivets. Rivets used in the wing leading edges, wing tips, and elevons may be substituted with different types of rivets according to the rivet substitution data given in Table 1-XXII.

CAUTION

Do not substitute screws for solid rivets in fuel tight repairs.

2-31. Interspar Structure Repairs (Spars, Ribs, and Wing Plating).

2-32. Interspar structure repairs involve the spars, ribs, and the plating covering these wing components. See figures 2-23, 2-24, 2-25, 2-27, and 2-29. All repairs in the fuel tank area will require sealing to provide fuel-tight seams. Figure 2-31 illustrates a typical fuel leak repair. Refer to paragraphs 2-35 through 2-50 for repair of wing plating areas that are bonded with Scotchweld tape. Refer to typical repair, Section X, for repairs to wing plating in non-fuel-tight areas. See figure 2-28 for replacement of damaged main landing gear side brace fitting.

2-33. Scotchweld Tape Application.

2-34. Scotchweld tape is applied to the faying surfaces of the wing structure in the fuel-tight areas during assembly. The completed wing assembly is then placed in a dry air circulating oven and is baked for one hour at 160°C (320°F). Any subsequent repairs made to the wing structure must be made with sealer, Military Specification MIL-S-8802,as shown on figure 2-31.
and baking process cannot be used for repairs to the wing structure. Separation of Scotchweld bonded components may be accomplished by the procedures given in paragraph 2-35.

2-35. Repairs Involving Scotchweld Tape.

2-36. Scotchweld is a very strong bonding medium which cannot readily be removed by any solvent currently available. Scotchweld bonded parts may be separated by using either one or a combination of the following methods:

1. First Method.
   a. Remove all fasteners from the parts to be separated.
   b. Fabricate a wedge from micarta or any non-metallic material that will not nick, dent or gouge the wing structure.
   c. Apply a layer of dry ice to the outside surface of the wing plating along the seam that is to be separated.
   d. Allow the dry ice to remain in contact with the outside surface of the wing plating until frost appears on the opposite or inside surface of the wing plating.
   e. Place the sharp edge of the wedge against the faying surface of the components to be separated.
   f. Using a leather or plastic mallet, strike the wedge with a single sharp blow.
   g. Repeat steps "e" and "f" at intervals of approximately 12 inches along the seam of the components being separated to assure positive breaking of the Scotchweld bond.

CAUTION

Do not attempt to drive the wedge completely between the components being separated.

2. Second Method.
   a. Same as step "a" in method one.
   b. Same as step "b" in method one.
   c. Apply CO_2 to the outer surface of the wing plating along the seam of the two components that are to be separated.
   d. Allow the CO_2 to remain in contact with the outer surface of the wing plating until frost appears on the opposite or inside surface of the wing plating.
   e. Same as step "e" in method one.
   f. Same as step "f" in method one.
   g. Same as step "g" in method one.

2-37. Identification of Leaks.

2-38. A discolored area around the source of trouble is usually a guide to the type of fuel leak. The discolored areas must be wiped dry and inspected frequently to evaluate and determine the exact source of the leak. The general types of fuel leaks are classified as follows:

a. Stain: A dry, discolored area.

b. Seep: A wet, discolored area.

c. Drip: Free droplets.

d. Run: A continuous stream.

2-39. Preparation for Fuel Leak Repair.

2-40. To prepare tank for fuel leak repair, proceed as follows:

a. Ground the airplane at four points: nose, tail, and each main landing gear.

NOTE

Use ohmmeter on each grounding connection; the resistance between any two grounding connections shall not exceed 10 ohms.

b. Defuel tank in accordance with T.O. 1F-106A-2-5-2-1.

c. Attach a portable ground blower duct to fuel access door area with a static bond connection as shown on figure 2-32.

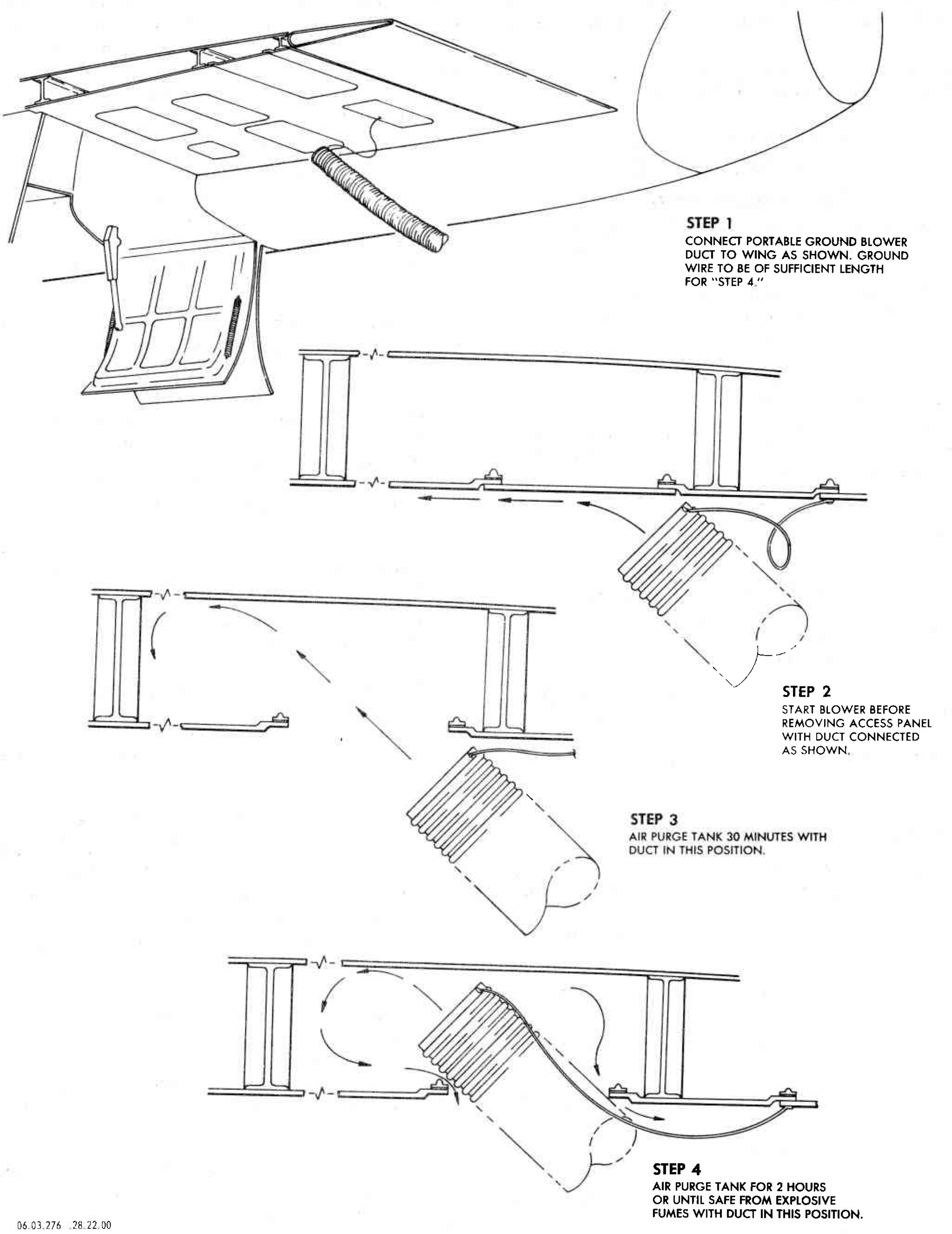


Figure 2-32. Air Purging of Fuel Tank

d. Start ground blower and remove access panel. Make sure the static bond between duct and airplane remains intact.

e. Allow blower to air purge tank for 30 minutes before mopping-up residual fuel in tank.

WARNING

JP-4 is a highly combustible fuel. It is also toxic when breathed for prolonged periods. Personnel working in fuel tank interior must wear a respirator connected to a fresh air supply, and an assistant must stand by at all times work is being performed in fuel tank interior.

f. Insert ground blower duct into tank as shown in figure 2-32 and air purge for two hours.

g. Stop blower and wait ten minutes; then check tank with combustible gas indicator, USAF Type R-1 (6665-530-0985) or equivalent.

NOTE

If the tank is safe, start the blower and begin repairs. If the tank is not safe, start the blower and continue air purging for two hours or until tank is free of explosive fumes when checked with the explosive vapor detector. Refer to T.O. 1-1-3 for additional information on air purging of fuel tanks.

h. Make repairs inside the fuel tank, observing the following precautions. Use an explosive-proof light. Use only air-powered drills and equipment. Refer to T.O. 1-1-3 for stock numbers of correct tools and equipment to be used in repair of fuel tanks.

2-41. Fuel System Purging.

2-42. Airplanes requiring a wing change or extensive maintenance to the fuel system should be purged with 100-octane aviation fuel Specification MIL-G-5572. Although the fuel system can be made explosion-proof by air purging, the toxious JP-4 fumes which remain when the airplane is placed in a closed hangar present a potential health hazard. To purge the fuel system, proceed as follows:

a. Defuel airplane in accordance with T.O. 1F-106A-2-5-2-1.

b. Refuel airplane with 100-octane aviation fuel, Specification MIL-G-5572.

c. Defuel airplane in accordance with T.O. 1F-106A-2-5-2-1.

d. Open all manual operated drain cocks, flush type drains, and filter drains.

e. Connect ground blower duct. Air-purge tanks as shown on figure 2-32.

f. Continue air-purging and enter "F" tank. Remove the manifold line between the CG fuel shutoff valve, the fuel check valve, and the refuel line to remove trapped fuel.

g. Use syphoning equipment and remove residual fuel trapped in tank areas.

NOTE

Any residual fuel which remains after syphoning may be mopped up, using only authorized lint free cloth, cotton only, cheesecloth, bleached, Type 2, Class 2, Stock Number 8305-205-3496, Specification Number CCC-C-440.

h. If airplane has CG fuel transfer system deactivated, remove flange from fuel transfer line in left and right "T" tanks.

NOTE

If CG transfer system is activated, attach a source of 5 to 10 psi dry air to one of the manifold check valves in the fuel pressurization and venting system to keep the CG fuel shutoff valve in each "T" tank open during purging operations.

i. Attach a source of 5 to 10 psi dry air at the left and right boost pump pressure switches, refueling adapter and the tank vent outlets.

j. Continue air purging precedure shown in figure 2-32 for approximately eight hours.

2-42A. Fuel System Purging, Alternate Procedure.

2-42B. The oil purge method can be used as an alternate for purging the fuel system. Refer to T.O. 1-1-3 for procedure.

2-43. Preparation of Surface for Sealing.

2-44. Prior to the initial mating of repair materials, all parts must be thoroughly cleaned. The parts must be completely free of moisture, grease, or other foreign matter to permit proper adhesion of the sealing materials. Suitable solvents and cleaners are listed in Table 11-X.

2-45. Application of Cements and Sealers.

2-46. All faying surfaces must be sealed at the time a repair is accomplished. Apply sealer, Military Specification MIL-S-8802, to the faying surfaces prior to riveting. After the riveting process, apply a smooth fillet of sealer, Military Specification MIL-S-8802, around the edges of the mated parts. The sealant should be tack free before the tank is filled with fuel. See figures 2-30 and 2-31 for examples of sealant application. Refer to Section XI for sealant material specifications.

2-47. Structural Adhesive Prime Repair In Fuel Tank.

2-48. The inner structure of the fuel tank is finished with structural adhesive prime EC-1290 at the time of fabrication. During the course of structural repair or inspection of the tanks, this prime may become scratched or portions of it removed. Scratched EC-1290 structural adhesive prime finish or exposed inner fuel tank structure, caused by removed finish, shall be coated with a thin brush coat of coating, MIL-S-4383 prior to refueling. Apply coating MIL-S-4383, as follows:

a. Thoroughly clean the surface to remove any moisture, grease, or foreign matter with clean cheesecloth dampened in aliphatic naphtha.

b. Spray or brush an even coat of MIL-S-4383 over the scratched or exposed area.

c. Do not disturb newly coated area for at least 55 minutes after application.

d. The MIL-S-4383 coating shall be allowed 12-hours drying time before the tank is filled with fuel.

2-48A. Corrosion Removal and Treatment, Integral Fuel Tank.

a. It is a permissible to mechanically remove corrosion pits from wings skins and access doors. Material removed shall not exceed 0.046 inch in depth. The following doors are exceptions:

(1) 8-18668 and 8-18667 doors -0.035 inch.

(2) 8-18664 door -0.039 inch.

Corrosion pits shall be ground out to form a saucer shaped depression within localized areas of one inch while maintaining nine inches between pit area centers.

b. Corrosion pits exceeding the permissible limits but not covering an area of more than 1/8 inch in diameter may be repaired by plugging with a MS20426DD6 rivet. Finish plug repair in accordance with Section II, paragraphs 2-44 and 2-46.

c. Remove residue from surfaces of secondary wing structure (tubing, clamps, fuel system components, etc.) using MEK, Specification TT-M-261.

d. Repair wing skin and access doors exceeding the above limits as per Section X, figure 10-1.

e. Treat and refinish all areas in accordance with T.O. 1F-106A-23 and this manual as applicable.

2-49. Repair of Leaks Around Rivets.

2-50. If leaks are found around rivets, the rivets must be replaced. Care must be exercised in drilling out the old rivet. When installing fuel-tight Straylor rivets, make certain that an undamaged 0.004-inch aluminum foil washer is installed under the head of the rivet. Refer to paragraphs 1-131 and 1-133, and see figures 1-34 and 1-35 for installation and removal of Straylor rivets.

2-51. Repair of Leaks Around Access Doors.

2-52. In the event that leaks occur around the fuel tank access door attachment screws, remove the screw in question and check the dome nut O-ring. If the dome nut O-ring has been damaged, the fuel tank will have to be drained, the access door removed and the dome nut in question removed and replaced with a new one. One of the most frequent causes of leaks around the fuel tank access doors is the insertion of overlength screws, thus causing damage to the dome nut. See figure 2-30 for dome nut replacement and leak sealing procedures.

2-53. Replacement of Access Door Gaskets.

2-54. When determining the source of fuel leaks around access doors, the gasket may be at fault. Two types of material are used for access door gaskets: neoprene and silicone rubber. Each of these materials has specific bonding requirements which must be met to ensure that access doors maintain a fuel-tight seal.

2-55. Replacement of Access Door Neoprene Gasket.

2-56. The following procedure is prescribed for replacement of the access door neoprene gasket:

a. Drain the fuel tank.

b. Remove access door.

c. Remove old gasket and thoroughly clean the door surface to which the new gasket will be applied. Refer to Section XI for list of cleaners, solvents, and repair materials.

d. Apply a thin, even coat of red or uncolored cement EC-776, Type I to surface of access door and to checked (cloth imprint) side of neoprene gasket.

e. Allow cement to dry until finger touch will not lift strings of cement.

f. Apply gasket to door, pressing firmly to assure positive contact.

g. Allow bond to cure approximately one hour before proceeding with step "h."

h. Using a hollow-end tool, cut screw holes through neoprene gasket from door face opposite gasket using existing screw holes in the door as a guide.

i. Trim excess neoprene neatly, avoiding ragged or torn edges that may cause splits to form under pressure.

2-57. Replacement of Access Door Silicone Rubber Gasket.

2-58. To replace access door silicone rubber gaskets, proceed as follows:

a. Perform steps "a" through "c" as outlined in paragraph 2-56.

b. Clean gasket with a lint-free cloth soaked with aliphatic naphtha, Specification TT-N-95.

c. Roughen gasket with 180-grit sandpaper.

d. Coat gasket with EC 1663 cement and allow to dry for one hour.

e. Spray or paint the seal faying surface of the access door with EC 1662 primer and allow to dry for at least three minutes.

f. Apply gasket to door and air cure for 24 hours.

g. Perform step "h," paragraph 2-56.

h. Install access door.

2-59. Air Pressure Test of Fuel Tank Structure.

2-60. For pressure testing purposes, after a completed repair in the main fuel tank area, wing fuel tanks No. 1, No. 2 and No. 3 must be tested as a single unit. If a repair is made in the wing transfer "T" tank, it must be pressure tested as an independent unit. Refer to paragraph 2-63 for pressure test procedures in the "T" tank. Refer to Section IV in this manual for repair and pressure testing information concerning the fuselage "F" tank.

2-61. Equipment Requirements.

FIGURE	NAME	TYPE	USE AND APPLICATION
Refer to T.O. 1F-106A-2-3-2-1.	High-Pressure Air Compressor.	MC-11 (4310-624-4457)	To provide air pressure to fuel tanks.
	Air pressure lines (3).	0 to 3,000 psi capacity.	To connect air pressure source and manometer to fuel tanks.
	CAUTION Mercury manometer shall not be used.		
	Gage (0 to 30 psi).	Calibrated in ¼-inch increments (6685-526-6881).	To measure air pressure in fuel tanks.

2-61. Equipment Requirements (Cont).

FIGURE	NAME	TYPE	USE AND APPLICATION
2-34	Rubber Plug (2).		To plug vent lines.
	Bubble Fluid.	MIL-L-25567.	To detect leaks.
	Freon 12 Gas.		
	Freon Leak Detector.	Type H-1 General Electric (10312596-H-1) or Type H-6 (4940-630-7947).	
2-33	Air Test Dummy Door.		To provide entrance of air pressure and Freon gas into fuel tanks and to provide fuel tank over pressure relief.
	Pressure Cap (4).	AN292-6 (4730-277-6539).	To plug sense lines.
	Pressure Cap (2).	AN929A-4 (4730-278-5006).	
	Union Plug (4).	AN815-6 (4730-187-0486).	
	Union Plug (2).	AN815-4 (4730-278-3242).	

2-62. Procedure for Pressure Testing Main Fuel Tank After a Completed Repair.

a. Remove access door No. 11 from underside of No. 2 fuel tank. See figure 1-2 for exact location of main fuel tank access doors. Insure that all other main fuel tank access doors are properly secured. Replace damaged door gaskets as directed in paragraph 2-53.

b. Disconnect the 3.00-inch fuel transfer line in main wheel well and cap that portion of the line leading into the main fuel tank.

c. Insert a rubber plug into the 1.50-inch vent line. Compress rubber plug until vent line is completely sealed. See figure 2-34 for fabrication of rubber plug.

d. Cap the 0.25-inch sense line with an AN815-4 plug and an AN929A-4 pressure cap.

e. Mount an air test dummy door in place of fuel tank access door No. 11. See figure 2-33 for fabrication of air test dummy door.

f. Adjust regulator on compressed air source to insure supply of dry air at 10 psi. Close all valves before proceeding with the next step.

g. Connect 10-psi air pressure line from the air source to the inlet fitting on air test dummy door.

h. Deleted.

i. Deleted.

j. Slowly open the air pressure valve on 10-psi air source and pressurize tank to 7.50 (+0.0 -0.5) psi.

CAUTION

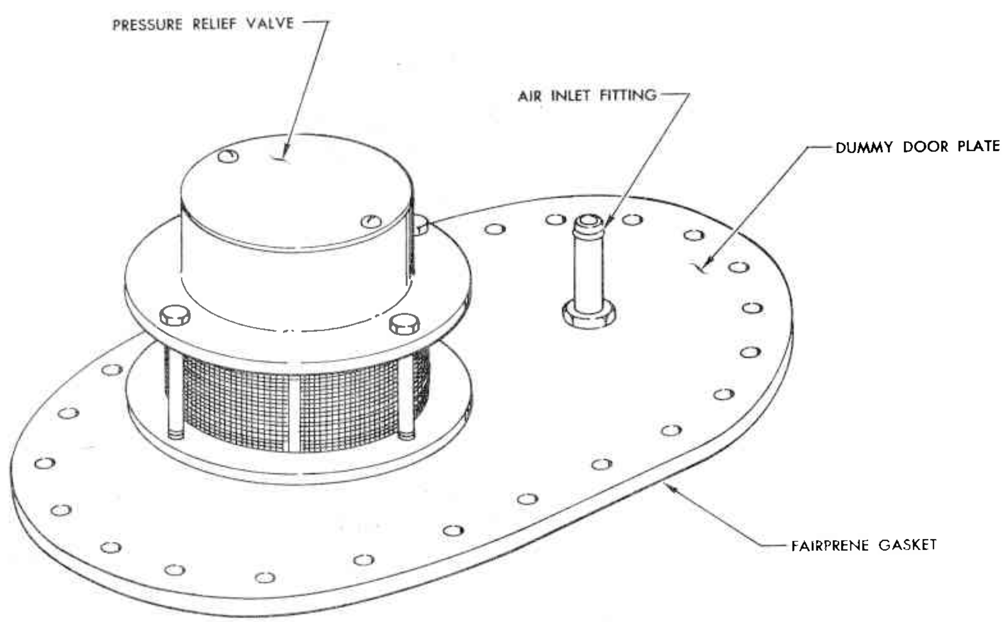
Open valve slowly to prevent movement and/or damage to air pressure gage.

k. Apply bubble fluid to repair area and check for leaks.

l. If a leak is detected, relieve all air pressure from fuel tank, remove air test dummy door, repair the leak, replace air test dummy door, and recheck for leaks according to steps "j" and "k." Repeat this process until no leaks are detected.

m. Upon completion of pressure test with no leaks detected, relieve the fuel tank of all air pressure and remove all test equipment.

n. Remove pressure cap from 3.00-inch fuel transfer line in main wheel well. Remove rubber plug from 1.50-inch vent line on under side of No. 2 tank. Remove



FABRICATION INSTRUCTIONS FOR MAIN FUEL TANK LEAK TEST DUMMY DOOR.

a. Fabricate dummy door from 0.125 7075-T6 sheet stock.
b. Use one of two existing doors, which are located in the number two tank at B.L. 145.00, Station 510.00 and B.L. 164.00, Station 547.00 as a pattern for door outline and screw hole attachment locations.
c. Fabricate 1.00-inch wide door flange seal from 0.032 Fairprene 5570. Cement seal to dummy door with red or uncolored EC-776.
d. Install AN807 hose fitting or equivalent in dummy door. Select fitting to match size of available air hose.
e. Secure fitting with AN-924 nut or equivalent.
f. Install pressure relief valve set to relieve pressure in excess of 7.5 psi. See Note 2 for relief valve size.
g. Install 0-30 psi pressure gage on dummy door.

NOTES:
1. REFER TO PARAGRAPH "FUEL TANK LEAK TEST" FOR TEST PROCEDURES.
2. ANY RELIEF VALVE USED MUST BE LARGE ENOUGH WHEN OPENED TO PERMIT A GREATER VOLUME OF AIR TO EXIT THROUGH THE VALVE THAN THE VOLUME OF AIR ENTERING THROUGH THE INLET FITTING. EXAMPLE: IF INLET FITTING HAS A 1.00-INCH INSIDE DIAMETER, THEN RELIEF VALVE SHOULD HAVE A 1.50-INCH OR GREATER INSIDE DIAMETER AT ITS SMALLEST OPENING.
3. USING EXAMPLE SHOWN ABOVE FOR MAIN TANK, FABRICATE AIR TEST DUMMY DOORS FOR TRANSFER TANK AND FUSELAGE TANK EXCEPT THAT RELIEF VALVES SHALL BE SET TO RELIEVE PRESSURE IN EXCESS OF 20 PSI.

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Figure 2-33. Fabrication of Main Fuel Tank Air Test Dummy Door

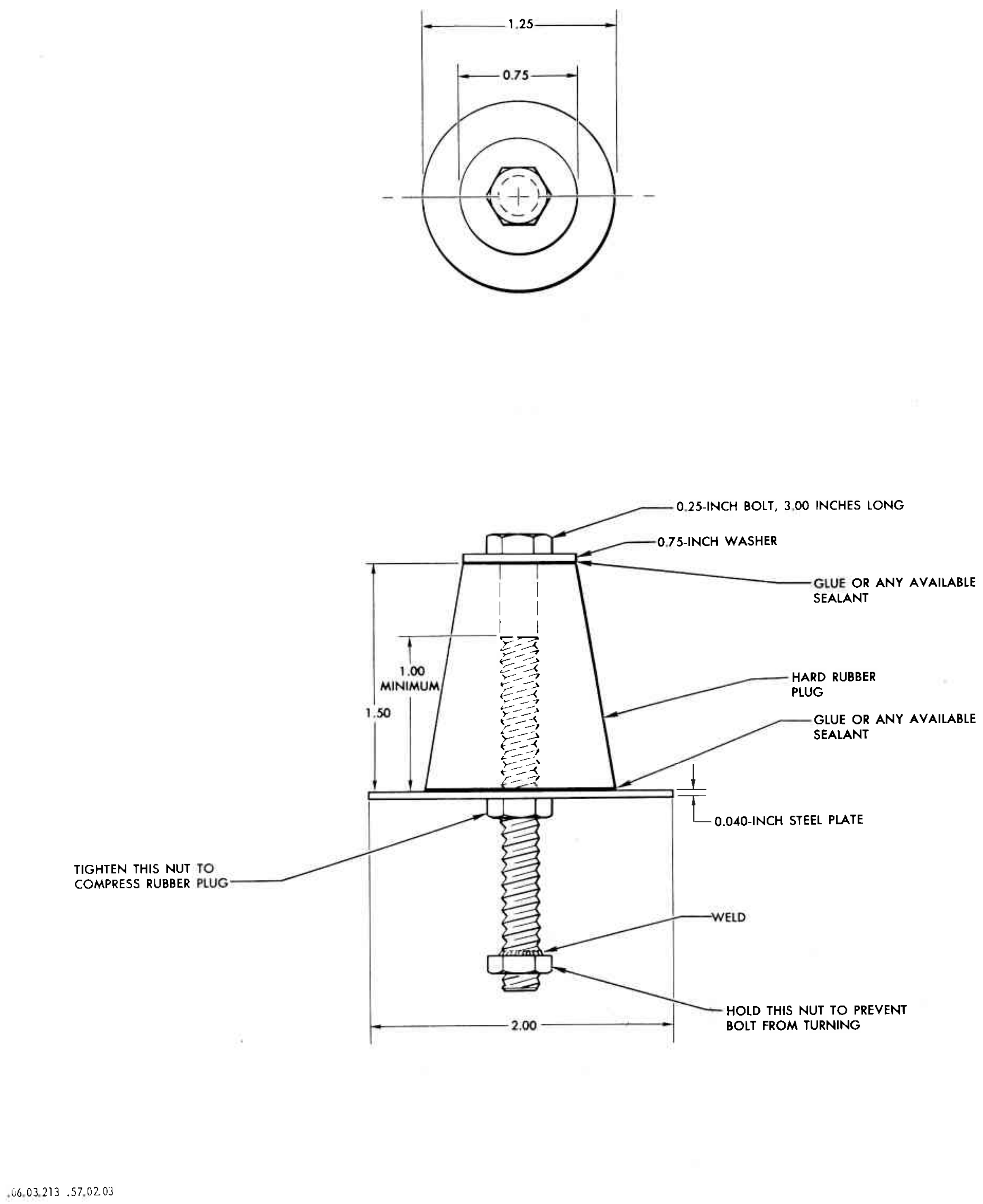

Figure 2-34. Rubber Plug

AN815-4 plug and AN929A-4 pressure cap from 0.25-inch sense line on underside of No. 2 tank.

CAUTION

Failure to remove all caps and plugs used during test could result in structural damage to fuel cell or failure of fuel system components during subsequent fueling, defueling and fuel system check out operations.

o. Reconnect 3.00-inch fuel transfer line in main wheel well.

p. A fuel system operational check must be performed after completion of step "o". Refer to T.O. 1F-106A-2-5-2-1 for procedure.

2-63. Procedure for Pressure Testing Wing Transfer "T" Tank After a Completed Repair.

a. Remove the engine. Refer to T.O. 1F-106A-2-4-2-1 for procedure.

b. Remove "T" tank access door No. 25 from underside of wing. See figures 1-2 and 1-3 for exact location of "T" tank access door No. 25.

c. Disconnect and cap the 3.00-inch fuel transfer line in main wheel well. Refer to paragraph 2-61 for equipment required for pressure testing and to T.O. 1F-106A-2-5-2-1 for a schematic of the fuel system.

d. Cap the following lines into the "T" tank with air pressure caps. These lines are centrally located on the inside of the fuselage between stations 556.75 and 593.46 at water line −16.00.

1. The 0.38-inch air pressure lines to the inboard and outboard CG fuel shutoff valves.
2. The 0.38-inch air pressure line to the inboard and outboard fuel low level pilot float valves.
3. The 0.25-inch air sense line from the "T" tank to the "T" tank air pressure regulator.
4. The 0.25-inch air sense line from the pressure relief valve to the "T" tank air pressure regulator.
5. The 0.38-inch air pressure lines to the emergency air pilot float valve (two lines for the left-hand side and one line for the right-hand side).
6. The 1.25-inch air pressure line.

CAUTION

Air pressure caps must have a red warning streamer at least one foot long.

e. Mount an air test dummy door in place of the No. 25 access door. See figure 2-33 for fabrication of air test dummy door. Insure that all other main fuel tank access doors are properly secured. Replace damaged door gaskets as directed in paragraph 2-53.

f. Adjust regulator on compressed air source to insure supply of dry air at 20 psi.

CAUTION

Close all valves before proceeding with the next step.

g. Connect 20-psi air pressure line from the air source to the inlet fitting on the air test dummy door.

h. Deleted.

i. Disconnect the 0.38-inch air pressure line to the vent valve line located on the inboard side of the "T" tank closing rib between stations 556.75 and 593.46 at water line −16.00.

j. Connect an air pressure line from second source of air pressure to vent valve.

k. Adjust safety relief valve on air test dummy door to relieve pressure at 20 (±0.5) psi.

l. Deleted.

m. Apply 15 (±1) psi from second source of air pressure to vent valve.

NOTE

Air pressure on vent valve to remain constant throughout pressure testing of "T" tank.

n. Slowly turn on primary air pressure and pressurize "T" tank to 7(±0.5) psi.

o. Apply bubble fluid over entire area of repair and check for leaks. If a leak is detected, relieve the "T" tank of all air pressure, remove the air test dummy door, repair the leak, replace the air test dummy door, and repeat steps "n" and "o" until no leaks are detected.

p. Repeat steps "n" and "o" using 10.0 (±0.5) psi. If a leak is detected, follow the process outlined in step "o". Recheck for leaks per steps "n" through "p" until no leaks are detected.

WARNING

High pressures are used in the following steps. Before proceeding with the remainder of test, rope off the pressure test area and hang warning signs indicating that a pressure test is in progress. Remove all personnel not actually engaged in the pressure test. All safety precautions must be observed.

q. Introduce Freon 12 gas into "T" tank for remainder of test.

NOTE

Prior to testing "T" tank for leaks, make a check of air inlet and pressure relief valves for escaping Freon gas, since escaped gas may contaminate the surrounding air and make the test invalid.

r. Increase pressure to 15 (±0.5) psi.

s. Inspect thoroughly around repair area for leaks with General Electric H-1 leak detector.

NOTE

If a leak is detected at this step, repair should be completely reworked and a new pressure test started.

t. If no leaks were detected after step "s," increase pressure to 20 (±0.5) psi and hold for three minutes.

u. Reduce pressure in "T" tank to 10 (±0.5) psi and check for leaks with bubble fluid or General Electric H-1 leak detector.

NOTE

A leak detected at this point usually indicates a structural failure of the repair. If a structural failure is determined, repair must be reworked and a new pressure test started.

v. After all leak tests prove satisfactory, release all air pressure from "T" tank and vent valve. Remove all test equipment.

w. Remove air pressure caps from all lines which were capped in steps "c" and "d."

CAUTION

Failure to remove all caps and plugs used during test could result in structural damage to fuel cell or failure of fuel system components during subsequent fueling, defueling and fuel system check out operations.

x. Reconnect all lines that were disconnected in steps "c," "d" and "i."

y. A fuel system operational check must be performed before final sealing of the "T" tank. Refer to T.O. 1F-106A-2-5-2-1 for procedure.

z. Reinstall the engine. Refer to T.O. 1F-106A-2-4-2-1 for procedure.

2-64. Wing Drag Angle Repair.

2-65. The wing drag angles which assist in attaching the wings to the fuselage also provide a smooth aerodynamic contour between the wings and the fuselage. The drag angles must be kept structurally sound within the negligible damage limits, as stated in Table 2-I. See figure 2-35 for repair of wing drag angles.

2-66. Wing Leading Edge Repairs.

2-67. The leading edge is the most critical part, aerodynamically, of the entire wing. The boundary layer is thinnest at this point, becoming progressively deeper downstream. As the boundary layer deepens, the wing becomes less critical aerodynamically. Consequently the least critical area of the wing is at the trailing edge. All repairs to the leading edge outer plating must be of the flush type to restore the aerodynamic contour. For information concerning repairs, see figures 2-36 through 2-38. When damage to a leading edge section exceeds 30 percent, it is considered more efficient to replace the component. Refer to Table 2-I for allowable limits of negligible damage.

2-68. Wing Tip Repairs.

2-69. See figure 2-39 for repair of the wing tip structure. Refer to Table 2-I for negligible damage limits.

2-70. Wing Attachment Repairs.

2-71. Repairs to the wing attachment section are shown on figure 2-7. Original installation of the wing does not require bushings in the wing attachment fittings and they are required only if tolerances are not met because of wear or damage. All new bushings shall receive a coating of zinc chromate primer before installation. Caution must be taken to insure that the attaching bolt and inside surface of the bushing are kept free of foreign matter. When possible, ream the inside diameter of the bushing after installation.

2-72. Elevon Repairs.

2-73. See figures 2-16 through 2-20 for repairs to the elevon structure other than the honeycomb section. Refer to Table 2-I to determine the negligible damage limits for wing group components. Repairs made on the elevon assemblies will not require rebalancing of the elevons. Refer to paragraph 2-76 for repairs to the honeycomb structure of the elevon.

2-74. Elevon Fairing Repairs.

2-75. Refer to Table 2-I for negligible damage limits to elevon fairing plating. See figure 2-17 for repair of actuator fairing skin.

2-76. Elevon Honeycomb Repairs.

2-77. Figure 2-21 illustrates the method of rebonding separation of the elevon honeycomb section from the upper or lower skin. See figure 2-22 for the procedure used to remove moisture from the honeycomb sections of the elevon. See figures 10-11 through 10-46 for general honeycomb repairs.

2-77A. Repair or Replacement of Skin and Core of Elevon Trailing Edge Wedges—Depot Level Only.

a. Materials.

(1) Core filler, meeting FMS 1026, Class VI, Type B.

(2) Film adhesive, meeting FMS 1013, Form III.

(3) Aluminum Sheet, 2024-T6,clad x 0.025".

(4) See Table 2-0 for substitute adhesives.

b. Equipment.

(1) The elevon has an inboard and outboard section (see figures 2-13 and 2-14). Each section consists of a covered torque box and a metal honeycomb trailing edge. The trailing edge is the item to be repaired.

(2) A tooling fixture for assembly of inboard and outboard elevons, RH and LH, bears the number 60SCJ2048. This fixture aligns the trailing edge with the torque box.

WARNING

- Accomplish all work in authorized areas with proper protection devices to safeguard health. Work involving application of potentially harmful materials must be accomplished in an exhaust ventilated area as required by SGB. Environment assesment of this process has been made.

- The solvents listed in this process order are hazardous. They evaporate quickly, producing toxic vapors in the air. They can cause dermatitis if skin contact occurs. T.O. 42A1-1-3 should be consulted for the proper handling and storage of these solvents.

- Use protective eye or face shields and rubber gloves when working with solvents. Use proper safety container for solvent to prevent splashes and continuing emanation of vapors. Adequate ventilation must be available.

- The sealant and adhesive materials listed in paragraph 3.0 contain chemicals and violatile materials known to be toxic and irritating to the skin and eyes. These sealants and adhesives can also cause dermatitis if skin contact occurs.

- The following protective actions should be taken in mixing sealants and adhesives.

  (a) Wear gloves, a rubber apron and protective eye glasses or a plastic face shield. Coveralls are recommended. Apply protective hand cream on the hands before putting on gloves. SGB recommends that cotton gloves be worn under the rubber or PVC gloves.

  (b) Mixing operations should be in accordance with SM-ALC MAOI 65-4.

  (c) If sealant or adhesive gets on the outer clothes, they should be immediately changed. The sealants and adhesives if left on may cause local skin irritation and dermatitis.

- At the end of the day or after completion of the adhesive or solvent handling operations for the day, wash hands thoroughly and then apply skin conditioner. Remember, cleanliness is the best defense against dermatitis.

c. Process.

(1) Replacement of one skin.

(a) Removal of elevon wedges from elevon assembly is described in paragraph 2-18A.

(b) Remove the damaged skin as follows:

1. Remove fasteners.

2. Remove the skin using dry ice and wooden wedges.

(c) Clean any fuel contaminated honeycomb with trichloroethylene; invert and dry for 8 hours at 180°F.

(d) Repair the core as necessary. Pot with core filler (Epocast 1310 + hardener 9228).

(e) Vapor degrease the assembly.

(f) Make a new skin from 2024-T6, clad, 0.025". Use the old skin as a template.

(g) Clean the skin for bonding in the plating shop.

(h) Bond the skin as follows:

1. Use FMS 1013 film adhesive. Align the skin to the wedge with tack rivets.

2. Using a flat plate as a bond form, vac-bag and put into the autoclave.

3. Retain in the autoclave for 1 hour at 350°F and 15 psig net.

4. Cool for 1 hour. Discard all the vac-pac materials except the bleeder cloth.

(i) Grind off the excess adhesive.

(j) Install rivets.

(k) Inspect the wedge for honeycomb bond integrity using tap test.

(2) Replacement of two skins requires a repeat of steps c(1)(b) throu c.(1)(k).

(3) Replacement of two skins and a core requires the following steps:

(a) Replace one skin as specified above.

(b) Remove the second skin and replace the core as required. Cut the wedge-shaped core on the Do-all saw, insert it into place, and pot it according to the process specified in paragraph c.

(c) Repeat steps c(1)(b) through c(1)(k).

d. Inspection and Testing.

(1) Subject all the skins to the "tap" test to determine the quality of honeycomb bonding.

(2) Make a random selection of wedges for other NDI inspection by the appropriate quality organization.

(3) Lay up a lap shear specimen in accordance with T.O. 1F-111A-3. Prepare the specimen with the same adhesive used in the honeycomb repair and cure it simultaneously with the repair piece. Use the lap shear specimen tester in the shop. A minimum value of 2000 psi must be attained.

e. References.

(1) T.O. 1F-106A-3, Structural Repair Instructions.

(a) Paragraph 2-17 describes elevon structure.

(b) Paragraph 2-18 describes inspection and repair of surfaces containing honeycomb core.

(c) Figures 2-13 and 2-14 show elevon structure, inboard and outboard panel.

(d) Figure 2-21 shows elevon honeycomb separation repair.

(e) Figure 10-41 shows honeycomb repair on trailing edge.

(f) Table 10-1 lists honeycomb repair adhesives and processing.

(2) T.O. 1F-111A-3, Structural Repair Instructions.

2-78. External Fuel Tank Repairs.

2-79. All repairs to the external tank must be of the flush-patch type to provide aerodynamic smoothness. The repairs must also provide positive fuel-tight sealing. See figure 2-40A for repair of the external fuel tank. Tank dents that do not exceed 3/16 inch in depth or do not exceed three inches in diameter may be filled with MIL-S-38228 aerodynamic smoothing compound. Fill dent with compound and smooth to contour. Cure 24 hours at 77°F; sand to smooth surface.

2-80. **Pylon Support Beam Elongated Hole Repair.**

2-81. Pylon support beams and rib found with elongated holes will be repaired by reaming the elongated hole to the next bolt size and installing existing bolt with sleeves made from stainless steel tubing cut to size for proper fit. Bolts will be removed one at a time and elongated holes reamed as follows: Holes for NAS1103 bolts (3/16) will be reamed to 0.250/0.255, holes for NAS1105 bolts (5/16) will be reamed to 0.375/0.380, holes for NAS1106 bolts (3/8) will be reamed to 0.438/0.433. Holes will be reamed with a hand reamer from the existing oversize hole in increments of 1/32 inch. After reaming cut the tube to the grip length of the bolt and deburr both ends of sleeve. Reinstall the same bolt, nut and washer. The sleeves will be made from the following tubing specification: MIL-T-8606 0.250 outside diameter with 0.028 wall thickness. MIL-T-6845 0.375 outside diameter with 0.028 wall thickness, and MIL-T-8606 0.438 outside diameter with 0.028 wall thickness.

2-82. Replacement of Forward Pylon Support Beams.

2.83 Procedures for support beams, Part Nos. 65E33321-3 and -4 are as follows:

a. Remove external fuel tank/pylon assemblies from the aircraft (if installed) in accordance with instructions in T.O. 1F-106A-2-5-2-1.

b. Remove wing access doors, Nos. 51 and 52 (Part Nos. 65J33342-1, -2, -3, -4, -5, or -6). Retain door assembly and screws for reinstallation.

c. Remove vent tubes, Part Nos. 65A35901-1 (LH) and 66A40633-1 (RH). Retain for reinstallation. Plug all open ends of tubing to prevent entry of foreign material.

d. Drill out all attachment rivets from support angles, Part Nos. 65J33330-23 (LH) and -24 (RH), and remove angles from wing. Retain for reinstallation.

e. Remove support beams, Part Nos. 65E33321-3, and -4. Retain all attaching hardware, consisting of bolts, Part Nos. NAS1103-7 (3 each), NAS1103-6 (3 each), NAS1106-11 (6 each), nuts, Part No. H19300-3 (6 each), washers K19301-3 (6 each), K19301-6 (6 each), and AN960D10 (3 each); and shims Part No. 66C40784, for use in installation of support beams.

f. Remove nut plates, Part No. NAS1067A3 (4 each), channel nuts, Part No. NAS689P7-5 (2 each), and shims Part No. 66C40784 (2 each), from support beams, Part Nos. 65E33321-3 and -4, and retain for reinstallation on support beams.

NOTES

If unthreaded portion (shank) of bolts provided with locating tool Part Nos. 65J34870-1 or -2, does not extend into support beams, Part Nos. 65E33321-3 or -4, at least 1/4-inch, use bolts and barrel nuts, which attach pylon to aircraft with the locating tool to provide proper alignment of support beams.

g. Position and secure support beam, Part No. 65E33321-3, on upper forward end of locating tool, Part No. 65J34870-1 (LH), and support beam, Part No. 65E33321-4, on upper forward end of locating tool, Part No. 65J34870-2 (RH), using the bolts and nuts (without washers) supplied with the tool.

h. Position locating tool, Part Nos. 65J34870-1 (LH) and 65J34870-2 (RH), with support beam attached, in the approximate location where old support beams were removed. Install bolts and nuts provided with tool in rear support fitting, Part Nos. 65E33322-1 (LH) and 65E33322-2 (RH), using the indexing bolts at the existing door attachment hole in the wing (aft inboard hole).

NOTE

If will be necessary to hold the forward end of the locating tool up until the locating tool, Part No. 65J34870-3, is installed. Bolts in rear support beam to rib may be loosened as required to allow alignment without deforming or deflecting the locating fixture.

i. Temporarily install support angles, Part Nos. 65J33330-23 (LH) and -24 (RH) in their original locations.

j. Install locating tool, Part No. 65J34870-3, under aft flange of support beams, Part Nos. 65E33321-3 and -4. Temporarily place shim, Part No. 66C40784, between support beam, Part Nos. 65E33321-3 and -4 aft flange and locating tool, Part No. 65J34870-3, thick end outboard. Secure locating tool Part No. 65J34870-3 to the existing door attachment holder, clamp support beams, Part No. 65E33321-3 and -4 to locating tool Part No. 65J34870-3.

k. Locate and mark holes (where bolts, Part Nos. NAS1103 and NAS1106, were removed in step e) through wing ribs onto new support beams, Part Nos. 6533321-3 and -4.

l. Remove locating tool, Part No. 65J34870-3, support angles, Part Nos. 65J33330-23 and -24, locating tools Part Nos. 65J34870-1 and -2, and support beams, Part Nos. 65E33321-3 and -4.

m. Drill and ream holes in support beams, Part Nos. 65E33321-3 and -4, which were marked in step k; rear holes 0.380/0.375 inch (6 places each beam), forward holes 0.196/0.191 inch (6 places each beam).

n. Install support beams, Part No. 65E33321-3 and -4, using the attachment hardware removed in step e. Fabricate shim (laminated shim stock), Part No. 65J33330-3, to extend to the lower edge of the support beam. Locating fixture must be installed while the support beam to rib hardware is being tightened.

NOTE

Trim the self aligning 3/8 inch diameter washer, Part No. K19301-6, in the lower outboard location in the support beam, as required, to permit the washer to seat on the flange and eliminate fillet radii interference.

o. Install support angles, Part Nos. 65J33330-23 (LH) and -24 (RH). Add a shim (laminated shim stock) 2.85 inches by 1.20 inches in gap between support beam and angle. Secure shim by applying MIL-S-8802 sealant to shim just prior to installation.

p. Temporarily install access doors, numbers 51 and 52. Drill 0.196/0.191-inch diameter holes (14 each) in support beams, Part Nos. 65E33321-3 and -4, to correspond with appropriate holes in the door. Remove doors from wing. Retain for reinstallation.

q. Position the nut plates, Part No. NAS1067A3, and channel nuts Part No. NAS689P7-5, on the upper surface of each flange of the support beams. Position the shims, Part No. 66C40784, to correspond with the holes on the lower side in each flange of the support beam. Attach the nut plates, channel nuts, and shims with rivets Part No. MS20426AD3.

r. Reinstall vent tube, Part Nos. 65A35901-1 (LH), and 66A40633-1 (RH).

s. Reinstall access doors numbers 51 and 52, utilizing attachment screws removed in step b.

t. Reinstall external fuel tank/pylon assemblied in accordance with instructions in T.O. 1F-106A-2-5-2-1.

2-84. Replacement of AFT Pylon Support Beams.

2-85. Procedures for Support Beam Part Nos. 65E333221-1 and -2 are as follows:

a. Remove external fuel tank/pylon assemblies from the aircraft (if installed) in accordance with instructions in T.O. 1F-106A-2-5-2-1.

b. Remove wing access doors, numbers 51 and 52 (Part Nos. 65J3334-1, -2, -3, -4, -5, or -6). Retain door assembly and screws for reinstallation.

c. Remove support beams Part Nos. 65E333322-1 and -2. Retain all attaching hardware consisting of bolts, NAS1103-11 (3 ea), NAS1105-9 (2 ea) NAS1105-10 (5 ea), NAS1105-11 (1 ea); nuts MS21059-L3 (3 ea), H19300-5 (5 ea), MS21061-L5 (3 ea), washers AN960D106 (3 ea) K19301-5 (5 ea), AN960D516 (1 ea); and shims Part No. 66C40784, for use in installation of new support beams.

d. Remove nut plates, Part No. MS21075L3 (3 ea), channel nuts, Part No. NAS689P7-3 (4 ea), and shims Part No. 66C40784 (4 ea) and retain for reinstallation on new support beams.

NOTE

If unthreaded portion (shank) of bolts provided with locating tool, Part Nos 65J34870-1 or -2, does not extend into support beams, Part Nos. 65E33322-1 or -2 at least 1/4 inch, use bolts and barrel nuts, which attach pylon to aircraft with the locating tool to provide proper alignment of support beams.

e. Position and secure support beam Part No. 65E33322-1, on upper aft end of locating tool, Part No. 65J34870-1 (LH), and support beam, Part No. 65E33322-2, on upper forward end of locating tool, Part No. 65J34870-2 (RH) using the bolts and nuts (without washers).

f. Position locating tool, Part Nos. 65J34870-1 (LH) and 65J34870-2 (RH), with support beam attached, in the approximate location where old support beams were removed. Install bolts and nuts provided with tool in front support fitting, Part Nos. 65E33321-3 (LH) and 65E33321-4 (RH), and install the indexing bolt at the existing door attachment hole in the wing (aft inboard hole).

NOTE

Bolts in forward support beam to rib may be loosened as required to allow alignment without deforming or deflecting the locating fixture.

g. Locate and mark holes (where bolts, Part Nos. NAS1103, and NAS1105, were removed in step c) through wing ribs onto new support beams Part Nos. 65E33322-1 and -3. Determine shim requirement between beams and ribs. Drill 0.191/0.196 holes (8 each) in aft of support beams to correspond with door attachment holes.

h. Remove locating tool, Part No.65J34870-3, locating tools Part Nos. 65J34870-1 and -2, and support beams, Part Nos. 65E33322-1 and -2.

i. Drill and ream holes in support beams, Part Nos. 65E33322-1 and -2, which were marked in step g forward holes 0.317/0.312 inch (5 places each beam) holes 0.196/0.191 inch (3 places each beam) and 0.317/0.312 inch (3 places each beam). Fabricate shims as required and drill to match support beams.

j. Install support beams, Part No. 65E33322-1 and -2 using the attached hardware removed in step c. Fabricate shim (laminated shim stock), Part No. 65J33330-3, to extend to the lower edge of the support beam. Locating fixture must be installed while the support beam to rib hardware is being tightened.

k. Temporarily install access doors, numbers 51 and 52. Drill 0.196/0.191 inch diameter holes (14 each) in support beams, Part Nos. 65E33322-1 and -2 to correspond with appropriate holes in the door. Remove doors from wing. Retain for installation.

l. Position the nut plates, Part No. NAS689P7-3 and channel nuts, Part No. MS21075L3 on the upper surface of each flange of the support beams. Position the shims, Part No. 66C40784, to correspond with the holes on the lower side in each flange of the support beam. Attach the nut plates, channel nuts, and shims with rivets Part No. MS20426AD3.

m. Reinstall access door numbers 51 and 52, utilizing attachment screws removed in step b.

n. Reinstall external fuel tank/pylon assemblies in accordance with instructions in T.O. 1F-106A-2-5-2-1.

TABLE 2-0. — MATERIALS USED IN ELEVON WEDGE (OLD AND NEW)

COMPONENT	ORIGINAL 1F-106A-3 MATERIAL	NEW REPAIR 1F-111A-3 MATERIAL
Skin	2024-T6 Clad sheet 0.025"	No change
Honeycomb, aluminum	Nonperforated 1/8" HEX, 0.0015" Foil	FMS-1019 Type I C (5052)
Primer for Honeycomb	FM-47	None required
Film Adhesive	AF-31	FMS 1013 Form III (AF-130)
Primer for Film Adhesive	FM-47	None required
Core Filler	E-3045	FMS 1026, Class VI Type B (Plastilock 654-HE)
Tape	Narmco	None required
Core Cleaner Solvent	MEK	Trico
Aerodynamic Smoother	EC 1653	FMS 1048 (EA-934)
Injection Adhesive for Delaminated Honeycomb	Epon VIII Epon 3119, Epon 1469	FMS 1102, FMS 1104 (Aerobond 2185)

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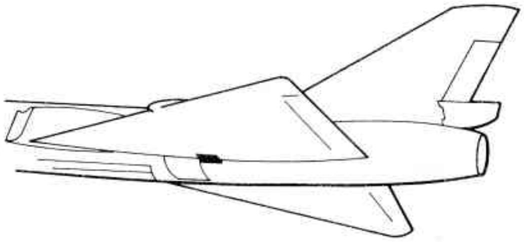


REPAIR PROCEDURE

a. Remove rivets and screws from damaged wing drag angle.

b. Insert 0.016 stainless steel strip under drag angle at cut line to protect wing/fuselage skin.

c. Cut away damages portion of wing drag angle.

d. Fabricate a repair angle from 7178-T6 bare sheet to match cutaway portion of wing drag angle, as shown in detail.

e. Install repair angle and drill holes to match existing holes in fuselage and wing, as shown in installation view.

f. Remove repair angle and machine countersink all holes.

g. Refer to Paragraph 1-185 for finish requirements. Seal fraying surfaces with MIL-S-81733 sealant.

h. Reinstall repair angle and rivet to wing. (NAS 1670 Blind bolts may be substituted for MS20426 rivets)

i. Install screws through repair angle and fuselage.

j. Fill all gaps in repair with MIL-S-38228 fairing compound.

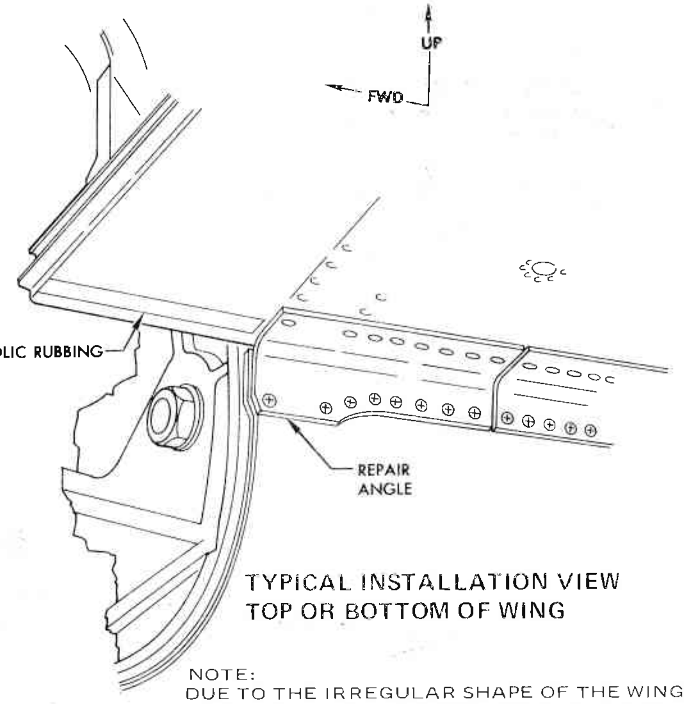
k. Deleted

l. Paint repair area according to applicable paint schedule.

m. Cracks up to 1 inch in length running along the radius and not closer than 6 inches to each other or the end of part may be stop drilled using a 1/8 inch drill. Cracks exceeding these limits or running across the angle shall be repaired per above repair procedure.

NOTE

IF PHENOLIC RUBBING STRIP SHOWN IN INSTALLATION VIEW IS DAMAGED, INSTALL A NEW STRIP OF THE SAME DIMENSIONS. INSTALL NEW STRIP WITH MIL-S-38228 SEALER. CLEAN INSTALLATION AREA BEFORE APPLICATION.



NOTE:
DUE TO THE IRREGULAR SHAPE OF THE WING DRAG ANGLE, ALL DIMENSIONS FOR THE REPAIR PARTS MUST BE TAKEN BY ACTUAL MEASUREMENT OF THE DAMAGED ANGLE. REPAIR PROCEDURE CAN BE USED ANYWHERE ON DRAG ANGLE.

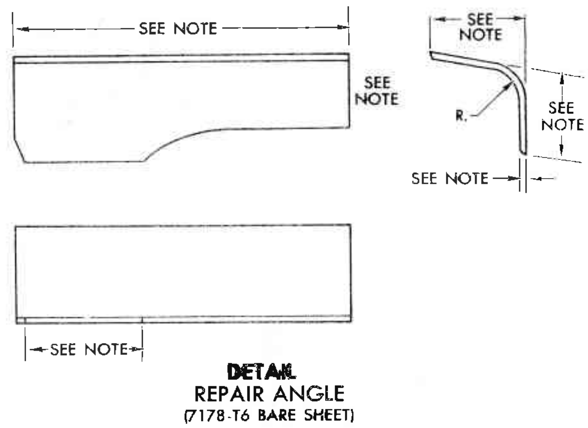


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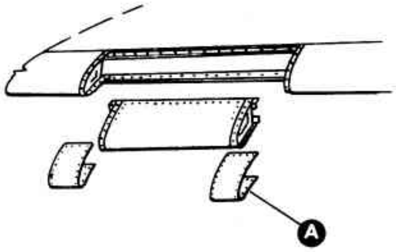
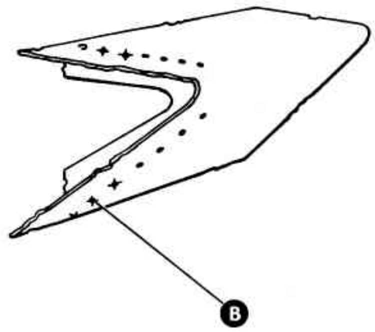
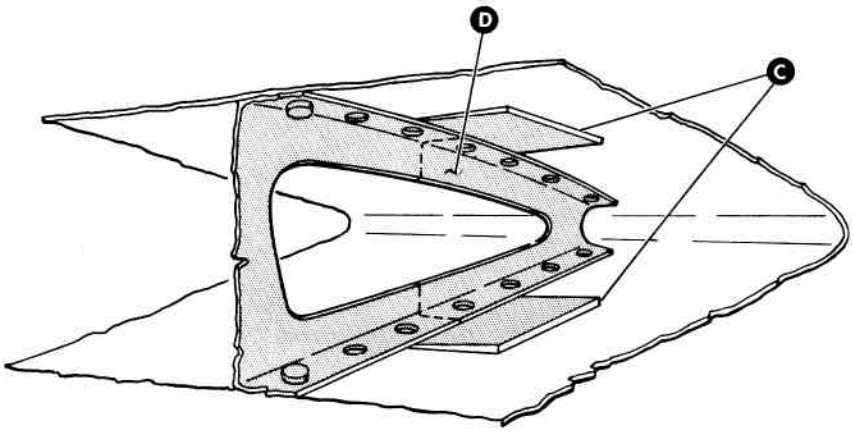
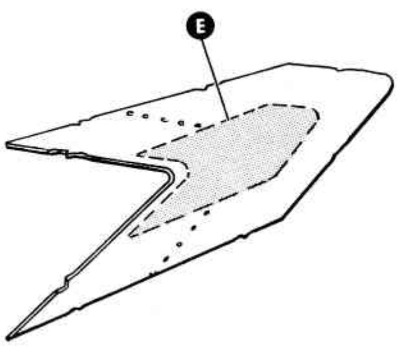
Figure 2-35. Wing Drag Angle Repair









A Remove attaching screws, panel assemblies, and damaged leading edge section.

B Drill out rivets in damaged portion of rib and skin as shown, and four additional rivets as indicated by (+).

C Wedge thin micarta blocks between skin and rib as shown.

D Cut away damaged portion of rib. Maximum trim allowance on rib 2.50 inches. Remove micarta blocks.

E Clean up damaged area in wing leading edge to regular shape. Break all sharp edges and round all corners.

NOTE:
WHEN DAMAGE TO RIB EXCEEDS 2.50 INCHES, REPLACE RIB.

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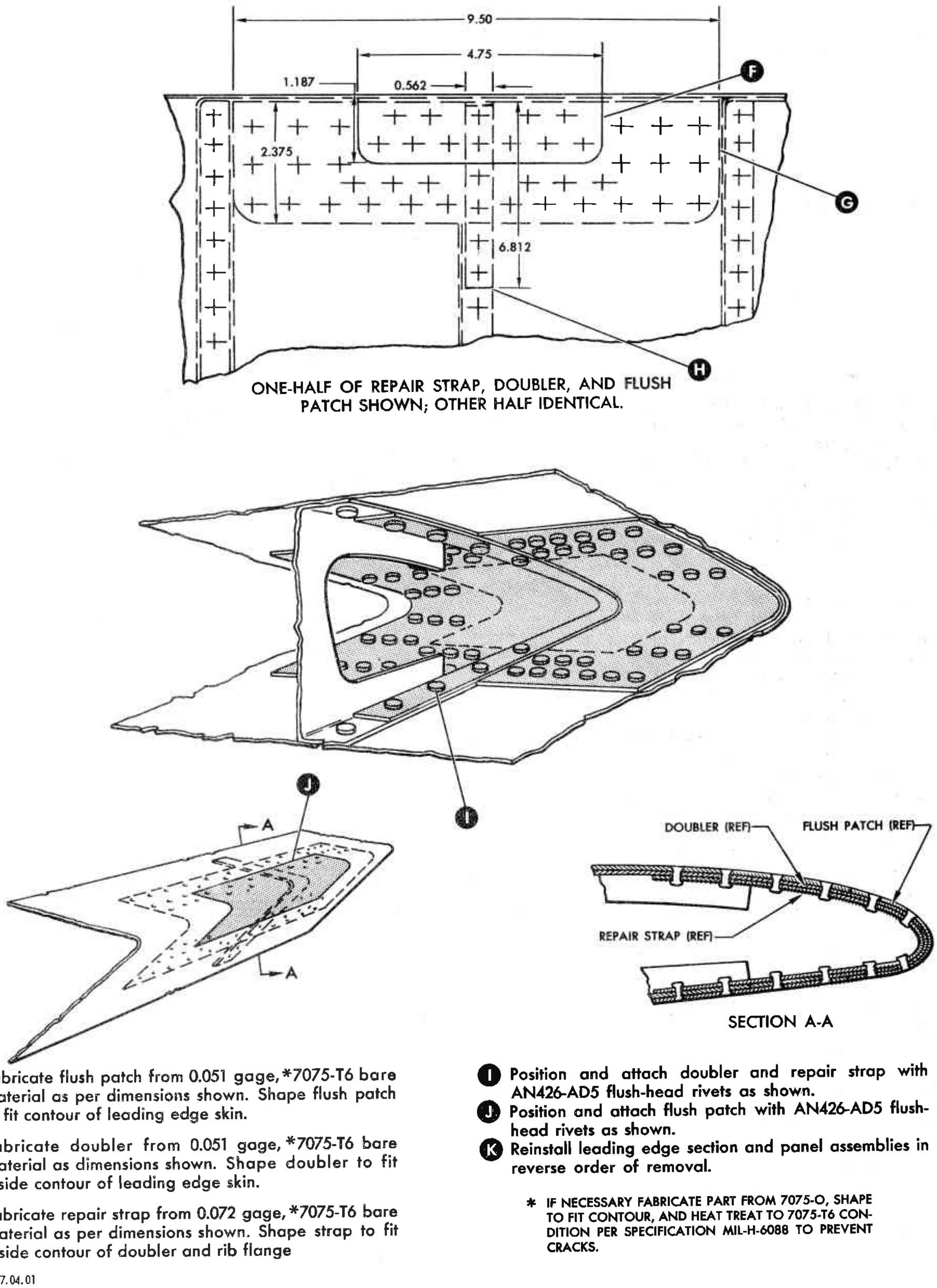
Figure 2-36. Wing Leading Edge Repair (Sheet 1 of 2)



F Fabricate flush patch from 0.051 gage, *7075-T6 bare material as per dimensions shown. Shape flush patch to fit contour of leading edge skin.

G Fabricate doubler from 0.051 gage, *7075-T6 bare material as dimensions shown. Shape doubler to fit inside contour of leading edge skin.

H Fabricate repair strap from 0.072 gage, *7075-T6 bare material as per dimensions shown. Shape strap to fit inside contour of doubler and rib flange

I Position and attach doubler and repair strap with AN426-AD5 flush-head rivets as shown.

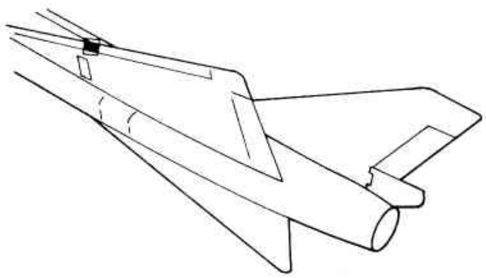
J Position and attach flush patch with AN426-AD5 flush-head rivets as shown.

K Reinstall leading edge section and panel assemblies in reverse order of removal.

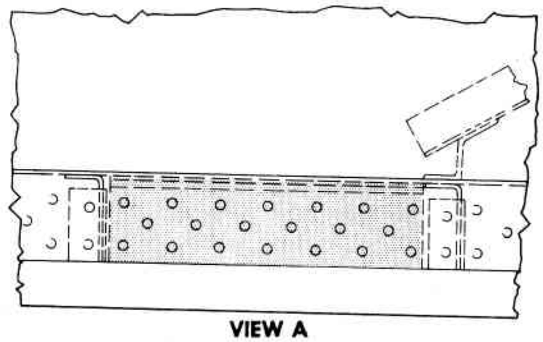
* IF NECESSARY FABRICATE PART FROM 7075-O, SHAPE TO FIT CONTOUR, AND HEAT TREAT TO 7075-T6 CONDITION PER SPECIFICATION MIL-H-6088 TO PREVENT CRACKS.

06.03.126-2 .57.04.01

Figure 2-36. Wing Leading Edge Repair (Sheet 2 of 2)



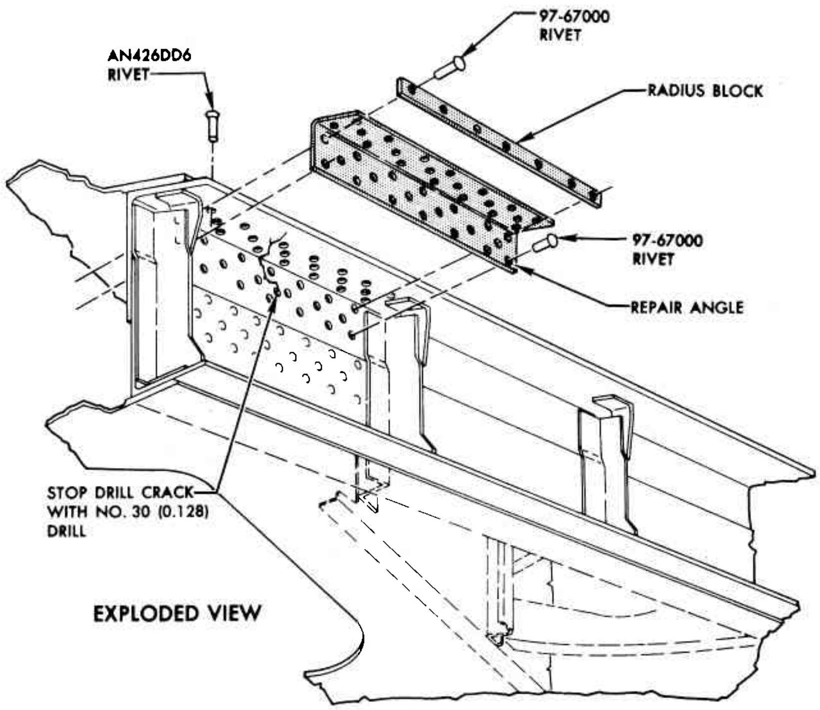
AREA OF ASSUMED DAMAGE



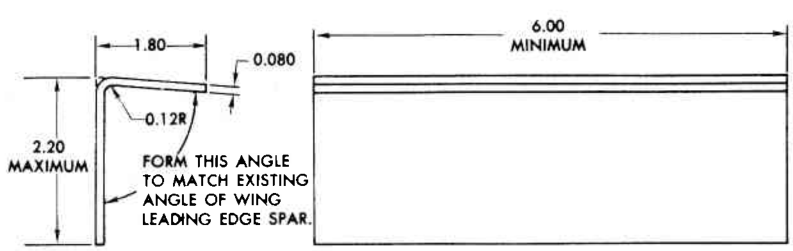
VIEW A
PLAN VIEW OF WING UPPER SURFACE

REPAIR PROCEDURE

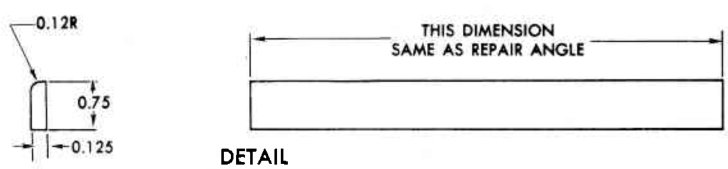
a. Remove damaged leading edge panel.
b. Remove all paint and primer from damaged area of leading edge spar. Use a clean cloth dampened with MEK, Federal Specification TT-M-261 for cleaning.
c. Inspect damaged leading edge spar and surrounding structure with fluorescent penetrant to determine extent of damage.
d. Stop-drill crack with a No. 30 (0.218) drill.
e. Fabricate repair angle from 7075-0 material using dimensions shown. Repair angle must be heat treated per Specification MIL-H-6088 to a -T6 condition after forming. Fabricate radius block from 7075-T6 sheet as shown.
f. Remove an access door from lower surface of wing plating to gain access to inner side of damaged area.
g. Using a No. 10 (0.1935) drill, remove rivets from upper surface of wing plating as indicated.
h. Install repair parts as shown and hold in place with clamps.
i. Using existing holes in wing plating as a guide, drill holes through repair angle. Use a No. 10 (0.1935) drill.
j. Layout and drill holes as indicated through repair parts and leading edge spar. Use a No. 10 (0.1935) drill.
k. Remove repair parts, then remove all drill chips and burrs from repair parts and from wing repair area.
l. Apply a protective coating to wing repair area and parts. See figure Primer and Paint Coatings in Section I.
m. Apply a coat of sealer, Military Specification MIL-S-8802, to faying surfaces of repair parts. Refer to T.O. 1-1-3 for mixing and application procedures for sealer.
n. Reinstall repair parts and hold in place with clecos.



EXPLODED VIEW



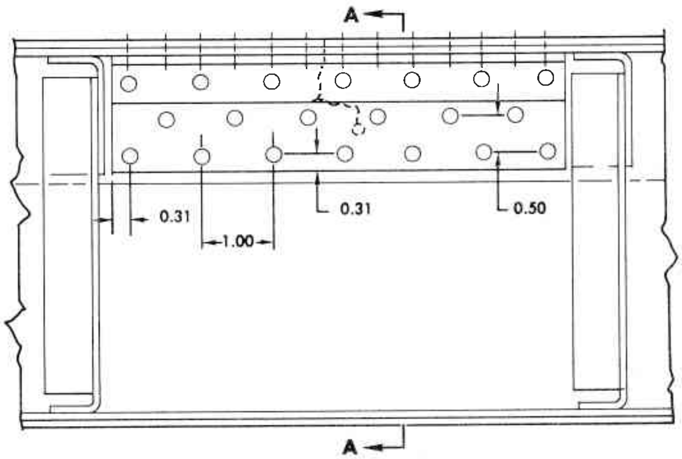
DETAIL
REPAIR ANGLE



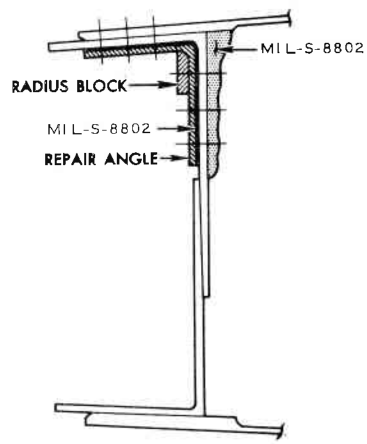
DETAIL
RADIUS BLOCK

06.03.231-1B

Figure 2–37. Repair of Wing Leading Edge Spar (Sheet 1 of 2)



ASSEMBLED VIEW



SECTION A-A

o. Rivet repair parts to wing plating with CY rivets. Refer to Table 1-XXIII for rivet basic code information.
p. Rivet repair parts to wing leading edge spar with XC rivets.
q. Apply a coat of sealer, Military Specification MIL-S-8802 over repair area on inside of wing fuel tank as shown.
r. Deleted.
s. Reinstall fuel-tight access door. Refer to paragraph Replacement of Access Door Gaskets in this section for fuel-tight access door sealing procedures.
t. Replace or repair wing leading edge panel and install with original screws. Replace any screws with damaged or stripped threads.

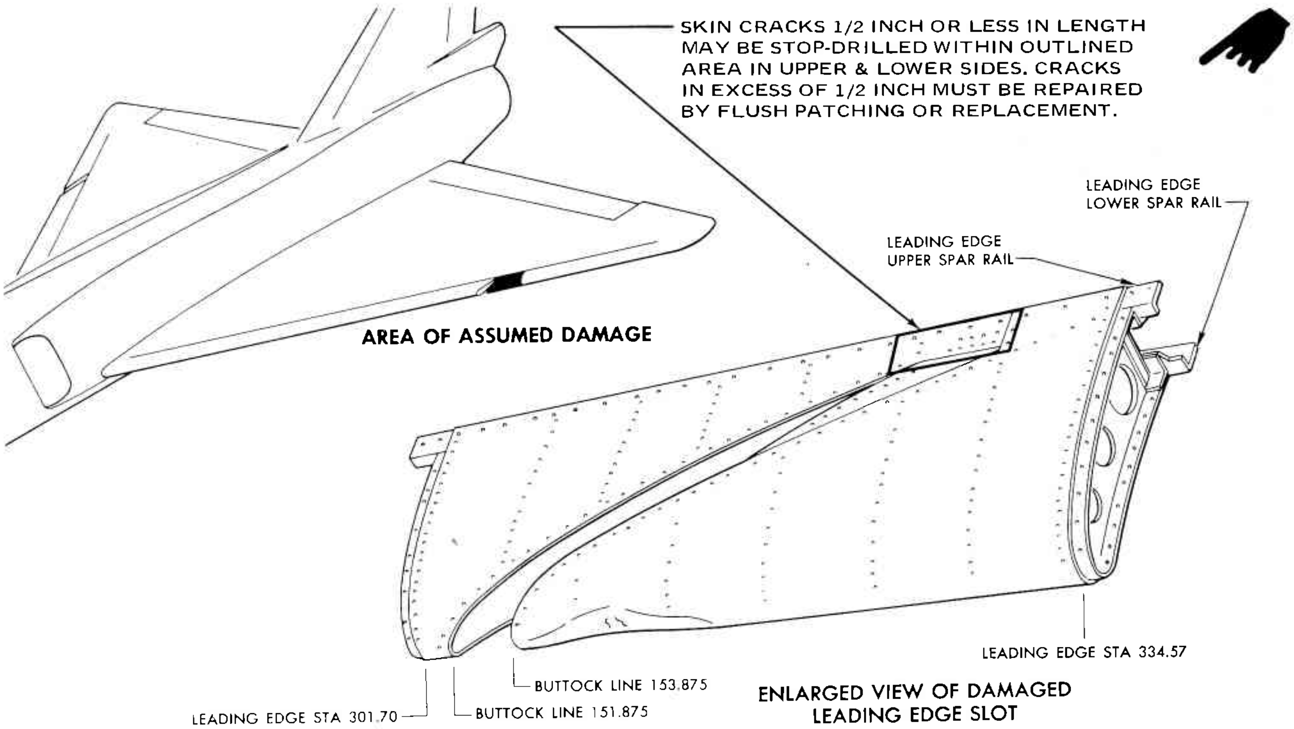
NOTE

IF DAMAGED WING LEADING EDGE SECTION IS REPARABLE, MAKE REPAIR ACCORDING TO PROCEDURES OUTLINED IN FIGURE 2-36.

u. Perform a fuel tank leak test according to procedures outlined in this section.
v. Finish exterior surface of repair area according to procedures outlined in Section I.

06.03.231-2B

Figure 2–37. Repair of Wing Leading Edge Spar (Sheet 2 of 2)



LEADING EDGE SLOT REMOVAL PROCEDURE

a. Remove aerodynamic smoothing compound from recessed heads of wing leading edge attaching screws.
b. Remove screws attaching leading edge slot section to wing and adjacent portions of leading edge. Hold screws for reinstallation.
c. Remove leading edge slot section.

TOOL BUILDUP PROCEDURE

a. Build tool base from steel pipe to dimensions shown. Secure pipes together by welding.
b. Using the outside surface of doubler at each end of leading edge slot section as a guide, form end locating plates. Locating plates are to be made from 0.125 gage 301 one-quarter hard steel.

NOTE

EXTEND ENDS OF LOCATING PLATES PAST LEADING EDGE SPAR RAILS TO ALLOW FOR WORKING SPACE.

c. Using existing holes in doublers as a guide, drill a minimum of three holes through each side of end locating plates.

CAUTION

USE AN UNDERSIZED DRILL TO PICK UP HOLES THROUGH NUTPLATES TO PREVENT DAMAGE TO NUTPLATE THREADS.

d. Secure end locating plates to doublers with AN526-1032 screws.
e. Attach an angle plate to each end of locating plate as shown and hold in place with clamps.
f. Drill holes through the locating brake and angle plate as shown. Use a No. 12 (0.189) drill.
g. Secure angle plates to locating plates with AN3-5 bolts and AN365-1032 nuts.
h. Place leading edge slot assembly, with locating and angle plates attached on tool base, and secure in place by welding angle plates to base.
i. Weld two 0.125 gage 301 one-quarter hard steel brace plates across each end of locating plates as shown.
j. Weld a holding channel to outside surface of end locating plates along top center line as shown. Holding channels are to be made from 0.125 gage, 301 one-quarter hard steel.
k. Place a locating channel on top of end holding channels as shown and hold in place with clamps.
l. Attach the outboard end of locating channel to the outboard holding channel with tooling pin and wing nut as shown.
m. Attach the inboard end of locating channel to inboard channel by welding hinge halves to channels as shown.
n. Attach, by welding, a hinge to side of locating channel. Attach to locating channel on concave side of leading edge.
o. Using an undamaged leading edge slot section as a guide, form drill plates for all ribs outboard of and including outboard diagonal slot closing rib. Drill plates are to be made from 0.125 gage 301 one-quarter hard steel.
p. Attach drill plates to tool by welding as shown.
q. Remove leading edge slot section from tool.
r. Using a No. 20 (0.161) drill remove all rivets from outboard skin panel. Remove skin panel.
s. Fabricate rib spacers from 0.050 gage soft aluminum alloy.
t. Form spacers to match contour of exposed ribs.
u. Install leading edge section in tool and place spacers between ribs and drill plates.
v. Using existing holes in undamaged ribs as a guide, drill holes in drill plates.

NOTE

HOLES FOR DAMAGED SECTION OF LEADING EDGE MUST BE LAID OUT BY PHYSICAL MEASUREMENT AND MUST MATCH ORIGINAL HOLE PATTERN.

w. Remove leading edge from tool.

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Figure 2-38. Wing Leading Edge Slot Repair (Sheet 1 of 4)

DAMAGE REMOVAL PROCEDURE

a. Remove three bolts and three nuts which attach outboard diagonal slot closing rib to slot wedge.
b. Using a No. 20 (0.161) drill, remove rivets attaching plug to slot closing rib. Remove plug and hold for reinstallation if not damaged.
c. Using a No. 20 (0.161) drill, remove rivets attaching damaged vertical ribs to attaching clips. Remove damaged ribs.
d. Using a No. 20 (0.161) drill, remove rivets attaching rib clips to diagonal rib. Remove ribs and clips and hold for reinstallation.
e. Using a 3/16" (0.187) drill, remove rivets attaching diagonal rib to spar clips. Remove diagonal rib.

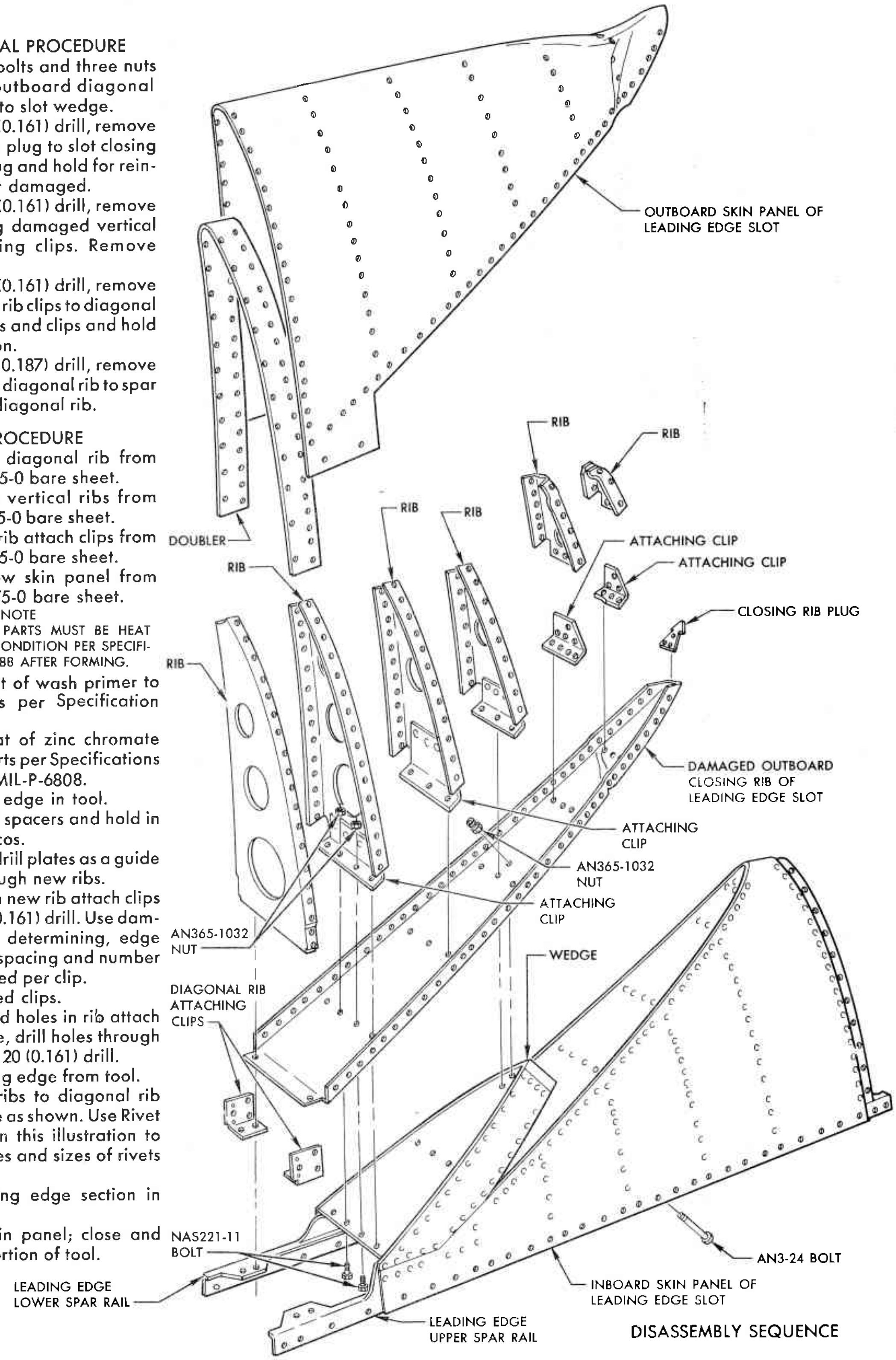
INSTALLATION PROCEDURE

a. Fabricate new diagonal rib from 0.063 gage 7075-0 bare sheet.
b. Fabricate new vertical ribs from 0.050 gage 7075-0 bare sheet.
c. Fabricate new rib attach clips from 0.063 gage 7075-0 bare sheet.
d. Fabricate a new skin panel from 0.050 gage 7075-0 bare sheet.

NOTE

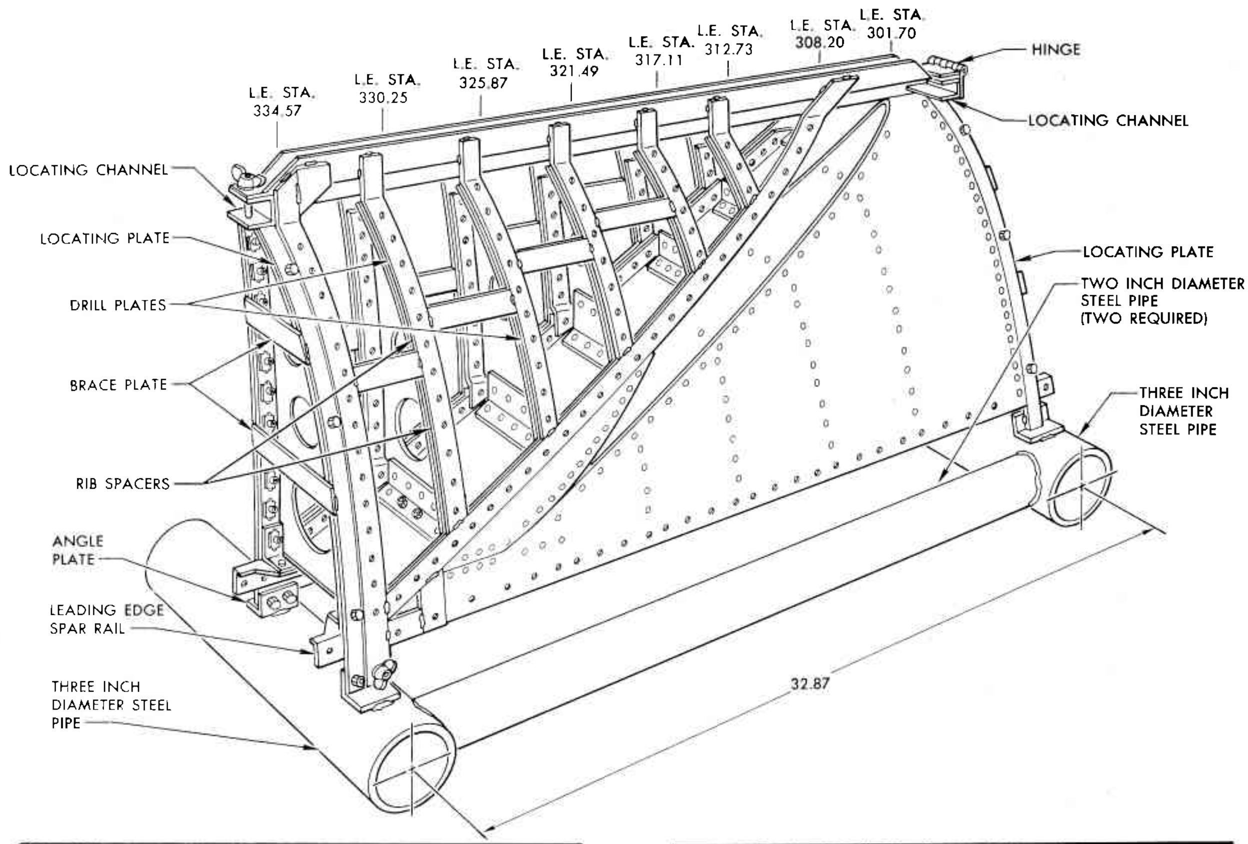
ALL FABRICATED PARTS MUST BE HEAT TREATED TO T6 CONDITION PER SPECIFICATION MIL-H-6088 AFTER FORMING.

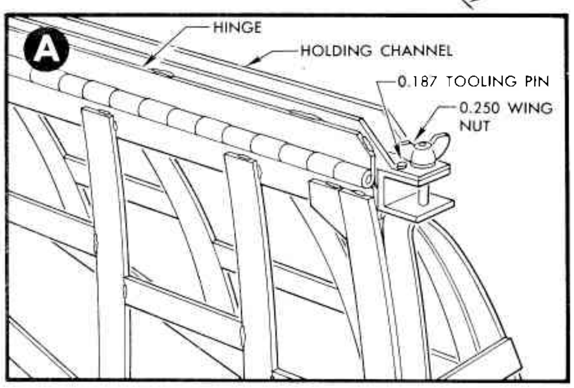
e. Apply one coat of wash primer to all repair parts per Specification MIL-C-8514.
f. Apply one coat of zinc chromate to all repair parts per Specifications MIL-P-8585 or MIL-P-6808.
g. Install leading edge in tool.
h. Install ribs and spacers and hold in place with clecos.
i. Using holes in drill plates as a guide drill holes through new ribs.
j. Predrill holes in new rib attach clips with a No. 20 (0.161) drill. Use damaged clips for determining, edge distance, hole spacing and number of holes required per clip.
k. Install predrilled clips.
l. Using predrilled holes in rib attach clips as a guide, drill holes through ribs. Use a No. 20 (0.161) drill.
m. Remove leading edge from tool.
n. Rivet vertical ribs to diagonal rib and slot wedge as shown. Use Rivet Table shown in this illustration to determine types and sizes of rivets to be used.
o. Reinstall leading edge section in tool.
p. Install new skin panel; close and lock hinged portion of tool.

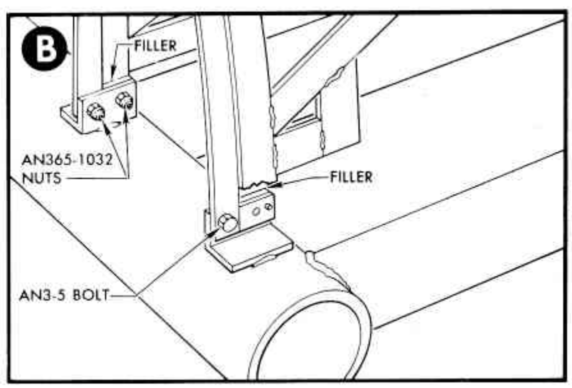


06.03.245-2A

Figure 2-38. Wing Leading Edge Slot Repair (Sheet 2 of 4)







q. Using holes in drill plates as a guide, drill holes through new skin panel.

CAUTION

USE A NO. 39 (0.099) DRILL IN PRIMARY DRILLING OF SKIN PANEL. TO ASSURE ALIGNMENT BETWEEN HOLES IN DRILL PLATE AND HOLES IN RIBS, REAM HOLES WITH A NO. 20 (0.161) DRILL AFTER ALIGNMENT IS ASSURED.

r. Remove leading edge section from tool.

s. Disassemble new skin panel and remove all burrs from leading edge.

t. Countersink all holes in exterior surface of new skin 100 degree X 0.286 ± 0.004.

u. Install plug in leading edge of diagonal rib and hold in place with clamps.

v. Using existing holes in plug as a guide, drill holes through diagonal rib. Use a No. 20 (0.161) drill.

w. Remove plug and remove all burrs.

x. Reinstall plug and rivet in place. Refer to Rivet Table shown in this illustration for type and size of rivets.

y. Reinstall new skin panel and hold in place with clecos.

z. Rivet skin panel in place. Refer to Rivet Table shown in this illustration for types and sizes of rivets to be used.

aa. Fill all cracks between slot closing rib and new skin with MIL-S-38228 aerodynamic smoothing compound.

ab. Paint the new portion of leading edge as required according to applicable paint schedule given in Section I.

ac. Install leading edge slot section to wing.

ad. Reinstall screws which were removed in "b" of Leading Edge Slot Removal Procedure.

ae. Fill all recessed heads of screws with MIL-S-38228.

06.03.245-3B

Figure 2-38. Wing Leading Edge Slot Repair (Sheet 3 of 4)

RIVET TABLE		
RIVETS THROUGH-	TYPE OF RIVET	DIAMETER OF RIVET
SKIN AND RIB NO'S 1, 2, 3, 4, 5, AND 6	AN426AD	5/32
RIB NO'S 1 AND 7 AND SPAR RAIL ATTACH CLIPS	AN470AD	5/32
SKIN AND RIB NO. 7	AN426DD	5/32
END HOLES IN RIB ATTACH CLIP NO. 2 AND RIB NO. 7	AN470DD	3/16
END HOLES IN RIB ATTACH CLIP NO. 3 AND RIB NO. 7	AN470DD	3/16
MIDDLE HOLE IN RIB ATTACH CLIP NO. 3 AND RIB NO. 7	Q4310	3/16
END HOLES IN RIB ATTACH CLIP NO. 4 AND RIB NO. 7	AN426DD	3/16
MIDDLE HOLE IN RIB ATTACH CLIP NO. 4 AND RIB NO. 7	Q4310	3/16
RIB ATTACH CLIP NO'S 5 AND 6 AND RIB NO. 7	AN426DD	3/16
RIB ATTACH CLIP NO'S 5 AND 6 AND RIB NO. 5 AND 6	AN470DD	3/16
PLUG AND RIB NO. 7	AN426AD	1/8

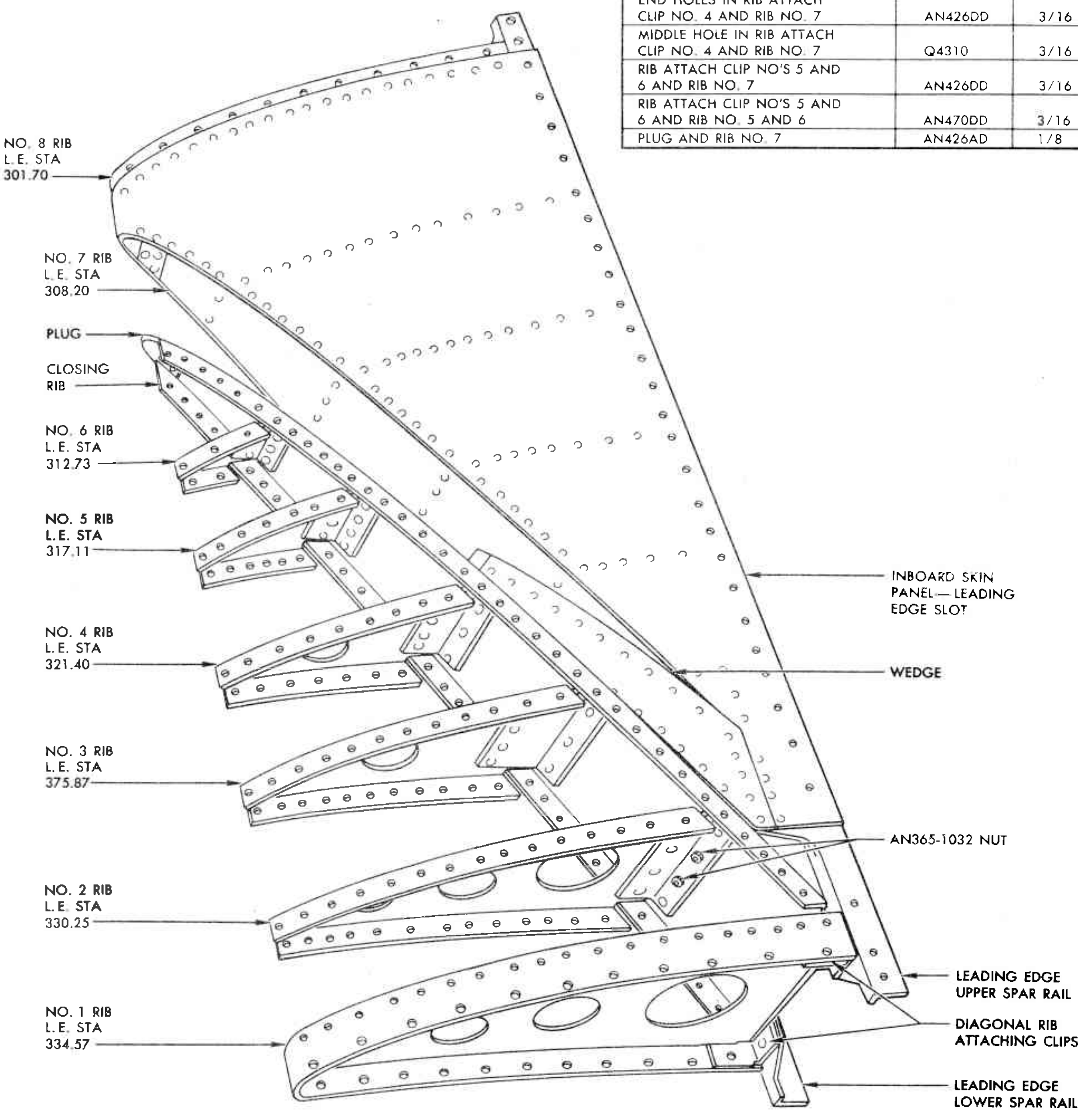
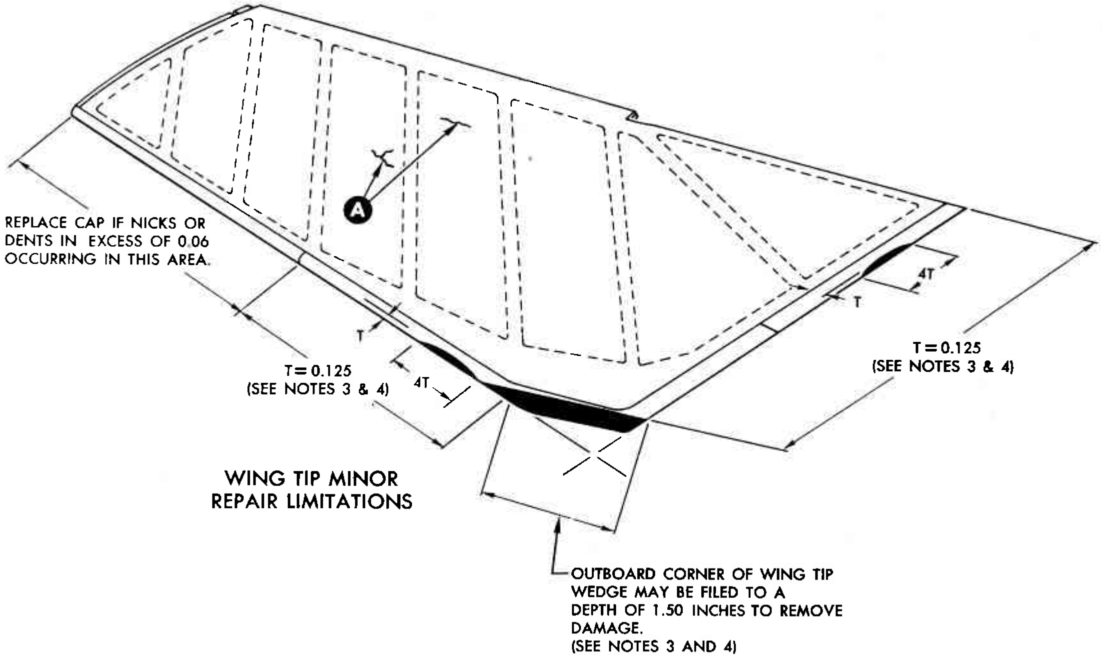
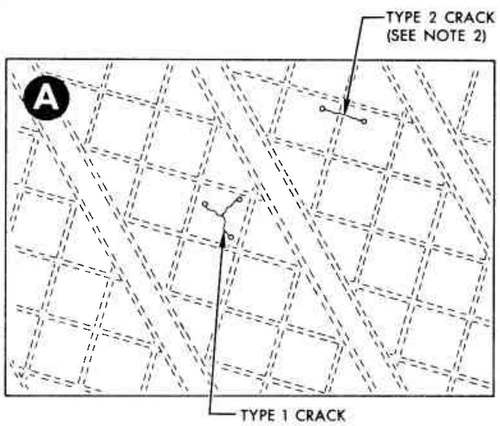


Figure 2-38. Wing Leading Edge Slot Repair (Sheet 4 of 4)



WING TIP MINOR REPAIR LIMITATIONS



REPAIR OF CRACKED WING TIP SKINS—TYPE 1

a. Remove all paint and primer from cracked area of wing tip skin.
b. Perform a fluorescent penetrant inspection to determine length of crack.
c. Stop drill crack at each end of crack with a No. 30 (0.1285) drill.
d. Rout out crack with a 0.128 router bit.
e. Fill routed area with MIL-S-38228 aerodynamic smoothing compound.
f. Apply primer and paint to repair area according to directions given in Section I.

NOTES:
1. DAMAGE IN EXCESS OF DIMENSIONS SHOWN WILL REQUIRE REPLACEMENT OF DAMAGED COMPONENT.
2. CRACKS ACROSS INTEGRAL WEBS OF WING TIP SKINS WILL REQUIRE REPLACEMENT OF SKIN.
3. WEDGES MAY BE FILED AS INDICATED TO REMOVE MINOR DAMAGE.
4. MAXIMUM ALLOWABLE MATERIAL TO BE REMOVED FROM LEADING EDGE OR TRAILING EDGE OF WING TIP WEDGES IS NOT TO EXCEED 2.50 SQUARE INCHES.
5. TOTAL AREA OF WING TIP WEDGES TO BE REMOVED BY FILING NOT TO EXCEED 15.00 SQUARE INCHES.

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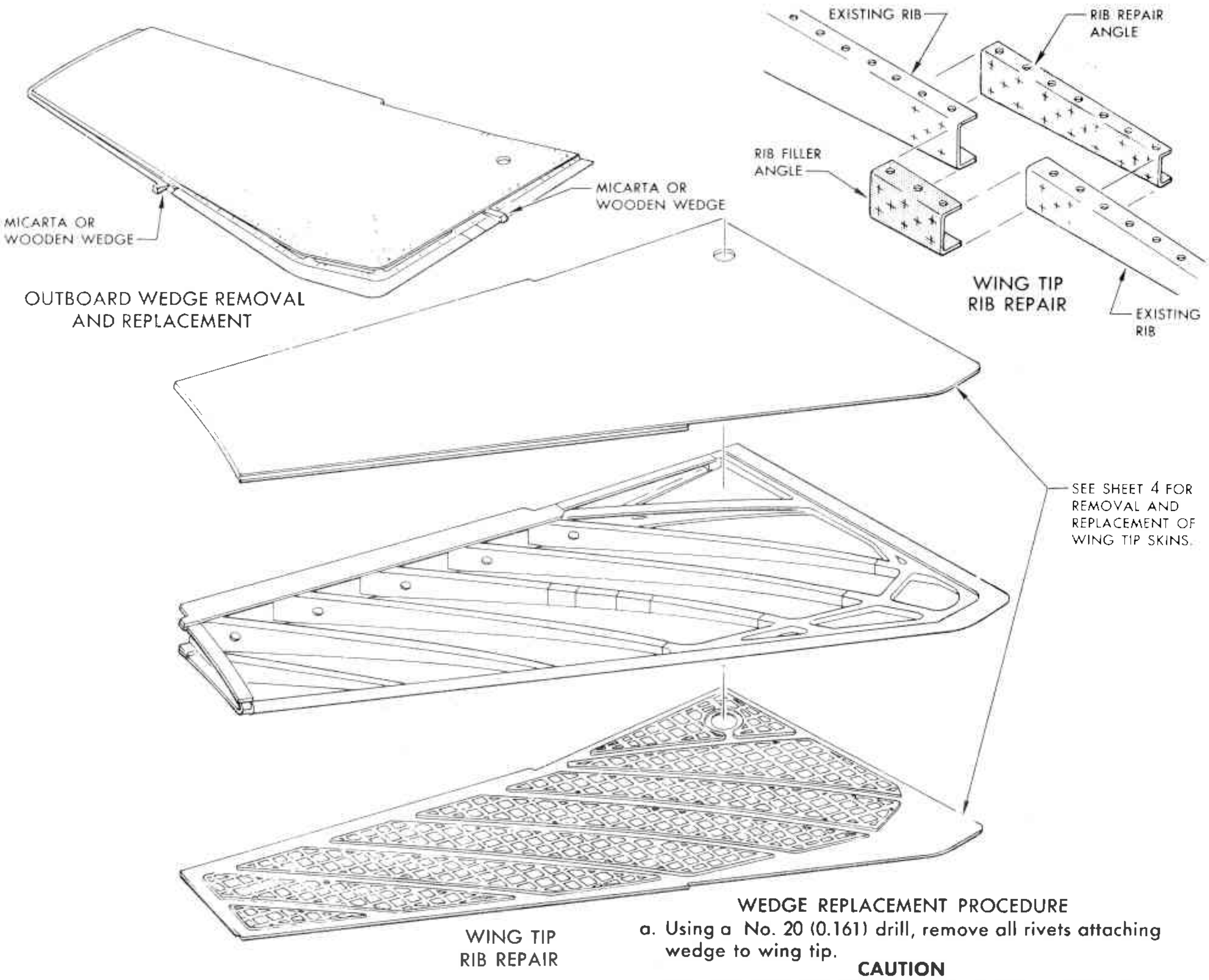
Figure 2-39. Wing Tip Repairs — Limitations and Procedures (Sheet 1 of 6)



WING TIP RIB REPAIR PROCEDURE

a. Remove either the upper or lower wing tip skin.
b. Cutout the damaged portion of wing tip rib.
c. Fabricate rib repair channels from 7075-0 bare sheet as shown. Repair angles must be heat treated to the -T6 condition per Specification MIL-H-6088 after forming.
d. Install rib repair channel and hold in place with clamps.
e. Install rib filler channel and hold in place with clamps.
f. Using predrilled holes in vertical flange of repair parts as a guide, perform drilling operation. Use a No. 30 (0.128) drill.
g. Ream holes drilled in step "f" with a No. 20 (0.161) drill.
h. Remove repair parts and remove all burrs from repair parts and wing tip.
i. Apply one coat of wash primer to repair parts per Specification MIL-C-8514.
j. Apply four coats of Zinc Chromate Primer to repair parts per Specification MIL-P-8585 or MIL-P-6808.
k. Reinstall rib repair parts and hold in place with clecos.
l. Rivet rib repair parts in place with AN370AD-5 rivets.
m. Complete repair of rib by wing tip skin replacement procedure.

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WEDGE REPLACEMENT PROCEDURE

a. Using a No. 20 (0.161) drill, remove all rivets attaching wedge to wing tip.

CAUTION

DO NOT ATTEMPT TO DRILL COMPLETELY THROUGH WING TIP TO REMOVE RIVETS. DRILL OUT MANUFACTURED HEAD OF RIVET, THEN REMOVE REMAINDER OF RIVET WITH A PUNCH.

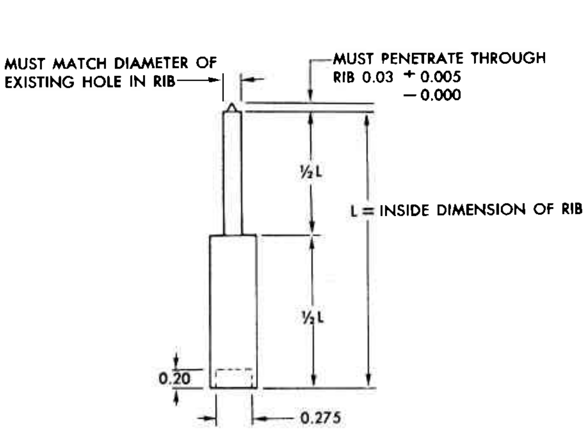
b. Fabricate two wedges as shown and insert between wing tip wedge and skin.
c. Remove wing tip wedge.
d. Install new wing tip wedge.
e. Remove fabricated wedges.
f. Align new wing tip wedge and hold in place with clamps.
g. Using existing holes in wing tip skins as a guide, drill holes through new wing tip wedge.

NOTE

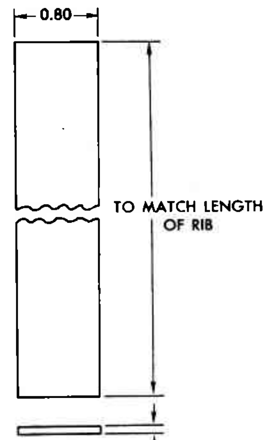
TO MAINTAIN ALIGNMENT BETWEEN EXISTING HOLES IN UPPER AND LOWER WING TIP SKINS; DRILL HOLES THROUGH WEDGE WITH A NO. 39 (0.099) DRILL FIRST, THEN REAM HOLES TO PROPER SIZE WITH A NO. 20 (0.161) DRILL.

h. Remove wing tip wedge and remove all burrs from wedge and wing tip.
i. Reinstall wing tip wedge and align by placing AN426B-5 rivets through holes drilled in step "g".
j. Secure rivets in place by standard riveting methods.
k. Mill shop formed rivet heads flush with wing tip skin.
l. Apply paint to wing tip according to applicable schedule given in Section I.

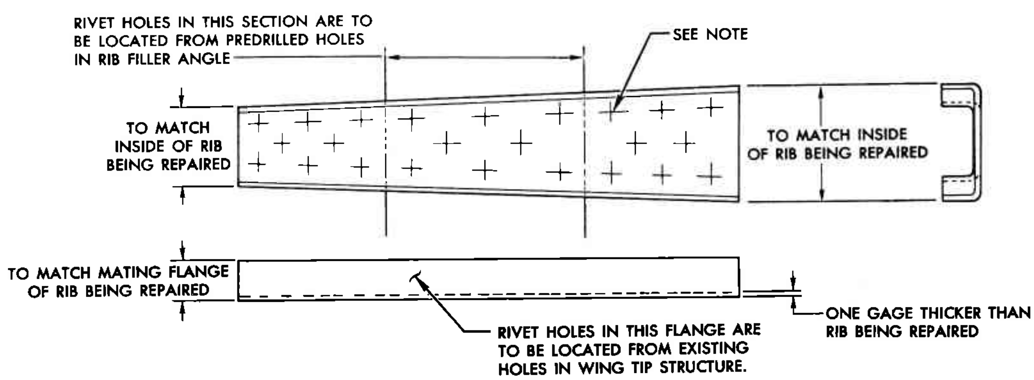
Figure 2-39. Wing Tip Repairs — Limitations and Procedures (Sheet 2 of 6)



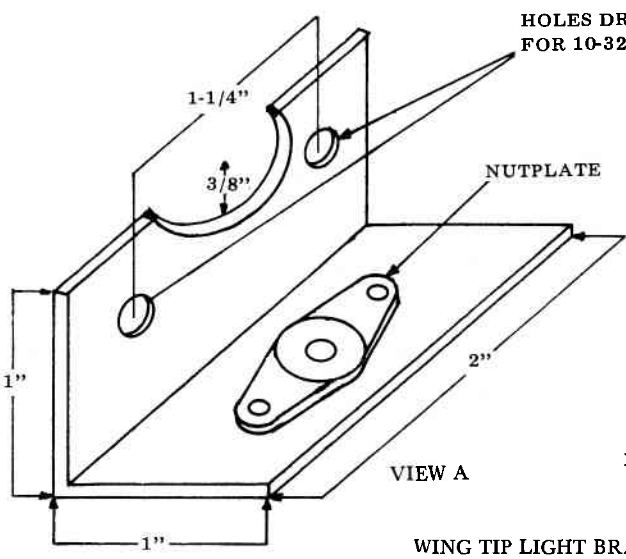
HOLE LOCATING PUNCH



DRILL TEMPLATE



RIB REPAIR ANGLE



VIEW A

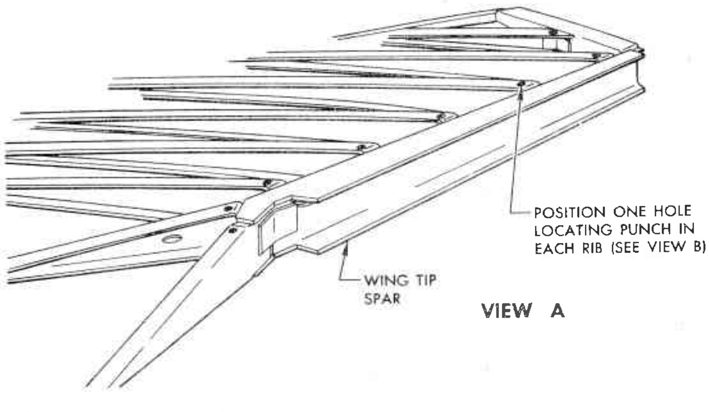
REPAIR PROCEDURE

1. REMOVE THE BROKEN OR STRIPPED LUGS THAT THE LIGHT SOCKET IS ATTACHED TO.
2. FABRICATE A BRACKET FROM A SECTION OF 1-INCH BY 1-INCH EXTRUDED ANGLE 2 INCHES LONG APPROXIMATELY .062 THICK.
3. MAKE A CUTOUT IN ONE LEG OF THE ANGLE TO ACCOMMODATE SOCKET AND DRILL 2 HOLES AS SHOWN IN VIEW A FOR ATTACHING SOCKET.
4. DRILL AND INSTALL A 10-32 NUTPLATE ON OTHER LEG OF ANGLE.
5. ATTACH TO WING TIP BY REMOVING ONE EXISTING RIVET FROM THE FORWARD EDGE OF ORIGINAL RETAINER. DRILL AND COUNTERSINK RIVET HOLE FOR A 10-32 SCREW. ATTACH TO WING TIP WITH A 10-32 SCREW.

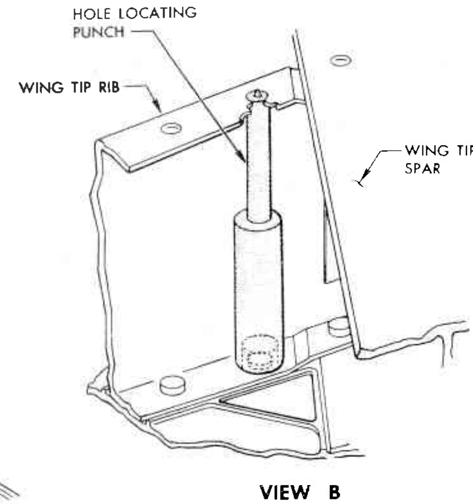
NOTE:
ATTACH LIGHT SOCKET TO BRACKET PRIOR TO INSTALLING ON WING TIP.

WING TIP LIGHT BRACKET REPAIR

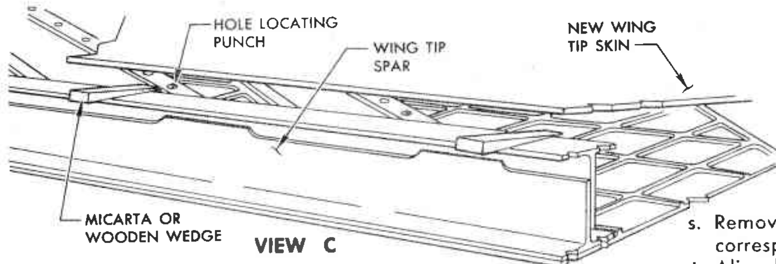
Figure 2-39. Wing Tip Repairs — Limitations and Procedures (Sheet 3 of 6)



VIEW A



VIEW B



VIEW C

WING TIP SKIN REPLACEMENT PROCEDURE

a. Remove the screws attaching the wing tip to the wing and hold for reinstallation.
b. Remove the wing tip. Disconnect position light wiring as wing tip is removed.
c. Using a No. 20 (0.161) drill, remove all rivets from damaged wing tip skin.

NOTE

DO NOT ATTEMPT TO DRILL COMPLETELY THROUGH WING TIP TO REMOVE RIVETS. DRILL OUT MANUFACTURED RIVET HEADS, THEN REMOVE REMAINDER OF RIVET WITH A PUNCH.

d. Remove damaged wing tip skin.
e. Make hole locating punches as shown from 0.375-inch diameter drill rod.
f. Make four wedges from micarta or wood.
g. Install hole locating punches in ribs as shown in views A and B.
h. Place wedges along spar as shown in view C.
i. Install new wing tip skin; align skin and hold in place place with clamps.
j. Using existing holes along outboard edge of wing tip as a guide, drill holes through new wing tip skin. Use a No. 39 (0.099) drill.
k. Remove wedges from between new skin and wing tip spar.

NOTE

REMOVE WEDGES CAREFULLY TO PREVENT NEW SKIN FROM BECOMING MISALIGNED.

l. Hold new skin firmly in place along wing tip spar with clamps.
m. Using a leather or plastic mallet, tap the new skin lightly over each of the hole locating punches.
n. Remove all clamps and remove the new skin.
o. Remove the hole locating punches from ribs.
p. Fabricate drill template strips to locate remainder of rib holes from soft aluminum as shown.
q. Place drill template strips in grooves of wing tip skin inner surface.
r. Using holes in new skin as a guide, drill holes through drill template strips. Use a No. 39 (0.099) drill.
s. Remove drill template strips from skin and place over corresponding ribs in wing tip.
t. Align holes in drill template strips over corresponding holes in wing tip ribs and hold securely in place with clamps.
u. Using existing holes in wing tip ribs and spar as a guide, drill holes in drill template strips. Use a No. 39 (0.099) drill.
v. Remove drill template strips from ribs and spar.
w. Place drill template strips in corresponding grooves of wing tip skin.

NOTE

ALIGN DRILL TEMPLATE STRIPS IN PLACE BY MATCHING HOLES IN SKIN WITH HOLES IN DRILL TEMPLATE STRIPS.

x. Using holes in drill template strips as a guide, drill holes through skin. Use a No. 39 (0.099) drill.
y. Remove drill template strips from skin.
z. Reinstall new skin on wing tip and hold securely in place with clamps.
aa. Ream holes in new skin with a No. 30 (0.128) drill.

CAUTION

CHECK EACH HOLE FOR ALIGNMENT BEFORE REAMING.

ab. Enlarge holes in skin to final size with a No. 20 (0.161) drill.
ac. Remove new skin and remove all burrs from skin and wing tip.
ad. Machine countersink all holes in skin 78 degrees.
ae. Reinstall skin and align by placing rivets through holes. Use DL rivets through blind area and CY rivets in non-blind area.

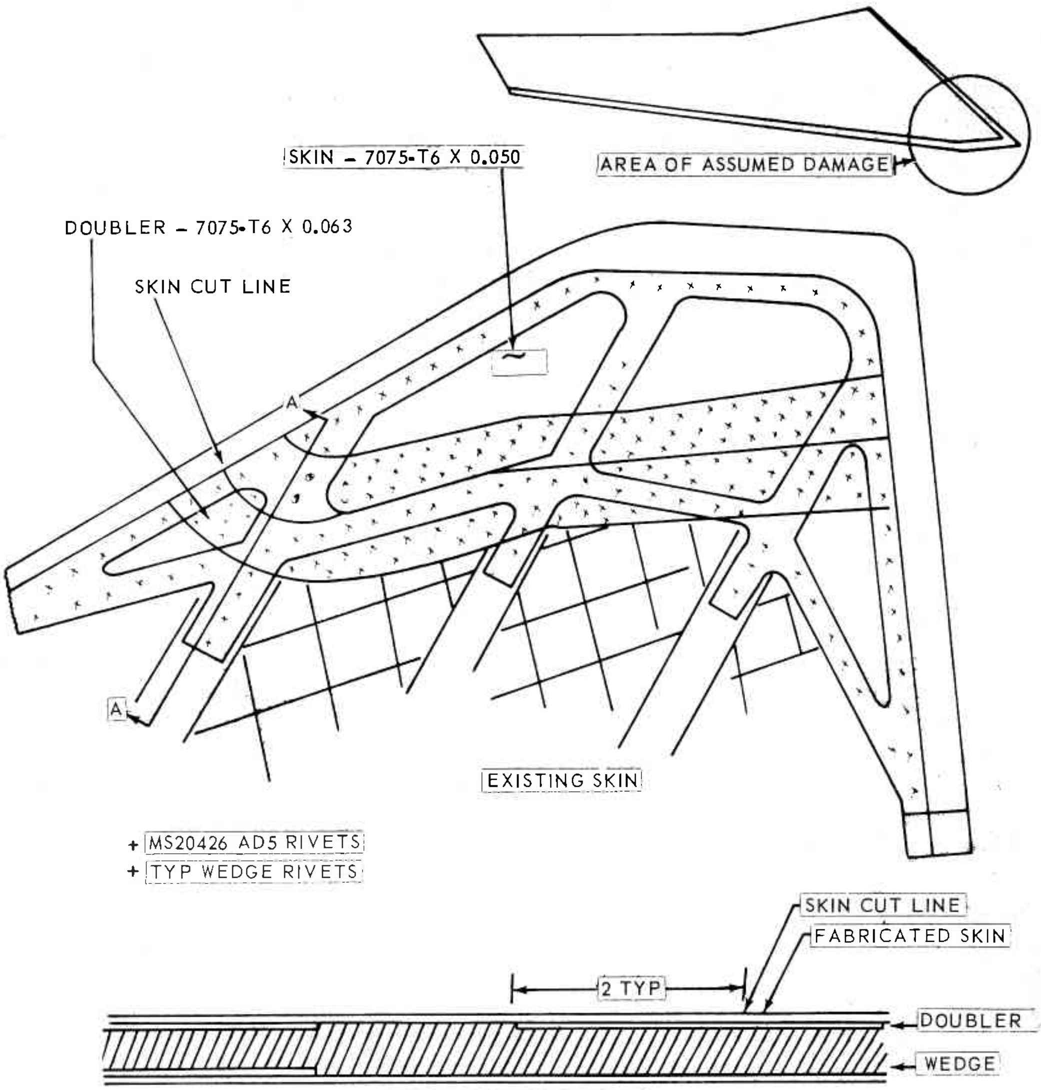
NOTE

REFER TO TABLE 1-XXIII FOR RIVET BASIC CODE.

af. Secure rivets in place by standard riveting methods.
ag. Apply paint to wing tip according to applicable paint schedule given in Section I.
ah. Reconnect position light wires.
ai. Reinstall wing tip with screws removed in step "a."
aj. Fill screw heads and gaps between wing tip and wing with aerodynamic smoothing compound. Refer to Table 11-VII for application of aerodynamic smoothing compound.

06.03.247-4B

Figure 2-39. Wing Tip Repairs — Limitations and Procedures (Sheet 4 of 6)



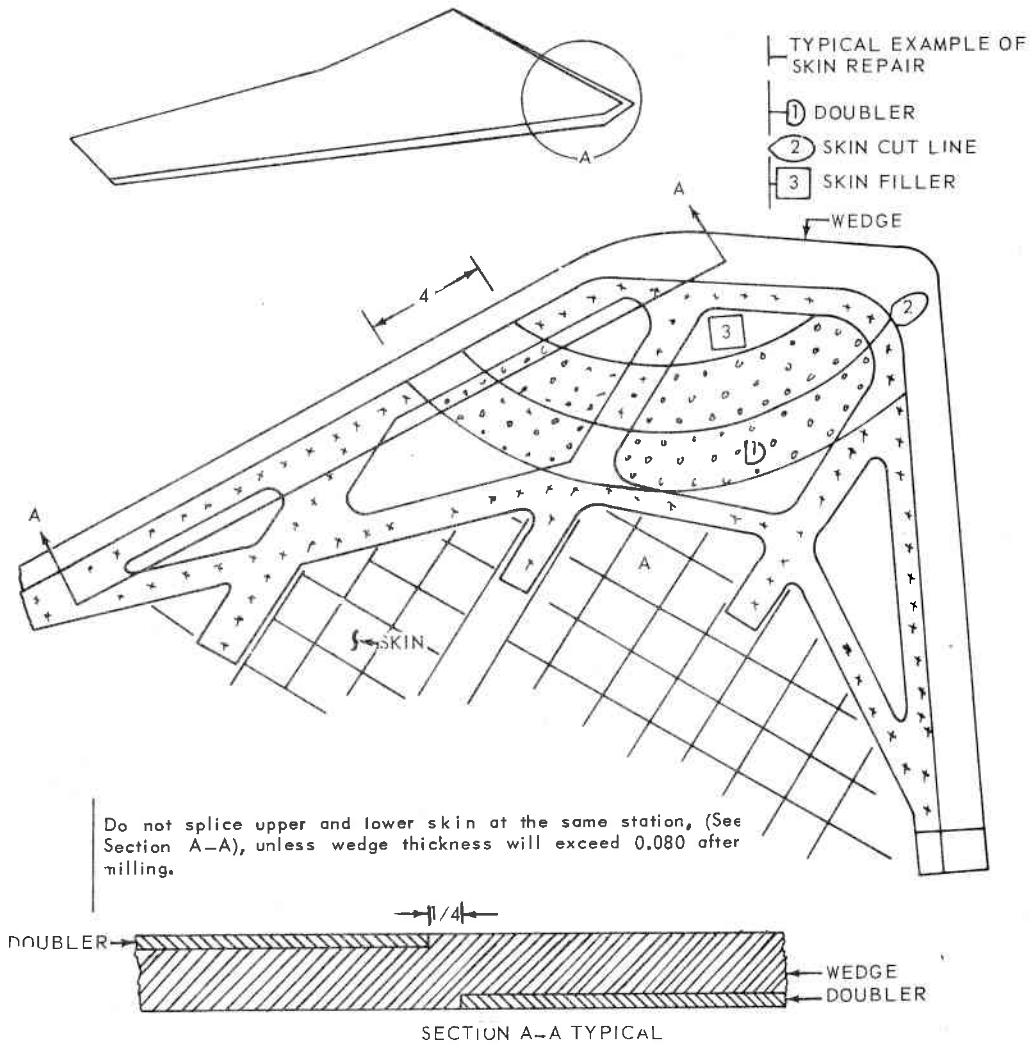
REPAIR PROCEDURE

a. Remove rivets in wedge and -10 lower skin as required to facilitate repair. Reference (Sheet 2 of 6) and (Sheet 4 of 6).
b. Cut broken and torn skin to approximate shape as shown.
c. Remove grid portion of -8 skin as required to maintain a flush surface under doubler. Apply protective coating per MIL-C-5541, wash primer MIL-C-8514, and primer MIL-P-8585, to exposed metal.
d. Fabricate doubler from 7075-T6 x 0.064 aluminum alloy sheet to fit contours, and as shown.
e. Fabricate skin filler from 7075-T6 x 0.051 aluminum alloy. Trim to fit contours of wedge and skin cut line.
f. Insert a new wedge in place using clecos. Temporarily install -10 skin, reference (Sheet 4 of 6). Drill holes in wedge from existing holes in skin. Drill holes in fabricated skin rod doubler as shown in figure 1-32 and table 1-11.
g. Remove fabricated skins, doublers and wedge. Temporarily locate doublers on wedge. Mark doubler location. Mill wedge surface 0.064 deep to recess doubler flush with wedge contour. Apply MIL-M-3171, Type 1, prepaint surface treatment to milled area followed by Primer to all mating surfaces. (Reference T.O. 1F-106A-23).
h. Countersink holes in skins. Install doubler to skin using MS20426AD5 rivets.
i. Reinstall skin and wedge.
j. Prime and paint wing tip surfaces as directed in T.O. 1F-106A-23.

NOTE

ALL MACHINED SURFACES TO BE 125 MIN. PER MIL-STD-10.

Figure 2-39. Wing Tip Repairs — Limitations and Procedures (Sheet 5 of 6)



REPAIR PROCEDURE

a. Remove rivets in wedge and skin as required to facilitate repair. Reference (Sheet 2 of 6) and (Sheet 4 of 6).
b. Cut broken and torn skins not to exceed limits as shown.
c. Grid portion of skin may be removed to provide a smooth skin surface. Apply protective coating per MIL-C-5541, wash primer MIL-C-8514, and primer MIL-P-8585 to exposed skin.
d. Fabricate reinforcing doubler and skin filler from 7075-T6, same guage as thickness of milled skin at cut line.
e. Insert a new wedge in place. Drill holes in wedge from existing holes in skin. Drill holes in fabricated doubler, filler and skin as per figure 1-32, and table 1-IV. (Rivet size and pattern in wedge shown on Drawing 8-18411.
f. Remove fabricated skins, doublers and wedge. Temporarily locate doubler on wedge. Mark doubler location. Mill wedge surfaces to recess doublers flush with wedge contour. Apply MIL-M-3171, Type 1, prepaint surface treatment to milled areas followed by primer to all mating surfaces, as per T.O. 1F-106A-23.
g. Countersink holes in skins. Install doublers and filler using MS-20426AD5 rivets. FA200 Jo-Bolts may be used as required. One row Jo-Bolts equal to two (2) rows of MS-20426-AD5 rivets. Maintain 2D edge distance.
h. Reinstall skins and wedge.
i. Prime and paint wing tip surfaces as directed in T.O. 1F-106A-23.

NOTE

ALL MACHINE FINISHES TO BE 125 MIN. PER MIL-STD-10.

Figure 2-39. Wing Tip Repairs — Limitations and Procedures (Sheet 6 of 6)

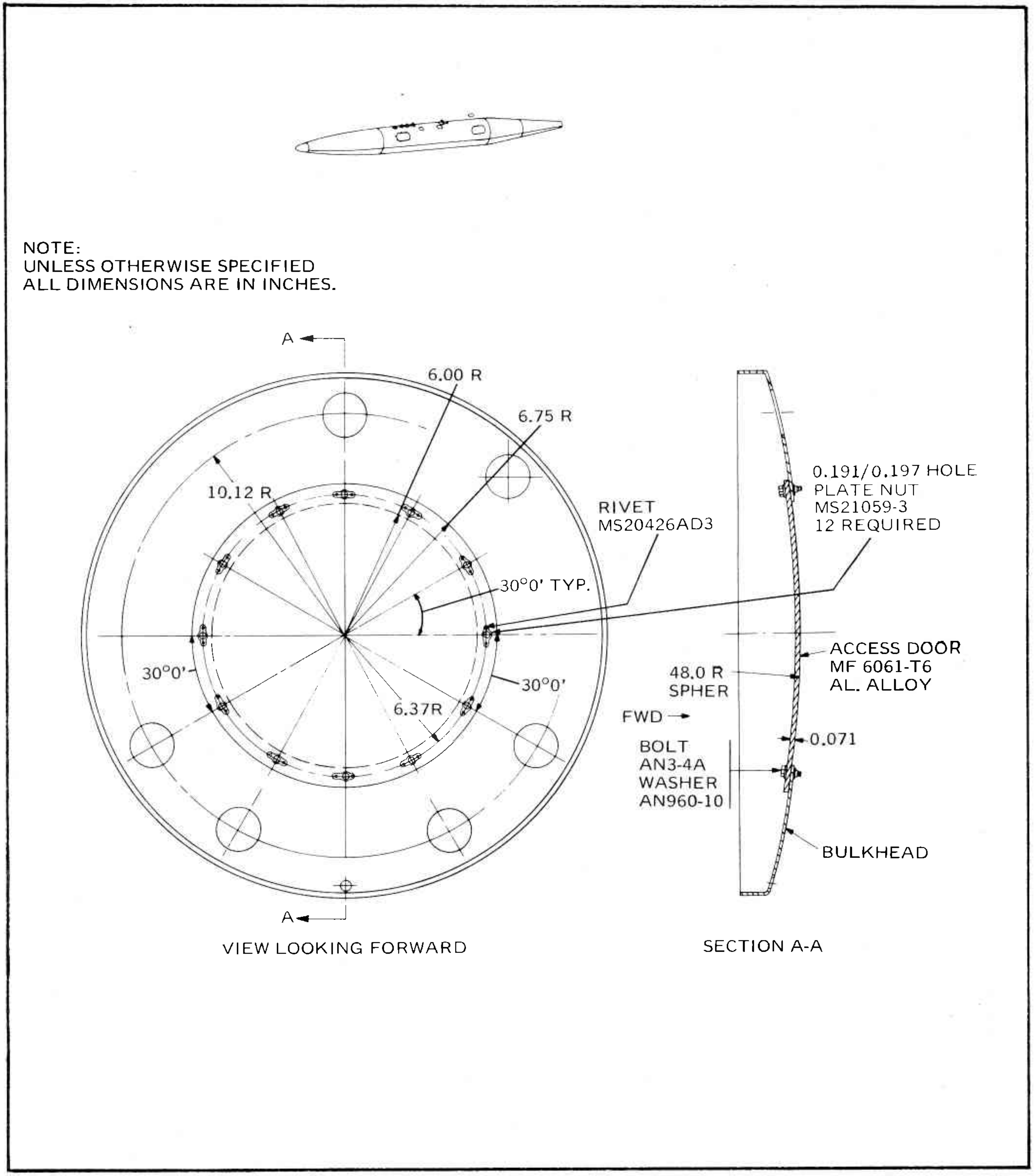
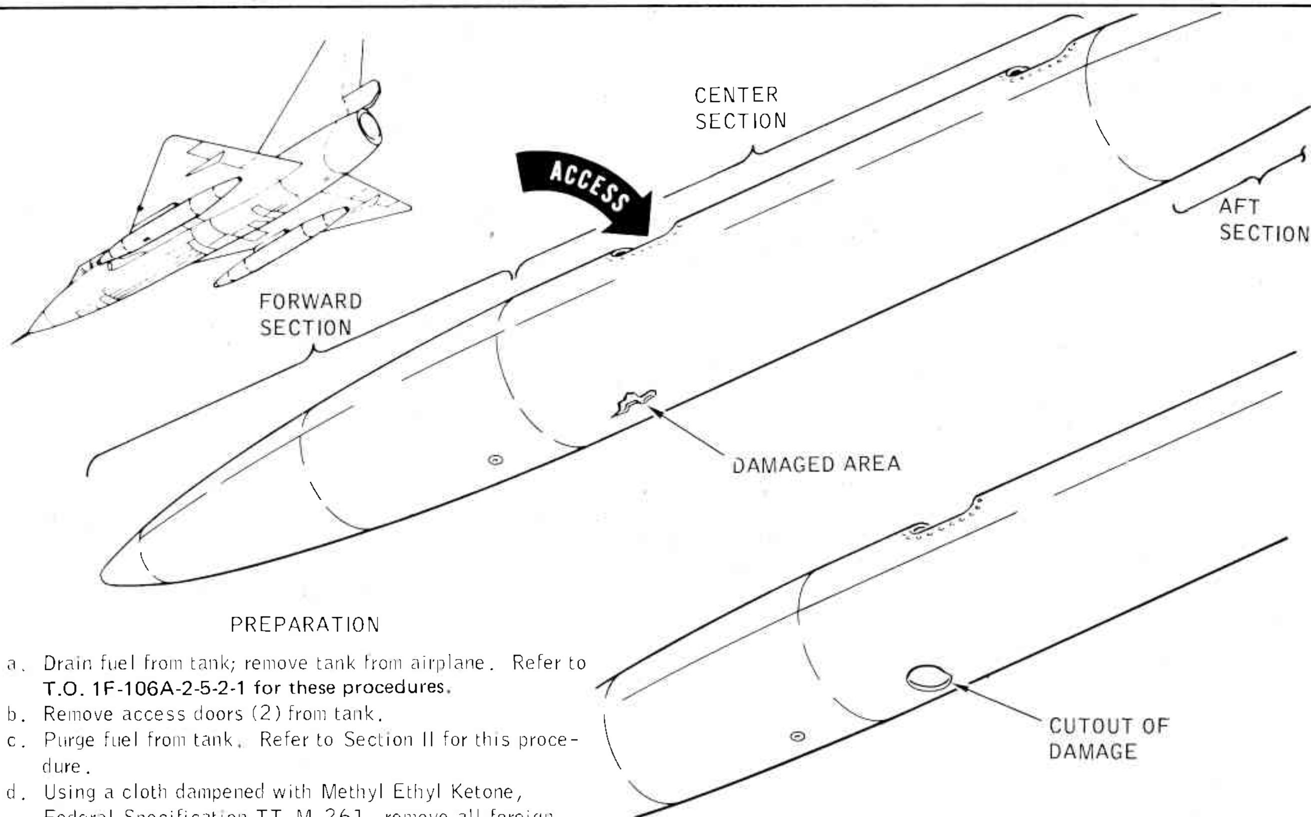


Figure 2-40. Nose Section Access for Internal Repairs.
Applicable after Incorporation of TCTO 1F-106-958.

All data on pages 2-79 and 2-80 deleted.



PREPARATION

a. Drain fuel from tank; remove tank from airplane. Refer to T.O. 1F-106A-2-5-2-1 for these procedures.
b. Remove access doors (2) from tank.
c. Purge fuel from tank. Refer to Section II for this procedure.
d. Using a cloth dampened with Methyl Ethyl Ketone, Federal Specification TT-M-261, remove all foreign material from damaged area.
e. Perform dye penetrant inspection to determine extent of damage. Refer to Section I for dye penetrant inspection procedures.

CENTER SECTION REPAIR

a. Cut out damaged material to leave a regular shape in existing skin.
b. Fabricate a doubler from 6061 T6 bare sheet, 0.090 inch thick. Doubler to be of sufficient size to allow 3 rivet diameters edge distance on outer row of rivets, 3/4 inch separation between inner and outer rows of rivets, and 3/4 inch edge distance on inner row of rivets.
c. Fabricate a flush skin patch from 6061 T6 bare sheet, 0.090 inch thick.
d. Predrill doubler as shown using a No. 20 (0.161) drill.
e. Install doubler and hold in place with clamps.
f. Using predrilled holes in doubler as a guide, drill holes in tank skin using a No. 10 (0.193) drill.
g. Remove clamps and install clecos to hold doubler in place.
h. Position flush patch in place.
i. Using doubler predrilled holes as a guide, drill holes in flush patch using a No. 10 (0.193) drill.

NOTE

INSTALL CLECOS THROUGH DOUBLER AND PATCH AS HOLES ARE DRILLED TO MAINTAIN HOLE ALIGNMENT.

j. Remove patch and doubler; remove all burrs.
k. Reinstall doubler and patch using clecos.
l. Countersink all holes using a 100° countersink.
m. Remove patch and doubler. Thoroughly clean repair parts and tank skin using the following procedure:

CAUTION

ALL PERSONNEL HANDLING REPAIR PARTS DURING AND AFTER THE FOLLOWING TREATMENT MUST WEAR CLEAN WHITE GLOVES TO PREVENT CONTAMINATION OF FAYING SURFACES.

1. Perform initial cleaning with a clean, lint-free cheese cloth dampened with Methyl Ethyl Ketone, Federal Specification TT-M-261.
2. Finish cleaning with a clean, lint-free cheese cloth dampened with Aliphatic Naphtha, Federal Specification TT-N-95.
3. Apply Deoxidine to faying surfaces of repair parts and existing skin. Application may be made with a clean, lint-free cheese cloth or a nylon bristle brush.

NOTE

DO NOT ALLOW DEOXIDINE TO DRY DURING APPLICATION.

4. Apply Alodine 1200 to faying surfaces of repair parts and existing skin. Application may be made with a clean, lint-free cheese cloth or a nylon bristle brush.

NOTE

DO NOT ALLOW ALODINE 1200 TO DRY DURING APPLICATION.

5. Rinse faying surfaces of repair parts and existing skin with a clean, lint-free cheese cloth dampened with clean, room temperature tap water.

NOTE

USE SEVERAL PIECES OF CLEAN CHEESE CLOTH FOR RINSING.

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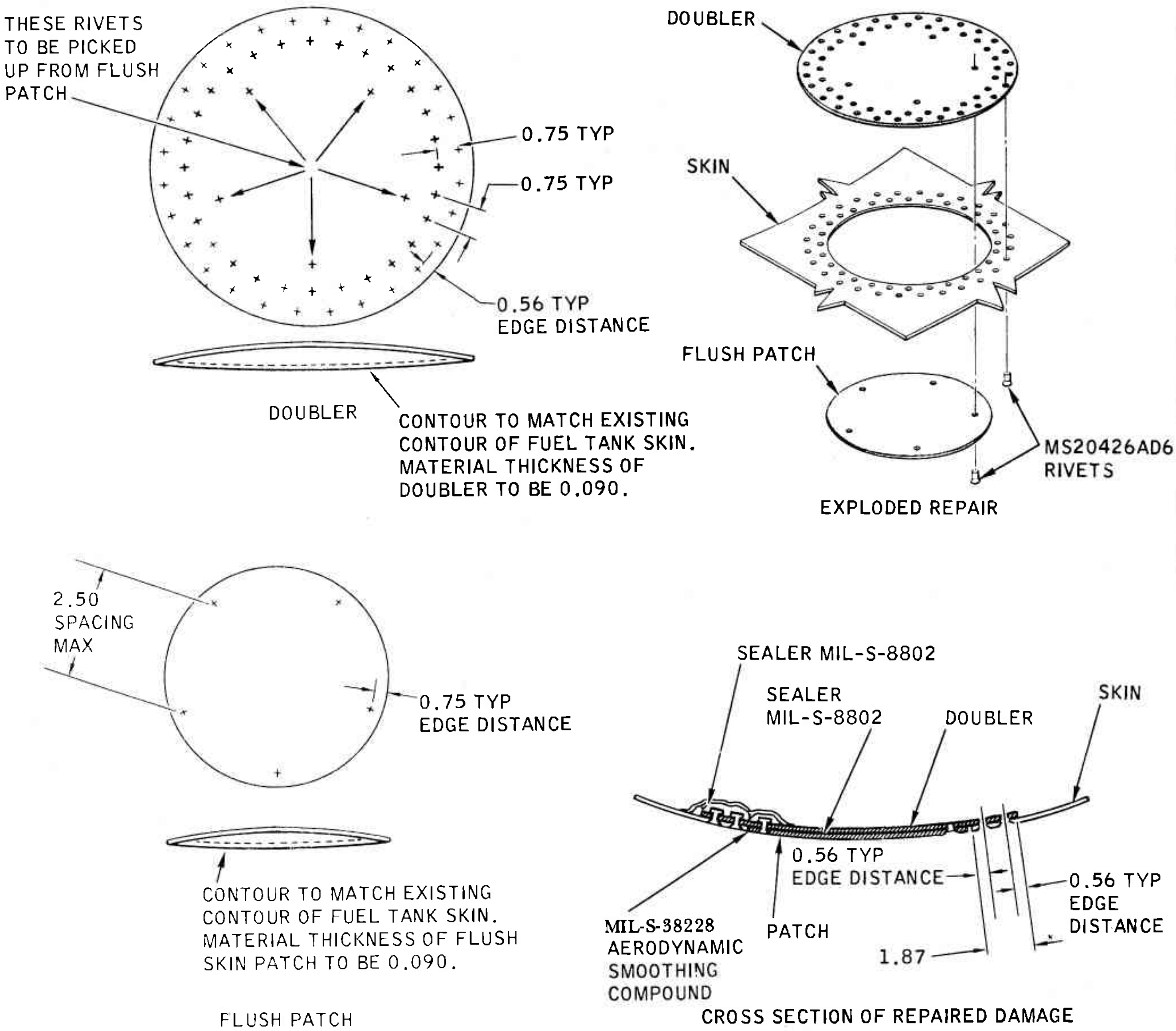
Figure 2-40A. External Fuel Tank Repairs (Sheet 1 of 3)
Applicable after incorporation of TCTO 1F-106-958.



n. Apply a coat of sealer, Specification MIL-S-8802, to faying surfaces of repair parts and existing skin. Refer to T.O. 1-1-3 for application procedures for sealer.

o. Install repair parts and hold in place with clecos. Rivet doubler to existing skin and flush patch to doubler with MS20426AD6 rivets.

NOTE

WIPE ALL MOISTURE FROM RIVET AND APPLY A COAT OF SEALER, SPECIFICATION MIL-S-8802 TO SHANK OF RIVET BEFORE EACH RIVET IS INSTALLED.

p. Apply a coat of sealer, Specification MIL-S-8802, to interior surface of repair as shown. Refer to T.O. 1-1-3 for application procedures for sealer.

q. Fill gap between flush patch and existing skin with MIL-S-38228.

FORWARD OR AFT SECTION REPAIR

a. Perform steps of preparation procedure.

b. Cut out damaged material. Make cutout enough of a rectangular or oval shape to permit insertion of doubler. Radius of cutout corners not to be less than two inches.

NOTE

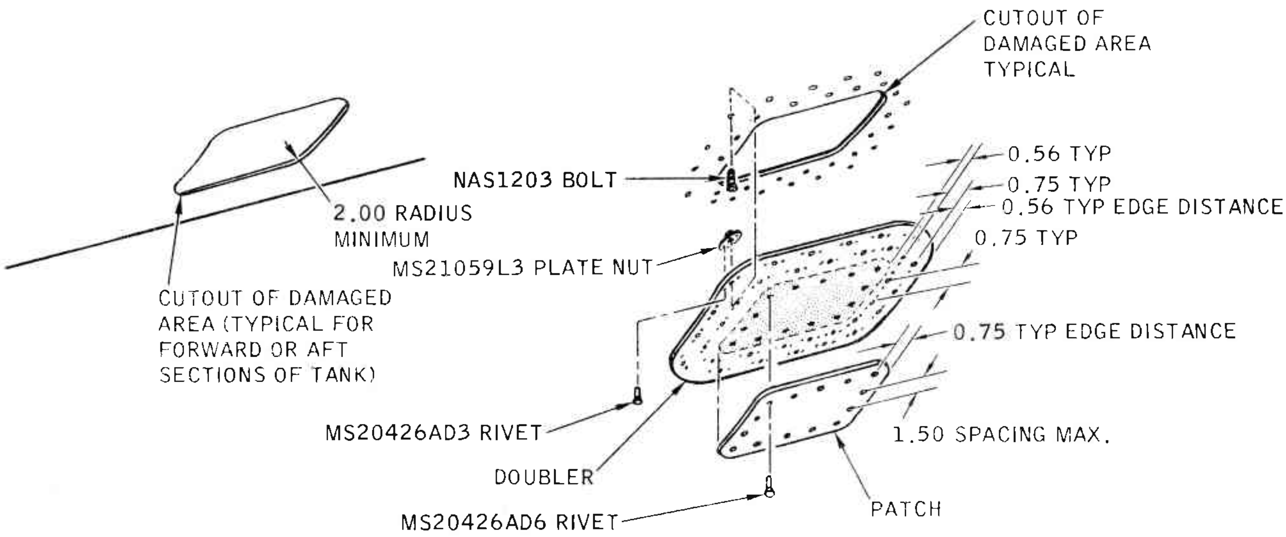
POSITION TANK SO THAT DAMAGE AREA IS AT THE LOW POINT. THIS IS TO KEEP CUTTINGS AND SHAVINGS CONCENTRATED NEAR THE CUT-OUT AND TO FACILITATE TANK CLEANING.

c. Fabricate doubler from 6061 T6 bare sheet, 0.090 inch thick. Doubler to be of sufficient size to allow 3 bolt diameters edge distance on outer row of bolts, 3/4 inch separation between inner and outer row of bolts, and 3/4 inch edge distance on inner row of bolts.

d. Fabricate a flush skin patch from 6061 T6 bare sheet, 0.090 inch thick.

Figure 2-40A. External Fuel Tank Repairs (Sheet 2 of 3)

Applicable after incorporation of TCTO 1F-106-958.



e. Predrill bolt hole pattern in tank using a No. 20 (0.161) drill.
f. Position doubler in tank and clamp in position.
g. Using tank predrilled holes as a guide, drill holes in doubler using a No. 10 (0.193) drill.

NOTE

INSTALL CLECOS THROUGH TANK AND DOUBLER AS HOLES ARE DRILLED TO MAINTAIN HOLE ALIGNMENT.

h. Using a soft lead pencil, lightly mark outline of tank cutout on doubler.
i. Remove doubler; remove burrs from tank and doubler.
j. Reinstall doubler using clecos.
k. Countersink holes using a 100° countersink; remove doubler.

NOTE

A SLIGHT PROTRUSION OF SCREW HEADS TO A MAXIMUM OF 0.002 INCH IS DESIRABLE.

l. Position patch on doubler; hold together with clamps.
m. Drill rivet pattern in patch and doubler using a No. 10 (0.193) drill.

NOTE

INSTALL CLECOS THROUGH DOUBLER AND PATCH AS HOLES ARE DRILLED TO MAINTAIN HOLE ALIGNMENT.

n. Remove clecos and clamps; remove all burrs.
o. Position patch on doubler; hold together with clecos.
p. Countersink rivet holes in patch using 100° countersink.
q. Remove clecos; separate patch and doubler.
r. Perform cleaning procedure in step m of center section repair procedure.
s. Apply a coat of sealer, Specification MIL-S-8802, on faying surfaces of doubler and patch. Refer to T.O. 1-1-3 for sealer application procedures.
t. Position patch on doubler and hold with clecos. Rivet patch to doubler using MS20426AD6 rivets.

NOTE

WIPE ALL MOISTURE FROM RIVETS AND APPLY A COAT OF MIL-S-8802 TO RIVET SHANK BEFORE RIVET INSTALLATION.

u. Attach MS21059L3 plate nuts to inside of doubler using setup bolts. Drill plate nut rivet holes using No. 40 (0.098) drill. Countersink holes using a 100° countersink. Remove burrs and shavings.
v. Install plate nuts on inside of doubler using MS20426AD3 rivets. Apply sealer, Specification MIL-S-8802, to faying surfaces of doubler and plate nuts.

NOTE

WIPE ALL MOISTURE FROM RIVETS AND APPLY A COAT OF MIL-S-8802 TO RIVET SHANK BEFORE RIVET INSTALLATION.

w. Apply coat of sealer, Specification MIL-S-8802, over rivets on inner surface of doubler and patch.
x. Apply coat of sealer, Specification MIL-S-8802, on faying surfaces of tank and doubler assembly. Position doubler assembly in tank and install bolts NAS1203 using sealer, Specification MIL-S-8802, applied to threads of bolts.
y. Fill gap between patch and tank skin with MIL-S-38228.

CONCLUSION

a. Verify that inside of tank is clean and free of foreign materials.
b. Reinstall tank access doors.
c. Install tank on airplane. Refer to T.O. 1F-106A-2-5-2-1 for this procedure.

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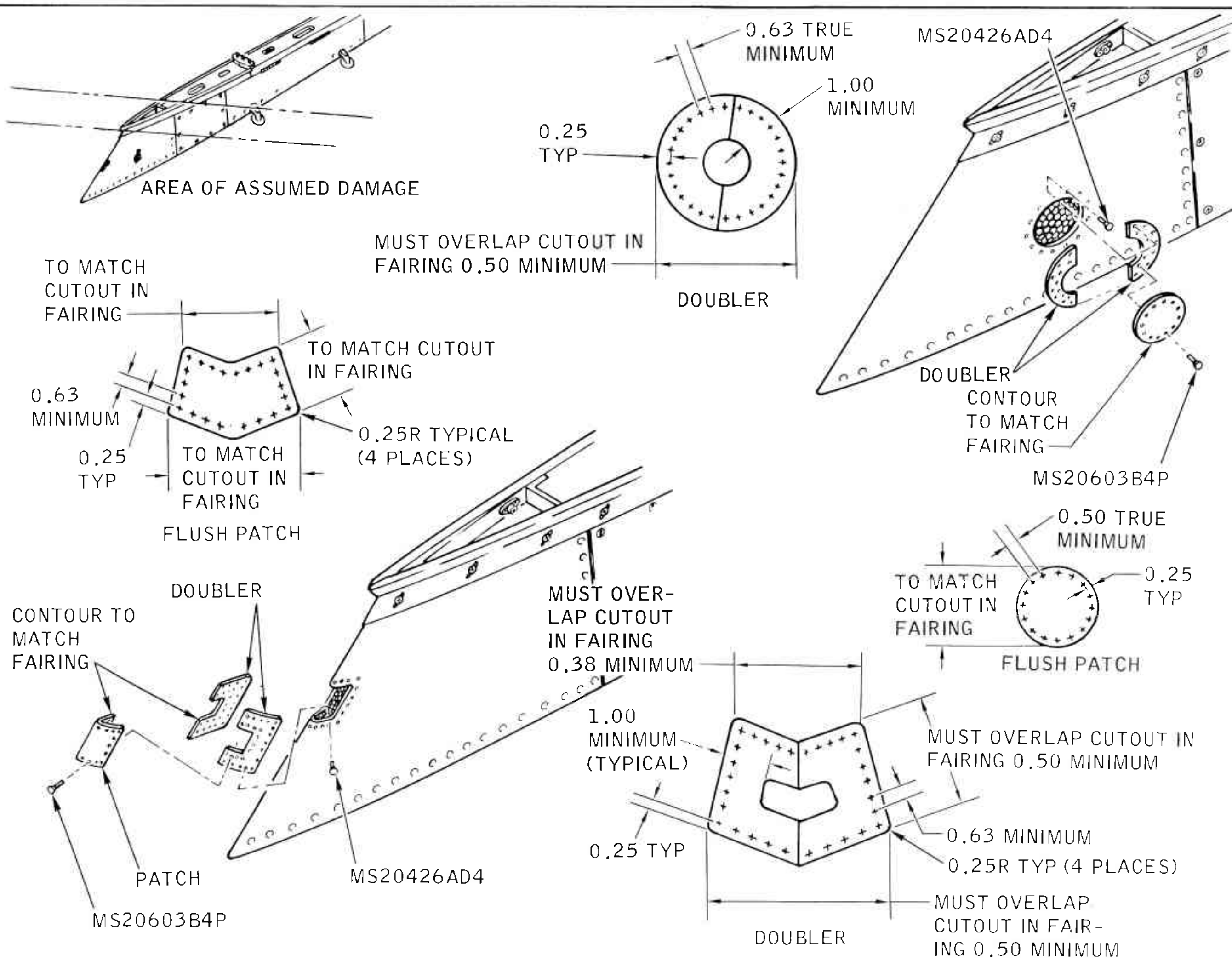
Figure 2-40A. External Fuel Tank Repairs (Sheet 3 of 3)

All data on pages 2-80D through 2-81, including figure 2-41 deleted.



REPAIR PROCEDURE

a. Perform a fluorescent penetrant inspection to determine extent of damage. Refer to Section I for this inspection procedure.
b. Cut out damaged area to leave a regular shape in existing skin. Cutout must be of sufficient size for insertion of doubler halves.
c. Working through cutout area, remove sufficient amount of the foam filler material to facilitate installation of repair parts.
d. Fabricate repair parts from 2024 T3 stock, 0.040 inch thick, as shown.

NOTE

FOR EASE OF FORMING, A SQUARE OR RECTANGULAR PATCH IS PREFERRED ALONG THE LEADING EDGE OF THE FAIRING. THE SHAPE OF THE CUTOUT FOR DAMAGE REMOVAL WILL DETERMINE THE PATCH SHAPE FOR THE FLAT AREAS OF THE FAIRING.

e. Install doubler halves and hold in place with clamps.
f. Drill in doubler attachment rivet pattern on 1/2 inch centers using a No. 30 (0.128) drill.
g. Remove doubler halves. Remove burrs from fairing and doubler halves.
h. Clamp flush patch to doubler halves and drill patch rivet pattern using a No. 30 (0.128) drill.
i. Remove clamps and remove burrs from repair parts.
j. Assemble doubler halves to patch and hold together using clecos.
k. Countersink patch-doubler holes using a 100° countersink.
l. Install doubler halves in fairing and hold with clecos.
m. Countersink fairing-doubler holes using 100° countersink.
n. Rivet doubler halves in place as shown.
o. Install flush patch as shown.
p. Fill gaps between fairing and patch using aerodynamic smoothing compound according to applicable schedule given in Table II-VII.

6SR-958-3

Figure 2-41A. External Fuel Tank Pylon Fairing Repair

All data on pages 2-83 thru 2-98 including Figure 2-42 thru 2-49 are deleted.

2-86. . **Packing and Crating.**

2-87. Packing and crating of the elevon, wing tip, and drop tank assemblies for shipping or storage may be accomplished as shown on figures 2-48 through 2-50. The airplane component being packaged and the materials used shall conform to the following specifications:

a. Preservation of airplane components shall be accomplished according to Specification MIL-P-116, Method III.

b. Packing of airplane components shall be accomplished according to Specification MIL-C-25731. Crates shall be Type I, Class II.

c. Components packed for air shipment shall be packed in open crates conforming to Specification MIL-C-25731, Type I, Class I.

d. Marking of crates shall be in accordance with Specification MIL-STD-129. In addition to marking required by MIL-STD-129, the words "Removable End" and "Reusable Crate" shall be stenciled on both ends of the crate. The words "Sling Here" and "Center of Balance" shall be stenciled in their proper place. See table 1-LII for center of balance data.

2-88. **PREPARATION OF WING FOR AIRLIFT.**

2-89. Preparation of a wing for airlift requires three major operations. The wing must be defueled and purged, removed from the airplane, and installed on a shipping stand. Since JP-4 is both toxic and flammable, it is of the utmost importance that all fumes be removed from the wing before airlift operation.

a. Defuel airplane in accordance with T.O. 1F-106A-2-5-2-1.

b. Purge wing with 100-octane gasoline. Refer to paragraph 2-41 for procedure.

c. Remove wing tip and leading edge. Refer to T.O. 1F-106A-2-2-2-2 for procedure.

d. Package all removed parts and components, other than elevon and main landing gear, in accordance with instructions contained in T.O. 00-85-16, or stow in boxes provided with the shipping stand for shipment with the wing. This includes all access doors and attaching screws.

e. Remove wing from fuselage. See figure 2-7 for procedure.

f. Cap all hydraulic and fuel lines; stow or remove all loose lines and electrical connectors.

g. Place wing on dolly or mattresses.

NOTE

The right wing must be inverted (turned upside down) prior to installation on shipping stand. The No. 2 and No. 6 spar adapters, provided with the shipping stand, can be used to invert the right wing. Shipping stand shown in Figure 2-51.

h. Attach the No. 2 spar adapter, AF Dwg. 58SAC589, to No. 2 spar and secure with lock pins.

i. Attach the No. 6 spar adapter, AF Dwg. 58SAC588, to No. 6 spar and secure with lock pins.

j. Attach the universal lifting eye, provided with the shipping stand, to the wing lift point. If the left wing is to be air shipped, attach universal lifting eye to the wing tiedown point. See figure 2-51.

k. Attach hoisting sling, AF Dwg. 58SAB591, to spar adapters and lifting eye. Attach two 15-foot lengths of rope, as shown in figure 2-51, to guide wing into place on shipping stand.

l. Hoist wing until No. 6 spar adapter clears the curved pivot hook at the top of the shipping stand. Lower and guide the wing until the pivot hook enters the 1½-inch diameter hole in the No. 6 spar adapter. Continued lowering will cause the wing to pivot into position on the shipping stand. See figure 2-51.

m. Bolt No. 6 spar adapter to shipping stand with four AN 12-24A bolts, four AN 960-1216 washers, and four AN 365-1216 nuts. Secure the No. 2 spar adapter to the shipping stand using four AN 960-1216 washers and four AN 365-1216 nuts.

n. Attach the wing support pad to the elevon attach fitting as shown on figure 2-51 and adjust to insure that the wing clears the shipping stand.

NOTE

The canted position of the wing, when installed on the shipping stand, may cause trapped fuel to drain into the lower portion of the tanks. Since all residual fuel must be removed prior to airlift, the wing should be placed on the shipping stand at least eight hours prior to airlift to allow for fuel drainage and subsequent mopping. Should the shipping stand not be available prior to arrival of transport aircraft, the intent of the foregoing shall be accomplished by otherwise devising a means of propping the wing in a canted position.

2-90. **JIGS.**

2-91. Figure 2-42 shows the jig used to support the elevon during a typical repair operation. Figures 2-43 and 2-44 show the master tooling fixture for the inboard and outboard elevon assembly. Figure 2-45 shows the master tooling fixture for the Case XIV wing assembly, and figure 2-46 shows the master tooling fixture for the Case XXIX wing assembly. Figure 2-47 shows the elevon contour boards used for checking elevon alignment.

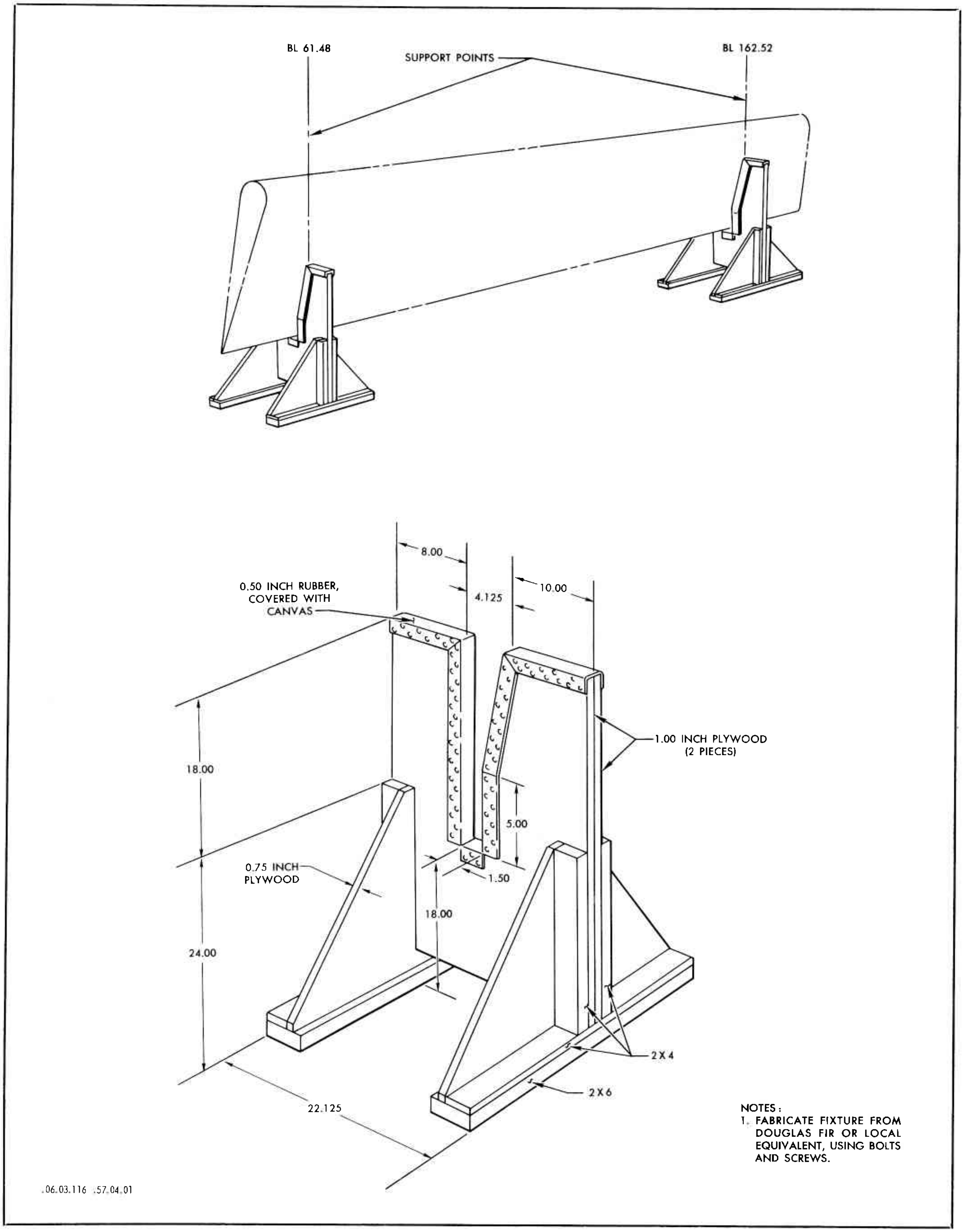
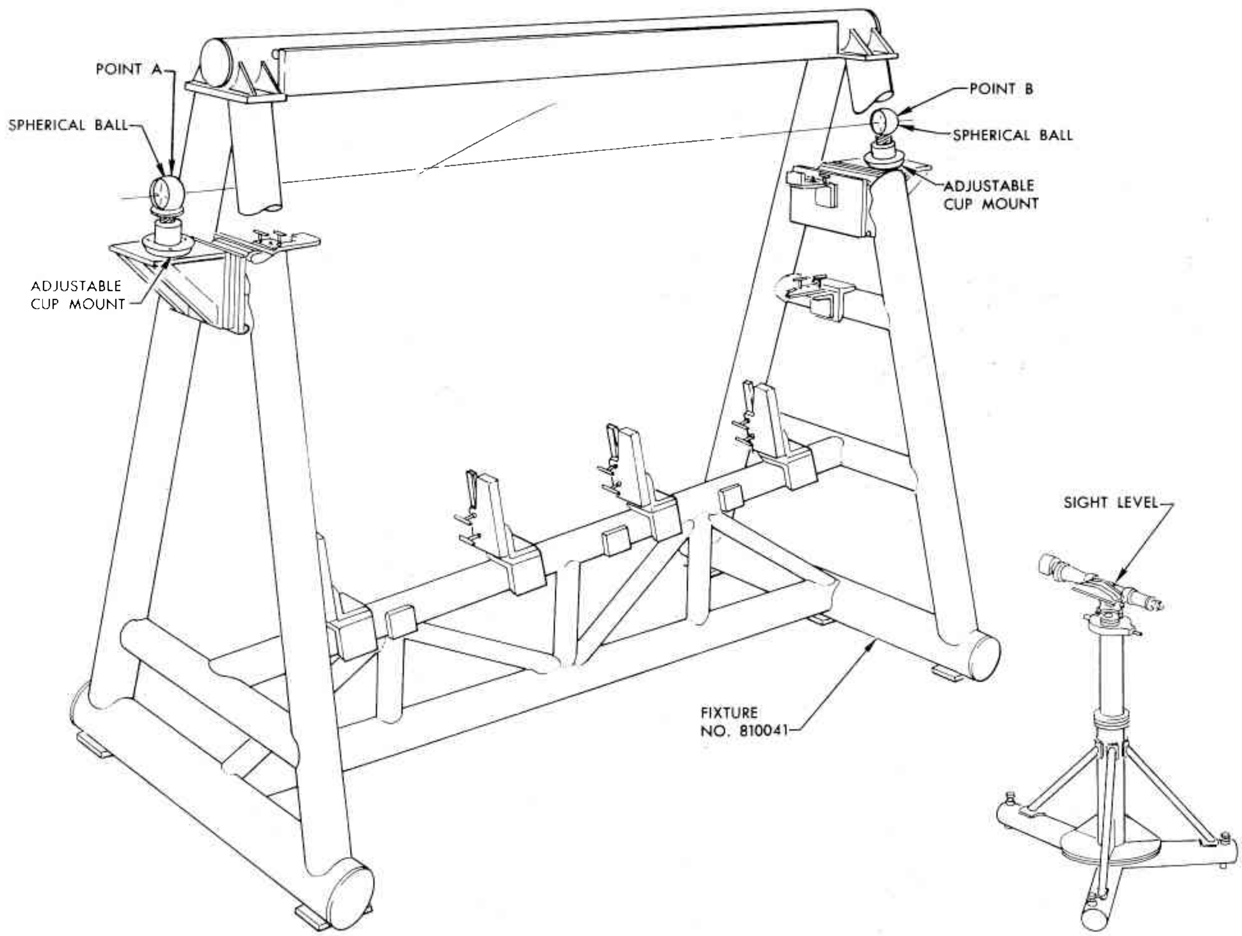


Figure 2-42. Elevon Repair Support Fixture



TOOL FUNCTION

To assemble elevon spar assembly, ribs, hinge fittings, wedge assembly and skins.

OPTICAL TOOLING EQUIPMENT REQUIRED

a. One sight level.
b. One alignment telescope.
c. One 60-inch inside micrometer.
d. Two spherical balls with 2¼-inch plastic targets.
e. One 1½-inch plastic target.
f. One 6-inch master spirit level.

FABRICATION PROCEDURE

a. Level fixture as follows:
   (1) Set up sight level away and to one side of fixture.
   (2) Refer to Section I for leveling procedure using sight level.

(NOTE: PROCEDURE CONTINUED ON SHEET 2).

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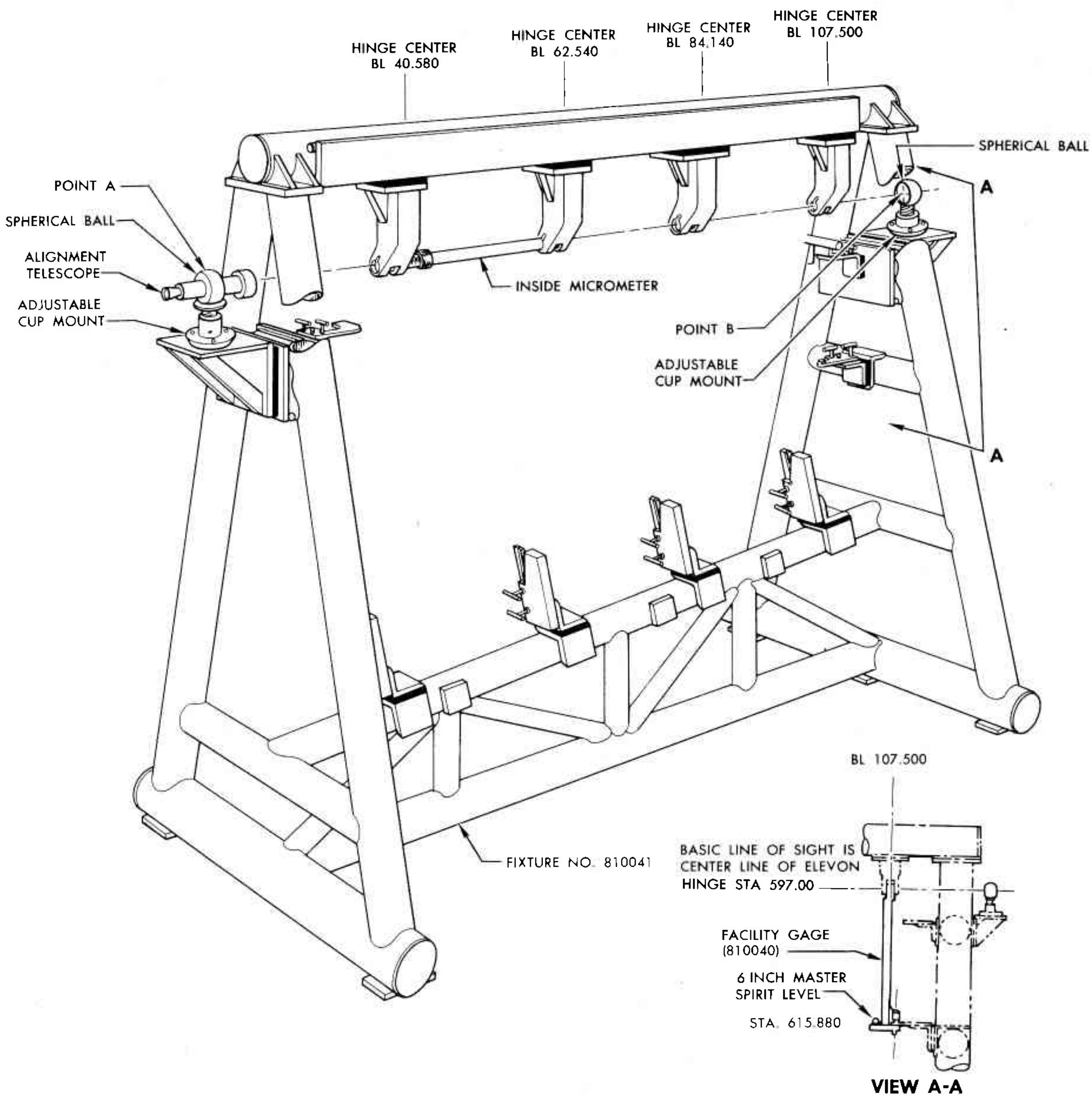
Figure 2-43. Master Tooling — Elevon Assembly, Inboard Left-Hand (Sheet 1 of 2)



b. Establish basic line of sight:
   (1) Mount alignment telescope at point A.
   (2) Mount 2¼-inch plastic target at point B.
   (3) Refer to Section I for use of alignment telescope.

NOTE

BASIC LINE OF SIGHT IS CENTERLINE OF ELEVON HINGE AT FUSELAGE STATION 597.000, WATER LINE -16.000.

c. Mount hinge locators as follows:
   (1) Set hinge locators in basic line of sight with 1½ inch plastic target.
   (2) Mount facility gage, 810040, then level gage with six-inch master spirit level to control rotation of gage. Leveling gage will establish centerline of hinge locator at buttock line 107.500.
   (3) Using the 60-inch inside micrometer, position the remaining hinge locators at buttock lines 84.140, 62.540 and 40.580 (position hinge locators in this order).

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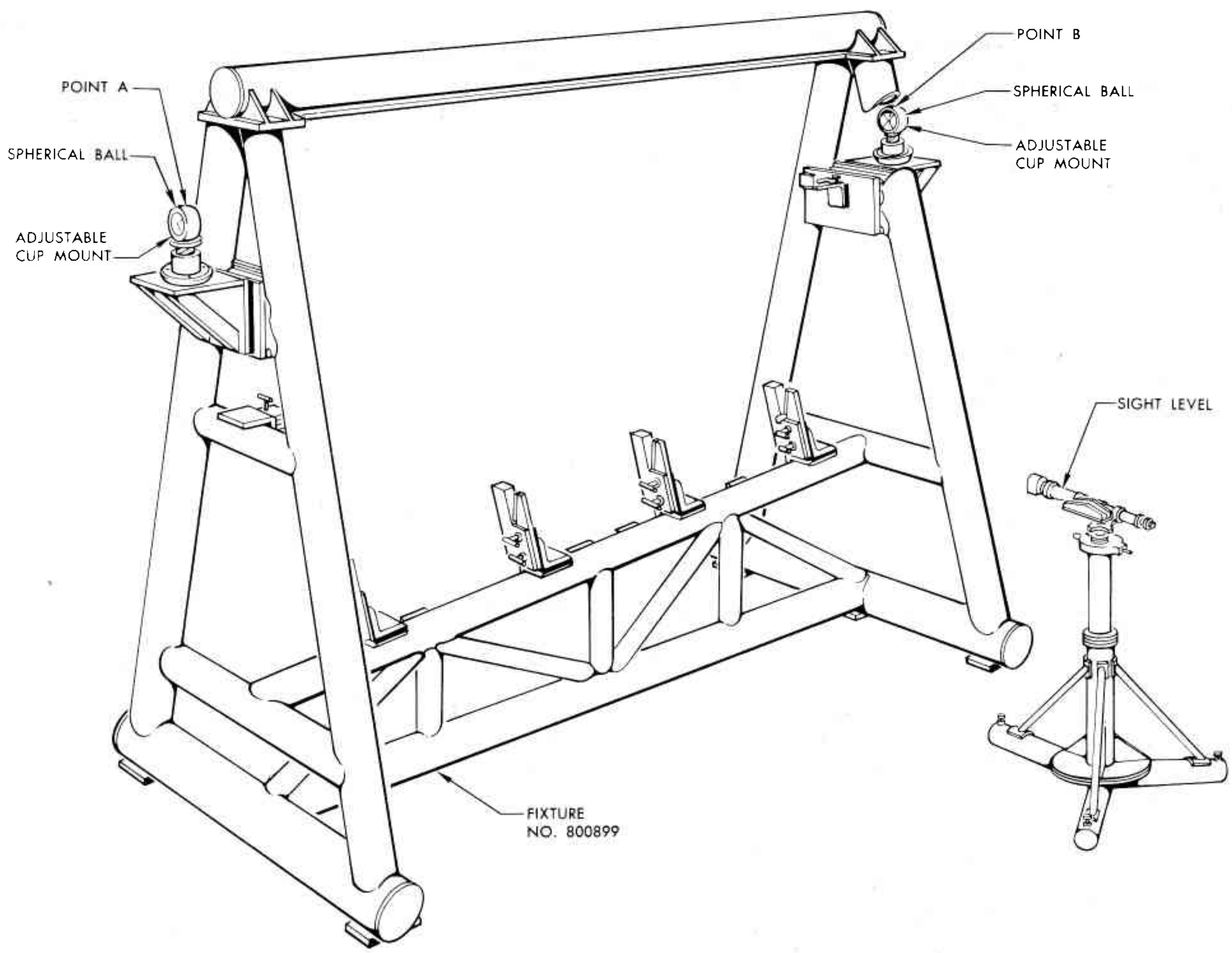
Figure 2-43. Master Tooling — Elevon Assembly, Inboard Left-Hand (Sheet 2 of 2)



TOOL FUNCTION

To assemble elevon spar assembly, ribs, hinge fittings, wedge assembly and skins.

OPTICAL TOOLING EQUIPMENT REQUIRED

a. One sight level.
b. One alignment telescope.
c. One 60-inch inside micrometer.
d. Two spherical balls with 2¼-inch plastic targets.
e. One 1½-inch plastic target.
f. One 6-inch master spirit level.

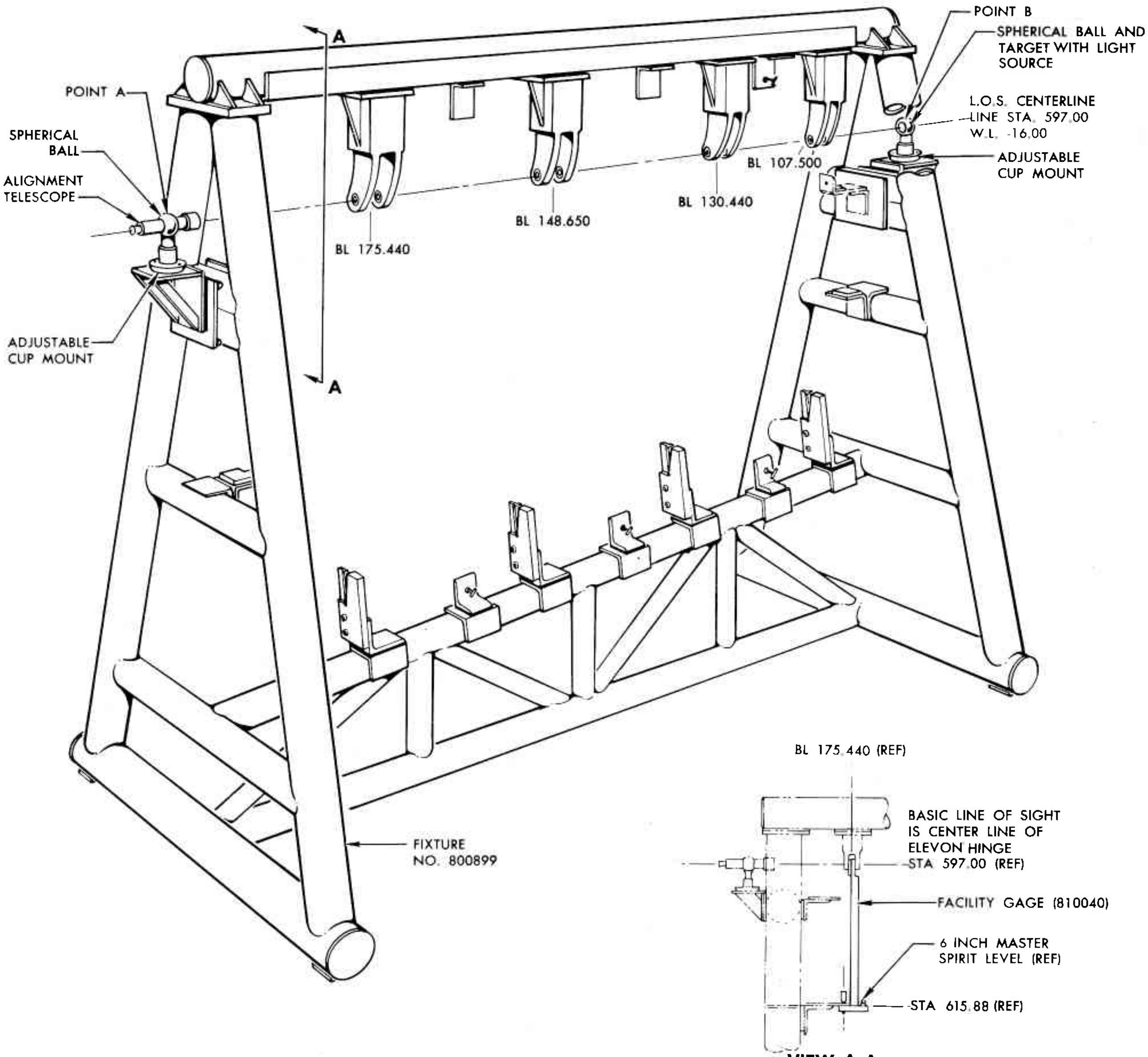
FABRICATION PROCEDURE

a. Level fixture as follows:
   (1) Set up sight level away and to one side of fixture.
   (2) Refer to Section I for leveling procedure using sight level.

NOTE: PROCEDURE CONTINUED ON SHEET 2.

06.03.207-1C

Figure 2-44. Master Tooling — Elevon Assembly, Outboard Left-Hand (Sheet 1 of 2)



VIEW A-A

b. Establish basic line of sight as follows:
   (1) Mount alignment telescope at point A.
   (2) Mount 2¼-inch plastic target at point B.
   (3) Refer to Section I for use of alignment telescope.

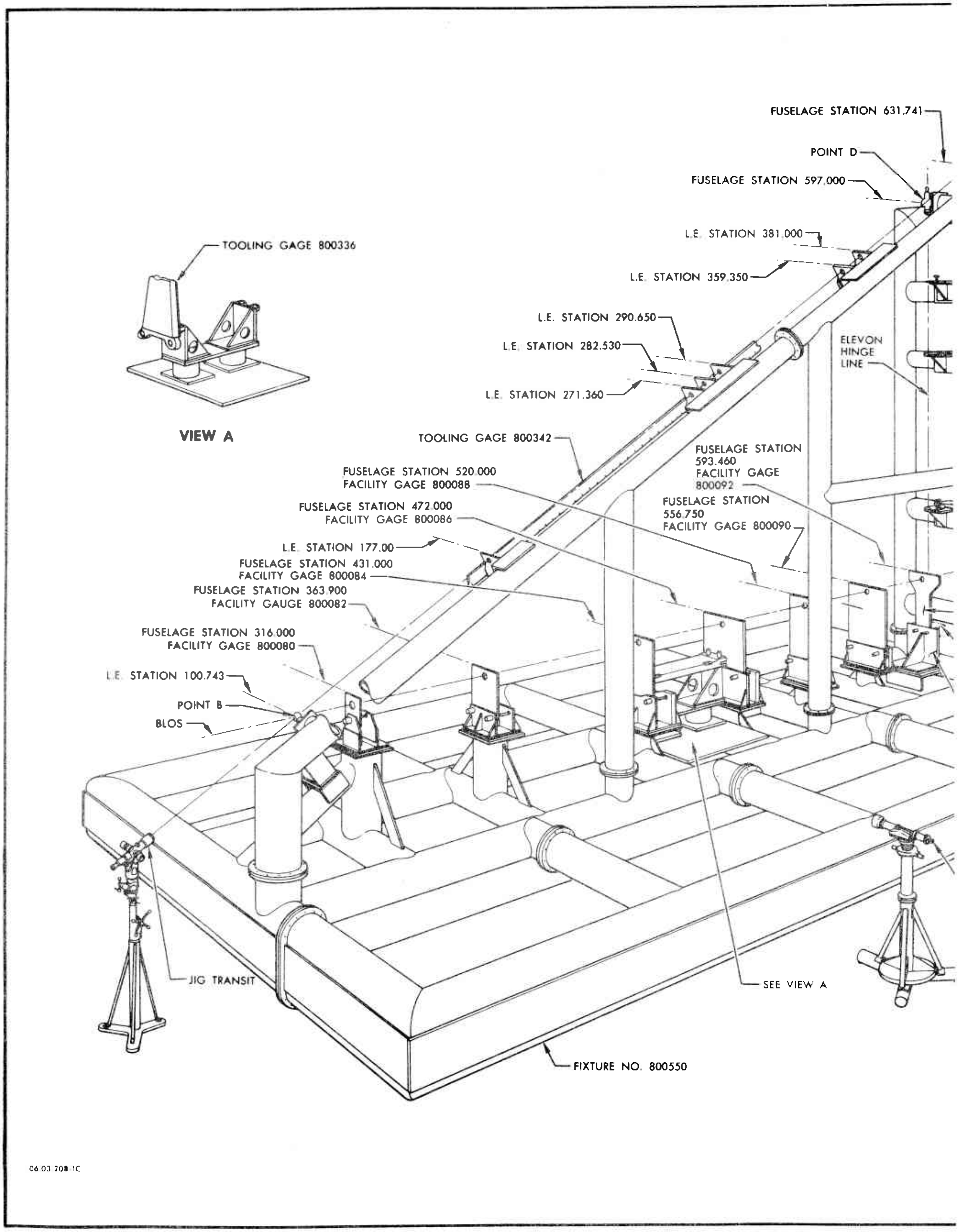
NOTE

BASIC LINE OF SIGHT IS CENTERLINE OF ELEVON HINGE AT FUSELAGE STATION 597.000, WATER LINE -16.000.

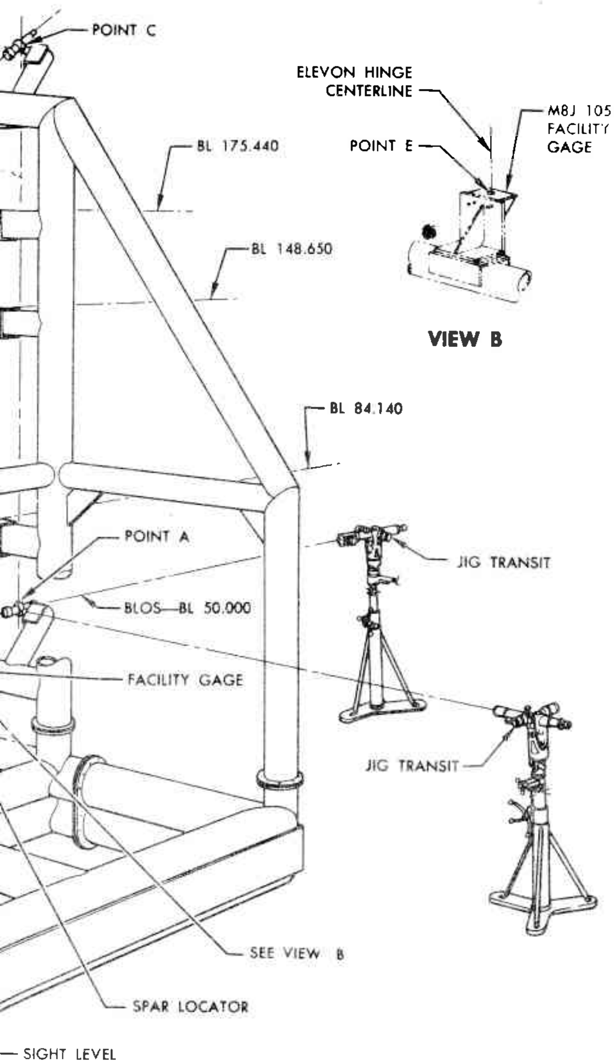
c. Mount hinge locators as follows:
   (1) Set hinge locators in basic line of sight with 1½ inch plastic target.
   (2) Mount facility gage, 810040, then level gage with 6-inch master spirit level to control rotation of gage. Leveling gage will establish centerline of hinge locator at buttock line 107.500.
   (3) Using the 60-inch inside micrometer, position the remaining hinge locators at buttock lines 130.440, 148.650, and 175.440 (hinge locators to be positioned in this order).

06.03.207.2C

Figure 2-44. Master Tooling — Elevon Assembly, Outboard Left-Hand (Sheet 2 of 2)



FUSELAGE STATION 631.741
POINT D
FUSELAGE STATION 597.000
L.E. STATION 381.000
L.E. STATION 359.350
TOOLING GAGE 800336
L.E. STATION 290.650
L.E. STATION 282.530
ELEVON HINGE LINE
L.E. STATION 271.360
VIEW A
TOOLING GAGE 800342
FUSELAGE STATION 593.460 FACILITY GAGE 800092
FUSELAGE STATION 520.000 FACILITY GAGE 800088
FUSELAGE STATION 472.000 FACILITY GAGE 800086
FUSELAGE STATION 556.750 FACILITY GAGE 800090
L.E. STATION 177.00
FUSELAGE STATION 431.000 FACILITY GAGE 800084
FUSELAGE STATION 363.900 FACILITY GAUGE 800082
FUSELAGE STATION 316.000 FACILITY GAGE 800080
L.E. STATION 100.743
POINT B
BLOS
JIG TRANSIT
SEE VIEW A
FIXTURE NO. 800550
06.03 208-1C



TOOL FUNCTION

a. To check wing alignment.
b. To check alignment of elevon attach points.

OPTICAL TOOLING EQUIPMENT REQUIRED.

a. Two alignment telescopes.
b. Two jig transits.
c. One sight level.
d. Two 2¼-inch diameter targets with light sources.
e. Two 1½-inch diameter plastic targets.
f. One 12-inch master spirit level.
g. One 6-inch master spirit level.
h. Four 3½-inch diameter spherical balls.
i. One set of 30-foot inside micrometers.
j. One auto-reflection mirror.

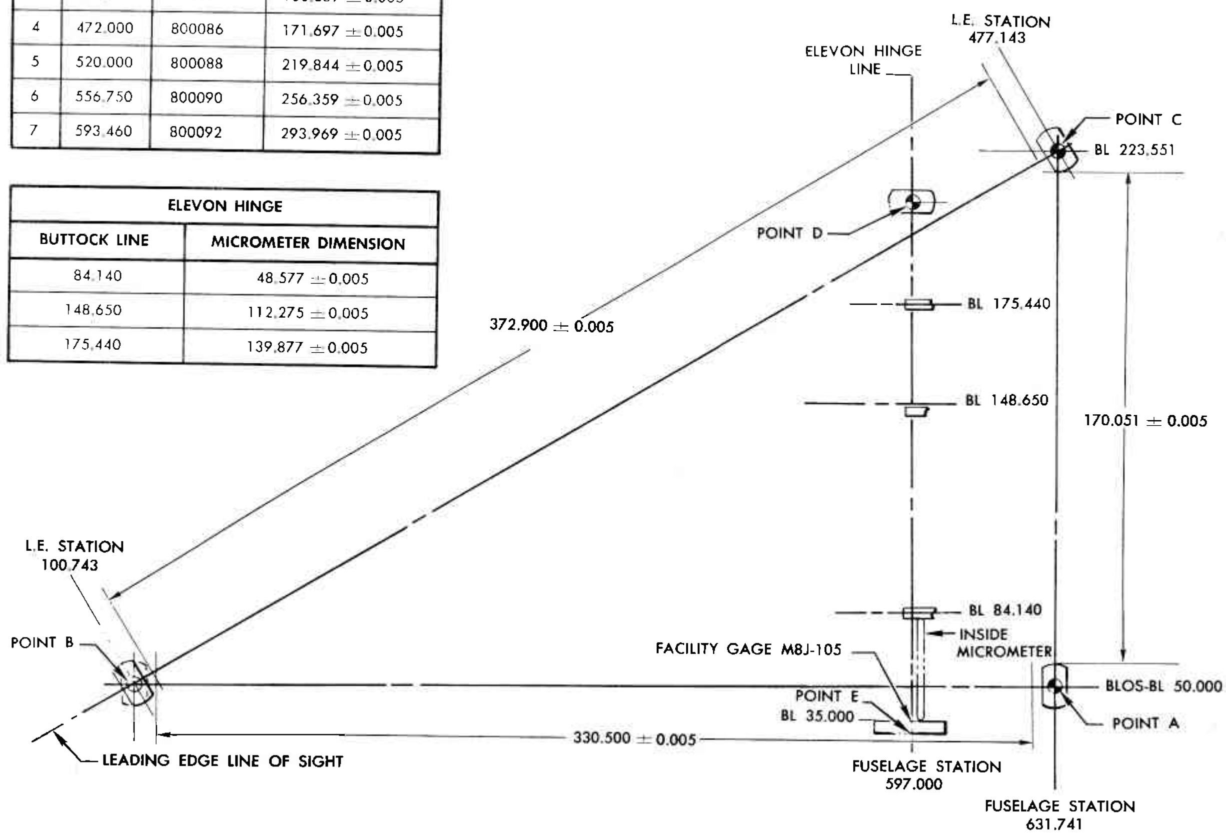
MASTER TOOLING PROCEDURE

a. Level fixtures as follows:
   (1) Set up sight level away from and to one side of fixture.
   (2) Refer to Section I for leveling procedure using a sight level.
b. Establish basic line of sight (BLOS) as follows:
   (1) Mount spherical balls at points A and B on adjustable cup mounts.
   (2) Establish a level horizontal plane between points A and B with sight level.
   (3) Using the 30-foot inside micrometer, adjust spherical balls at points A and B to dimension shown on sheet 2.
c. Establish point C as follows:
   (1) Set up one jig transit away from and to one side of fixture at an approximate 90 degree angle from point A. This jig transit establishes a true vertical plane between points A and C. Refer to Erection of a Vertical Plane from a Horizontal Plane using the Jig Transit in Section I.
   (2) Set up a second jig transit away from fixture and in line with points A and B. Use method referred to in previous step and move point C into vertical plane in line with points A and B.
d. Establish the linear distance between points B and C as follows:
   (1) Using the 30-foot inside micrometer, adjust the spherical ball at point C in its proper relation with point A, as shown on sheet 2.
   (2) Again using the 30-foot inside micrometer, measure the distance between points B and C. If this dimension agrees with the dimension given on sheet 2, proceed with setting up wing geometry. If dimension does not check, repeat steps "b" through "d" and recheck dimension between points B and C.
e. Establish points D and E as follows:
   (1) Mount an alignment telescope at point A.
   (2) Mount a 2¼-inch diameter target with light source at point B.
   (3) Set spar locator at fuselage station 593.46 using facility gage, 800092. Measure distance from point B to facility gage as shown on Sheet 2 to establish the correct location of the facility gage. Insert 1½-inch plastic target in facility gage and align gage with BLOS.
   (4) Mount the M8J 105 facility Gage on locator at station 593.46, as shown in view B. Centerline of the M8J 105 facility gage establishes point E at fuselage station 597.000.
   (5) Set up a jig transit away from fixture and opposite point E.
   (6) Mount an alignment telescope at point C.
   (7) Mount a 2¼-inch diameter target with light source at point B.
   (8) Using the jig transit, establish a vertical plane parallel to the line of sight between points A and C.
   (9) Establish a line of sight between points C and B.
   (10) Establish point D where the vertical plane from point E intersects the line of sight between points B and C.
f. Establish the elevon hinge points by using the 30-foot inside micrometer and measuring the distances shown on Sheet 2. All dimensions are taken from the upper surface of the M8J 105 facility gage (point E).
g. Set remaining spar locators in the BLOS by method used in step "e". (3) and maintain correct dimensions between them with an inside micrometer, as shown on Sheet 2.

Figure 2-45. Master Tooling — Case XIV Wing Assembly (Sheet 1 of 2)
Applicable to F-106A airplanes 56-453 thru 56-466, and F-106B airplane 57-2507

JIG DIMENSIONS			
SPAR NO.	FUSELAGE STATION	FACILITY GAGE NO.	MICROMETER DIMENSION
1	316.000	800080	16.919 ±0.005
2	363.900	800082	63.909 ±0.005
3	431.000	800084	130.659 ±0.005
4	472.000	800086	171.697 ±0.005
5	520.000	800088	219.844 ±0.005
6	556.750	800090	256.359 ±0.005
7	593.460	800092	293.969 ±0.005

ELEVON HINGE	
BUTTOCK LINE	MICROMETER DIMENSION
84.140	48.577 ±0.005
148.650	112.275 ±0.005
175.440	139.877 ±0.005



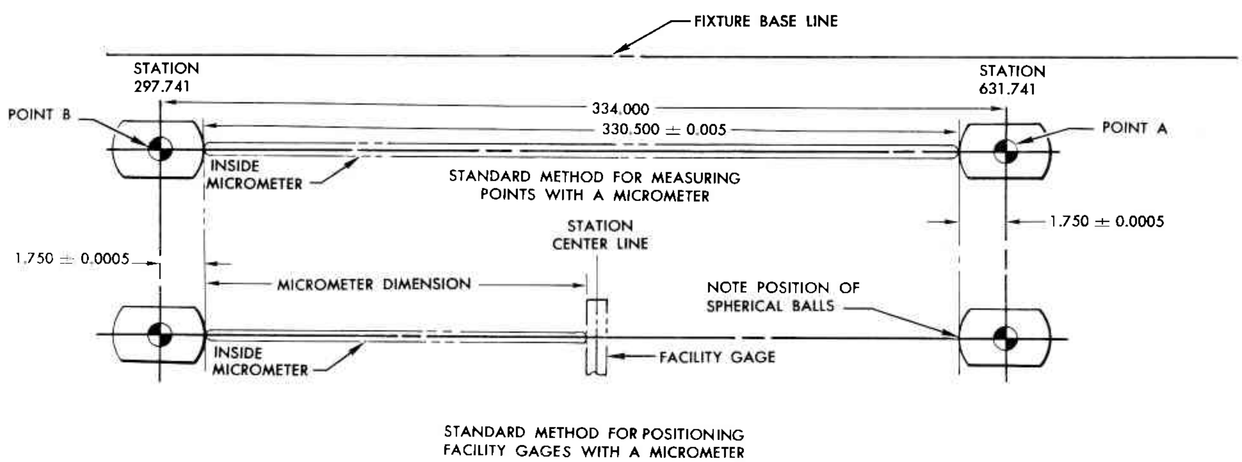


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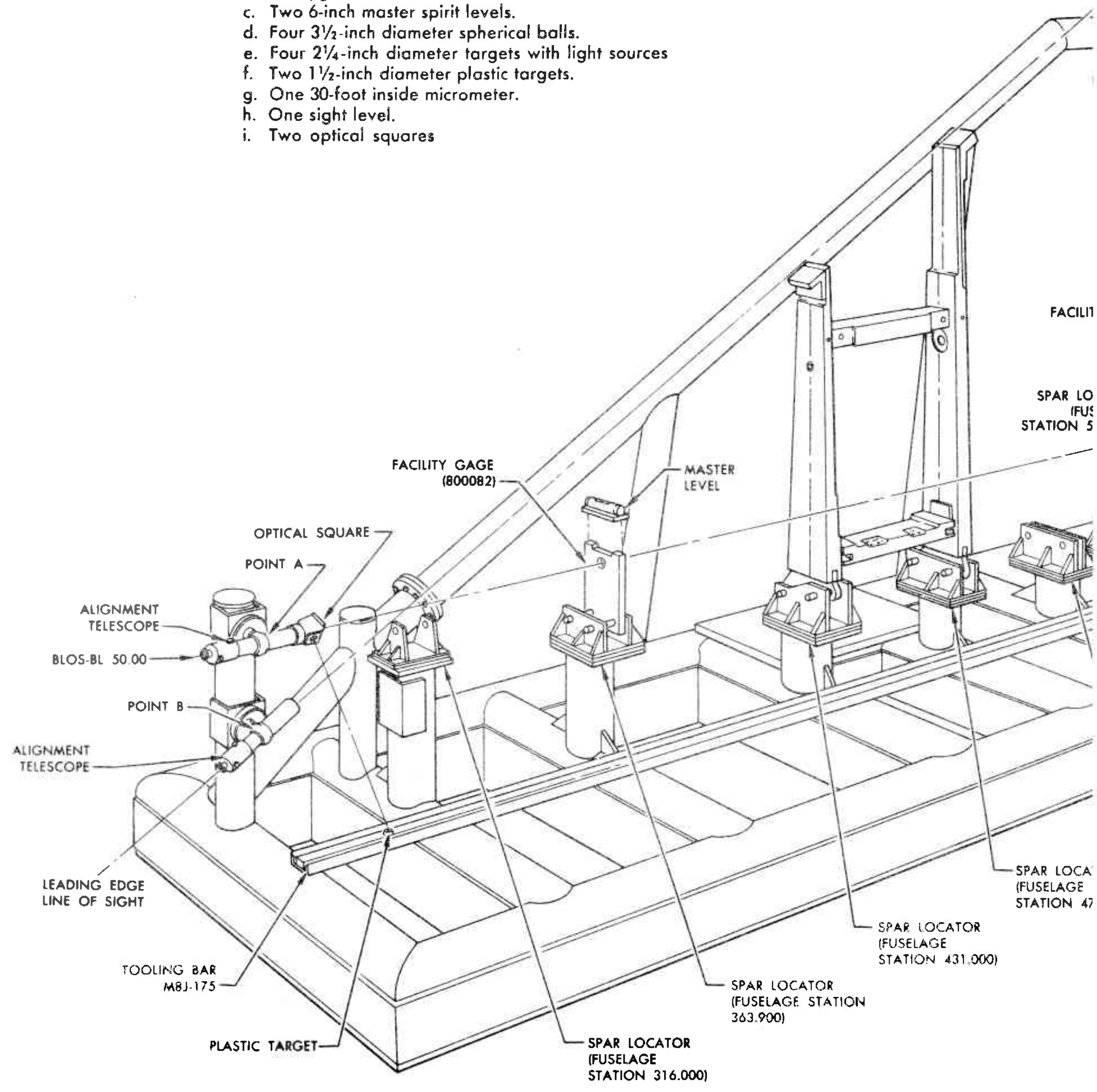
Figure 2-45. Master Tooling — Case XIV Wing Assembly (Sheet 2 of 2)
Applicable to F-106A airplanes 56-453 thru 56-466, and F-106B airplane 57-2507

TOOL FUNCTION

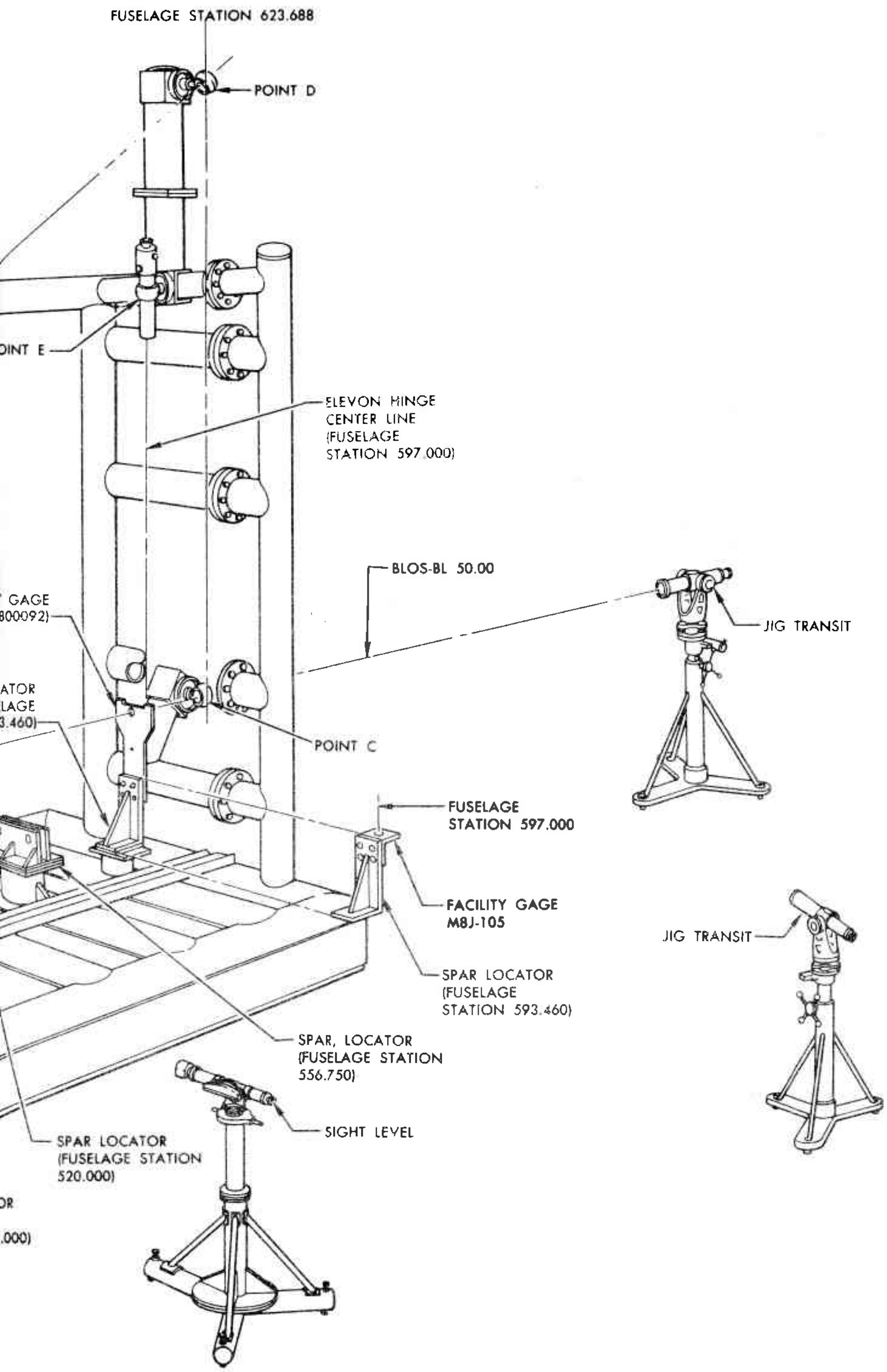
To check alignment of wing.

OPTICAL TOOLING EQUIPMENT REQUIRED.

a. Two alignment telescopes.
b. Two jig transits.
c. Two 6-inch master spirit levels.
d. Four 3½-inch diameter spherical balls.
e. Four 2¼-inch diameter targets with light sources
f. Two 1½-inch diameter plastic targets.
g. One 30-foot inside micrometer.
h. One sight level.
i. Two optical squares



06 03 209 1C



MASTER TOOLING PROCEDURE

a. Level fixture and establish a basic line of sight (BLOS) as follows:
   (1) Set up sight level away from and to one side of fixture.
   (2) Refer to Section I for leveling procedure using sight level.
   (3) Mount spherical balls with targets and light source at points A and C.
   (4) Establish horizontal alignment of points A and C.
   (5) Remove target with light source at point A and insert an alignment telescope.
   (6) Align point A with point C.
   (7) Using the 30-foot inside micrometer, establish point C as shown on sheet 2.

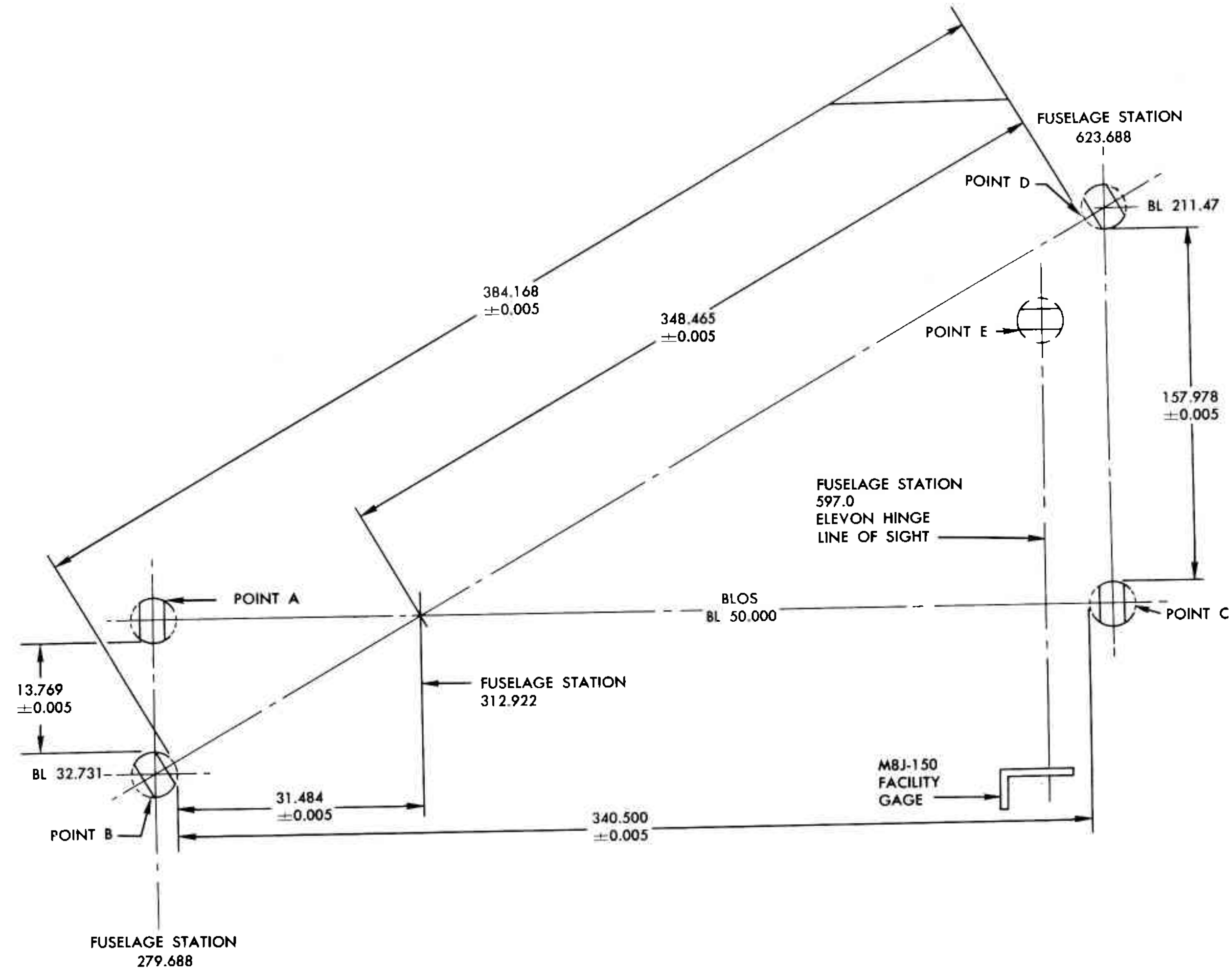
b. Establish points B and D as follows:
   (1) Set up one jig transit away from fixture and approximately halfway between points A and C.
   (2) Set up second jig transit away from fixture and in line with points A and C.
   (3) Refer to Section I for Erection of a Vertical Plane from a Horizontal Plane using the Jig Transit.

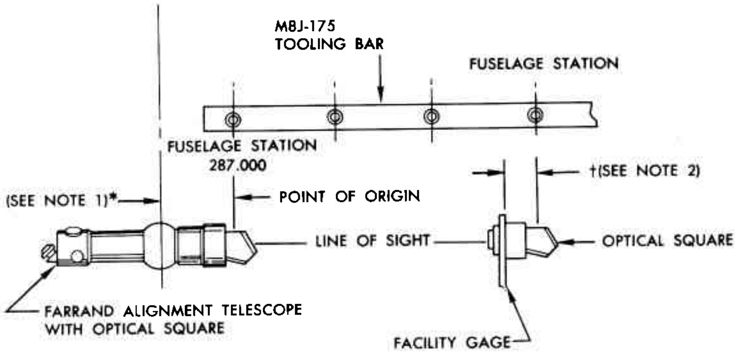
NOTE

USE THE 30-FOOT INSIDE MICROMETER TO ESTABLISH THE TRUE DIMENSIONS BETWEEN POINTS. USE THE JIG TRANSIT SET UP AWAY FROM AND TO ONE SIDE OF THE FIXTURE TO ESTABLISH A TRUE VERTICAL PLANE. USE THE JIG TRANSIT SET UP AWAY FROM FIXTURE AND IN LINE WITH POINTS A AND C TO HOLD ALL POINTS IN A PARALLEL PLANE.

c. Establish the spar reference points as follows:
   (1) Mount an alignment telescope with an optical square at point A.
   (2) Establish the location of the M8J-175 tooling bar.
   (3) Rough set the spar No. 1 locator.
   (4) Mount the facility gage, 800080 within the spar locator and insert a plastic target in the facility gage line-of-sight hole.
   (5) Bring the facility gage and spar locator into line of sight of optical square and control rotation with 6-inch master spirit level.
   (6) Repeat steps "c. (1)" through "c. (5)" to locate spars No. 2, No. 3, No. 4, No. 5, No. 6 and No. 7 using the facility gages and station callouts shown on sheet 2.

Figure 2-46. Master Tooling — Case XXIX Wing Assembly (Sheet 1 of 2)
Applicable to F-106A airplanes 56-467, 57-229 and subsequent, and F-106B airplanes 57-2508 and subsequent





SPAR	FUSELAGE STATION	FACILITY GAGE NUMBER
1	319.910	800080
2	367.430	800082
3	434.850	800084
4	475.860	800086
5	523.695	800088
6	560.680	800090
7	596.460	800092

NOTES:
1. * DENOTES THE STANDARD DISTANCE (7.312 INCHES) FROM THE EXIT LINE OF SIGHT OF THE OPTICAL SQUARE AND THE CENTERLINE OF THE INTEGRAL SPHERICAL BALL ON THE FARRAND ALIGNMENT TELESCOPE.
2. † DENOTES THE STANDARD DISTANCE (3.000 INCHES) FROM THE EXIT LINE OF SIGHT OF THE OPTICAL SQUARE AND ITS MOUNTING FACE.
3. REFER TO PARAGRAPH "THE OPTICAL SQUARE" IN SECTION 1 FOR INFORMATION REGARDING THE USE OF THE OPTICAL SQUARE.

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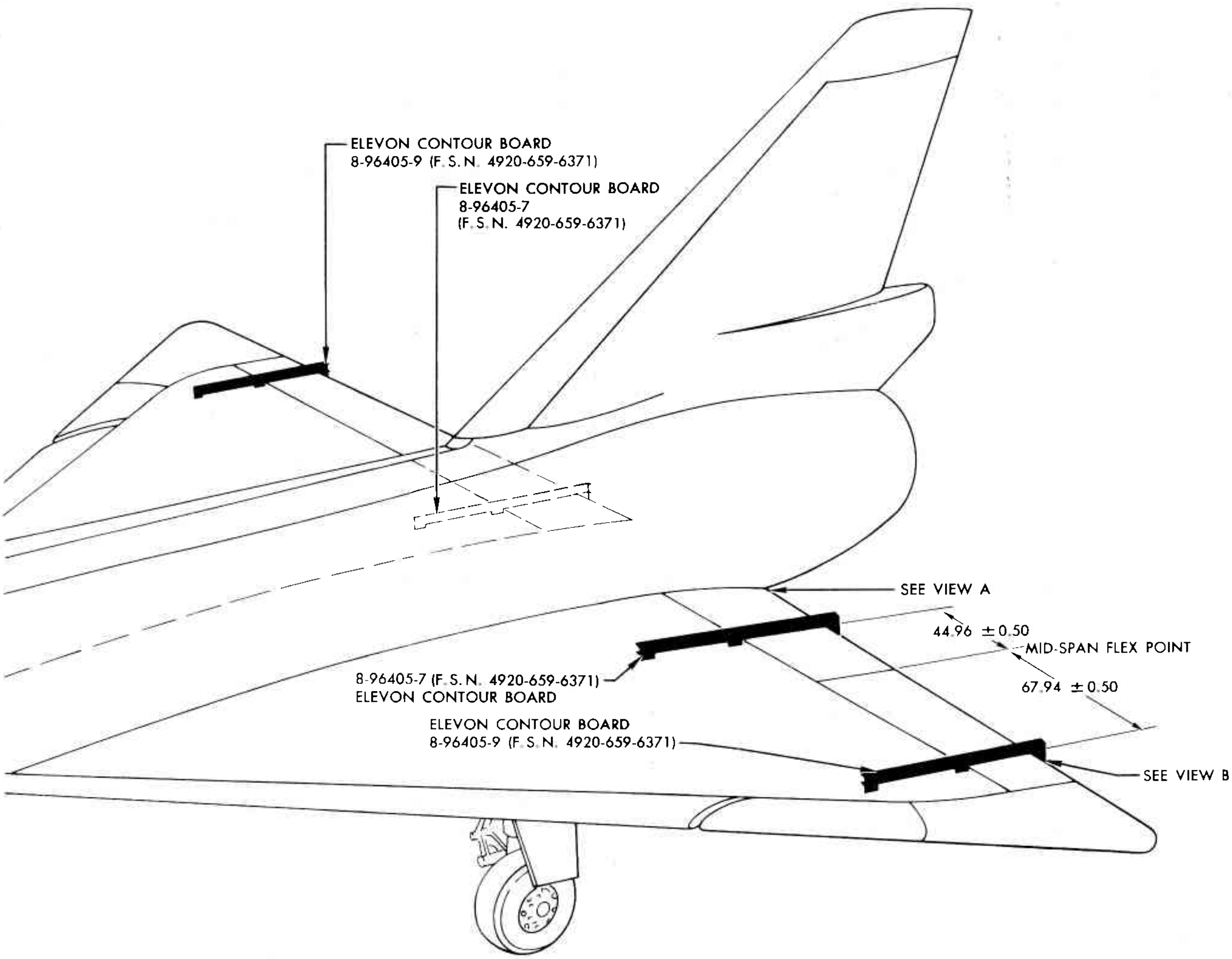
Figure 2-46. Master Tooling — Case XXIX Wing Assembly (Sheet 2 of 2)
Applicable to F-106A airplanes 56-467, 57-229 and subsequent, and F-106B airplanes 57-2508 and subsequent



CONTOUR BOARD ALIGNMENT CHECK PROCEDURE:

a. Prepare airplane as required for elevon operation procedure given in T.O. 1F-106A-2-7-2-1.

b. Trim elevons to neutral (streamline position). Elevons are in neutral when inboard tips are in line with positioning rivets on fuselage (see view A) under the following conditions: hydraulic power on, engine not installed, no fuel in tanks and elevator and aileron mixer rig pin holes aligned.

c. Position the 8-96405-7 (F. S. N. 4920-659-6371) contour board 44.96 (±0.50) inches inboard from the centerline of the mid-span flex point of both elevons.

NOTE

WITH ENGINE INSTALLED AND FUEL IN TANKS, TRAILING EDGE MAY DROOP ONE-EIGHTH INCH BELOW CENTER OF POSITIONING RIVET.

d. Position the 8-96405-9 (F. S. N. 4920-659-6371) contour board 67.94 (±0.50) inches outboard from the centerline of the mid-span flex point of both elevons.

NOTE

CONTOUR BOARDS MUST BE PLACED PARALLEL TO LINE OF FLIGHT.

e. Check and record deviation from wing chord line as calibrated on plate at aft end of each contour board (see view B). Total deviation between inboard and outboard contour board readings should not exceed ±0.25 inch on either wing.

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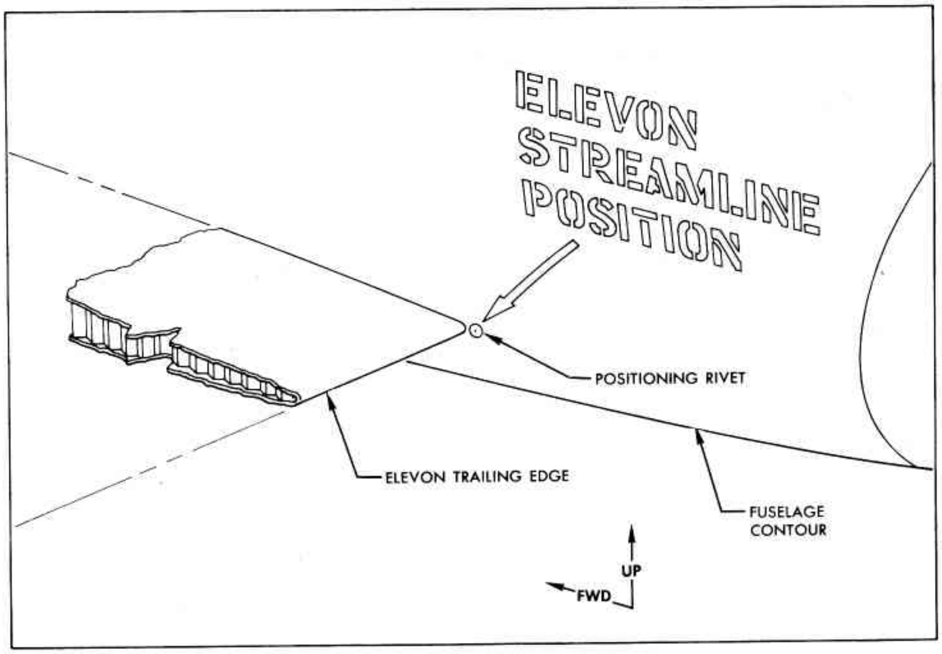
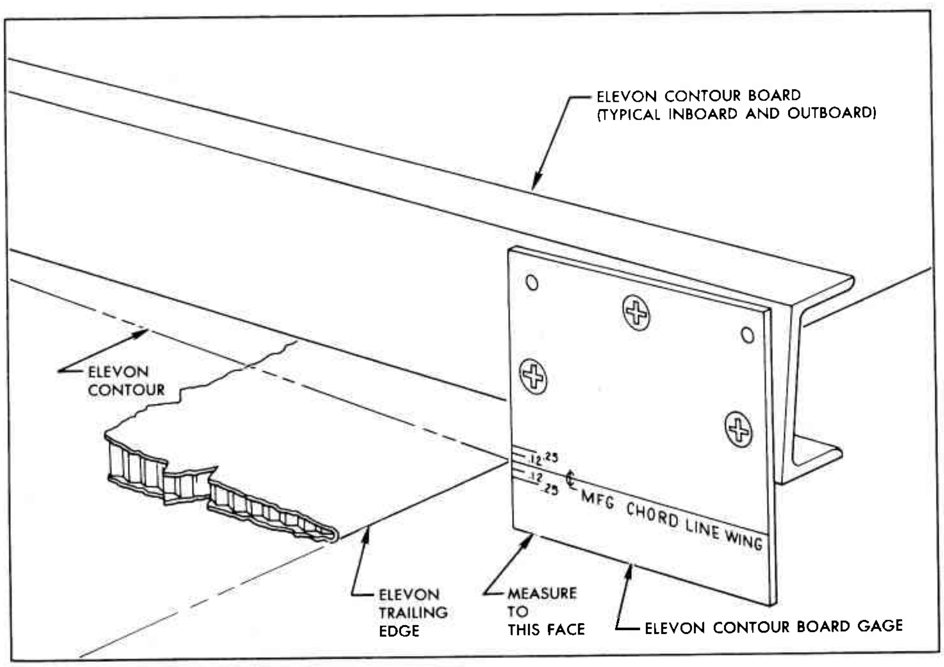
Figure 2-47. Elevon Contour Boards (Sheet 1 of 2)



VIEW A



VIEW B

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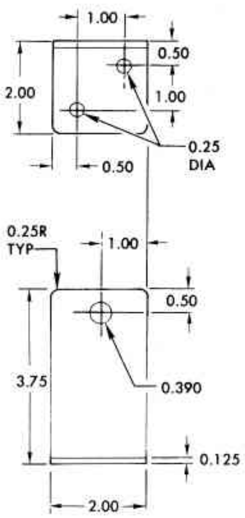
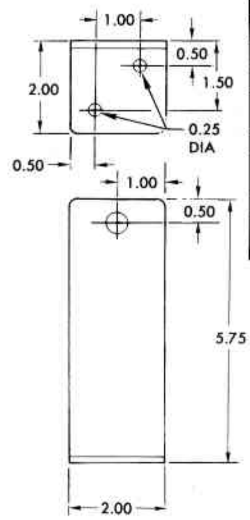
Figure 2-47. Elevon Contour Boards (Sheet 2 of 2)

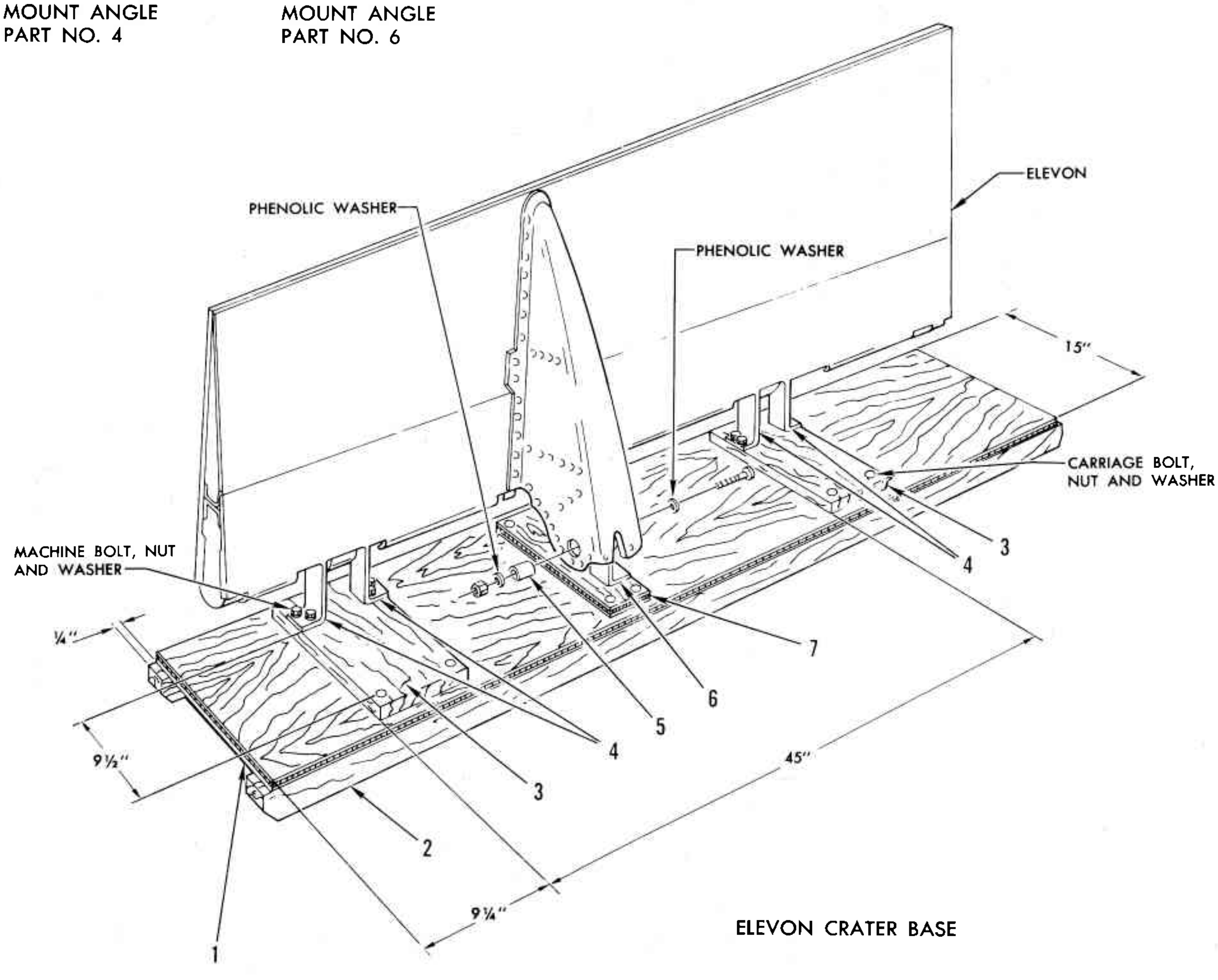
TABLE I				
NAME	DESCRIPTION	TYPE	PART NO.	NO. REQUIRED
FLOOR	¼ x 15 x 83	PLYWOOD	1	1
SKID	1¾ x 1¾ x 83½	LUMBER	2	2
LOAD BEARING MEMBER	¾ x 7½ x 13½	LUMBER	3	2
MOUNT ANGLE	0.125 x 2.00 x 8.00	STEEL	4	4
PHENOLIC BUSHING	$^{+0.750}_{-0.740}$ O.D. x 0.359I.D. x 1.00	PHENOLIC	5	3
MOUNT ANGLE	0.125 x 2.00 x 6.00	STEEL	6	1
LOAD BEARING MEMBER	½ x 4 x 13½	PLYWOOD	7	1



MOUNT ANGLE
PART NO. 4



MOUNT ANGLE
PART NO. 6



ELEVON CRATER BASE

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Figure 2-48. Packing and Crating — Elevon (Sheet 1 of 2)

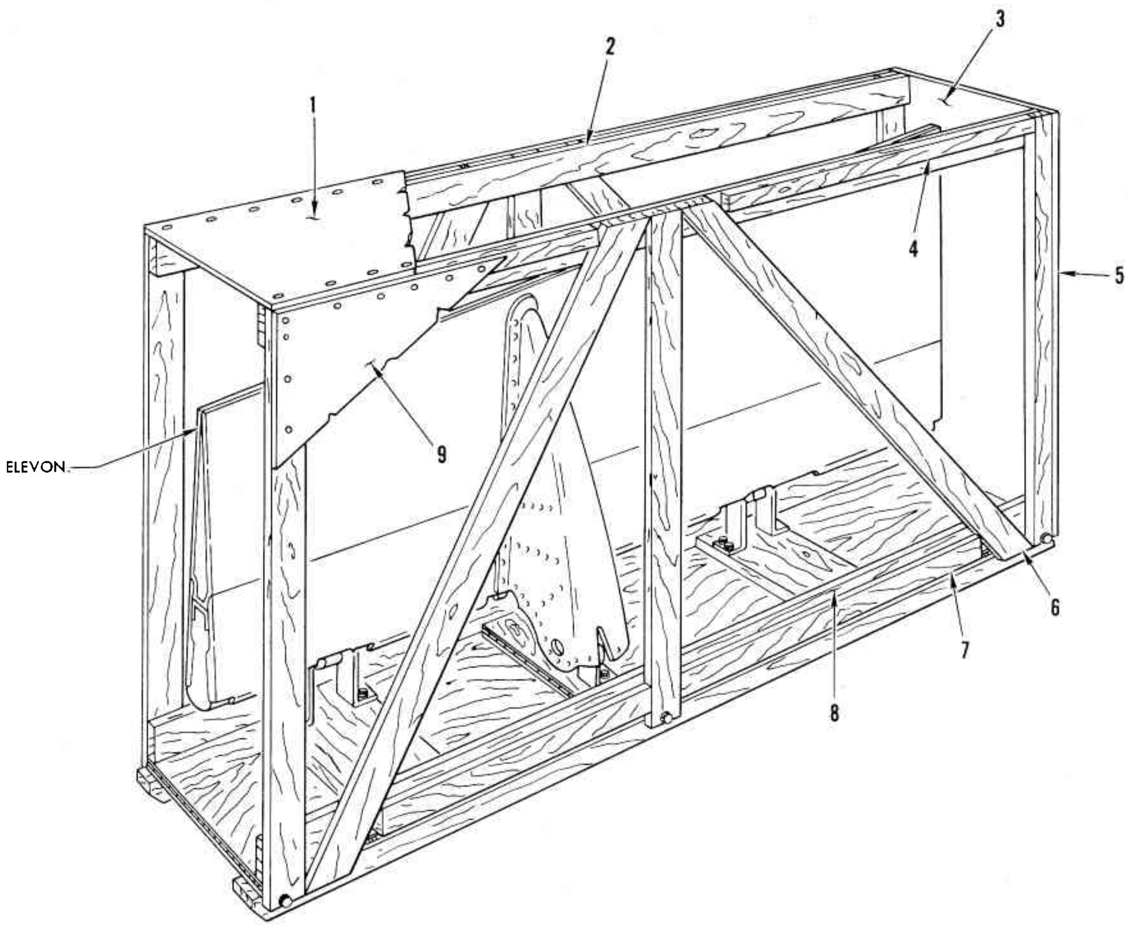

TABLE II				
NAME	DESCRIPTION	TYPE	PART NO.	NO. REQUIRED
SHEATHING	¼ x 16½ x 83½	PLYWOOD	1	1
LONGERON CLEAT	¾ x 3⅝ x 83	LUMBER	2	2
SHEATHING	¼ x 16½ x 38¼	PLYWOOD	3	2
FILLER CLEAT	¾ x 1¾ x 32	LUMBER	4	4
VERTICAL CLEAT	¾ x 2⅝ x 39¾	LUMBER	5	6
DIAGONAL CLEAT	¾ x 2⅝ x 55¼	LUMBER	6	4
FILLER CLEAT	¾ x 1¾ x 31	LUMBER	7	4
LONGERON CLEAT	¾ x 2⅝ x 83	LUMBER	8	2
SHEATHING	⅛ x 38¼ x 83½	PAPER-OVERLAID VENEER	9	2

TABLE III			
NAME	SIZE	TYPE	NO. REQUIRED
NUT	AN365-624	STEEL	2
WASHER	0.375	PHENOLIC	8
LAG SCREW	0.312 x 2.00	STEEL	6
WASHER	0.312	STEEL	6
CARRIAGE BOLT	0.312 x 3.00	STEEL	12
MACHINE BOLT	0.312 x 1.50	STEEL	10
WASHER	0.250	STEEL	22
NUT	0.250	STEEL	22
BOLT	AN6-20	STEEL	2
MACHINE BOLT	0.375 x 3.50	STEEL	1
NUT	0.375	STEEL	1

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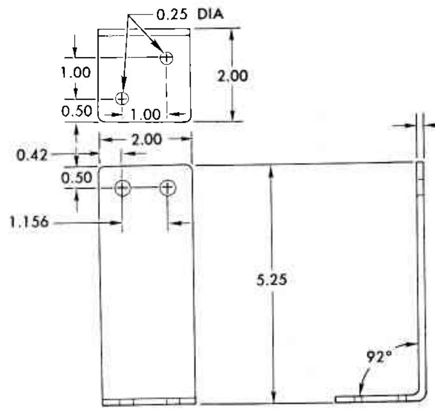
Figure 2-48. Packing and Crating — Elevon (Sheet 2 of 2)



MOUNT ANGLE
PART NO. 1

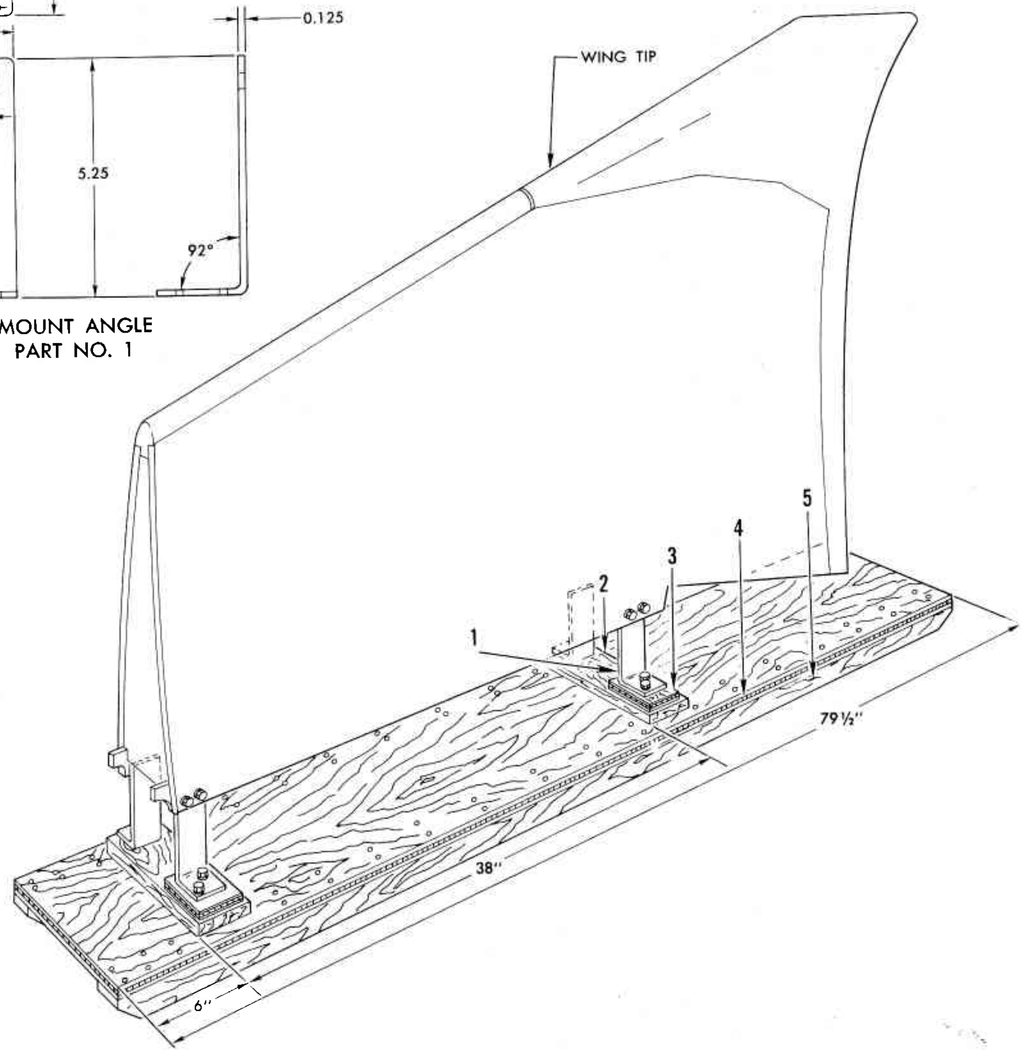


TABLE I				
NAME	DESCRIPTION	TYPE	PART NO.	NO. REQUIRED
MOUNT ANGLE	0.125 x 2.00 x 7.375	STEEL	1	4
LOAD BEARING MEMBER	¾ x 3⅝ x 9½	LUMBER	2	2
BLOCK	5/16 x 3 x 3⅝	PLYWOOD	3	2
FLOOR	¼ x 11 x 79	PLYWOOD	4	1
SKID	1¾ x 1¾ x 79½	LUMBER	5	2

06.03.255-1B

Figure 2-49. Packing and Crating — Wing Tip (Sheet 1 of 2)

TABLE II				
NAME	DESCRIPTION	TYPE	PART NO.	NO. REQUIRED
SHEATHING	¼ x 12½ x 79½	PLYWOOD	1	1
LONGERON CLEAT	¾ x 3⅝ x 79	LUMBER	2	2
SHEATHING	⅛ x 46½ x 79	PAPER-OVERLAID VENEER	3	2
VERTICAL CLEAT	¾ x 2⅝ x 48½	LUMBER	4	6
DIAGONAL CLEAT	¾ x 2⅝ x 60	LUMBER	5	4
FILLER CLEAT	¾ x 1¾ x 30	LUMBER	6	4
LONGERON CLEAT	¾ x 2⅝ x 79	LUMBER	7	2
CONTOUR BOARD	2 x 6 x 18	LUMBER	8	1
CONTOUR BOARD	2 x 8 x 18	LUMBER	9	1
FILLER CLEAT	¾ x 3⅝ x 31	LUMBER	10	4
SHEATHING	¼ x 12½ x 46½	PLYWOOD	11	2

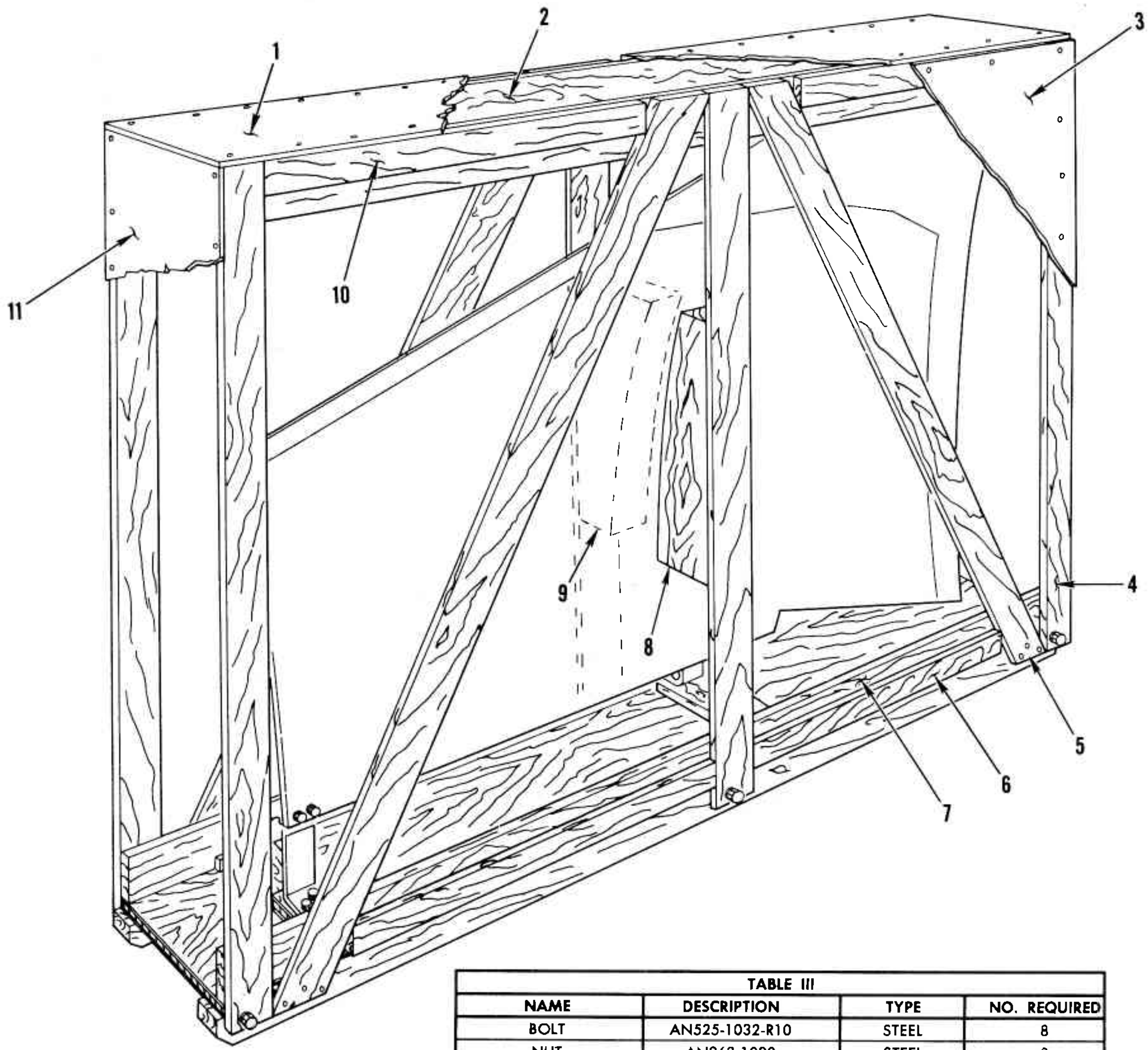


TABLE III			
NAME	DESCRIPTION	TYPE	NO. REQUIRED
BOLT	AN525-1032-R10	STEEL	8
NUT	AN363-1032	STEEL	8
WASHER	AN960-10	STEEL	8
LAG SCREW	0.312 x 2.00	STEEL	6
WASHER	0.312	STEEL	6
CARRIAGE BOLT	0.250 x 3.00	STEEL	4
MACHINE BOLT	0.250 x 2.00	STEEL	14
WASHER	0.250	STEEL	10
NUT	0.250	STEEL	10
MACHINE BOLT	0.250 x 1.50	STEEL	2

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Figure 2-49. Packing and Crating — Wing Tip (Sheet 2 of 2)

 ALL DATA ON PAGES 2-99 THRU 2-101 INCLUDING FIGURE 2-50 DELETED.

2-86. PREPARATION OF WING FOR AIRLIFT.

2-87. Preparation of a wing for airlift requires three major operations. The wing must be defueled and purged, removed from the airplane, and installed on a shipping stand. Since JP-4 is both toxic and flammable, it is of the utmost importance that all fumes be removed from the wing before airlift operation.

a. Defuel airplane in accordance with T.O. 1F-106A-2-5-2-1.

b. Purge wing with 100-octane gasoline. Refer to paragraph 2-41 for procedure.

c. Remove wing tip and leading edge. Refer to T.O. 1F-106A-2-2-2-2 for procedure.

d. Package all removed parts and components, other than elevon and main landing gear, in accordance with instructions contained in T. O. 00-85-16, or stow in boxes provided with the shipping stand for shipment with the wing. This includes all access doors and attaching screws.

e. Remove wing from fuselage. See figure 2-7 for procedure.

f. Cap all hydraulic and fuel lines; stow or remove all loose lines and electrical connectors.

g. Place wing on dolly or mattresses.

NOTE

The right wing must be inverted (turned upside down) prior to installation on shipping stand. The No. 2 and No. 6 spar adapters, provided with the shipping stand, can be used to invert the right wing. Shipping stand shown in Figure 2-51.

h. Attach the No. 2 spar adapter, AF Dwg. 58SAC589, to No. 2 spar and secure with lock pins.

i. Attach the No. 6 spar adapter, AF Dwg. 58SAC588, to No. 6 spar and secure with lock pins.

j. Attach the universal lifting eye, provided with the shipping stand, to the wing lift point. If the left wing is to be air shipped, attach universal lifting eye to the wing tiedown point. See figure 2-51.

k. Attach hoisting sling, AF Dwg. 58SAB591, to spar adapters and lifting eye. Attach two 15-foot lengths of rope, as shown in figure 2-51, to guide wing into place on shipping stand.

l. Hoist wing until No. 6 spar adapter clears the curved pivot hook at the top of the shipping stand. Lower and guide the wing until the pivot hook enters the 1½-inch diameter hole in the No. 6 spar adapter. Continued lowering will cause the wing to pivot into position on the shipping stand. See figure 2-51.

m. Bolt No. 6 spar adapter to shipping stand with four AN 12-24A bolts, four AN 960-1216 washers, and four AN 365-1216 nuts. Secure the No. 2 spar adapter to the shipping stand using four AN 960-1216 washers and four AN 365-1216 nuts.

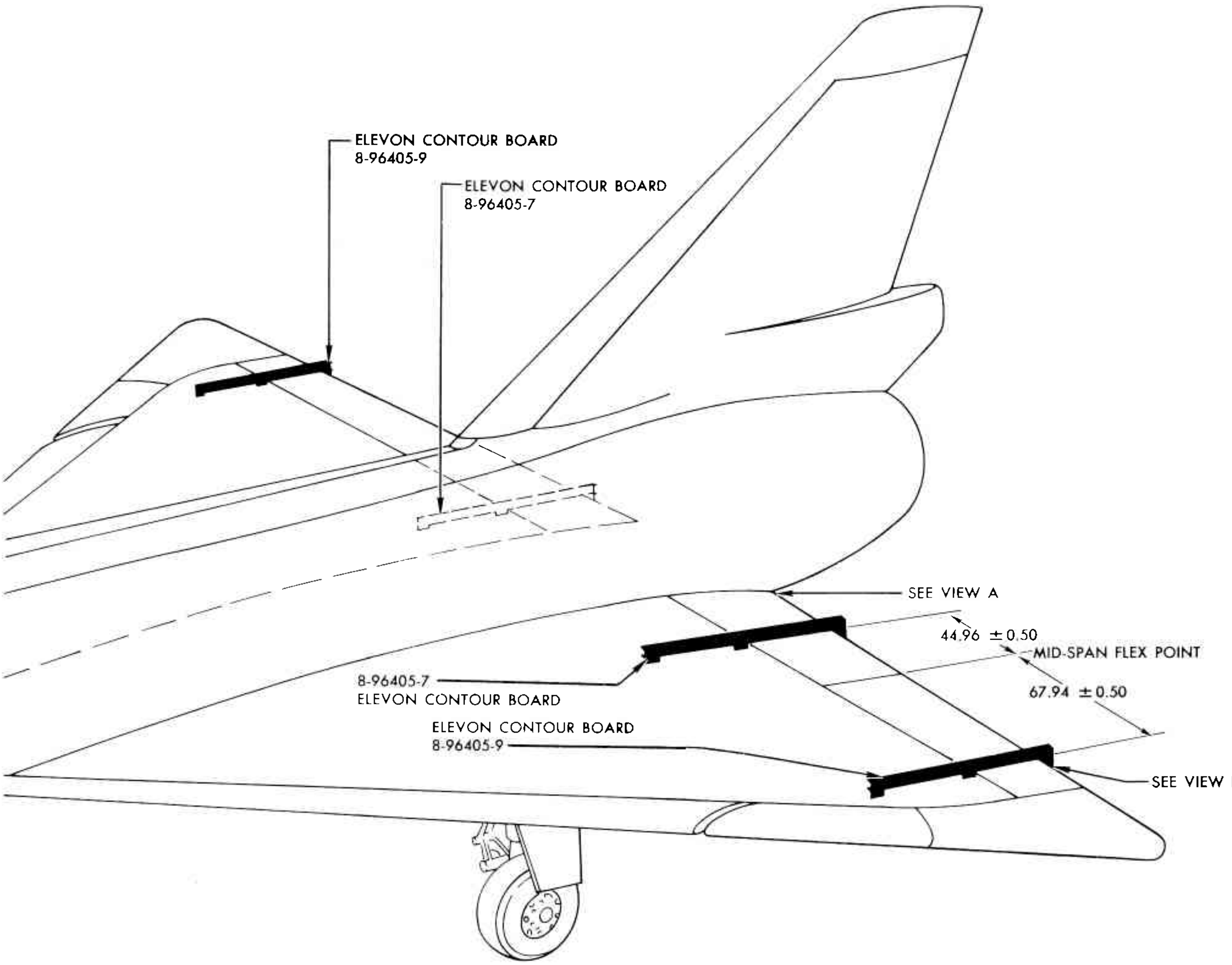
n. Attach the wing support pad to the elevon attach fitting as shown on figure 2-51 and adjust to insure that the wing clears the shipping stand.

NOTE

The canted position of the wing, when installed on the shipping stand, may cause trapped fuel to drain into the lower portion of the tanks. Since all residual fuel must be removed prior to airlift, the wing should be placed on the shipping stand at least eight hours prior to airlift to allow for fuel drainage and subsequent mopping. Should the shipping stand not be available prior to arrival of transport aircraft, the intent of the foregoing shall be accomplished by otherwise devising a means of propping the wing in a canted position.

2-88. ELEVON ALIGNMENT.

2-89. Figure 2-50 shows the elevon contour boards used for checking elevon alignment.



CONTOUR BOARD ALIGNMENT CHECK PROCEDURE:

a. Prepare airplane as required for elevon operation procedure given in T.O. 1F-106A-2-7-2-1.
b. Trim elevons to neutral (streamline position). Elevons are in neutral when inboard tips are in line with positioning rivets on fuselage (see view A) under the following conditions: hydraulic power on, engine not installed, no fuel in tanks and elevator and aileron mixer rig pin holes aligned.
c. **Position the 8-96405-7 contour board 44.96 (±0.50) inches inboard from the centerline of the mid-span flex point of both elevons.**

NOTE
WITH ENGINE INSTALLED AND FUEL IN TANKS, TRAILING EDGE MAY DROOP ONE-EIGHTH INCH BELOW CENTER OF POSITIONING RIVET.

d. **Position the 8-96405-9 contour board 67.94 (±0.50) inches outboard from the centerline of the mid-span flex point of both elevons.**

NOTE
CONTOUR BOARDS MUST BE PLACED PARALLEL TO LINE OF FLIGHT.

e. Check and record deviation from wing chord line as calibrated on plate at aft end of each contour board (see view B). Total deviation between inboard and outboard contour board readings should not exceed ±0.25 inch on either wing.

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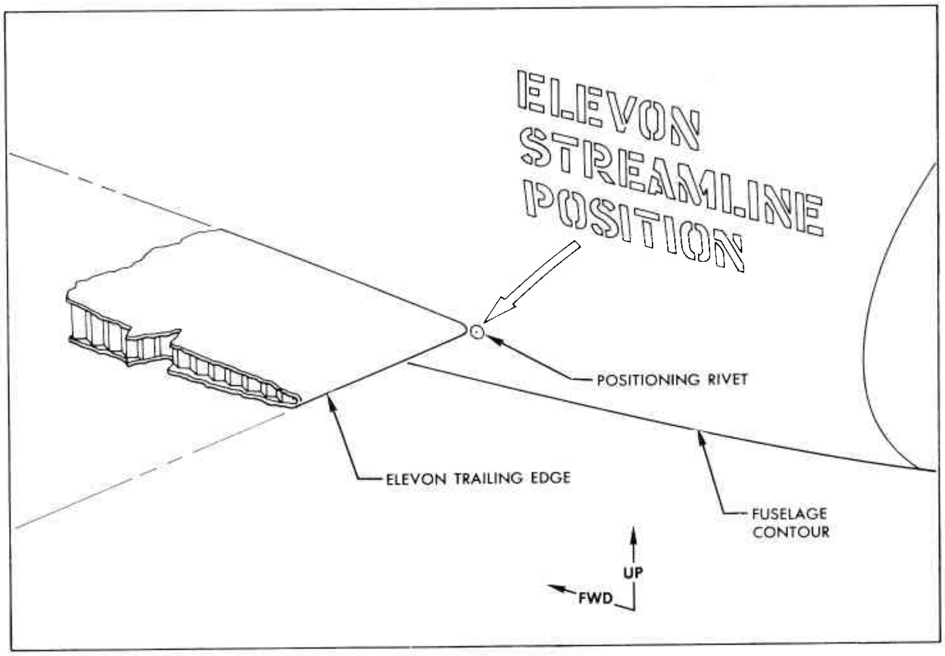
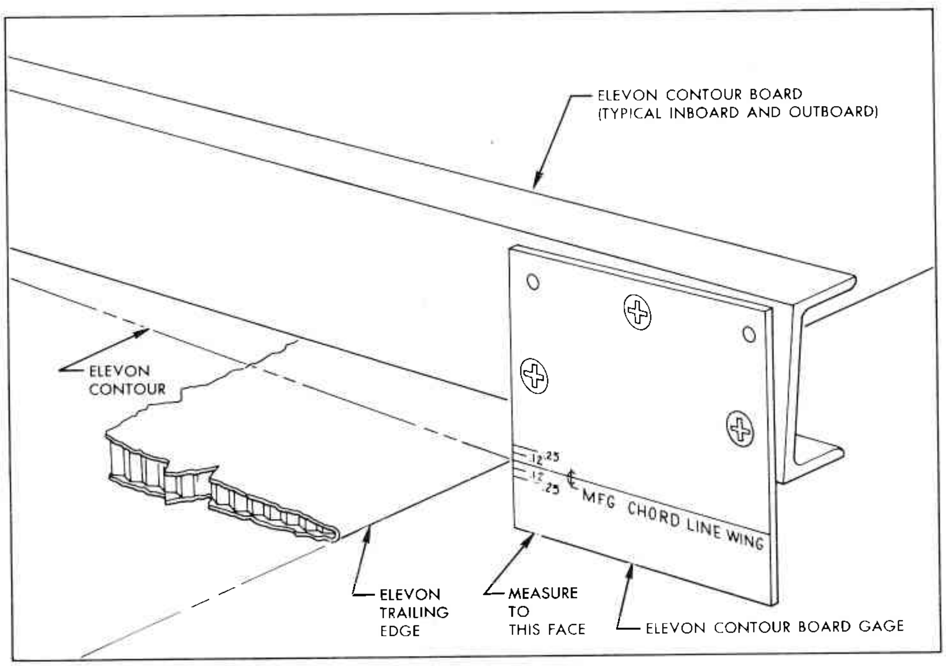
Figure 2-50. Elevon Contour Boards (Sheet 1 of 2)



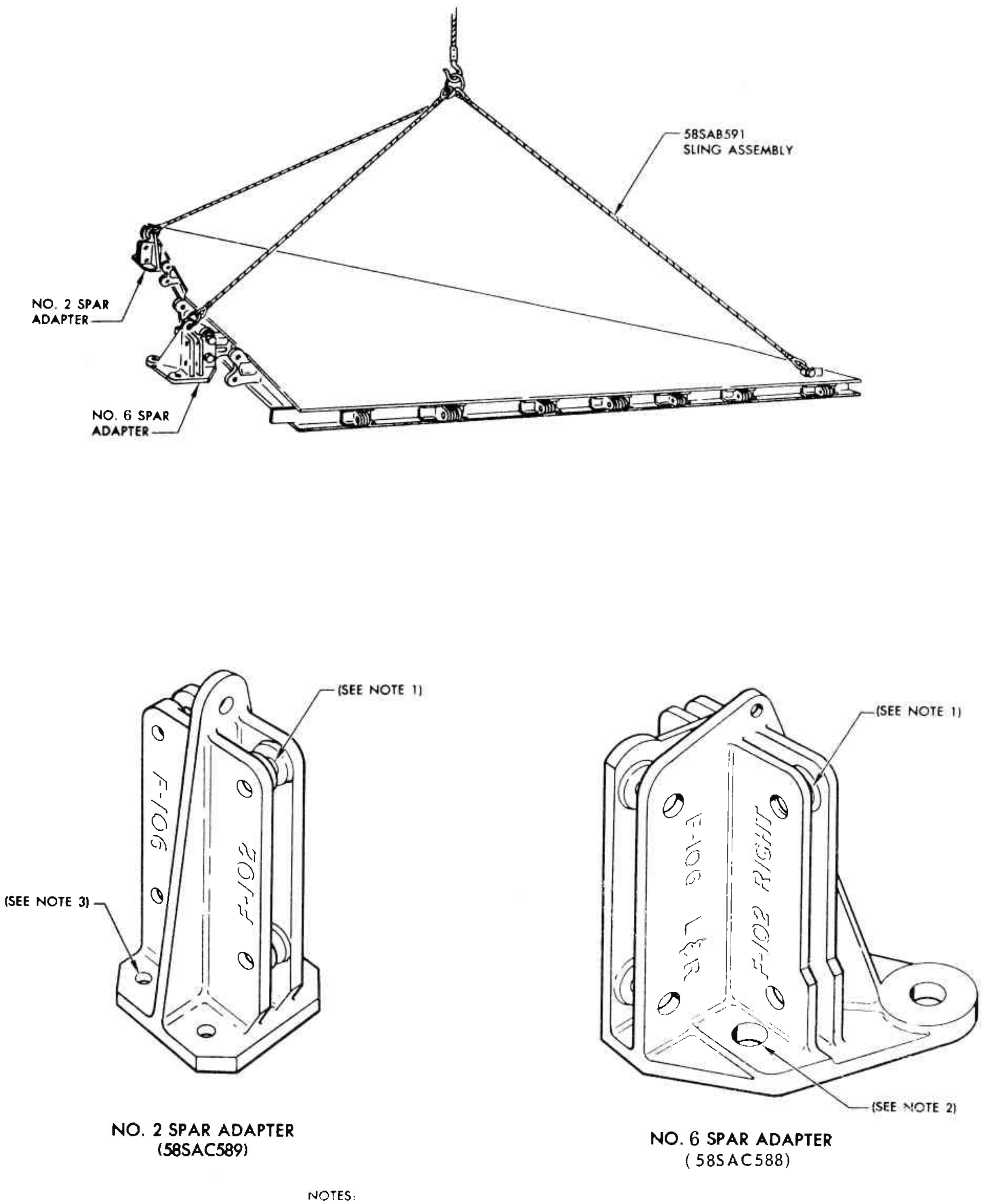
VIEW A



VIEW B

06.03.243-2 .57.04.01

Figure 2-50. Elevon Contour Boards (Sheet 2 of 2)



NOTES:
1. COTTER PINS MUST BE INSTALLED IN ALL ATTACHING PINS.
2. USE AN315-12R NUTS (4) AN960-1216 WASHERS (4) AND AN12-24A BOLTS (4) TO ATTACH WING TO STAND.
3. USE AN315-12R NUTS (4) AND AN960-1216 WASHERS (4) TO ATTACH WING TO STAND.

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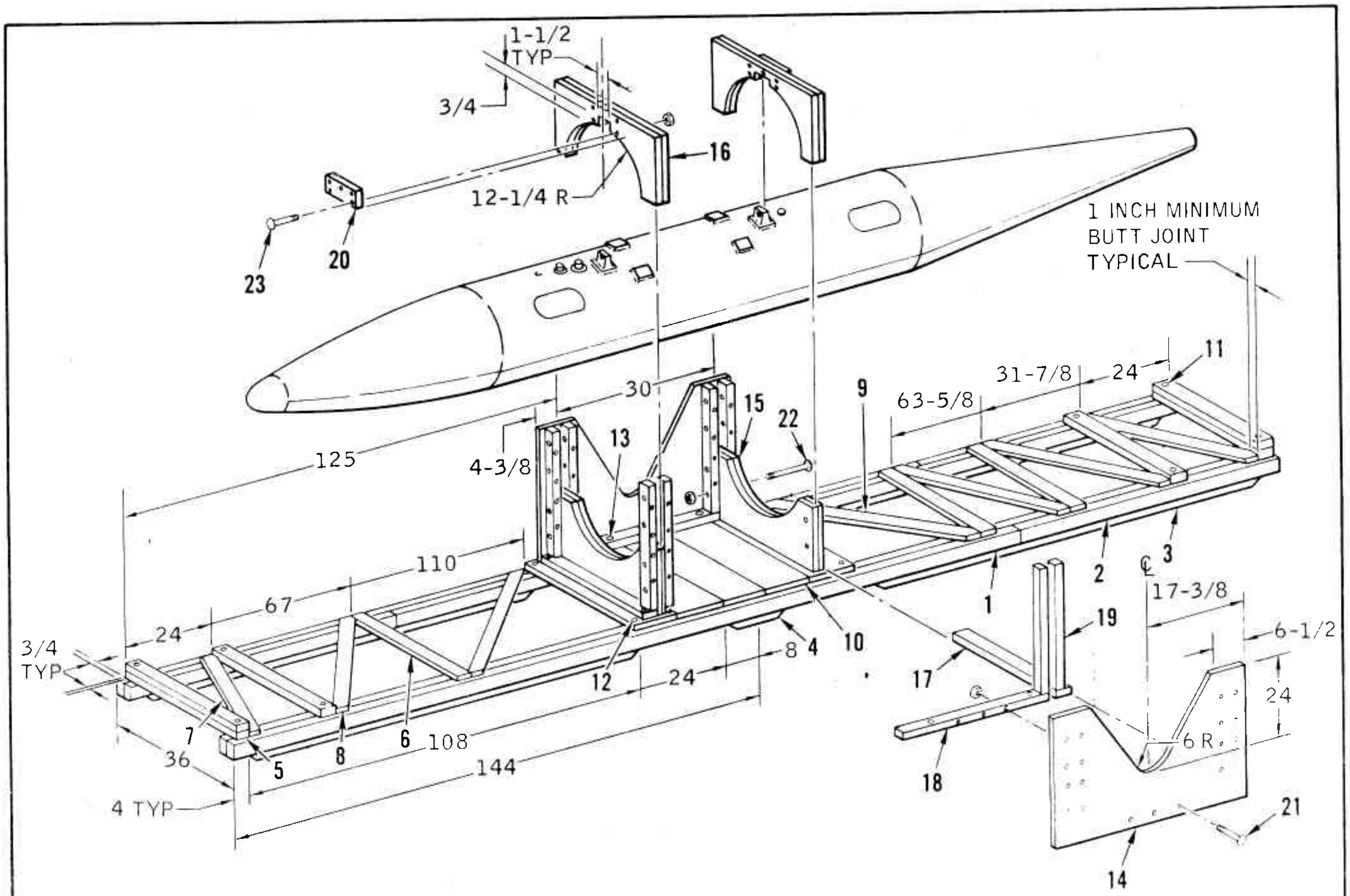
Figure 2-51. Preparation of Wing Airlift (Sheet 1 of 3)

 All data on page 2-102A and 2-102B, including figures 2-50A and 2-50B are deleted.



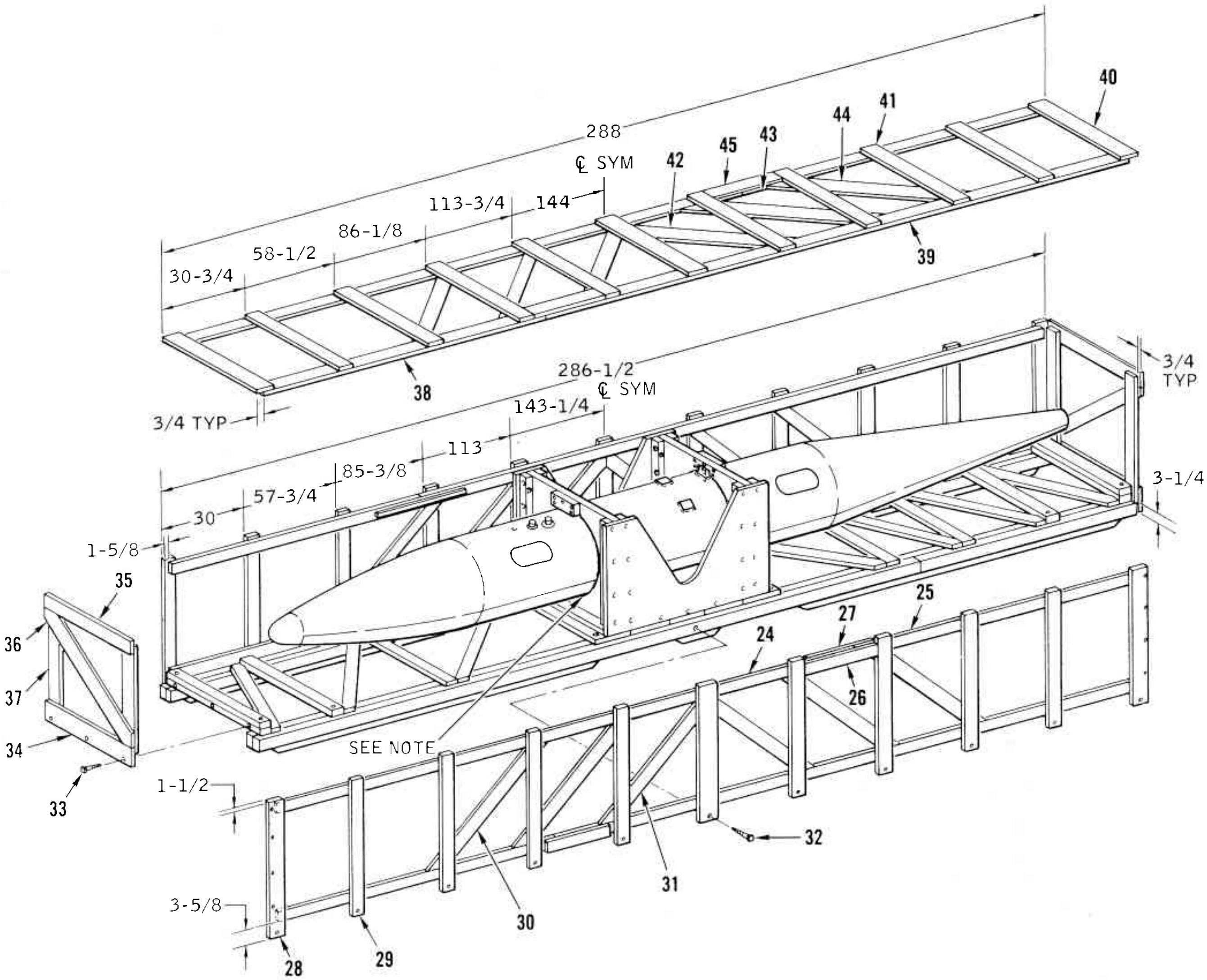
MATERIAL TABLE				
NAME	DESCRIPTION	TYPE OF MATERIAL	PART NO.	NO. REQUIRED
SKID	2 x 4 x 192	LUMBER	1	4
SKID	2 x 4 x 96	LUMBER	2	4
RUBBING STRIP	2 x 4 x 108	LUMBER	3	4
RUBBING STRIP	2 x 4 x 16	LUMBER	4	2
HEADER & FLOOR MEMBER	2 x 4 x 34-1/2	LUMBER	5	6
FLOOR MEMBER	1 x 4 x 34-1/2	LUMBER	6	3
DIAGONAL	1 x 4 x 40	LUMBER	7	2
DIAGONAL	1 x 4 x 52-3/8	LUMBER	8	2
DIAGONAL	1 x 4 x 44-1/2	LUMBER	9	3
LOAD BEARING MEMBER	2 x 12 x 34-1/2	LUMBER	10	5
CARRIAGE BOLT	3/8 x 7-1/2	STEEL	11	12
CARRIAGE BOLT	3/8 x 6	STEEL	12	4
CARRIAGE BOLT	3/8 x 9	STEEL	13	16
GUSSET	3/8 x 35 x 34-3/4	PLYWOOD	14	2
CRADLE	3/4 x 16 x 32-3/4	PLYWOOD	15	4
CRADLE	3/4 x 19 x 32-3/4	PLYWOOD	16	4
LAG STRIP (LATITUDE)	2 x 4 x 33	LUMBER	17	2
LAG STRIP (LONGITUDE)	2 x 2 x 28-1/2	LUMBER	18	2
UPRIGHT	2 x 2 x 33-3/8	LUMBER	19	8
END PAD	3/4 x 5-3/4 x 10	PLYWOOD	20	2
CARRIAGE BOLT	1/4 x 3	STEEL	21	38
CARRIAGE BOLT	1/4 x 5	STEEL	22	16
CARRIAGE BOLT	1/4 x 2-1/2	STEEL	23	10

NOTE: PAD ALL POINTS OF CONTACT WITH TANK USING CELLULOSE OR RUBBER CUSHIONING.

65R-958-4-1

Figure 2-50A. Packing and Crating-Drop Tank (Sheet 1 of 2)

Applicable after incorporation of TCTO 1F-106-958



MATERIAL TABLE (CONTINUED)				
NAME	DESCRIPTION	TYPE OF MATERIAL	PART NO.	NO. REQUIRED
LONGERON CLEAT	1 x 6 x 192	LUMBER	24	4
LONGERON CLEAT	1 x 6 x 96-1/4	LUMBER	25	4
FACE SPLICE PLATE	1 x 6 x 24	LUMBER	26	4
REAR SPLICE PLATE	1 x 4 x 24	LUMBER	27	4
CENTER & END STRUT	1 x 6 x 42-3/4	LUMBER	28	6
INTERMEDIATE STRUT	1 x 4 x 42-3/4	LUMBER	29	16
DIAGONAL	1 x 6 x 50	LUMBER	30	8
DIAGONAL	1 x 6 x 44-5/8	LUMBER	31	4
LAG SCREW	3/8 x 4	STEEL	32	22
LAG SCREW	5/16 x 3	STEEL	33	11
HORIZONTAL FRAME	1 x 6 x 37-1/2	LUMBER	34	2
HORIZONTAL FRAME	1 x 4 x 37-1/2	LUMBER	35	2
DIAGONAL	1 x 6 x 48	LUMBER	36	2
VERTICAL CLEAT	2 x 4 x 34-3/8	LUMBER	37	4
LONGERON CLEAT	1 x 6 x 192	LUMBER	38	2
LONGERON CLEAT	1 x 6 x 94-1/2	LUMBER	39	2
CENTER & END STRUT	1 x 6 x 36	LUMBER	40	3
INTERMEDIATE STRUT	1 x 4 x 36	LUMBER	41	8
DIAGONAL	1 x 6 x 43-1/2	LUMBER	42	2
DIAGONAL	1 x 6 x 49-1/2	LUMBER	43	4
SPLICE CLEAT	1 x 4 x 24	LUMBER	44	2
SPLICE CLEAT	1 x 6 x 24	LUMBER	45	2

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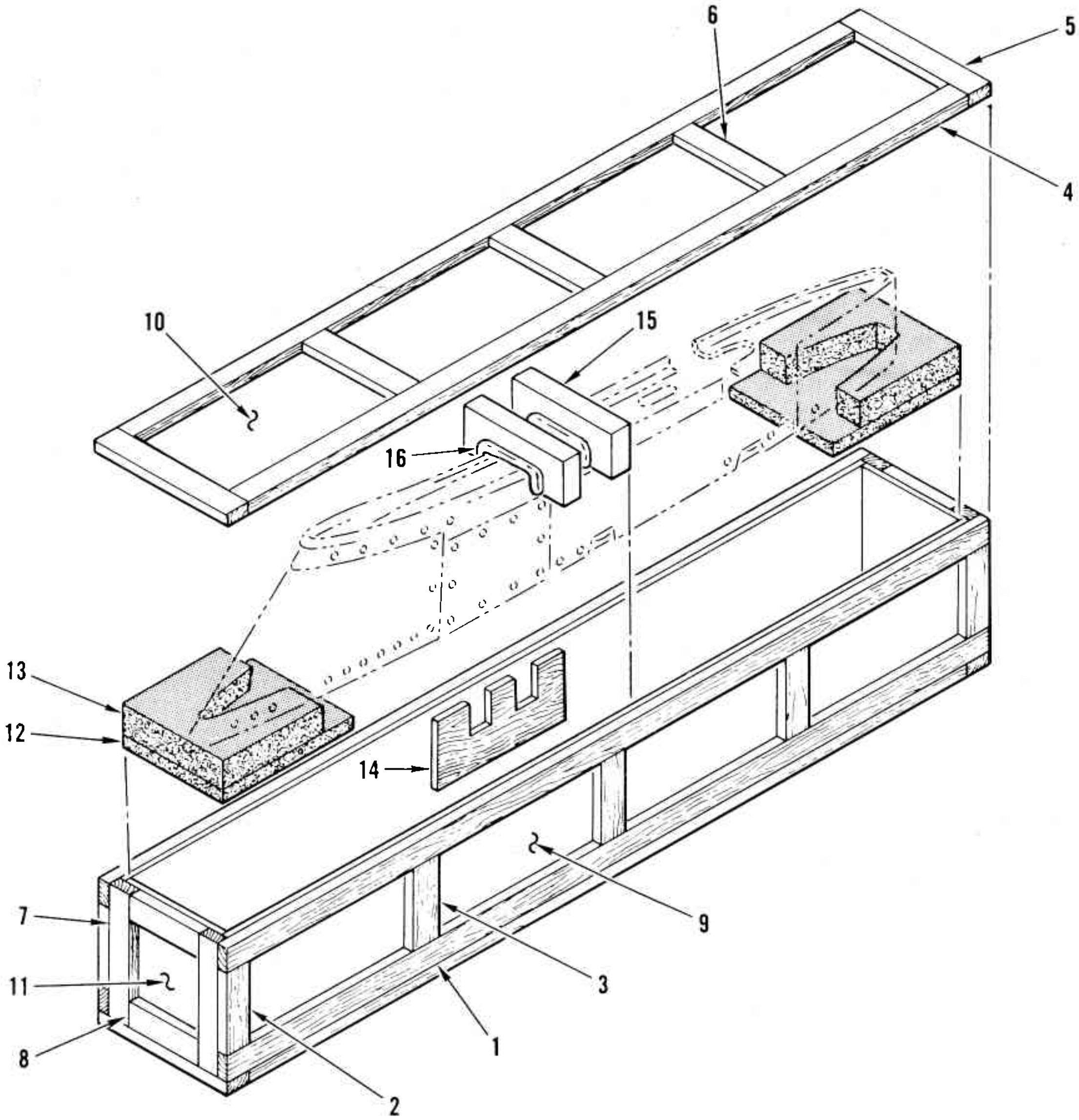
Figure 2-50A. Packing and Crating-Drop Tank (Sheet 2 of 2)
Applicable after incorporation of TCTO 1F-106-958.



MATERIAL TABLE				
NAME	DESCRIPTION	TYPE OF MATERIAL	PART NO.	NO. REQUIRED
LONGERON CLEAT	1 x 2 x 82	LUMBER	1	4
FILLER CLEAT	1 x 2 x 11-1/4	LUMBER	2	4
INTERMEDIATE CLEAT	1 x 2 x 11-1/4	LUMBER	3	6
LONGERON CLEAT	1 x 2 x 78-3/4	LUMBER	4	4
HORIZONTAL CLEAT	1 x 2 x 14	LUMBER	5	4
FILLER CLEAT	1 x 2 x 10-3/4	LUMBER	6	6
VERTICAL CLEAT	1 x 2 x 12-1/2	LUMBER	7	4
FILLER CLEAT	1 x 2 x 8-3/4	LUMBER	8	4
SHEATHING	1/8 x 82 x 12-1/2	PAPER-OVERLAID VENEER	9	2
SHEATHING	1/8 x 82 x 14	PAPER-OVERLAID VENEER	10	2
SHEATHING	1/8 x 12 x 12	PAPER-OVERLAID VENEER	11	2
FILLER	1 x 8-3/4 x 14	CANE AND WOOD FIVER	12	2
CONTOUR FILLER	2 x 8-3/4 x 12	CANE AND WOOD FIBER	13	2
SOCKET	3/4 x 12 x 6	PLYWOOD	14	2
RETAINER	2 x 4 x 12	LUMBER	15	2
WADDING	UU-C-843	CELLULOSE	16	AS REQUIRED

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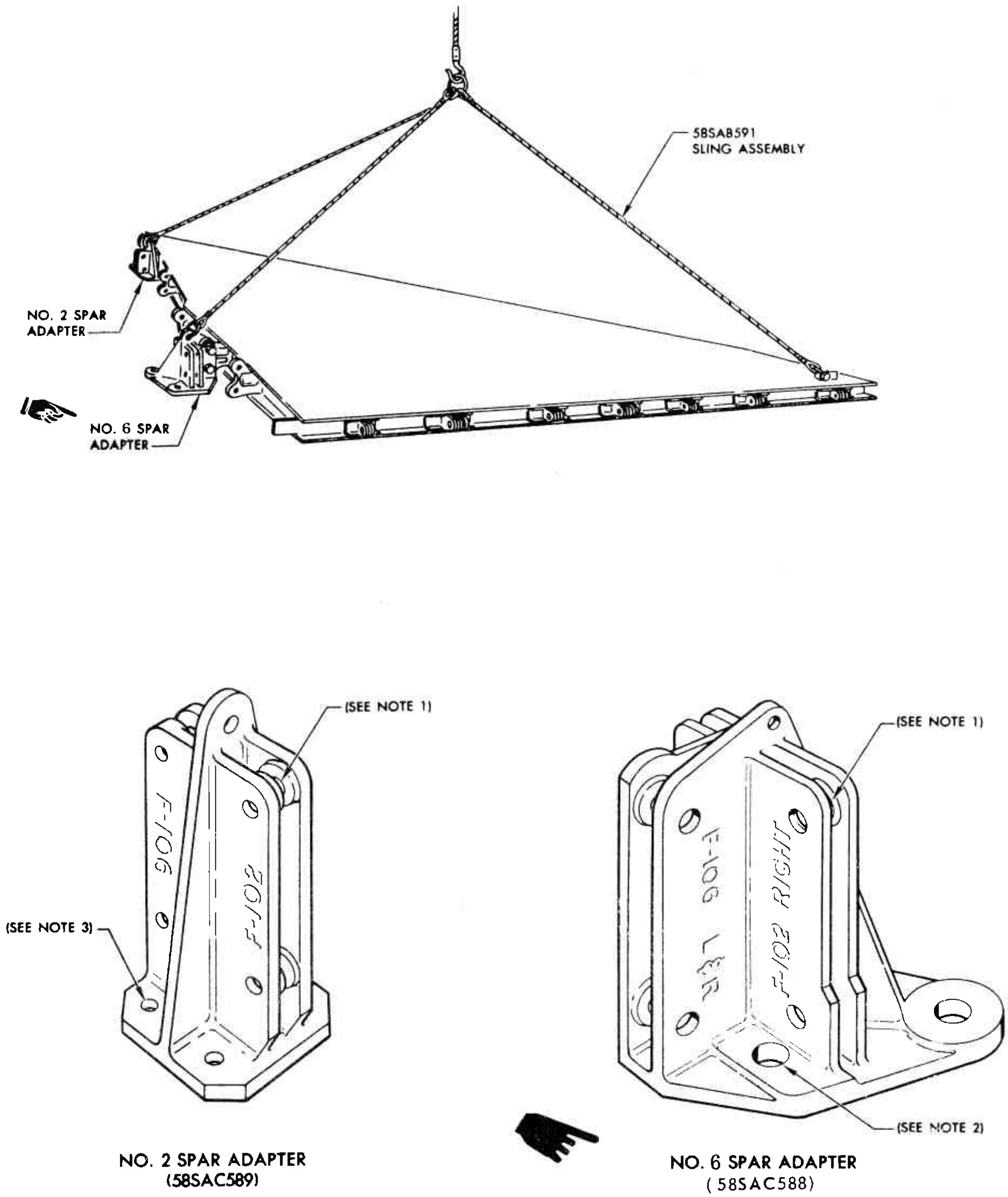
Figure 2-50B. Packing and Crating-External Tank Pylon

Applicable after incorporation of TCTO 1F-106-958



NOTES:
1. COTTER PINS MUST BE INSTALLED IN ALL ATTACHING PINS.
2. USE AN315-12R NUTS (4) AN960-1216 WASHERS (4) AND AN12-24A BOLTS (4) TO ATTACH WING TO STAND.
3. USE AN315-12R NUTS (4) AND AN960-1216 WASHERS (4) TO ATTACH WING TO STAND.

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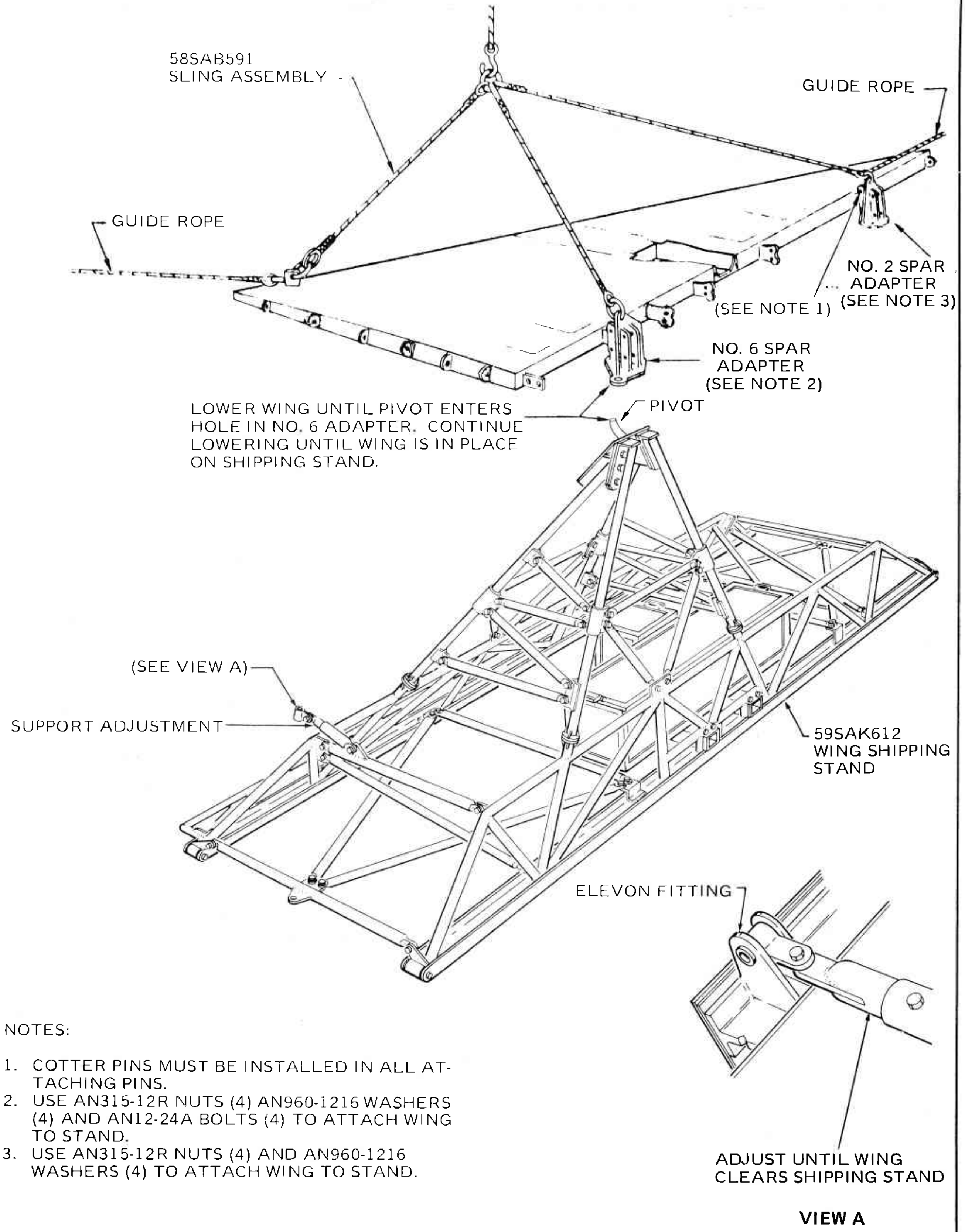
Figure 2-51. Preparation of Wing Airlift (Sheet 1 of 3)



NOTES:

1. COTTER PINS MUST BE INSTALLED IN ALL ATTACHING PINS.
2. USE AN315-12R NUTS (4) AN960-1216 WASHERS (4) AND AN12-24A BOLTS (4) TO ATTACH WING TO STAND.
3. USE AN315-12R NUTS (4) AND AN960-1216 WASHERS (4) TO ATTACH WING TO STAND.

Figure 2-51. Preparation of Wing Airlift (Sheet 2 of 3)

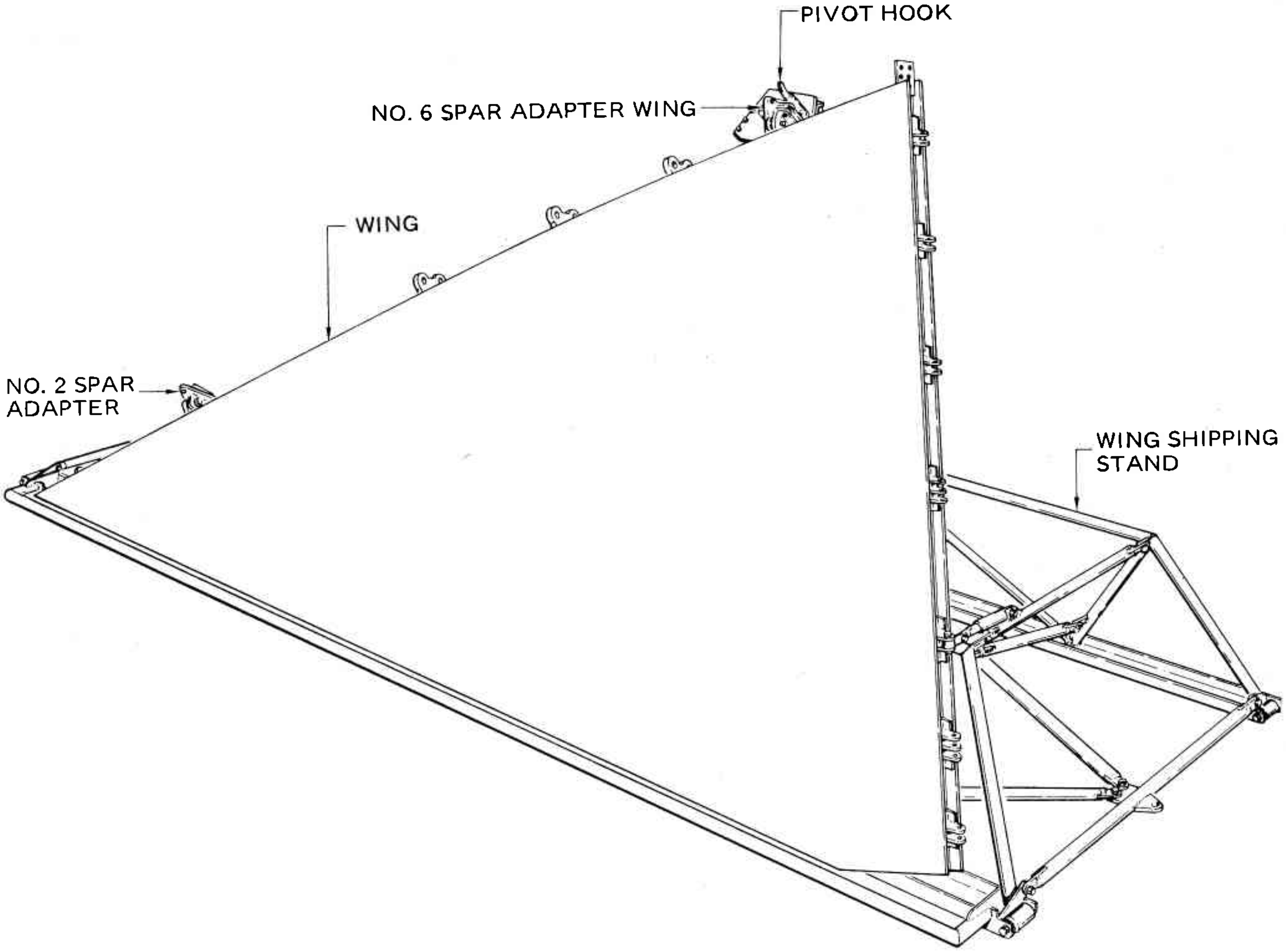


Figure 2-51. Preparation of Wing Airlift (Sheet 3 of 3)

TABLE 2-I

Negligible Damage Limits — Wing Group

COMPONENT	TYPE AND CLASS OF DAMAGE ALLOWED AFTER REWORK					REMARKS
	Scratch	Nick	Dent	Hole	Crack	
WING STRUCTURE						
Plating—Upper	I	I	I	*	*	
Plating—Lower	I	I	I	*	*	
Leading Edge Spar Rail	I	I	*	I	*	
Leading Edge Spar Web	II	II	I	I	*	
Spar Attachment Fittings, Nos. 1 thru 6	I	I	*	*	*	
Spars, Nos. 1 thru 6 Flange	I	I	*	I	*	
Spars, Nos. 1 thru 6 Web	II	II	I	I	*	
Bulkhead Wing Fuselage Intersection	II	II	II	II	II	
Rib Rails	II	II	II	II	I	
Rib Webs	III	III	III	III	II	
Rib Stiffeners	III	III	III	III	II	
Fuel-Tight Corners	III	III	III	*	*	
Drag Angles	II	II	II	*	*	
LEADING EDGE STRUCTURE						
Plating	II	II	I	*	*	
Doublers	II	II	I	I	I	
Ribs	III	III	III	III	II	
Attachment Angles	II	II	II	II	I	
TRAILING EDGE AND TRIM TANK STRUCTURE						
Spar No. 7 Flange	I	I	*	I	*	
Spar No. 7 Web	I	II	I	*	*	
Spar No. 7 Attachment	I	I	*	*	*	
Plating	I	I	I	*	*	
Integral Stiffeners	I	I	*	*	*	
Rib Rails	I	I	*	*	*	
Truss Members	I	I	*	*	*	
Transverse Stiffeners B.L. 170.00 to 188.00	I	I	*	*	*	
Hinge Fittings		Refer to an Aeronautical Structural Engineer				
Actuator Fitting		Refer to an Aeronautical Structural Engineer				
Actuator Fairing Skin	III	III	III	II	II	

*Component must be repaired or replaced.

TABLE 2-1

Negligible Damage Limits — Wing Group (Cont)

COMPONENT	TYPE AND CLASS OF DAMAGE ALLOWED AFTER REWORK					REMARKS
	Scratch	Nick	Dent	Hole	Crack	
ELEVON STRUCTURE—INBOARD AND OUTBOARD						
Plating	I	I	I	*	*	
Spar Assembly Rail	I	I	*	I	*	
Spar Assembly Web	II	II	I	I	I	
Ribs	III	III	II	II	I	
Hinge Fittings		Refer to an Aeronautical Structural Engineer				
Outboard Horn Fitting		Refer to an Aeronautical Structural Engineer				
Actuator Fitting		Refer to an Aeronautical Structural Engineer				
Actuator Fairing	III	III	III	II	II	
Trailing Edge Wedges	II	II	*	*	I	
WING TIP STRUCTURE—BUILT-UP TYPE						
Spar	I	I	I	*	*	
Plating	I	I	I	*	*	
Ribs	II	II	II	II	II	
Leading Edge Slug	II	II	II	II	I	
Trailing Edge Slug	II	II	II	II	I	
EXTERNAL FUEL TANK STRUCTURE						
Skin	II	II	III	Refer to an Aeronautical Structural Engineer		
Fairing	II	II	III	Refer to an Aeronautical Structural Engineer		
Pylon Ribs	II	II	I	II	II	
Pylon Post	II	II	I	II	II	
Pylon Angles	II	II	I	II	II	
Rails	II	II	I	Refer to an Aeronautical Structural Engineer		
Rings	II	II	Refer to an Aeronautical Structural Engineer			

*Component must be repaired or replaced.

Section III

TAIL GROUP

3-1. TAIL GROUP.

3-2. The basic tail group structure consists of a full cantilever vertical fin with provisions for the attachment of a fin tip, leading edge, and rudder. The fin assembly is an integral part of the fuselage structure and cannot be detached for repair or replacement without extensive separation of the tail and fuselage structure. Figure 3-1 shows the tail group components and incorporates a reference figure index. Figure 3-2 locates the stations in the fin and rudder structure.

3-3. Fin Interspar Structure.

3-4. The fin assembly consists of riveted high-strength aluminum alloy construction. The structure consists of a built-up leading edge spar and four forged-type spars. Figure 3-3 shows the details of the fin structure. The spars are interconnected by a series of chordwise shear-web ribs and attaching angle clips. The fin inner structure is enveloped by flush riveted aluminum honeycomb core sandwich panels. Additional reinforcement is provided by doublers bounded within the sandwich panel at points of attachment to the spars. Doors are incorporated in the fin plating on the left side to provide access for maintenance and inspection of the bay areas. Figure 3-4 shows the fin and rudder plating. Doors are located at the base of the structure on the right side to provide access for inspection and servicing of equipment within the fin.

3-5. Fin Leading Edge.

3-6. The fin leading edge structure consists of a series of press-formed ribs, reinforcing doublers, and angles as shown on figure 3-5. The structure is divided into four sections, including a "Q" intake section, and a stub section which mates with and attaches to the dorsal fairing. The outer plating is flush riveted to the ribs and angles. The leading edge sections are attached to the fin by screws through gang channel nuts which are riveted to the inner flange of the fin leading edge spar. The basic function of the leading edge is to maintain the airfoil in a given contour to meet aerodynamic requirements. The leading edge has no provisions for thermal anti-icing.

3-7. Fin Tip.

3-8. The fin tip structure is divided into an upper and lower section as shown on figure 3-6. The upper section consists of a fiberglass plastic honeycomb core preformed by routing-out a portion in the center to accommodate the Tacan antenna. The core is sandwiched between glass cloth laminates and bonded with plastic resins. The lower section is constructed of an aluminum honeycomb core sandwiched between aluminum outer skin panels. The panels are flush riveted to the internal structure of the lower section. The upper and lower sections are joined by screws installed through a plate nut channel attached to the inner flange of the lower section's chordwise, preformed channel. The fin tip leading edge spar, located in the lower section, consists of a built-up type "I" section. Gang channel nuts are riveted to the inside of the spar flanges to provide a means of attachment for the upper end of the vertical fin's leading edge. The fin tip is attached to the vertical fin structure with screws installed through a plate nut channel riveted to the inner flanges of a channel that forms the base of the fin tip's lower section. The lower fin tip section houses the IFF antenna.

3-9. Rudder.

3-10. The rudder structure consists basically of perforated aluminum honeycomb core sandwich construction. The details of rudder construction are shown on figure 3-3. The rudder leading edge consists of a press-formed channel spar fitted with three detachable forged hinged fittings and a cast actuator arm near the base of the spar. The upper and lower ends are fitted with channel-type ribs reinforced by doublers. The trailing edge consists of an extruded wedge bonded to the rudder plating. The rudder plating is straight-tapered from 0.072-inch gage at the bottom to 0.023-inch gage at the top. The rudder plating is attached with rivets at the perimeter of the structure and metal bonded in the honeycomb core area.

3-11. INDEXING.

3-12. Figure 3-1 is keyed by tail group components to figures in this section which illustrate and describe either structural features or repairs. Structural components referenced in the index are: fin plating, rudder plating

KEY	FIGURE REFERENCE	STRUCTURAL COMPONENT
1.	Fin	
	Plating	3-3
	Structures	3-4
2.	Rudder	
	Plating	3-4
	Structures	3-4
3.	Leading edge	
	Plating	3-3
	Structures	3-5
4.	Fin tip plating and	
	Structures	3-6

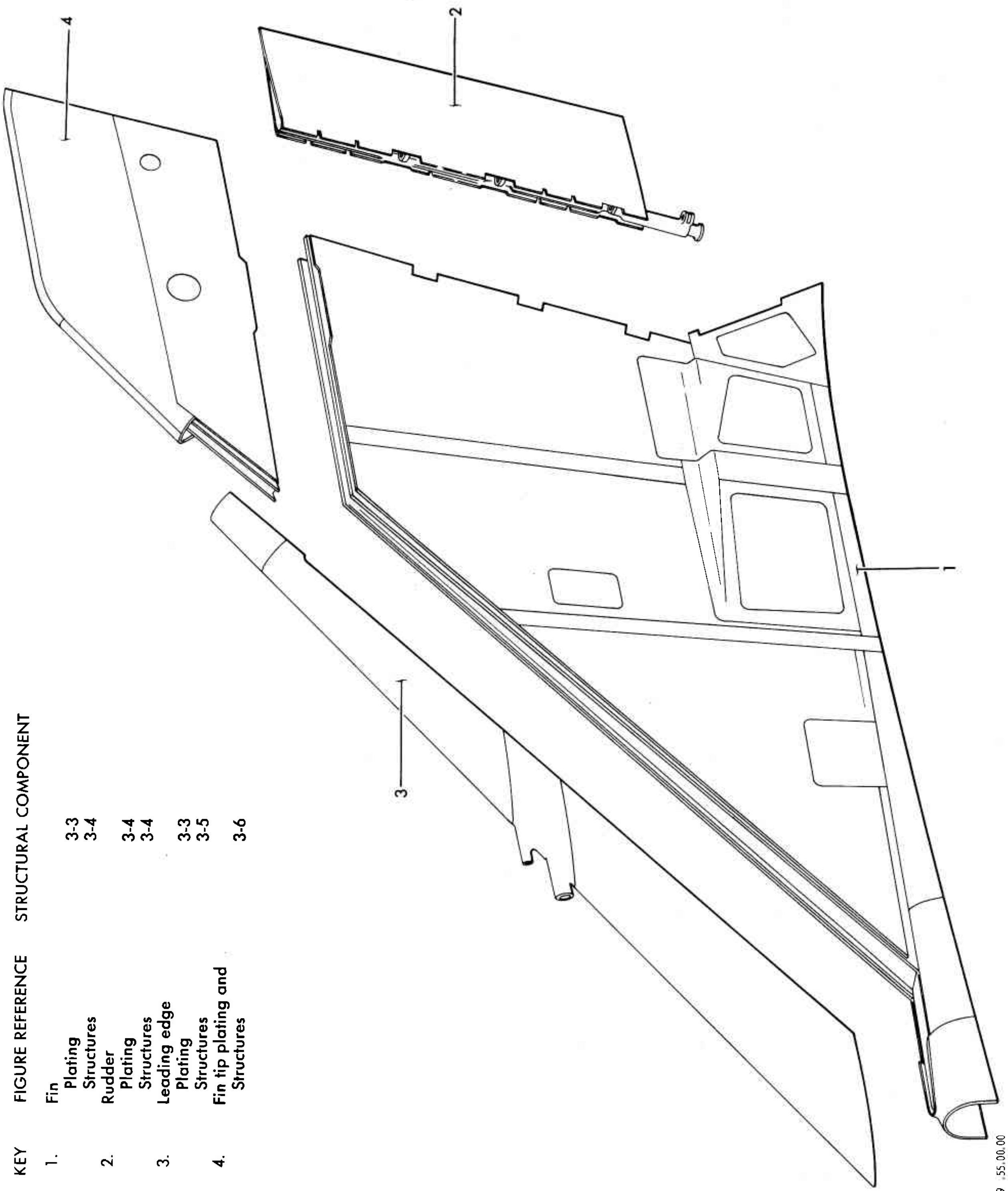


Figure 3-1. Tail Group Components and Figure Index

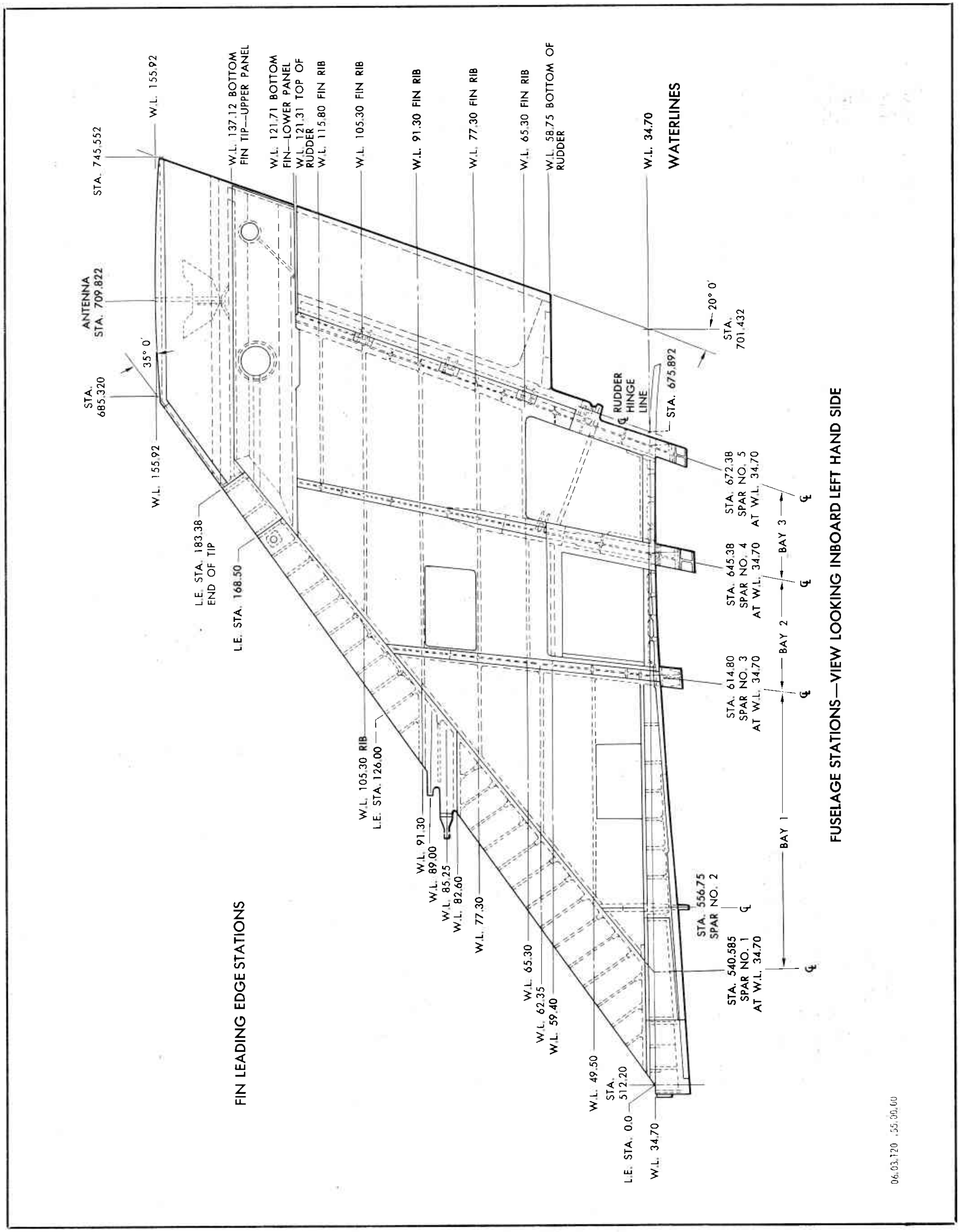


Figure 3-2. Fin and Rudder Station Diagram

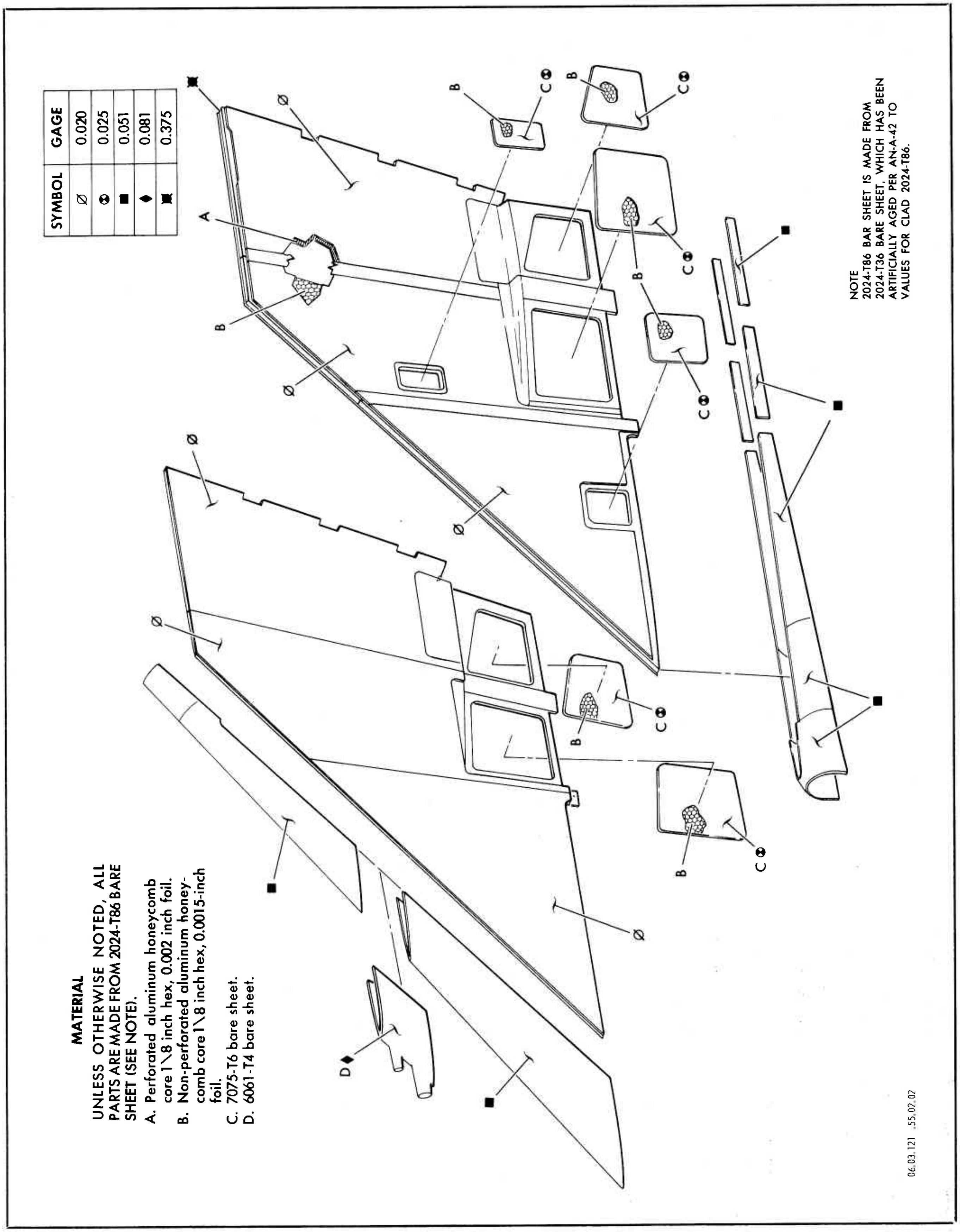


Figure 3-3. Fin Plating Diagram

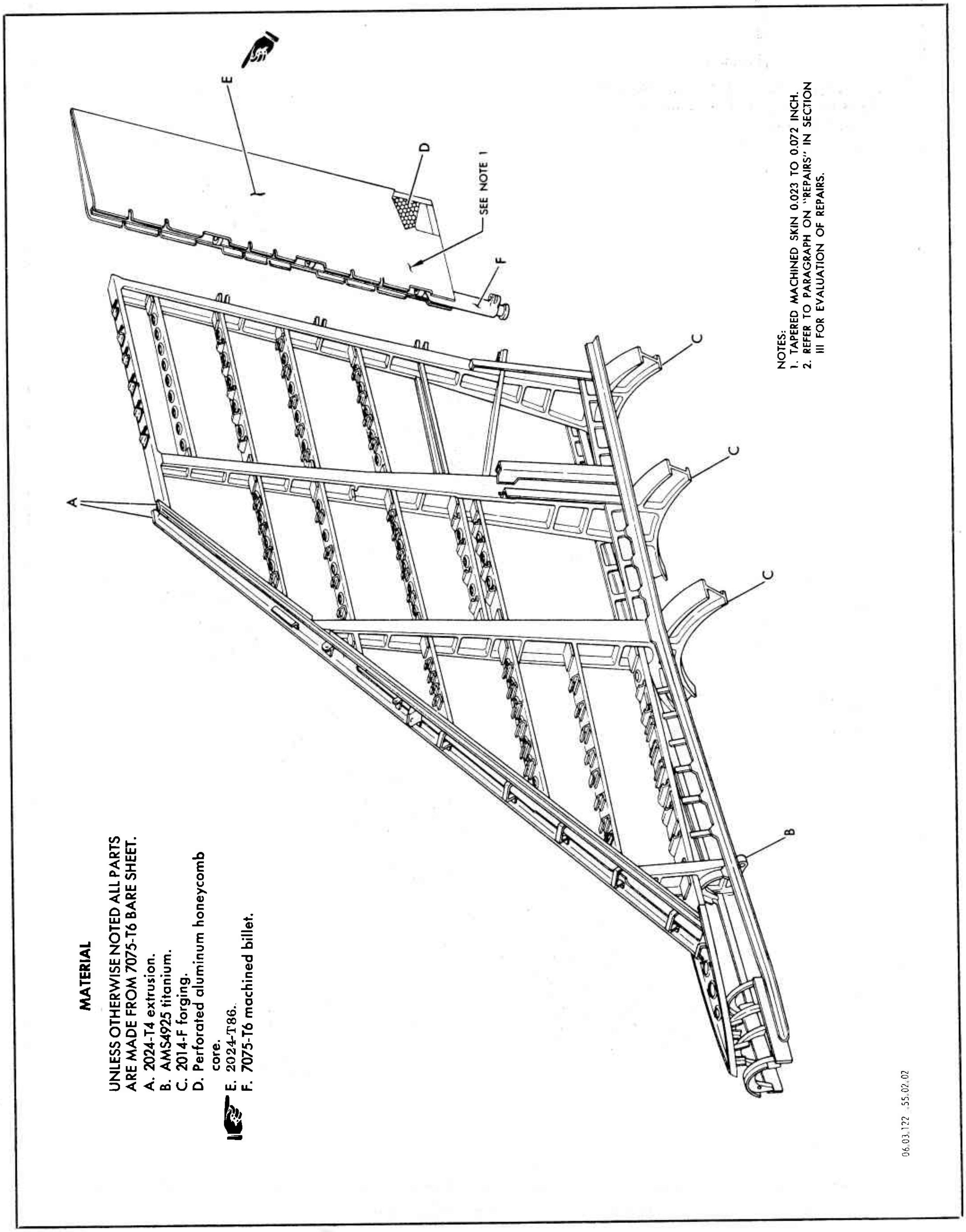
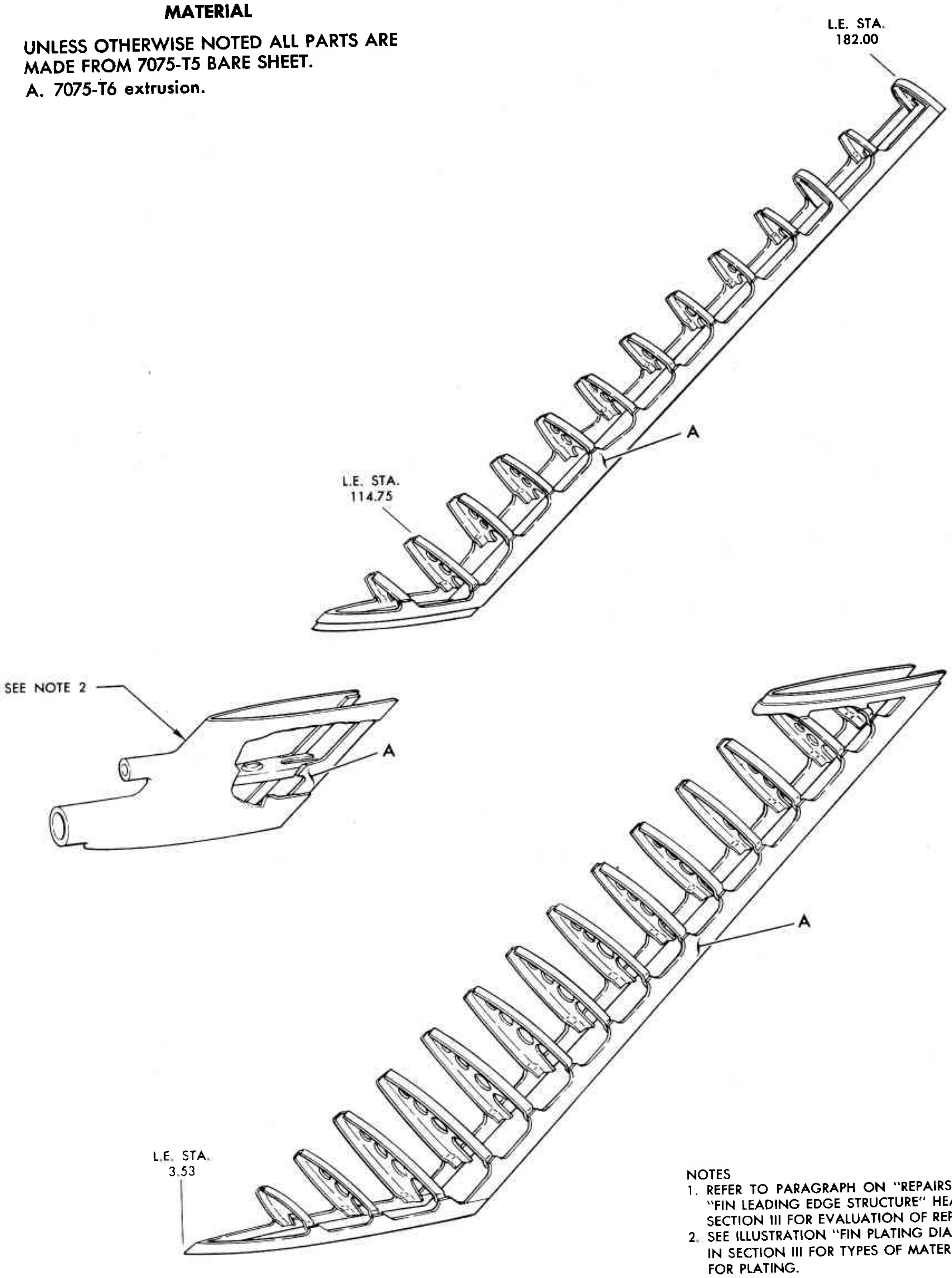


Figure 3-4. Fin and Rudder Structure

MATERIAL

UNLESS OTHERWISE NOTED ALL PARTS ARE MADE FROM 7075-T5 BARE SHEET.

A. 7075-T6 extrusion.



NOTES

1. REFER TO PARAGRAPH ON "REPAIRS" UNDER "FIN LEADING EDGE STRUCTURE" HEADING IN SECTION III FOR EVALUATION OF REPAIRS.
2. SEE ILLUSTRATION "FIN PLATING DIAGRAM" IN SECTION III FOR TYPES OF MATERIAL USED FOR PLATING.

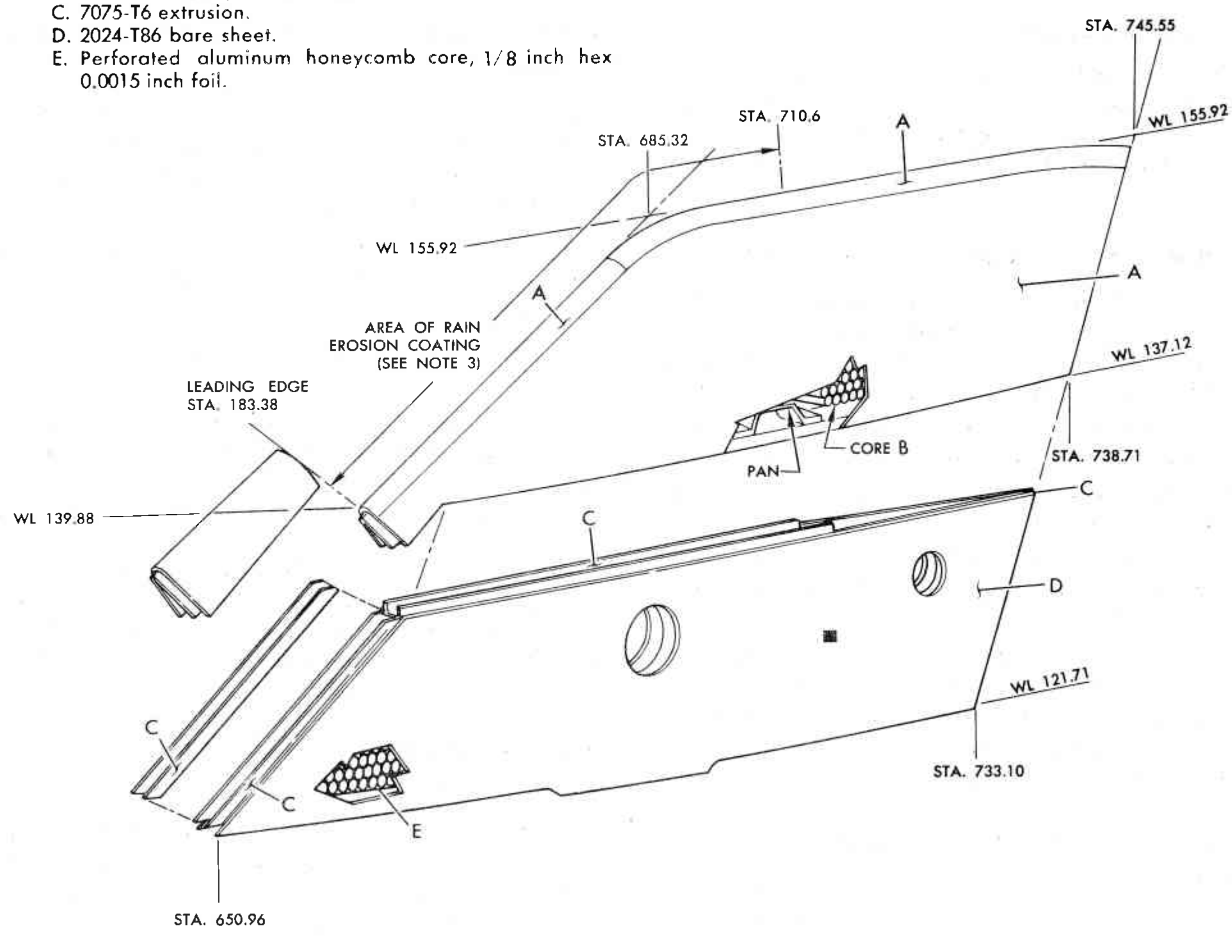
Figure 3-5. Fin Leading Edge Structure

MATERIAL

UNLESS OTHERWISE NOTED ALL PARTS ARE MADE FROM 7075-T6 BARE SHEET.

A. Laminated fiberglas skin, made in accordance with Specification MIL-P-8013A, Type III, of 181 glass cloth conforming to Specification MIL-F-9084, Type VIII and Selectron 5016 resin conforming to Specification MIL-R-7575, Type III.
B. Phenolic honeycomb core, 3/16 inch hex. Specification MIL-C-8073A, Type I-B, Class I.
C. 7075-T6 extrusion.
D. 2024-T86 bare sheet.
E. Perforated aluminum honeycomb core, 1/8 inch hex 0.0015 inch foil.

SYMBOL	GAGE
■	0.051



NOTES:
1. REFER TO PARAGRAPH ON "REPAIRS" IN THIS SECTION FOR REPAIR INFORMATION.
2. REFER TO PARAGRAPH ON "NEGLIGIBLE DAMAGE LIMITS, TAIL GROUP" IN THIS SECTION FOR DISPOSITION OF NEGLIGIBLE DAMAGE TO THIS COMPONENT.
3. REFER TO SECTION I FOR REPAIRS TO RAIN EROSION COATING.

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Figure 3-6. Fin Tip Plating and Structure structures, leading edge plating structures, and fin tip plating and structures. These components are also located as to area on figure 3-1.

3-13. REPAIRS.

3-14. The repairs to the tail group components are unique because of high stress level on the components and the installation of radio gear in certain portions. Consequently, before attempting a repair on any tail group component, carefully read the information concerning the repair of the particular component.

3-15. Rivet Substitution.

3-16. All rivets used in the vertical fin of F-106 airplanes may be substituted with different types of rivets which have either the same or a greater strength value than the original rivet. Refer to Table 1-XXII for rivet substitution data.

3-17. Negligible Damage Limits—Tail Group.

3-18. Table 3-I indicates the maximum allowable classification of five types of negligible damage (damage allowed to remain "as is" after minor rework such as stop drilling cracks, and fairing nicks or scratches). The maximum allowable damage classification will be found to the right of the component name in the vertical column under "Type of Damage." After classification is determined, see figures 1-17 through 1-19 for the damage limits allowed for each class: I, II, or III. The limits given on figures 1-17 through 1-19 apply only for a damaged area after rework, as shown on figures 1-20 and 1-21. An aeronautical structures engineer must be consulted for damages exceeding the limits given on figures 1-17 through 1-19 and for damage to components not listed in this table. Refer to paragraph 3-23 for negligible damage limits for upper fin tip.

3-19. Repairs to Fin Interspar Structure.

3-20. Unless otherwise approved by an aeronautical structures engineer, all repairs to the fin outer surface must be of the flush-type. See figures 3-6, 1-28, and 10-6 for various types of applicable repairs. See figures 3-14, 3-15, and 10-14 through 10-46 for repairs to the honeycomb sandwich panels. Repairs to fin aft spar hinge lug are shown on figure 3-7 and repairs to fin ribs are shown on figure 3-8.

3-21. Fin Leading Edge Repairs.

3-22. Unless otherwise approved by an aeronautical structures engineer, all repairs to the fin leading edge plating must be of the flush-type. See figures 1-25 and 3-9 for details concerning repairs. See figures 10-1 and 10-6 for typical repairs applicable to these components. Refer to paragraph 3-17 for the prescribed limits of negligible damage.

3-23. Fin Tip Repairs—Negligible Damage Limitations, Upper Section.

3-24. Minor scratches and abrasions in fiberglass laminate skin may be considered as negligible and allowed to remain "as is" where the following limitations are not exceeded:

a. Depth of damage: 0.010-inch maximum, provided the underlying glass fibers are not exposed.

b. Location of damage: Within area not covered by rain-erosion coating.

3-25. Fin Tip Repairs—Advanced Base Repairs to Upper Section.

3-26. Minor surface repairs are recommended for scratches and abrasions in the fiberglass laminate skin which cannot be considered as negligible and do not exceed the following limitations:

a. Depth of damage: 0.030-inch maximum, provided no more than three layers of the underlying glass cloth are damaged.

b. Location of damage: Nonrestricted areas shown on figure 3-13.

c. Total area of repair: Six square inches after repair is completed. Refer to paragraph 3-27 for information concerning major damage, if the damage exceeds the above limits. The procedures for the removal and replacement of the rain-erosion coating are given in paragraph 1-197. Minor surface repairs shall consist of filling in the damaged area with a mixture of catalyzed resin and chopped glass fibers, or the laying-in of several layers of catalyzed resin-impregnated glass cloth as specified in paragraph 1-199.

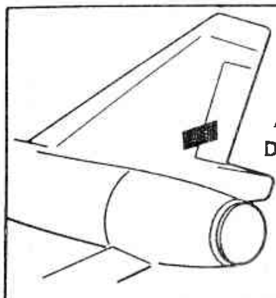
3-27. Fin Tip Repairs—Depot Repairs to Upper Section.

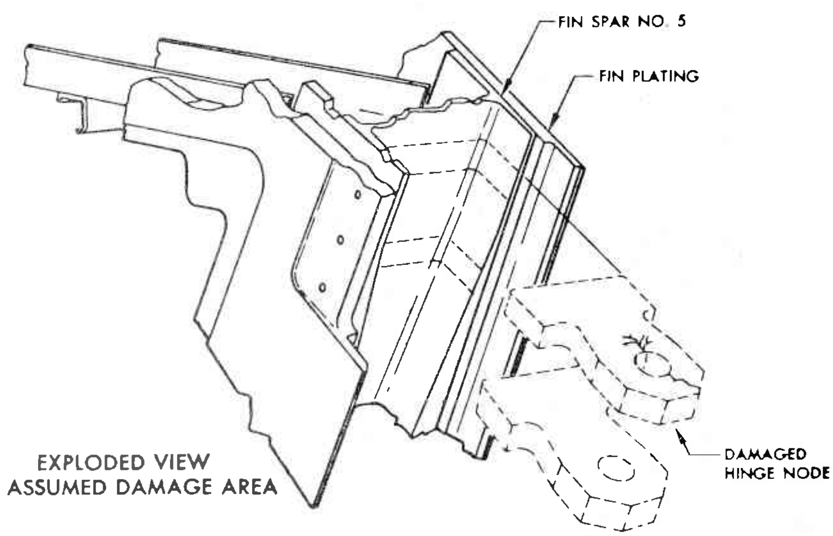
3-28. All damage to the fin tip upper section which exceeds the negligible damage or advanced base repair limitations shall be considered as major damage and shall be corrected at depot level. Types of major damage are as follows:

a. Deep scratches, abrasions, or dents.

b. Cracks or holes through laminate skin.

c. Delamination, which may be defined as a breaking down of the bond between the laminate and the core or between the individual layers of the glass cloth in the laminate. Delamination may be detected by lightly rapping with the edge of a coin on the suspected spot and surrounding areas. Listen for a definite change in sound over the affected area. Bulges in the surface will also indicate delamination areas. Request assistance from the nearest prime AMA for review and disposition of major damage by qualified aeronautical structures and antenna engineers. See figure 3-13 for areas that are restricted to the following types of repair:





REPLACEMENT OF FIN AFT SPAR HINGE NODE

a. Remove the rudder from the airplane. Refer to T.O. 1F-106A-2-7-2-1 for rudder removal procedure.

b. Using a metal saw, remove as much of the damaged hinge node as possible.

c. File the remaining portions of the hinge node not removed by sawing, until the hinge surfaces are flush with the spar. Using emery cloth, remove all file marks from the spar. Refer to Section I for paint and primer procedures.

d. Fabricate a new hinge node from a piece of 3.50x3.50x5.00-inch 2014-F bar. Outside dimensions of new hinge node will vary according to the amount of material removed from the spar and the natural taper of the spar.

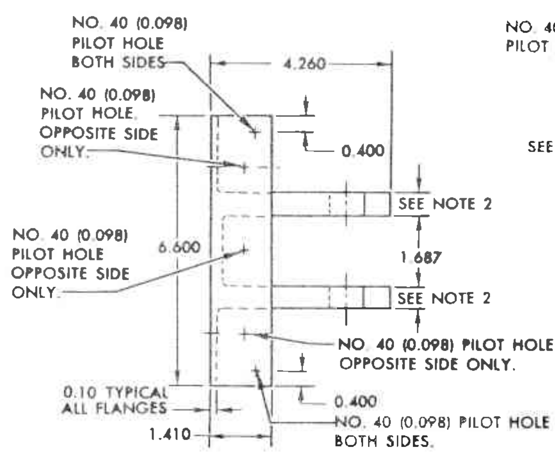
e. Drill 12 No. 40 (0.098) pilot holes and two No. 30 (0.128) pilot holes, as shown in Detail A.

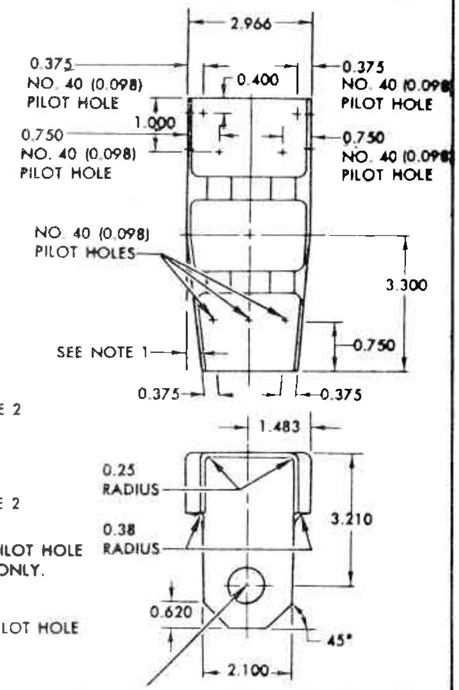
f. Position and align the new hinge node in the area provided in steps B and C. Secure the new hinge node in place with clamps. See figure Master Tooling—Vertical Fin for correct position of hinge node.

g. Using the seven No. 40 (0.098) pilot holes shown in the front view of detail A as a guide, drill seven No. 15 (0.180) holes. Ream holes from 0.199 to 0.202 and install seven F-200-2 Jo-bolts.

CAUTION

USE EXTREME CARE TO AVOID CUTTING INTO THE SPAR FLANGES.





DETAIL A

PRE-DRILL (UPPER, CENTER AND LOWER HINGE NODES) WITH NO. 30 (0.128) DRILL. FINISH DRILL CENTER AND LOWER NODES WITH 13/16 INCH DIAMETER DRILL; FINISH REAM TO 0.8750 (±0.0005) INCH. FINISH DRILL UPPER HINGE NODES WITH 3/4-INCH DIAMETER DRILL; FINISH REAM TO 0.8125 (±0.0005) INCH (REFERENCE DRAWING 8-14795).

h. Using the five No. 40 (0.098) pilot holes shown in the side view of Detail A as a guide, drill five No. 15 (0.180) holes. Ream holes from 0.199 to 0.202 and install five F-200-5 Jo-Bolts.

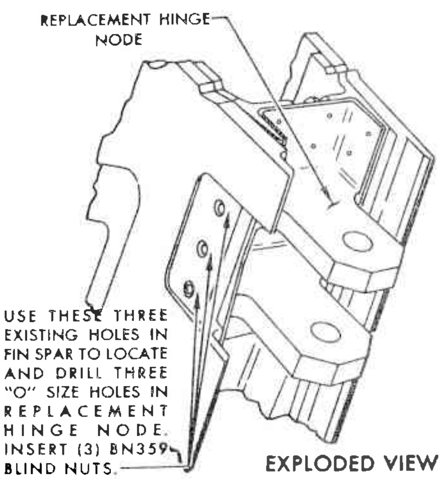
i. Using the three existing holes in the fin plating and fin spar, drill "O" (0.316) size holes (3) in the replacement hinge node and insert three BN359 blind nuts.

j. Using the two No. 30 (0.128) pilot holes as a guide, drill and ream rudder pin holes to the dimensions shown in Detail A.

k. Reinstall the rudder according to procedures outlined in T.O. 1F-106A-2-7-2-1.

NOTES

1. REPLACEMENT OF DAMAGED HINGE NODE AT WATER LINE 65.30 WILL REQUIRE A TAPERED REPLACEMENT PART. ALL OTHER HINGE NODES WILL NOT REQUIRE A TAPERED REPLACEMENT PART.
2. UPPER HINGE NODES 0.350, CENTER AND LOWER HINGE NODES 0.500 (REFERENCE DRAWING 8-14795).



EXPLODED VIEW
OF REPLACEMENT PART
AFTER INSTALLATION

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Figure 3-7. Fin Aft Spar Hinge Lug (Nodes) Repair and Replacement (Sheet 1 of 2)

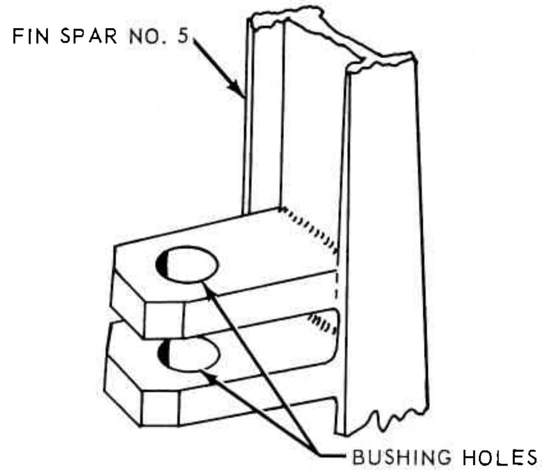
REPAIR OF FIN AFT SPAR HINGE NODE BUSHING RETAINER HOLES

a. When bushing holes are oversize or galled, clean up damaged hole by reaming until damage is removed. The diameter of the bushing hole shall not be increased beyond 1.010 inch.

b. Repair hinge nodes in accordance with Figure 3-7 (Sheet 1 of 2) if the above limit is exceeded.

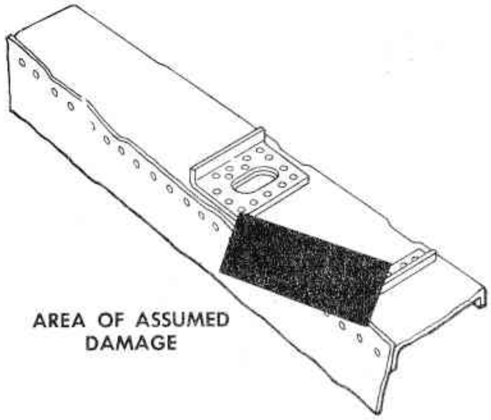
c. Manufacture repair bushings (one piece) in accordance with drawings 8-15805 (upper hinge nodes) and 8-15810 (center and lower hinge nodes) with the exception of increasing the O.D. diameter as required to maintain an interference fit of 0.0005 to 0.0015 inch.

d. Install bushing in accordance with T.O. 1F-106A-2-7-2-1.



EXPLODED VIEW
OF DAMAGED HOLES

Figure 3–7. Fin Aft Spar Hinge Lug (Nodes) Repair and Replacement (Sheet 2 of 2)



A Drill out existing 3/16 inch diameter rivets indicated by (O), attaching skin, doubler, and stiffener using a No. 11 (0.191) drill. Drill out rivets attaching skin four places beyond damaged area.

B Clean up damaged area to regular shape. Break all sharp edges and round all corners to 0.125 radius.

C Fabricate doubler of same type material and gage as rib assembly web. Drill No. 11 (0.191) rivet holes (19) to match existing rivet holes indicated by (O) and new rivet holes (40) as indicated by (-)

D Replace stiffener. Drill No. 11 (0.191) rivet holes indicated by (-) to match existing rivet holes indicated by (O) on doubler.

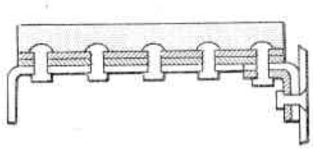
E Fabricate filler angle of same type material and gage as rib. Drill No. 11 (0.191) holes (20). Pick up rivet pattern from existing holes on doubler and rib.

F Fabricate splice angle of same type material and gage as rib. Drill No. 11 (0.191) holes (36). Pick up rivet pattern from existing holes on doubler, rib, and angle.

G Install doubler, stiffener, filler angle, and splice angle with AN470DD6 rivets.

H Attach skin with AN426DD6 flush head rivets (18) through filler angle and splice angle.

I Skin omitted for clarity. See Section A-A. If skin requires replacement, machine countersink rivet holes 100° x 0.365.



SECTION A-A

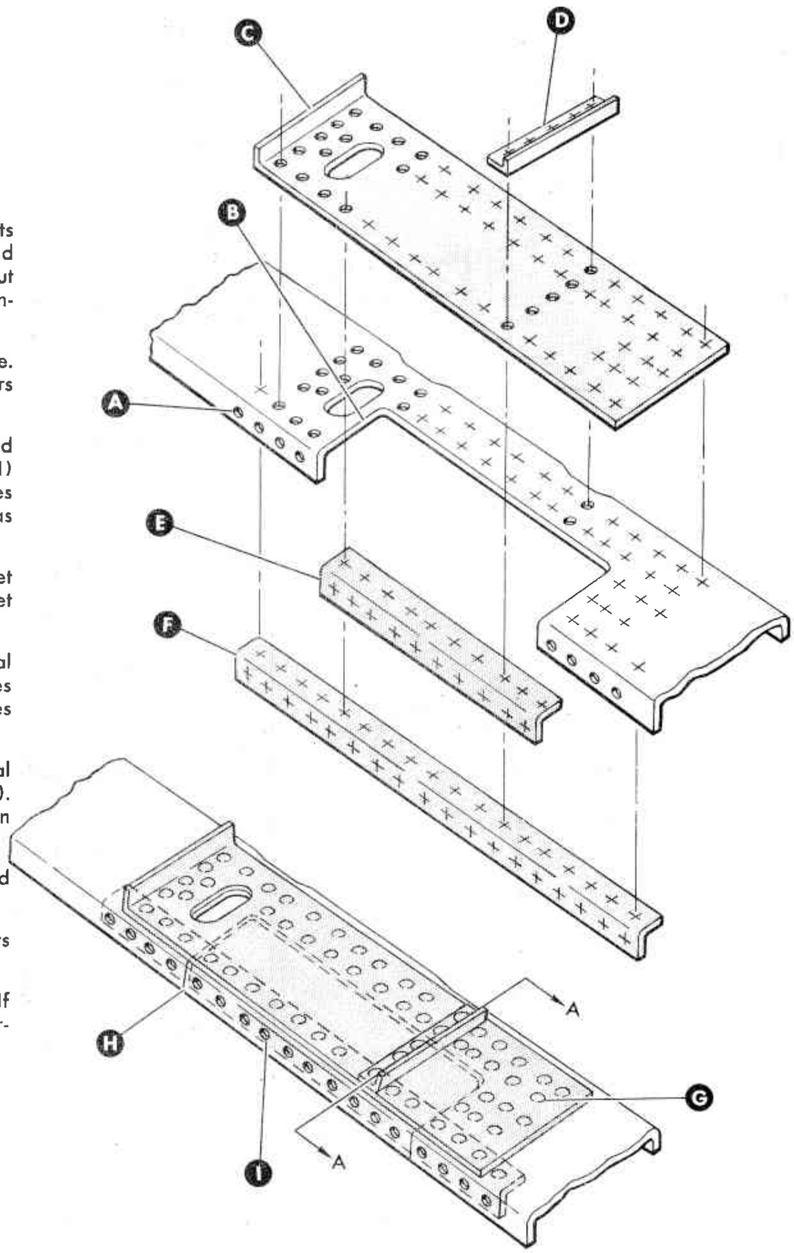


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Figure 3-8. Fin Rib Repair

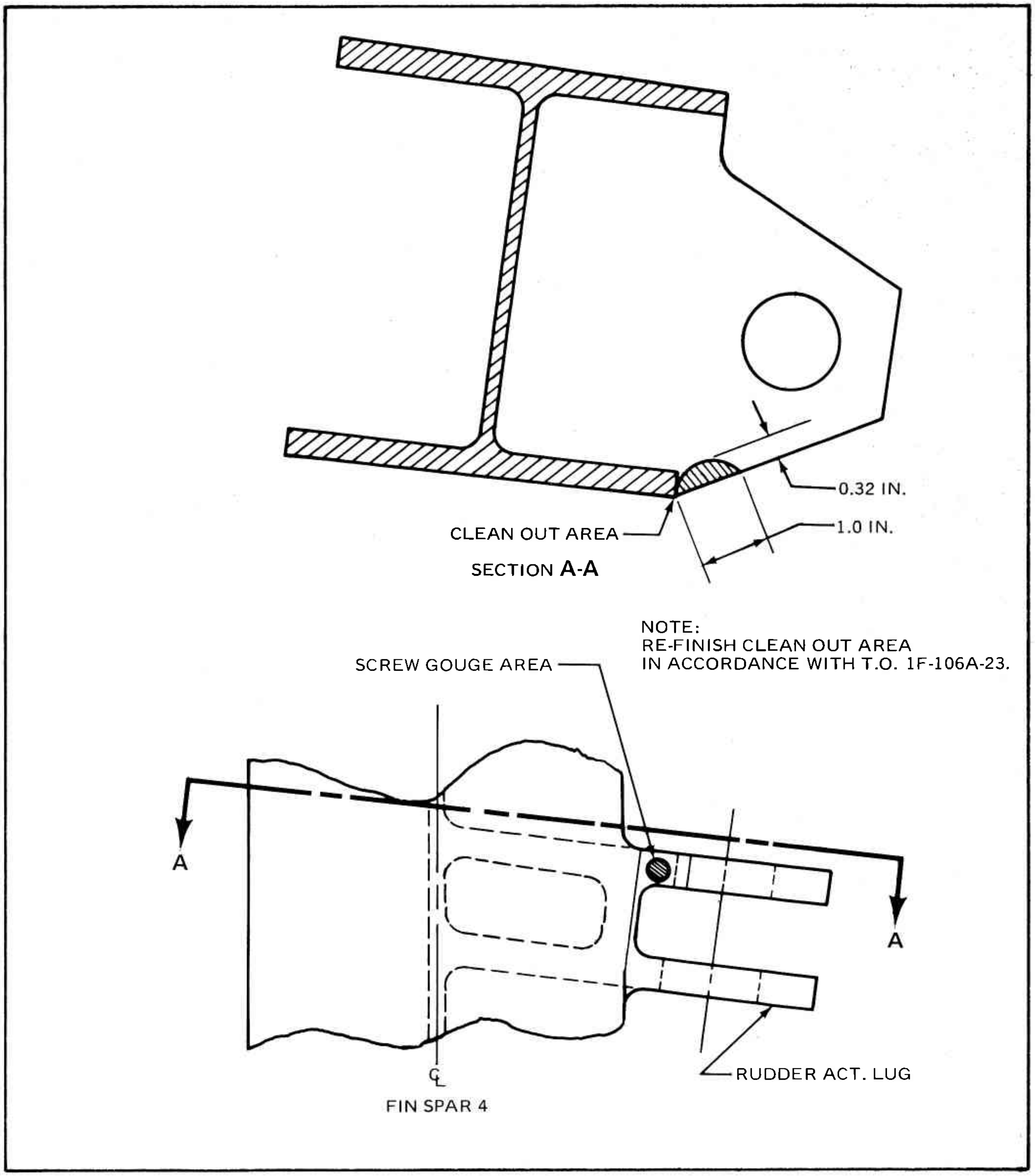
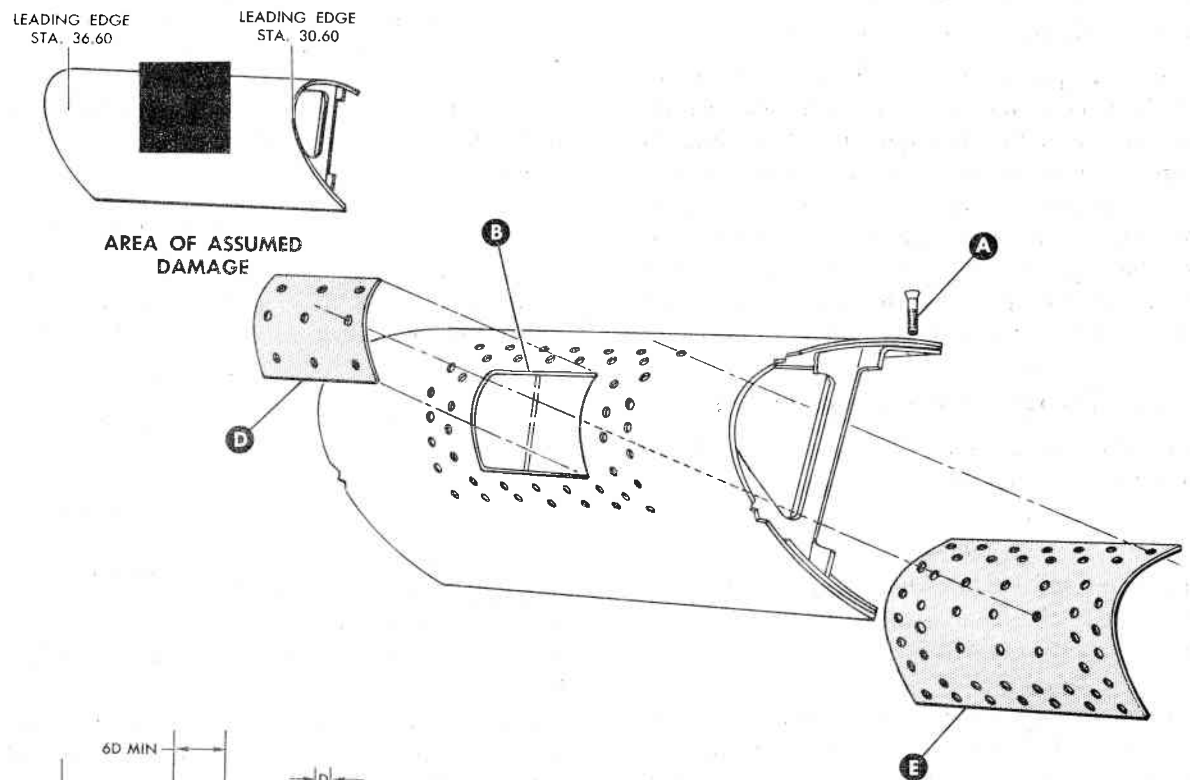
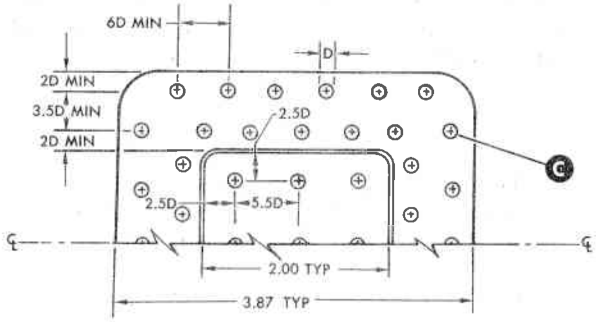


Figure 3-8A. Fin Spar 5 Repair





A Remove NAS333 screws attaching fin leading edge containing damaged area. Remove fin section.

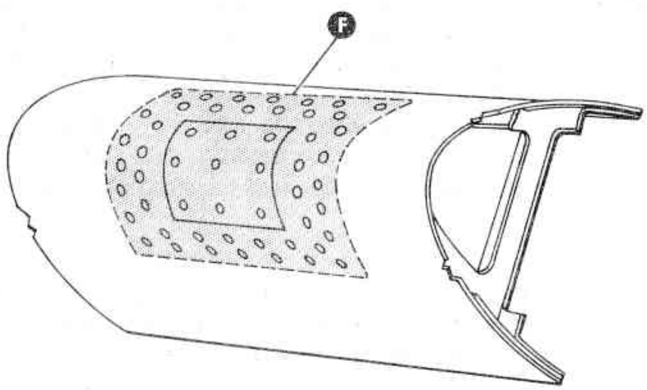
B Clean up damaged area to regular shape. Break all sharp edges and round all corners to a 0.125 radius.

C Layout and drill No. 30 (0.128) rivet holes (53) as shown. Machine countersink rivet holes 100° x 0.297.

D Fabricate patch plate from 0.051 gage, type 7075-T6 bare material.

E Fabricate doubler of 0.064 gage, type *7075-T6 bare material.

F Install patch plate and doubler with AN426-AD5 flush head rivets (53).



*IF NECESSARY, FABRICATE PART FROM 7075-0, SHAPE TO FIT CONTOUR, AND HEAT TREAT TO 7075-T6 CONDITION PER SPECIFICATION MIL-H-6088 TO PREVENT CRACKS.

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Figure 3-9. Fin Leading Edge Repair

a. Upper fin tip repair — leading edge.

b. Upper fin tip repair — step joint method.

c. Upper fin tip repair — scarf joint method.

If damage to the upper fin tip extends through both face plies and the honeycomb core, repair one side first and allow to cure, then repair the opposite side making the second repair at least two inches larger in diameter than the first. If the types of repair resin mentioned in the repair procedures are not available, any polyester resin that meets the requirements of Specification MIL-R-7575, type III, may be used provided that the manufacturer's mixing and curing instructions are strictly followed.

3-29. Upper Fin Tip Repair — Leading Edge.

3-30. The following procedure is to be used with figure 3-10, for repair of the fin tip leading edge:

NOTE

Information will be furnished when available concerning combination repair of leading edge and honeycomb core of upper fin tip.

a. Remove rain erosion coating six inches on each side of damaged area. Refer to paragraph 1-197 for removal procedure for rain erosion coating. If the damage occurs in that portion of the leading edge not covered with rain erosion coating, omit this step and proceed with step "b."

b. Cut or rout out the damaged area to a regular shape.

CAUTION

Do not cut or rout into the honeycomb core area in excess of one-half inch in depth.

c. Using a vacuum cleaner or compressed air, thoroughly remove all loose particles from the damaged area.

WARNING

If compressed air is used for removing loose particles, a face shield shall be worn by the operator to protect against flying particles.

d. Using an undamaged fin tip as a guide, prepare a mold at least four inches longer than the length of damage (two inches overlap at each end) to simulate the contour of the damaged area.

e. Using sheet wax, build up the inside of the mold approximately 0.080 inch to simulate the thickness of the fin tip leading edge laminate.

NOTE

Sheet wax may be obtained in varying thicknesses from the Kindt-Collins Company, 12651 Elmwood Ave., Cleveland, Ohio.

f. Insert one layer of No. 181 glass cloth, Specification MIL-F-9084, into the mold; impregnate with catalyzed resin and allow to cure.

REPAIR RESIN AND CATALYST

100 parts of resin Epon 815 or 828 Shell Chemical Corp.	and	25 parts of catalyst curing agent "T-I" Shell Chemical Corp.

or as an alternate use:

100 parts of resin Epon 828 Shell Chemical Corp.	and	10 parts maximum of curing agent "Hardener 951" Furane Plastics Inc.

g. Prepare the damaged fin tip leading edge for repair by scarfing the edges of the undamaged fiberglass laminate. Edges should be scarfed back at least one inch to provide sufficient bonding area for the fiberglass repair plies.

h. Remove the cured layer of fiberglass from the mold; cut to fit into the damaged area; clean thoroughly with methyl ethyl ketone, Specification TT-M-261, and allow to dry.

NOTE

The cured ply of fiberglass should be cut to length so as to butt up against but not overlap the innermost layer of laminate.

i. Cut nine additional layers of No. 181 fiberglass cloth. Each layer of fiberglass should be cut to sufficient size so that it will overlap the preceding layer by a proportionate amount. Proportionate amount of overlap will be determined by length of scarf. For instance, if the edge of the leading edge laminate has been scarfed back one and one-half inches and nine layers of cloth are to be applied, the overlap will be determined by the following formula: 1.50 inches divided by 9 equals 0.166 inches, which equals total overlap. 0.166 inches divided by 2 equals 0.083 inches, which equals overlap at each end.

j. Mix up enough resin with proper catalyst to impregnate the entire nine layers of repair plies. To mix the proper amount of resin, weight of base resin should equal the dry weight of fiberglass repair plies.

k. Apply a thin brush coat of catalyzed resin to the exposed surface of pre-cured repair ply and to the scarfed area.

l. Apply the first repair ply of fiberglass (first ply will be the smallest piece) and thoroughly impregnate the pores of repair ply with catalyzed resin.

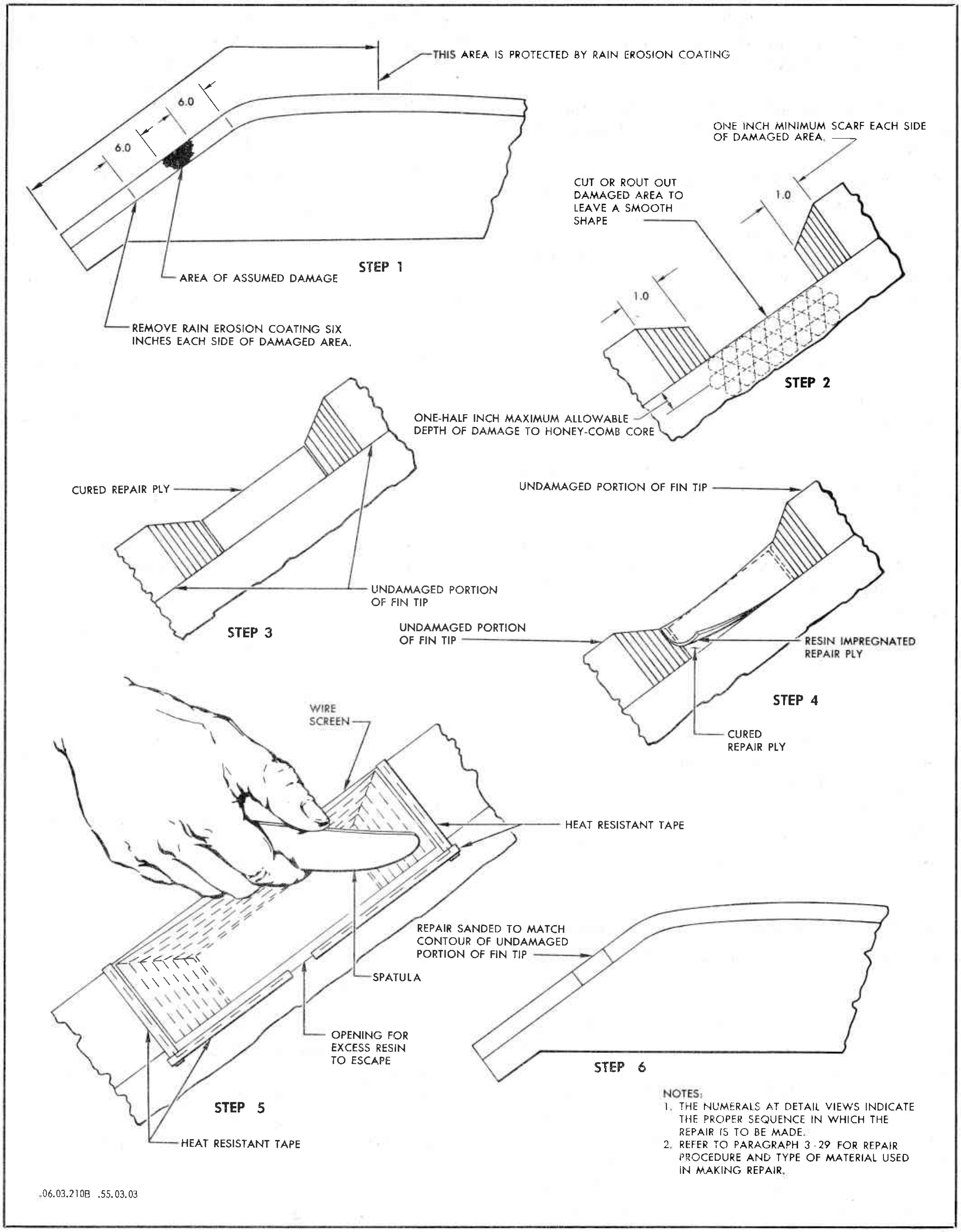


Figure 3-10. Upper Fin Tip Leading Edge Repair

NOTE

A nylon bristle brush will provide good results in applying resin to fiberglass cloth.

m. Apply the next larger repair ply and repeat resin application. Follow this method until all repair plies are in place.

n. Form a piece of fine wire screen over the repair area, and hold in place with heat resistant tape.

o. Using a spatula or similar flat object, wipe out the excess resin. The working or wiping of the laminate will be stopped when all air bubbles have been removed and the patch plies are firmly packed together.

CAUTION

Excessive working or wiping will cause voids and/or dry spots in the laminate.

p. Wipe up the excess resin from around the repair area with a lint free cloth moistened with methyl ethyl ketone.

q. Remove heat resistant tape and wire screen and allow repair to cure from one and one-half to two hours at room temperature, or from 25 to 35 minutes at 300°F (148.90C).

r. Using 180-grit sandpaper, sand repair area to a smooth contour with the rest of the fin tip.

s. Clean the fin tip with a cloth moistened with methyl ethyl ketone and allow to dry thoroughly.

t. Mix a small amount of base resin with catalyst and apply a thin finish coat to the repair area. Finish coat should overlap onto the undamaged portion of the fin tip on all sides a minimum of one-half inch.

u. If the damage occurred in the area protected by rain erosion coating, replace the rain erosion coating as outlined in paragraph 1-197.

3-31. Upper Fin Tip Repair — Step Joint Method.

3-32. Removal of Damaged Face Plies — Step Joint Method.

3-33. The step joint method of repair is recommended for those repairs which are larger than four square inches and extend either into the honeycomb core or completely through the core and both face plies. The following procedure may be used as a guide for the step joint type of repair. See figure 3-13 for areas restricted to this type of repair and figure 3-11 for an illustration of this method.

a. Outline the damaged area, with the aid of a straight edge and scribe, by scribing a rectangle or square with rounded corners that will require a minimum removal of sound material.

b. Extend the sides of the rectangle or square a distance in inches equal to the number of plies to be removed, less one inch, from the edge of the damaged area (seven inches if eight plies are to be removed, etc.). Overlap of each step should be at least one inch.

NOTE

In case of core damage, allow a minimum of one-half inch between the edge of the innermost ply and the perimeter of the core material that is to be removed.

c. With the aid of a straight edge, use a sharp knife or other especially prepared cutter and cut out a square area in the center of the damaged area approximately two square inches in area.

d. Remove the plug cut from the center of the damaged area.

e. With the aid of a straight edge, cut along the lines that were scribed in step "b."

CAUTION

Use extreme care not to cut or score the underlying ply of glass cloth. A suggested method is to cut half-way through the overlying ply and then lift this ply using a sharp thin chisel, as shown on figure 3-10 so that it will break away cleanly. If the layer of glass cloth underneath is cut or scored, the strength of the repair will be lessened.

f. When the outermost ply is removed, scribe a similar outline on the next exposed ply, except reduce the original dimensions one inch in all directions. Remove this ply as described previously and continue this procedure until all the damaged plies have been removed.

3-34. Replacement of Face Plies — Step Joint Method.

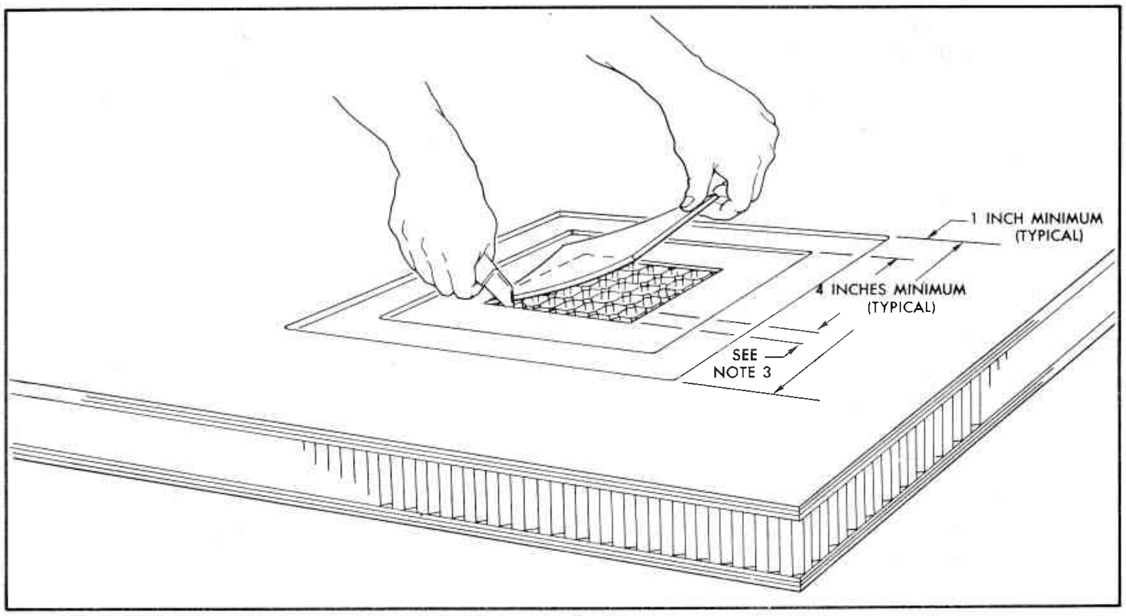
3-35. The following procedure may be used as a guide for replacing face plies in the step joint method:

a. Lightly sand exposed plies using 180-grit sandpaper.

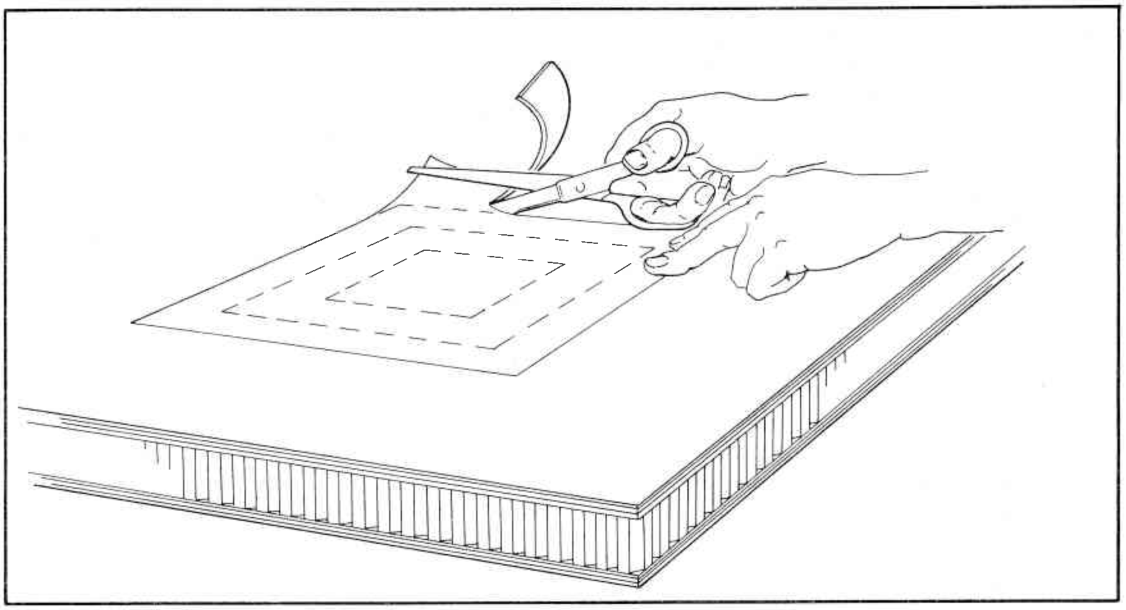
WARNING

The sanding operation on laminated glass cloth gives off a fine dust that may cause skin irritations, and breathing an excessive amount of this dust may be injurious. Therefore, precautions as to skin and respiration protection must be observed.

b. Replace face plies as shown on figure 3-10 by cutting patches from resin impregnated glass cloth (cloth



REMOVAL OF DAMAGED FACE PLIES (SEE NOTE 1)



REPLACEMENT OF REPAIR PLIES (SEE NOTE 2)

NOTES:
1. REFER TO PARAGRAPH 3-32 FOR PROCEDURE ON REMOVING DAMAGED FACE PLIES BY THE STEP JOINT METHOD OF REPAIR.
2. REFER TO PARAGRAPH 3-34 FOR PROCEDURE ON REPLACING FACE PLIES BY THE STEP JOINT METHOD OF REPAIR.
3. IN CASES OF CORE DAMAGE, ALLOW A MINIMUM OF ONE-HALF INCH BETWEEN THE EDGE OF THE INNERMOST PLY AND THE PERIMETER OF THE CORE MATERIAL TO BE REMOVED.

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Figure 3-11. Fiberglass Repair — Step Joint Method

conforming to Specification MIL-F-9084, with finishes as outlined in Specification MIL-F-9118), to conform to the first or innermost outline of exposed cutout plies and butting up to existing layers of plies but not overlapping. The impregnated glass cloth should contain 45 to 50 percent of catalyzed resin:

100 parts of resin Epon 815 or 828 Shell Chemical Corp.	and	25 parts of catalyst curing agent "T-1" Shell Chemical Corp.
	or	
100 parts of resin Epon 828 Shell Chemical Corp.	and	10 parts maximum of catalyst curing agent "Hardener 951" Furane Plastics Inc.

(Weight of resin to be equal to weight of dry glass cloth.)

NOTE

If the damage extends to the honeycomb core, the innermost layer of cloth should be cured before placing in damaged area in order to make a void-free repair to the laminate.

c. Lightly sand the cured repair ply surfaces and clean thoroughly as outlined in step "a."

d. Brush a thin coat of catalyzed resin on the surface of the cured ply that will bond to the honeycomb core, then brush a thin coat of catalyzed resin on the contact edges of the honeycomb core.

e. Place the cured ply in place and allow to cure from one and one-half to two hours at room temperature, or from 25 to 35 minutes at 300°F (148.9°C).

f. Cut the remaining patch face plies to fit the remaining steps.

NOTE

A recommended method for cutting resin impregnated glass cloth is to sandwich each piece between two sheets of colored cellophane larger than the patch by at least two inches for all sides. The patch plies are then cut to desired shape without the usual fraying of the edges. Remove cellophane sheets as patch plies are applied.

g. Cut from polyvinyl alcohol sheeting (PVA) a sheet or cover conforming to the shape of the repair area and extending at least two inches beyond the edge of the repair.

h. Place the PVA sheeting over the wet laminate; then using heat resistant tape or zinc chromate paste, seal the edges of the PVA sheeting except for a small opening at one edge.

i. Using a spatula or similar flat object, wipe out the excess resin and all air bubbles toward the opening left in the sheeting. The working or wiping of the laminate will be stopped when all air bubbles have been removed and the patch plies are firmly packed together.

CAUTION

Excessive working or wiping will cause voids and/or dry spots in the laminate.

j. When the repaired laminate has been cured, one and one-half to two hours at room temperature or 25 to 30 minutes at 300°F (148.90C), lightly sand, if necessary, to obtain smoothness and contour, to remove excess resin and to prepare surface for any subsequent painting or coating. Use 180-grit sandpaper and observe skin and respiration precautions as previously noted.

k. Mix a small portion of catalyzed resin and brush on a thin finish coating of clean resin. Allow finish coat to cure 8 to 24 hours before subjecting fin tip to use.

3-36. Upper Fin Tip Repair — Scarf Joint Method.

3-37. Removal of Damaged Face Plies — Scarf Joint Method.

3-38. This type of repair is not recommended for damage in excess of four square inches and extending into the honeycomb core. The following procedure may be used as a guide for the scarf method of repair:

a. Sand out the damaged face plies to a circular or oval dish shape using 180-grit sandpaper, as shown on figure 3-10.

b. The damaged face plies shall be scarfed back a minimum of fifty times the face ply or wall thickness from the outer edge of the damaged area. For instance, if wall thickness equals 0.045 inch, scarf back 50 times 0.045 inch, or 2-250 inches.

3-39. Replacement of Face Plies — Scarf Joint Method.

3-40. The following procedure may be used as a guide for replacing patch plies in the scarf type repair:

a. Thoroughly clean the damaged area with methyl ethyl ketone, Specification TT-M-261. Allow to dry completely before proceeding with repair.

b. The glass cloth laminations for the scarf type repair are prepared with the largest piece being cut to the exact shape of the outside of the scarfed area, and the smallest piece being cut so that it overlaps the scarfed area by its proportionate amount (proportionate amount of overlap will depend upon the number of plies being replaced). The intermediate pieces are cut to have equal taper. See figure 3-12 for an illustration of this method.

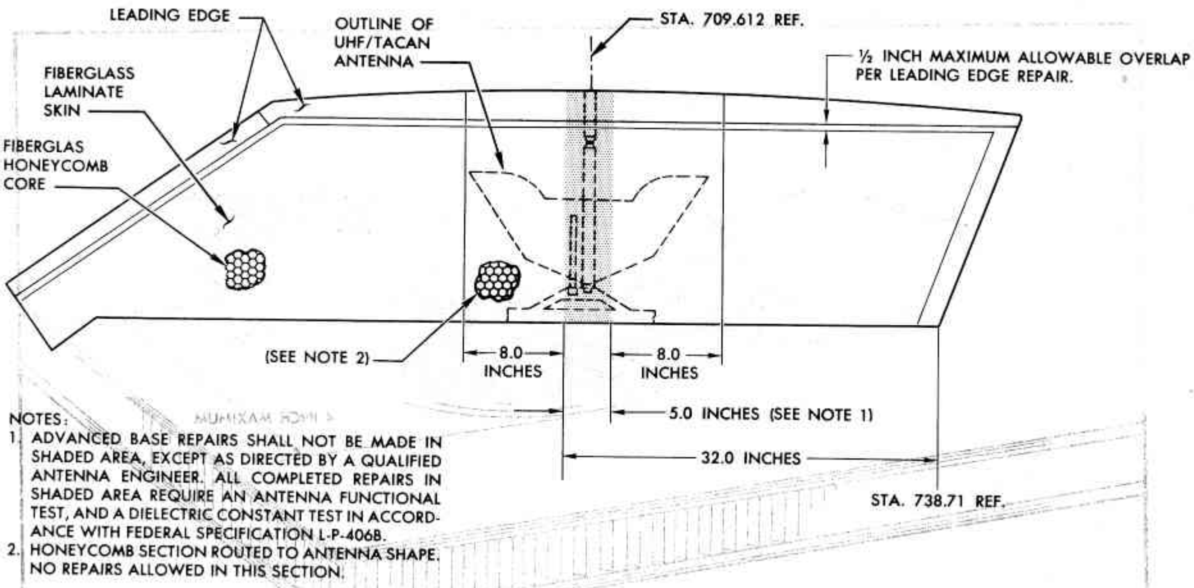


Figure 3-13. Repair Limitations — Upper Fin Tip Fiberglass Laminate

NOTE

A convenient method of preparing the glass cloth patches is to impregnate the repair patches with catalyzed resin in any convenient method. Use the same type and mixture of resin as used in the step joint type of repair. Sandwich repair patches between two sheets of colored cellophane larger than the repair patch by at least two inches on all sides. The patch pieces are then cut to desired shape without the usual fraying at the edges.

c. Coat the scarfed surface of the repair area with one brush coat of catalyzed resin before applying patches.

d. When all the patch plies are ready for assembly, place them in the proper sequence for installation with the smallest piece first. Remove the first sheet of cellophane and insert the patch in place, then remove the second sheet of cellophane.

e. Complete the buildup, cure and finish as outlined in paragraph 3-34.

3-41. Replacement of Damaged Honeycomb Core.

3-42. When making repairs to the honeycomb core, do not remove both sides of laminated face plies at the same time. Remove one surface of face plies by either the step joint or scarf method, depending upon the extent of damage, then repair honeycomb core by the following procedure:

a. Cut out damaged core and lightly sand the exposed surface of the inner ply of cloth.

b. Thoroughly clean the damaged area with methyl ethyl ketone, Specification TT-M-261, and allow to dry thoroughly.

c. Cut a piece of honeycomb core material to the exact size of the damaged area. The thickness and cell size of the repair honeycomb core shall conform to Specification MIL-C-8073A.

NOTE

If slicing of the repair honeycomb core is necessary, use a band saw for convenience. The bandsaw blade should have about 32 teeth per inch and be operated at a speed of about 5000 fpm.

d. Apply a thin coat of catalyzed resin to the cutout area and a like amount on all surfaces of the replacement core, then insert core material into the damaged area making certain that replacement core conforms closely to the existing undamaged honeycomb core.

e. Insert the first impregnated patch ply over the replacement core material, then apply a light and uniform pressure by means of heat resistant tape, sand bags or vacuum bags.

f. Cure the honeycomb core repair with heat lamps, or by inserting the entire assembly in an air-circulating oven at temperatures specified in paragraph 3-34.

g. Finish face ply buildup by repeating procedures outlined in either paragraph 3-34 or 3-39.

h. Finish outside surface as outlined in paragraph 3-34.

3-43. Fin Tip Repairs to Lower Section.

3-44. See figure 1-18, and refer to typical repairs in Section X for repairs to the ribs and channels in the fin tip lower section. Refer to Section X for repairs to the honeycomb sandwich panels. Refer to Section II, Paragraph 2-77A, which uses same type procedures and materials.

3-45. Rudder Repairs.

3-46. All repairs to the rudder are considered to be of a critical nature because of the high stress level imposed on the structure. Any repairs to the spar, the skin adjacent to the rudder hinge, and to the rudder actuator will require approval by an aeronautical structures engineer. The use of external patches or the protruding of rivet heads will require the approval of an aeronautical structures engineer. Refer to Table 3-1 for the prescribed limits of negligible damage. Refer to Section X for repairs to the honeycomb structure. See figure 3-14 for rudder honeycomb separation repair. Refer to Section II, paragraph 2-77A, which uses same type procedures and materials.

3-46A. Inspection, Dehydration, Repair, and Seal Surfaces Containing Honeycomb Core Applicable to Vertical Stabilizer Fin Tip Lower Panel and Rudder will be accomplished as follows:

a. Remove vertical stabilizer fin tip lower panel and rudder in accordance with instructions contained in T.O. 1F-106A-2-2-2-2 and T.O. 1F-106A-2-7-2-1.

b. Drill out JO-Bolts, NAS1674-3L, attaching tip assembly lower panels to vertical stabilizer.

NOTE

Rudders and fin tip lower panels which have been reworked by depot, to replace the perforated core with non-perforated core do not require dehydration and sealing.

c. X-ray fin tip lower panel and rudder to determine if moisture is present and/or core is damaged.

d. Using a "Q" drill bit (0.3320-inch diameter) drill hole in center of upper rib on longitudinal centerline of vertical stabilizer fin tip lower panel. For rudder, on longitudinal centerline on bottom rib two inches aft of aft edge of support fitting.

CAUTION

Use drillstop to assure that bit does not penetrate more than 1/8 inch into honeycomb core.

e. Tap hole with 3/8 -24 tap and install union fitting AN815-3 (FSN 4730-187-0483) in tapped hole using sealant MIL-S-8802 class B.

f. Utilizing low temperature boil procedure, dehydrate assembly using set moisture removal FSN 4920-ND00062P. Vacuum shall be between 10 PSIG to 12 PSIG to assure boiling of water at as low as 160°F and temperature shall not exceed 180°F. 4 to 8 hours may be required for dehydration of core assembly, dependent on the amount of water in core.

NOTE

A vacuum chamber capable of safely maintaining 12.3 PSIG vacuum at 160°F to 180°F with adequate safety provisions to assure preventing excess temperature or collapsing from excess vacuum may be used as a suitable substitute.

g. Re-x-ray assembly to determine that all moisture has been removed.

h. When all moisture has been removed from assembly or x-ray reveals no moisture present, repair any damage in accordance with applicable procedures outlined in this technical order.

i. Test and seal all assemblies as follows:

(1) Connect air line to fitting (Part No. AN815-3).

(2) Pressurize assembly between 10 to 12 PSIG with dry oil free air and submerge in water to locate leaks.

NOTE

For vertical stabilizer lower fin tip, fabricate rubber wedges to plug wave guide and IFF antenna openings when panel is to be submerged in water.

(3) Mark all leaks with black grease pencil or suitable substitute.

REPAIR PROCEDURE

a. Determine exact area of separation by rapping outer surface of rudder with a coin. **(STEP 1)**

b. Outline exact area of separation with a pencil and measure area.

c. Remove rudder in accordance with T.O.1F-106A-2-7-2-1.

NOTE

IF AREA OF SEPARATION EXCEEDS 6 SQUARE INCHES, REPLACE RUDDER.

d. Drill holes through skin in area where separation has occurred. Use a No. 40 (0.098") drill. Holes should be drilled one inch apart throughout area of separation. **(STEP 2)**

CAUTION

DO NOT ALLOW DRILL TO PENETRATE MORE THAN 1/16 INCH THROUGH SKIN TO PREVENT DAMAGING HONEYCOMB SEGMENTS.

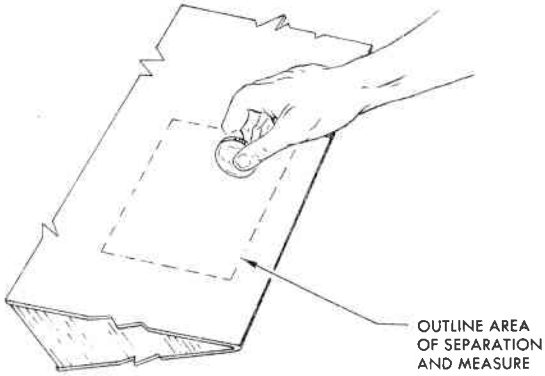
e. Mix adhesive. Refer to Table II-VIII for mixing procedure. Use one of the following adhesives: Epon VIII, Epon 3119 or EC1469.

f. Use a sealant gun with a small tip or a large type hypodermic needle and inject enough mixed adhesive to completely saturate area of separation and flush the drilled holes. **(STEP 3)**

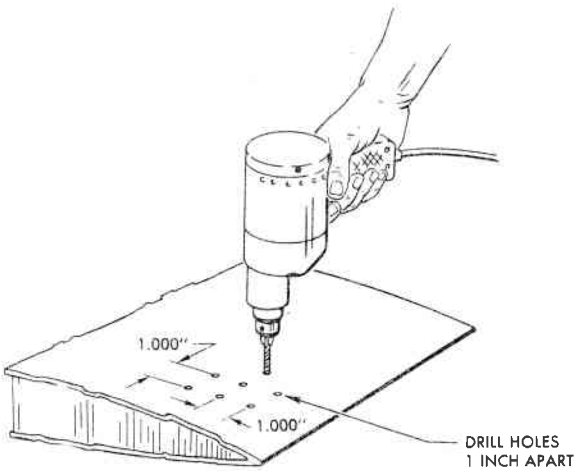
NOTE

IF AREA OF SEPARATION IS SMALL (2 SQUARE INCHES OR LESS) ADHESIVE MAY BE INJECTED FIRST WITH ONE NEEDLE FOLLOWED BY THE ACCELERATOR IN ANOTHER NEEDLE. THIS METHOD WILL ELIMINATE PRE-MIXING. HOWEVER, PROPER MIXTURE RATIOS BETWEEN ADHESIVE AND ACCELERATOR MUST BE OBSERVED.

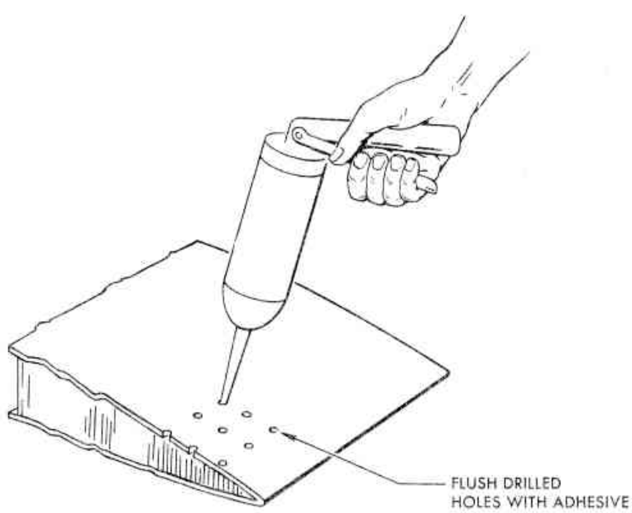
g. Clamp a block over repair area and allow adhesive to cure for time recommended by the manufacturer. **(STEP 4)**



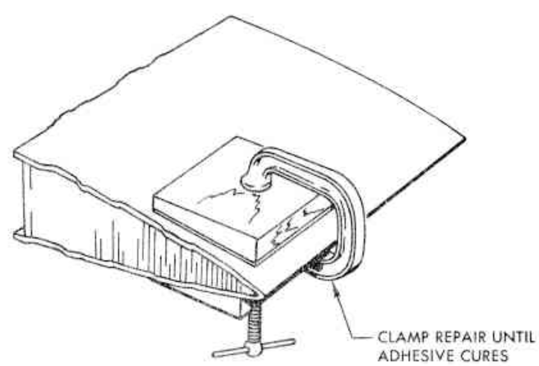
STEP 1



STEP 2



STEP 3



STEP 4

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Figure 3-14. Rudder Honeycomb Separation Repair

(4) Seal all leaks with sealant MIL-S-81733 class B 1/2. Surfaces to be cleaned with clean cloth and methyl-ethyl-ketone prior to applying sealant.

NOTE

To assure that sealant penetrates into all openings, apply sealant while assembly is subjected to a slight vacuum (approximately 7 PSIG). After sealant has been cured, recheck assembly (submerged) and repeat sealing process until all leaks are sealed.

j. Remove union fitting (Part No. AN815-3) and install plug machine thread FSN 4730-528-4871 or suitable substitute with sealant MIL-S-81733 class B.

k. To identify sealed fin tip panel, paint or stencil "SEALED" and date on the upper rib.

l. To identify sealed rudder paint or stencil date and a red dot in center of bottom rib.

CAUTION

Position light, IFF antenna access door, wave guide antenna slot, and forward channel must be sealed, using sealant, MIL-S-81733.

m. Reinstall fin tip panel and rudder in reverse order of removal.

3-47. Vertical Stabilizer – Sealing to prevent migration of foreign objects into rudder controls area.

3-48. Seal all openings, including structural mismatches, tooling holes, lightning holes, and holes caused by removal of equipment in the following areas: (1) Rib at WL 65.30 between spar 4 and 5 (Door 161). (2) Rib at WL 59.40 between spar 4 and 3 (Door 157). (3) Rib at WL 49.50 between spar 3 and 1 (Door 153). (4) Rib at WL 36.50 between spar 3 and 2 (Door 153). (5) Leading edge spar between WL 36.50 and WL 121.71. (6) Along spar 3 between WL 49.50 and WL 59.40. (7) Along spar 4 between WL 59.40 and WL 65.30. All holes 1/2 inch in diameter or less will be sealed with sealant MIL-S-8802. All holes larger than 1/2 inch in diameter will be sealed by fabricating appropriate size patch plate from QQ-A-250/5 (2024T3) 0.040 thick. Attach plate with a minimum of 4 each MS20600AD4 rivets. Apply sealant MIL-S-8802 to faying surfaces of patch plates prior to installation.

NOTE

Perform one time inspection for debris prior to sealing.

3-49. JIGS.

3-50. Figure 3-16 shows the master tooling fixture for the vertical fin assembly.

3-51. Packing and Crating — Fin Tip and Rudder.

3-52. Figures 3-17 and 3-18 illustrate the approved packing and crating methods for shipment of the rudder and fin tip. These containers are made in accordance with Specification MIL-C-25731, Type 1, Class 1 and 2. Class 1 is for air shipment; Class 2 is for domestic shipment. Specifications MIL-P-116 and MIL-B-121 explain the approved methods and materials used in packing these items. Crates shall be marked in accordance with Specification MIL-STD-129, the words "Removable End," and "Reusable Crate" shall be stenciled in their proper places. See table 1-LII for center of balance data.

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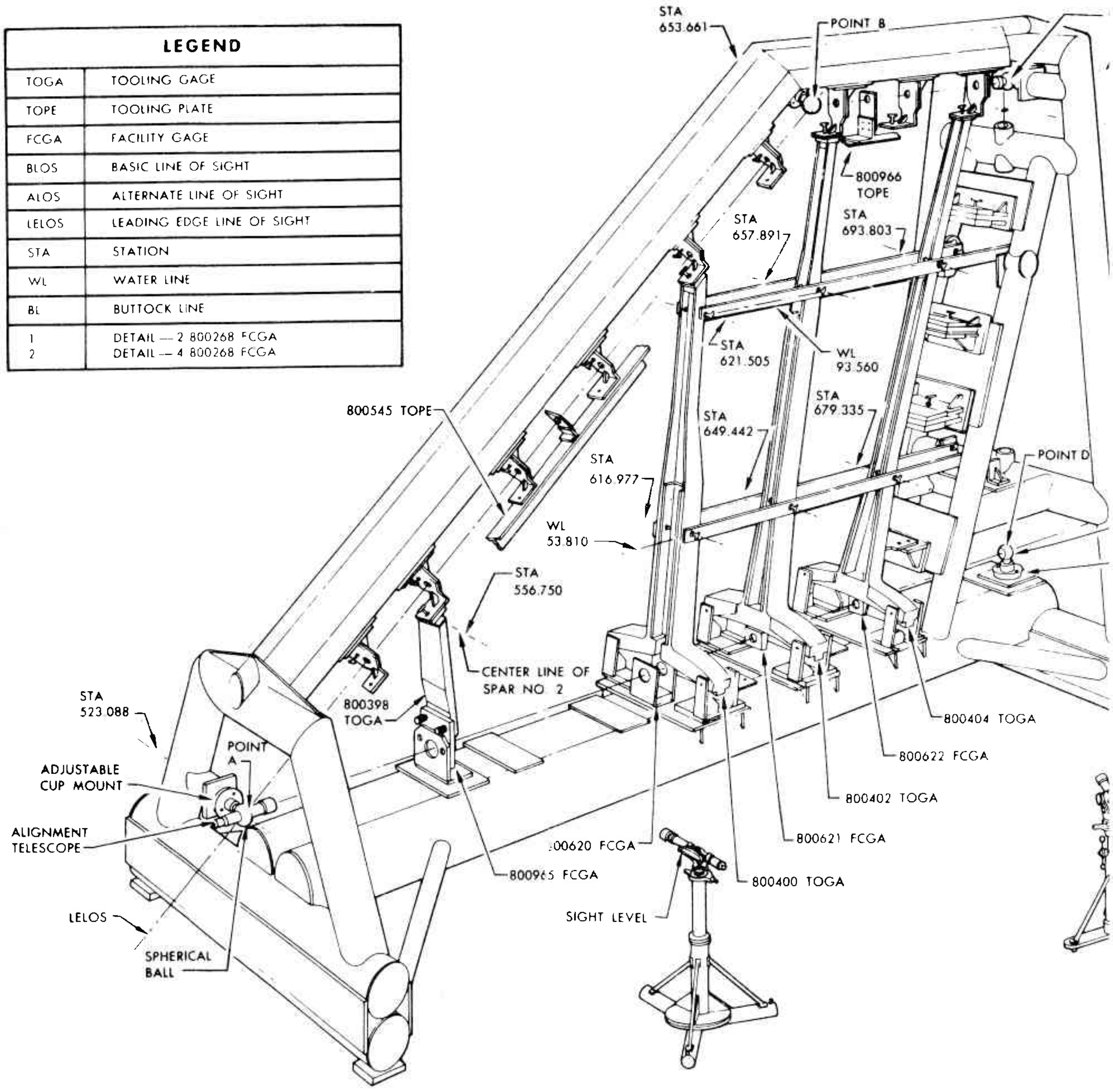
All data on page 3-21 and 3-22 including figures 3-15 and 3-15A Deleted.

LEGEND	
TOGA	TOOLING GAGE
TOPE	TOOLING PLATE
FCGA	FACILITY GAGE
BLOS	BASIC LINE OF SIGHT
ALOS	ALTERNATE LINE OF SIGHT
LELOS	LEADING EDGE LINE OF SIGHT
STA	STATION
WL	WATER LINE
BL	BUTTOCK LINE
1 2	DETAIL — 2 800268 FCGA DETAIL — 4 800268 FCGA



GAGES USED FOR MASTERING FIXTURE.

a. **Tooling Plate, 800545.**
b. **Tooling Plate, 800966.**
c. **Tooling Gage 800398 with Facility Gage 800965.**
d. **Tooling Gage 800400 with Facility Gage 800620.**
e. **Tooling Gage 800402 with Facility Gage 800621.**
f. **Tooling Gage 800404 with Facility Gage 800622.**

OPTICAL TOOLING EQUIPMENT REQUIRED

a. Two jig transits.
b. Two alignment scopes.
c. One sight level.
d. Two $2\frac{1}{4}$-inch diameter targets with light sources.
e. Two $3\frac{1}{2}$-inch diameter spherical balls.
f. Three $1\frac{1}{2}$-inch diameter plastic targets.
g. One auto-reflection mirror.
h. One set of 30 foot inside micrometers.

a. To be
b. To as
No. 4
c. To lin
No. 5

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R
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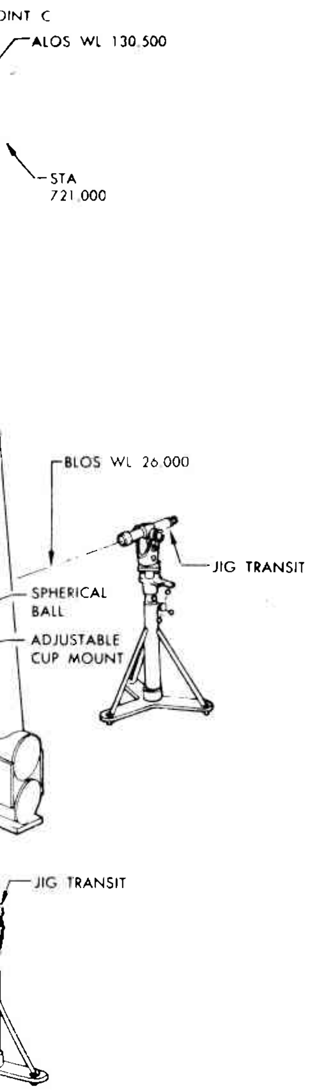
06 03 215 IC



TOOL FUNCTION

...sed as a master mating fixture. ...emble and drill fin spars No. 2, No. 3, ...nd No. 5 and leading edge spar No. 1. ... bore the rudder hinge fittings on spar

...TE
...ER TO SECTION I FOR USE OF OPTICAL ...UIPMENT

MASTERING PROCEDURE FOR MASTERING FIXTURE

NOTE

STEPS A AND B ESTABLISH POINT A AT STATION 523.086 AND POINT D AT STATION 721.000. THE HORIZONTAL ALIGNMENT BETWEEN POINTS A AND D ESTABLISHES WATER LINE 26.000, THE BASIC LINE OF SIGHT.

a. Level fixture as follows:
   (1) Set up sight level away from and to one side of fixture.
   (2) Level fixture and establish a horizontal alignment of points A and D.
   (3) Refer to Section I for leveling procedures using the sight level.

b. Establish the linear distance between points A and D by using the 30-foot inside micrometer and measuring the distance between points A and D, as shown on sheet 2.

c. Establish point C in a vertical plane and 90 degrees to the basic line of sight as follows:
   (1) Set up one jig transit away from and to one side of point D to establish a true vertical plane between points D and C.
   (2) Set up the second jig transit in line with the basic line of sight at the aft end of the fixture to hold point C in a horizontal plane with points A and D.
   (3) Refer to Section I for procedure on leveling the jig transit.
   (4) To gain the proper dimension between points D and C, use the 30-foot inside micrometer as shown on sheet 2. This establishes point C at station 721.000 and water line 130.500.

d. Establish point B in horizontal plane with point C and perpendicular to the basic line of sight as follows:
   (1) Mount the sight level away from and to one side of the fixture.
   (2) Mount a jig transit at the aft end of the fixture and in line with the basic line of sight.
   (3) Using the 30-foot inside micrometer, establish point B at station 655.661 and water line 135.500.
   (4) Using the 30-foot inside micrometer, measure the distance between points A and B. If this dimension checks with the dimension given on sheet 2, geometry of fixture is complete.

e. Install tooling and facility gages for spars No. 2, 3, 4 and 5 and leading edge spar No .1.
   (1) Mount an alignment telescope with a spherical ball at point A.
   (2) Mount the 2¼ inch diameter targets with light source in spherical balls at point B.
   (3) Align scope at point A with target at point B. Refer to Section I for use of alignment telescope.
   (4) Mount tooling angle, with facility gages attached, on the fixture and align facility gages approximately with basic line of sight.
   (5) Install 1½-inch plastic target on facility gages and align facility gages with basic line of sight.
   (6) Install the facility gage details, -2 and -4 to establish the proper angularity of spars No. 3, 4, and 5.
   (7) Mount the alignment telescope with a spherical ball at point C.
   (8) Mount the 2¼-inch diameter target with light source in the spherical ball at point B.
   (9) Align telescope at point A with target at points B and D.
   (10) Align telescope at point C with targets at points B and D.
   (11) Install the 800966 tooling plate in the fixture and pin as shown to top of spars No. 4 and 5.
   (12) Install the 800545 tooling plate in the fixture and pin as shown to top of spar No. 3.
   (13) Install the 1½-inch plastic targets in tooling plates 800545 and 800966.
   (14) Using an alignment telescope at point A, align tooling plate 800545 with target at point B.
   (15) Using alignment telescope at point C, align tooling plate 800966 with target at point B.
   (16) Index tooling gage 800398 to tooling plate 800545 and bring into line of sight between points A and B. Control the vertical and horizontal station planes by autoreflection (see figure "Principle of Autoreflection" in Section I.

Figure 3-16. Master Tooling — Vertical Fin (Sheet 1 of 2)

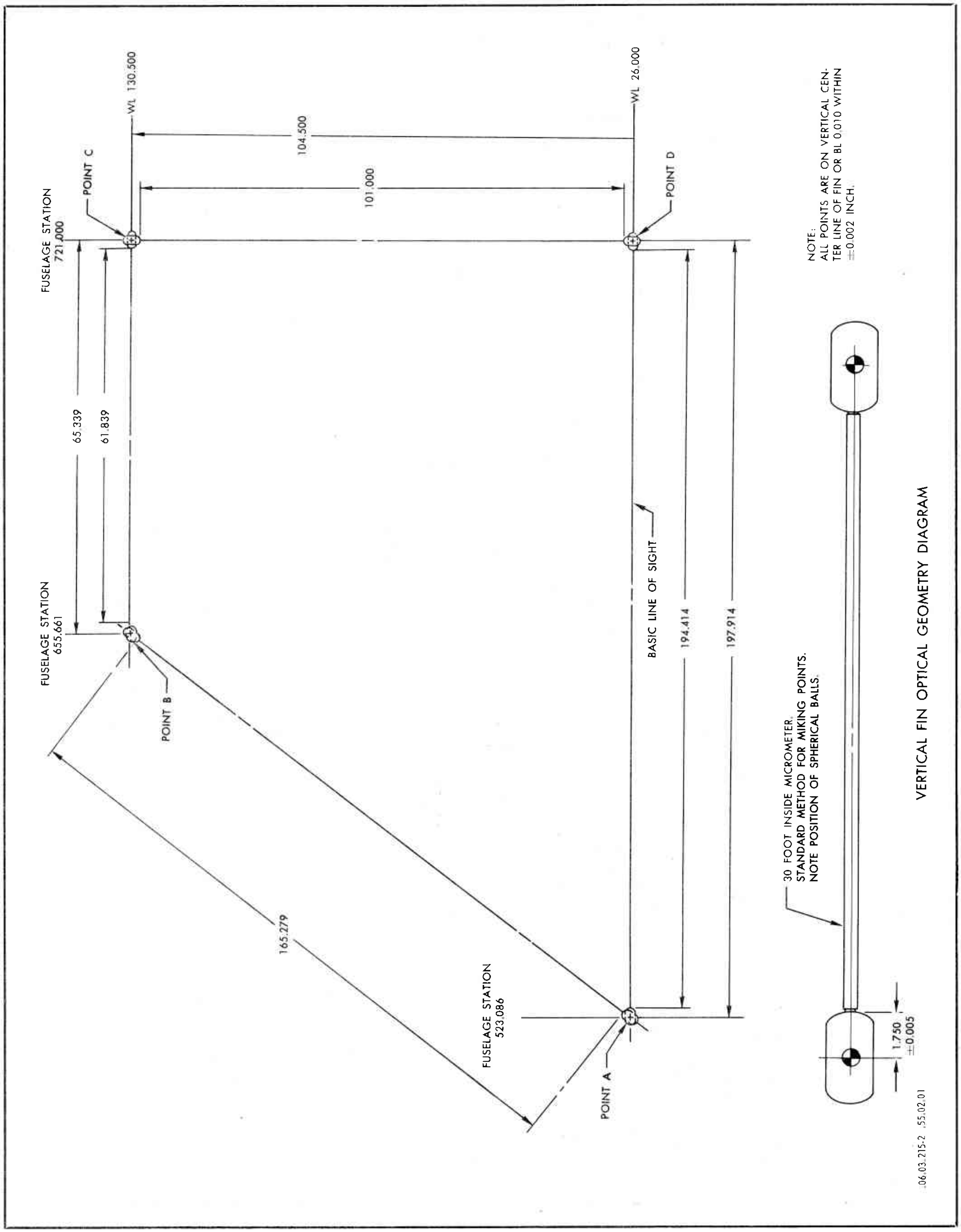
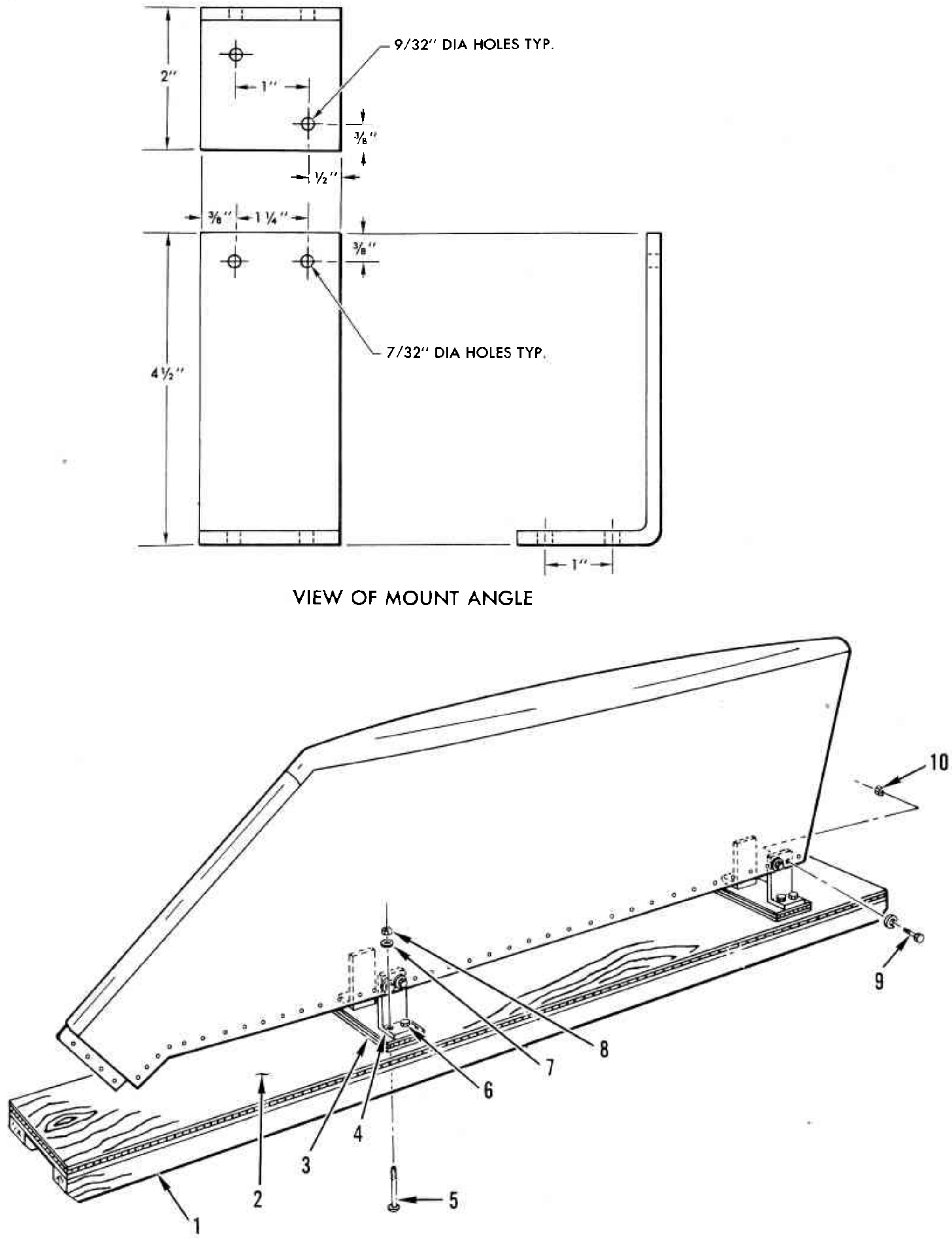


Figure 3-16. Master Tooling — Vertical Fin (Sheet 2 of 2)



MATERIALS				
NAME	DESCRIPTION	TYPE	PART NO.	NO. REQUIRED
SKID	1-¾" x 1-¾" x 87-½"	LUMBER	1	2
FLOOR	¼" x 8" x 87"	PLYWOOD	2	1
LOAD BEARING MEMBER	¾" x 3-⅝" x 6-½"	LUMBER	3	2
MOUNT ANGLE	⅛" x 2" x 6-½"	STEEL	4	4
CARRIAGE BOLT	¼" x 3"	STEEL	5	4
MACHINE BOLT	¼" x 1-½"	STEEL	6	4
WASHER	¼"	STEEL	7	4
NUT	¼"	STEEL	8	8
BOLT	AN507-1032-10	STEEL	9	8
NUT	AN364-1032	STEEL	10	8

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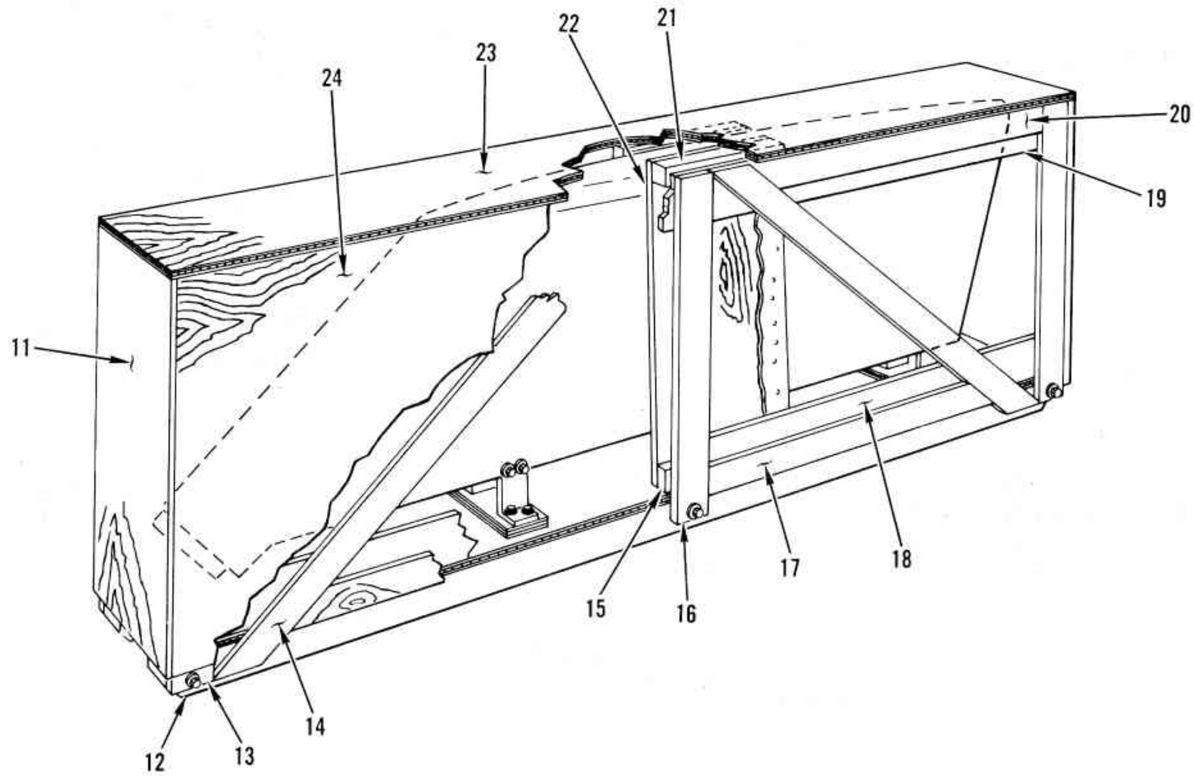
Figure 3-17. Packing and Crating — Fin Tip (Sheet 1 of 2)



MATERIALS				
NAME	DESCRIPTION	TYPE	PART NO.	NO. REQUIRED
SHEATHING	¼" x 9-¾" x 24"	PLYWOOD	11	2
LAG SCREW	5/16" x 2"	STEEL	12	6
WASHER	5/16"	STEEL	13	6
DIAGONAL CLEAT	¾" x 3" x 49"	LUMBER	14	4
BLOCK	¾" x 1" x 8"	LUMBER	15	2
MOUNT ANGLE	⅛" x 2" x 6-½"	STEEL	16	4
FILLER CLEAT	¾" x 1-¾" x 30"	LUMBER	17	4
LONGITUDINAL CLEAT	¾" x 3" x 87"	LUMBER	18	2
FILLER CLEAT	¾" x 1-¾" x 32-½"	LUMBER	19	4
LONGITUDINAL CLEAT	¾" x 3-⅝" x 87"	LUMBER	20	2
BLOCK	1-¾" x 1-¾" x 8"	LUMBER	21	2
HOLDING FIXTURE	¼" x 8" x 24"	PLYWOOD	22	2
SHEATHING	¼" x 9-¾" x 87-½"	PLYWOOD	23	1
SHEATHING	⅛" x 23-¾" x 87"	PAPER-OVERLAID VENEER	24	2

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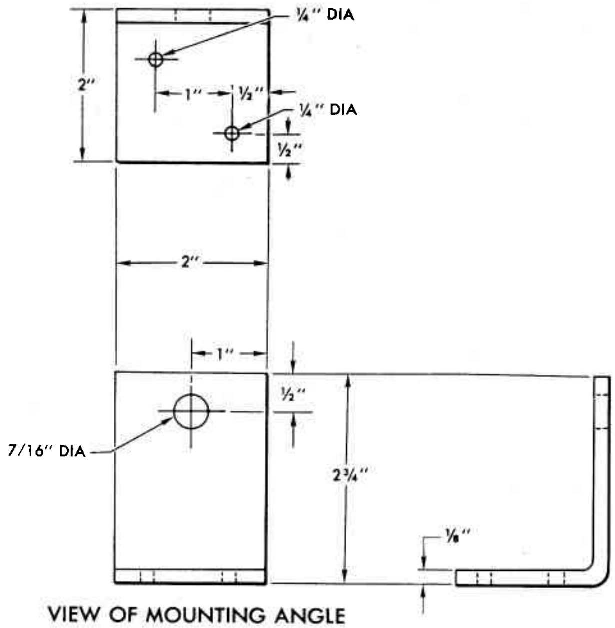
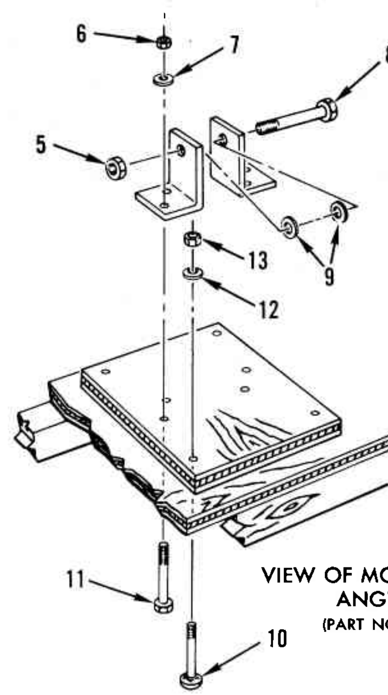
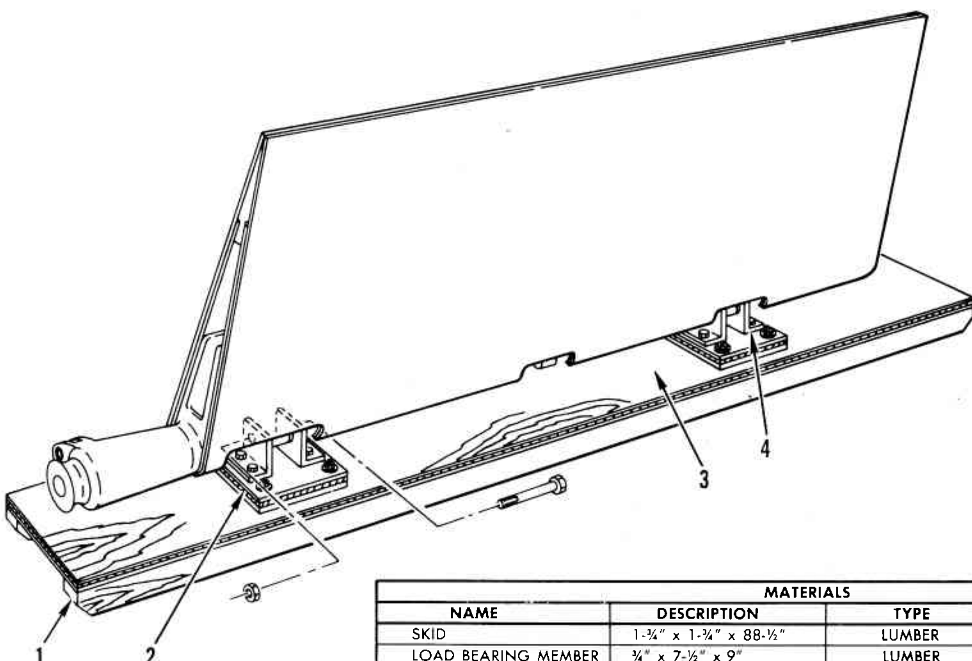
Figure 3-17. Packing and Crating — Fin Tip (Sheet 2 of 2)



VIEW OF MOUNTING ANGLE
(PART NO. 4)



VIEW OF MOUNTING ANGLE
(PART NO. 4)



MATERIALS				
NAME	DESCRIPTION	TYPE	PART NO.	NO. REQUIRED
SKID	1-¾" x 1-¾" x 88-½"	LUMBER	1	2
LOAD BEARING MEMBER	¾" x 7-½" x 9"	LUMBER	2	2
FLOOR	¼" x 10-½" x 88"	PLYWOOD	3	1
MOUNT ANGLE	⅛" x 2" x 4⅞"	STEEL	4	4
NUT	AN-365-720	STEEL	5	2
NUT	¼"	STEEL	6	8
WASHER	¼"	STEEL	7	8
BOLT	AN-22	STEEL	8	2
WASHER	⅜"	PHENOLIC	9	2
CARRIAGE BOLT	¼" x 3"	STEEL	10	8
MACHINE BOLT	¼" x 1-½"	STEEL	11	8
WASHER	¼"	STEEL	12	8
NUT	¼"	STEEL	13	8
SHEATHING		PAPER-OVERLAID VENEER	14	2

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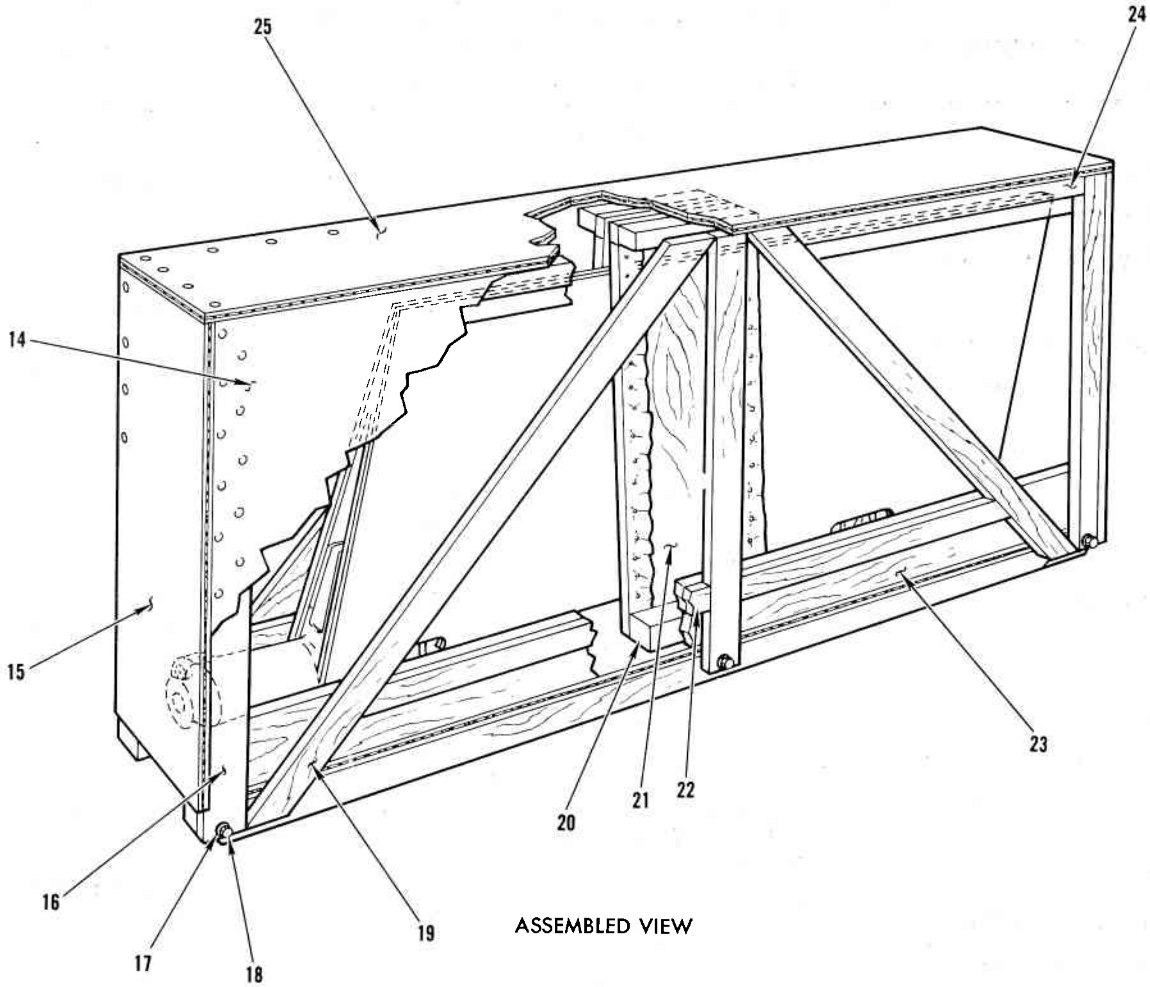
Figure 3-18. Packing and Crating — Rudder (Sheet 1 of 2)



MATERIALS				
NAME	DESCRIPTION	TYPE	PART NO.	NO. REQUIRED
SHEATHING	¼" x 12-¼" x 31-¼"	PLYWOOD	15	2
VERTICAL CLEAT	¾" x 3" x 33"	LUMBER	16	6
LAG SCREW	5/16" x 2"	STEEL	17	6
WASHER	5/16"	STEEL	18	6
DIAGONAL CLEAT	¾" x 3" x 52"	LUMBER	19	4
BLOCK	1-¾" x 1-¾" x 12"	LUMBER	20	4
HOLDING FIXTURE	¼" x 12" x 31-¼"	PLYWOOD	21	2
FILLER CLEAT	¾" x 1-¾" x 32"	LUMBER	22	4
LONGITUDINAL CLEAT	¾" x 3" x 88"	LUMBER	23	2
FILLER CLEAT	¾" x 1-¾" x 34"	LUMBER	24	4
SHEATHING	¼" x 12-¼" x 88-½"	PLYWOOD	25	1

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Figure 3-18. Packing and Crating — Rudder (Sheet 2 of 2)

TABLE 3-I

Negligible Damage Limits — Tail Group

COMPONENT	TYPE AND CLASS OF DAMAGE ALLOWED AFTER REWORK					REMARKS
	Scratch	Nick	Dent	Hole	Crack	
FIN STRUCTURE						
Plating	I	I	*	*	*	Honeycomb Core.
Leading Edge Spar Rail	I	I	*	I	*	
Leading Edge Spar Web	II	II	I	I	I	
Spar No. 2	I	I	*	I	*	Titanium Forging.
Flange Spar Nos. 3, 4, and 5	I	I	*	I	*	
Web Spar Nos. 3, 4, and 5	II	II	I	I	*	
Rib Flange	II	II	II	II	I	
Rib Web	III	III	III	III	II	
Rib Stiffeners	III	III	I	III	II	
Canted Rib—Bay 3	I	I	*	*	*	
Hinge Fittings		Refer to an Aeronautical Structural Engineer				
LEADING EDGE STRUCTURE						
Plating	I	I	*	*	*	
Doublers	I	I	*	*	*	
Ribs	I	III	II	II	II	
FIN TIP STRUCTURE (LOWER PANEL)						
Plating	I	I	*	*	*	Honeycomb Core.
Channel Leading Edge	II	II	I	I	I	
Wedge Trailing Edge	III	III	II	II	II	
Lower Channel	I	I	I	I	*	
FIN TIP STRUCTURE (UPPER PANEL)						
Laminate Skin, Fiberglass.	—	—	*	*	*	Refer to paragraph 3-21 for limitations.
Forward Rib	II	II	I	II	II	
Lower Ribs	II	II	I	II	II	
RUDDER STRUCTURE						
Plating	I	I	*	*	*	Honeycomb Core.
Front Spar	I	I	*	*	*	
Upper Rib	III	III	II	II	II	
Lower Rib	I	I	*	*	*	
Trailing Edge Wedge	III	III	II	II	II	
Hinge Fittings		Refer to an Aeronautical Structural Engineer				
Actuating Arm		Refer to an Aeronautical Structural Engineer				

*Component must be repaired or replaced

3-53. VERTICAL STABILIZER ALUMINUM HONEYCOMB PANEL REPAIRS – DEPOT LEVEL ONLY.

3-54. General Information - Materials:

a. Honeycomb core, aluminum alloy 5052, 1/8 inch hexagonal cell, 0.0002 inch foil, meeting specification MIL-C-7438D, Type 1D, non-perforated:

(1) Size 28 inch x 96 inch x 0.329 ± 0.005 inch, FSN 5680L9642002049.

(2) Size 28 inch x 96 inch x 0.319 ± 0.005 inch, FSN 5680L9689152049.

b. Aluminum sheet, 2024 T86, bare, 0.020 inch by 48 inch by 144 inch, FSN 9535-684-1429.

c. Aluminum sheet, 2024 T86, bare, 0.025 inch by 48 inch by 144 inch, FSN 9535-684-1429.

d. Adhesive Film, metal to core and metal to metal.

(1) Requirements:

(a) Usage. This repair procedure is basically written for use of the Aerobond 3030 (see paragraph 3-54. d. (2)) adhesive. The adhesives listed in paragraph 3-54. d. (3) are alternates for use when the paragraph 3-54.d. (2) adhesive is not available. If paragraphs 3-54. d. (2) or 3-54. d. (3) materials are not available, and a work stoppage occurs, the substitutes listed under paragraph 3-54. d. (4) may be used. These substitutes should not be normally used in regular production. If adhesives other than the one listed under paragraph 3-54. d. (2) are used, changes will have to be made in the process to accomodate the different cure requirements.

(b) Storage and Handling. The adhesives listed under paragraph 3-54. d. are perishable and require low temperature shipment and storage (0°F or less).

(c) Adhesive Properties. The adhesives listed in paragraphs 3-54. d. (2) and 3-54. d. (3) are considered serviceable if they meet the requirements of MIL-A-25463, Type II and MMM-A-132, Type II and in addition, develop a metal-to-metal T peel strength in excess of 10 lbs. per inch width.

(d) Age Control. Age control shall be based on a 6 month shelf life for storage at 0°F. Testing of incoming adhesives for conformance to the requirements for the adhesive shall be at the descretion of Quality Control.

(2) Principal Adhesive: Aerobond 3030, Adhesive Engineering Company, 1411 Industrial Road, San Carlos, California 94070. Order in a nominal 0.090 lb./sq. ft. density.

(3) Alternates (order in a nominal 0.090 lb/sq. ft. density).

(a) Adhesive films from Adhesive Engineering.

1 Aerobond 3034.

2 Aerobond 3037.

3 Aerobond 3038.

(b) Plastilock 729, B.F. Goodrich Adhesive Products, 500 S. Main Street, Akron, Ohio 44318.

(4) Substitutes.

NOTE

These materials are normally stocked and will meet Convair Specification 8-01319, but will not meet the requirements of paragraph 3-54. d. (1) (c). Serviceability requirements shall be based on the requirement of paragraph 4.2.2. of Convair Spec. 8-01318.

(a) FR-7035, Fiber Resin Corporation, 170 West Providence Avenue, Burbank, California 91502.

(b) EA 9601, Hysol Division, The Dexter Corporation, 2850 Willow Pass Road, Pittsburg, California 94565.

e. Core edge potting compound, low expansion type.

(1) Requirements.

(a) Usage. This repair procedure has been prepared for use of the film type materials listed in paragraph 3-54.e.(2). Paste type materials are listed in paragraphs 3-54.e.(3) and 3-54.e.(4) and may be used in place of the paragraph 3-54.e.(2) materials if desired for convenience and for a reduction in production time. If the paste type materials are used, changes will have to be made in the process to accomodate the different cure requirements.

(b) Storage and Handling. The potting compounds listed in paragraph 3-54. e. (2) are perishable and require low temperature shipment and storage (0°F or less).

(c) Physical Properties.

1 Core Splice Films (paragraph 3-54. e. (2) materials) only.

a A standard 1/2 inch overlap shear strength equal to or in excess of 1050 psi at -65°F and at 77°F, and 900 psi at 216°F. (Aluminum 2024 T81 adherents, 0.065 inch thick by 1 inch wide, a 0.050 inch thick by 0.4 x 1.0 inch bond line, vacuum bagged and cured at the conditions of paragraph 3-56. m.)

b Cured thickness (free expansion during cure at 20 inches of vacuum following the paragraph 5.13 cure process) between 0.050 and 0.090 inches.

2 Core splice films and pastes (paragraphs 3.54. d. (2), 3-54. d. (3) and 3-54. d. (4) materials). When used as a core splice material to splice two sections of core together to form a continuous sandwich panel, and then assembled with the splice line perpendicular to the long panel direction and located halfway between the load and support axis, the resulting cured panel when tested following MIL-A-25462, shall develop a core shear strength equal to or better than that required by paragraphs 4.2.2.2 and 4.2.4.2 of Convair specification 8-01319.

(d) Age Control. Age control for the film type potting compounds of paragraph 3-54. d. (2) shall be based on 6 month shelf life for storage at 0°F. Age control for paragraphs 3-54. d. (3) and 3-54. d. (4) materials shall be 6 months at 77°F testing of incoming potting compounds for conformance to the requirements of paragraph 3-54. d. (1) (c) shall be at the discretion of Quality Control.

(2) Principal Core Splice Films:

(a) AF-3002, 3M Company, Adhesives, Coatings and Sealers Division, 3M Center, St. Paul, Minnesota 55101.

(b) Plastilock 654GD, B.F. Goodrich Adhesive Products, 500 S. Main St., Akron, Ohio 44318.

(c) Aerobond 3026, Adhesive Engineering Company, 1411 Industrial Road, San Carlos, California 94070.

NOTE

Request a nominal thickness of 0.050 inch.

(3) Paste, no expansion, two component syntactic foam type.

(a) FR-13 Paste and No. 8 Hardener, Fiber-Resin Corporation (see paragraph 3-54. d. (4) (a)).

(b) Epocast 1310 Mod 2 and No. 9228 Hardner, Furane Company, 5121 San Fernando Road, West Los Angeles, California 90039.

(4) Pourable liquid, expansion, one component foam type.

(a) Thermofoam 3051, Adhesive Engineering (see paragraph 3-54. d. (2)).

f. Cleaning Solvents.

(1) Methyl ethyl ketone, specification TT-M-261, FSN 6810-281-2762, 5 gallon can.

(2) Tetrachloroethylene (perchloroethylene), Specification 0-T-236, FSN 6810-819-1128, 5 gallon can.

g. Corrosive Chemicals.

(1) Pasa-Jell 105, metal surface cleaner and conditioner, SEMCO Division, PRC Corporation, PO Box 61037, Los Angeles, California 90061.

(2) Sulfuric Acid, 96%, Specification 0-S-809, Type 1, Class I.

(3) Sodium Dichromate, Specification 0-S-595, FSN 6810-281-2686, 100 lb. bag.

h. Tank Type Solutions.

(1) Oakite Stripper SA, Oakite Products, Inc., So. Valley Road, Berkeley Heights, N.J. 07922.

(2) Aluminum Deoxidating Solution.

i. Release Agents (Purchase as Spray Cans).

(1) Mold Release (one of the following):

(a) Number 1711 Release Agent, Contour Chemical Company, 4 Draper Street, Woburn, Mass 01801.

(b) FreKote-33, FreKote Incorporated, P.O. Box 825, Boca Raton, Florida 33432.

(c) Number 3070 Epoxy Mold Release, Crown Industrial Products Company, 198 Stateline Road, Hebron, Illinois 60043.

(2) Fluorocarbon Surface Release Agent (one of the following):

(a) Vydax AR, E.I. duPont de Nemours and Company, Petroleum Chemicals Division, Wilmington, Delaware 19898.

(b) CHR Rulon Spray No. 2, Connecticut Hard Rubber Company, New Haven, Connecticut 06509.

(c) Number 6075 Dry Fluorocarbon Lubricant, Crown Industrial Products Company, 198 Stateline Road, Hebron, Illinois 60034.

j. Identification Tag Adhesive, Specification MIL-A-8623, Type I, FSN 8040-270-8137, 1-quart kit or FSN 8040-900-6296, 1-pint kit.

3-55. General Information - Equipment.

a. Autoclave, Bldg. 655, with bondform cart and accessories.

b. Panel Bondforms (one with a flat surface and one with a curved surface to match the shape of Part Number 8-14152-3 and -4).

c. Vacuum Bag Materials. Vacuum bag film, sealing tape, glass cloth, bleeder cloth, hoses, etc. (see reference in paragraph 3-59. a. (5)).

d. Adhesive handling and layup facilities. Clean layup room with 0°F adhesive storage freezer (see reference in paragraph 3-59. a. (4)).

e. Standard FPL Etch System for Aluminum.

(1) Etch Solution Tank

WARNING

Spattering or a violent explosion may occur if water is added to sulfuric acid. Sulfuric acid should be added to water slowly and carefully with stirring. An exhaust system consistent with OSHA Standards and approved by base bio-environmental engineers must be provided.

(a) Set up a corrosion resistant, agitated and heated tank to contain the etch solution. Add sufficient sulfuric acid (paragraph 3-54. g. (2)), sodium dichromate (paragraph 3-54. g. (3)) and deionized water to make an initial solution as follows:

H_2O	30 parts by weight
H_2SO4	10 parts by weight
$Na_2Cr_2O_7$	4 parts by weight

The tank should be large enough to accommodate the frames with complete immersion.

(b) Set up heating controls on the tank to maintain a 155 + 5°F temperature during processing. (Heat may be turned off when tank is not in use.)

(c) During etch treatment of parts, the tank must be under agitation to obtain a satisfactory and uniform etch.

(d) Add deionized water, acid and sodium dichromate as required to maintain solution within the following limits during etching of aluminum parts:

H_2SO_4	280 to 290 gms/liter
$Na_2Cr_2O_7$	27 to 75 gms/liter
Cl (Chloride)	0 to 0.36 gms/liter
$Cr_2(SO_4)_3$	0 to 40 gms/liter
SO_4 (Sulfate)	274 to 530 gms/liter

NOTE

Operating controls on the tank can be based on solution specific gravity, adding deionized water to the tank to make up for losses so that the specific gravity is between 1.18 and 1.20 at operating temperatures.

If solution limits can not be obtained by additions of chemicals, the solution should be partially dumped and a fresh solution made up.

(e) Tank shall not be released for production use until solution is aged and approved (based on adhesive bond strength of test specimens).

(2) Rinse tank or spray rinse area, (final rinse capability) with deionized water.

(3) Drying area (or oven at 160°F if available) to dry off rinse water.

f. Personnel protective equipment. Rubber gloves, clean lint-free cotton gloves, aprons, face masks or shields, protective hand creams and dust respirators.

g. Clean part drying and temporary storage racks (to hold parts in a clean area while they are drying).

h. Cleaning Supplies. Clean cotton rags (lint and oil free) or clean cheesecloth, wiping tissues, solvent dispensing cans (safety type) and small plastic wash bottles.

i. Pasa-Jell application area. Facilities including drain to industrial sewer, water hoses for washing off the applied Pasa-Jell and a supply of deionized water for a final rinse.

j. Vapor Degreaser. Operate with tetrachloroethylene (paragraph 3-54. f. (2)). Degreaser should be located close to the layup room.

k. Stripper Tank. Set up and operate per Local Procedures, "Shop Service Test of Solvent Based Structural Adhesive Stripper."

l. Aluminum Deoxidating Cleaner Tank. As per Local Procedures, "Cleaning, Chromate Conversion Coating, Anodizing and Removing Anodic Coatings of Aluminum and Aluminum Alloys."

NOTE

The combined maximum limits of 160°F and a 6% solution concentration should be avoided as rapid discoloration of honeycomb core foil surfaces starts to occur at this condition.

3-56. PROCESS.

NOTE

- This process is abbreviated. Shop workers are assumed to be trained in structural adhesive bonding techniques.
- Absolute identification of part to aircraft must be retained since each panel has a unique hole pattern and must go back on the original aircraft in the same location.

a. Remove existing inner and outer skins from the panels. Use dry ice on skins to aid in the removal of skins. Identify the frame with the aircraft by engraving the aircraft tail number and the part serial number on an exposed side edge of the frame.

b. Remove old honeycomb core from frame and discard.

c. Remove old adhesive from frame. Use methods in paragraphs 3-56. c. (1) or 3-56. c. (2) as available. Avoid method in paragraph 3-56. c. (1) if possible.

(1) Move frame to the grinding area and mechanically abrade off the old adhesive. Do not gouge the metal frame or remove metal to where the drawing dimension tolerances are exceeded. Do not grind off identification numbers.

WARNING

Wear approved face shield and dust respirator while abrading or sanding off the old adhesive.

(2) Stripping of old adhesive from frames, using the Oakite S-A soak process (paragraph 3-55. k.).

a. Soak degreased frames for 24 hours or longer as required.

NOTE

The tank should be large enough for complete immersion of the frames in the stripper, below the solution-seal layer interface. If a smaller tank is used, and only part of the frame is immersed at any given time, a corrosive condition will occur at both the air-seal layer and seal layer-solution interfaces. The rate of attack at the solution-water interface line is on the order of 1 to 3 mils (0.001")/day. (Within the solution the attack is neglibile, about 10 mils/year). Therefore, if small tanks are used, the frame should be moved every few hours to minimize the attack at any given spot.

(b) Once or twice every shift remove frame from solution and inspect for removal of adhesive and for the degree of corrosive attack on the metal by the stripper. Use aluminum scrapers to remove loosened core and adhesive residues. Wear rubber (neoprene) gloves to handle frames.

NOTE

The stripper attacks the primer underneath the adhesive layer and only softens the outer adhesive layer. The softening action is slow and may require immersion over a period of days. Since the stripper is corrosive, the frames must be regularly inspected for corrosion attack.

(c) Wash off stripper residues.

(d) If frame is discolored (smut), immerse briefly in one of the aluminum deoxidizing solutions to brighten the surface. Rinse off deoxidizing solution.

(e) Dry frame before transporting out of shop.

d. Layout. Cut and trim new outer and inner skins to fit the frame. Drill six holes in both skins for alignment during the bonding process.

e. Cut and trim honeycomb core (paragraph 3-54. a.) to fit frame openings. Observe drawing requirements for core ribbon direction and dimensional tolerances.

f. Preliminary cleaning of frames and skins (3-56. f. (1) or 3-56. f. (2)).

WARNING

Perform solvent cleaning operations in an approved cleaning cabinet or in a well ventilated area. Avoid prolonged breathing of vapors. Avoid eye and repeated skin contact. Keep solvents away from sparks and flames.

(1) Hand wipe method. Hand wipe frames and skins with MEK (paragraph 3-54. f. (1) and 3-58.) wet cloths or wiping tissues to remove mill markings, ink, grease or dust. Abrade off persistent deposits if MEK wipe treatment is not successful in removing the deposit.

(2) Solution Cleaning.

(a) Vapor degrease skins and frames.

(b) Solution clean parts in an aluminum deoxidizing cleaner (paragraph 3-55. j.).

(c) Rinse parts and inspect for contamination by the water break test method. Remove any spot contamination and repeat the cleaning and rinsing steps.

(d) Proceed to solution etch if available, otherwise dry part and route back to work area.

g. Surface Conditioning. Use method in paragraph 3-56. g. (1) if available, otherwise use method in paragraph 3-56. g. (2).

(1) Immerse outer skin and frame in the standard acid etch solution (paragraph 3-55. e.) for a period of from 12 to 16 minutes. Then remove and rinse in the rinse tank and finally rinse with deionized water. (See paragraphs 3-58. b. and 3-58. c.)

(2) Mechanically abrade or roughen the inner side of the outer skin. Wipe off any sanding residues with solvent

(paragraph 3-54. f.) and wiping cloths or tissues. (The frame does not need to be roughened.) Treat inner surface of skin and frame with Pasa-Jell 105 (paragraph 3-54. g. (1)). (See paragraphs 3-58. b. and 3-58. c.)

(3) Inspect the surface for contamination using the water break test immediately after the final rinse with tap water.

(4) Move the wet parts to a drying rack and let dry naturally in a dust-free, oil and contamination free area.

h. Inspect core parts for core burrs and defects (see reference, paragraph 3-59. a. (2)). Moderate and heavy burr should be corrected. Distorted cells should be straightened out. Parts with broken foil or with node separations interior to an outside area of two cells from the edges should be discarded and a new part obtained.

i. Clean honeycomb parts by immersion in a vapor degreaser for 30 seconds or more. Use only a clean degreaser that has a clean boiling pool. Remove to cool and re-imerse two or more times. Remove core and route to work area.

j. Move frame, outer skin and core parts to the layup room for adhesive application.

k. Apply adhesive to parts (see reference, paragraph 3-59. a. (4)).

(1) Inspect core sections for handling damage.

(2) Assemble core parts into frame using strips of edge to core adhesive (paragraph 3-54. e.).

NOTE

Adhesive film is unsupported and fractures easily when handled at low temperatures.

(3) Apply adhesive (paragraph 3-54. d.) and fit outer skin to honeycomb core and frame. Adhesive film should overlay core and frame structure. Avoid overlapping peices of adhesive film. (See paragraphs 3-58. f. and 3-58. g.)

(4) Insert temporary fasteners (previously coated with a mold release) through frame and outer skin to establish skin to frame alignment.

l. Vacuum Bag Panel to Bondform (see reference, paragraph 3-59. a. (5)).

(1) Bondform surface should have a realease agent on the surface. (See paragraph 3-58. d.)

(2) A fluorocarbon release agent may be applied lightly to the outer edges of the frame where adhesive bleed-out will occur during the cure cycle.

(3) Place panel so that the outer skin is against the bond form surface.

(4) Place one or more thermocouples against the bond-form adjacent to the adhesive bond line.

(5) The pink silicone treated bleeder cloth may be used directly over the open side of the core and panel.

m. Cure Adhesive in Autoclave.

(1) Re-establish vacuum on vacuum bag with autoclave vacuum system.

(2) Check vacuum bag for leaks and close off any located leaks.

(3) Roll cart and bondform into autoclave, connect up the thermocouples and close the door.

(4) Set pressure controller to 5 psi and pressurize autoclave.

(5) When pressure has reached 5 psi, vent vacuum system and vacuum bag to the atmosphere.

(6) Reset pressure controller to 15 psi.

(7) Start heating autoclave. Set temperature controller set point to 365°F.

(8) Observe heat up rate. Limit rate of rise to between 10° and 2°F per minute.

(9) When bondline temperature has reached 350°F, reset temperature controller to 350°F. Maintain steady state conditions (cure) for a period of 45 minutes.

(10) Following cure period, shut off heat and cool autoclave.

(11) Start releasing pressure when bondline temperature is below 180°F.

(12) When autoclave has cooled to 100°F or less, open door and remove bondform and cart.

CAUTION

Wear asbestos gloves when removing hot items from the autoclave.

n. Remove vacuum bag and temporary fasteners.

o. Transport partially bonded panel to work area.

NOTE

Be careful in handling the panel to avoid damaging the exposed honeycomb core surface.

p. Using the frame as a drill guide, drill attaching holes through outer skin. Remove all drillings.

CAUTION

Use extreme care in drilling through the frame to be sure that the drill is properly aligned with the existing hole at all times.

q. Check core surface for proper mating with inner skin. Sand down any high points. Note any depressions around the frame region that will require special buildup to avoid "steps" in the inner skin. Contact SM/ALC/MANCA methods to perform this buildup.

WARNING

Perform solvent cleaning operations in an approved cleaning cabinet or in a well ventilated area. Avoid prolonged breathing of vapors. Avoid eye and repeated skin contact. Keep solvents away from sparks and flames. Use approved personal protective equipment (eye goggles/face shield) when using compressed air. Maximum allowable air pressure for cleaning operations is restricted to less than 30 psi. Provide protection from flying particles when using compressed air. Do not direct airstream towards self or other personnel.

r. Inspect inner panel side for core burrs and defects (paragraph 3-59. d. (3)). Moderate and heavy burr should be corrected. Distorted cells should be straightened out if possible and broken foil areas potted. Contact for details. Remove any drilling chips with a light flow of clean air. Use a small stream of tetrachloroethylene (paragraph 3-54. f. (2)) to flush out any drill or sanding residues that appear to be adherent.

s. Wipe the frame surface to be bonded with clean rags or wiping tissues dampened with clean MEK (see paragraph 3-58).

t. Acid etch the inner skin.

u. Return the panel and treated parts to the layup room.

v. Apply film adhesive (paragraph 3-54. d.) to core and frame. Fit inner skin to panel using temporary fasteners to maintain alignment.

w. Vacuum bag panel to bondform per paragraph 3-56. l.

x. Cure panel per paragraph 3-56. m., only hold 350°F for one hour instead of 45 minutes indicated in paragraph 3-56. m. (9).

y. Remove vacuum bag, temporary fasteners, and transport panel to work area.

z. Using frame as a guide, drill attaching holes through inner skin. Remove all drilling chips.

CAUTION

Use extreme care in drilling through the frame to be sure that the drill is properly aligned with the existing holes at all times.

aa. Remove any adhesive flash and finish panel.

ab. Apply identification tag.

(1) Prepare an AN-7510-1-Rev. 9 identification tag with the following information:

AIRCRAFT MOD.	F-106
PART NO.	(obtain from drawing)
CONT NO.	(enter aircraft tail number from frame)
SERIAL NO.	(part serial number from frame engraved number)
MODIFICATION INCORPORATED	TO 1F-106A-3 (date, month-year rebuilt)

(2) Roughen up back side of tag and clean back side.

(3) Prepare a small quantity of adhesive (paragraph 3-54. j.) and bond tag to inner skin. (Adhesive is a room temperature curing epoxy adhesive.)

ac. Apply chemical conversion coating, MIL-C-5541, then prime and overcoat in accordance with T.O. 1-1-4 and T.O. 1-1-8.

3-57. Inspection or Testing.

a. Perform ultrasonic inspection of completed panels for delaminations only at the request of Quality Control. Panels will not normally require ultrasonic inspection. Allowable defects uncovered during inspection are as follows:

(1) No single unbonded area of more than one inch in diameter.

(2) Less than 10% cumulative bonding void on any straight line across the surface of the panel in any direction.

b. Defects found in panels shall be repaired in accordance with directions given in Sections 10 and 11 of this publication.

c. The adhesives, the parts, the cleaning processes and the bonding processes are to be controlled in accordance with the standards referenced in paragraph 3-59. a.

3-58. Health and Safety Notes.

a. MEK (paragraph 3-54. f. (1)) is hazardous and flammable. Tetrachloroethylene (paragraph 3-54. f. (2)) is hazardous, but is not flammable. T.O. 42A1-1-3 should be consulted for the proper handling and storage of these solvents.

b. Pasa-Jell 105 (paragraph 3-54. g. (1)), Oakite S-A (paragraph 3-54. h. (1)), and sulfuric acid (paragraph 3-54. g. (2)) are acidic corrosive chemicals. Any spills of these materials should be immediately washed away with water. Residues may be treated with sodium bicarbonate to neutralize any remaining acid. When handling these materials in bulk, wear rubber aprons, rubber or PVC gloves and protective face shields.

c. Pasa-Jell 105 (paragraph 3-54. g. (1)) and sodium dichromate (paragraph 3-54. g. (3)) are oxidizing chemicals and may cause fires if these materials are combined with organic solvents (MEK) or other combustible materials (wiping tissues and cloths).

d. The spray release agents (paragraph 3-54. i.) will form hazardous mists when applied. The mists from the paragraph 3-54. i. (1) materials may be explosive and should not be applied in the vicinity of ovens or electrical switch gear. The mists from all the release agents should not be breathed, and the agents preferably applied in a well ventilated area. If the area is not ventilated, organic vapor protective masks should be worn when the agents are applied.

e. MEK (paragraph 3-54. f. (1)), tetrachloroethylene and Oakite S-A (paragraph 3-54. h. (1)) are considered volatile materials. (The volatile components in Oakite S-A are sulfur dioxide, dichloromethane and trace amounts of mercaptans. A water layer on top of Oakite S-A must be maintained at all times to block the evaporation of these volatile ingredients.) Do not inhale the vapors from these volatile materials and avoid skin contact. Wear protective gloves and approved organic vapor protective masks.

f. The adhesive materials listed in paragraphs 3-54. d. and 3-54. e. contain chemicals and volatile ingredients known to be toxic and irritating to the skin and eyes. These adhesives can cause dermatitis if skin contact occurs. Avoid skin contact with these adhesives by using gloves.

g. Adequate local ventilation should be provided where the volatile materials discussed in paragraph 3-58. e. are handled on work benches. If possible these materials should not be used in the layup room.

(1) The ventilation system shall provide lateral exhaust ventilation of 100 feet per minute at the work edge of the work bench.

(2) The type of ventilation system shall be coordinated with Bioenvironmental Engineering prior to planning and installation.

h. At the end of the day or after completion of the adhesive or solvent handling operations for the day, wash your hands thoroughly and then apply skin conditioner. Remember cleanliness is the best defense against dermatitis.

3-59. References.

a. Adhesive Bonding Standards. (Until these can be established, current shop practices and the referenced documents will be used as an interim measure.)

(1) Standard Acid Etch Process for Treating Aluminum and Aluminum Alloys Prior to Adhesive Bonding. (Reference North American Specification LA0106-006, paragraphs 8.1.4 and 15.2.2, and General Dynamics Specification FPS 1009D, paragraphs 3.3.1 and 4.1.1.2.)

(2) Standard Pasa-Jell Process for Treating Aluminum, Aluminum Alloys and Titanium Prior to Adhesive Bonding.

(3) Quality Standard for Honeycomb Core Defects. (Reference General Dynamics Specification FPS-1017 and FPS-1009D, paragraphs 3.1.1 through 3.1.1.4.)

(4) Standard Process for Storing, Handling and Applying Film Adhesives and Preimpregs. (Reference General Dynamics Specification FMS-1009D, paragraphs 2.1, 3.2, 5.1, and FPS-1016B.)

(5) Standard Vacuum Bag Process. (Reference General Dynamics Specifications FPS-1067A and FPS-1016B.)

(6) Quality Standard for Control of Adhesive Bonding Materials and Processes. (Reference General Dynamics Specifications FPS-1009D and FPS-1016B.)

(7) Process for Ultrasonic Inspection of Adhesive Bonded Metal Faced Sandwich Panels.

Section IV

BODY GROUP

4-1. BODY GROUP.

4-2. The main fuselage sections of the F-106A and F-106B airplanes are semi-monocoque in design and of riveted, high-strength, all-metal construction. The fuselage structures differ only in that portion of the structure from station 102.00 to station 472.00. The difference is basically due to the single-place cockpit in the F-106A and the two-place, tandem cockpit in the F-106B, with their attendant differences in fuselage fuel tank and electronic bay locations.

4-3. DESCRIPTION OF FUSELAGE SECTIONS.

4-4. The fuselages of the F-106A and F-106B airplanes are divided into four main sections. These sections are the fuselage nose section, the forward and aft intermediate sections, and the fuselage aft section or tail cone. All sections except the aft or tail cone sections are joined by manufactured splices; the tail cone is pinned to the aft intermediate section and is readily removable. Figure 4-1 shows the fuselage components and gives a figure index for individual component illustrations. Fuselage stations for the F-106A and F-106B are shown on figures 4-2 and 4-3.

4-5. Fuselage Nose Section—F-106A.

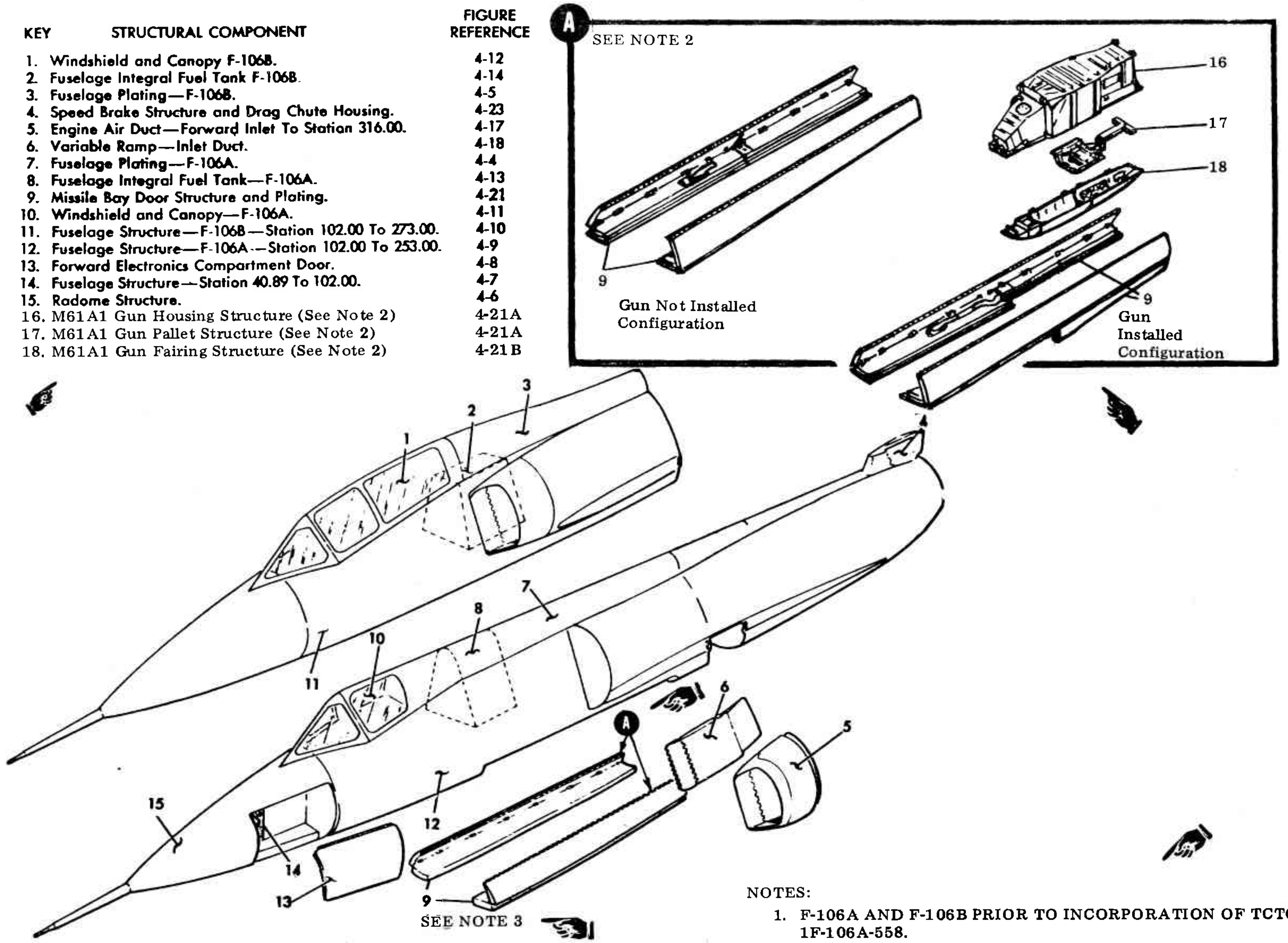
4-6. The fuselage nose section extends from the airspeed boom aft to station 253. Located in this nose section is the removable fiberglass radome, the forward electronics compartment, cockpit, nose wheel well, fuselage fuel tank, and the upper and lower aft electronic compartments. The cone-shaped radome is of the continuous fiberglass, filament wound type. The fiberglass is reinforced with low-pressure laminating resin, Specification MIL-R-7575, or equivalent, used as a bonding agent. See figure 4-6 for repair limitations and refer to paragraph 4-33 for repair information. The radome is equipped with an aluminum mounting ring at the large end. Four bolts connect the radome to the fuselage bulkhead at station 40.89. A neoprene seal, extruded inserts, and retainers fitted to the radome ring assembly provide the required sealing at the joint between the radome and the fuselage structure. The forward tip of the radome is fitted with an adapter for the attachment of the boom and the pitot tube. The forward radar electronics compartment is constructed basically of stressed aluminum alloy.

4-7. The forward bulkhead at station 40.89 is of the built-up type construction and provides the framework for attachment of the radome. Four longerons, two upper and two lower, extend aft from this bulkhead at station 40.89 to the bulkhead at station 102.00. A shear web, made of stressed aluminum alloy and reinforced with extruded aluminum angles, extends from the fuselage bulkhead at station 40.89 to the fuselage bulkhead at station 102.00. This shear web is considered a structural member and should be treated as such when designing any repair for it. The forward radar electronics compartment is enveloped by stressed aluminum alloy plating on the upper and lower surfaces; this plating is attached to the fuselage structure with flush-head rivets. Access doors, on the left and right sides of the forward radar electronics compartment, complete the envelopment of the fuselage structure in this area. These doors are of the panel type and consist of a stressed aluminum alloy outer skin attached to the door ribs and angles with flush-head rivets. The doors are hinged to the upper portion of the fuselage structure and are held in the closed position by stressed panel Camloc fasteners. See figures 4-4, 4-5, 4-7 and 4-8 for illustrations showing fuselage structure, plating and door structure in this area. Refer to Section I for information pertaining to stressed panel Camloc fasteners.

4-8. The lower section of the fuselage structure from station 102.00 to station 253.00 (shown on figure 4-9) consists principally of a series of semicircular zee and channel spliced beltframes with forged and built-up bulkheads connected by longerons, gussets and intercostals. The fuselage framework is enveloped by stressed aluminum alloy skins attached with flush-head rivets and other fasteners. This portion of the fuselage provides the space for the nose landing gear wheel well and attachment framework for the nose landing gear and gear door.

4-9. The pilot's compartment for the F-106A is pressurized and its structural framing consists of a series of closely spaced vertical frame assemblies reinforced by longitudinal stiffeners. The web flooring is supported by channels and zee stiffeners riveted to the built-up type bulkheads at stations 102.00 and 171.50. The two upper,

KEY	STRUCTURAL COMPONENT	FIGURE REFERENCE
1.	Windshield and Canopy F-106B.	4-12
2.	Fuselage Integral Fuel Tank F-106B.	4-14
3.	Fuselage Plating—F-106B.	4-5
4.	Speed Brake Structure and Drag Chute Housing.	4-23
5.	Engine Air Duct—Forward Inlet To Station 316.00.	4-17
6.	Variable Ramp—Inlet Duct.	4-18
7.	Fuselage Plating—F-106A.	4-4
8.	Fuselage Integral Fuel Tank—F-106A.	4-13
9.	Missile Bay Door Structure and Plating.	4-21
10.	Windshield and Canopy—F-106A.	4-11
11.	Fuselage Structure—F-106B—Station 102.00 To 273.00.	4-10
12.	Fuselage Structure—F-106A—Station 102.00 To 253.00.	4-9
13.	Forward Electronics Compartment Door.	4-8
14.	Fuselage Structure—Station 40.89 To 102.00.	4-7
15.	Radome Structure.	4-6
16.	M61A1 Gun Housing Structure (See Note 2)	4-21A
17.	M61A1 Gun Pallet Structure (See Note 2)	4-21A
18.	M61A1 Gun Fairing Structure (See Note 2)	4-21B



NOTES:

1. F-106A AND F-106B PRIOR TO INCORPORATION OF TCTO 1F-106A-558.
2. F-106A AFTER INCORPORATION OF TCTO 1F-106A-558.
3. TCTO 1F-106A-558 IS APPLICABLE TO F-106A VERTICAL INSTRUMENTED AIRCRAFT.

Figure 4-1. Fuselage Components and Figure Index

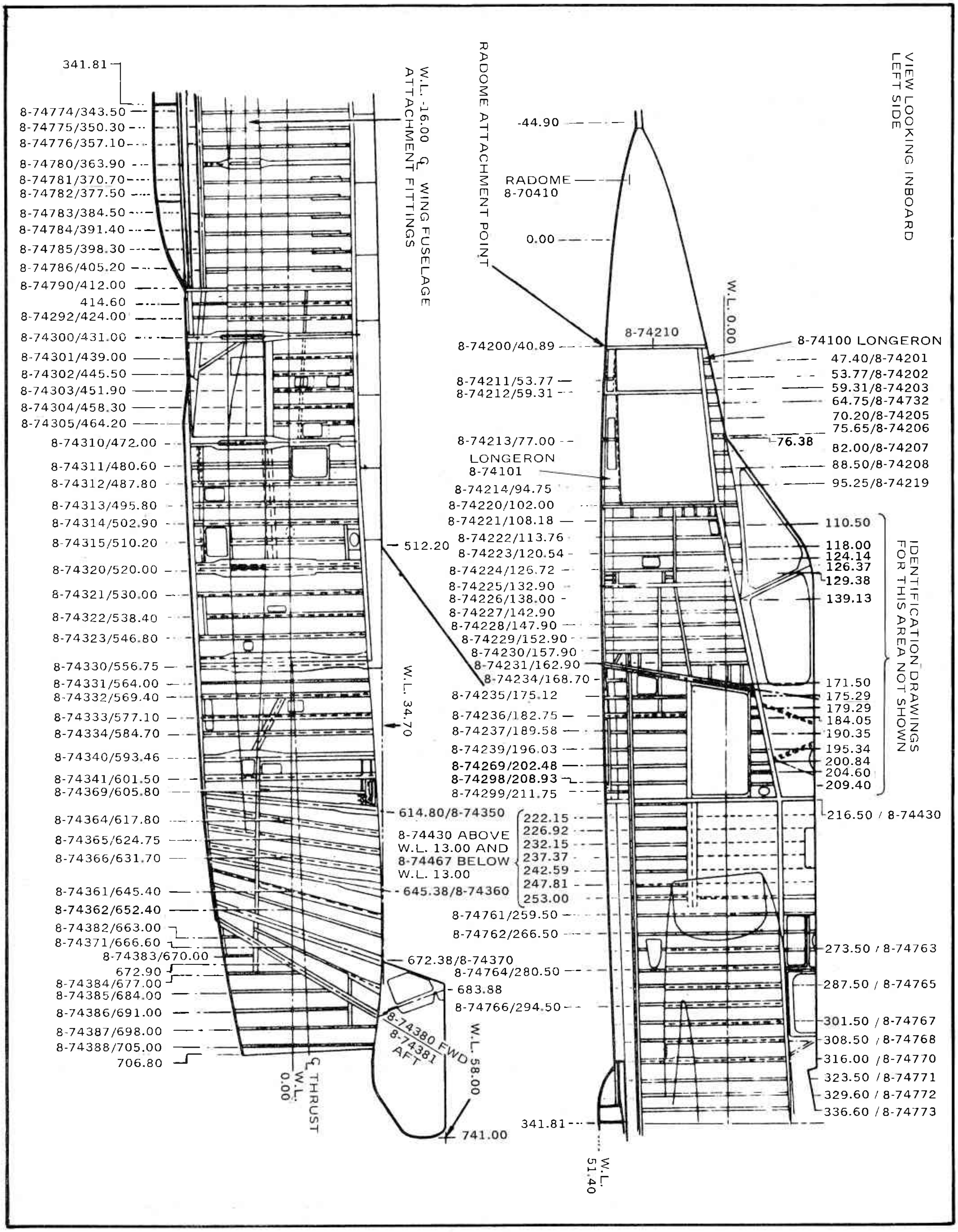


Figure 4-2. Fuselage Station Diagram — F-106A

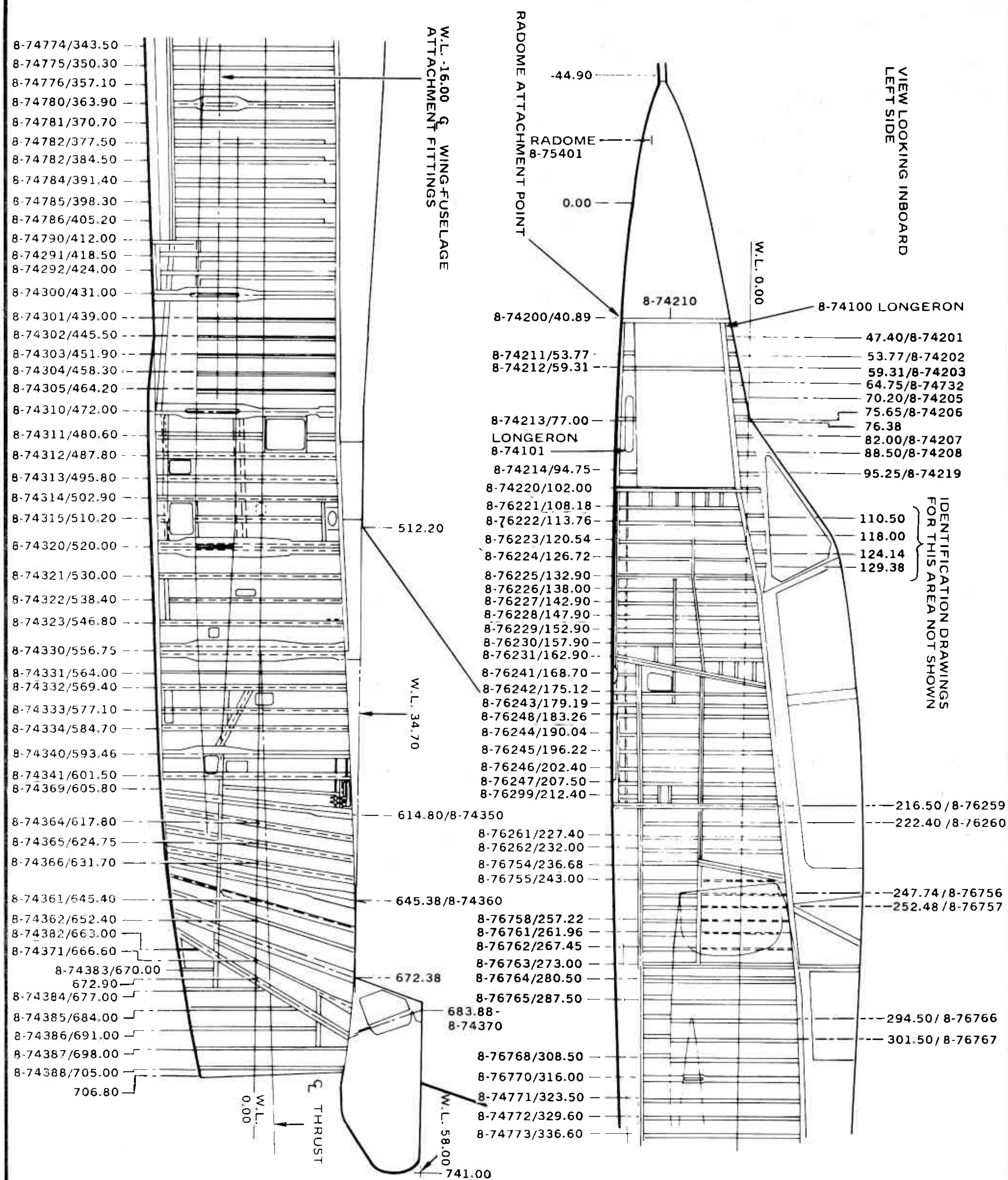
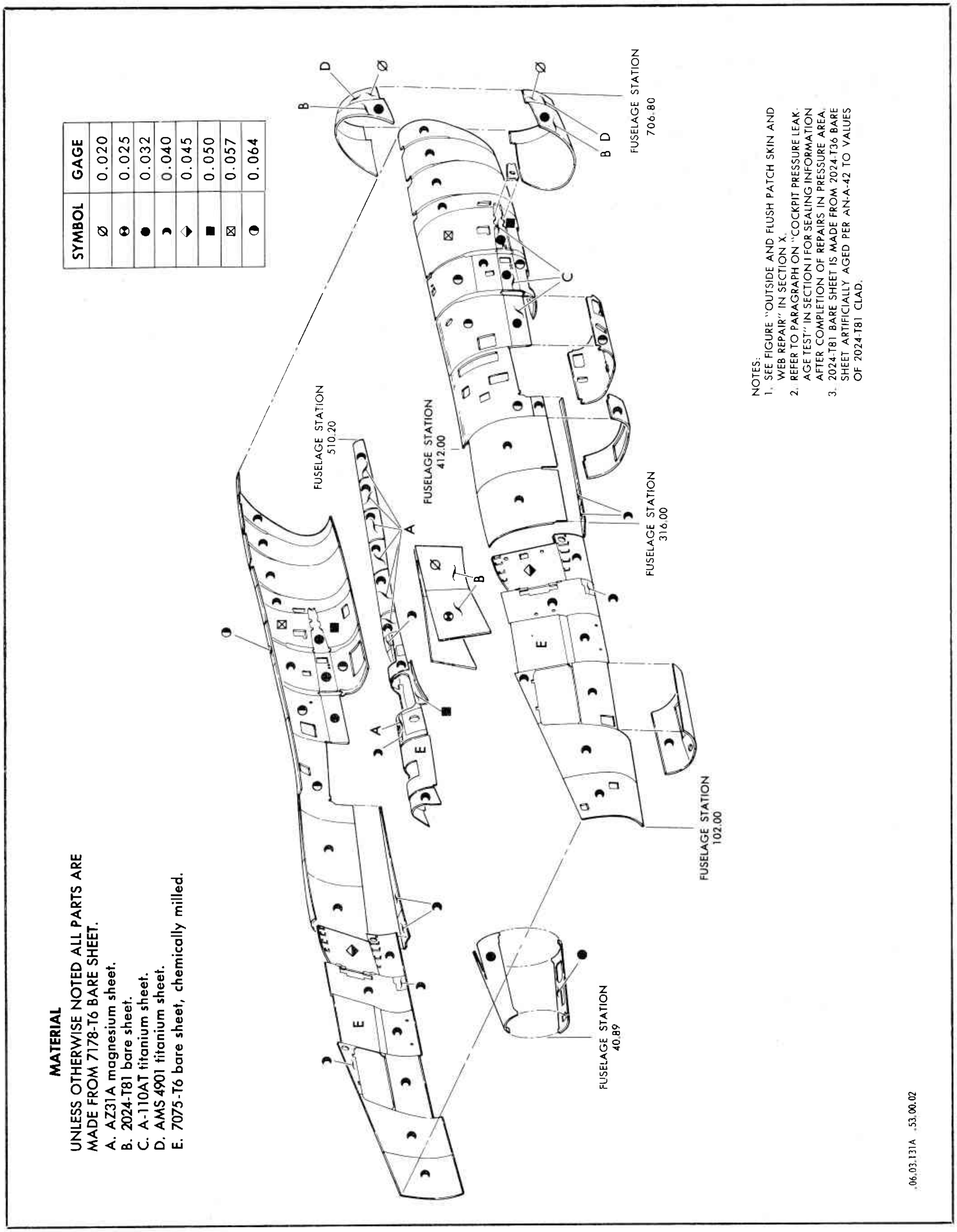


Figure 4-3. Fuselage Station Diagram — F-106B



SYMBOL	GAGE
Ø	0.020
⊗	0.025
●	0.032
◗	0.040
◆	0.045
■	0.050
⊠	0.057
◐	0.064

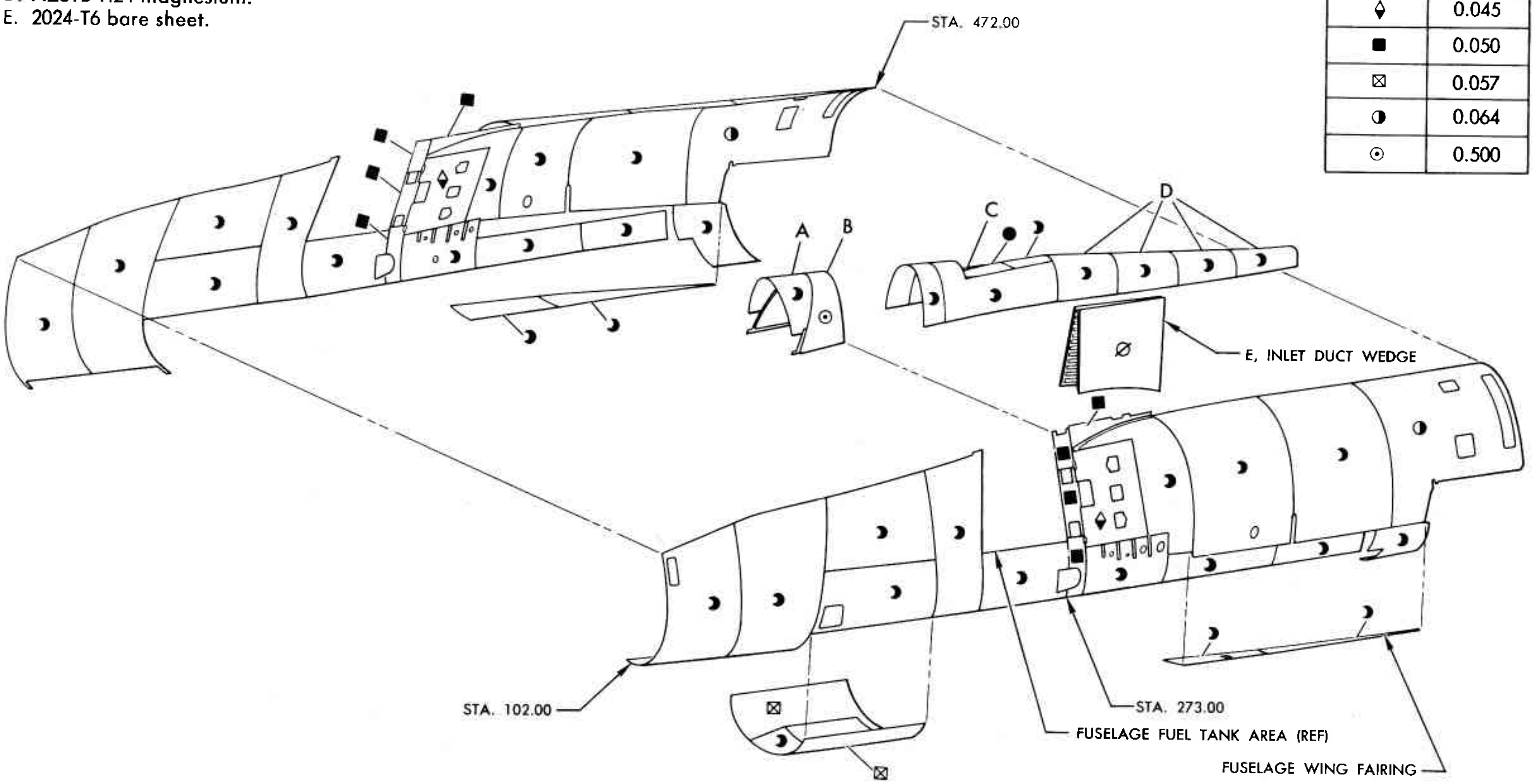
Figure 4-4. Fuselage Plating Diagram — F-106A

MATERIAL

UNLESS OTHERWISE NOTED ALL PARTS ARE MADE FROM 7178-T6 BARE SHEET.

A. AZ31A-O magnesium.
B. Fiberglass honeycomb core enveloped by number 181 glass cloth laminate.
C. AMS4901 titanium.
D. AZ31B-H24 magnesium.
E. 2024-T6 bare sheet.

SYMBOL	GAGE
∅	0.020
●	0.032
◗	0.040
◈	0.045
■	0.050
⊠	0.057
◑	0.064
⊙	0.500



EXPLODED VIEW—F-106B FUSELAGE PLATING

NOTES:
1. REFER TO PARAGRAPH ON "NEGLIGIBLE DAMAGE LIMITS—FUSELAGE GROUP" IN THIS SECTION FOR DISPOSITION OF NEGLIGIBLE DAMAGE.
2. REFER TO SECTION X FOR TYPICAL REPAIR INFORMATION AND FIGURE TITLED "FLUSH PATCH-SKIN REPAIR."

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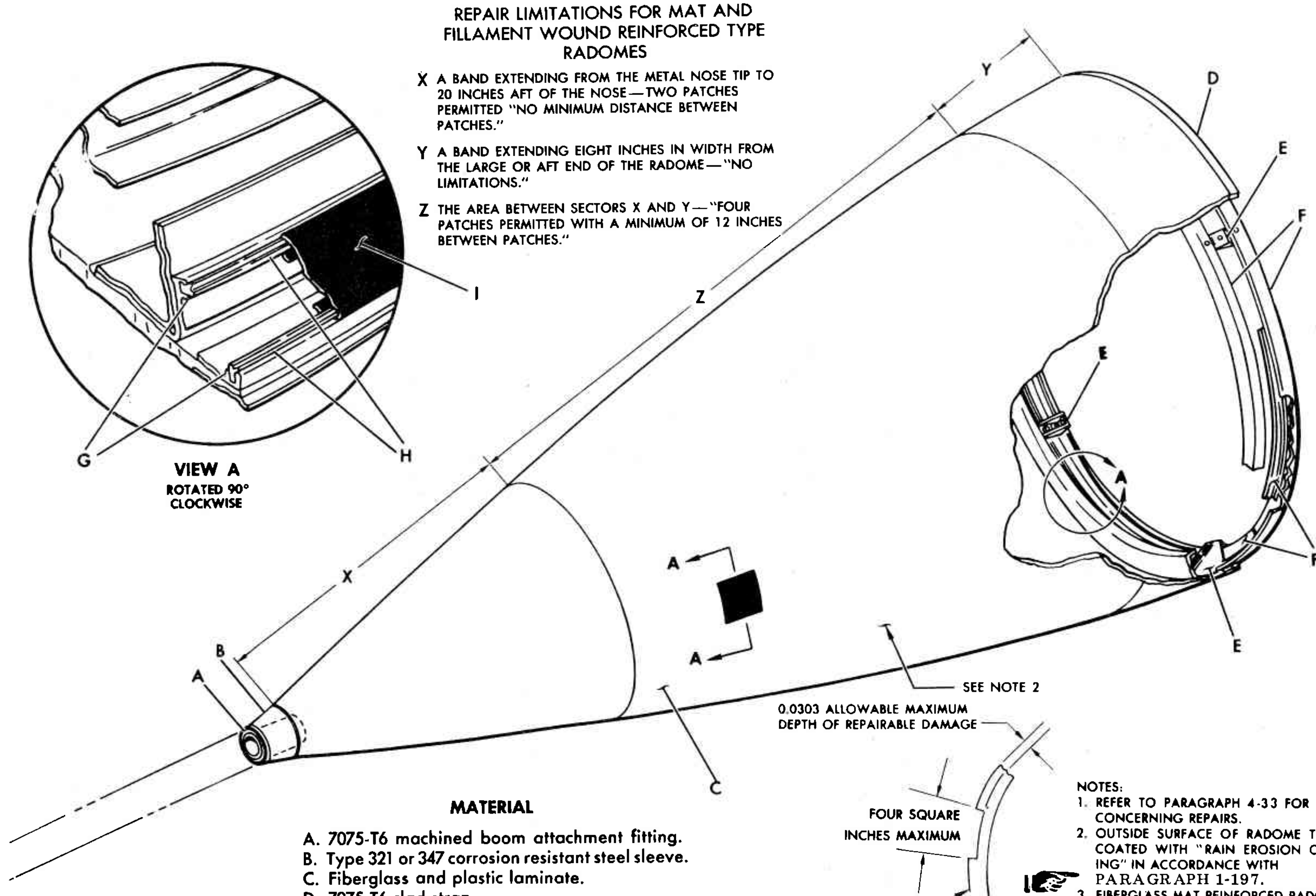
Figure 4-5. Fuselage Plating Diagram — F-106B



REPAIR LIMITATIONS FOR MAT AND FILLAMENT WOUND REINFORCED TYPE RADOMES

X A BAND EXTENDING FROM THE METAL NOSE TIP TO 20 INCHES AFT OF THE NOSE—TWO PATCHES PERMITTED "NO MINIMUM DISTANCE BETWEEN PATCHES."

Y A BAND EXTENDING EIGHT INCHES IN WIDTH FROM THE LARGE OR AFT END OF THE RADOME—"NO LIMITATIONS."

Z THE AREA BETWEEN SECTORS X AND Y—"FOUR PATCHES PERMITTED WITH A MINIMUM OF 12 INCHES BETWEEN PATCHES."

MATERIAL

A. 7075-T6 machined boom attachment fitting.
B. Type 321 or 347 corrosion resistant steel sleeve.
C. Fiberglass and plastic laminate.
D. 7075-T6 clad strap.
E. ZK51-T5 magnesium alloy casting.
F. 7075-T6 clad frame.
G. 6061-T6 extrusion.
H. INSERT PART NO. 340-8210303-9.
I. Neoprene-impregnated nylon seal.

NOTES:
1. REFER TO PARAGRAPH 4-33 FOR DATA CONCERNING REPAIRS.
2. OUTSIDE SURFACE OF RADOME TO BE COATED WITH "RAIN EROSION COATING" IN ACCORDANCE WITH PARAGRAPH 1-197.
3. FIBERGLASS MAT REINFORCED RADOMES ARE USED ON F-106A AIRPLANES 56-454 AND 56-457. CONTINUOUS GLASS FILAMENT WOUND REINFORCED RADOMES ARE USED ON F-106A AIRPLANES 56-453, -455, -456, -458 AND SUBSEQUENT, AND F-106B AIRPLANES 57-2507, AND SUBSEQUENT.
4. REPAIR LIMITATIONS AND REPAIR PROCEDURES ARE THE SAME FOR BOTH TYPES.

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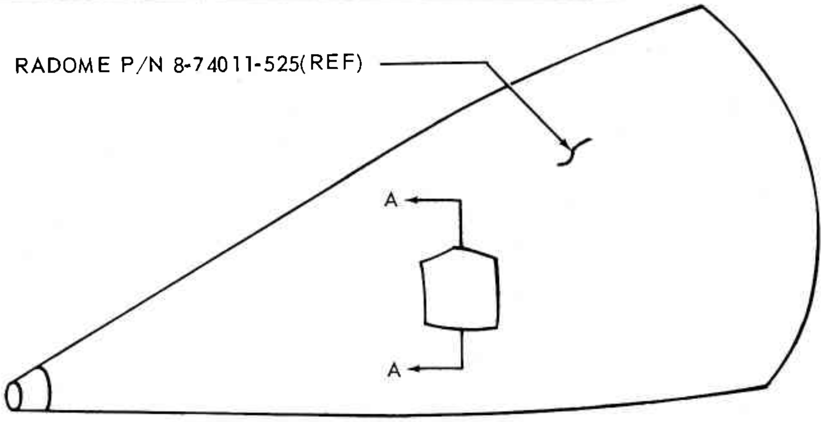
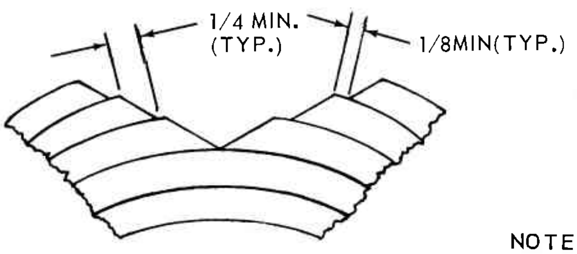
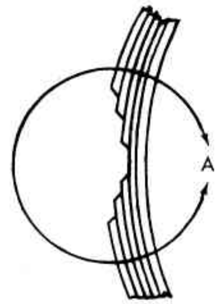
Figure 4-6. Radome Structure





VIEW A



A—A
REPAIR PREPARATION
TYPICAL

NOTE

1. 1/4 INCH MINIMUM BEVEL ALL EDGES OF EACH LAYER OF THE REWORKED AREA.

a. Inspect the radome to determine extent of damage and method of repair.
   1. Condemn the radome if the damage extends into the fourth lamination.

NOTE

THE RADOME IS CONSTRUCTED WITH 5 LAMINATIONS. THE OUTSIDE (FIRST), CENTER (THIRD) AND INSIDE (FIFTH) LAMINATIONS ARE CIRCUMFERENTIALLY WOUND. THE SECOND AND FORTH LAMINATIONS CONSIST OF ROVING LAID LONGITUDINALLY.

   2. DELETED
   3. Condemn the radome if it is permanently distorted.
   4. Repair all other damage within the limits of Paragraphs a (1), a (2), and a (3) using the following procedures.

b. Prepare the damaged area for lay-up.
   1. Remove all loose strands. Short frayed ends and short loose fibers on strands will aid adhesion and are acceptable.
   2. Scarf the edges of the reworked area as illustrated in View A.
   3. Remove the outer layer of circumferentially wound filament from the entire circumference when the damage exceeds 45 degrees of the circumference and/or 5 inches in length. Chamfer the edges as illustrated in View A.
   4. Remove resin dust and loose material with a soft brush and low pressure blow gun.
   5. Clean the area with methyl-ethyl-keytone. Do not allow the methyl-ethyl-keytone to dry on the surface. Blot up the liquid with lint free paper towels. Do not rub.
   6. Inspect for and remove any lint, paper particles, etc., from the area with tweezers.

CAUTION

DO NOT HANDLE THE ROVING OR THE REWORKED AREA WITH THE BARE HANDS. USE LINT FREE COTTON OR NYLON GLOVES. SKIN OILS AND OTHER CONTAMINATION WILL PREVENT ADHESION.

c. Lay-up. Small repairs up to 5 inches in length and 45 degrees width may be built up with layers of glass cloth and resin. Maintain 45 to 50 percent glass cloth to resin ratio.

d. Lay-up of third and second layers.
   1. Cut strands of roving, wet them with activated resin, and lay them parallel to the original roving.
      (a) Use varying lengths of roving to match the taper of the rewprked area. Strands shall lie flat and straight.
      (b) Work out all bubbles and excess resin.
      (c) Fill the reworked area of third and/or second layer. The outer surface of the fill shall be smooth and confrom to the contour of the second layer.

Figure 4-6A. Randome Fiberglass Repair For Overhaul Facilities (Sheet 1 of 2)

e. Lay-up of the outer layer, circumferentially wound.
1. Mount the radome on the Fixture, SMAMA Drawing Number 65SCJ6237.
2. Start the winding in the bottom of the reworked area by passing the roving around the radome and crossing over the free end of the roving at the end of the first turn.
3. Apply resin sparingly but adequately to assure complete wetting of the roving. The resin may be applied before winding or during the winding.
4. Carefully feed the roving on to the radome maintaining a ½ to 2 pound tension on the roving. Do not relax this tension until the entire area is filled with the roving. Anchor the loose end of a spool of roving by lapping with the first turn of the next spool. Build up the outer layer with even layers of roving, filling all voids. Occasional inspections shall be made to detect and eliminate bubbles in the resin. The finished surface shall be smooth and contoured to the original radome shape.

f. Cover the repaired area with PVC film and apply 10 to 20 psi load (20 to 30 in HG vacuum) to the repaired area.

CAUTION

DO NOT APPLY TAPE TO THE SURFACE OF THE RADOME.

g. Cure the resin.
h. Testing of repaired radomes:
1. Subject the radome to a bursting pressure of of 6 psig at room temperature.
2. Subject the radome to a collapsing load of 9 psi (18 in HG vacuum) at room temperature.
3. With the radome securely attached to aircraft attaching points (similated aircraft installation) slowly apply a 250 lb load at right angles to the axis of the radome at the boom attach fitting, P/N 8-72973.
4. Apply one load in a direction away from and towards the repaired area. Use Fixture, SMAMA Drawing Number 65SCJ6237.

i. Inspect for evidence of failure. Cracks, delamination, bulging and/or distortion are causes for rejection.
j. Subject the radome to electrical tests according with TO-1-1-24 and 1-1-24A.

MATERIALS

1. Roving, Fiberglass Cordage, MIL-Y-1140, Class C. Form 2.
2. Resin and Catalyst, MIL-R-25042, Type III, Epocast Resin, H1468, and Hardener, 9816, manufactured by Furane Plastics.
3. Glass Cloth, MIL-C-9084.
4. Film (PVC), or equal.

EQUIPMENT

1. Fixture - Repair and Test, F-106 Radome, SMAMA Drawing Number 65SCJ6237.
2. Plate, plug and fittings for pressure and vacuum testing.

SAFETY NOTES

Contact with some of the resin formulation may cause nasal or skin irritations, therefore, adequate ventilation will be provided in impregnating and lay-up areas. To prevent skin irritation, personnel handling resins should wash with soap and water at frequent intervals. Solvents may cause infections and irritations when used to remove resins from the skin.

REFERENCES

T.O. 1-1-24, and 1-1-24A Maintenance, Repair and Electrical Requirements of Fiber Laminate and Sandwich Constructed Airborne Radomes.

Figure 4-6A. Randome Fiberglass Repair For Overhaul Facilities (Sheet 2 of 2)

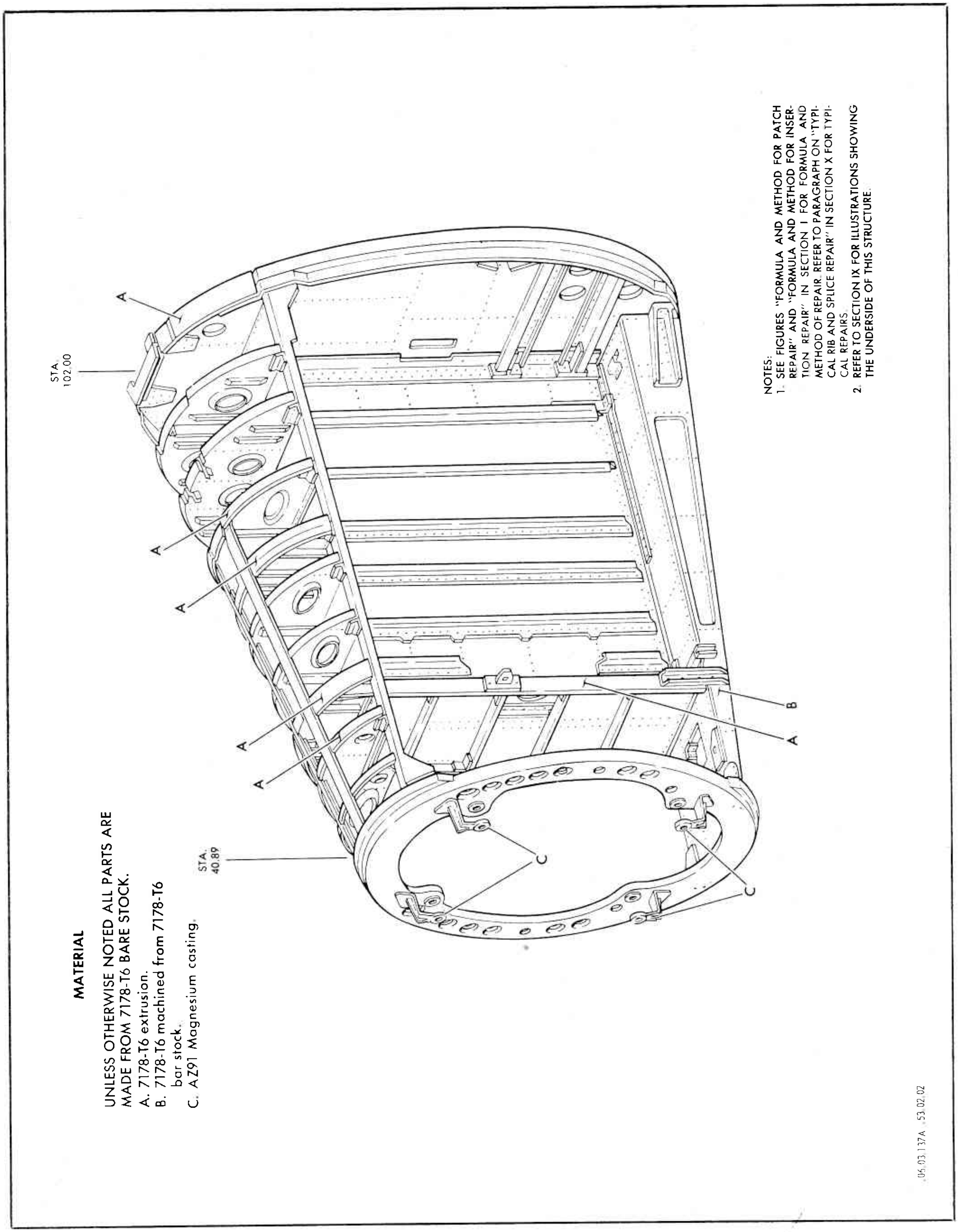
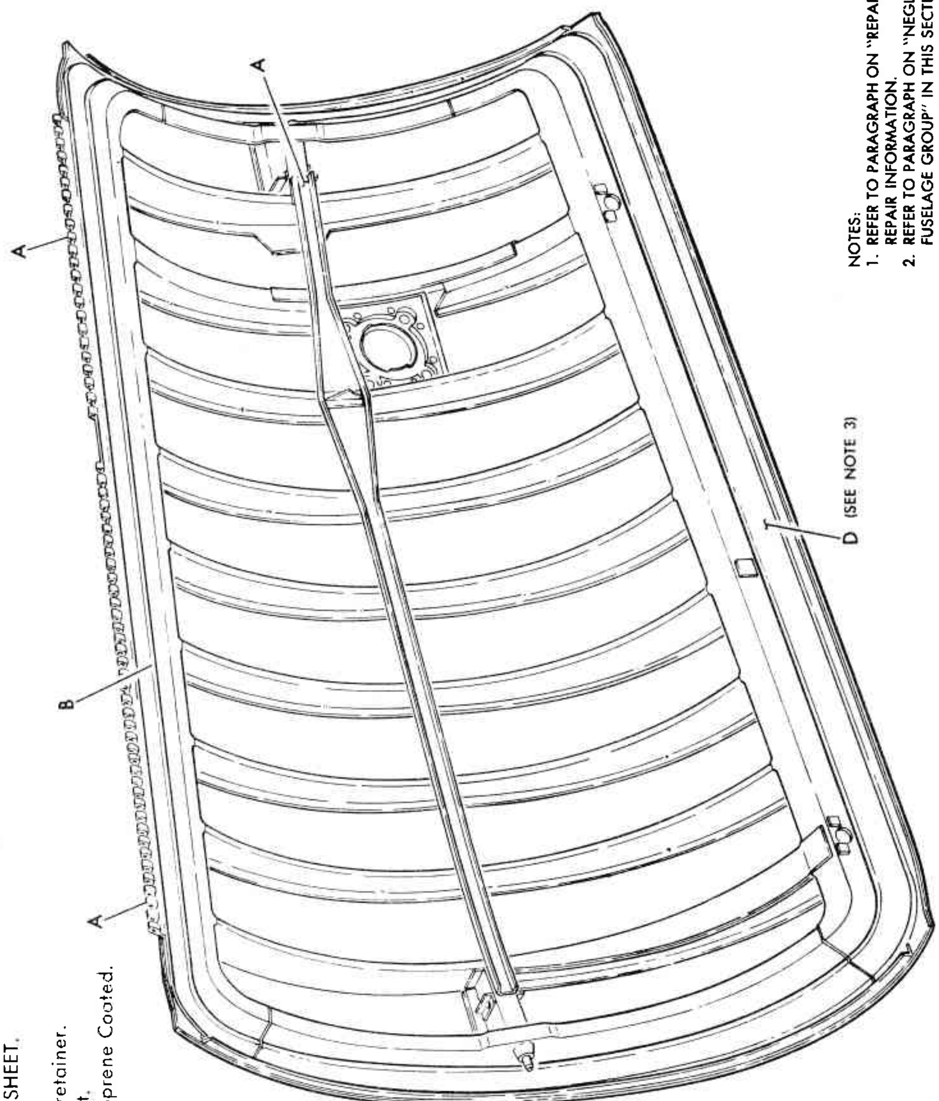


Figure 4-7. Fuselage Structure — Station 40.89 to 102.00



MATERIAL

UNLESS OTHERWISE NOTED ALL PARTS ARE MADE FROM 7178-T6 BARE SHEET.

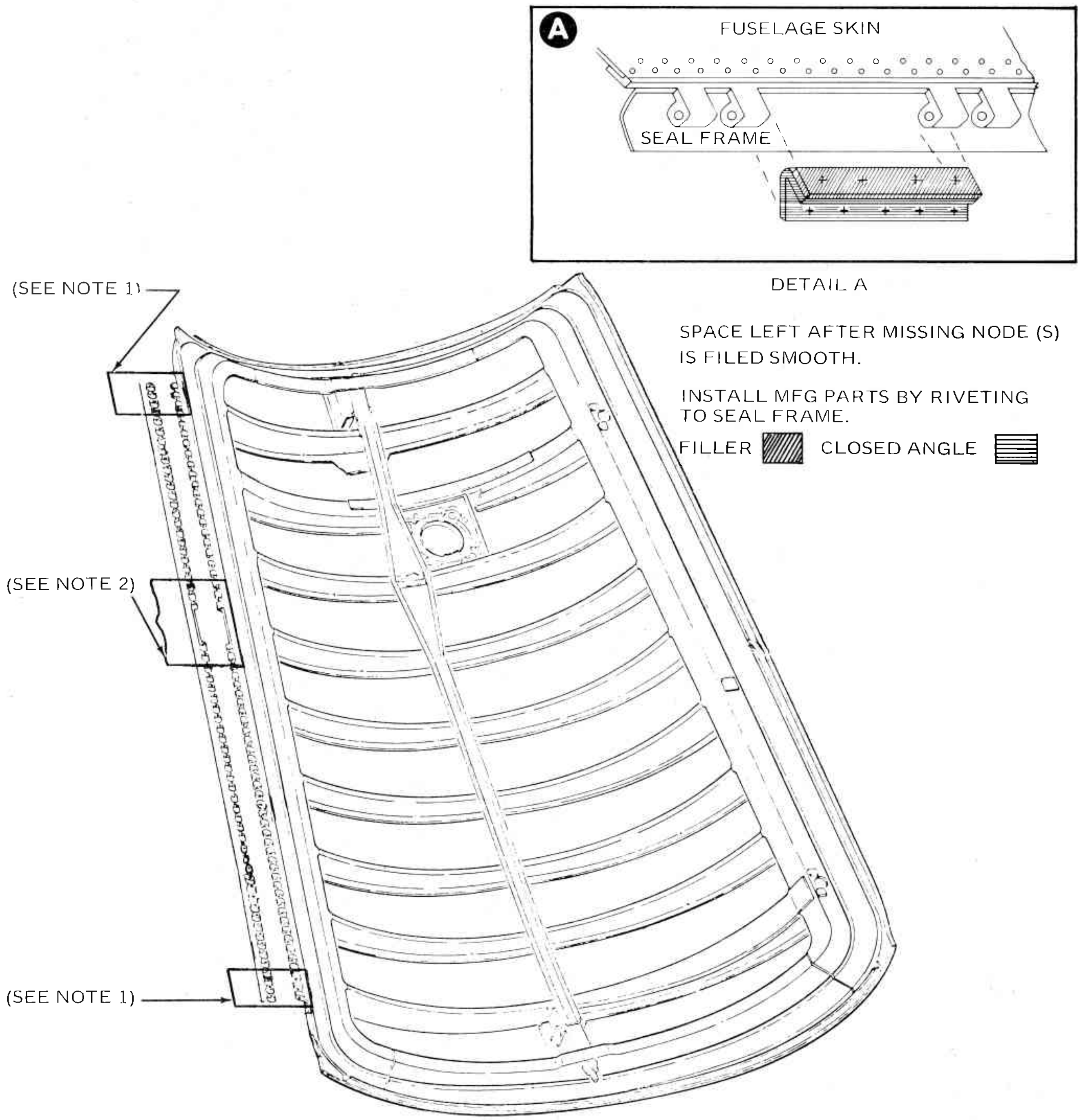
A. 7178-T6 extrusion.
B. 6061-T6 extruded retainer.
C. 7075-T6 bare sheet.
D. Nylon Fabric, Neoprene Coated.

NOTES:
1. REFER TO PARAGRAPH ON "REPAIRS" IN THIS SECTION FOR REPAIR INFORMATION.
2. REFER TO PARAGRAPH ON "NEGLIGIBLE DAMAGE LIMITS—FUSELAGE GROUP" IN THIS SECTION FOR DISPOSITION OF NEGLIGIBLE DAMAGE TO THIS COMPONENT.
3. NYLON FABRIC, NEOPRENE COATED SEAL HELD IN PLACE BY INSERT PART NO. 340-8210303-9.

06.03.136 .53.02.02

Figure 4-8. Forward Electronics Compartment Door Structure

IN AREA WHERE NODES ARE MISSING/REMOVED PER— LIMITATIONS SHOWN IN NOTES. FABRICATE ANGLE FILLER TO FAIR IN GAPS LEFT BY NODE REMOVAL AS SHOWN IN DETAIL A



NOTES

1. MAXIMUM OF ONE NODE MAY BE MISSING IN FIRST THREE NODES.
2. MAXIMUM OF TWO NODES MAY BE MISSING EITHER SIDE OF CUT OUT.
3. MAXIMUM OF THREE NODES MAY BE MISSING FROM ENTIRE HINGE WITH AT LEAST THREE GOOD NODES BETWEEN MISSING NODES, EXCEPT AS NOTED IN NOTES 1 AND 2.

Figure 4-8A. Forward Electronics Compartment Door

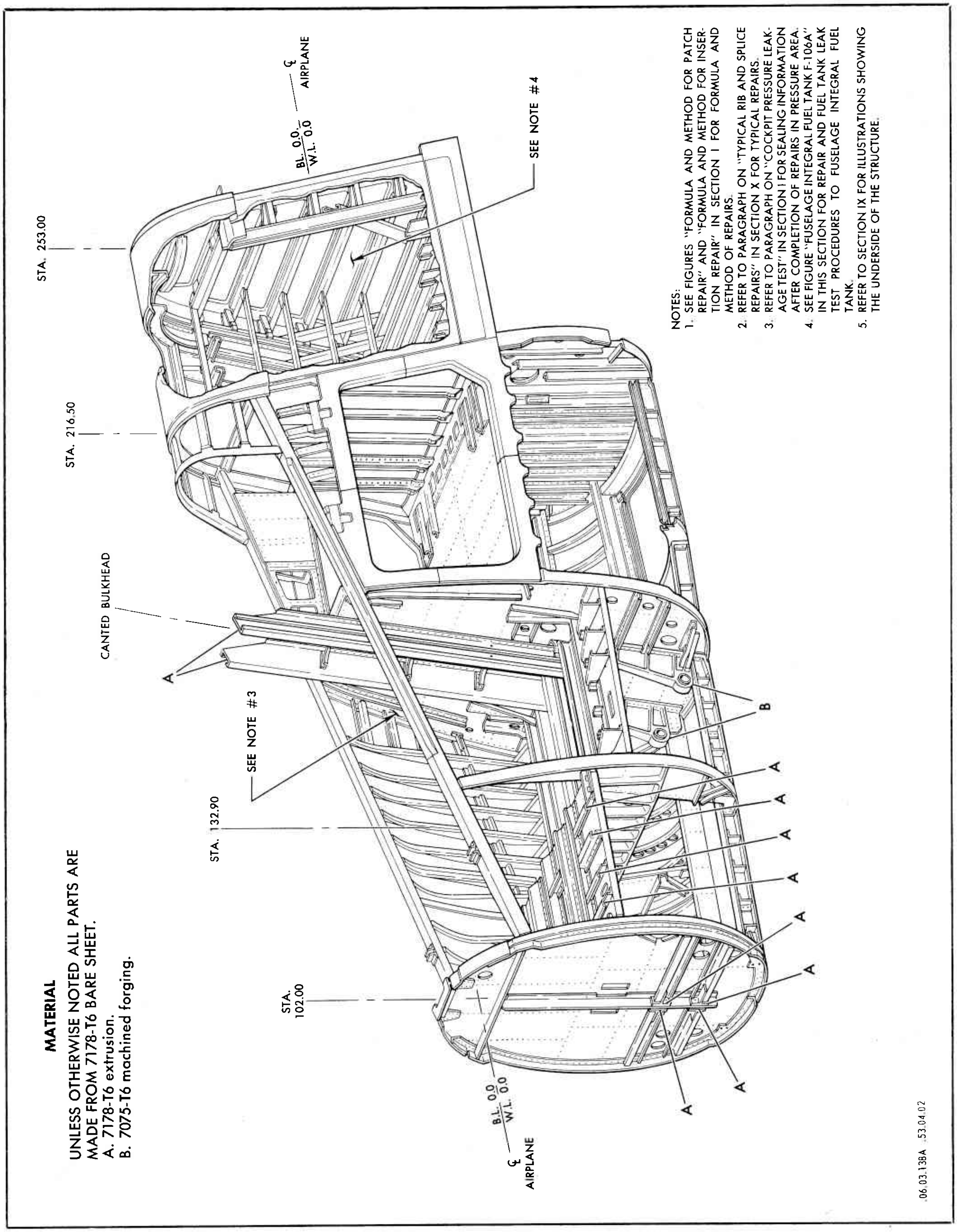


Figure 4-9. Fuselage Structure — Station 102.00 to 253.00 — F-106A

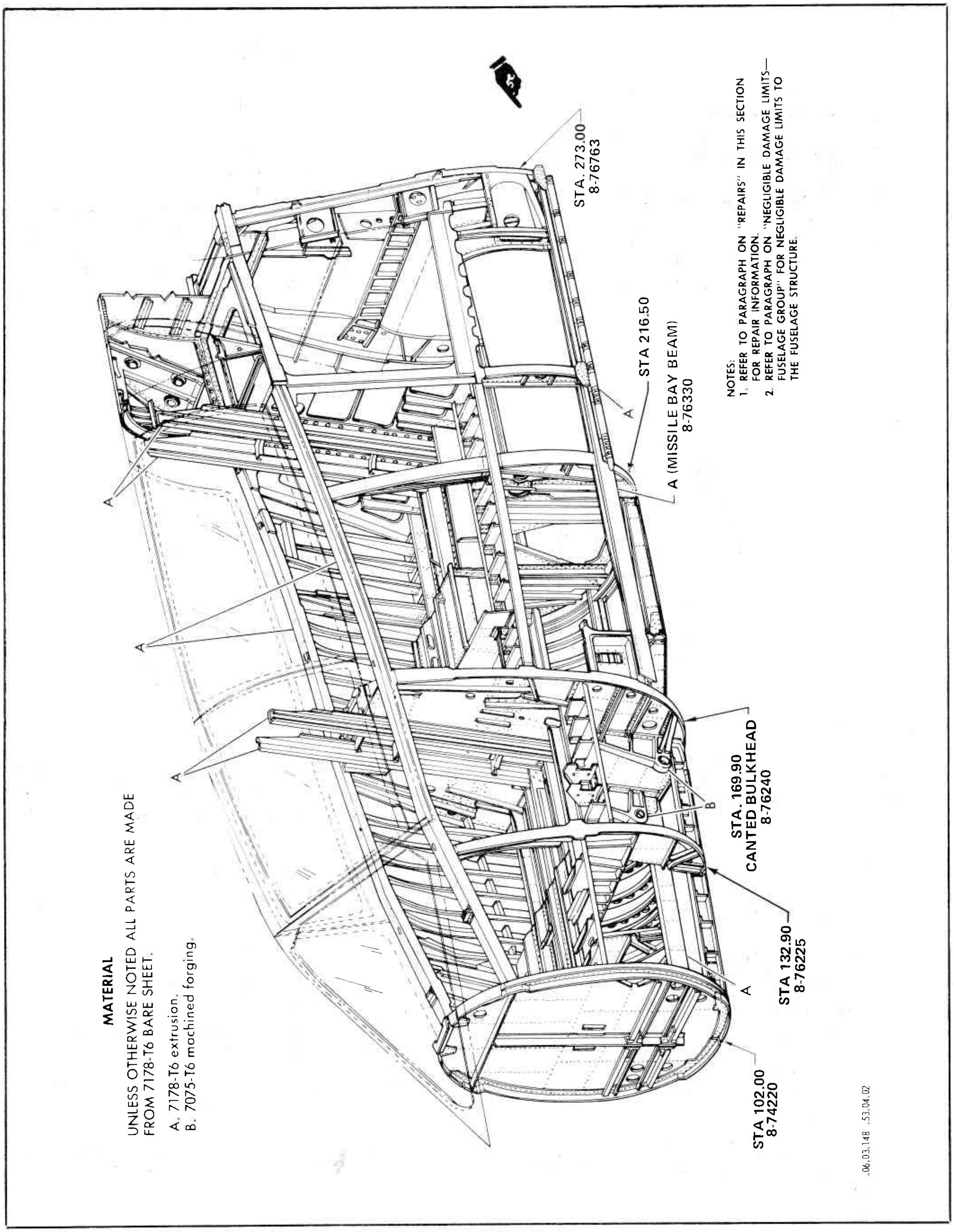
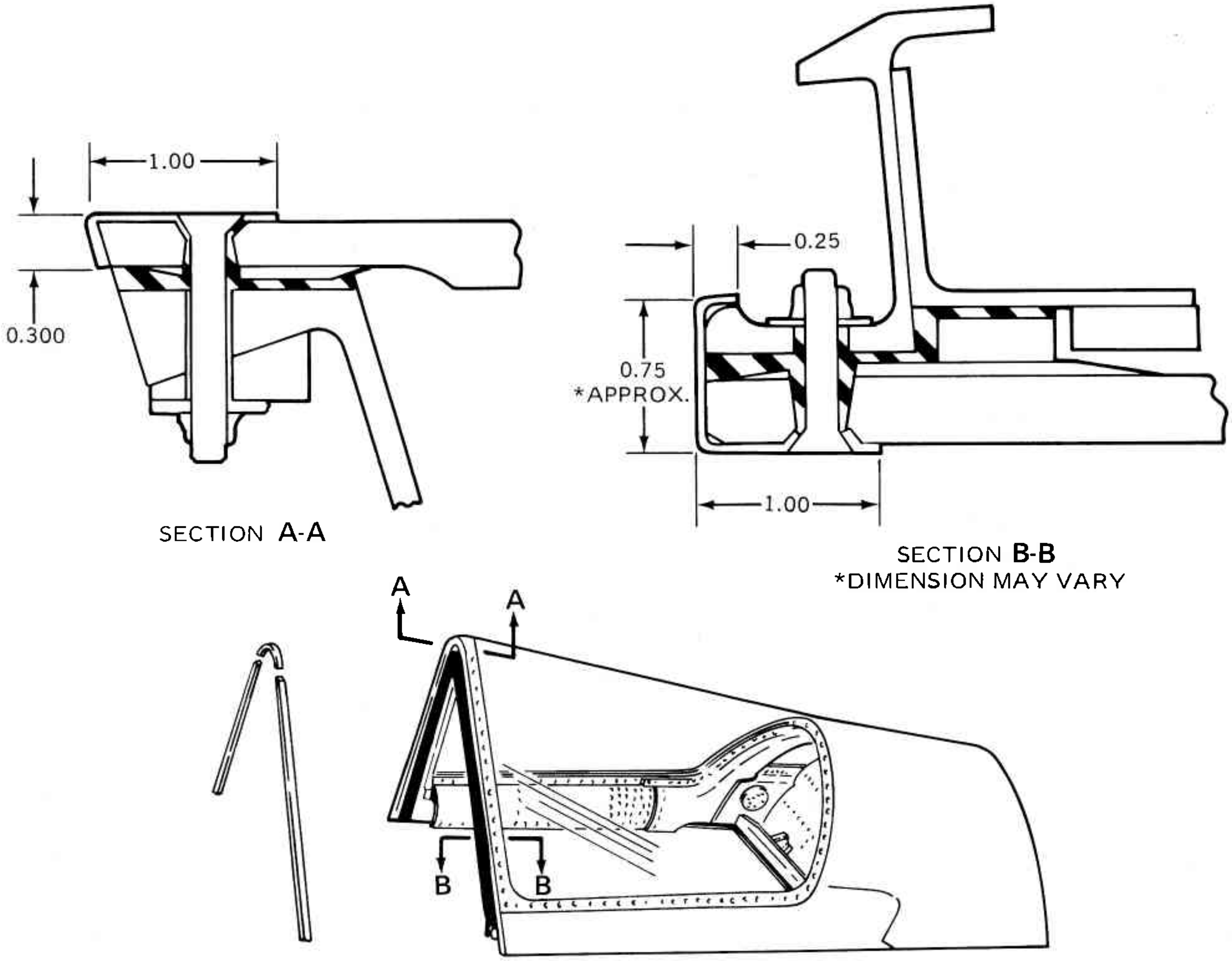


Figure 4-10. Fuselage Structure — Station 102.00 to 273.00 — F-106B



REPAIR PROCEDURE

1. CLOSE CANOPY AND LOCK. CHECK CLEARANCE BETWEEN CANOPY LEADING EDGE AND WINDSHIELD FRAME, MARK AREA TO BE TRIMMED.

NOTE

- ALL CANOPY GLASSES WHICH HAVE EROSION BEYOND THE FORWARD EDGE OF THE SCREW HOLES SHALL BE REPLACED.
- ALL NEW CANOPY GLASSES SHALL HAVE STAINLESS STEEL STRIPS INSTALLED AT TIME OF INSTALLATION.

2. REMOVE CANOPY.
3. SHAVE LEADING EDGE OF CANOPY GLASS TO OBTAIN PROPER CLEARANCE FOR STRIP. SEE FIGURE 1-25.
4. MANUFACTURE THE APEX STRIP, 0.016 TYPE 321 MIL-S-6721 STAINLESS STEEL, PICKING UP FOUR HOLES ACROSS APEX AND DIMENSIONS SHOWN IN SECTION A-A. HAND FIT TO CANOPY CONTOUR.
5. REMOVE SCREWS FROM CANOPY APEX. (RETAIN FOR REINSTALLATION.) PLACE APEX STRIP FLAT AGAINST THE GLASS LEADING EDGE PRIOR TO PICKING UP THE SCREW HOLES.
6. DIMPLE HOLES IN APEX STRIP, REMOVE ALL BURRS AND SHARP EDGES AND INSTALL USING RETAINED HARDWARE. (DISCARD DIMPLED WASHERS.)
7. MANUFACTURE THE SIDE STRIPS FROM 0.016 TYPE 321 MIL-S-6721 STAINLESS STEEL, 28 INCHES LONG AND DIMENSIONS SHOWN IN SECTION B-B.
8. REPEAT STEPS 5 AND 6 FOR EACH SIDE STRIP ONE SIDE AT A TIME.
9. REINSTALL CANOPY, CHECK FOR FIT AND PROPER CLEARANCE AT THE WINDSHIELD FRAME.
10. REPAIR FOR F-106B AIRCRAFT IS THE SAME AS ABOVE EXCEPT APEX STRIP IS NOT REQUIRED.

Figure 4-11. Canopy Glass Erosion Repair

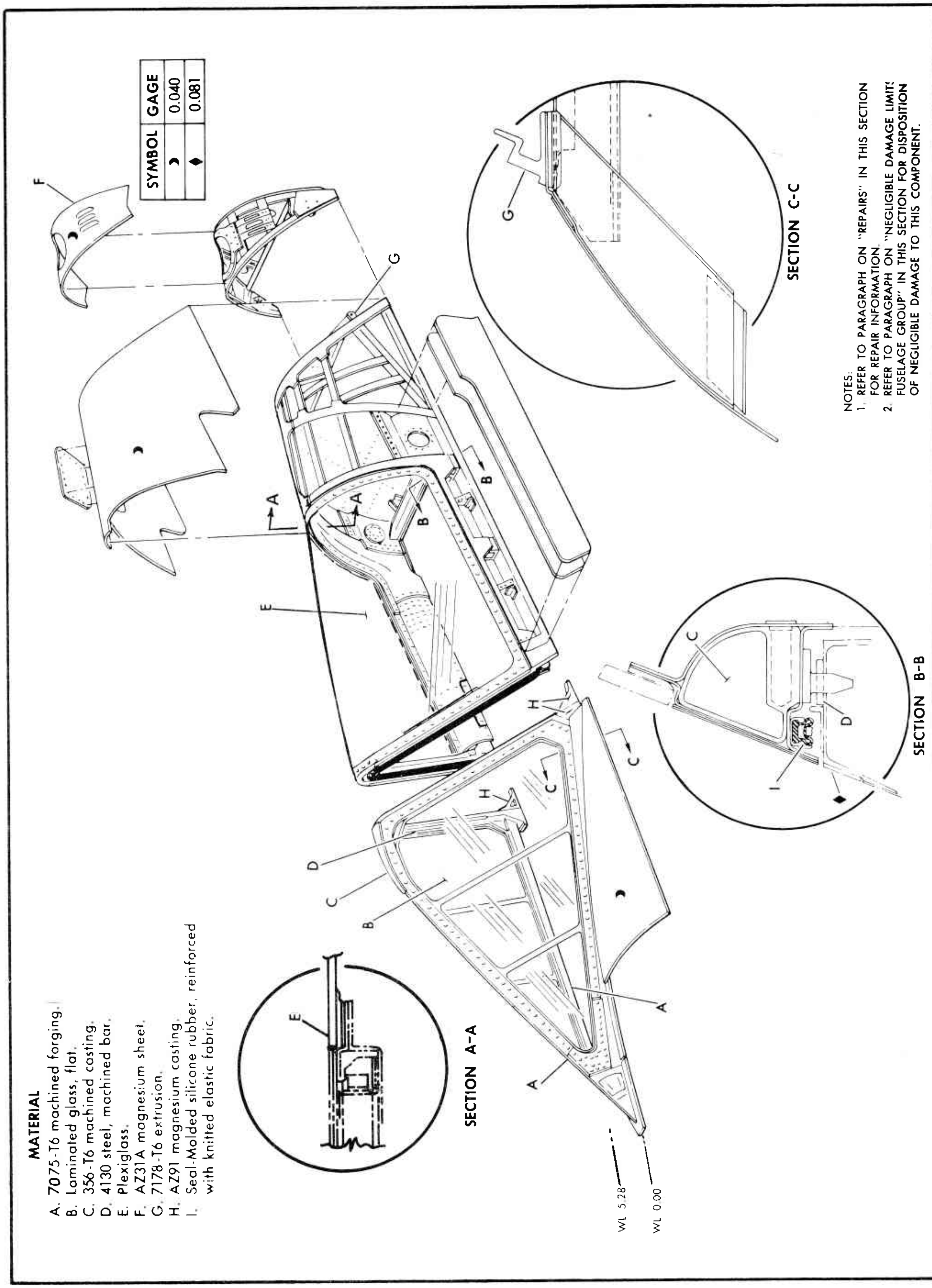


Figure 4-11A. Windshield and Canopy Structure –F-106A

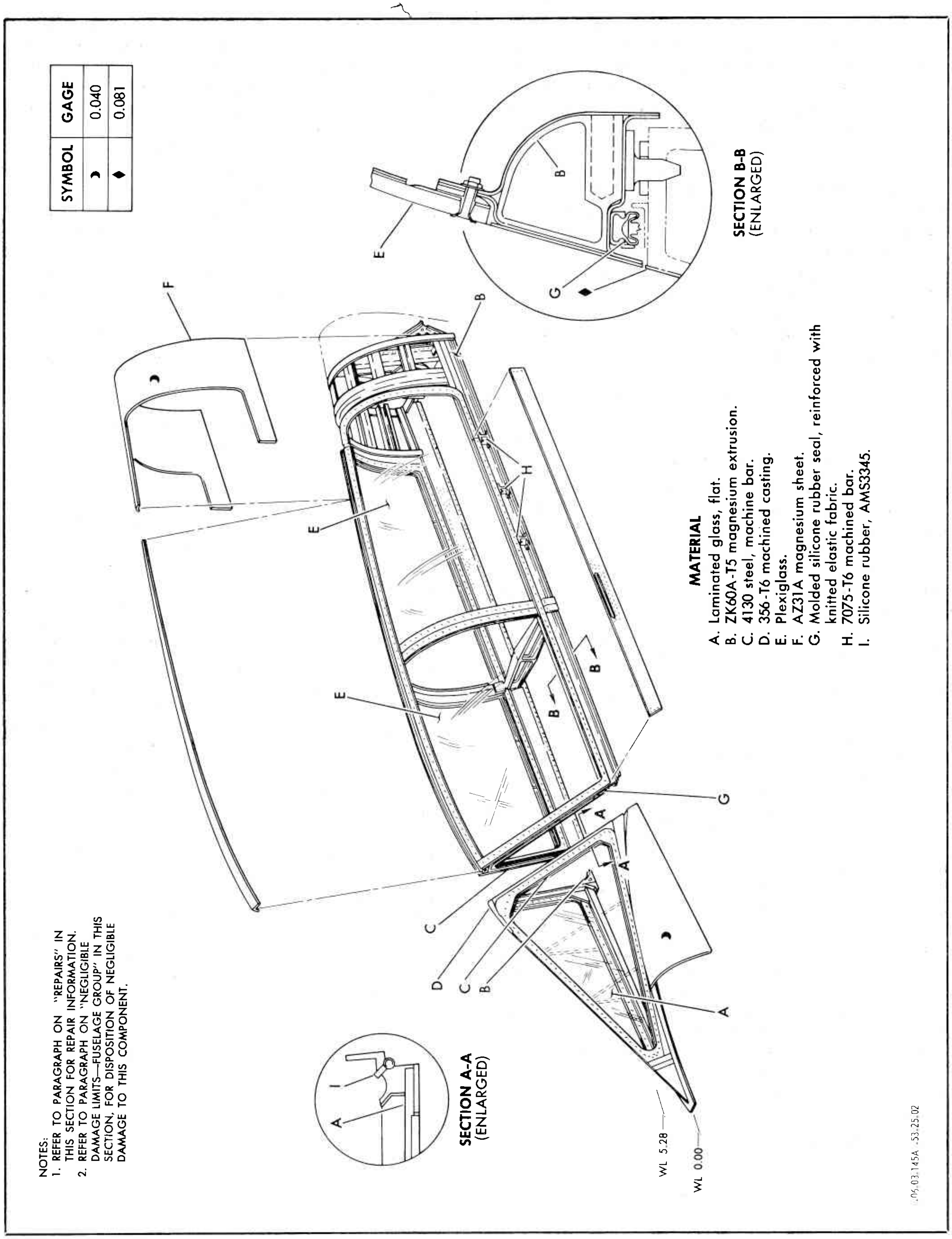


Figure 4-12. Windshield and Canopy Structure — F-106B

heavy-gage, parallel longerons interconnect the bulkheads and provide the support for windshield and canopy structures. The windshield and canopy assemblies for the F-106A are commonly referred to as the optimum vision type due to their "V"-shaped configuration and large single transparent panel assemblies. The windshield assembly consists basically of a framework of special extrusions and two laminated glass panel assemblies. The glass panels are secured to an extruded sill of the framework by retainers and flush-head screws. Rubber seals, attached to the structure with an application of EC-524 sealer, provide airtight seams. The canopy is constructed of built-up framework incorporating a plexiglas panel assembly in each side of the forward section, and a belt-frame and reinforced bulkhead closed in by three sections of outer skin in the aft section. The plexiglas panel assemblies are fitted with laminated orlon strips along the perimeter and are secured to the canopy framework longerons by means of flush-head screws. The windshield and canopy are shown on figure 4-11. Refer to T.O. 1F-106A-2-2-2-2 for canopy and windshield panel removal and replacement procedures.

4-10. The fuselage integral fuel tank is located in the upper portion of the fuselage nose section just aft of the upper aft electronics compartment and consists basically of a forward bulkhead at station 216.00, and an aft bulkhead at station 253.00. These bulkheads are of the integral rib sculptured type. The two bulkheads are connected by a bottom horizontal skin at water line −12.90, and by left and right vertical skins. The bottom and the left and right skins are of the integral-rib, sculptured type. The fuselage integral fuel tank is made fuel-tight by the use of machined aluminum alloy bar corner fittings, Scotchweld structural adhesive bonding tape and Straylor fuel-tight rivets. See figure 4-13. Refer to paragraphs 1-131 and 1-133 for installation and removal procedures for Straylor rivets. Refer to paragraphs 2-33 and 2-35 for information pertaining to Scotchweld tape. Refer to paragraph 4-43 for the fuel tank leak testing procedure after repairs.

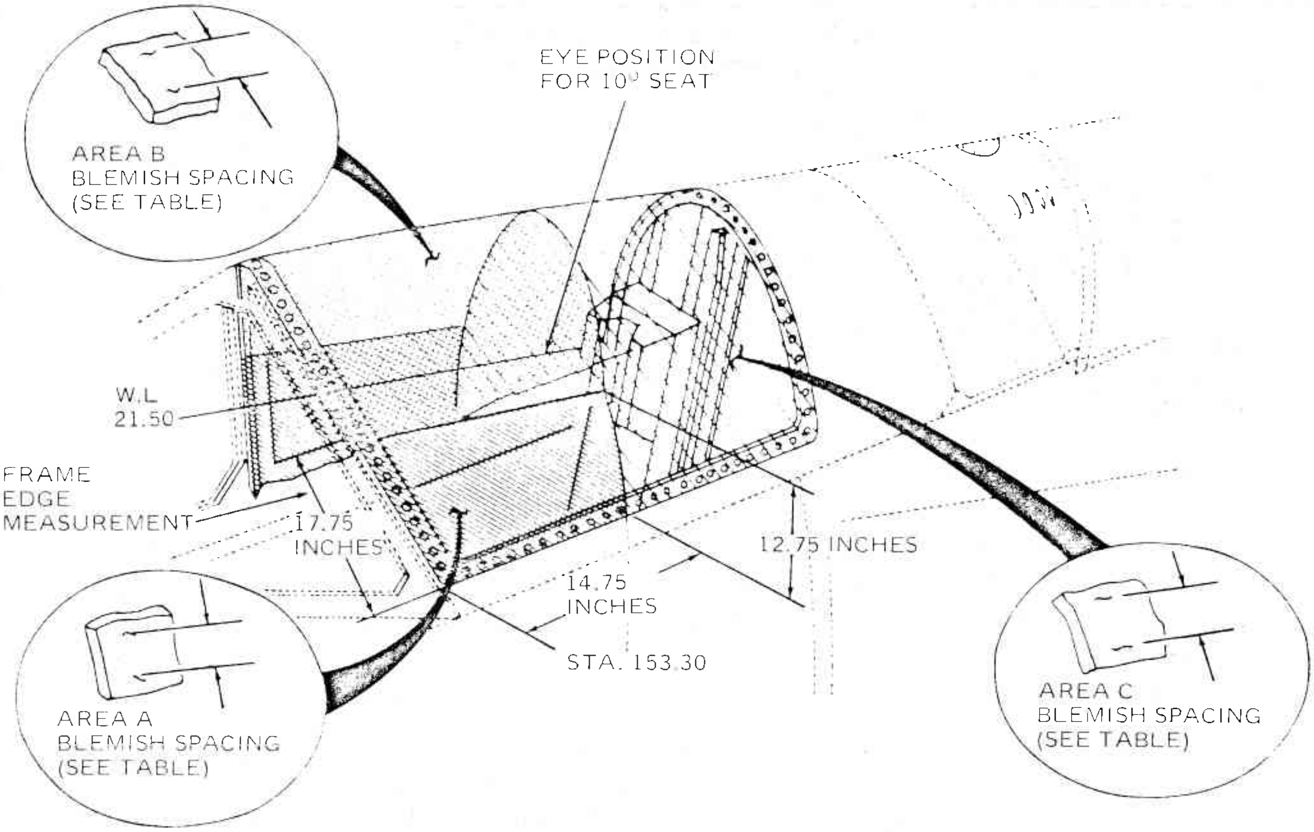
4-11. The upper aft electronics compartment doors on the left and right sides of the fuselage are located between stations 171.50 and 216.00, and are constructed principally of 7178-T6 bare sheet. See figure 4-15 for an illustration of the upper aft electronics compartment door structure. The doors are attached to the fuselage framework by two hinges along the upper edge of the doors, and are secured in the closed position by Camloc fasteners around the perimeter of the doors. The fuselage plating for the F-106A in the fuselage nose section area consists of 7178-T6 bare sheet; see figure 4-4 for the plating arrangement and material type and gage. To provide the required aerodynamic smoothness, all plating is attached with flush-head rivets and fasteners.

4-11A. F-106A CLEAR TOP CANOPY. *Applicable after incorporation of TCTO 1F-106A-556.*

4-11B. The F-106A airplane canopy is an aluminum frame structure consisting of a machined arch assembly at the forward end and built up frames and sheeting in the aft portion. The canopy panel consists of a one-piece, single-ply stretched acrylic clear top transparency having a thickness of 0.330 to 0.380 inch except at the forward apex area which has a thickness of 0.215 to 0.245 inch. The panel is reinforced at the attachment areas with a laminate of nylon fabric which is bonded to the panel with acrylic resin. Acrylic shear strips are bonded to the inside forward and side edges of the panel assembly which engage frame installed aluminum shear strips upon panel installation. The panel is attached to the canopy frame with countersunk screws spaced approximately one inch apart. This improved canopy configuration provides the pilot with increased visibility above and to the rear, and provides additional head room above and to the sides. The panel does not incorporate the coated surface defog system.

4-11C. F-106A CLEAR TOP CANOPY BLEMISH LIMITS. *Applicable after incorporation of TCTO 1F-106A-556.*

4-11D. The canopy panel should be inspected and reworked or replaced as necessary to premit continued operation of the aircraft with the panel having a minimum number of blemishes. Small separate scratches having a maximum depth of 0.003 inch are acceptable but should be waxed with paste wax to minimize appearance. Figure 4-12A defines the use areas of the canopy panel and provides spacing limitations for blemishes that do not require rework. Refer to paragraph 4-11G for blemish depth measuring procedures. If the limits of figure 4-12A are exceeded, rework of the affected areas to the requirements of paragraphs 4-11E through 4-11J will be required. Scratches that exceed 0.003 inch in depth and nicks that exceed the limits of figure 4-12A that do not exhibit evidence of cracks, are acceptable if they can be sanded and/or buffed out without reducing the panel thickness to less than 0.31 inch in all areas except the forward apex area. The forward apex area which comprises a radius area of five inches from the panel top forward point, must not be reduced in thickness to less than 0.20 inch when working out scratches or nicks. Any sanding or polishing out of scratches or nicks must have a blend out slope of at least ten to one. Occasional areas of light crazing are acceptable providing no area is larger than three inches in any direction and no closer than 15 inches to the next crazed area. Crazed areas should not exceed four per canopy. Any conditions that exceed the rework limitations in paragraphs 4-11E through 4-11J, or any cracks in the panel assembly will be cause for panel replacement.



BLEMISH LIMITS

AREA	AREA VISION USE	BLEMISH QUANTITY LIMIT PER AREA	BLEMISH MINIMUM SPACING PER AREA
A	Taxiing and flying	8	3 Inches
B	Overhead and outboard scanning	10	3 Inches
	Rear view scanning	12	2 Inches

Note

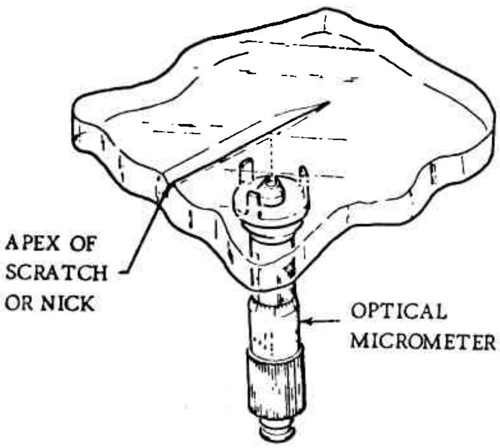
There are two A areas in the canopy panel. Where two or more blemishes appear in adjacent areas, limits of the area of least permissible blemishes shall apply.

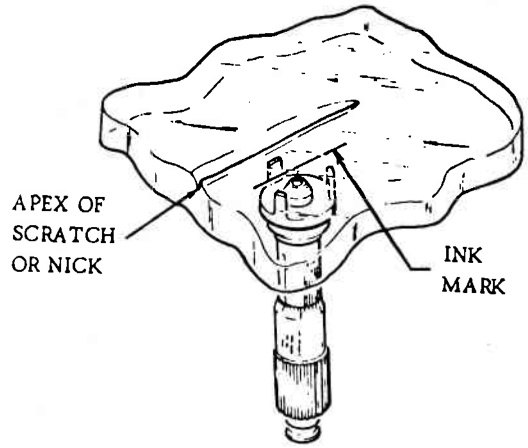
ACCEPTABLE BLEMISHES

SCRATCHES	Not exceeding 0.003 inch depth and are within blemish spacing limits. Wax affected area to make scratches less noticeable.
NICKS	Not exceeding 0.02-inch depth and 0.04-inch diameter, having no evidence of cracking and are within blemish spacing limits. Wax affected area to make nick less noticeable.
RAIN EROSION	Canopy glasses which have erosion beyond the forward edge of the screw holes shall be replaced.

Figure 4-12A. F-106A Clear Top Canopy Limits of Blemishes Not Requiring Rework

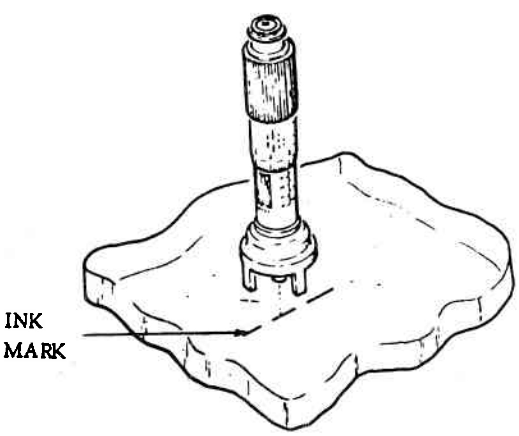
DAMAGED AREA MEASUREMENT
(PRIOR TO REWORK)

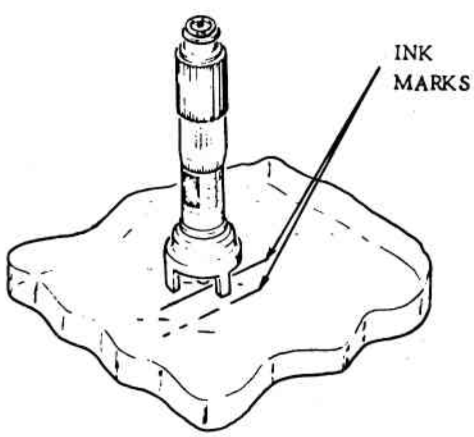




a. Stain apex of scratch or nick with ink.
b. Focus micrometer on scratch or nick apex from far side.
c. Record micrometer reading.
d. Make ink mark with felt tip pen on side opposite scratch or nick.
e. Focus micrometer on ink mark.
f. Record micrometer reading.
g. Find the difference between the two readings by subtracting the second reading from the first reading.
h. Multiply the difference by the refraction constant 1.5. Result is thickness of undamaged area.
i. Remove ink marks.

REWORKED AREA MEASUREMENT





a. Make ink mark on far side.
b. Focus micrometer on ink mark.
c. Record micrometer reading.
d. Make ink mark on near side.
e. Focus micrometer on ink mark.
f. Record micrometer reading.
g. Find the difference between the two readings by subtracting the second reading from the first reading.
h. Multiply the difference by the constant 1.5. Result is thickness of refinished area.
i. Remove ink marks.

Figure 4-12B. Canopy Panel Thickness Measurements

4-11E. CLEAR TOP CANOPY PANEL BLEMISH REFINISHING.

4-11F. EQUIPMENT REQUIRED.

FIGURE	NAME	TYPE	ALTERNATE	USE
	Buffing Wheel	Unstitched canton flanel - 6 inches diameter x 20 ply		For buffing blemishes.
	Drill Motor Arbor	1/4 inch		To hold buffing wheel.
	Flo-Master Felt Tips	1/4 inch round, N10, Esterbrook Pen Co. (81697)		For buffing blemishes.
	Drill Motor	Pneumatic, 1300 rpm or less		To power buffing equipment.
	Sanding Block	Hard rubber		To hold abrasive paper.
	Optical Micrometer	Model 966A1 (6650-A31-5532)		To measure depth of blemishes.
	Polishing Compound	MIL-W-18767, Type 1 (RM 7930-634-5340-6500)	Wilco Anti-Static Plastic Cleaner - Wilco Co., Los Angeles	For polishing canopy panel.
	Buffing Compound	Plastipol - Lasalco, Inc., St. Louis, Mo.	Equivalent	To buff blemishes (Ref. AN01-1A-12).
	Rouge	Chrome Rouge XZ- Lasalco, Inc., St. Louis, Mo.		To buff blemishes.
	Felt Tip Pen	Local Procurement		For marking canopy surfaces for thickness measurements.
	Wax	Simonize Paste Wax Simonize Co., Chicago, Ill.		To wax scratches.
	Disposable Paper Wipes	Brand 05316 Scott Paper Co., Philadelphia, Pa.	Equivalent	For cleaning panel surfaces.
	Abrasive Paper, wet or dry	No. 400		For sanding blemishes.
	Abrasive Paper, wet or dry	No. 600		For sanding blemishes.
	Polishing Cloth	Fed. Spec. DD-C-441, Type I (G7920-2051656)		For polishing panel surfaces.

4-11G. THICKNESS MEASUREMENT OF CLEAR TOP CANOPY PANEL.

4-11H. Prior to and after refinishing of a deep nick or scratch area in the canopy panel, measurements must be taken to determine that an acceptable thickness of material will or does remain after the nick or scratch has been worked out. See figure 4-12B for this procedure. In all areas of the canopy panel except the forward apex area, a minimum of 0.31-inch thickness of material must remain. In the forward apex area (five-inch radius from the top forward point), a minimum of 0.20-inch thickness of material must remain.

4-11J. PROCEDURE.

a. Small scratch or nick refinishing (scratch or nick exceeding 0.003-inch depth but not requiring sanding).

(1) Install a Flo-Master tip in the drill motor chuck.

(2) Sharpen the tip with a file.

(3) Apply Chrome Rouge XZ buffing compound to the tip.

(4) With motor running as slowly as possible, apply the tip point into the scratch or nick and work along the length of the scratch or to the depth of the nick. Resharpen tip and apply compound as required until a fingernail will no longer hang on the scratch or nick.

(5) Blend the reworked area into the surrounding local area using the six inch buffing wheel and Plastipol buffing compound. Do not run motor over 1300 rpm.

b. Large scratch or nick refinishing (requiring sanding).

(1) Measure undamaged panel depth at scratch or nick to determine advisability of rework. See figure 4-11G for this procedure. Replace panel if it is determined that rework will create an area in panel not acceptable in thickness. Refer to T.O. 1F-106A-2-2-2-2 for this procedure.

(2) To refinish a scratch, sand out scratch using rubber block and No. 400-A abrasive paper and water. Using a circular motion, sand an area approximately twice the diameter as the length of the scratch until the scratch is removed.

NOTE

All reworked areas must have a blend-out slope of at least ten to one.

(3) To refinish a nick, sand out nick using rubber block and No. 400-A abrasive paper and water. Using a circular motion, sand the area around the nick to create a minimum of at least a ten to one blend-out slope.

(4) Flush area thoroughly with water.

(5) Repeat step (2) or (3) using No. 600A abrasive paper and water over a slightly larger area until all No. 400 A sanding scratches are gone.

(6) Flush area thoroughly with water; dry area.

(7) Buff sanded area using six inch buffing wheel and Plastipol buffing compound. Apply buffing compound to wheel and apply wheel lightly to the sanded area. Move wheel and motor assembly in a direction 90° to the motor rotation. Do not let motor rpm exceed 1300 rpm. Buff until sanded area regains original luster.

(8) Clean area with plastic cleaner.

(9) Wax area to a high gloss finish.

c. After rework completion, check the canopy panel from the pilot's position that visual impairment or severe optical distortion was not created by the rework. When vision through a reworked area is distracted and is caused to focus on the panel surface, then a vision impairment condition exists. Severe optical distortion or vision impairment that cannot be alleviated by additional sanding and buffing is cause for canopy panel replacement.

d. Check for acceptable thickness of the panel reworked area using the optical micrometer. Refer to parageaph 4-11J for this procedure. Refer to paragraph 4-11E for panel thickness limitations.

4-11K. F-106A CANOPY PANEL BONDED COMPONENT PARTIAL SEPARATION REPAIR, *Applicable after incorporation of TCTO 1F-106A-556.*

4-11L. In the event that inspection reveals partial separation of a panel shear strip or of the nylon edge reinforcement, rebonding of the affected area is required prior to continued use of the canopy panel.

NOTE

Complete separation of shear strips from the panel panel will require panel replacement. Depot level repair is required for proper relocation of shear strips.

4-11M. PROCEDURE.

a. Remove panel from the canopy frame. Refer to T.O. 1F-106A-2-2-2-2 for this procedure.

b. Mask off area adjacent to area being bonded using tape, Federal Specification PPP-T-60, Type 3, Class I.

c. Clean area being bonded using Aliphatic Naphtha, Federal Specification TT-N-95

d. Inject bonding cement PS-18 (77902) into area being bonded using a clean metal point syringe. Refer to paragraph 11-45D for cement mixing and handling procedures.

Exercise care that all air pockets have been eliminated by application of the bonding cement.

e. Clamp bond area using spring clamps spaced to provide even pressure along entire length of bond area.

f. Remove any excess cement from bond area before cement has had time to set.

g. Allow repair area to cure for eight hours at 70°F temperature.

CAUTION

Do not use heat lamps for bond curing. Uncotrolled heating can cause the panel material to soften and distort.

h. Clean cement from the syringe before residual cement has had time to harden using methylene chloride.

WARNING

Methylene chloride is toxic if its vapors are inhaled over a period of time. Provide adequate ventilation when working with this material.

i. Remove clamps and masking tape. Clean the panel with plastic cleaner.

j. Install panel in canopy frame. Refer to T.O. 1F-106A-2-2-2-2 for this procedure.

4-11N. F-106A CANOPY LEDGE (PN 7144504-01) REPAIR.

1. Remove canopy assembly Part No. 7144504-01 from aircraft in accordance with T.O. 1F-106A-2-2-2-1.

2. Remove glass assembly from canopy frame in accordance with T.O. 1F-106A-2-2-2-2, Task 6-214.

NOTE

Measure glass position in frame for reinstallation.

3. Remove splices, Part No. 7144504-05 and 7144504-06, frame, Part No. 7144504-03, shear strip, Part No. 7144504-17, splices, Part No. 7144504-07 and 7144505-08, and ledge Part No. 7144504-01. Note: All parts are reusable except 7144504-01.

4. Clean all surfaces of old sealant, check and treat for corrosion, and paint as required.

5. Manufacture spacers in accordance with Air Force Drawing 8044104.

6. Manufacture new ledge in accordance with Air Force Drawing 8044105.

7. Fit and install new spacers and ledge in accordance with Air Force Drawing 8044106.

8. Reidentify repaired canopy ledge to Part No. 8044106.

9. Install glass assembly in accordance with T.O. 1F-106A-2-2-2-2, Task 6-215.

10. Install canopy assembly in accordance with T.O. 1F-106-2-2-2-1.

11. Pressure check cockpit for leaks.

4-12. Fuselage Nose Section—F-106B.

4-13. The F-106B fuselage nose section is similar to that of the F-106A, except between stations 102.00 and 472.00. See figures 4-5, 4-10, 4-12, and 4-14 for illustrations showing the structure and plating of the F-106B airplane. The principal difference between the F-106A and F-106B airplanes is basically the lengthening of the pilot's compartment and canopy to provide the two-place, tandem seating arrangement of the F-106B. Due to the increase in length of the pilot's compartment, the fuselage integral fuel tank is reduced in size from that of the F-106A and moved aft between stations 236.00 and 273.00. The equipment located in the upper aft electronic bay of the F-106A is moved aft to the center of the forward missile bay area in the F-106B. The lower halves of the missile bay doors are shortened to conform to this change in equipment location.

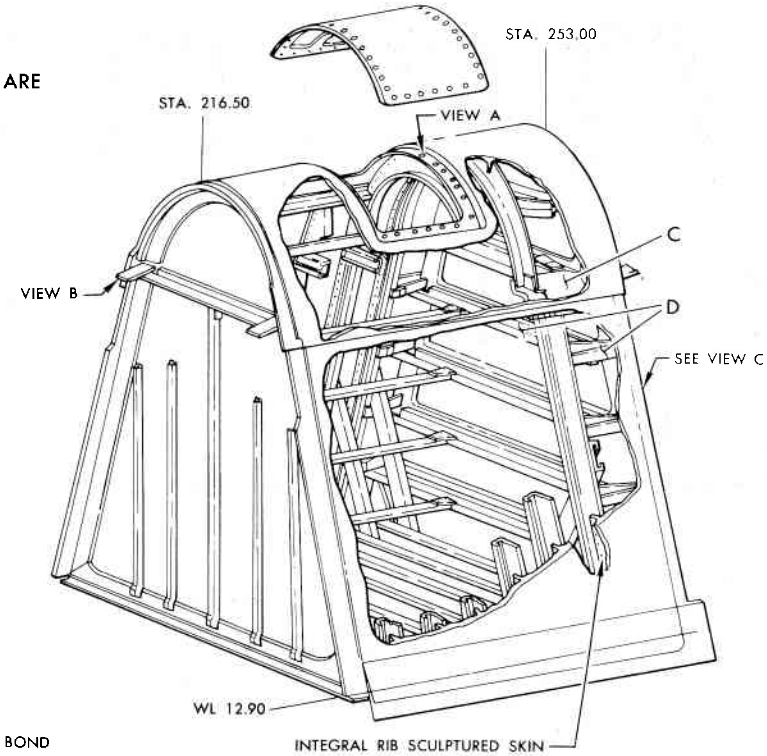
4-14. Fuselage Forward Intermediate Section—F-106A.

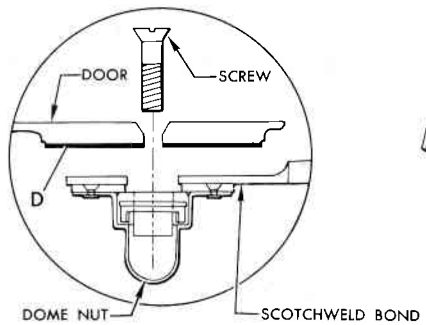
4-15. The forward intermediate section of the fuselage includes the fuselage structure, the engine air intake ducts, the variable ramp, the missile bay area with its missile bay doors, the engine air scroll, the main landing gear compartment, and the fuselage inner and outer plating. The wing attaching fittings for spars 1, 2, and 3 are located in this section of the fuselage structure. The fuselage structural framework in this section consists principally of a series of semicircular zee and channel spliced belt frames and forged and built-up type bulkheads connected by longerons, gussets and intercostals. See figure 4-16 for an illustration of fuselage framework. The fuselage plating in this section consists of stressed aluminum alloy. All plating in this area is attached with flush-head rivets and fasteners to meet the aerodynamic smoothness requirements as outlined in Section I. The engine air intake ducts are joined to the fuselage in such a way as to make them an integral part of the fuselage framework (see figures 4-17 and 4-18). The framework of the engine air ducts is made of a series of evenly spaced semicircular zee and channel spliced belt-frames interconnected by gussets and intercostals. The inner and outer plating of the engine air duct is made of stressed aluminum alloy and is attached by flush-head rivets and fasteners. Thermal anti-icing is provided in the forward area of the engine air intake duct. The variable ramp consists of three separate panels constructed of stressed aluminum alloy plating, press-formed channel ribs, and extruded channel stiffeners. Extruded hinges and steel hinge pins connect the forward and aft panels to the center panel. The center panel is slotted to bleed off boundary layer air when the variable ramp is in operation. The slot members are riveted to the plate and reinforced with 2024-T6 aluminum alloy clips. Cracked clips may be replaced and slot members repaired. See figure 4-19A for variable ramp panel hinge node and pin retention repair procedures. Expanded hinge node repairs may be accomplished in accordance with Air Force Drawing No.7645110. Refer to paragraph 4-46 for repair procedures of peeled nickel plate on the variable ramp duct leading edge. Damage exceeding those illustrated shall be referred to an aeronautical structures engineer. See figure 4-19 for an illustration of the variable ramp structure and plating. Refer to T.O. 1F-106A-2-4-2-1 for removal and installation of the variable ramp. The engine air scroll consists principally

MATERIAL

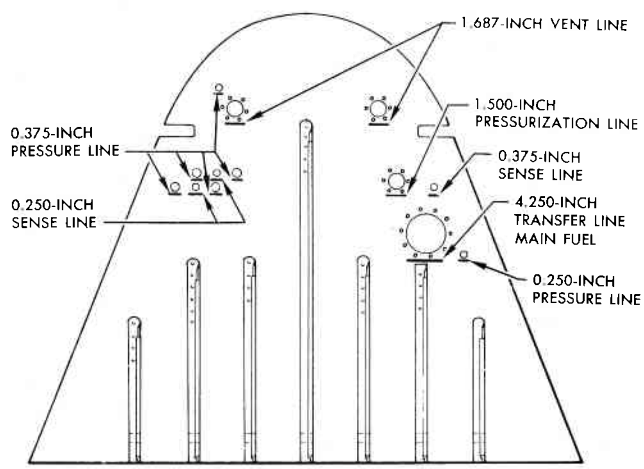
UNLESS OTHERWISE NOTED ALL PARTS ARE MADE FROM 7075-T6 BARE SHEET.

A. 7075-T6 extrusion.
B. 2014-T6 machined bar.
C. 7178-T6 bare sheet.
D. Fairprene seal.

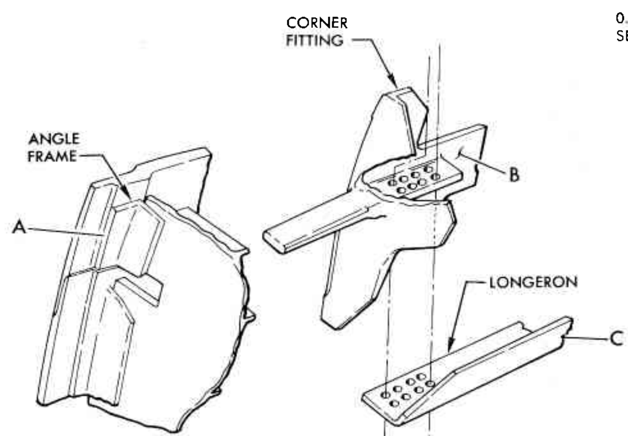




VIEW A



VIEW C

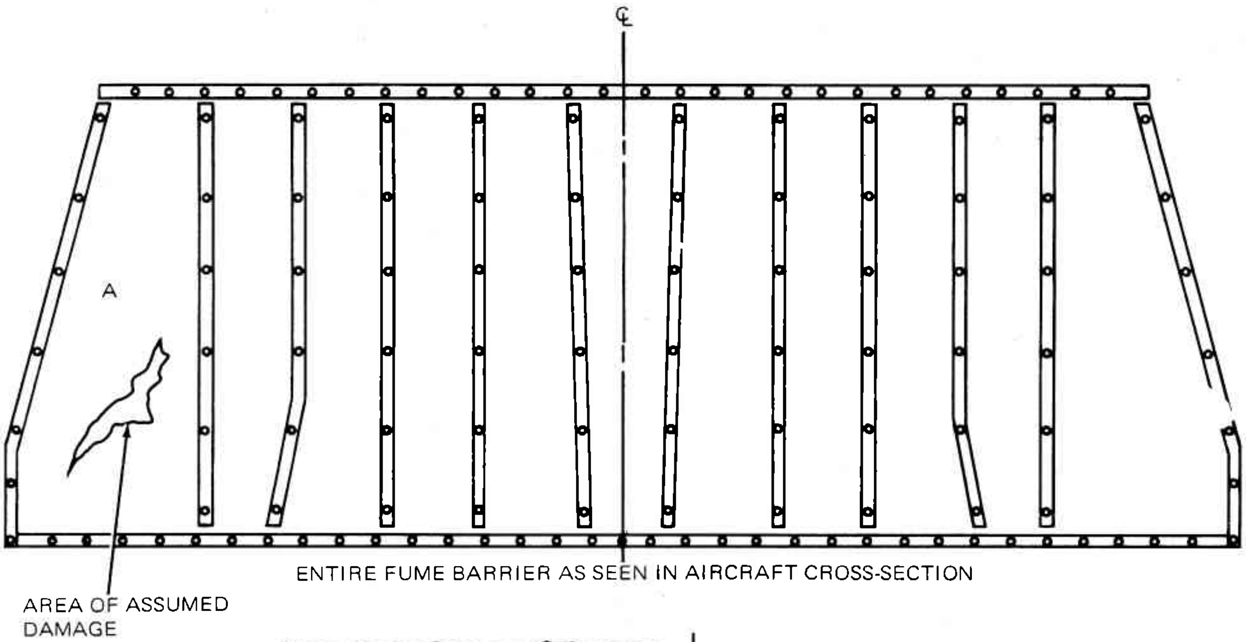


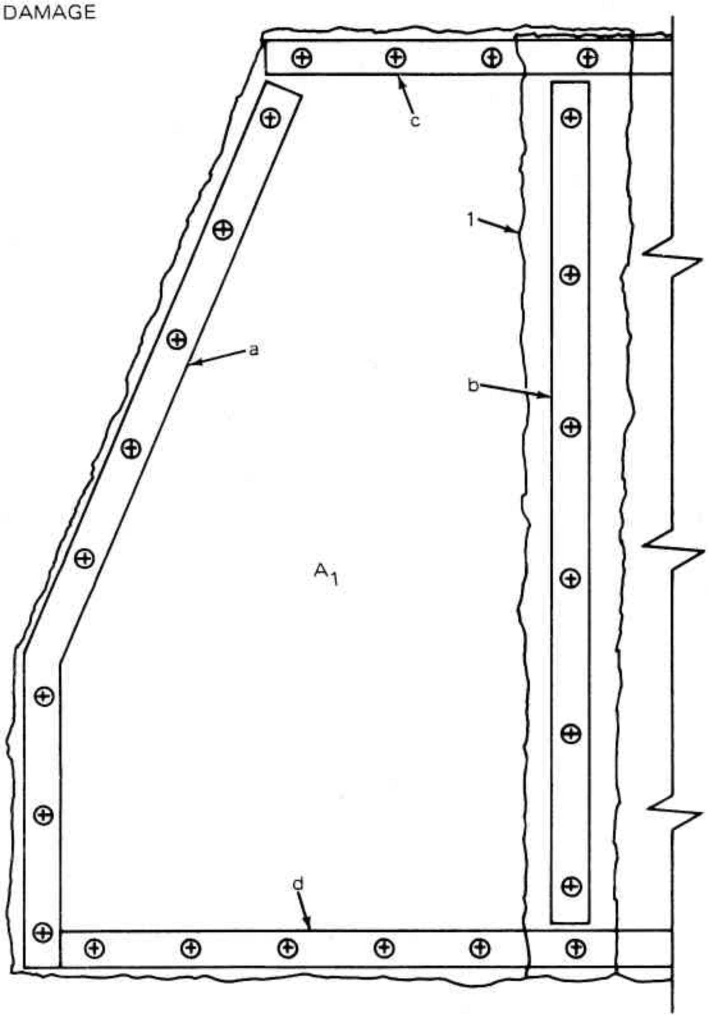
VIEW B
(EXPLODED)

NOTES:
1. REFER TO PARAGRAPH ON "REPAIRS" IN THIS SECTION FOR REPAIR INFORMATION.
2. REFER TO PARAGRAPH ON "NEGLIGIBLE DAMAGE LIMITS—FUSELAGE GROUP" IN THIS SECTION FOR DISPOSITION ON NEGLIGIBLE DAMAGE TO THIS COMPONENT.
3. REFER TO PARAGRAPH ON "REPAIRS" IN SECTION II FOR DOME NUT REPLACEMENT DATA.
4. REFER TO PARAGRAPH "FUSELAGE INTEGRAL FUEL TANK PRESSURE TEST" IN THIS SECTION FOR FUSELAGE FUEL TANK PRESSURE TEST PROCEDURE.

.06.03.132 A .53.06.02

Figure 4-13. Fuselage Integral Fuel Tank — F-106A





REPAIR PROCEDURE

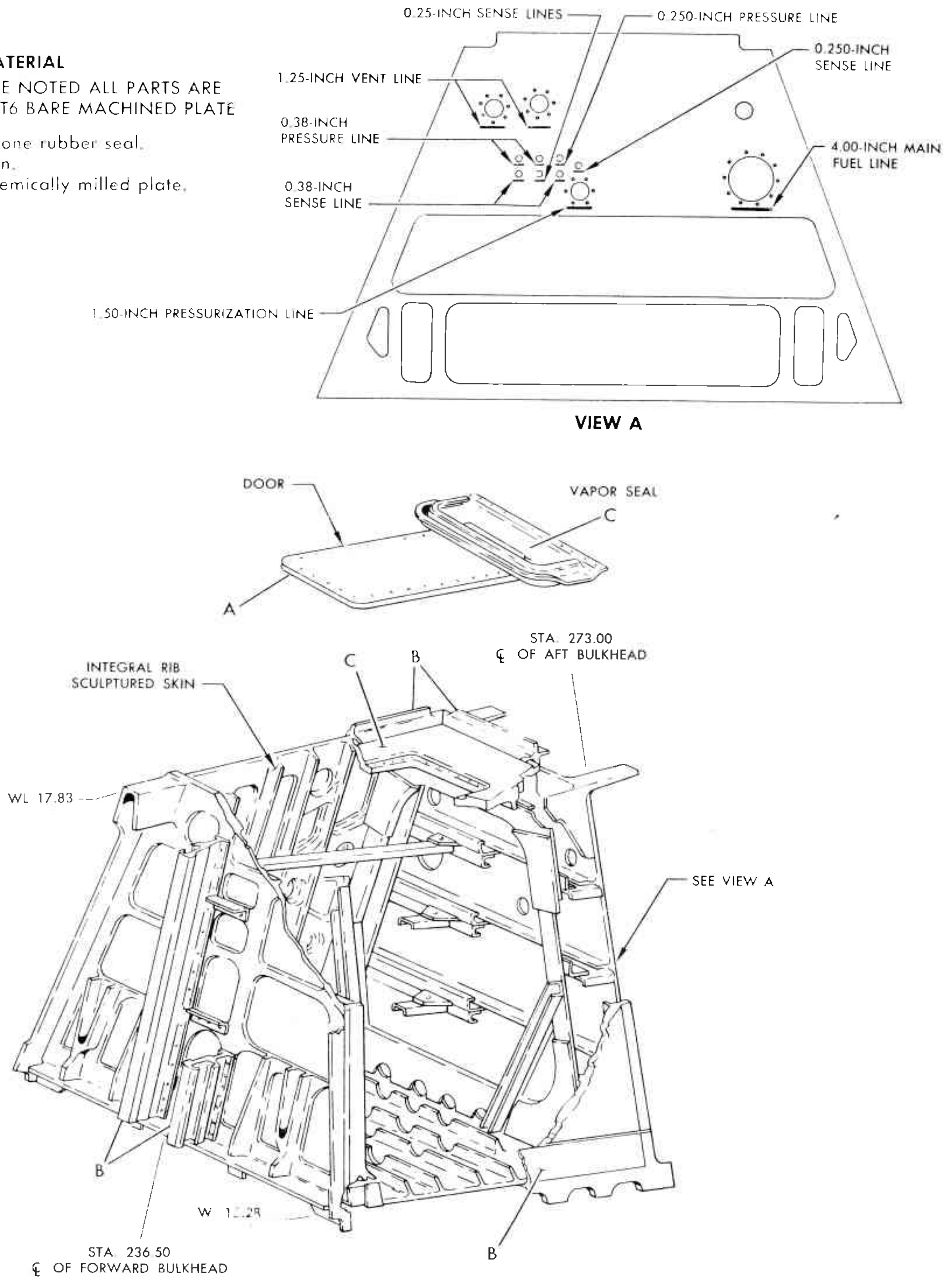
1. REMOVE a & b , LOOSEN c & d .
2. CUT DAMAGED BARRIER ALONG LINE 1 & REMOVE DAMAGED SECTION, AS ILLUSTRATED, A
3. CUT NEW PIECE A1 ; PUNCH HOLES AS NEEDED AND INSTALL.
4. APPLY THIN COAT MIL-S-8802 (OR EQUAL) SEALER TO ALL FAYING SURFACES
5. REINSTALL a & b , TIGHTEN c & d.

Figure 4-13A. Fume Barrier-Typical Repair-F-106A

MATERIAL

UNLESS OTHERWISE NOTED ALL PARTS ARE MADE FROM 7075-T6 BARE MACHINED PLATE

A. W54P white silicone rubber seal.
B. 7178-T6 extrusion.
C. 7075-T6 bare chemically milled plate.



NOTES:
1. REFER TO PARAGRAPH ON "REPAIRS" IN THIS SECTION FOR REPAIR INFORMATION.
2. REFER TO PARAGRAPH ON "NEGLIGIBLE DAMAGE LIMITS—FUSELAGE GROUP" IN THIS SECTION FOR DISPOSITION OF NEGLIGIBLE DAMAGE TO THIS COMPONENT.
3. REFER TO PARAGRAPH ON "REPAIRS" IN SECTION II FOR DOME NUT REPLACEMENT DATA.
4. REFER TO PARAGRAPH "FUSELAGE INTEGRAL FUEL TANK PRESSURE TEST" IN THIS SECTION FOR FUEL TANK PRESSURE TEST PROCEDURE.

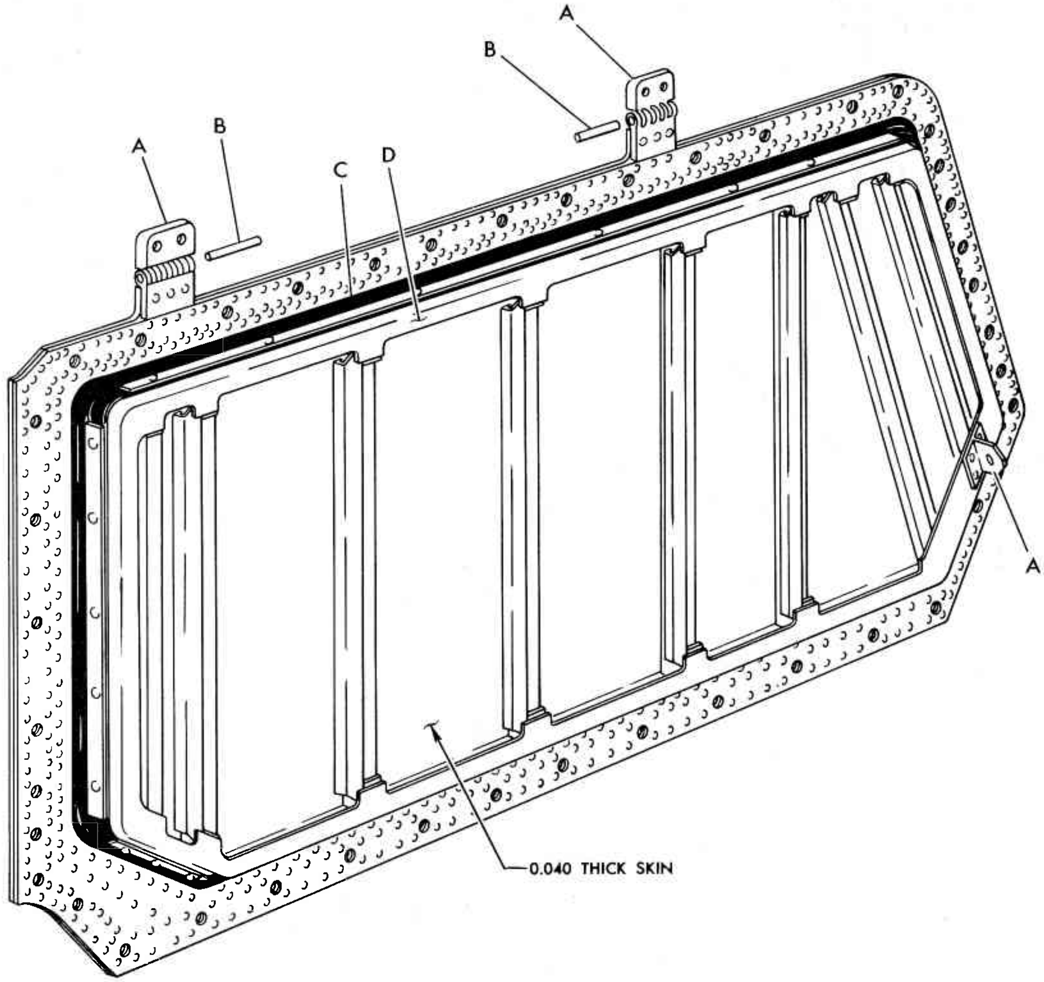
53.06.02

Figure 4-14. Fuselage Integral Fuel Tank — F-106B

MATERIAL

UNLESS OTHERWISE NOTED ALL PARTS ARE MADE FROM 7178-T6 BARE SHEET.

A. 7075-T6 extrusion.
B. 302 corrosion resistant steel.
C. Silicone rubber.
D. 7075-T6 clad sheet.



NOTES:
1. REFER TO PARAGRAPH ON "REPAIRS" IN THIS SECTION FOR REPAIR INFORMATION.
2. REFER TO PARAGRAPH ON "NEGLIGIBLE DAMAGE LIMITS—FUSELAGE GROUP" IN THIS SECTION FOR DISPOSITION OF NEGLIGIBLE DAMAGE TO THIS COMPONENT.
3. NUTS SECURING HINGE TO FUSELAGE MAY BE REPLACED WITH PLATE NUTS OF SAME SIZE AND MATERIAL WHEN REPLACEMENT OF HINGE IS NECESSARY.

.06.03.182 .53.18.02

Figure 4-15. Upper Aft Electronics Compartment Door Structure—F-106A

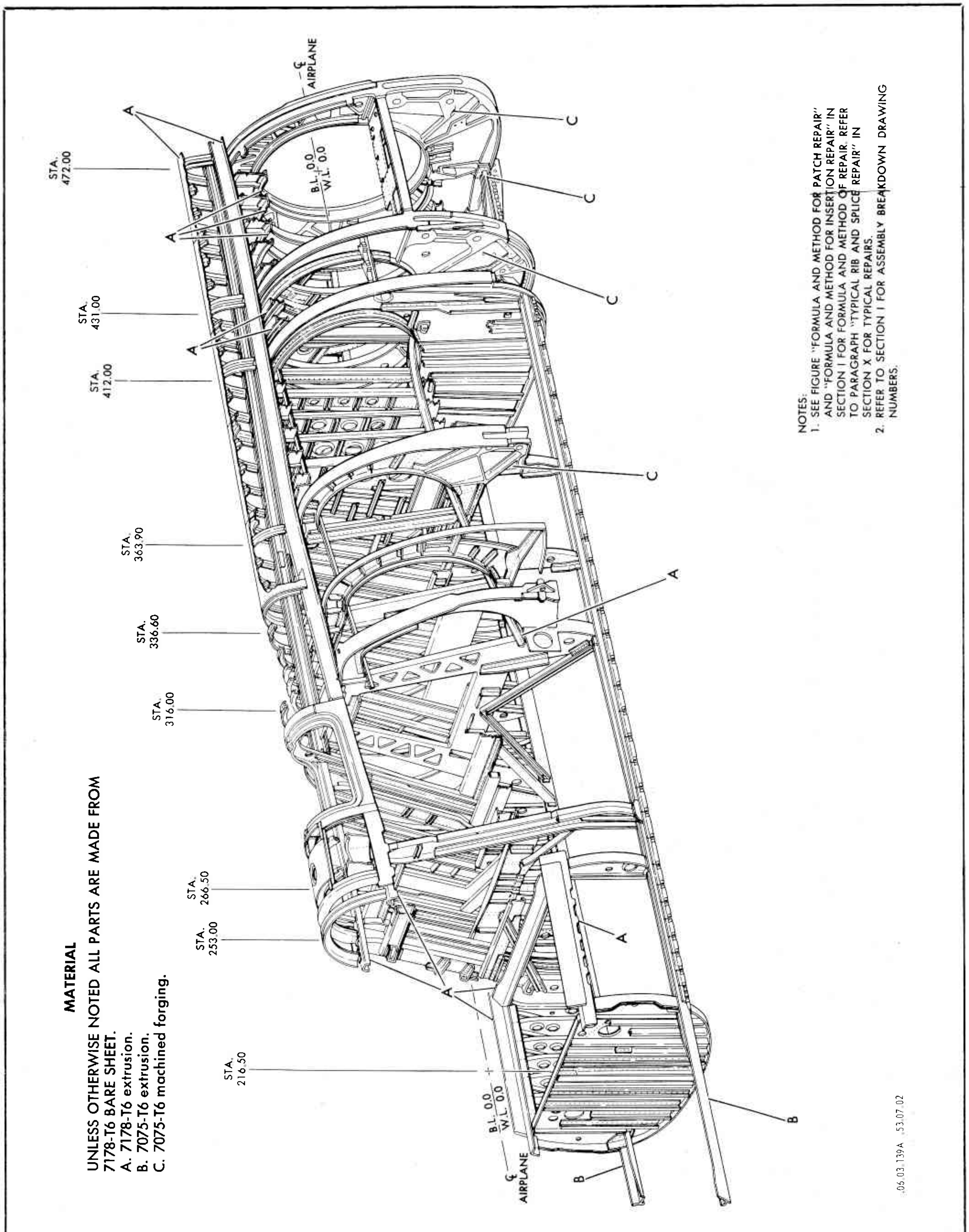
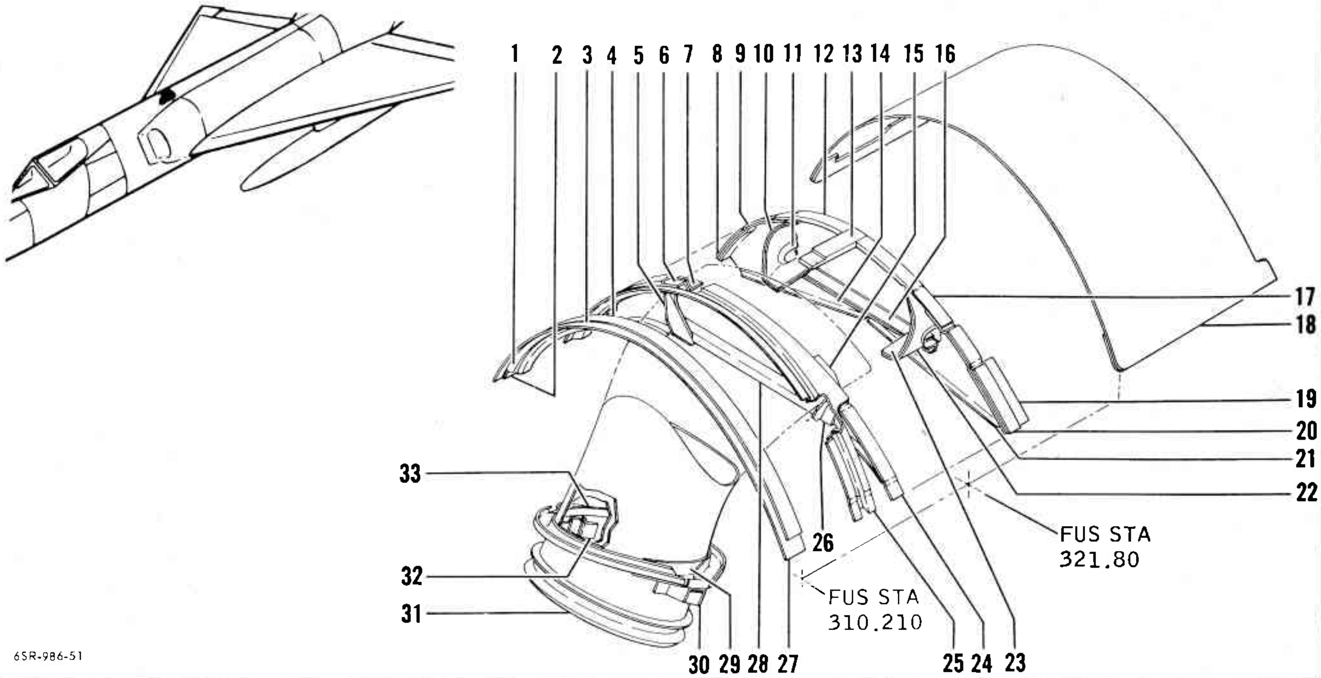


Figure 4-16. Fuselage Structure — Station 216.00 to 472.00 — F-106A



KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
1	66C40239-1	Seal		Silicone Rubber	MIL-R-25988	Replace
2	66J40690-15	Retainer	0.025 x 0.5 x 30.8	Cres Typ 301	MIL-S-5059	Replace
					Cond 1/4 Hd	
3	66D39731-1	Zee	0.04 x 2.0 x 50.0	7178-T6	QQ-A-250/14(T6)	10-7, 10-10A
4	66J40690-14	Filler	0.056 x 0.78 x 28.4	7178-T6	QQ-A-250/14(T6)	Replace
5	66J40690-12	Angle	0.032 x 1.71 x 3.75	Cres Typ 301	MIL-S-5059	Replace
					Cond 1/4 Hd	
6	66J40690-8	Support	0.032 x 6.8 x 7.0	Cres Typ 302	MIL-S-5059	10-10A, 10-10B,
					Cond 1/4 Hd	10-10C
7	66J40690-10	Angle	0.032 x 1.3 x 3.7	Cres Typ 301	MIL-S-5059	Replace
					Cond 1/4 Hd	
8	66J40690-2	Filler	0.05 x 0.750 x 3.63	2024-T3	QQ-A-250/4(T3)	Replace
9	66J40690-3	Filler	0.05 x 1.6 x 6.85	2024-T3	QQ-A-250/4(T3)	Replace
10	66D40693-2	Doubler	0.062 x 13.0 x 13.0	Cres Typ 301	MIL-S-5059	Replace
					Cond 1/4 Hd	
11	66C40692-1	Seal	0.062 x 3.0 x 6.0	Rubber Asbestos	MIL-A-7021CL Medium	Replace
				0.062		
12	66E40694-2	Shim	0.75 x 1.5 x 10.5	2024-T4	QQ-A-225/6(T4)	Replace
13	66J40690-17	Filler	0.19 x 1.2 x 4.1	7178-T6	QQ-A-250/14(T6)	Replace
14	66J40690-6	Support	0.032 x 6.0 x 11.5	Cres Typ 302	MIL-S-5059	10-10A, 10-10B
					Cond 1/4 Hd	
15	66J40690-9	Angle	0.032 x 1.3 x 3.7	Cres Typ 301	MIL-S-5059	Replace
					Cond 1/4 Hd	
16	66J40690-4	Channel	0.032 x 2.2 x 10.3	Cres Typ 301	MIL-S-5059	Replace
					Cond 1/4 Hd	
17	66E40694-1	Shim	0.75 x 1.5 x 10.5	2024-T4	QQ-A-225/6(T4)	Replace
18	66J40691-2	Skin	0.04 x 11.59 x 38.0	7178-T6	QQ-A-250/14(T6)	4-18F, 10-1
19	66J40690-3	Filler	0.05 x 1.6 x 6.25	2024-T3	QQ-A-250/4(T3)	Replace
20	66J40690-2	Filler	0.05 x 0.75 x 3.63	2024-T3	QQ-A-250/4(T3)	Replace
21	66J40690-5	Support	0.032 x 6.0 x 11.5	Cres Typ 302	MIL-S-5059	10-10A, 10-10B
					Cond 1/4 Hd	

Figure 4-16A. Aerial Refueling Dorsal Section, F-106A Stations 308.5 to 322.0 (Sheet 1 of 2)

Applicable after incorporation of TCTO 1F-106-986

KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
22	66C40692-1	Seal	0.062 x 3.0 x 6.0	Asbestos Rubber	MIL-A-7021 Medium	Replace
23	66D40693-1	Doubler	0.062 x 13.0 x 13.0	Cres Typ 301	MIL-S-5059	Replace
					Cond 1/4 Hd	
24	66J40690-7	Support	0.032 x 6.8 x 7.0	Cres Typ 302	MIL-S-5059	10-10A
					Cond 1/4 Hd	
25	66D39732-1	Hat	0.05 x 3.0 x 50.0	7178-T6	QQ-A-250/14(T6)	10-10A, 10-10C
26	66J40690-11	Angle	0.032 x 1.71 x 3.75	Cres Typ 301	MIL-S-5059	Replace
					Cond 1/4 Hd	
27	66J40690-16	Retainer	0.032 x 1.34 x 30.8	7178-T6	QQ-A-250/14(T6)	*
28	66J40690-13	Channel	0.032 x 1.88 x 19.7	Cres Typ 301	MIL-S-5059	10-5
					Cond 1/4 Hd	
29	66J40235-11	Gusset	0.04 x 2.8 x 4.8	Cres Typ 321/347	MIL-S-6721	Replace
30	66J40235-9	Clip	0.04 x 3.1 x 5.6	Cres Typ 321/347	MIL-S-6721	Replace
31	66J40235-2	Duct (Weldment)		Cres Typ 321/347	MIL-S-6721	10-10D
32	66J40235-10	Clip	0.04 x 3.1 x 5.6	Cres Typ 321/347	MIL-S-6721	Replace
33	66J40235-12	Gusset	0.04 x 2.8 x 4.8	Cres Typ 321/347	MIL-S-6721	Replace

*Remove damaged area. Insert new piece picking up three rivets (minimum)

Figure 4-16A. Aerial Refueling Dorsal Section, F-106A Stations 308.5 to 322.0 (Sheet 2 of 2)
Applicable after incorporation of TCTO 1F-106-986

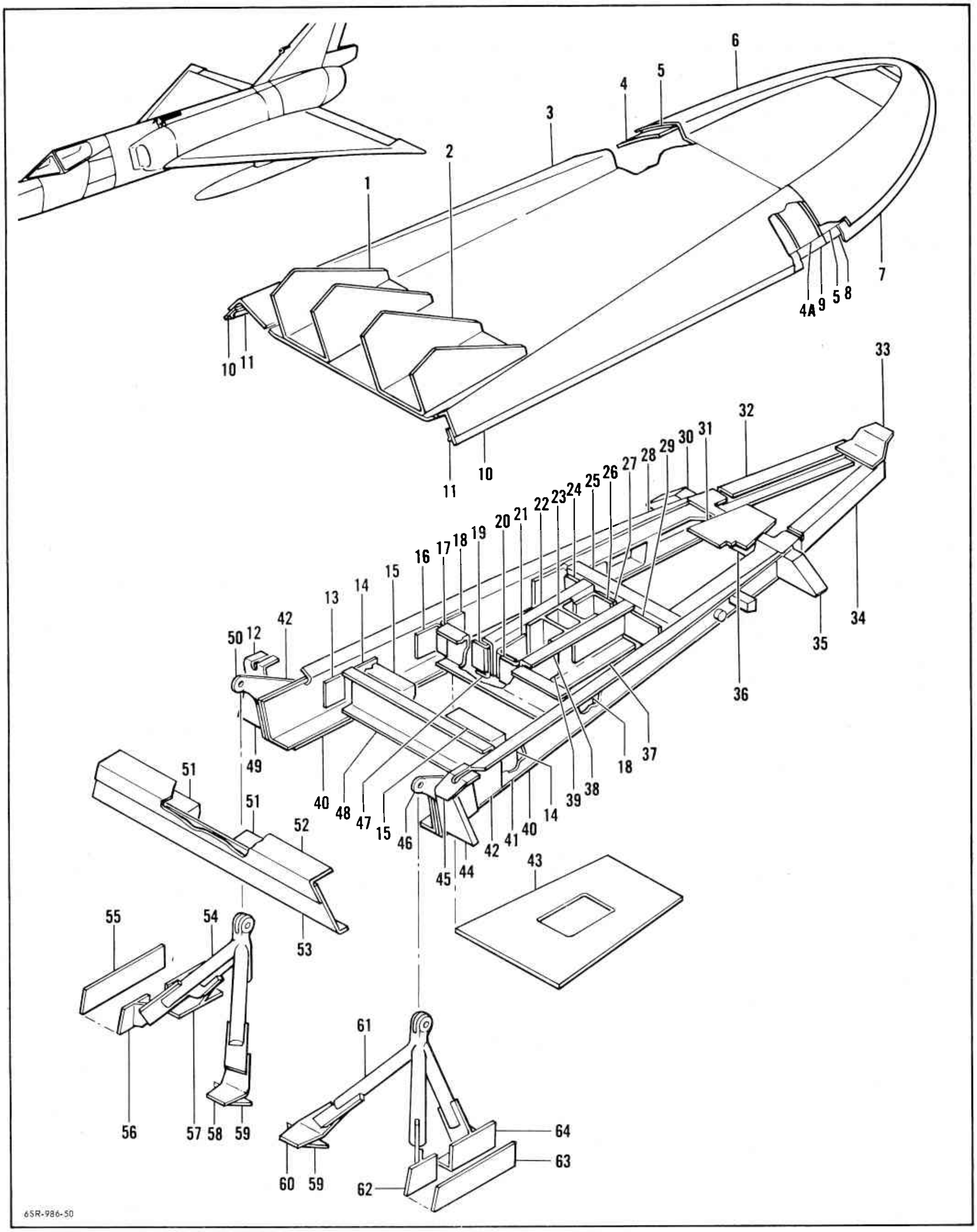


Figure 4-16B. Aerial Refueling Slipway Door and Hinge Supports, F-106A (Sheet 1 of 3)
Applicable after incorporation of TCTO 1F-106-986

KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
1	66D40674-6	Channel	0.19 x 8.5 x 12.0	4130 Cond A	MIL-S-18729	10-10E
		(Cad. Plat)				
2	66D40674-5	Channel	0.19 x 8.5 x 12.0	4130 Cond A	MIL-S-18729	10-10B
		(Cad. Plat)				
3	66J40525-1	Fairing	0.125 x 24.0 x 31.0	Cres Typ 301	MIL-S-5059	10-2, 10-10B
					Cond 1/4 HD	
4	66C39710-1	Plate	0.25 x 2.41 x 4.10	Cres Typ 301		Replace
4A	66C39710-2	Plate	0.25 x 2.41 x 4.10	Cres Typ 301	MIL-S-5059	Replace
					Cond 1/4 HD	
5	66J40259-3	Filler	0.032 x 1.0 x 3.0	2024-T3	QQ-A-250/4(T3)	Replace
6	66J40689-1	Fairing	0.09 x 20.5 x 26.0	Cres Typ 302	MIL-S-5059	10-10E*
					Cond A	
7	66J40259-8	Seal	0.093 x 12.5 x 15.0	Syn Rub	AMS 3208	Replace
					Shore 50	
8	66D39721-1	Retainer	0.060 x 0.7 x 17.0	2024-T3	QQ-A-250/4(T3)	Replace
9	66J40259-7	Retainer	0.063 x 0.7 x 3.0	7075-T6	QQ-A-250/12(T6)	Replace
10	66J40259-9	Seal	0.093 x 12.5 x 15.0	Syn Rub	AMS 3208	Replace
					Shore 50	
11	66D39720-1	Retainer	E491002 x 24.5	2024-T4	QQ-A-200/3(T4)	Replace
12	66C39722-2	Bracket	0.071 x 1.7 x 6.0	7075-T6	QQ-A-250/12(T6)	Replace
13	66C40688-1	Angle	0.071 x 2.0 x 2.7	7075-T6	QQ-A-250/12(T6)	Replace
14	66C40688-2	Angle	0.071 x 2.0 x 5.4	7074-T6	QQ-A-250/12(T6)	Replace
15	66C40673-1	Adapter	1.0 x 2.5 x 3.5	7075-T651	QQ-A-250/12(T651)	Replace
16	66C40688-3	Angle	0.071 x 1.6 x 2.9	7075-T6	QQ-A-250/12(T6)	Replace
17	66C40684-2	Rib	0.071 x 3.8 x 11.2	7075-T6	QQ-A-250/12(T6)	10-10A, 10-10C
18	66C40688-4	Angle	0.071 x 1.8 x 3.4	7075-T6	QQ-A-250/12(T6)	Replace
19	66C39717-4	Bracket	0.071 x 1.8 x 3.7	7075-T6	QQ-A-250/12(T6)	Replace
20	66C39723-1	Bracket	0.071 x 3.0 x 6.8	7075-T6	QQ-A-250/12(T6)	Replace
21	66C39717-2	Bracket	0.071 x 1.8 x 4.3	7075-T6	QQ-A-250/12(T6)	Replace
22	66C39723-2	Bracket	0.071 x 3.0 x 6.8	7075-T6	QQ-A-250/12(T6)	Replace
23	66D40429-1	Support		7079-T651	QQ-A-250/17(T651)	Replace
24	66C39715-4	Bracket	0.071 x 2.2 x 3.0	7075-T6	QQ-A-250/12(T6)	Replace
25	66D40685-2	Rib	0.071 x 4.0 x 7.5	7075-T6	QQ-A-250/12(T6)	10-10A, 10-10C
26	66C39718-2	Bracket	0.71 x 1.5 x 3.4	7075-T6	QQ-A-250/12(T6)	Replace
27	66C39718-1	Bracket	0.071 x 1.5 x 3.4	7075-T6	QQ-A-250/12(T6)	Replace
28	66E39724-2	Beam	0.071 x 10.0 x 20.0	7075-T6	QQ-A-250/12(T6)	10-10A, 10-10C
29	66C39715-3	Bracket	0.071 x 2.2 x 3.0	7075-T6	QQ-A-250/12(T6)	Replace
30	66E40671-2	Stop	2.5 x 5.0 x 12.25	7079-T651	QQ-A-250/17(T651)	Replace
31	66C39711-1	Plate	0.125 x 3.45 x 3.18	Cres Typ 301	MIL-S-5059	Replace
					Cond 1/4 HD	
32	66D39713-4	Channel	0.071 x 1.44 x 8.71	7075-T6	QQ-A-250/12(T6)	10-10C
33	66C39725-1	Beam	0.100 x 2.9 x 2.9	7075-T6	QQ-A-250/12(T6)	Replace
34	66D39713-3	Channel	0.071 x 1.44 x 8.71	7075-T6	QQ-A-250/12(T6)	10-10C
35	66E40671-1	Stop	2.5 x 5.0 x 12.25	7079-T651	QQ-A-250/17(T651)	Replace
36	66C39712-2	Channel	0.8 x 1.25 x 6.06	7075-T6	QQ-A-250/12(T6)	Replace
37	66C39716-1	Bracket	0.071 x 3.0 x 6.8	7075-T6	QQ-A-250/12(T6)	Replace
38	66C39717-1	Bracket	0.071 x 1.8 x 4.3	7075-T6	QQ-A-250/12(T6)	Replace
39	66C39717-3	Bracket	0.071 x 1.8 x 3.7	7075-T6	QQ-A-250/12(T6)	Replace
40	66J40259-6	Strap	0.071 x 1.3 x 27.6	7075-T6	QQ-A-250/12(T6)	10-10E
41	66E39724-1	Beam	0.071 x 13.7 x 25.8	7075-T6	QQ-A-250/12(T6)	10-10A, 10-10C
42	66C39719-1	Filler	0.1 x 2.36 x 5.30	7075-T6	QQ-A-250/12(T6)	Replace
43	66J40259-2	Plate	0.16 x 7.0 x 8.3	2024-T3	QQ-A-250/4(T3)	Replace
44	66C39714-1	Channel	0.071 x 4.2 x 3.9	7075-T6	QQ-A-250/12(T6)	Replace
45	66C39722-1	Bracket	0.071 x 1.7 x 6.0	7075-T6	QQ-A-250/12(T6)	Replace
46	66C40675-1	Hinge Plate	0.25 x 4.1 x 5.2	4130 Cond A	MIL-S-18729	Replace
47	66C39716-2	Bracket	0.071 x 3.0 x 6.8	7075-T6	QQ-A-250/12(T6)	Replace
48	66C40684-1	Rib	0.071 x 3.8 x 11.2	7075-T6	QQ-A-250/12(T6)	10-10A, 10-10C
	*Leave in as welded condition.					

Figure 4-16B. Aerial Refueling Slipway Door and Hinge Supports, F-106A (Sheet 2 of 3)

Applicable after incorporation of TCTO 1F-106-986

KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
49	66C39714-2	Channel	0.071 x 4.2 x 3.9	7075-T6	QQ-A-250/12(T6)	Replace
50	66C40675-2	Hinge Plate	0.25 x 4.1 x 5.2	4130 Cond A	MIL-S-18729	Replace
51	66C40672-1	Adapter	1.5 x 3.0 x 4.0	7075-T651	QQ-A-250/12(T651)	Replace
52	66C40686-1	Angle	0.071 x 2.7 x 11.0	7075-T6	QQ-A-250/12(T6)	10-10A
53	66C40687-1	Angle	0.08 x 4.5 x 16.8	7075-T6	QQ-A-250/12(T6)	10-10A
54	66E40248-2	Support		4130 Cond A or N	MIL-S-18729	Replace
		(Weldment)				
55	66J40429-7	Plate	0.125 x 1.7 x 6.0	4130 Cond A or N	MIL-S-18729	Replace
56	66D40255-2	Support		4130 Cond A or N	MIL-S-18729	10-10E
		(Weldment)				
57	66D40262-2	Support		4130 Cond A or N	MIL-S-18729	10-10E
		(Weldment)				
58	66J40249-4	Angle	0.19 x 2.31 x 4.5	4130 Cond A or N	MIL-S-18729	Replace
59	66J40249-6	Plate	0.125 x 1.7 x 2.3	7075 Cond A or N	QQ-A-250/12	Replace
60	66J40249-3	Angle	0.19 x 2.31 x 4.5	4130 Cond A or N	MIL-S-18729	Replace
61	66E40248-1	Support		4130 Cond A or N	MIL-S-18729	Replace
		(Weldment)				
62	66D40255-1	Support		4130 Cond A or N	MIL-S-18729	10-10E
		(Weldment)				
63	66J40249-5	Plate	0.125 x 1.7 x 6.0	4130 Cond A or N	MIL-S-18729	Replace
64	66D40262-1	Support		4130 Cond A or N	MIL-S-18729	10-10E
		(Weldment)				

Figure 4-16B. Aerial Refueling Slipway Door and Hinge Supports, F-106A (Sheet 3 of 3)
Applicable after incorporation of TCTO 1F-106-986

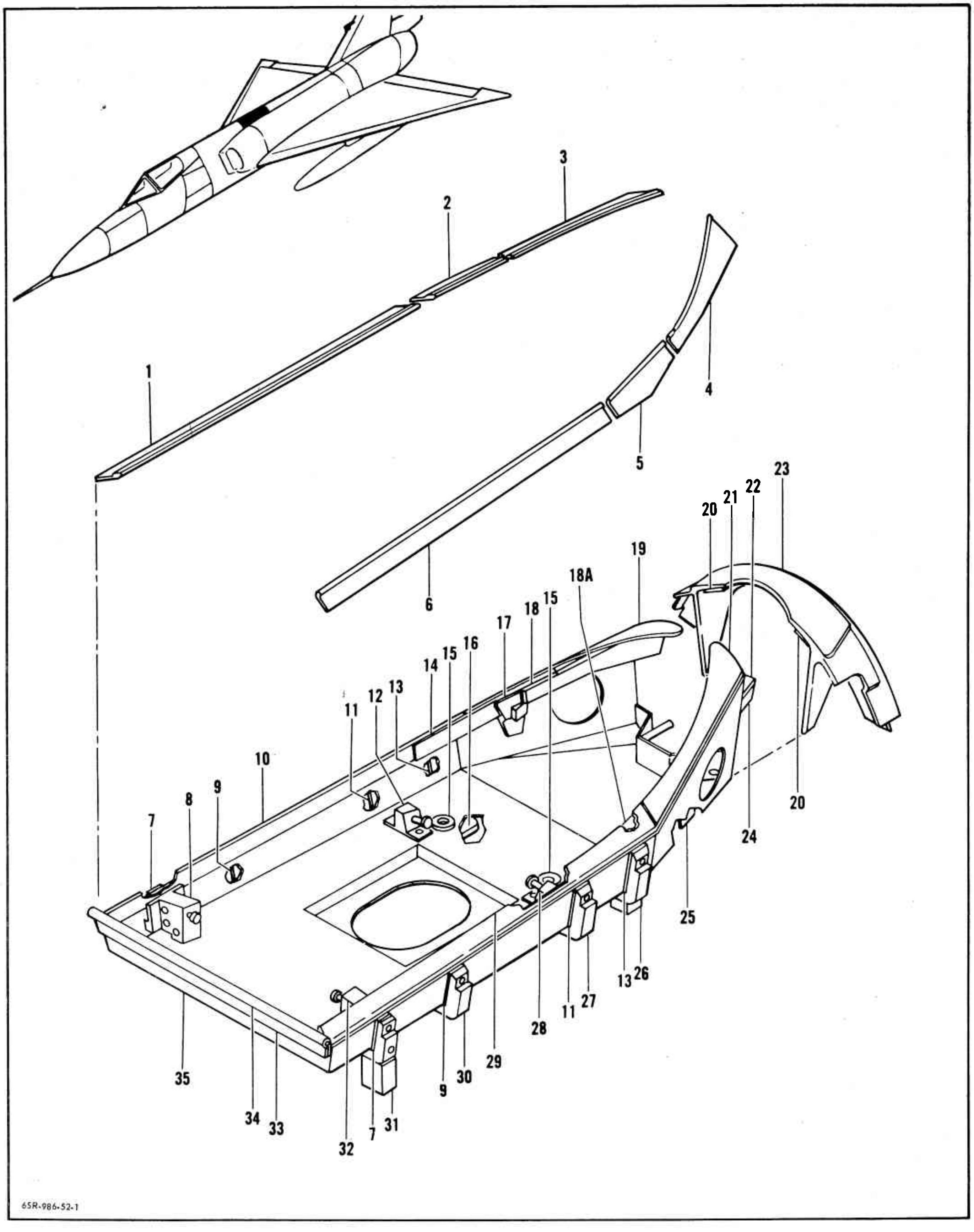


Figure 4-16C. Aerial Refueling Dorsal Sections, F-106A, Stations 320.0 to 370.7 (Sheet 1 of 4)
Applicable after incorporation of TCTO 1F-106-986

KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
1	66C39842-2	Guard	0.25 x 1.4 x 24.0	7075-T6	QQ-A-250/12(T6)	Replace*
2	66C39845-2	Striker	0.25 x 3.0 x 7.0	7075-T6	QQ-A-250/12(T6)	Replace
3	66C39843-2	Striker	0.25 x 2.5 x 10.0	7075-T651	QQ-A-250/12(T651)	Replace
4	66C39843-1	Striker	0.25 x 2.5 x 10.0	7075-T651	QQ-A-250/12(T651)	Replace
5	66C39845-1	Striker	0.25 x 3.0 x 7.0	7075-T6	QQ-A-250/12(T6)	Replace
6	66C39842-1	Guard	0.25 x 1.4 x 24.0	7075-T6	QQ-A-250/12(T6)	Replace*
7	66J40246-5	Shim	0.032 x 2.40		AMS 4013	Replace
8	66D39811-2	Stop	2.5 x 3.5 x 3.5	7079-T651	QQ-A-250/17(T65)	Replace
9	66J40246-4	Shim	0.032 x 2.60		AMS 4013	Replace
10	66D39818-2	Angle	0.10 x 4.0 x 28.0	7075-T6	QQ-A-250/13(T6)	Replace
11	66J40246-3	Shim	0.032 x 2.90		AMS 4013	Replace
12	66D39810-2	Stop	1.25 x 1.375 x 2.75	7079-T651	QQ-A-250/17(T651)	Replace
13	66J40246-2	Shim	0.032 x 3.00		AMS 4013	Replace
14	66D39819-4	Angle	0.10 x 8.0 x 10.0	7075-T6	QQ-A-250/13(T6)	Replace
15	MS35489-14	Grommet	1/16 x 5/16	Rubber	MIL-G-3036	Replace
16	66J39816-7	Channel	0.050 x 3.50 x 3.50	2024-T3	QQ-A-250/4(T3)	Replace
17	66C39804-2	Receptacle	1.0 x 2.75 x 3.5	7075-T651	QQ-A-250/12(T651)	Replace
18	66C39851-2	Probe	0.875 x 2.25 x 9.0	7075-T651	QQ-A-250/12(T651)	Replace
18A	66D39819-3	Angle	0.10 x 8.0 x 11.0	7075-T6	QQ-A-250/13(T6)	Replace
19	66D39817-6	Angle	0.10 x 5.0 x 12.0	7075-T6	QQ-A-250/13(T6)	Replace
20	66J39840-27	Filler	0.125 x 0.8 x 1.5	2024-T6	QQ-A-250/12(T6)	Replace
21	66D39817-5	Angle	0.10 x 5.0 x 12.0	7075-T6	QQ-A-250/13(T6)	Replace
22	66C39851-1	Probe	0.875 x 2.25 x 9.0	7075-T651	QQ-A-250/12(T651)	Replace
23	66J40443-1	Dorsal Section	4.0 x 7.75 x 19.5	7079-T651	QQ-A-250/17(T651)	10-9
24	66J40443-3 ***	Dorsal Section	4.0 x 7.75 x 19.5	7075-T651	QQ-A-250/12(T651)	10-9
25	66J39840-24	Filler	0.160 x 1.35 x 1.5	2024-T3	QQ-A-250/4(T3)	Replace
26	66C39804-1	Receptacle	1.0 x 2.75 x 3.5	7075-T651	QQ-A-250/12(T651)	Replace
27	66D39824-1	Strap (Weldment-		4130	MIL-S-18729	Replace
		Cad. Plate)				
28	66D39826-1	Strap	0.625 x 1.25 x 21.5	4130-Cond N	MIL-S-18729(N)	Replace
29	66D39810-1	Stop	1.25 x 1.375 x 2.75	7079-T651	QQ-A-250/17(T651)	Replace
30	66D39818-1	Angle	0.10 x 4.0 x 28.0	7075-T6	QQ-A-250/13(T6)	Replace
31	66D39856-1	Strap	0.625 x 1.25 x 20.5	4130-Cond N	MIL-S-18729(N)	Replace
32	66D39825-1	Strap (Weldment-		4130	MIL-S-18729	Replace
		Cad. Plate)				
33	66D39811-1	Stop	2.5 x 3.5 x 3.5	7079-T651	QQ-A-250/17(T651)	Replace
34	66J39816-5	Strap	0.050 x 0.63 x 20.0	2024-T3	WW-T-700/6"	Replace
					Typ 1(T3)	
35	66J39816-11	Seal	18.0	Silicone Rubber	MIL-R-25988	Replace
36	66J39816-9	Pan (Weldment)		6061-T6	QQ-A-250/11(T6)	10-2, 10-10B,
						10-10E**
37	66J39806-2	Skin Assy		7075-T6	QQ-A-250/12(T6)	4-18G
		(Weldment)				
38	66E39841-1	Skin Assy		2024-T3	QQ-A-250/4(T3)	4-16F, 4-16G,
		(Weldment)				10-10C
39	66J39840-29	Clip	0.5 x 1.16 x 2.68	2024-T6	QQ-A-250/12(T6)	Replace
40	66J39840-28	Clip	0.5 x 1.16 x 2.68	2024-T6	QQ-A-250/12(T6)	Replace
41	67E36507-1	Hat	0.040 x 2.5 x 50.0	7075-T6	QQ-A-250/12(T6)	10-10A, 10-10C
42	66J39806-1	Skin Assy		7075-T6	QQ-A-250/12(T6)	4-18G
		(Weldment)				
43	66J39840-17	Angle	0.063 x 1.6 x 2.67	2024-T42	QQ-A-250/4(T42)	Replace
44	66C39796-2	Intercostal	0.063 x 5.0 x 5.0	7075-T6	QQ-A-250/12(T6)	Replace
45	66J39840-4	Angle	0.063 x 1.6 x 2.67	2024-T42	QQ-A-250/4(T42)	Replace
46	66J39840-19	Angle	0.063 x 1.6 x 2.64	2024-T42	QQ-A-250/4(T42)	Replace
47	66C39798-2	Intercostal	0.063 x 4.2 x 5.4	7075-T6	QQ-A-250/12(T6)	Replace
48	66J39840-6	Angle	0.063 x 1.6 x 2.64	2024-T42	QQ-A-250/4(T42)	Replace
49	66J39840-21	Angle	0.063 x 1.6 x 2.74	2024-T42	QQ-A-250/4(T42)	Replace

*Portion of guard may be replaced. Butt joints must be above dorsal ribs.

Seal repair using EC1291. * Suitable sub for 66J40443-1

Figure 4-16C. Aerial Refueling Dorsal Sections, F-106A, Stations 320.0 to 370.7 (Sheet 2 of 4)

Applicable after incorporation of TCTO 1F-106-986

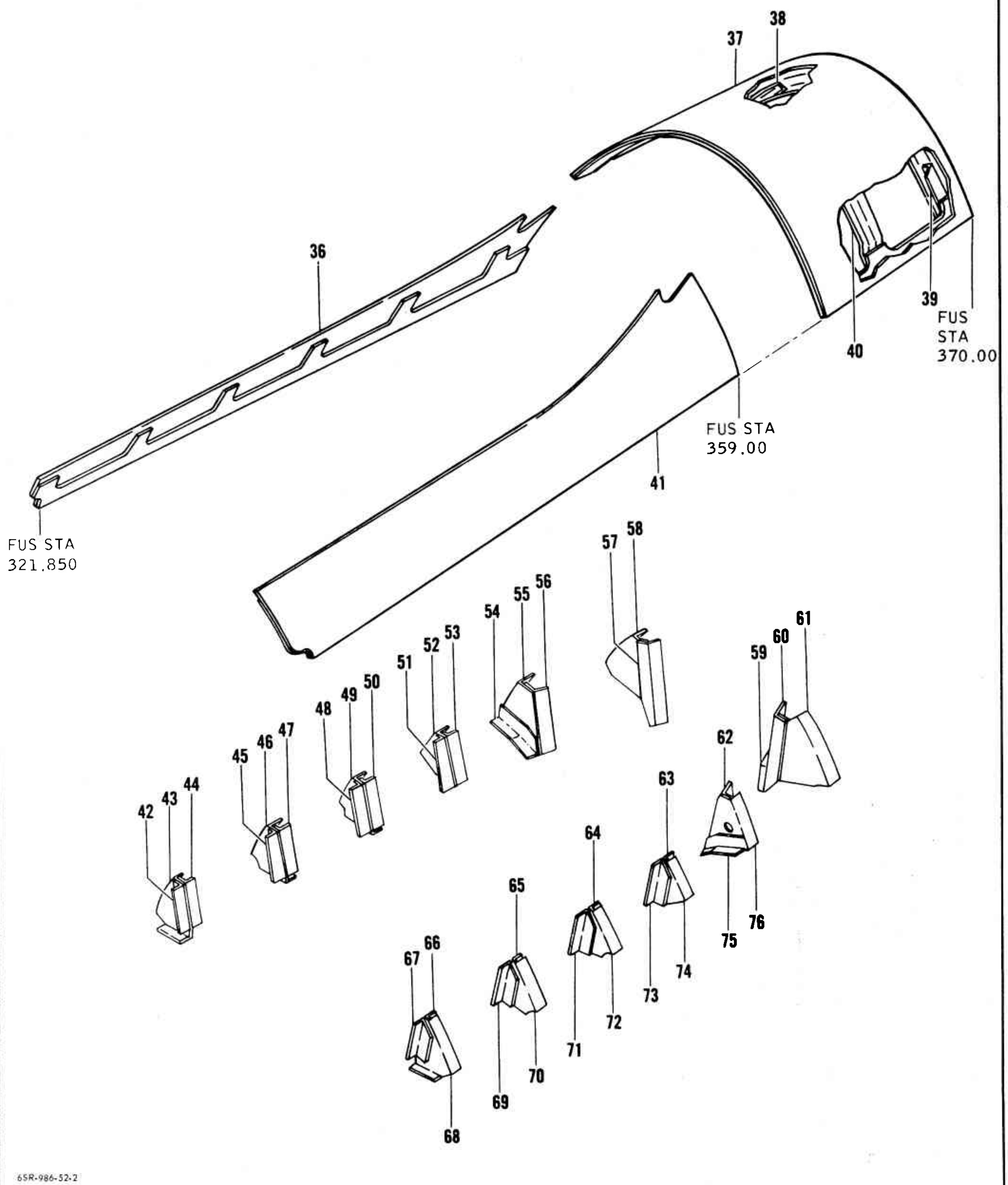
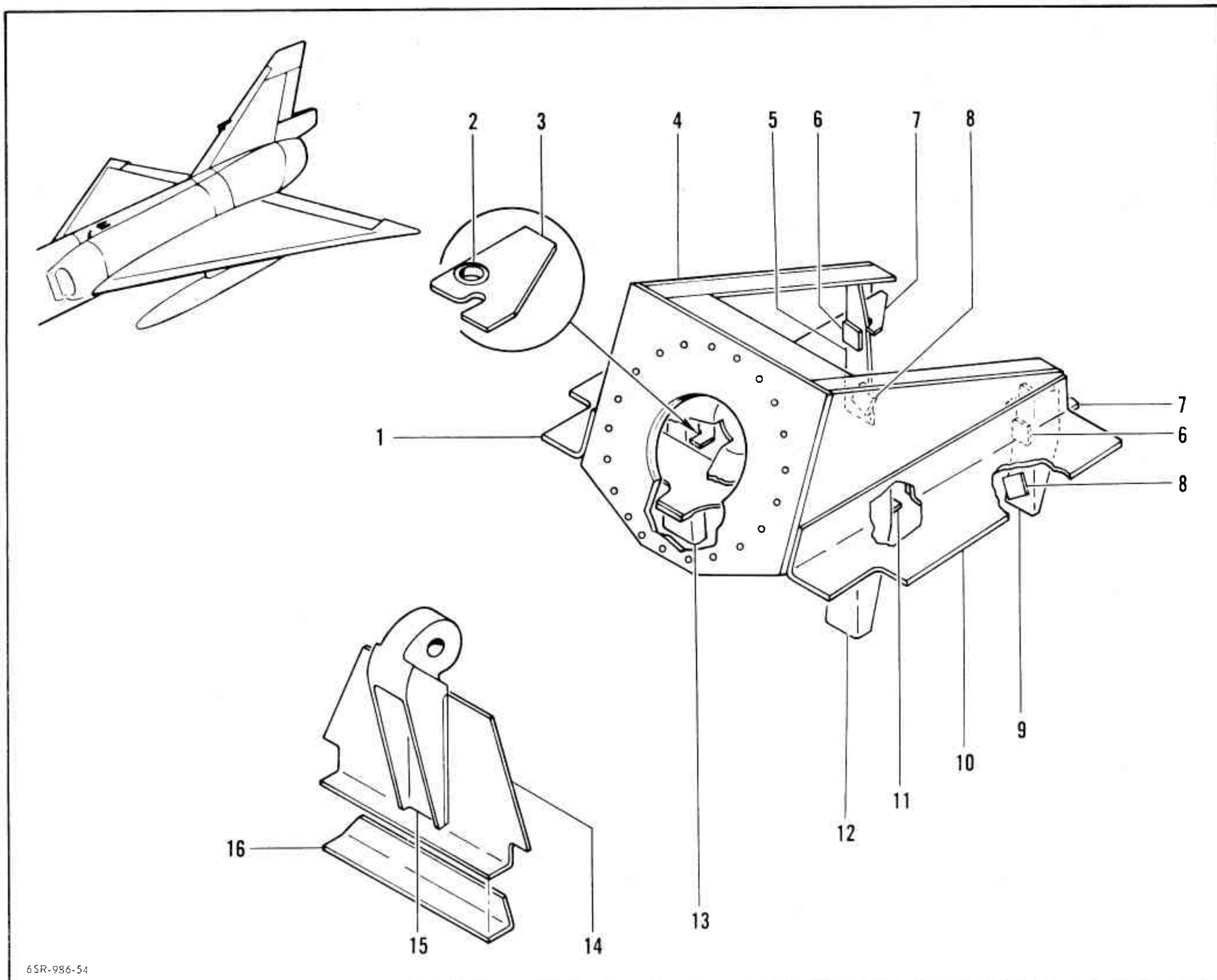


Figure 4-16C. Aerial Refueling Dorsal Sections, F-106A, Stations 320.0 to 370.7 (Sheet 3 of 4)
Applicable after incorporation of TCTO 1F-106-986

KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
49	66C39798-4	Intercostal	0.063 x 4.5 x 5.5	7075-T6	QQ-A-250/12(T6)	Replace
50	66J39840-8	Angle	0.063 x 1.6 x 2.74	2024-T42	QQ-A-250/4(T42)	Replace
51	66J39840-23	Angle	0.063 x 1.6 x 2.93	2024-T42	QQ-A-250/4(T42)	Replace
52	66C39799-2	Intercostal	0.063 x 3.3 x 4.3	7075-T6	QQ-A-250/12(T6)	Replace
53	66J39840-10	Angle	0.063 x 1.6 x 2.93	2024-T42	QQ- A-250/4(T42)	Replace
54	66J39840-26	Angle	0.163 x 1.5 x 3.5	2024-T42	QQ-A-250/4(T42)	Replace
55	66C39797-2	Support	0.050 x 4.4 x 5.8	7075-T6	QQ-A-250/12(T6)	Replace
56	66J39840-12	Angle	0.063 x 1.5 x 4.12	2024-T42	QQ-A-250/4(T42)	Replace
57	66J39840-14	Angle	0.063 x 1.5 x 6.5	2024-T42	QQ-A-250/4(T42)	Replace
58	66C39803-2	Support	0.05 x 5.8 x 5.8	7075-T6	QQ-A-250/12(T6)	10-10A, 10-10B
59	66J39840-2	Filler	0.160 x 0.8 x 1.0	7075-T6	QQ-A-250/12(T6)	Replace
60	66J39840-13	Angle	0.063 x 1.5 x 6.5	2024-T42	QQ-A-250/4(T42)	Replace
61	66C39803-1	Support	0.05 x 5.8 x 5.8	7075-T6	QQ-A-250/12(T6)	10-10A, 10-10B
62	66J39840-11	Angle	0.063 x 1.5 x 4.12	2024-T42	QQ-A-250/4(T42)	Replace
63	66J39840-9	Angle	0.063 x 1.6 x 2.93	2024-T42	QQ-A-250/4(T42)	Replace
64	66J39840-7	Angle	0.063 x 1.6 x 2.74	2024-T42	QQ-A-250/4(T42)	Replace
65	66J39840-5	Angle	0.063 x 1.6 x 2.64	2024-T42	QQ-A-250/4(T42)	Replace
66	66J39840-3	Angle	0.063 x 1.6 x 2.67	2024-T42	QQ-A-250/4(T42)	Replace
67	66J39840-16	Angle	0.063 x 1.6 x 2.67	2024-T42	QQ-A-250/4(T42)	Replace
68	66C39796-1	Intercostal	0.063 x 5.0 x 5.0	7075-T6	QQ-A-250/12(T6)	Replace
69	66J39840-18	Angle	0.063 x 1.6 x 2.64	2024-T42	QQ-A-250/4(T42)	Replace
70	66C39798-1	Intercostal	0.063 x 4.2 x 5.4	7075-T6	QQ-A-250/12(T6)	Replace
71	66J39840-20	Angle	0.063 x 1.6 x 2.74	2024-T42	QQ-A-250/4(T42)	Replace
72	66C39798-3	Intercostal	0.063 x 4.5 x 5.5	7075-T6	QQ-A-250/12(T6)	Replace
73	66J39840-22	Angle	0.063 x 1.6 x 2.93	2024-T42	QQ-A-250/4(T42)	Replace
74	66C39799-1	Intercostal	0.063 x 3.3 x 4.3	7075-T6	QQ-A-250/12(T6)	Replace
75	66J39840-25	Angle	0.163 x 1.5 x 3.5	2024-T42	QQ-A-250/4(T42)	Replace
76	66C39797-1	Support	0.050 x 4.4 x 5.8	7075-T6	QQ-A-250/12(T6)	Replace

Figure 4-16C. Aerial Refueling Dorsal Sections, F-106A, Stations 320.0 to 370.7 (Sheet 4 of 4)

Applicable after incorporation of TCTO 1F-106-986



KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
1	66J40250-9	Angle	0.16 x 4.4 x 11.2	7075-T6	QQ-A-250/12(T6)	Replace
2	MS35489-11	Grommet	1/16 x 7/32	Rubber	MIL-G-3036	Replace
3	66J40250-13	Plate	0.05 x 2.0 x 3.0	7075-T6	QQ-A-250/13(T6)	Replace
4	66J40971-1	Support (Weldment)		6061-T651	QQ-A-250/11(T651)	10-10E
5	66J40250-18	Support	0.09 x 2.5 x 4.95	7075-T6	QQ-A-250/12(T6)	Replace
6	66J40250-6	Filler	0.08 x 0.8 x 0.8	7075-T6	QQ-A-250/12(T6)	Replace
7	66J40250-12	Plate	0.05 x 2.0 x 2.5	7075-T6	QQ-A-250/13(T6)	Replace
8	66J40250-7	Filler	0.125 x 0.8 x 0.8	7075-T6	QQ-A-250/12(T6)	Replace
9	66J40250-17	Support	0.09 x 2.5 x 4.95	7075-T6	QQ-A-250/12(T6)	Replace
10	66J40250-8	Angle	0.16 x 4.4 x 11.2	7075-T6	QQ-A-250/12(T6)	Replace
11	66J40250-14	Plate	0.05 x 2.0 x 3.0	7075-T6	QQ-A-250/13(T6)	Replace
12	66J40250-2	Support	0.09 x 3.7 x 9.0	7075-T6	QQ-A-250/12(T6)	Replace
13	66J40250-3	Support	0.09 x 3.7 x 9.0	7075-T6	QQ-A-250/12(T6)	Replace
14	66J40250-16	Support	0.063 x 7.0 x 9.0	7075-T6	QQ-A-250/12(T6)	Replace
15	66D40251-1	Support	2.0 x 2.5 x 7.0	7075-T651	QQ-A-250/12(T651)	Replace
16	66J40250-10	Angle	0.063 x 1.8 x 6.6	7075-T6	QQ-A-250/12(T6)	Replace

Figure 4-16D. Aerial Refueling Receptacle Support, F-106A
Applicable after incorporation of TCTO 1F-106-986

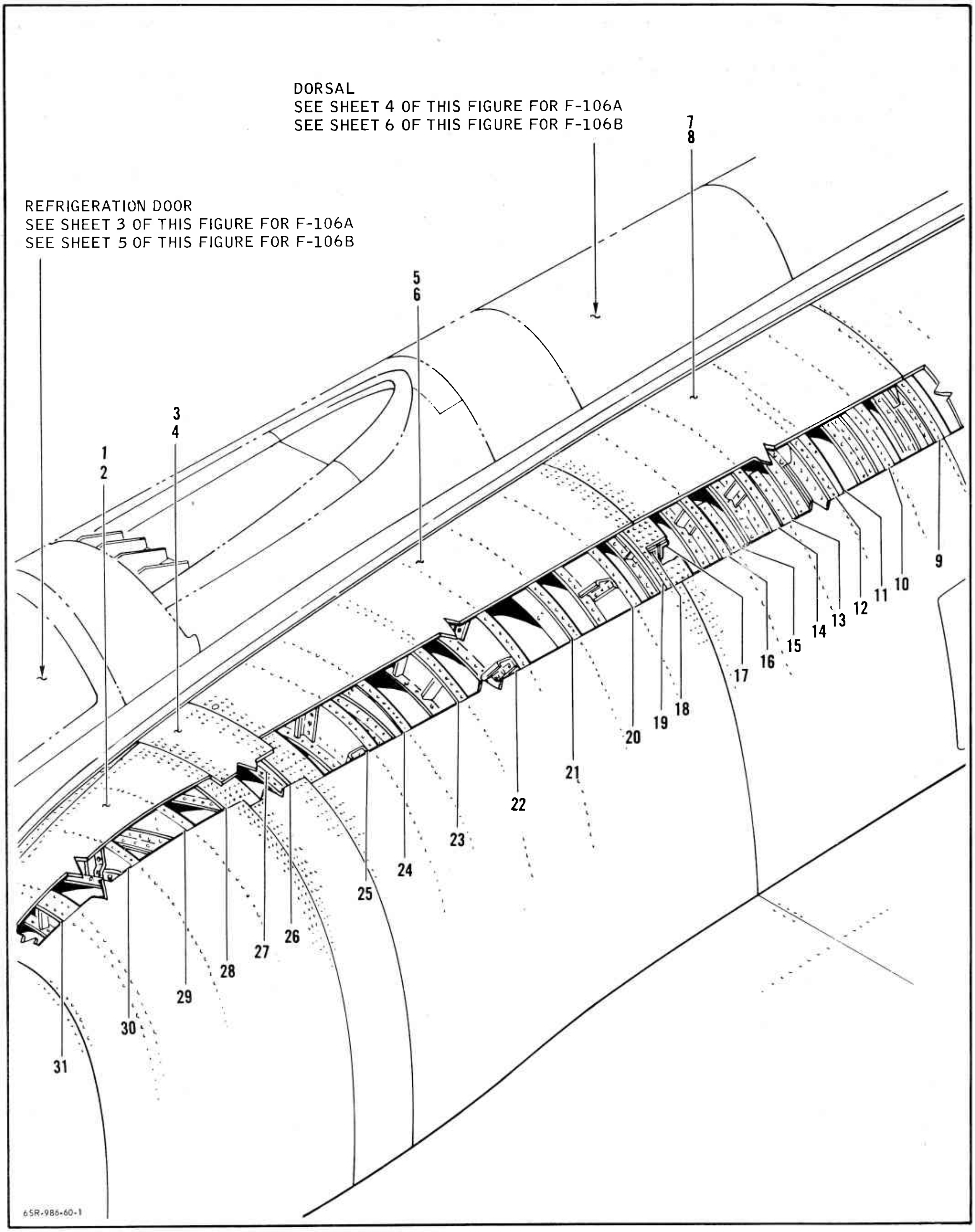


Figure 4-16E. Aerial Refueling Adjacent Area Structure F-106A and F-106B (Sheet 1 of 6)

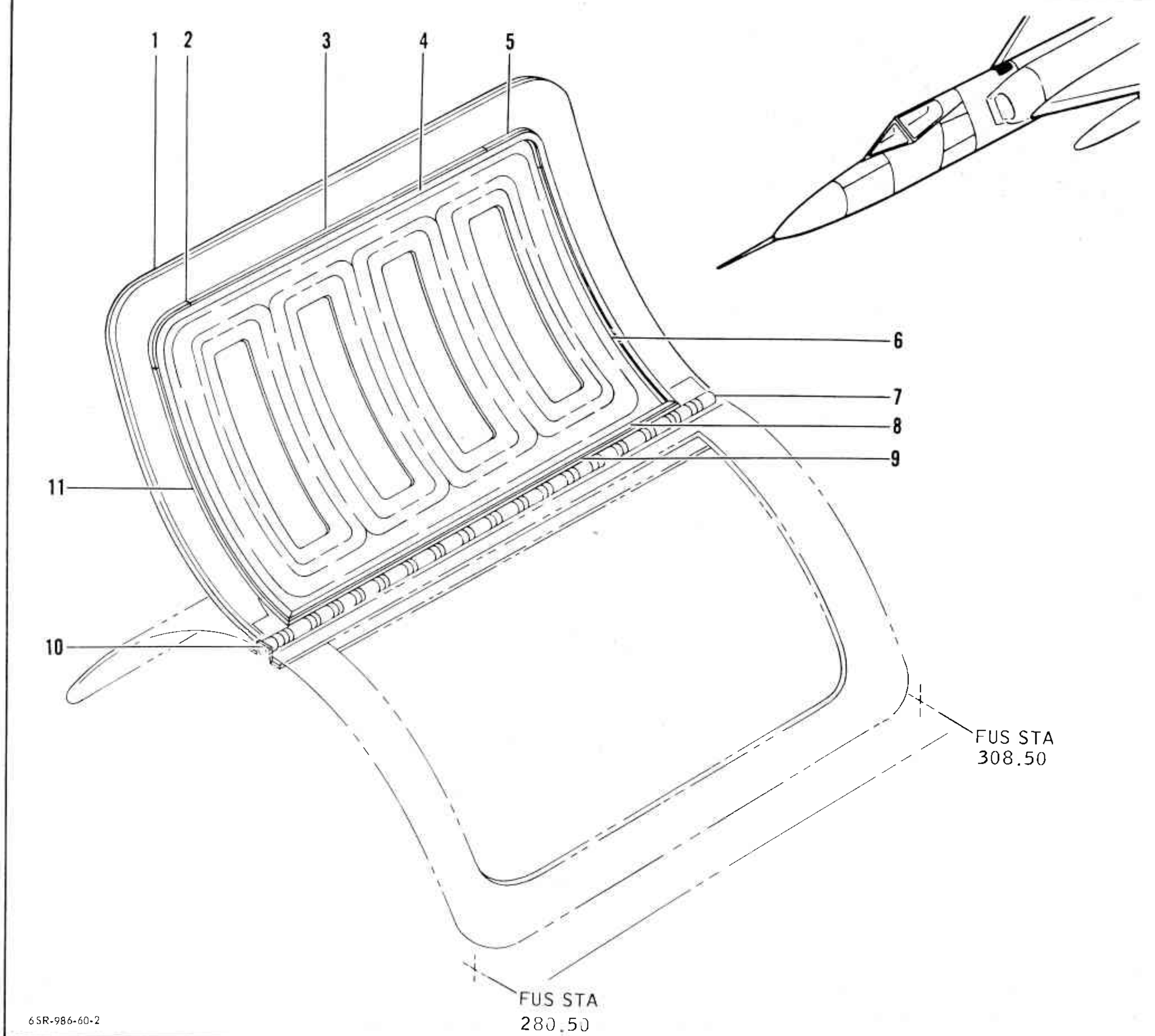
KEY NO.	*DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
1	8-74566-7(-17)	Skin, F-106A	0.04 x 28.5 x 71.5	7178-T6	MIL-A-9180(T6)	4-18B
2	8-76566-69(-71)	Skin, F-106B	0.04 x 28.5 x 71.5	7178-T6	MIL-A-9180(T6)	4-18B
3	8-74566-11(-12)	Skin, F-106A	0.04 x 8.0 x 66.0	7178-T6	MIL-A-9180(T6)	4-18B
4	8-76566-11(-12)	Skin, F-106B	0.04 x 8.0 x 66.0	7178-T6	MIL-A-9180(T6)	4-18B
5	8-74476-7(-8)	Skin, F-106A	0.04 x 49.0 x 66.0	7178-T6	MIL-A-9180(T6)	4-18B**
6	8-76476-7(-8)	Skin, F-106A	0.04 x 49.0 x 66.0	7178-T6	MIL-A-9180(T6)	4-18B**
7	8-74477-733(-734)	Skin, F-106B	0.05 x 48.1 x 65.0	7178-T6	MIL-A-9180(T6)	4-18B**
8	8-74477-739(-739)	Skin, F-106B	0.05 x 48.1 x 65.0	7178-T6	MIL-A-9180(T6)	4-18B**
9	8-74785-7(-8)	Frame Channel	0.045 x 30.3 x 40.1	7178-T6	MIL-A-9180(T6)	10-10A
10	8-74784-7(-8)	Frame Channel	0.045 x 31.8 x 39.4	7178-T6	MIL-A-9180(T6)	10-10A
11	8-74783-7(-8)	Frame	0.05 x 35.0 x 59.0	7178-T6	MIL-A-9180(T6)	10-10A
12	8-74783-111(-112)	Channel	0.05 x 17.5 x 52.9	7178-T6	MIL-A-9180(T6)	10-10A
13	8-74782-129(-130)	Frame Channel	0.05 x 15.2 x 50.0	7178-T6	MIL-A-9180(T6)	10-10A
14	8-74782-85(-86)	Channel	0.05 x 34.7 x 59.0	7178-T6	MIL-A-9180(T6)	10-10A
15	8-74781-11(-12)	Frame	0.05 x 34.6 x 58.1	7178-T6	MIL-A-9180(T6)	10-10A
16	8-74781-97(-98)	Channel	0.05 x 20.6 x 47.0	7178-T6	MIL-A-9180(T6)	10-10A
17	8-74780-9(-10)	Bulkhead Angle	0.08 x 3.0 x 62.0	7178-T6	MIL-A-9180(T6)	10-10A
18	8-74780-11(-12)	Bulkhead Angle	0.08 x 3.0 x 62.0	7178-T6	MIL-A-9180(T6)	10-10A
19	8-74780-13(-14)	Strap	0.188 x 4.5 x 71.2	7178-T6	MIL-A-9180(T6)	Table 4-1
20	8-74776-19(-20)	Frame	0.063 x 43.5 x 58.2	7178-T6	MIL-A-9180(T6)	10-10A
21	8-74775-35(-36)	Frame	0.063 x 44.2 x 46.8	7178-T6	MIL-A-9180(T6)	10-10A
22	8-74774-15(-16)	Frame	0.063 x 32.5 x 60.0	7178-T6	MIL-A-9180(T6)	10-10A
23	8-74773-7(-8)	Frame	0.063 x 29.1 x 57.3	7178-T6	MIL-A-9180(T6)	10-10A
24	8-74772-83(-84)	Frame	0.063 x 46.0 x 46.0	7178-T6	MIL-A-9198(T6)	10-10A
25	8-74771-7(-8)	Frame	0.064 x 45.5 x 50.0	7178-T6	MIL-A-9198(T6)	10-10A
26	8-74770-7(-8)	Doubler	0.05 x 5.2 x 63.0	7178-T6	MIL-A-9180(T6)	Table 4-1
27	8-74798-83(-84)	Bulkhead	1.75 x 50.0 x 60.0	7075-T6	0-01027	10-9
28	8-74580-7(-9)	Frame	0.05 x 32.0 x 51.0	7178-T6	MIL-A-9180(T6)	10-10A
29	8-74579-67(-68)	Frame	0.063 x 29.0 x 51.0	7178-T6	MIL-A-9180(T6)	10-10A
30	8-74578-7(-8)	Frame	0.063 x 29.0 x 50.0	7178-T6	MIL-A-9180(T6)	10-10A
31	8-74577-75(-76)	Frame	0.05 x 28.5 x 49.0	7178-T6	MIL-A-9180(T6)	10-10A

*Dash numbers following drawing numbers are for left inlet duct.
Dash number in parenthesis is for right inlet duct.

**This repair is limited to the upper 25 inches of this skin with the following exceptions:

(1) These repairs do not apply where CRES rivets (XA4) Q4304 (97-67001-XXX) are used.
(2) No cutout hole permitted in repair doubler.
(3) Use 2-1/2 rows of NAS 1399MW4 rivets.
(4) Two inner rows of rivets to be at 0.50 inch pitch; outer row at 1.00 inch pitch.

Figure 4-16E. Aerial Refueling Adjacent Area Structure F-106A and F-106B (Sheet 2 of 6)

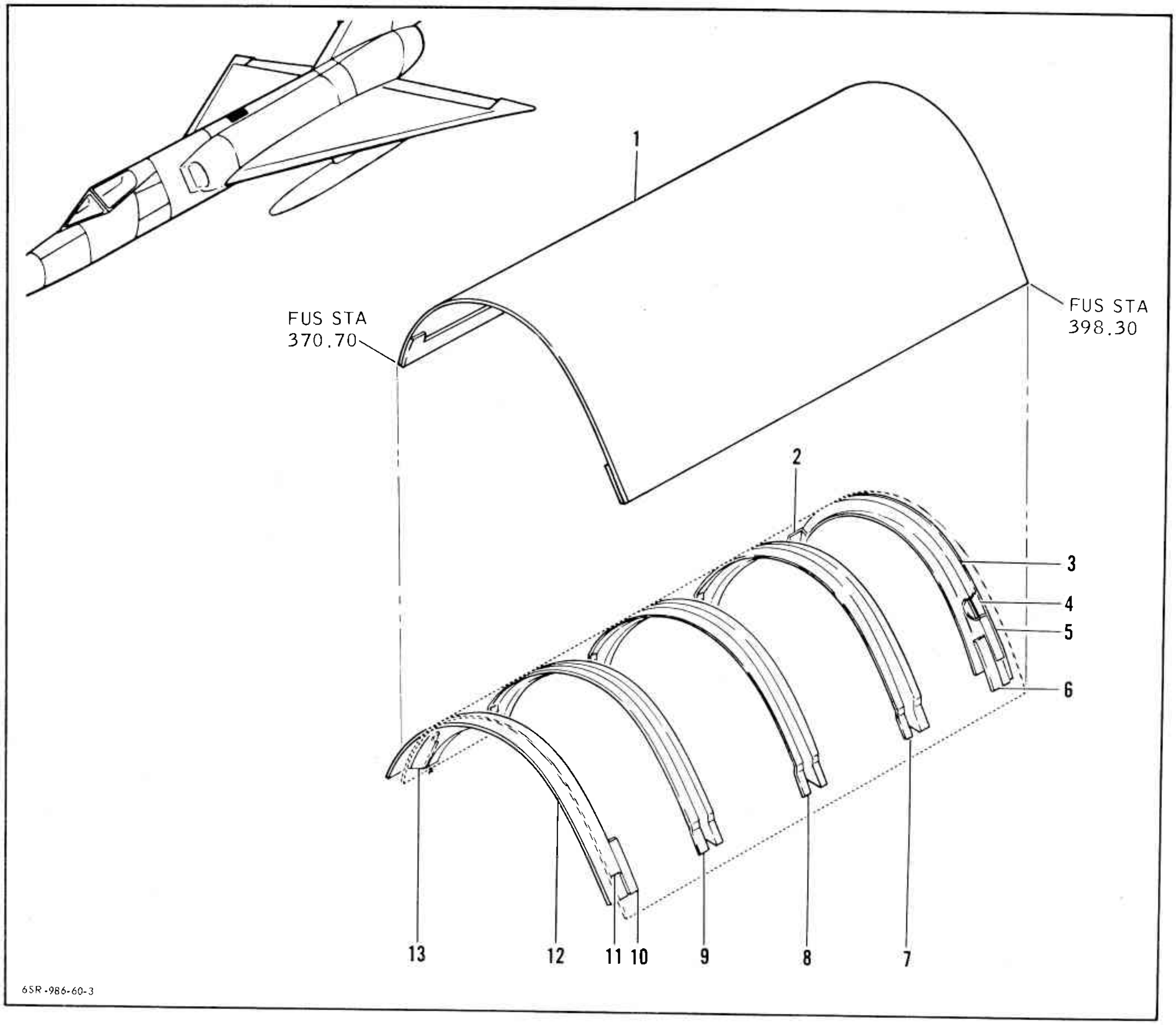


KEY NO.	*DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
1	8-74946-19(-20)	Door (Weldment)	0.063 x 18.9 x 27.8	7178-T6	MIL-A-9180	4-16F
2	8-74946-33(-34)	Retainer	E571001 x 6.1	6061-T6	QQ-A-270(T6)	Replace
3	8-74946-35(-36)	Retainer	E571001 x 16.9	6061-T6	QQ-A-270(T6)	Replace
4	8-74946-15(-15)	Seal	MFQ2531 x 53.8	Rubber	Rub. Extr. 351975	Replace
5	8-74946-39(-40)	Retainer	E571001 x 6.1	6061-T6	QQ-A-270(T6)	Replace
6	8-74946-37(-38)	Retainer	E571001 x 11.7	6061-T6	QQ-A-270(T6)	Replace
7	8-77726-7(-7)	Hinge	E491134 x 27.6	7178-T6	MIL-A-9186	**
8	8-74946-17(-17)	Seal	MFQ2531 x 23.9	Rubber	Rub. Extr. 351975	Replace
9	8-74946-29(-30)	Retainer	E571001 x 24.1	6061-T6	QQ-A-270(T6)	Replace
10	8-74946-23(-23)	Pin	0.120 x 27.6	Cres Typ 302	QQ-W-423 Cond B	Replace
11	8-74946-31(-32)	Retainer	E571001 x 12.7	6061-T6	QQ-A-270(T6)	Replace

*Dash numbers following drawing number is for left inlet duct. Dash numbers in parenthesis is for right inlet duct.

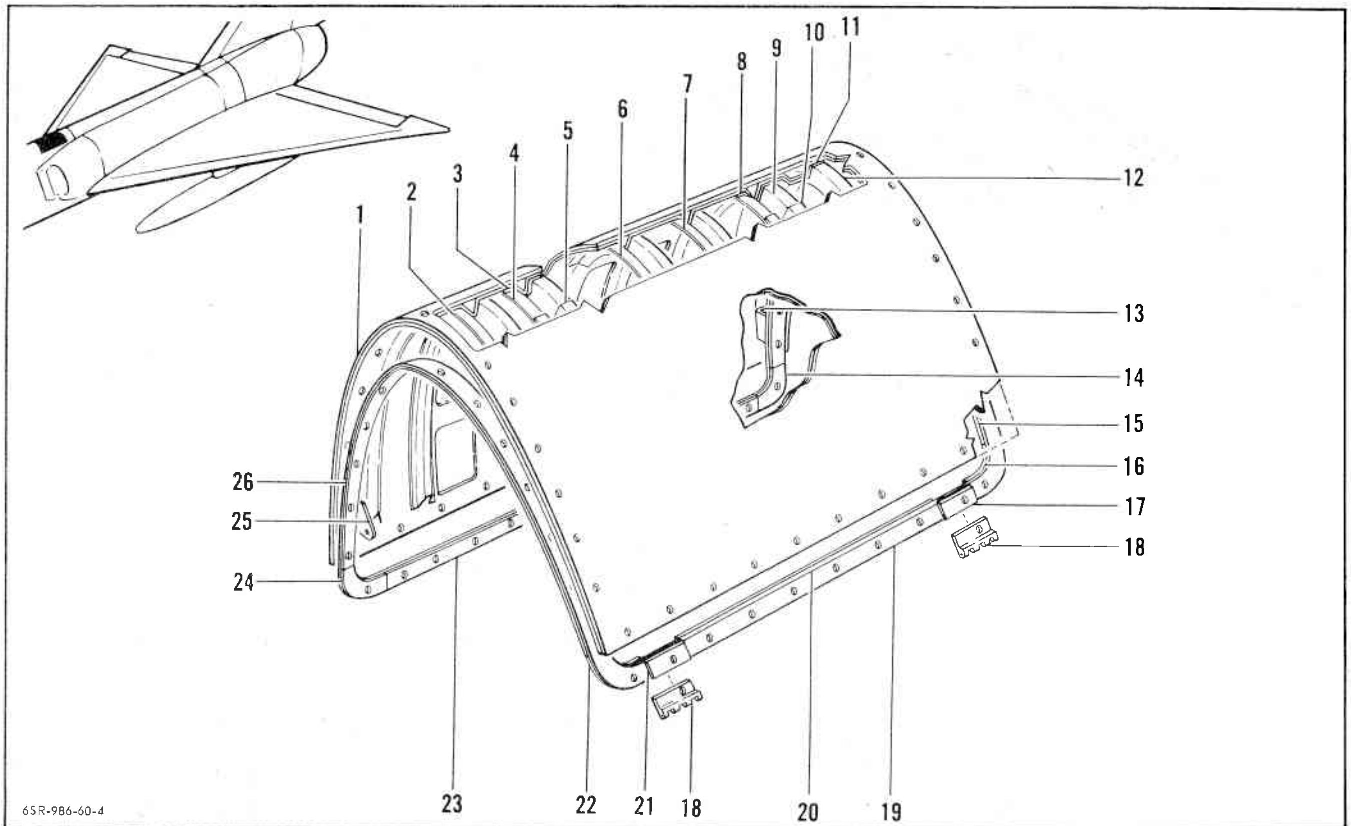
**Cut out damaged section and replace; no splice required.

Figure 4-16E. Aerial Refueling Adjacent Area Structure F-106A and F-106B (Sheet 3 of 6)



KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
1	8-74498-141	Skin (Weldment)	.040 x 25.00 x 19.00	AZ31A	QQ-M-44	10-10C, 10-10E
2	8-74498-152	Clip	0.05 x 1.20 x 2.70	7178-T6	MIL-A-9180(T6)	Replace
3	8-74498-49	Zee	0.04 x 2.40 x 21.20	7178-T6	MIL-A-9180(T6)	10-10C
4	8-74498-69	Filler	0.025 x .50 x 21.10	7178-T6	MIL-A-9180(T6)	*
5	8-74498-79	Retainer	0.02 x 1.10 x 20.90	Cres Typ 301	MIL-S-5059(1/4 Hd)	*
6	8-74498-151	Clip	0.05 x 1.20 x 2.70	7178-T6	MIL-A-9180(T6)	Replace
7	8-74498-109	Stiffener	0.036 x 2.5 x 21.7	7178-T6	MIL-A-9180(T6)	10-10A, 10-10C
8	8-74498-107	Stiffener	0.036 x 2.5 x 21.8	7178-T6	MIL-A-9180(T6)	10-10A, 10-10C
9	8-74498-105	Stiffener	0.036 x 2.5 x 22.10	7178-T6	MIL-A-9180(T6)	10-10A, 10-10C
10	8-74498-135	Clip	0.05 x 1.2 x 2.7	7178-T6	MIL-A-9180(T6)	Replace
11	8-74498-47	Zee	0.04 x 2.00 x 22.50	7178-T6	MIL-A-9180(T6)	10-10C
12	8-74498-89	Plate	0.025 x 1.00 x 22.50	Cres Typ 301	MIL-S-5059(1/4 Hd)	*
13	8-74498-136	Clip	0.05 x 1.2 x 2.7	7178-T6	MIL-A-9180(T6)	Replace
	*Cut out damaged section and replace; no splice required.					

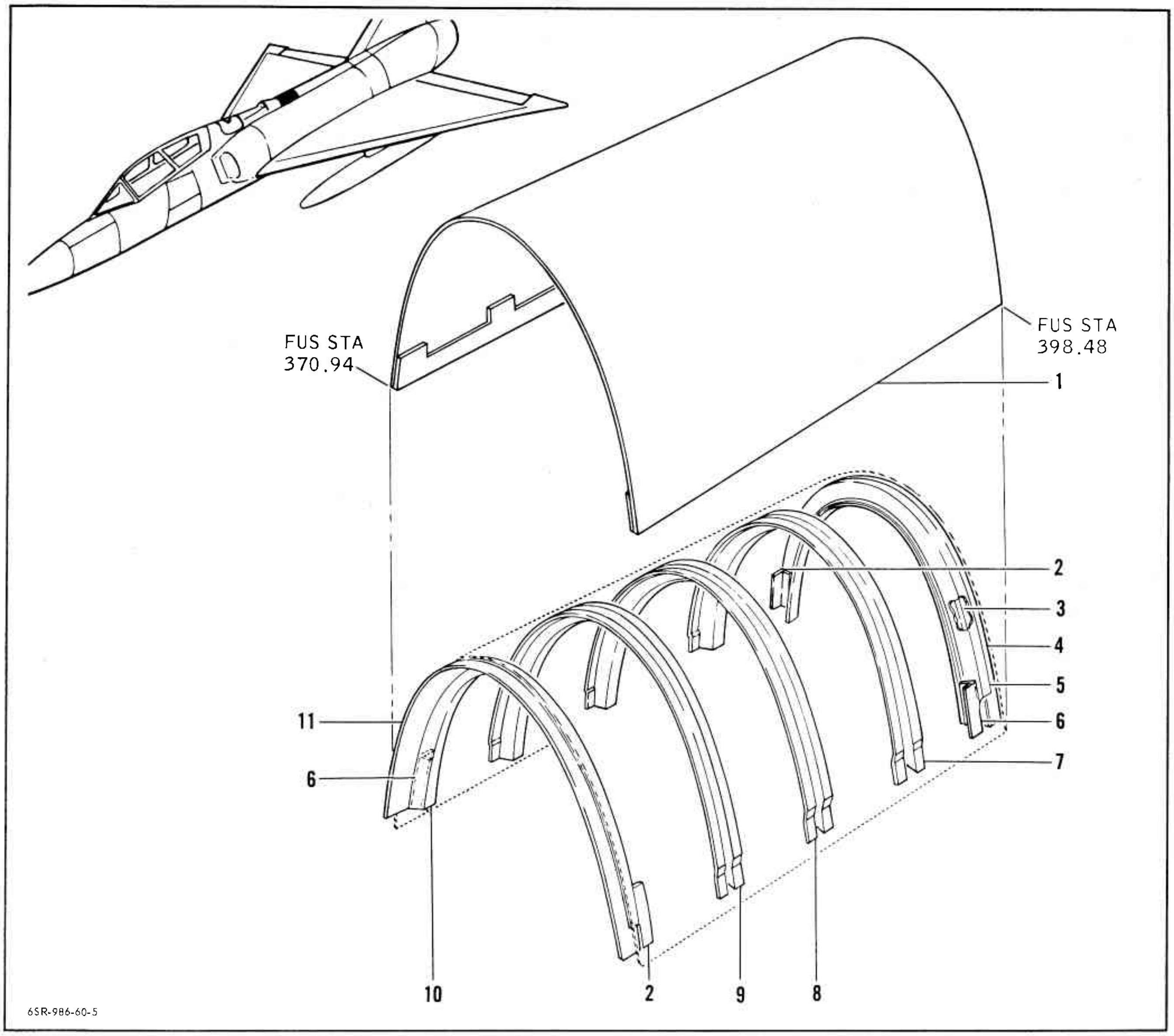
Figure 4-16E. Aerial Refueling Adjacent Area Structure F-106A and F-106B (Sheet 4 of 6)



KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
1	8-76942-87	Skin (Weldment)	0.063 x 18.9 x 27.8	7178-T6	MIL-A-9180(T6)	4-16F
2	8-76942-65	Splice	0.04 x 3.8 x 8.0	7178-T6	MIL-A-9180(T6)	Replace
3	8-76942-58	Frame	0.04 x 5.0 x 34.6	7178-T6	MIL-A-9180(T6)	10-10A, 10-10C
4	8-76942-61	Splice	0.04 x 5.0 x 8.0	7178-T6	MIL-A-9180(T6)	Replace
5	8-76942-57	Frame	0.04 x 5.0 x 34.6	7178-T6	MIL-A-9180(T6)	10-10A, 10-10C
6	8-76942-11	Hat	0.04 x 3.8 x 8.0	7178-T6	MIL-A-9180(T6)	10-10C
7	8-76942-67	Splice	0.04 x 3.8 x 8.0	7178-T6	MIL-A-9180(T6)	Replace
8	8-76942-60	Frame	0.04 x 5.0 x 34.6	7178-T6	MIL-A-9180(T6)	10-10A, 10-10C
9	8-76942-63	Splice	0.04 x 5.0 x 8.0	7178-T6	MIL-A-9180(T6)	Replace
10	8-76942-59	Frame	0.04 x 5.0 x 34.6	7178-T6	MIL-A-9180(T6)	10-10A, 10-10C
11	8-76942-13	Hat	0.04 x 3.8 x 8.0	7178-T6	MIL-A-9180(T6)	10-10C
12	8-76958-1	Stiffener	0.05 x 18.4 x 25.9	7075-T6	QQ-A-283(T6)	10-5
13	8-76942-15	Clip	0.063 x 1.3 x 1.8	7075-T6	QQ-A-283(T6)	Replace
14	8-76942-81	Retainer	0.09 x 3.3 x 3.4	2024-T6	QQ-A-362(T3)	Replace
15	8-76942-69	Retainer	0.09 x 0.8 x 40.5	2024-T3	QQ-A-362(T3)	*
16	8-76942-83	Retainer	0.09 x 2.9 x 3.0	2024-T3	QQ-A-362(T3)	Replace
17	8-76942-39	Filler	0.063 x 0.8 x 3.0	2024-T6	QQ-A-362(T3)	Replace
18	8-76915-7	Hinge Half	0.25 x 1.5 x 3.0	7075-T6	QQ-A-282(T6)	Replace
19	8-76942-93	Seal	Q2756 x 84.0	SUITABLE SUB-BAC1521-144		Replace
20	8-76942-71	Retainer	0.09 x 0.8 x 23.8	2024-T3	QQ-A-362(T3)	*
21	8-76942-37	Filler	0.063 x 0.8 x 3.0	2024-T6	QQ-A-362(T3)	Replace
22	8-76942-73	Retainer	0.09 x 2.6 x 3.3	2024-T3	QQ-A-362(T3)	Replace
23	8-76942-79	Retainer	0.09 x 0.8 x 30.4	2024-T6	QQ-A-362(T3)	*
24	8-76942-77	Retainer	0.09 x 3.3 x 3.4	2024-T3	QQ-A-362(T3)	Replace
25	8-76942-41	Bracket	0.063 x 2.0 x 2.4	2024-T6	QQ-A-362(T3)	Replace
26	8-76942-75	Retainer	0.09 x 0.8 x 47.0	2024-T3	QQ-A-362(T3)	*

*Cut out damaged section and replace; no splice required.

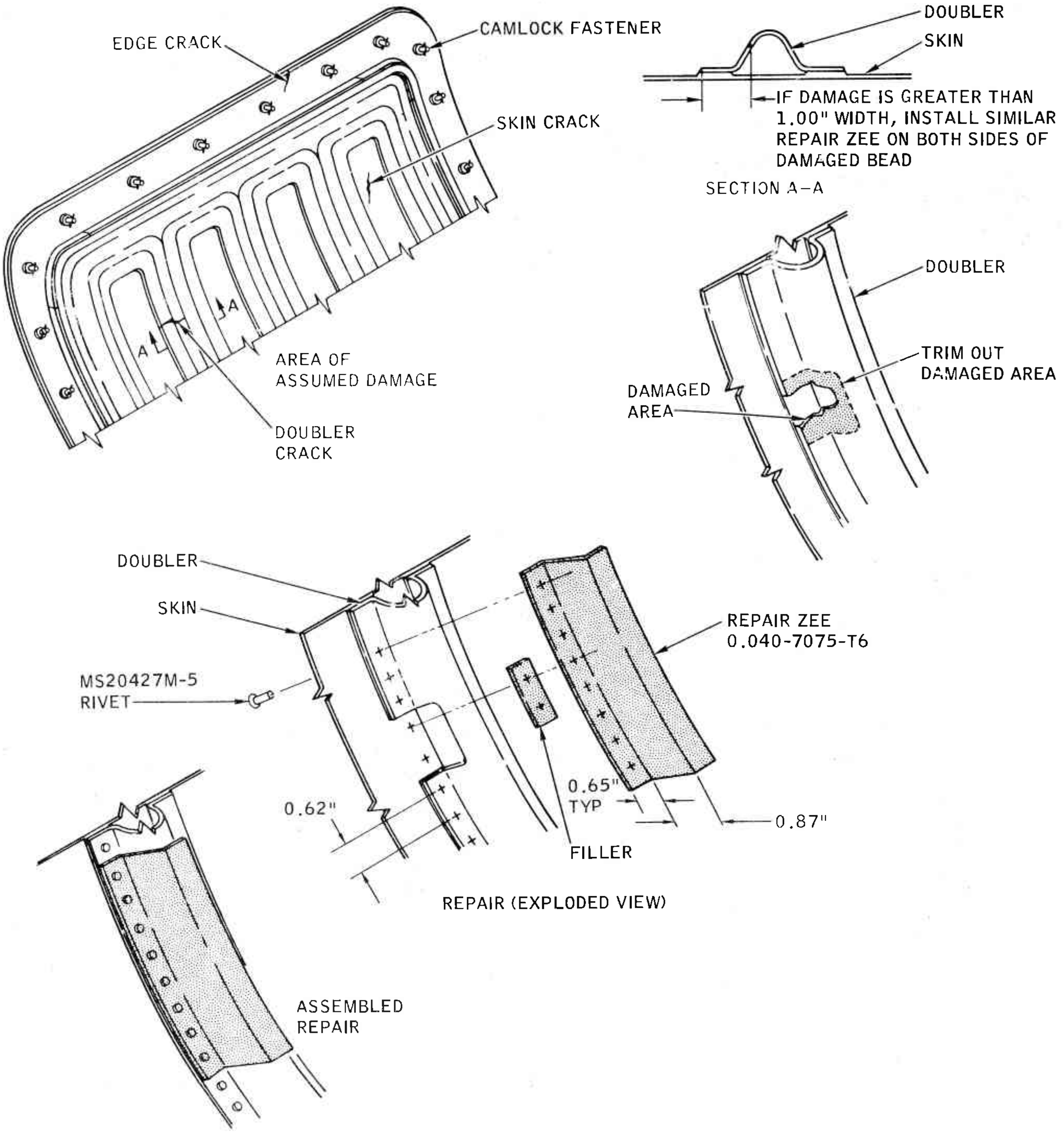
Figure 4-16E. Aerial Refueling Adjacent Area Structure F-106A and F-106B (Sheet 5 of 6)



KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
1	8-76498-15	Panel Assembly	0.040 x 27.7 x 37.0	AZ31B-H24	QQ-M-44(H24)	10-10C, 10-10E
		(Weldment)				
2	8-76498-39	Clip	0.05 x 1.2 x 2.7	7178-T6	MIL-A-9180	Replace
3	8-76498-41	Filler	0.025 x 50 x 29.0	7178-T6	MIL-A-9180	*
4	8-76498-97	Retainer	0.025 x 1.10 x 29.0	Cres Typ 301	MIL-S-5059 (1/4 HD)	*
5	8-76498-57	Stiffener	0.040 x 2.2 x 29.0	7178-T6	MIL-A-9180	10-7, 10-10A
6	8-76498-40	Clip	0.05 x 1.2 x 2.7	7178-T6	MIL-A-9180	Replace
7	8-76498-69	Stiffener	0.040 x 2.3 x 30.1	7178-T6	MIL-A-9180	10-10A, 10-10C
8	8-76498-67	Stiffener	0.040 x 2.3 x 31.1	7178-T6	MIL-A-9180	10-10A, 10-10C
9	8-76498-65	Stiffener	0.040 x 2.3 x 30.0	7178-T6	MIL-A-9180	10-10A, 10-10C
10	8-76498-49	Stiffener	0.040 x 2.0 x 34.4	7178-T6	MIL-A-9180	10-7, 10-10A
11	8-76498-89	Plate	0.025 x 1.0 x 34.0	Cres Typ 301	MIL-S-5059(1/4 HD)	*

*Cutout damaged section and replace; no splice required.

Figure 4-16E. Aerial Refueling Adjacent Area Structure F-106A and F-106B (Sheet 6 of 6)



REPAIR PROCEDURE

a. Repair skin per flush patch skin repair or flat plate edge repair in Section X. Patch doubler may be installed on external side of skin where installation difficulty is encountered. Chamfer edges of external patch doubler 45 degrees. Install flush plate where needed as a filler.

b. Edge cuts or cracks of less than 0.75 inch length and not closer than 0.50 inch from Camloc fasteners, may be stop drilled and edges smoothed.

c. Repair doubler bead by trimming out damaged area.

d. Fabricate zee repair part from 0.040 - 7075 stock. Dimension and heat treat as shown.

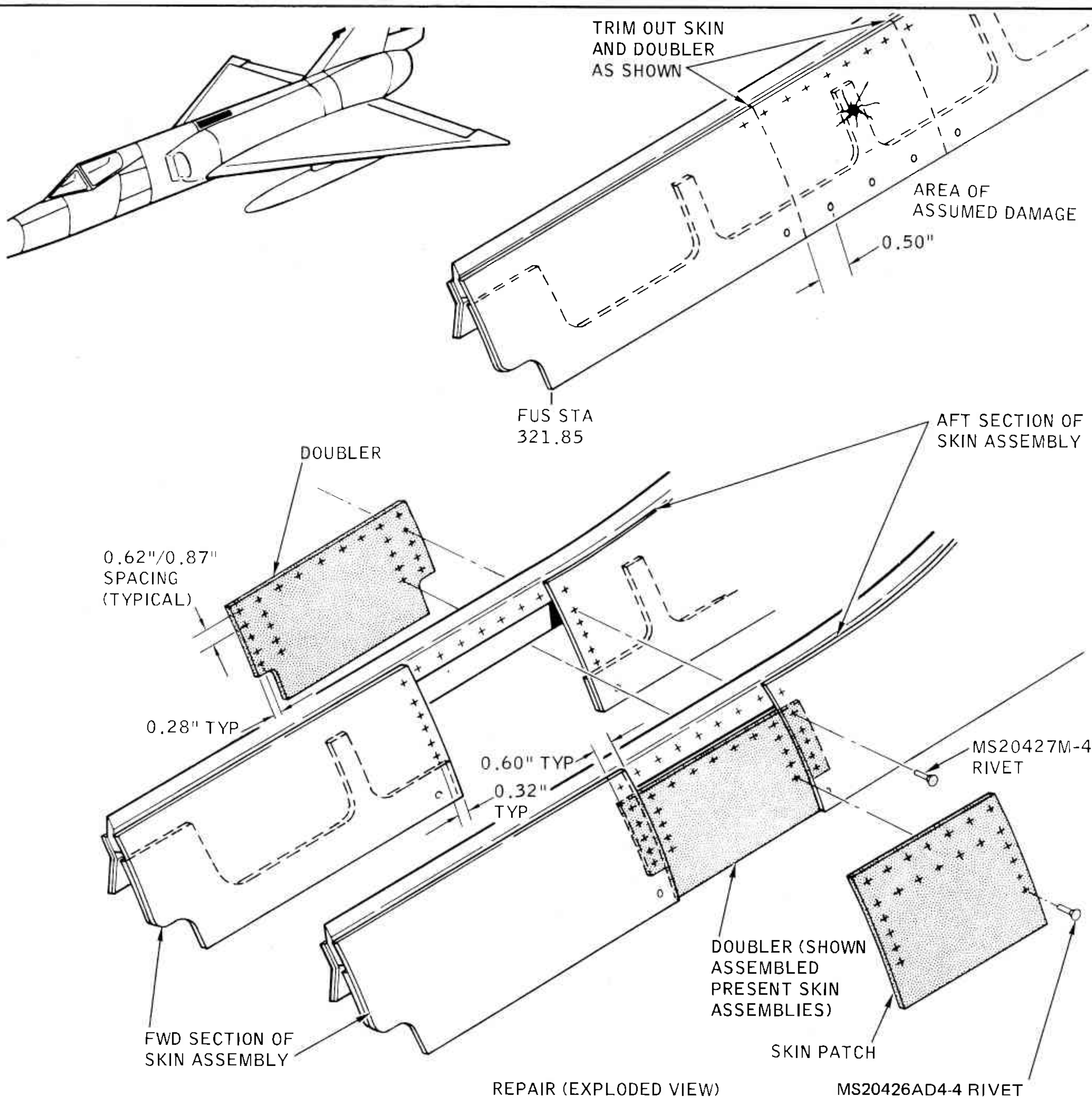
e. Refer to Section I for finish requirements of repair parts.

f. Drill rivet pattern as shown; countersink holes.

g. Install repair.

6SR-986-68

Figure 4-16F. Refrigeration Compartment Access Door Repair, Typical



REPAIR PROCEDURE

a. Trim out skin and doubler in area of damage making the cuts so that the cut edge falls halfway between two rivets on upper edge, and halfway between screw holes on lower edge or one half inch from screw hole whichever is less. Cutout must include one screw below and one screw hole on each side of doubler vertical leg.

NOTE

IF DAMAGE OCCURS OUTSIDE DOUBLER AREA, USE FIGURE 10-1 REPAIR EXCEPT USE ONLY.A SINGLE ROW OF MS20427M-4 RIVETS TO ATTACH DOUBLER. ATTACH PATCH TO DOUBLER USING MS20426AD4-4 RIVETS.

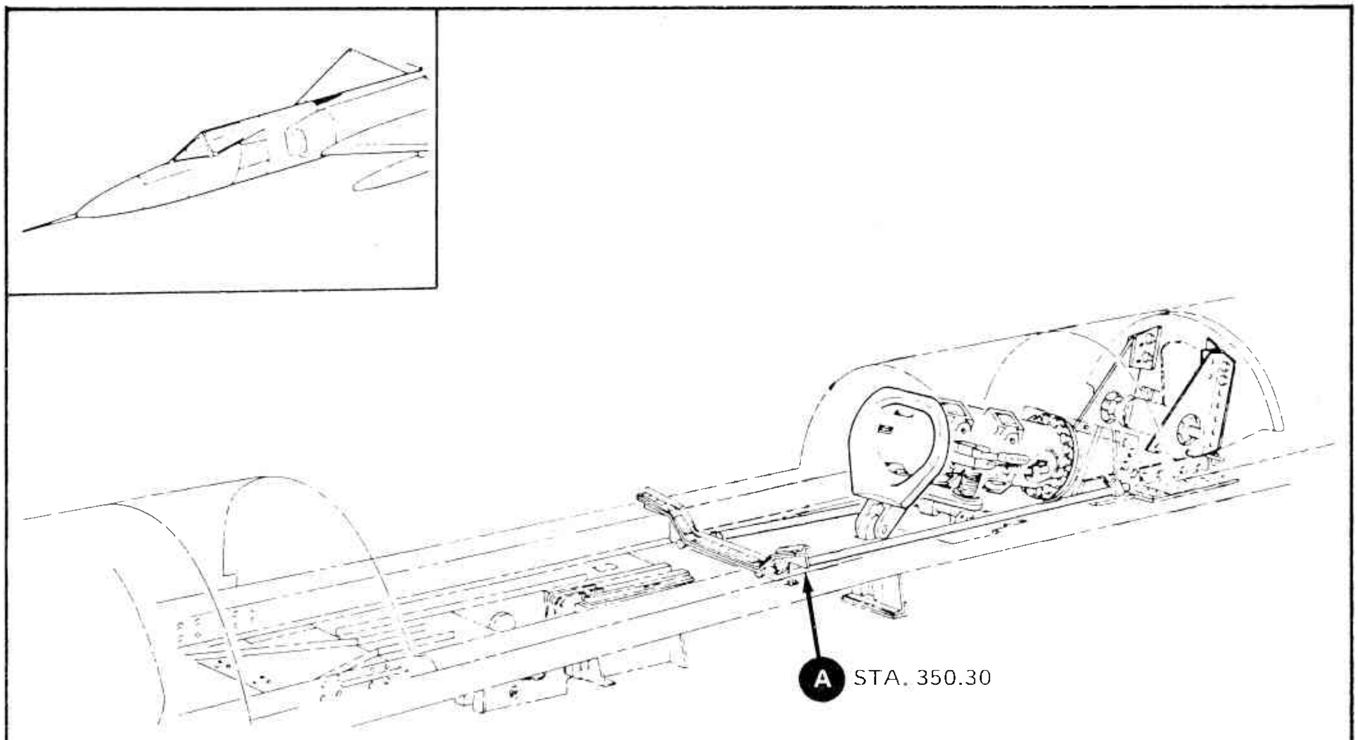
b. Fabricate repairs of same material and gage as damaged area.

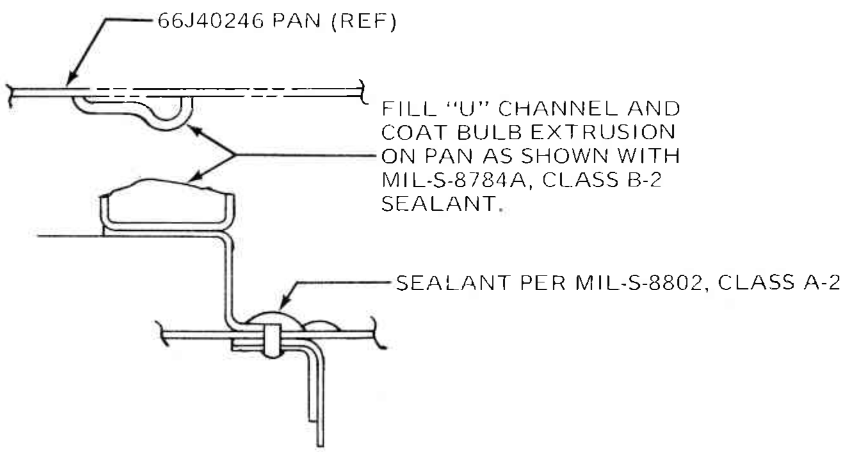
c. Refer to Section I for finish requirements of repair surfaces.

6SR-986-70

Figure 4-16G. Dorsal Skin and Doubler Repair, F-106A, Station 321.0 to 359.0

Applicable after incorporation of TCTO 1F-106-986





Detail **A**

NOTE

SHOULD ADDITIONAL INFORMATION BE REQUIRED - REFER TO AIR FORCE DRAWING 67J36505.

TYP APPLICATION OF SEALANT FWD DAM. CLEAN PAN AND CHANNEL WITH NAPTHA, DRY. ASSEMBLE IMMEDIATELY AND ALLOW TO CURE. HEAT, 100° TO 130°F FOR 2 HOURS MAY BE USED TO SPEED CURE.

NOTE: THIS INSTALLATION TO BE ACCOMPLISHED ONLY AFTER FINAL PREFIT OF DORSAL INSTALLATION.

Figure 4-16H. Dorsal Area Repair - Station 350.30

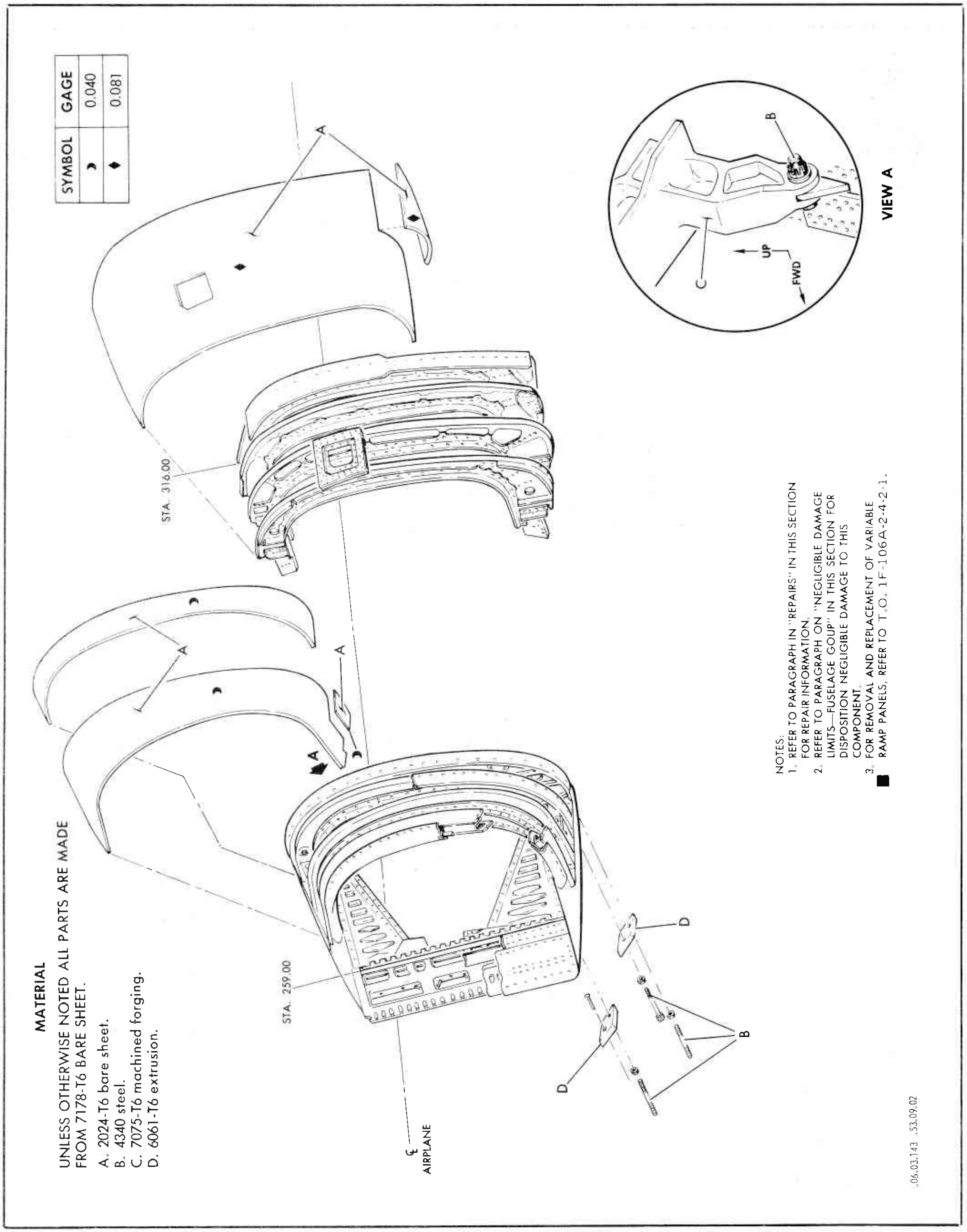


Figure 4-17. Engine Air Duct — Forward Inlet to Station 316.00
Applicable to F-106A airplane 56-455 and F-106B airplane 57-2507

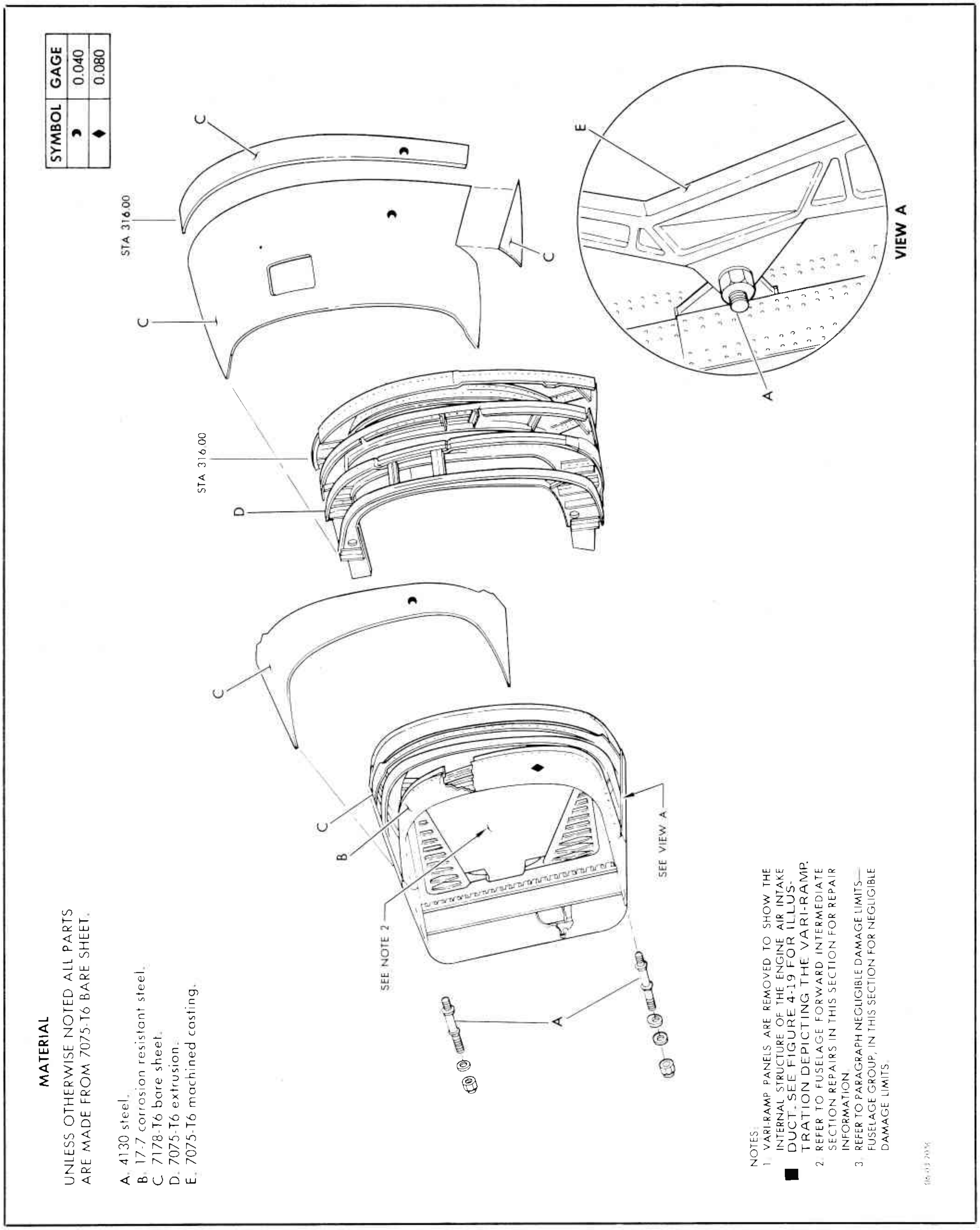
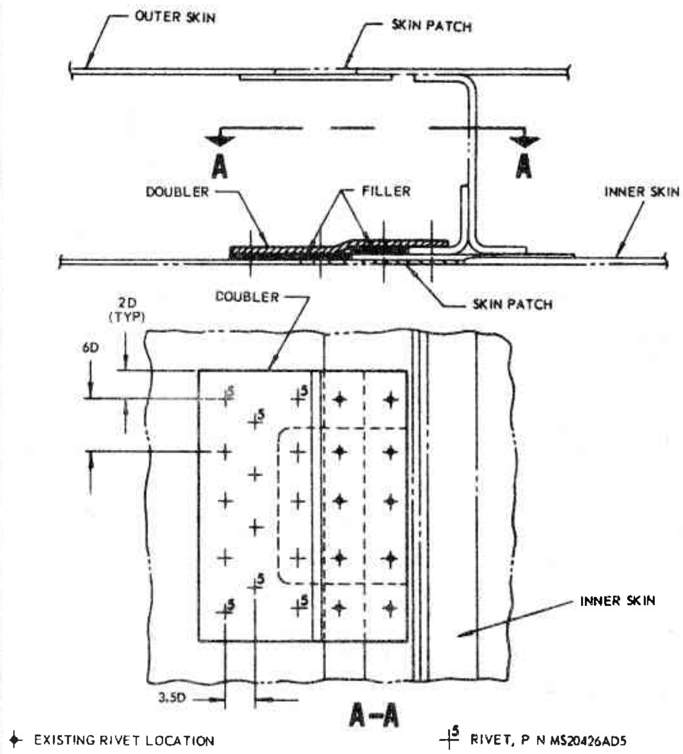


Figure 4-18. Engine Air Duct — Forward Inlet to Station 316.00
Applicable to F-106A airplanes 56-453, -454, 56-456 and subsequent, and F-106B airplanes 57-2508 and subsequent



DUCT INNER SKIN REPAIR PROCEDURES

A. ROUT OUT THE DAMAGED AREA TO A SMOOTH CONTOUR. CARE MUST BE EXERCISED NOT TO DAMAGE THE UNDERLYING STRUCTURE. WHENEVER A DAMAGED AREA IS TRIMMED AND IT IS NOT MORE THAN 4.5 INCHES FROM ANOTHER DAMAGED AREA (OLD OR NEW) THE TWO AREAS SHOULD BE CONSOLIDATED INTO ONE LARGE PATCH. IF A SERIES OF PATCHES EXIST AT A BUTT JOINT, CONSOLIDATE INTO ONE LARGE PATCH.

B. CUT ACCESS HOLES AS SHOWN IN FIGURES 4–18A AND 4–18B.

C. FABRICATE THE INNER SKIN PATCH, FILLERS, AND DOUBLERS FROM THE SAME MATERIAL AS THE INNER SKIN. THE SKIN PATCH SHOULD BE THE SAME THICKNESS AS THE EXISTING SKIN. THE FILLERS SHOULD BE THE SAME THICKNESS AS THE STIFFENER FLANGE. THE DOUBLERS SHOULD BE ONE SIZE THICKER THAN THE SKIN PATCHES.

D. REMOVE ENOUGH RIVETS FROM THE INNER SURFACE OF THE DUCT, IF REQUIRED, TO ALLOW ATTACHMENT OF THE REPAIR ASSEMBLY. THESE RIVETS ARE TO BE REPLACED WITH RIVETS OF THE SAME SIZE AND TYPE.

E. DRILL HOLES THROUGH THE INNER SKIN PATCH AND DOUBLERS USING EXISTING HOLES IN DUCT SKIN AS A GUIDE.

F. REMOVE THE INNER SKIN PATCH AND DOUBLERS. CLEAN ALL BURRS AND CHIPS FROM THEM AND THE DUCT STRUCTURE.

G. INSERT THE INNER SKIN PATCH, FILLERS, AND DOUBLERS AND INSTALL CLECOS FOR THE RIVETS REMOVED IN STEP E.

H. DRILL THE REMAINING HOLES THROUGH THE INNER SKIN PATCH, FILLERS, AND DOUBLERS AS INDICATED. USE A NO. 21 (0.159-INCH) DRILL. MAINTAIN AN EDGE DISTANCE OF 2D AND A RIVET SPACING OF 6D.

DUCT INNER SKIN REPAIR PROCEDURES

I. REMOVE THE REPAIR ASSEMBLY AND CLEAN ALL BURRS AND CHIPS FROM IT AND THE DUCT STRUCTURE.

J. APPLY A COAT OF SEALER, SPECIFICATION MIL-S-81733, TO THE FAYING SURFACES OF THE REPAIR ASSEMBLY AND THE DUCT SKIN AND STRUCTURE.

K. INSERT THE REPAIR ASSEMBLY AND INSTALL THE RIVETS. APPLY A COAT OF SEALER, SPECIFICATION MIL-S-81733, TO EACH RIVET AS IT IS INSTALLED.

L. FILL THE GAP BETWEEN THE SKIN PATCH AND THE INNER DUCT SKIN WITH SEALER, SPECIFICATION MIL S 81733 OR MIL S 8802

M. APPLY PAINT FINISH TO REPAIR AREA AS INDICATED IN T.O. 1F-106A-23.

N. INSTALL OUTER SKIN PATCHES AND DOUBLERS. USE PROCEDURES OUTLINED ON FIGURE 4–18B.

O. TOLERANCES: RIVET SPACING: ± 0.1
RIVET EDGE DISTANCE: 2D ± .03 MINIMUM
X – ± 0.1
XX – ± 0.03
XXX – ± 0.010

NOTE

INTAKE DUCT INNER SKINS, WHICH HAVE AREAS OF MINOR CORROSION, SHOULD BE CLEANED AND TREATED IN ACCORDANCE WITH T.O. 1-1-2. AREAS WITH DEEP CORROSION SHOULD BE REPAIRED IN THE SAME MANNER AS SHOWN ON FIGURES 4–18A AND 4–18B.

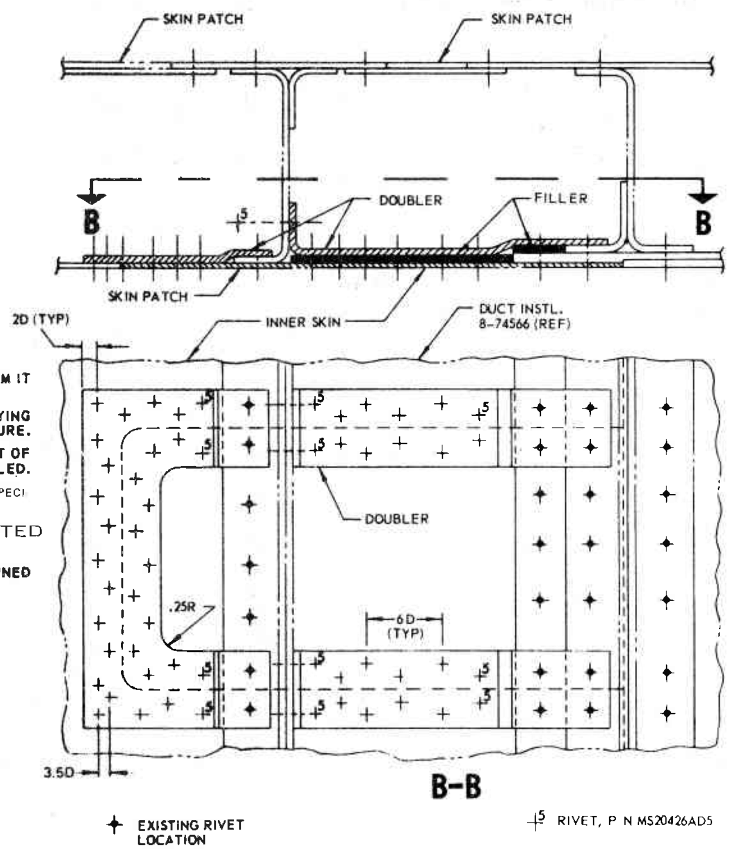


Figure 4-18A. Intake Duct Inner Skin Typical Repair (Sheet 1 of 2)

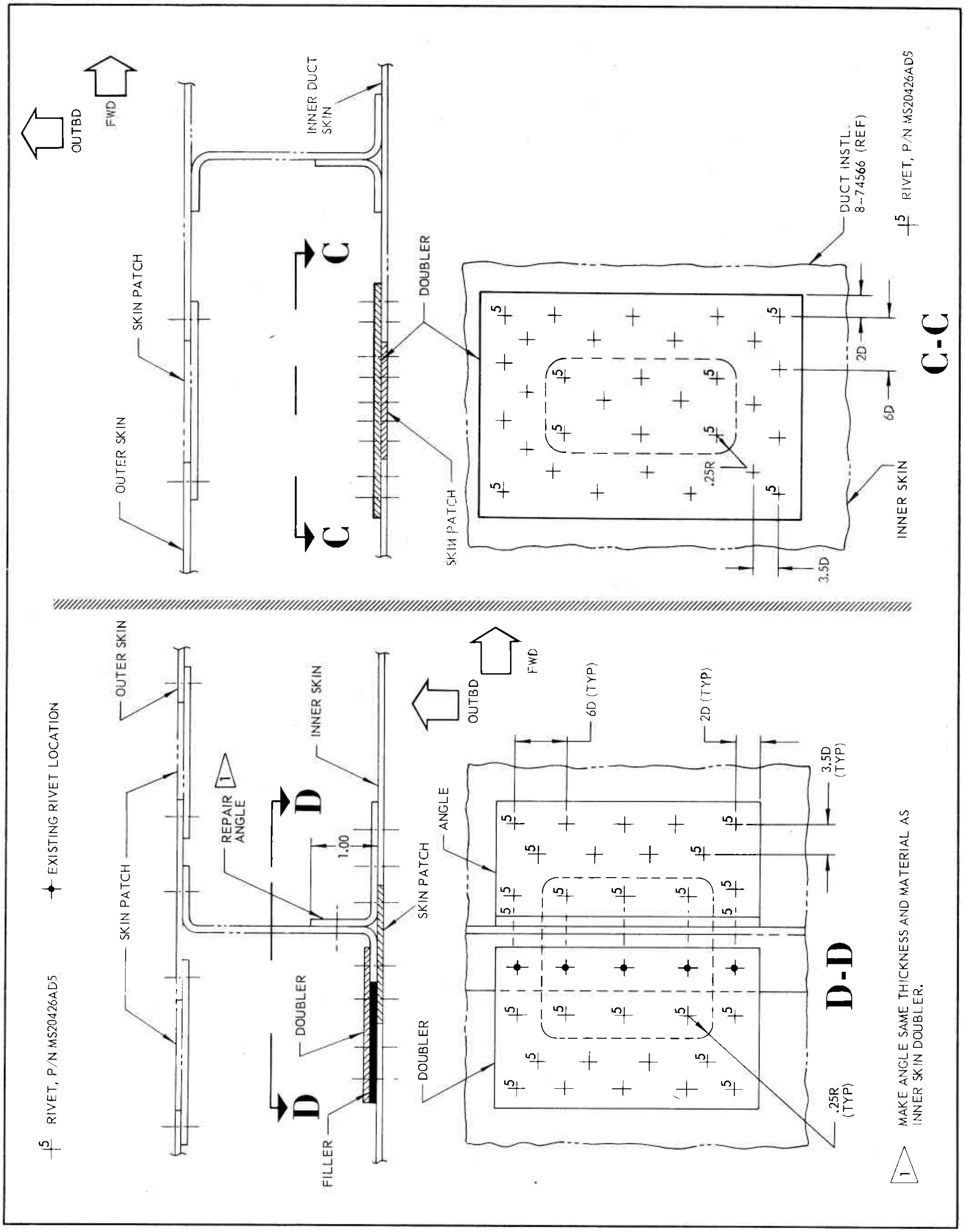


Figure 4-18A. Intake Duct Inner Skin – Typical Repair (Sheet 2 of 2)

DUCT OUTER SKIN REPAIR PROCEDURES

A. FABRICATE A SKIN PATCH FROM 7075–T6 ALUMINUM SHEET OF SAME THICKNESS AS EXISTING SKIN, IN ACCORDANCE WITH DIMENSIONS SHOWN IN VIEW A.

B. FABRICATE TWO DOUBLERS AND TWO SPLICE STRAPS FROM 7075–T6 ALUMINUM SHEET, ONE GAGE THICKER THAN THE SKIN, IN ACCORDANCE WITH DIMENSIONS SHOWN IN VIEW A.

NOTE

FABRICATE THE DOUBLER IN ONE PIECE AND DELETE THE SPLICE STRAPS IN LOCATIONS WHERE THERE IS ROOM TO INSTALL A ONE-PIECE DOUBLER.

C. LOCATE AND DRILL ONE ROW OF HOLES IN THE SKIN PATCH. SEE FIGURE 1–32 FOR RIVET SPACING AND RIVET EDGE DISTANCE INFORMATION. USE A NO. 39 (0.0995-INCH) DRILL.

D. LOCATE AND DRILL TWO ROWS OF HOLES IN EACH DOUBLER AND SPLICE STRAP. SEE FIGURE 1–32 FOR RIVET SPACING AND RIVET EDGE DISTANCE INFORMATION. USE A NO. 39 (0.0995-INCH) DRILL.

E. CENTER THE TWO DOUBLERS AND SPLICE STRAPS OVER THE ACCESS OPENING IN THE DUCT SKIN AND CLAMP IN PLACE.

F. USING THE PRE-DRILLED HOLES IN THE DOUBLERS AS A GUIDE, DRILL HOLES IN THE DUCT SKIN. USE A NO. 39 (0.0995-INCH) DRILL.

NOTE

TO ASSURE ALIGNMENT OF HOLES BETWEEN THE DOUBLER AND DUCT SKIN, INSERT CLECOS THROUGH DOUBLERS AND DUCT SKIN IMMEDIATELY AFTER DRILLING EACH HOLE.

G. REMOVE THE DOUBLER AND REMOVE ALL BURRS, CHIPS, AND OTHER FOREIGN MATERIAL FROM THE DOUBLERS AND DUCT STRUCTURE.

H. PREPARE THE DOUBLER FOR INSTALLATION BY TREATING THE REPAIR MATERIAL AS OUTLINED IN T.O. 1F-106A-23.

I. APPLY A COAT OF SEALER, SPECIFICATION MIL-S-81733, TO THE FAYING SURFACES OF THE DOUBLER AND DUCT SKIN.

J. INSERT THE DOUBLERS AND SPLICE STRAPS THROUGH THE ACCESS OPENING IN THE DUCT SKIN AND ALIGN THE HOLES IN THE DOUBLER WITH THOSE IN THE DUCT SKIN. HOLD THE DOUBLERS AND SPLICE STRAPS IN PLACE WITH CLECOS.

K. REAM THE HOLES IN THE DUCT SKIN AND DOUBLERS THAT ARE NOT FILLED WITH CLECOS, WITH A NO. 30 (0.1285-INCH) DRILL.

L. RIVET THE DOUBLERS TO THE DUCT WITH RIVETS, PART NO. MS20426AD4.

M. REMOVE THE CLECOS AND PERFORM STEPS K AND L IN THAT ORDER TO COMPLETE RIVETING OF DOUBLERS TO DUCT SKIN.

N. APPLY A FILLET OF SEALER, SPECIFICATION MIL-S-81733, AROUND THE PERIPHERY OF THE DOUBLERS AND FILL THE GAP BETWEEN THE DOUBLERS.

NOTE

APPLY A SUFFICIENT AMOUNT OF SEALER, SPECIFICATION MIL-S-81733, TO OVERLAP THE DOUBLERS 0.25-INCH AND OVERLAP THE DUCT SKIN 0.25 INCH. ALSO OVERLAP SEALER 0.25 INCH EACH SIDE OF GAP BETWEEN DOUBLERS.

O. INSERT THE SKIN PATCH INTO THE DUCT SKIN ACCESS OPENING.

P. USING THE PRE-DRILLED HOLES IN THE SKIN PATCH AS A GUIDE, DRILL HOLES THROUGH SKIN PATCH AND DOUBLERS. USE A NO. 39 (0.0995-INCH) DRILL.

Q. REMOVE THE SKIN PATCH AND REMOVE ALL BURRS, CHIPS, AND OTHER FOREIGN MATERIAL FROM THE SKIN PATCH AND DOUBLERS.

R. APPLY A COAT OF SEALER, SPECIFICATION MIL-S-81733, TO THE FAYING SURFACES OF THE SKIN PATCH AND DOUBLERS.

S. REINSERT THE SKIN PATCH AND HOLD IN PLACE WITH CLECOS.

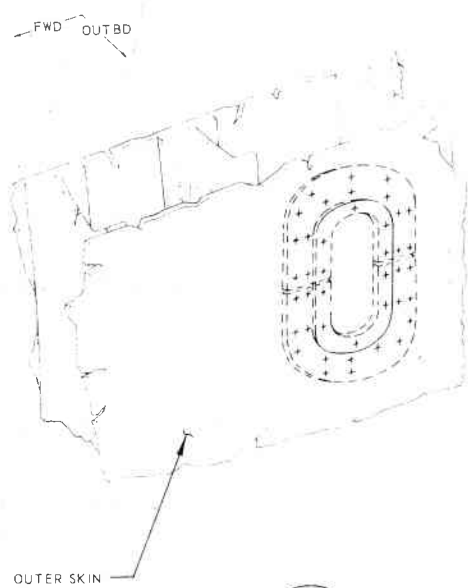
T. REAM THE HOLES IN THE SKIN PATCH AND DOUBLERS THAT ARE NOT FILLED WITH CLECOS WITH A NO. 30 (0.1285-INCH) DRILL.

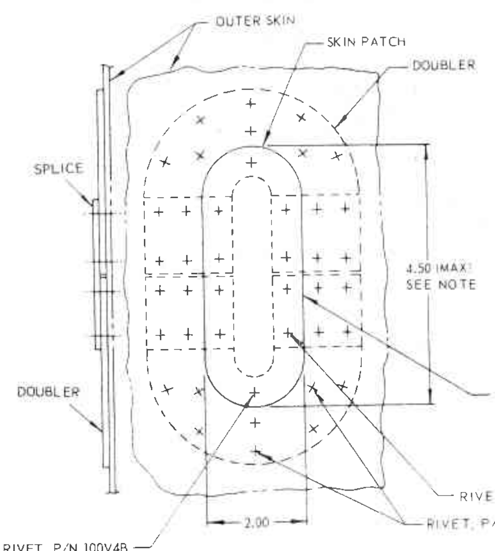
U. RIVET THE SKIN PATCH TO THE DOUBLERS WITH HUCK BLIND RIVETS. REFER TO TABLES I–XLII AND I–XLIII IN SECTION I FOR RIVET INFORMATION.

V. REMOVE THE CLECOS AND COMPLETE RIVETING THE SKIN PATCH TO THE DOUBLERS BY THE PROCEDURES OUTLINED IN STEPS T AND U.

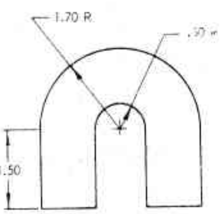
W. FILL THE GAP BETWEEN THE SKIN PATCH AND DUCT SKIN WITH SEALER, SPECIFICATION MIL-S-81733.

X. FINISH THE REPAIR IN ACCORDANCE WITH PROCEDURES OUTLINED IN T.O. 1F-106A-23.

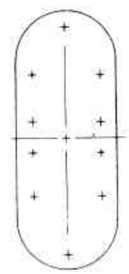




VIEW A



DOUBLER



SKIN PATCH
SEE VIEW 'A'

NOTE

CUT ACCESS HOLE AS REQUIRED BUT DO NOT EXCEED MAXIMUM (4.50 INCHES)

Figure 4-18B. Intake Duct Outer Skin Repair

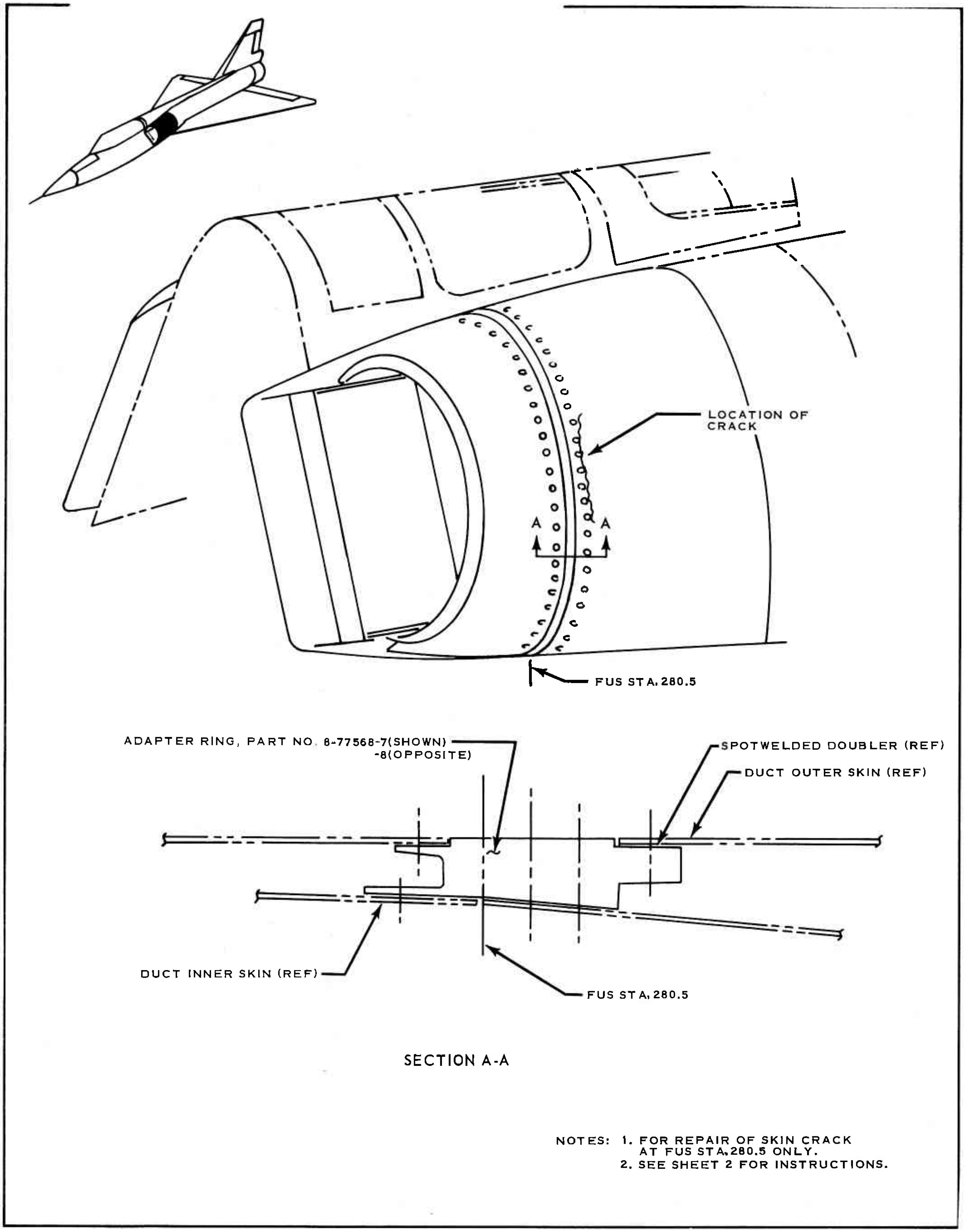


Figure 4-18D. Air Intake Duct, Skin Crack – Typical Repair (Sheet 1 of 3)

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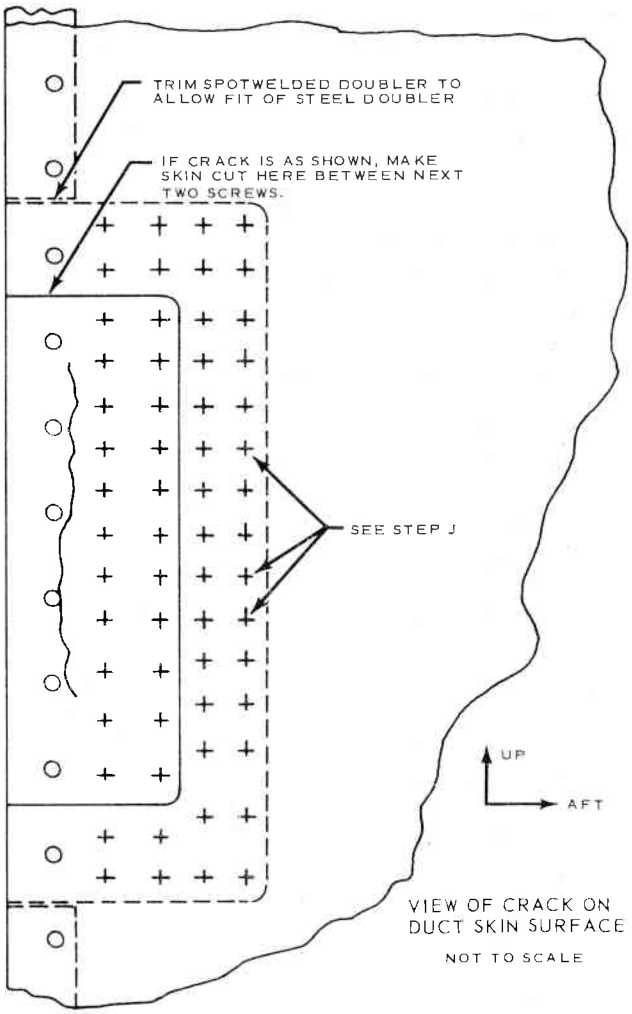
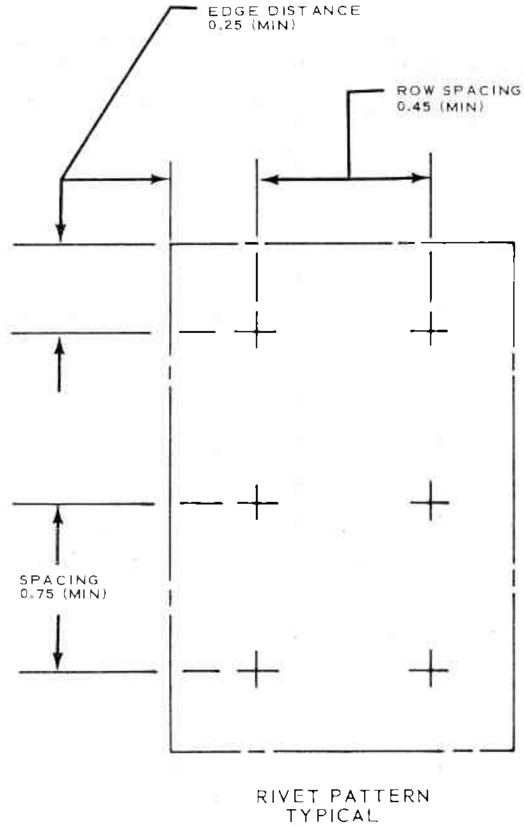




REPAIR PROCEDURES.

a. Determine the length of skin crack. Make the outlines for the repair. Allow for two rows of rivets in both the flush filler patch and the doubler.

NOTE
THIS REPAIR IS LIMITED TO CRACKS 20 INCHES IN LENGTH OR LESS, 10 INCHES ABOVE AND BELOW W.L. 0.00.

b. Fabricate the doubler as shown (width 3.00 minimum) using one of the following:
   1. 0.020 inch thick 301-¼ hard corrosion resistant steel.
   2. 0.020 inch thick 302-¼ hard corrosion resistant steel.

c. Fabricate the filler patch (width 2.00 minimum) from 0.040 inch thick 7075-T6 aluminum alloy.

d. Cut the skin midway between the screws.

e. Lift up the corners of the existing skin and trim back the spotwelded doubler to allow a flush fit (0.032 maximum clearance) of the new steel doubler.

CAUTION
LIFT THE CORNERS OF THE SKIN JUST ENOUGH TO ACCOMPLISH TRIMMING OF THE DOUBLER.

f. Mark the rivet pattern on the skin. Drill rivet holes through the skin and doubler using a No. 30 (0.1285) drill.

g. Attach doubler with Clecos. Using a hole finder or similar instrument, locate and drill the screws holes with No. 11 (0.191) drill through the filler patch and the doubler; mark and drill rivet holes through filler patch and doubler using a No. 30 (0.1285) pilot drill and a No. 29 (0.136) finish drill. Drill "skin to doubler" holes with a No. 29 (0.136) finish drill.

h. Remove all parts and clean up all chips and burrs.

i. Apply primer to all new surfaces as indicated in T.O. 1F-106A-23. Seal all surfaces with sealer, MIL-S-81733.

j. Attach doubler and filler patch using 100 degree countersunk explosive rivets, Dupont part number PN-134-100-X 1/16 or blind locked spindle rivet part number NAS1399-MW4-2.

k. Fill all voids with sealer, MIL-S-81733 and finish in accordance with T.O. 1F-106A-23.

Figure 4-18D. Air Intake Duct, Skin Crack – Typical Repair (Sheet 2 of 3)

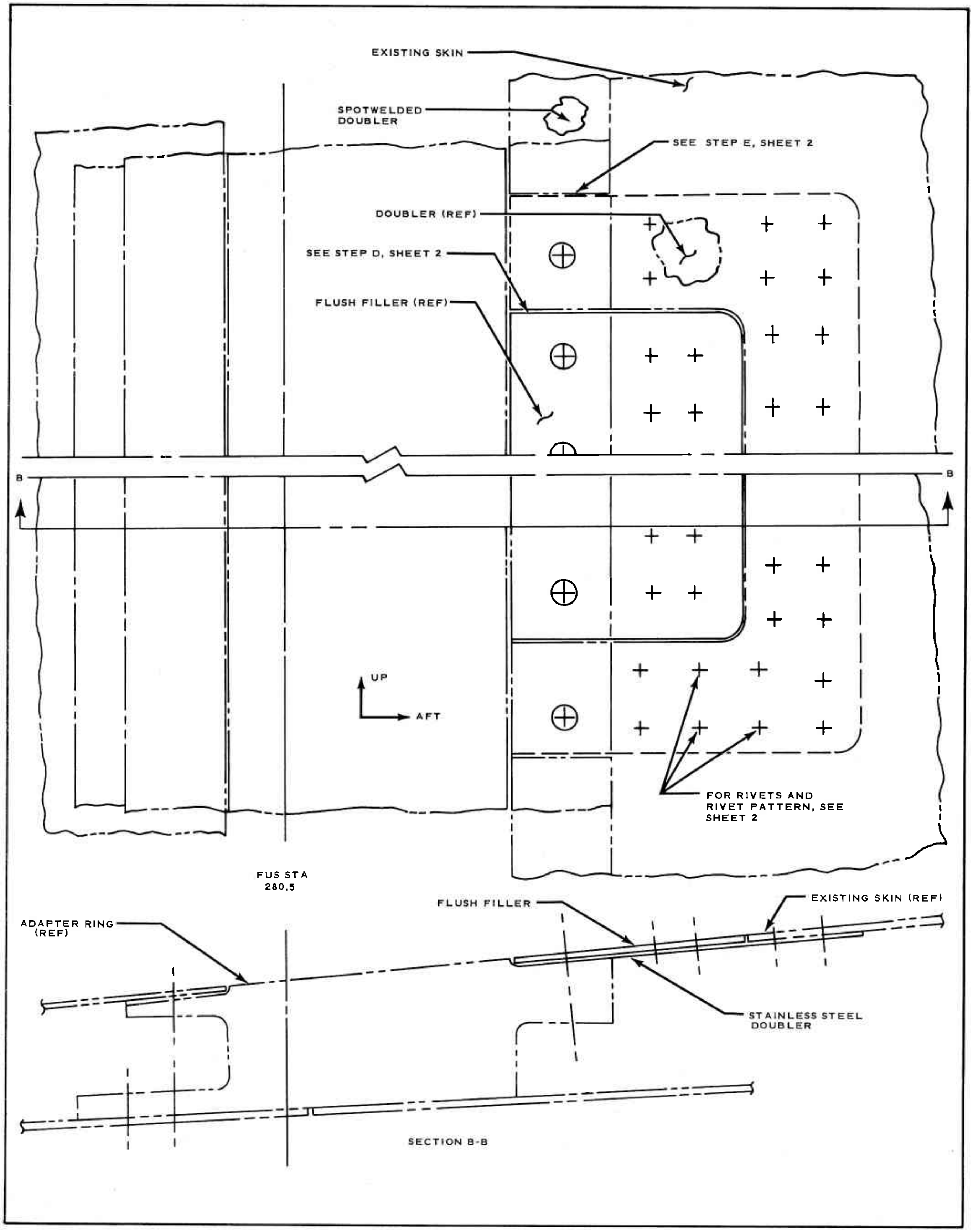
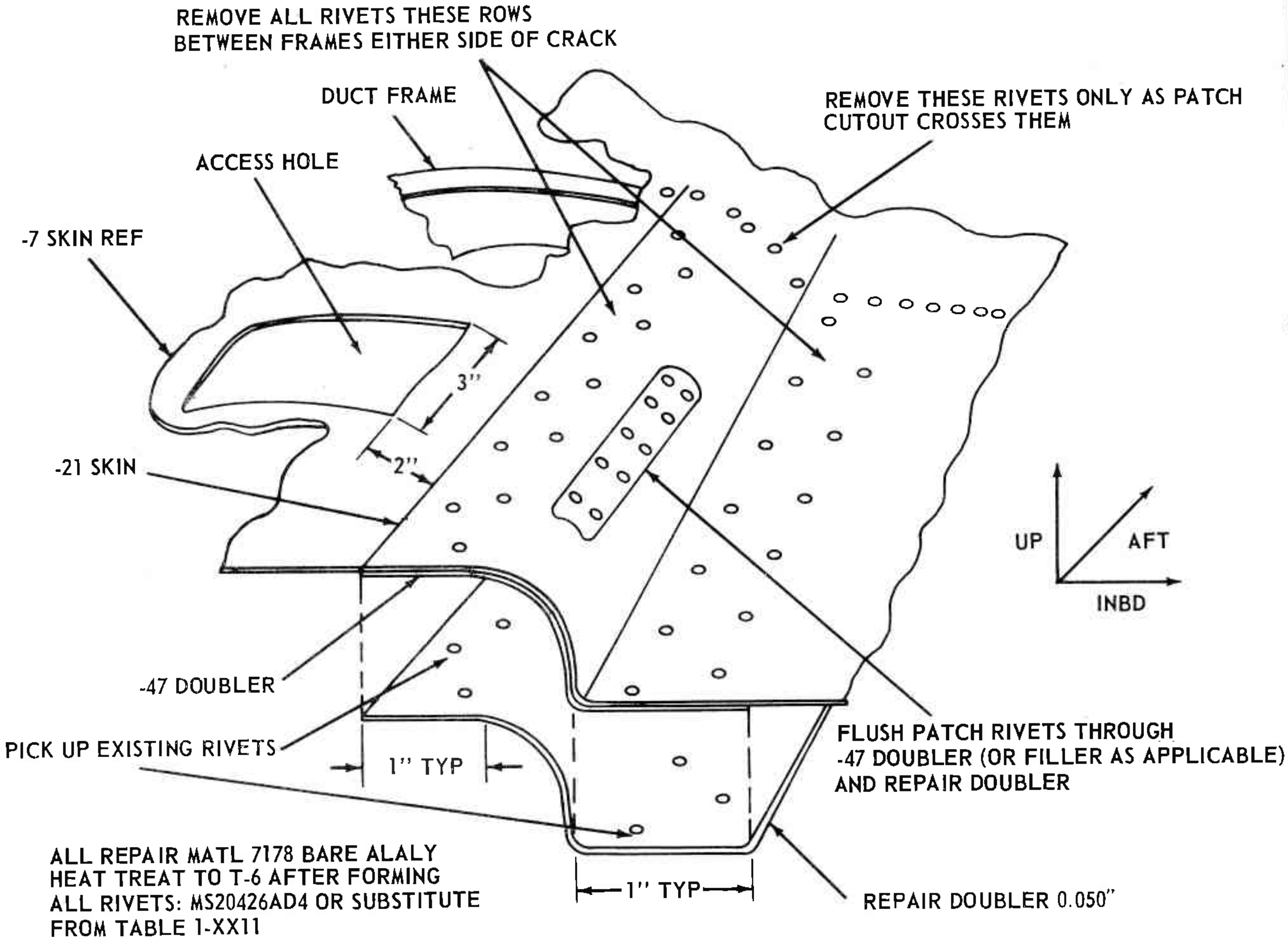


Figure 4-18D. Air Intake Duct, Skin Crack – Typical Repair (Sheet 3 of 3)



TYPICAL CROSS SECTION F.S. 293

REPAIR PROCEDURE
INTAKE DUCT OUTER SKIN (P/N 8-77586) FS280.5 to 316.0

1. Cut an access hole (3" wide and as long as necessary to clean out debris and buck rivets) centered between frames and starting 2" outboard of 8-77586-21 skin.
2. Cut out skin the length of crack and at least one inch wide. If more than one crack exists in adjacent areas it may be desirable to cut out one large piece of skin rather than several smaller ones.

CAUTION

DO NOT DAMAGE DOUBLER BENEATH SKIN

3. Remove rivets indicated in sketch – clean out all debris.
4. Fabricate doubler contoured to nest beneath the 8-77586-47 doubler long enough to extend between any two frames bounding the crack and wide enough to pick up existing rivets, as shown in sketch.

NOTE

IF CRACK EXTENDS ACROSS A FRAME, PUT A DOUBLER ON EACH SIDE OF FRAME EXTENDING EITHER TO THE NEXT FRAME OR AT LEAST THREE RIVETS BEYOND THE END OF THE PATCH CUTOUT.

5. Fabricate flush patch of the same size and contour as the skin cutout. Maximum gap between patch and cutout will be 0.032".
6. Apply primer to all new surfaces as indicated in T.O. 1F-106A-23. Seal all surfaces with sealer MIL-S-81733.
7. Install repair doubler under existing -47 doubler picking up existing rivet holes. Where it is impossible to buck rivets, use substitute blind fasteners (Ref Table 1-XX11).
8. Secure flush patch using at least two rows of rivets to maintain the contour.

NOTE

WHERE THE -47 DOUBLER DOES NOT CONTINUE BETWEEN THE PATCH AND THE REPAIR DOUBLER, INSTALL A 0.050" FILLER BETWEEN PATCH AND REPAIR DOUBLER.

9. Repair access hole IAW Fig 4-18B.

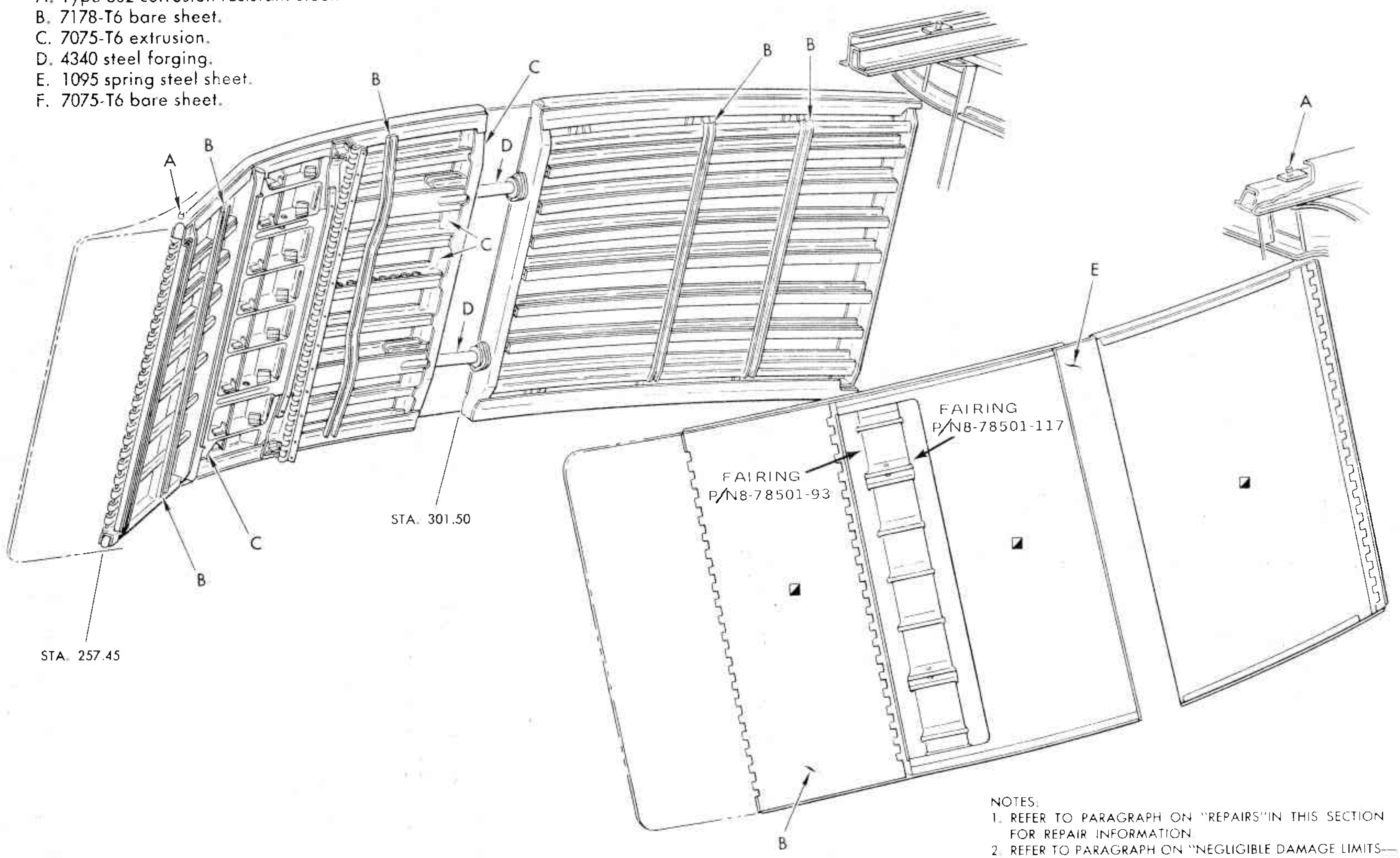
Figure 4–18E. Air Intake Outer Skin (P/N 8-77586) Repair F.S. 280.5-316.0

MATERIAL

UNLESS OTHERWISE NOTED ALL PARTS ARE MADE FROM 2024-T6 BARE SHEET.

A. Type 302 corrosion resistant steel.
B. 7178-T6 bare sheet.
C. 7075-T6 extrusion.
D. 4340 steel forging.
E. 1095 spring steel sheet.
F. 7075-T6 bare sheet.

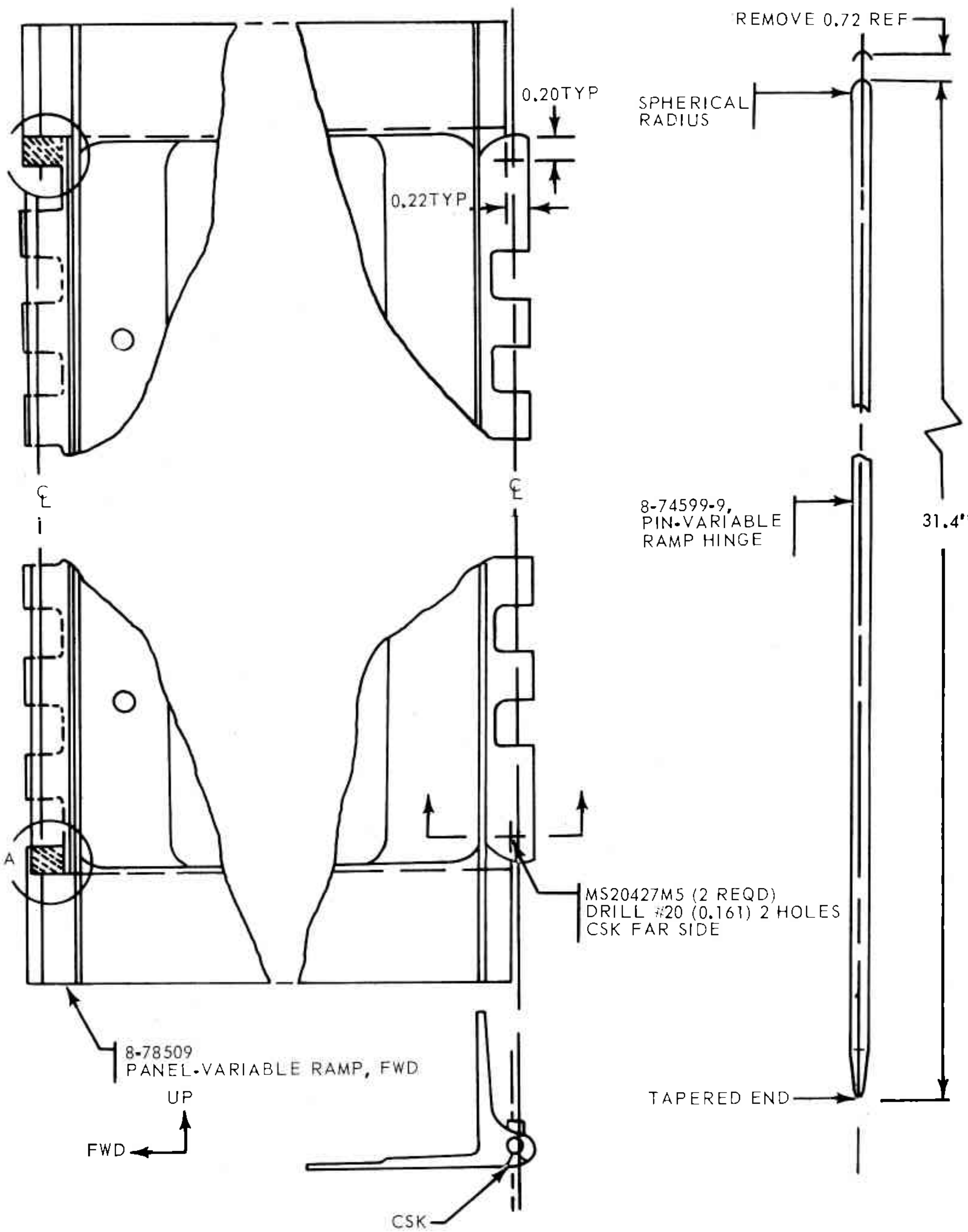
SYMBOL	GAGE
◪	0.071



NOTES:
1. REFER TO PARAGRAPH ON "REPAIRS" IN THIS SECTION FOR REPAIR INFORMATION.
2. REFER TO PARAGRAPH ON "NEGLIGIBLE DAMAGE LIMITS—FUSELAGE GROUP" IN THIS SECTION FOR DISPOSITION OF NEGLIGIBLE DAMAGE TO THIS COMPONENT.
3. FOR REMOVAL AND REPLACEMENT OF VARIABLE RAMP PANELS, REFER TO T.O. 1F-106A-2-4-2-1.

06.03.1427 .53.09.02

Figure 4-19. Variable Ramp Inlet Duct



HINGE NODE REWORK INSTRUCTIONS

- a. Remove variable ramp as required per T.O. 1F-106A-2-4-2-1 Fig 4-9 to gain access to nodes.
- b. Remove cracked, broken off, or excessively worn upper or lower (one only) hinge node (Detail A) flush with surface.
- c. Treat reworked metal in accordance with Section I.

NOTE

WHEN UPPERMOST OR LOWERMOST HINGE NODE OF FUSELAGE ATTACHED HINGE, PART NUMBER 8-74536, IS CRACKED, BROKEN OFF OR EXCESSIVELY WORN, REPAIR DAMAGED NODE IN ACCORDANCE WITH ABOVE INSTRUCTIONS, STEPS A THROUGH C.

HINGE PIN RETENTION REWORK INSTRUCTIONS

- a. Remove the variable ramp per T.O. 1F-106A-2-4-2-1 Fig. 4-9.
- b. Remove intermediate ramp hinge-pin and modify as shown.
- c. Drill and countersink rivet holes.
- d. Reinstall modified hinge-pin so as not to block rivet holes.
- e. Drive rivets and shape rivet heads to conform to external variable ramp panel contour.

Figure 4–19A. Variable Ramp Panel Hinge Node and Pin Retention Repair Procedures

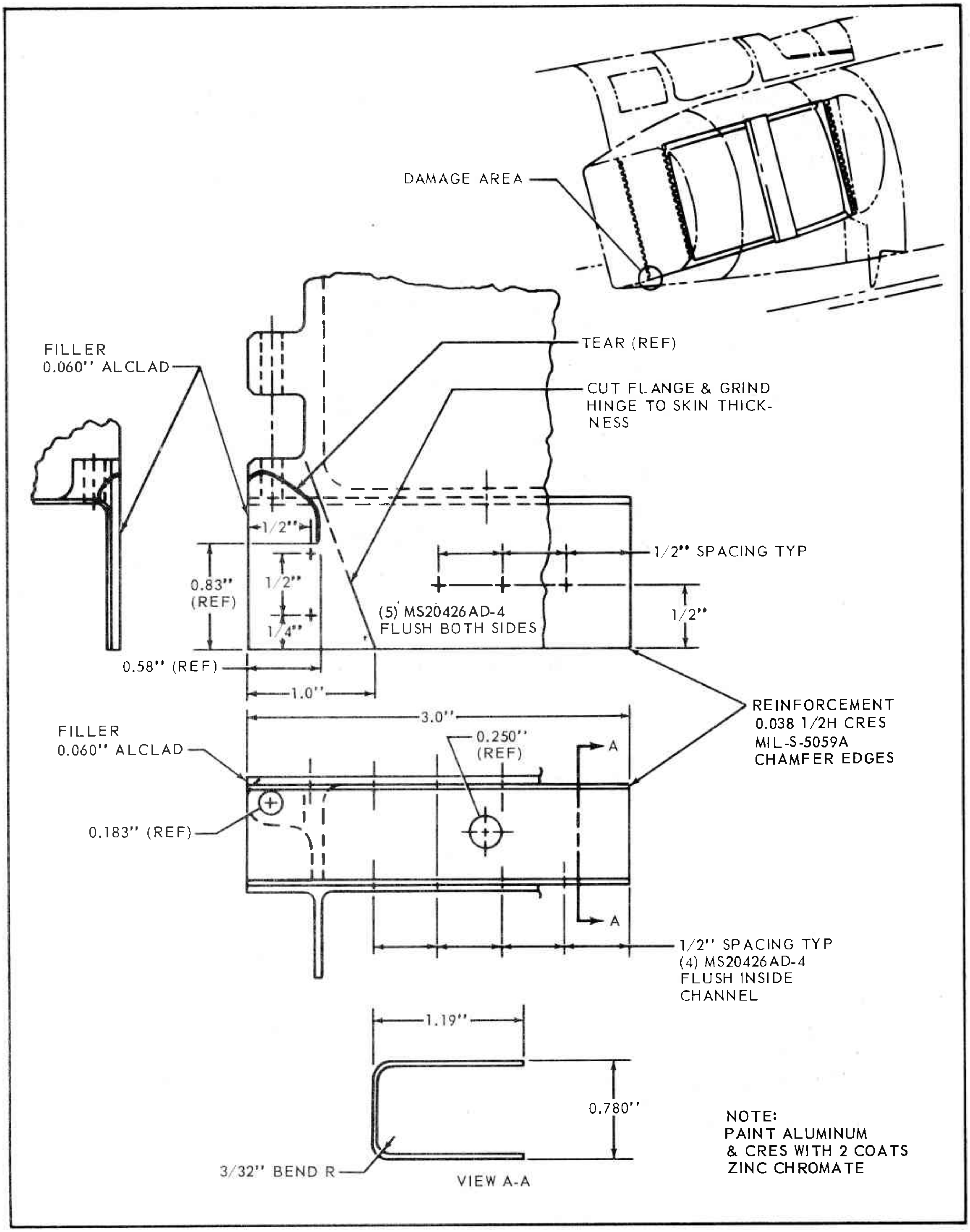


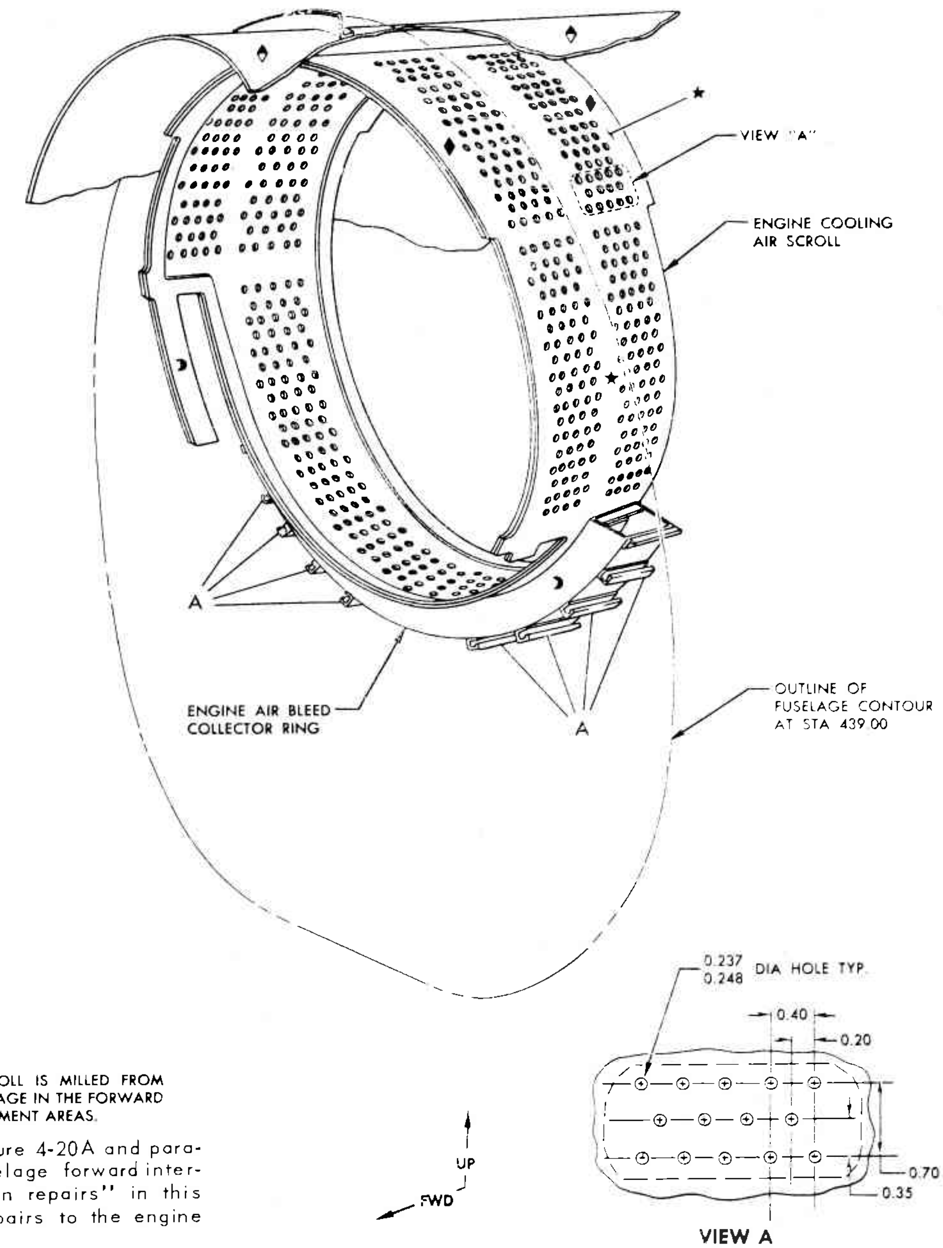
Figure 4-19B Variable Ramp Panel End Lobe and Channel Repair

MATERIAL

UNLESS OTHERWISE NOTED ALL PARTS ARE MADE FROM 7178-T6 BARE SHEET.

A. 7178-T6 extrusion.

SYMBOL	GAGE
☽	0.040
⌀	0.045
♦	0.080
★	0.125

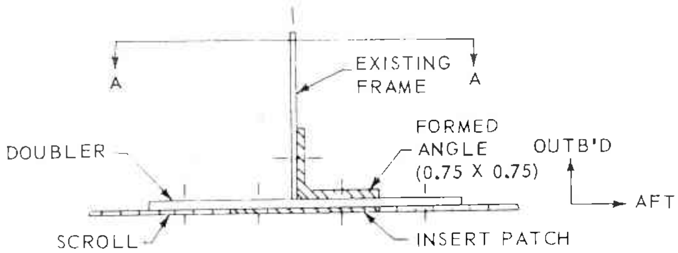


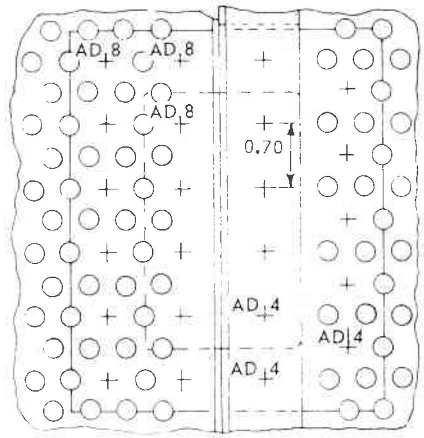
NOTES:

1. ENGINE AIR SCROLL IS MILLED FROM 0.125 TO 0.080 GAGE IN THE FORWARD AND AFT ATTACHMENT AREAS.
2. Refer to figure 4-20A and paragraph on "fuselage forward intermediate section repairs" in this section for repairs to the engine air scroll.

.06.03.164 .53.07.02

Figure 4-20. Engine Cooling Air Scroll





VIEW A – A

REPAIR A
STATION 439.0 or 445.5
REPAIR INSTRUCTIONS

a. Determine the extent of corrosion damage and mark outline for the repair as shown in repair A. Allow for two and a half rows of scroll holes in the insert patch on the aft side of the existing frame and two and a half rows of scroll holes for installation of rivets around the insert patch in the doubler.

b. Remove enough rivets attaching the scroll to the existing frame to allow removal of damage. These rivets will be replaced with MS20426AD4.

c. Cut the scroll on each side of the existing frame as required to remove the damaged area as shown in repair A.

d. Cut the flange of the existing frame to allow the doubler to be positioned between the scroll skin and the existing frame as shown in repair A.

e. Fabricate the insert patch, doubler, formed angle, and shims from 7178-T6 aluminum alloy. The insert patch should be the same thickness as the cut out portion of the scroll. The doubler shall be one guage thicker than the insert patch and the angle shall be the same thickness as the existing frame. The angle shall be long enough to pick up two existing rivets through frame flange on each side of repair cut out. Fabricate shims as required to go under ends of angle to match thickness of doubler.

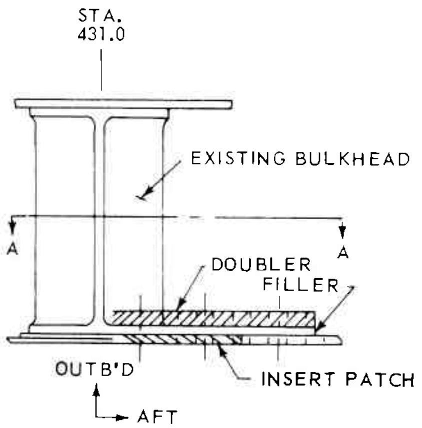
f. Drill rivet hole pattern thru the formed angle and existing frame as shown in above sketch. Drill existing scroll hole pattern thru the insert patch and doubler as shown in figure 4-20 of T.O. 1F-106-3 with 1/4 drill.

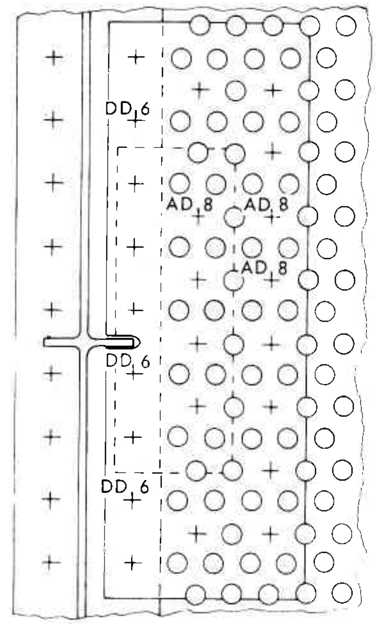
g. Drill rivet hole as shown in above sketch pattern in the insert patch, doubler, and formed angle using 1/8 drill. Select scroll holes to be used for rivet holes and countersink all rivet holes in the insert patch and scroll skin 100°.

h. Remove the insert patch, doubler, and formed angle. Clean all burrs and chips from them and the duct structure. Apply primer per T.O. 1F-106A-23 to all new surfaces. Seal all surfaces with sealer, MIL-S-81733.

i. Attach formed angle to existing frame using MS20470AD4 rivets, attach insert patch and doubler to formed angle using MS20426AD4 rivets, attach insert patch or scroll to doubler using MS20426AD8 rivets.

Figure 4-20A. Typical Repairs for Engine Cooling Air Scroll (Sheet 1 of 2)





VIEW A–A

REPAIR B
STATION 431.0
REPAIR INSTRUCTIONS

a. Determine the extent of corrosion damage and mark outline for the repair as shown in repair B. Allow for two rows of scroll holes in the insert patch and two and a half rows of scroll holes for installation of rivets around the insert patch in the doubler.
b. Remove enough 5/32 stainless steel rivets attaching the scroll to the aft flange of bulkhead to allow removal of damaged area and installation of doublers. These rivets will be replaced with MS20426DD6.
c. Cut the scroll along the center of the bulkhead and aft as required to remove the damaged area as shown in the above sketch.
d. Fabricate the insert patch, filler and doubler from 7178-T6 sheet. The insert patch shall be the same thickness as the cut out portion of scroll. The doubler shall be one guage thicker than the insert patch and the filler thick enough to allow the doubler to fit flush on top of the bulkhead flange. Notch doubler and filler as required to clear bulkhead web stiffeners.
e. Drill existing scroll hole pattern through the doubler and filler and add scroll hole pattern in insert patch, doubler and filler as shown in figure 4-20 of T.O. 1F-106-3 with 1/4 drill.
f. Select scroll holes to be used for rivet holes as shown in repair B and countersink 100° for MS20426AD8 rivets.
g. Enlarge 5/32 holes through bulkhead flange to 3/16 and countersink for MS20426DD6 rivets.
h. Remove the insert patch, filler and doubler. Clean all burrs and chips from them and the duct structure.
i. Apply primer per T.O. 1F-106A-23 to all new surfaces. Seal all surfaces with sealer, MIL-S-81733.
j. Install insert patch, filler and doubler with MS20426AD8 and MS20426DD6 rivets in previously drilled holes.

Figure 4-20A. Typical Repairs for Engine Cooling Air Scroll ***(Sheet 2 of 2)***

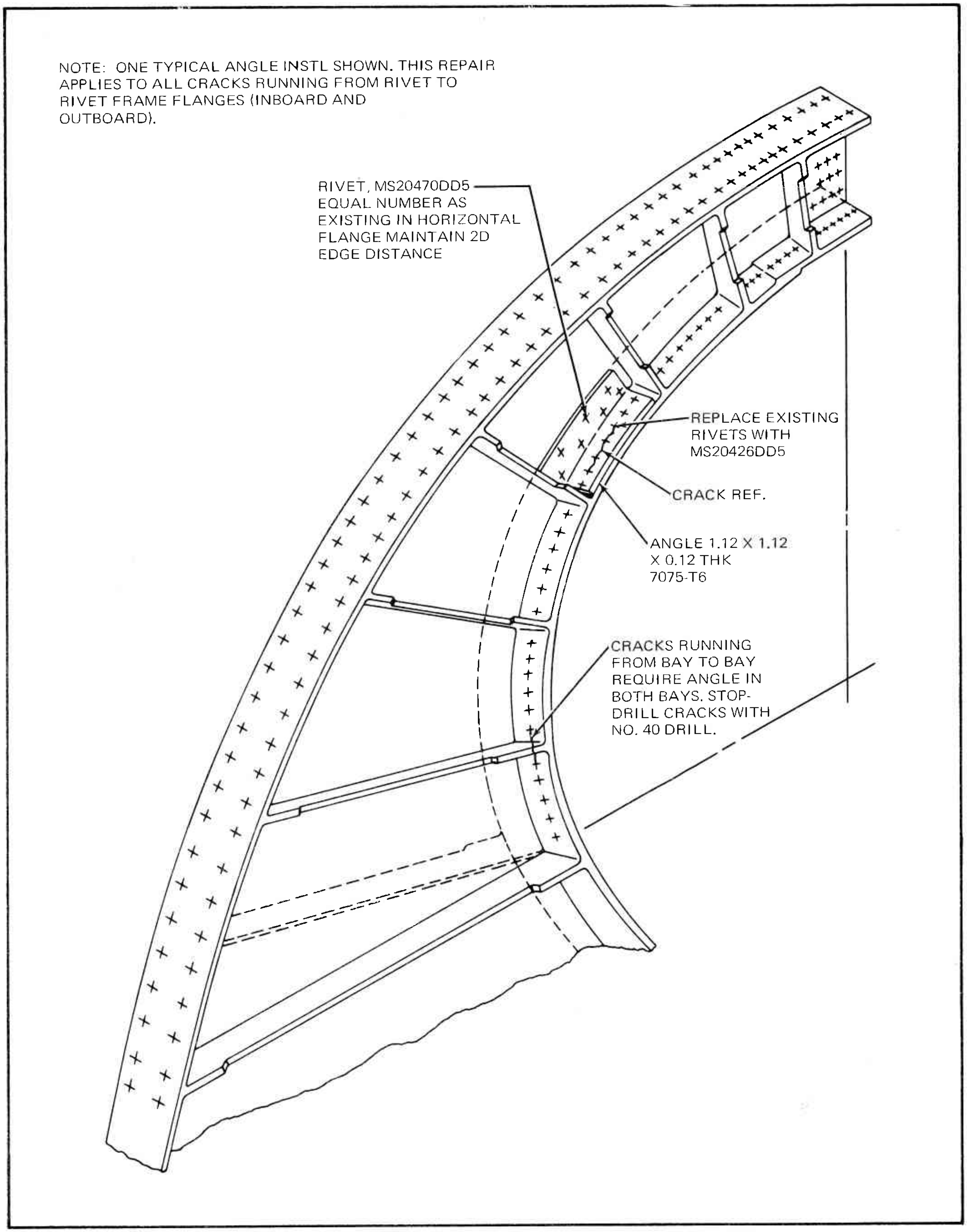


Figure 4-20B. Typical Repairs-Bulkhead Fuselage Station 431

of 7178-T6 bare sheet and is attached with flush-head rivets to the fuselage framework between stations 431.00 and 445.50. The engine air scroll supplies cooling air to the engine by bleeding air from the engine air intake duct. This bleed air passes through the perforated scroll into the engine air bleed collector ring located at the bottom of the engine air scroll and then aft to the engine. See figure 4-20 for an illustration of the engine cooling air scroll.

4-16. The missile bay area is contained within the lower section of the fuselage between stations 216.00 and 412.00, and is enclosed by hinged, double missile bay doors on each side of the fuselage center line. The missile bay doors are constructed so as to be aerodynamically smooth with the outside contour of the airplane when in the closed position. The missile bay doors on each side of the fuselage center line consist principally of an upper and lower door made of aluminum alloy inner and outer plating and a series of evenly spaced press-formed channel ribs interconnected by gussets and intercostals enclosed within the plating. The two missile bay doors on each side are hinged together by hinge fittings and pins, and each set of doors is attached to the fuselage structure in the same manner. See figure 4-21 for an illustration of the missile bay door structure and plating.

4-17. Fuselage Forward Intermediate Section—F-106B.

4-17A. Fuselage Forward Intermediate Section *Applicable after incorporation of TCTO 1F-106-986.* After incorporation of the above specified modification, the airplanes are equipped for aerial refueling. The changed areas are in the fuselage dorsal sections aft of the refrigeration compartment access door. New structure consists of dorsal fairing sections, slipway door, and support structure for the aerial refueling receptacle. See figures 4-16A through 4-16H and 4-22A through 4-22F for illustrations of the aerial refueling structure, and for repair data and reference.

NOTE

All repairs made in the dorsal aerial refueling fuel-tight area are to be sealed using sealer, Specification MIL-S-8802, Class B2.

4-18. The fuselage forward intermediate section of the F-106B is similar to that of the F-106A, except for the location of the electronic equipment in the forward section of the missile bay area. See figure 4-21 for the differences between the F-106A and F-106B missile bay doors, and figure 4-22 for differences in fuselage structure.

4-19. Fuselage Aft Intermediate Section.

4-20. The aft intermediate sections of the fuselage on the F-106A and F-106B are identical. This section consists principally of a series of spliced semicircular zee and channel beltframes, and forged and built-up type bulkheads interconnected by longerons, gussets, and intercostals. The fuselage framework is enveloped by stressed skins of aluminum, magnesium, and titanium that are attached to the structure by means of flush-head rivets and fasteners. The wing attach fittings for spars 4, 5, 6, and 7 are located in this section of the fuselage structure. The vertical fin is also permanently attached to this section of the fuselage. This aft intermediate section is illustrated on figure 4-23. The speed brake and drag chute housing is attached to spar number 5 of the fin structure and to the upper fuselage structure; this section is shown on figure 4-24. The drag chute housing structure consists of rectangular frame assemblies attached to the fuselage structure by means of longitudinal channels and angles. The built-up structure of the drag chute housing is reinforced by canted channels, and is enveloped by stressed aluminum alloy sculptured plating. Two doors of clam-type configuration are hinged to the aft end of the housing structure. These doors serve a dual function: that of enclosing the drag chute housing, and as speed brakes when opened. The speed brake doors are constructed of magnesium alloy sand castings. A seal is located along the inside edge of the doors and around their perimeter to provide weathertightness to the drag chute housing and to serve as a buffer between the two fast-acting speed brake doors. See figure 4-26 for an illustration of the tail hook installation.

4-21. Fuselage Aft Section.

4-22. The fuselage aft section is commonly referred to as the tail cone and is constructed principally of a series of semicircular zee and channel belt frames interconnected by gussets and intercostals. The tail cone is enveloped by aluminum and titanium stressed skins attached to the framework by means of flush-head rivets and other fasteners. The tail cone is attached by pin-type latches to the main portion of the fuselage structure. Figure 4-25 shows the tail cone structure.

4-23. INDEXING.

4-24. Figure 4-1 shows the components of the fuselage and gives a figure index for individual drawings of the various fuselage components. These individual component drawings in turn reference applicable repair illustrations and instruction.

4-25. REPAIRS.

4-25A. Repairs to cracks in fuselage bulkhead, Station 593.46, in AFT engine mount area, will be accomplished as shown in figure 4-23B.

4-26. Although there are some structural differences between the F-106A and F-106B airplanes between stations 102.00 and 472.00, all repairs may be made in the same manner for both airplanes. For repairs of the lower surface of the airplane, refer to Section IX for illustrations of the underside of the fuselage structure.

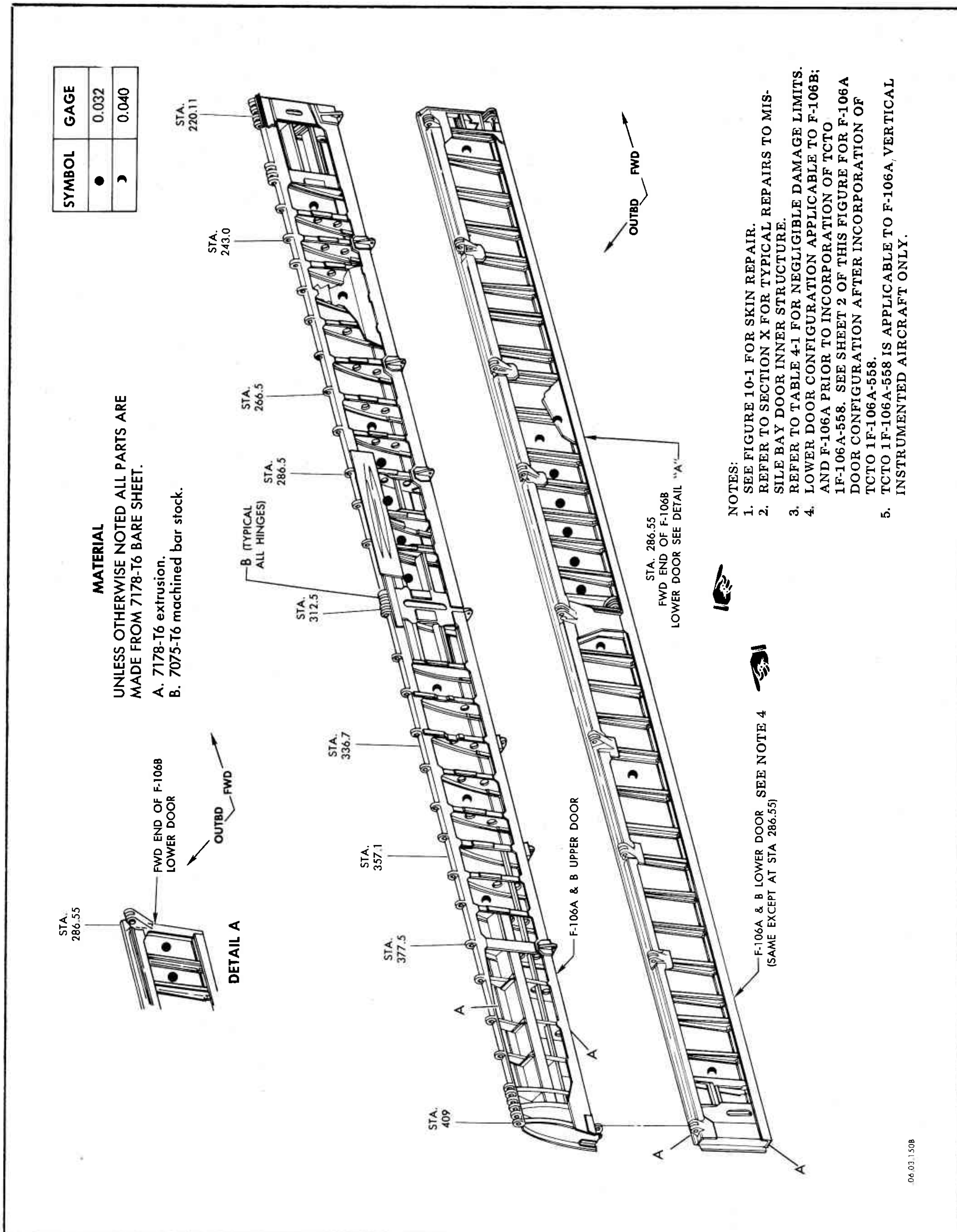
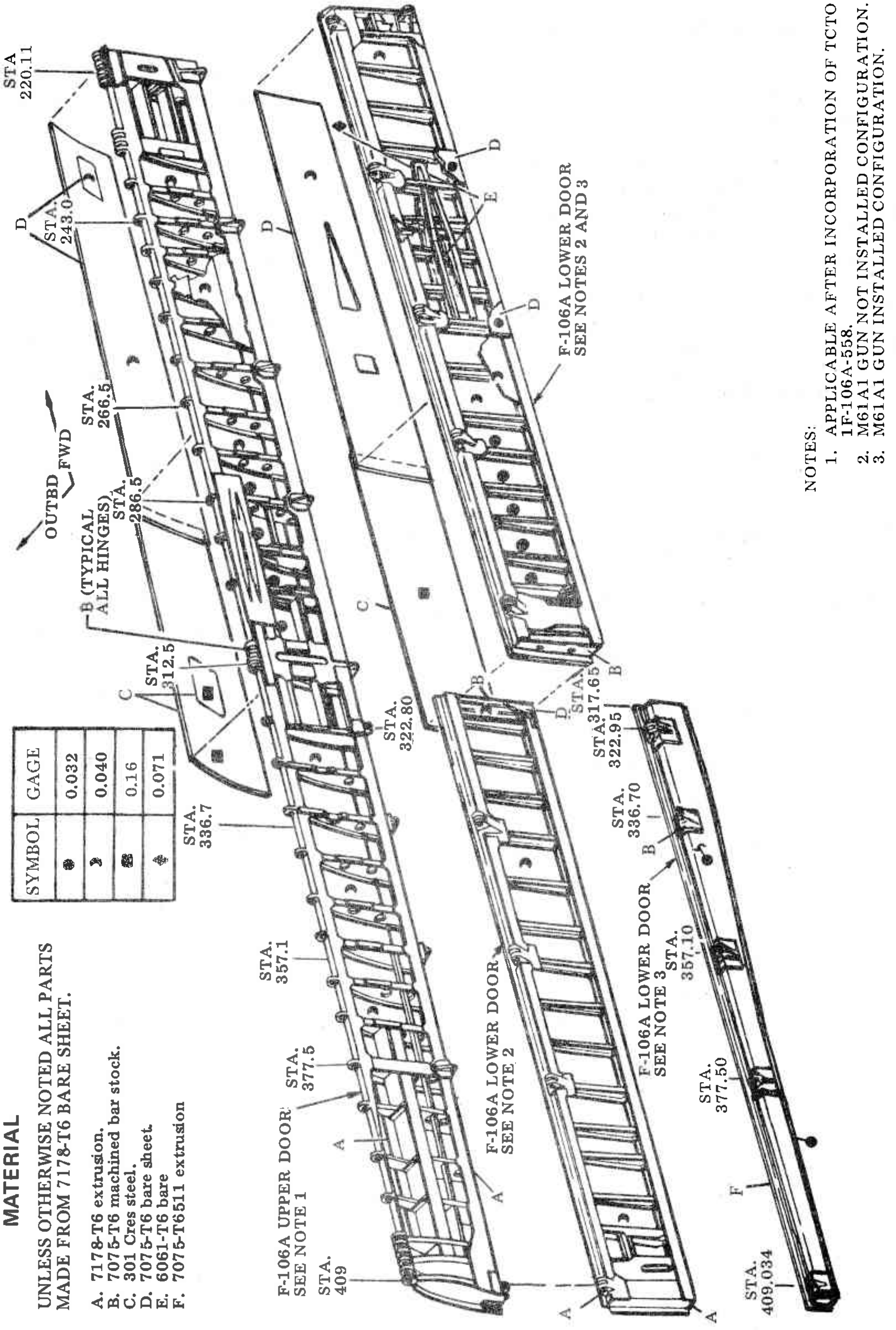


Figure 4-21. Missile Bay Door Structure and Plating (Sheet 1 of 2)



SYMBOL	GAGE
●	0.032
◗	0.040
▩	0.16
◆	0.071

Figure 4-21. Missile Bay Door Structure and Plating (Sheet 2 of 2)
Applicable to F-106A Vertical Instrumented Aircraft after incorporation of TCTO 1F-106A-558.

4-26A. Expanded Metal Repair (F-106A Gun Air Outlet Covers).

4-26B. The F-106A gun assembly external fairing incorporates four gun cooling air outlet openings equipped with expanded metal covers shaped to the contour of the gun fairing assembly. The two forward side-by-side openings each have expanded metal covers having dimensions of 6.25 inches by 18.50 inches (115.62 square inches area each). The two aft side-by-side openings each have expanded metal covers having dimensions of 5.50 inches by 7.50 inches (41.25 square inches area each). The combined area of the four expanded metal covers is approximately 313.74 square inches. The covers are made of type 304-2B 0.050-inch stainless steel sheet expanded into diamond shaped openings of approximately 0.750 inch by 1.812 inches. Each expanded metal cover is bonded into a fiberglass frame which is riveted to the gun fairing assembly.

4-26C. Two types of repairs are recommended for the cover assemblies expanded metal.

a. Misshaped and broken strands of expanded metal with no metal lost may be repaired as follows:

(1) Reshape distorted area to original contour using a plastic mallet with a wood block or shot bag backup. Straighten strands to original position and contour.

WARNING

Keep Methyl-ethyl-ketone (MEK) away from sparks and flames. Use only in well-ventilated area. Avoid prolonged breathing of vapors. Avoid eye and repeated skin contact.

(2) Clean paint from area to be welded by sanding and wire brushing. Finish paint removal using a cloth dampened with methy ethyl ketone (MEK), Specification TT-M-261.

(3) Cut piece of copper sheet (0.040) slightly larger than area of damage.

Note

Copper sheet backup plate is required to hold expanded metal contour during welding.

(4) Drill holes (2) for 1/8-inch bolts (one at each end of copper plate).

(5) Shape plate to outside contour of stretched metal cover.

(6) Bolt plate to outside surface of stretched metal cover over damaged area. Use washers under nuts to aid gripping grid of the stretched metal.

Note

If requirement for welding occurs within 0.75 inch of the fiberglass frame material, pack fiberglass area adjacent to area to be welded using wet asbestos, Specification MIL-A-17472 or equivalent, to prevent heat transfer into fiberglass bond.

(7) On the inside surface of the cover, heli-arc weld the broken strands to their original position using 321/347 stainless welding rod.

(8) Remove copper backing plate and asbestos packing if used.

(9) Shape and remove excess welding by filing.

(10) Refinish cover inner surface in area of repair by applying two coats of MIL-C-83268 urethane coating, insignia white color number 17875 of Federal Standard 595 per Specification MIL-F-18264.

(11) Refinish cover exterior surface in area of repair by applying two coats of MIL-C-83286 urethane coating, aircraft color number 16473 of Federal Standard 595 per Specification MIL-F-18264.

b. Damaged stretched metal covers having metal broken out may be repaired as follows:

Note

This repair is to used only if the total area of all existing and required repair plates does not exceed 2 percent of the total square inch area of all four air outlets combined as specified in paragraph 4-26B. Required repairs exceeding this limit will necessitate replacement of damaged parts.

(1) Reshape damaged area to original contour using a plastic mallet with a wood block or shot bag backup. Straighten strands to original position and contour.

WARNING

Keep methyl-ethyl-ketone away from sparks and flames. Use only in well-ventilated area. Avoid prolonged breathing of vapors. Avoid eye and repeated skin contact.

(2) Clean paint from area to be welded by sanding and wire brushing. Finish paint removal using a cloth dampened with methyl ethyl ketone (MEK), Specification TT-M-261.

(3) Cut piece of 0.050 inch 304-2B stainless sheet large enough to cover area of lost expanded metal with 0.50-inch overlap. Keep plate as small as possible, yet adequate to cover area and to facilitate welding.

(4) Chamfer edges of repair plate and shape plate to inside contour of repair area.

(5) Clamp repair plate to inside surface to cover assembly over repair area.

Note

If requirement for welding occurs within 0.75 inch of the fiberglass frame material, pack fiberglass area adjacent to area to be welded using wet asbestos, Specification MIL-A-17472 or equivalent, to prevent heat transfer into fiberglass bond.

(6) Heli-arc weld repair plate to cover assembly expanded metal using 321/374 stainless welding rod.

(7) Remove clamps and asbestos packing if used. Shape and remove excess weld by filing.

(8) Refinish cover inner surface in area of repair by applying two coats of MIL-C-83286 urethane coating, insignia white color number 17875 of Federal Standard 595 per specification MIL-F-18264.

(9) Refinish cover exterior surface in area of repair by applying two coats of MIL-C-83286 urethane coating, aircraft grey color number 16473 of Federal Standard 595 per Specification MIL-F-18264.

4-27. Rivet Substitution.

4-28. All rivets used in the fuselage of the F-106A and F-106B airplanes may be substituted with different types of rivets which have either the same or a greater strength value than the original rivet, except for the following restrictions:

a. Blind rivets may be used inside of air intake ducts and forward of air induction system only when impossible to use solid rivets. Refer to Section X, paragraph 10-7 for use and substitution of blind rivets/fasteners in these areas.

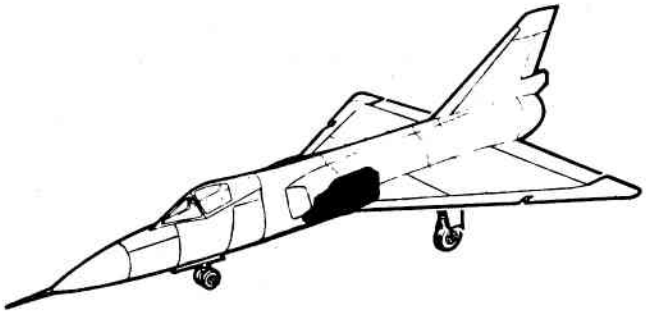
b. Flush rivets must be replaced by flush rivets. Refer to Table I-XXII for rivet substitution data.

4-29. Negligible Damage Limits — Fuselage Group.

4-30. Table 4-1 indicates the maximum allowable classifications of the five types of negligible damage for the fuselage (damage allowed to remain "as is" after minor rework, such as stop drilling cracks and fairing in nicks or scratches). The maximum allowable damage classification will be found to the right of the component name of the vertical column under the "Type of Damage" heading. After classification is determined, see figures 1-17 through 1-19 for the damage limits allowed for each class; I, II, or III. The limits given on figures 1-17 through 1-19 apply only for a damaged area after rework, as shown on figures 1-20 and 1-21. An aeronautical structures engineer must be consulted for damages exceeding the limits on figures 1-17 through 1-19, and for damage to components not listed in this table.

4-30A. SPEED BRAKE HINGE BUSHING REPAIR AND LOWER HINGE FITTING INSTALLATION HARDWARE.

4-30B. Refer to figure 4-41G for repair of speed brake hinge and speed brake door bushings and Figure 4-41H for lower hinge fitting installation hardware.



MATERIAL

UNLESS OTHERWISE NOTED, ALL PARTS ARE MADE FROM 7075 STOCK.

A. 7075 T-0, HEAT TREATED TO T-73
B. 7075-T-6, HEAT TREATED TO T-73
C. 7075 T-651, HEAT TREATED TO T-7351
D. 6061 T-6

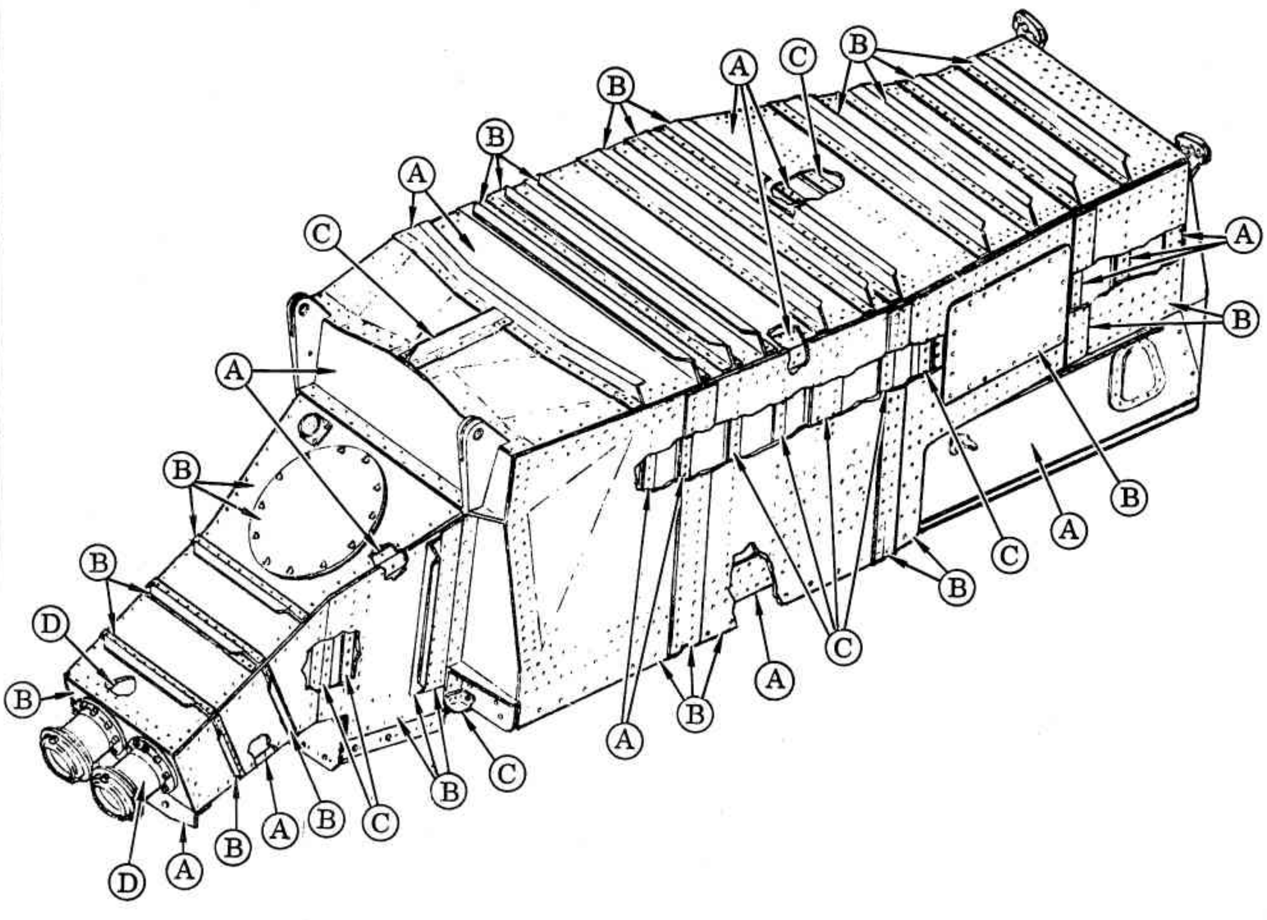
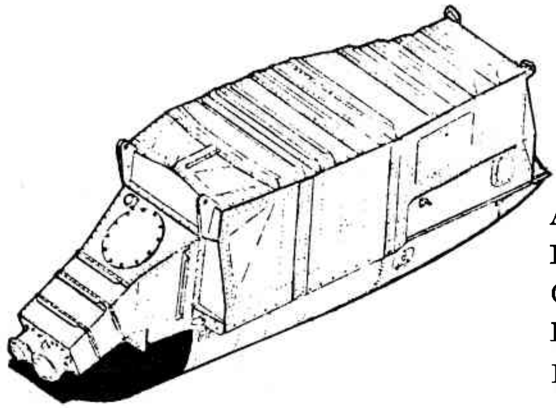


Figure 4-21A. F-106A Gun Housing Structure
Applicable to F-106A Vertical Instrumented Aircraft after incorporation of **TCTO 1F-106A-558.**



MATERIAL

A. 7075 T-0 HEAT TREATED TO T-73
B. 7075 T651 HEAT TREATED TO T73511
C. 321 CRES, COMPOSITION T-1
D. 6061 T-0 HEAT TREATED TO T-6
E. 321 CRES, FINISH 2D

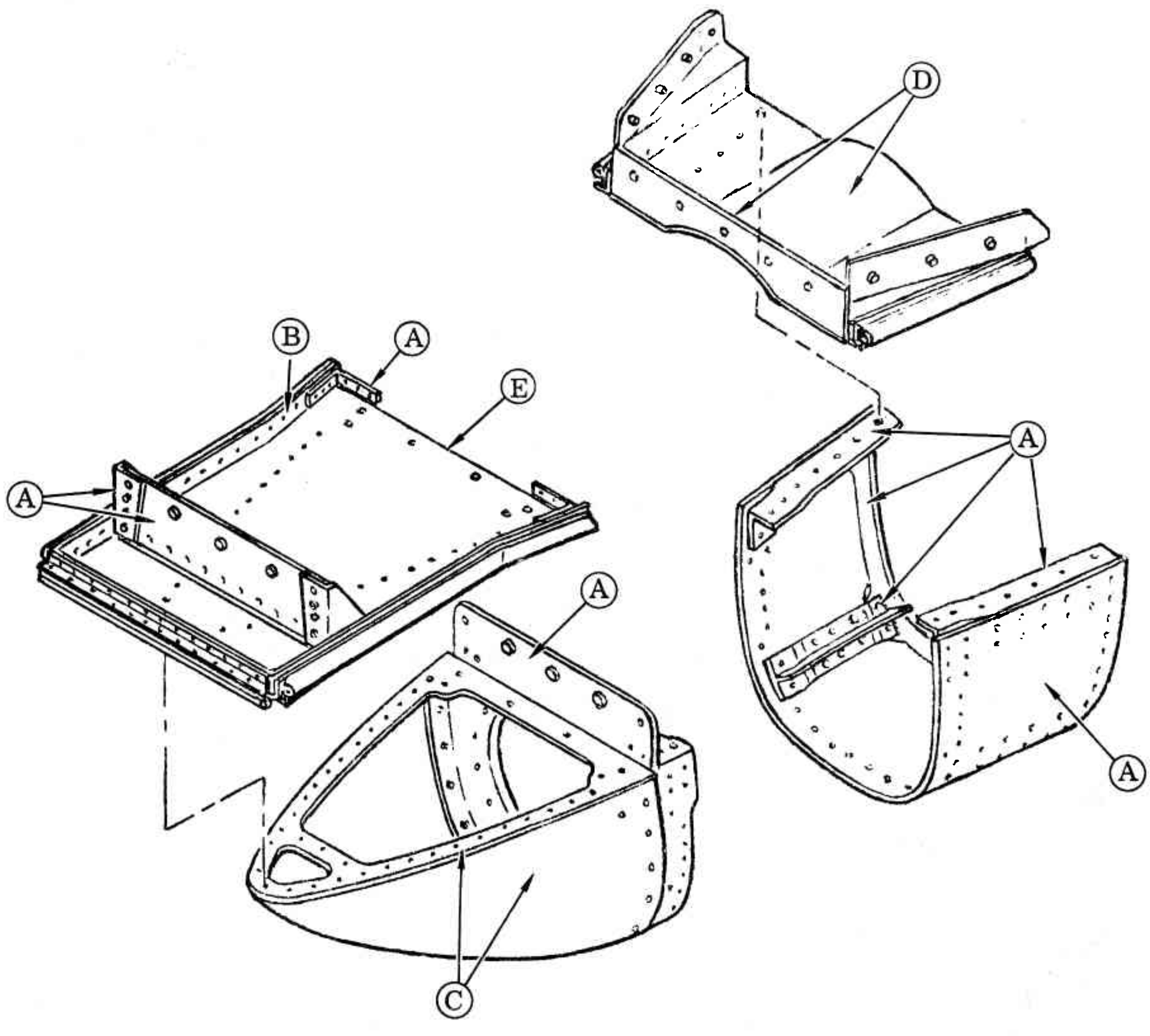
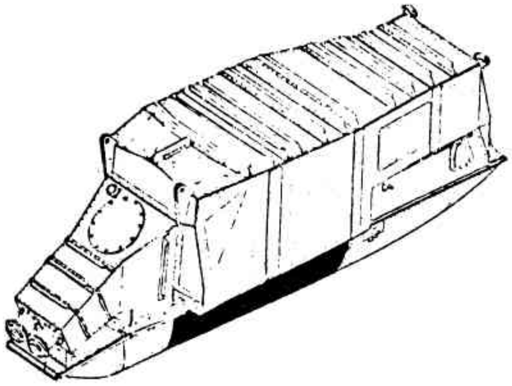


Figure 4-21B. F-106A Gun Fairing Structure (Sheet 1 of 3)
Applicable to F-106A after incorporation of TCTO 1F-106A-558.



MATERIAL

UNLESS OTHERWISE NOTED, ALL PARTS ARE MADE FROM 7075-T0 STOCK THEN HEAT TREATED TO THE T73 CONDITION

A. 7075-T651 MACHINED FROM BAR STOCK AND HEAT TREATED TO T7351
B. 17-4PH CRES, AMS5643, COND. A, H1100 MACHINED FROM 17-4PH CRES, AMS5643, COND. A BAR STOCK
C. 7075-T73 MACHINED FROM 7075-T6 EXTRUSION
D. 7075T-0 MACHINED FROM BAR STOCK AND HEAT TREATED TO T73

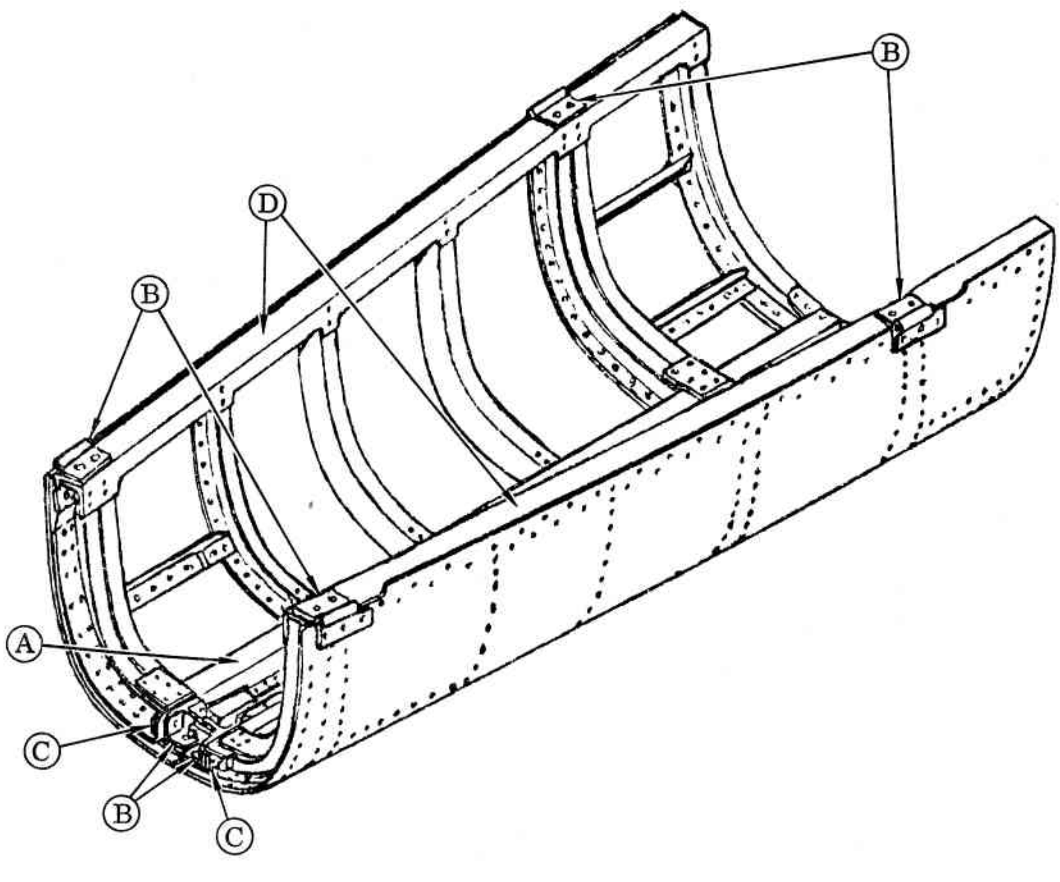
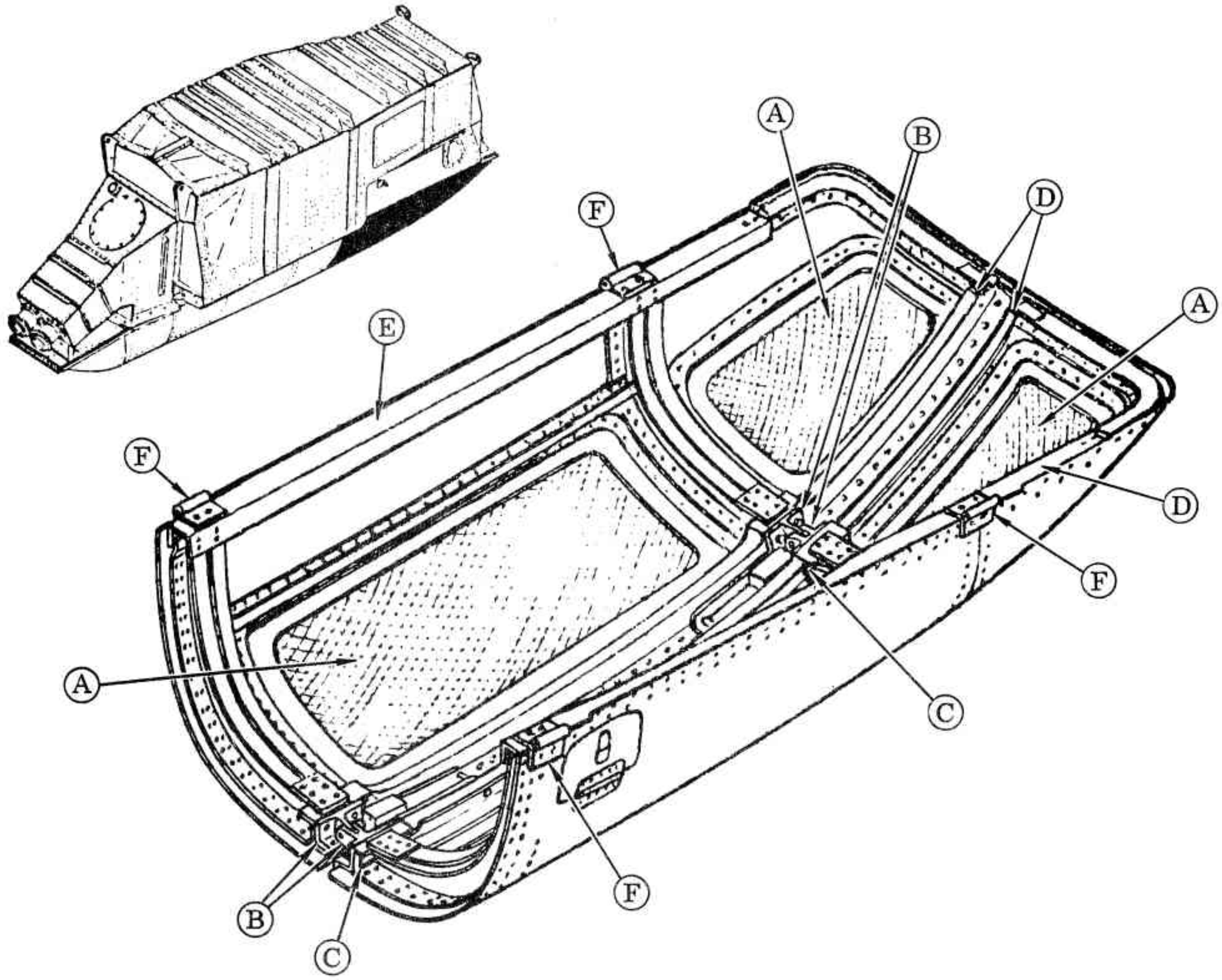


Figure 4-21B. F-106A Gun Fairing Structure (Sheet 2 of 3)
Applicable to F-106A after incorporation of TCTO 1F-106A-558.



MATERIAL

UNLESS OTHERWISE NOTED, ALL PARTS ARE MADE FROM 7075 STOCK THEN HEAT TREATED TO THE T73 CONDITION

A. 3/4 NO. 16 GLOBE (11432)
B. 17-4PH CRES, AMS5643, H1100 MACHINED FROM 17-4PH CRES, AMS5643, COND. A BAR STOCK
C. 7075-T73 MACHINED FROM 7075-T6 EXTRUSION
D. 7075-T651 MACHINED FROM BAR STOCK AND HEAT TREATED TO T-7351
E. 7075-T6 MACHINED FROM BAR STOCK AND HEAT TREATED TO T-73
F. 17-4PH CRES, AMS5643, COND. A, H1150 MACHINED FROM 17-4PH CRES, AMS5643, COND. A BAR STOCK

Figure 4-21B. F-106A Gun Fairing Structure (Sheet 3 of 3)
Applicable to F-106A after incorporation of TCTO 1F-106A-558.

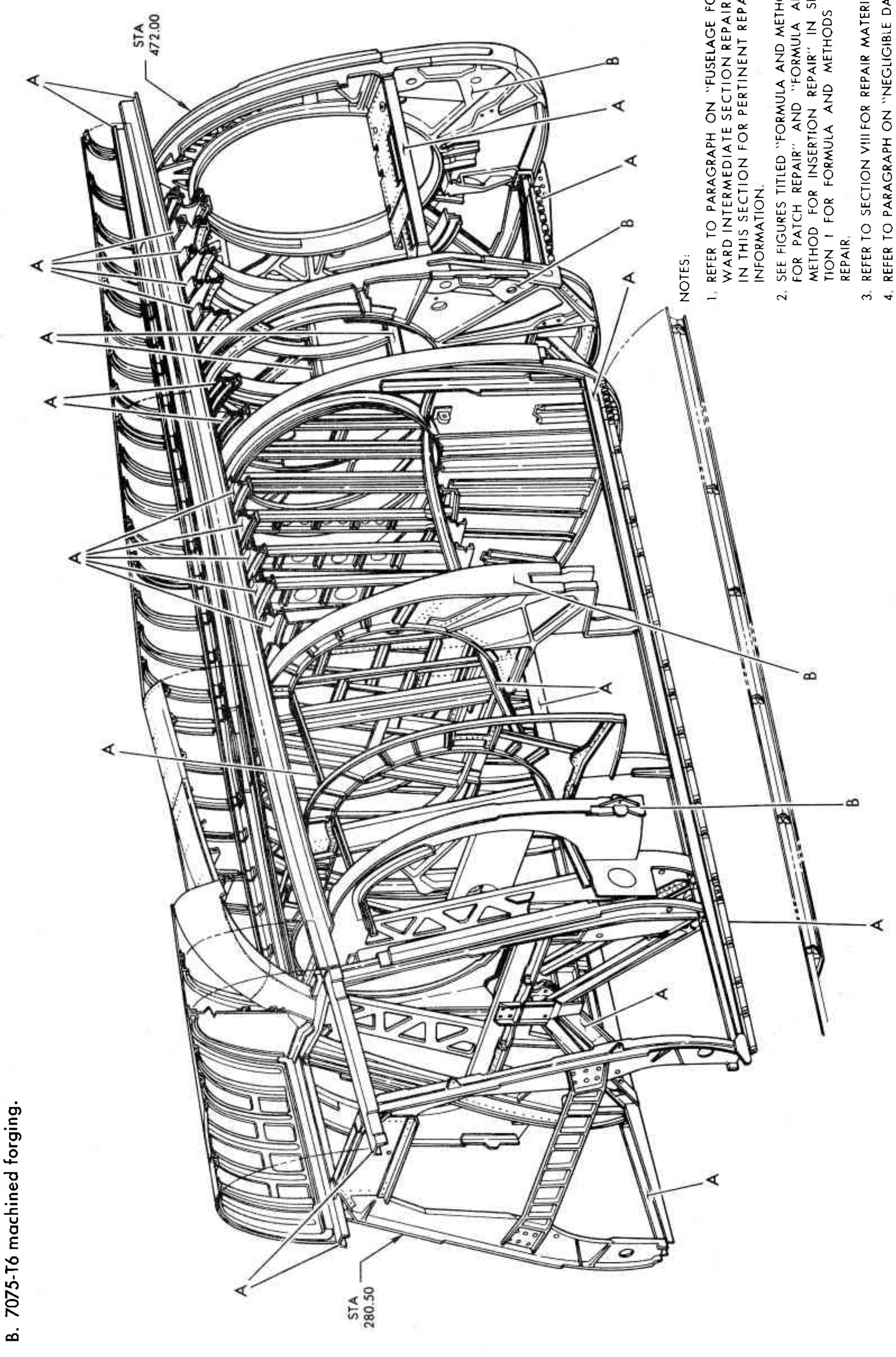
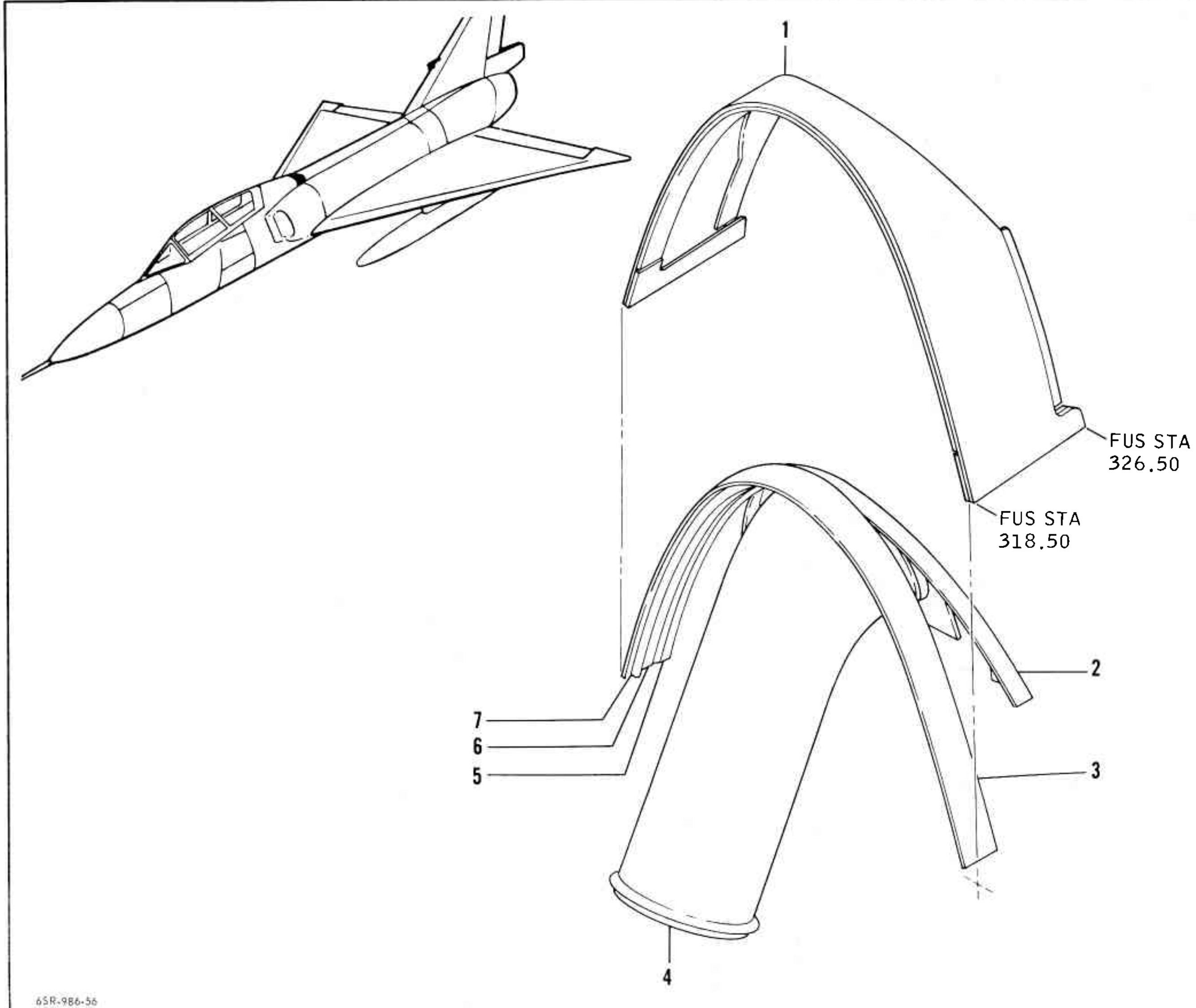


Figure 4-22. Fuselage Structure — Stations 273.00 to 472.00 — F-106B



KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
1	66J39734-2	Skin (Weldment)	0.05 x 9.4 x 52.5	7075-T6	QQ-A-250/12(T6)	4-16F, 10-1
2	66E39736-1	Support	0.032 x 11.7 x 20.5	Cres Typ 302	MIL-S-5059 Cond A	10-10A
3	66E39738-1	Retainer	0.025 x 2.1 x 43.0	Cres Typ 301	MIL-S-5059	**
					Cond 1/4 Hd	
4	66J40237-1	Duct (Weldment)		Cres Typ 321/347	MIL-S-6721	10-10D
5	66E39737-1	Support	0.04 x 2.5 x 43.0	7075-T6	QQ-A-250/12(T6)	10-10C*
6	66J40666-8	Retainer	E491002-24-0 x 53.0	2024-T0	QQ-A-200/3(T0)	Replace
7	66J40666-6	Seal		Silicone Rubber	MIL-R-25988	Replace

*Use portion of repair as required.

**Remove damaged area. Insert new piece picking up three rivets (minimum)

Figure 4-22A. Aerial Refueling Dorsal Section, F-106B, Station 318.5 to 326.5

Applicable after incorporation of TCTO 1F-106-986

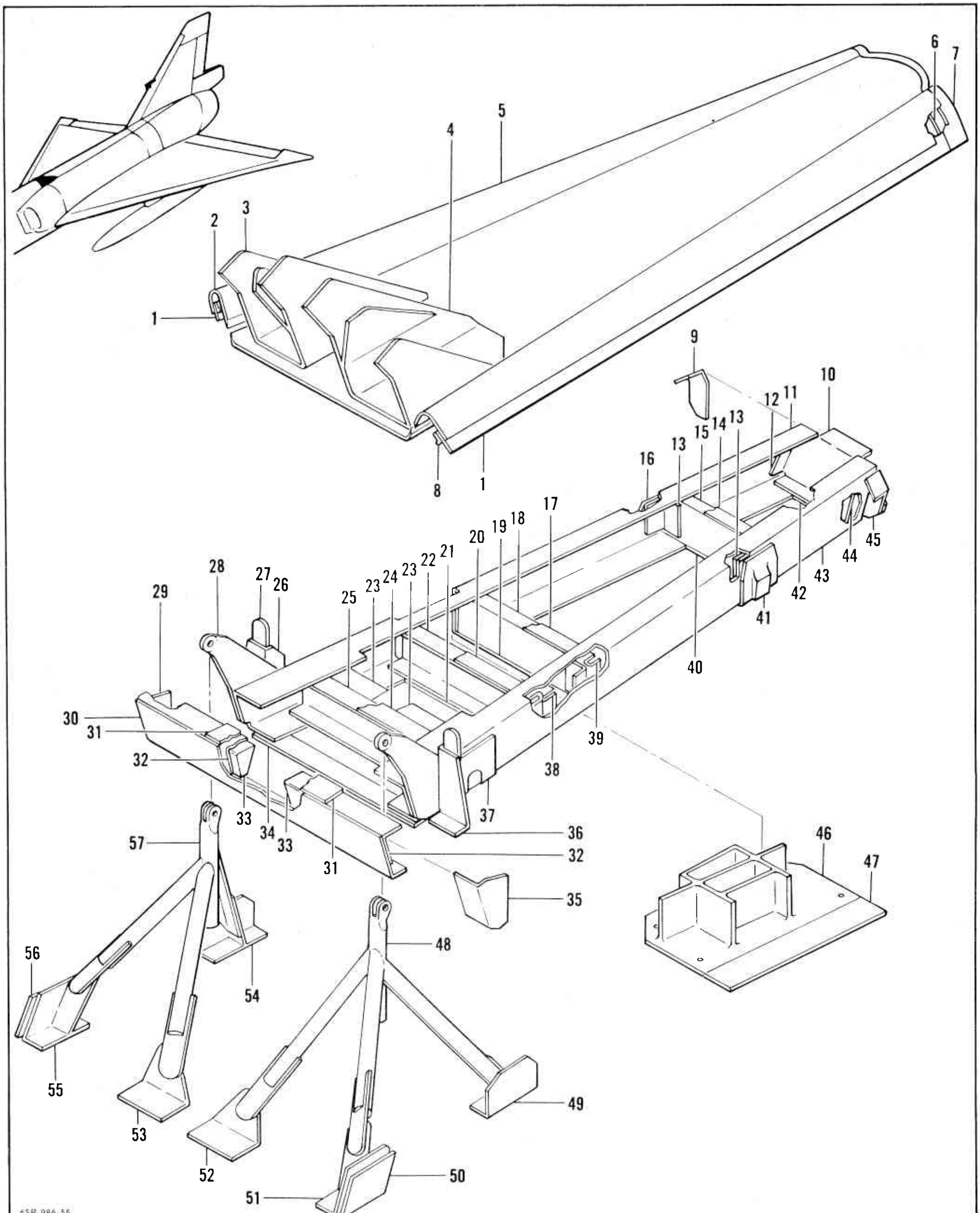


Figure 4-22B. Aerial Refueling Slipway Door and Hinge Supports, F-106B (Sheet 1 of 3)

Applicable after incorporation of TCTO 1F-106-986

KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
1	66J40670-23	Seal	3/32 x 35.0	Syn. Rubber	AMS3208	Replace
2	66J40670-6	Retainer	E491002 x 35.0	2024-T4	QQ-A-200/3(T4)	Replace
3	66E40547-4	Vane (Weldment)		4130 Cond A	MIL-S-18729	10-10B, 10-10E
4	66E40547-3	Vane (Weldment)		4130 Cond A	MIL-S-18729	10-10B, 10-10E
5	66J40372-1	Trough	0.125 x 26.0 x 40.0	Cres Typ 301	MIL-S-5059 Cond 1/4HD	10-1, 10-10B
6	66J40670-10	Retainer	0.04 x 1.5 x 10.0	2024-T3	QQ-A-250/4(T3)	Replace
7	66J40670-24	Seal	3/32 x 35.0	Syn. Rubber	AMS3208	Replace
8	66J40670-5	Retainer	E491002 x 35.0	2024-T4	QQ-A-200/3(T4)	Replace
9	66C40702-4	Attachment	0.071 x 3.0 x 4.5	7075-T6	QQ-A-250/12(T6)	Replace
10	66D40366-1	Rib	0.08 x 4.5 x 7.1	7075-T6	QQ-A-250/12(T6)	10-10A
11	66D40016-1	Channel	0.08 x 6.0 x 32.5	7075-T6	QQ-A-250/12(T6)	10-5, 10-10A,
						10-10C
12	66C40702-6	Attachment	0.071 x 3.0 x 3.5	7075-T6	QQ-A-250/12(T6)	Replace
13	66C40701-1	Attachment	0.071 x 1.8 x 2.6	7075-T6	QQ-A-250/12(T6)	Replace
14	66J40670-18	Filler	0.08 x 0.9 x 3.9	2024-T3	QQ-A-250/4(T3)	Replace
15	66C40683-1	Rib	0.071 x 4.3 x 8.8	7075-T6	QQ-A-250/12(T6)	10-10A
16	66D40261-2	Stop	2.5 x 2.5 x 2.5	7079-T651	QQ-A-250/17(T651)	Replace
17	66J40670-20	Filler	0.08 x 0.9 x 5.9	2024-T3	QQ-A-250/4(T3)	Replace
18	66J40670-19	Filler	0.08 x 0.9 x 4.5	2024-T3	QQ-A-250/4(T3)	Replace
19	66C40682-1	Rib	0.071 x 4.1 x 9.0	7075-T6	QQ-A-250/12(T6)	10-10A
20	66J40670-16	Filler	0.08 x 0.9 x 6.4	2024-T3	QQ-A-250/4(T3)	Replace
21	66J40670-15	Filler	0.08 x 1.9 x 4.8	2024-T3	QQ-A-250/4(T3)	Replace
22	66C40681-1	Rib	0.071 x 4.1 x 9.5	7075-T6	QQ-A-250/12(T6)	10-10A
23	66C40361-1	Fitting	1.5 x 4.0 x 2.25	7079-T651	QQ-A-250/17(T651)	Replace
24	66J40670-22	Filler	0.08 x 1.96 x 6.2	2024-T3	QQ-A-250/4(T3)	Replace
25	66C40371-1	Rib	0.071 x 3.9 x 9.4	7075-T6	QQ-A-250/12(T6)	10-10A
26	66C40698-2	Attachment	0.08 x 2.98 x 4.2	7075-T6	QQ-A-250/12(T6)	Replace
27	66C40370-1	Filler	0.19 x 1.3 x 4.1	2024-T3	QQ-A-250/4(T3)	Replace
28	66D40359-2	Hinge	0.25 x 6.5 x 8.0	4130 Cond MA	MIL-S-18729	Replace
29	66C40697-2	Attachment	0.071 x 4.0 x 4.0	7075-T6	QQ-A-250/12(T6)	Replace
30	66C40365-1	Angle	0.08 x 3.34 x 14.6	7075-T6	QQ-A-250/12(T6)	10-10A, 10-10B
31	66J40670-14	Filler	0.08 x 2.0 x 8.2	2024-T3	QQ-A-250/4(T3)	Replace
32	66C40364-1	Angle	0.08 x 4.0 x 10.4	7075-T6	QQ-A-250/12(T6)	10-10A, 10-10B
33	66C40360-1	Fitting	2.0 x 2.5 x 3.6	7079-T651	QQ-A-250/17(T651)	Replace
34	66J40670-4	Skin	0.05 x 10.1 x 30.3	2024-T3	QQ-A-250/4(T3)	10-2, 10-10B
35	66C40697-1	Attachment	0.071 x 4.0 x 4.0	7075-T6	QQ-A-250/12(T6)	Replace
36	66C40698-1	Attachment	0.08 x 2.98 x 4.2	7075-T6	QQ-A-250/12(T6)	Replace
37	66D40359-1	Hinge	0.25 x 6.5 x 8.0	4130 Cond MA	MIL-S-18729	Replace
38	66C40699-1	Attachment	0.071 x 2.2 x 1.7	7075-T6	QQ-A-250/12(T6)	Replace
39	66C40700-1	Attachment	0.071 x 1.8 x 2.25	7075-T6	QQ-A-250/12(T6)	Replace
40	66J40670-17	Filler	0.08 x 1.2 x 2.9	2024-T3	QQ-A-250/4(T3)	Replace
41	66D40261-1	Stop	2.5 x 2.5 x 2.5	7079-T651	QQ-A-250/17(T651)	Replace
42	66J40670-11	Filler	0.08 x 0.9 x 1.1	2024-T3	QQ-A-250/4(T3)	Replace
43	66E40362-1	Channel	0.08 x 6.0 x 32.5	7075-T6	QQ-A-250/12(T6)	10-5, 10-10A,
						10-10C
44	66C40702-5	Attachment	0.071 x 3.0 x 3.5	7075-T6	QQ-A-250/12(T6)	Replace
45	66C40702-3	Attachment	0.071 x 3.0 x 4.5	7075-T6	QQ-A-250/12(T6)	Replace
46	66E40358-1	Support	4.0 x 6.5 x 10.8	7079-T651	QQ-A-250/17(T651)	Replace
47	66J40670-21	Filler	0.12 x 1.62 x 8.8	2024-T3	QQ-A-250/4(T3)	Replace
48	66J39730-1	Support		4130 Cond N	MIL-S-18729	Replace
		(Weldment)				
49	66D39729-1	Support		4130 Cond N	MIL-S-18729	10-10E
		(Weldment)				
50	66J40264-9	Plate	0.25 x 1.7 x 3.7	4130 Cond N	MIL-S-18729	Replace
51	66D39727-1	Support		4130 Cond N	MIL-S-18729	10-10E
		(Weldment)				
52	66D39728-6	Support		4130 Cond N	MIL-S-18729	10-10E
		(Weldment)				

Figure 4-22B. Aerial Refueling Slipway Door and Hinge Supports, F-106B (Sheet 2 of 3)

Applicable after incorporation of TCTO 1F-106-986

KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
53	66D39728-7	Support		4130 Cond N	MIL-S-18729	10-10E
		(Weldment)				
54	66D39729-2	Support		4130 Cond N	MIL-S-18729	10-10E
		(Weldment)				
55	66D39727-2	Support		4130 Cond N	MIL-S-18729	10-10E
		(Weldment)				
56	66J40264-10	Plate	0.25 x 1.7 x 3.7	4130 Cond N	MIL-S-18729	Replace
57	66J39730-2	Support		4130 Cond N	MIL-S-18729	Replace
		(Weldment)				

Figure 4-22B. Aerial Refueling Slipway Door and Hinge Supports, F-106B (Sheet 3 of 3)

Applicable after incorporation of TCTO 1F-106-986

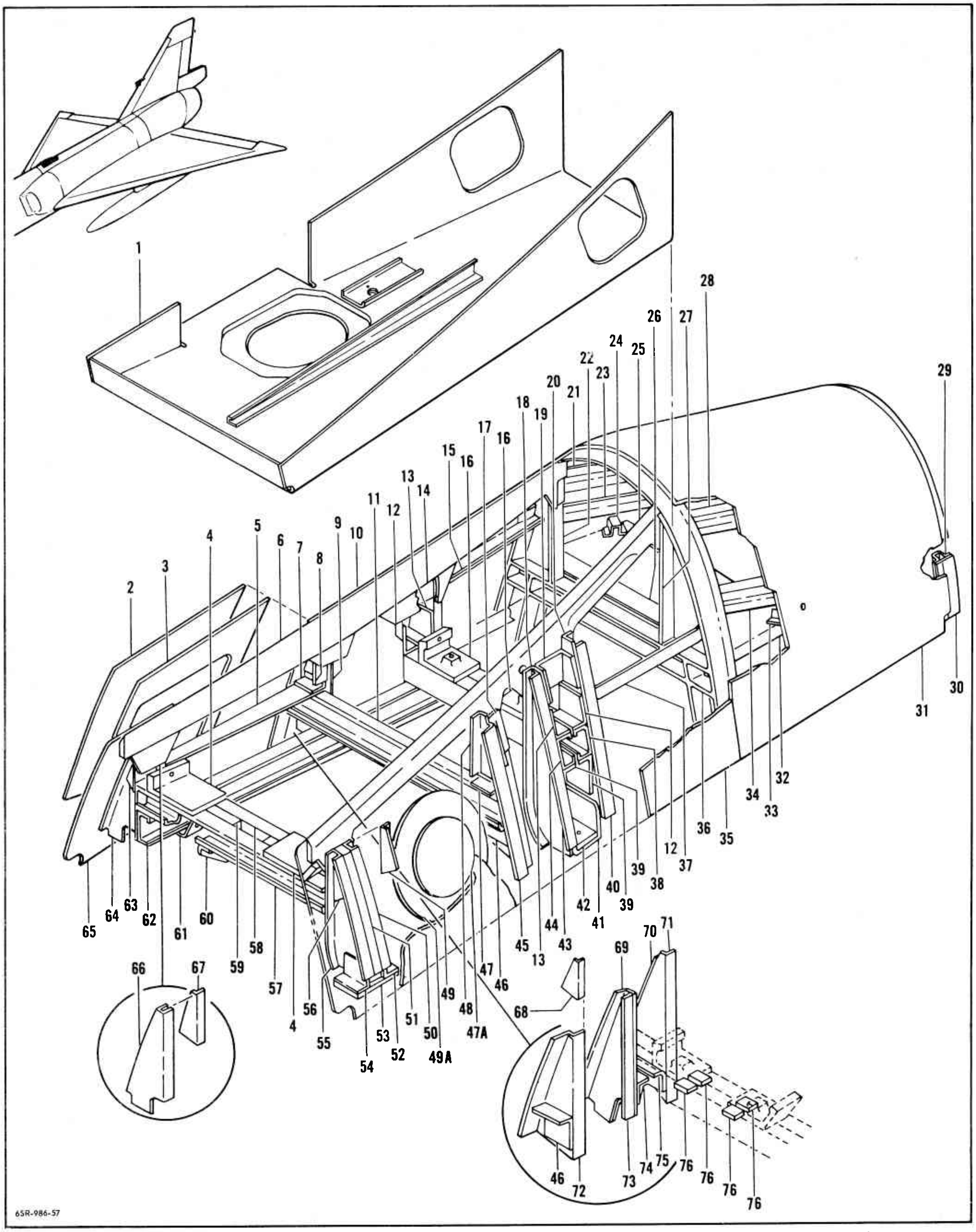


Figure 4-22C. Aerial Refueling Dorsal Sections, F-106B, Stations 325.0 to 377.5 (Sheet 1 of 3)

Applicable after incorporation of TCTO 1F-106-986

KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
1	66J40241-1	Pan (Weldment)		6160-T6	QQ-A-250/11(T6)	10-2, 10-10B,
						10-10E
2	66J39753-3	Skin	0.05 x 11.0 x 12.5	7075-T6	QQ-A-250/12(T6)	10-1
3	66J39753-2	Doubler	0.05 x 11.0 x 12.5	7075-T6	QQ-A-250/12(T6)	*
4	66D39801-1	Support	1.5 x 4.0 x 3.0	7079-T651	QQ-A-250/17(T651)	Replace
5	66C39774-1	Angle	0.08 x 3.5 x 13.0	7075-T6	QQ-A-250/12(T6)	10-10A
6	66J39772-1	Longeron	1.37 x 5.4 x 35.75	7075-T651	QQ-A-250/12(T651)	Replace
7	66C40704-2	Angle	0.08 x 1.9 x 3.7	7075-T6	QQ-A-250/2(T6)	Replace
8	66C40705-1	Angle	0.09 x 3.0 x 4.5	4130	MIL-S-18729 Cond A	Replace
9	66J40666-7	Plate	0.032 x 4.12 x 9.5	7075-T6	QQ-A-250/12(T6)	10-2, 10-10B
10	66J39755-1	Skin (Weldment)		7075-T6	QQ-A-250/12(T6)	4-16G
11	66D39744-4	Channel	0.1 x 4.3 x 18.0	7075-T6	QQ-A-250/12(T6)	10-8
12	66C39763-2	Angle	0.08 x 2.5 x 3.0	7075-T6	QQ-A-250/12(T6)	Replace
13	66C39763-1	Angle	0.08 x 2.5 x 3.0	7075-T6	QQ-A-250/12(T6)	Replace
14	66C39764-2	Channel	0.08 x 1.5 x 4.8	7075-T6	QQ-A-250/12(T6)	Replace
15	66C39777-2	Bracket	0.05 x 2.3 x 10.8	7075-T6	QQ-A-250/12(T6)	10-10A
16	66D39802-1	Support	2.0 x 2.5 x 4.0	7079-T651	QQ-A-250/17(T651)	Replace
17	66C40705-3	Angle	0.063 x 2.4 x 2.4	4130	MIL-S-18729 Cond A	Replace
18	66D39770-1	Support	0.08 x 4.7 x 10.1	7075-T6	QQ-A-250/12(T6)	Replace
19	66C39764-1	Channel	0.08 x 1.5 x 4.8	7075-T6	QQ-A-250/12(T6)	Replace
20	66D39761-1	Support	0.08 x 6.0 x 11.5	7075-T6	QQ-A-250/12(T6)	Replace
21	66C39786-4	Stiffener	0.04 x 3.7 x 9.5	2024-T3	QQ-A-250/4(T3)	10-10A, 10-10C
22	66C39773-2	Angle	0.10 x 4.5 x 8.8	7075-T6	QQ-A-250/12(T6)	10-10A
23	66C39786-2	Stiffener	0.04 x 3.7 x 7.8	2024-T3	QQ-A-250/4(T3)	10-10A, 10-10C
24	66C39853-2	Bracket	0.04 x 2.0 x 2.1	2024-T3	QQ-A-250/4(T3)	Replace
25	66J39771-1	Longeron	0.10 x 4.5 x 8.8	7075-T651	QQ-A-250/12(T651)	Replace
26	66C39787-1	Plate	0.1 x 1.6 x 16.4	7075-T6	QQ-A-250/12(T6)	Replace
27	66C39773-1	Angle	0.10 x 4.5 x 8.8	7075-T6	QQ-A-250/12(T6)	10-10A
28	66C39786-3	Stiffener	0.04 x 3.7 x 9.5	2024-T3	QQ-A-250/4(T3)	10-10A, 10-10C
29	66E39750-1	Support	0.04 x 3.0 x 40.0	7075-T6	QQ-A-250/12(T6)	10-10C
30	66J40666-16	Doubler	0.04 x 2.0 x 47.0	7075-T6	QQ-A-250/13(T6)	**
31	66J39751-1	Skin (Weldment)		7075-T6	QQ-A-250/12(T6)	10-10E
32	66E39783-1	Pan	0.05 x 6.8 x 24.7	7075-T6	QQ-A-250/12(T6)	10-2, 10-10B,
						10-10E
33	66C39853-1	Bracket	0.04 x 2.0 x 2.1	2024-T3	QQ-A-250/4(T3)	Replace
34	66C39786-1	Stiffener	0.04 x 3.7 x 7.8	2024-T3	QQ-A-250/4(T3)	10-10A, 10-10C
35	66D39752-1	Skin (Weldment)	0.05 x 15.0 x 35.0	7075-T6	QQ-A-250/12(T6)	10-1
36	66J40260-1	Dorsal Sect	2.5 x 18.0 x 19.0	7079-T651	QQ-A-250/17(T651)	10-9
		(Machined)				
37	66C39777-1	Bracket	0.05 x 3.0 x 11.5	7075-T6	QQ-A-250/12(T6)	10-10A
38	66D39744-2	Channel	0.1 x 4.3 x 18.0	7075-T6	QQ-A-250/12(T6)	10-8
39	66C40703-1	Angle	0.08 x 2.9 x 2.9	7075-T6	QQ-A-250/12(T6)	Replace
40	66D39762-1	Support	0.08 x 5.0 x 10.7	7075-T6	QQ-A-250/12(T6)	Replace
41	66D39766-2	Channel	0.9 x 3.75 x 7.0	7075-T6	QQ-A-250/4(T6)	Replace
42	66C39781-1	Plate	0.371 x 2.0 x 1.7	7075-T6	QQ-A-250/4(T6)	Replace
43	66D39769-1	Support	0.08 x 4.5 x 10.1	7075-T6	QQ-A-250/12(T6)	Replace
44	66D39744-3	Channel	0.1 x 4.3 x 18.2	7075-T6	QQ-A-250/12(T6)	10-8
45	66D39749-1	Support	0.08 x 6.5 x 10.0	7075-T6	QQ-A-250/12(T6)	10-10A
46	66C40703-2	Angle	0.08 x 2.5 x 3.0	7075-T6	QQ-A-250/2(T6)	Replace
47	66C40704-1	Angle	0.08 x 1.9 x 3.7	7075-T6	QQ-A-250/2(T6)	Replace
47A	66J39752-6	Door	0.1 x 5.0 x 5.0	7075-T6	QQ-A-250/12(T6)	Replace
48	66C40705-2	Angle	0.09 x 3.0 x 4.5	4130	MIL-S-18729 Cond A	Replace
49	66C40706-1	Angle	0.063 x 2.7 x 4.0	4130	MIL-S-18729 Cond A	Replace
49A	66J39752-5	Retainer	0.063 x 9.0 x 9.0	7075-T6	QQ-A-250/12(T6)	Replace
50	66D39779-1	Support	0.08 x 4.8 x 7.8	7075-T6	QQ-A-250/12(T6)	Replace

*Stop drill cracks where 1.0 edge distance.

**Remove damaged area. Insert new piece picking up four rivets (minimum).

Figure 4-22C. Aerial Refueling Dorsal Sections, F-106B, Stations 325.0 to 377.5 (Sheet 2 of 3)

Applicable after incorporation of TCTO 1F-106-986

KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
51	66D39745-1	Stiffener	0.75 x 8.0 x 9.0	7075-T651	QQ-A-250/12(T651)	Replace
52	66C39746-1	Angle	AND10133-1403 x	7075-T6511	QQ-A-200/11(T6511)	Replace
			1.65			
53	66C39747-1	Plate	0.5 x 1.8 x 3.0	7075-T651	QQ-A-250/12(T651)	Replace
54	66C39746-2	Angle	AND10133-1403 x	7075-T6511	QQ-A-250/11(T6511)	Replace
			1.65			
55	66D39780-1	Support	0.08 x 4.8 x 7.8	7075-T6	QQ-A-250/12(T6)	Replace
56	66D39778-1	Bracket	0.063 x 3.5 x 5.0	4130 Cond A	MIL-S-18729	Replace
57	66D39744-1	Channel	0.1 x 4.4 x 15.8	7075-T6	QQ-A-250/12(T6)	10-8
58	66C40708-1	Channel	0.1 x 4.8 x 15.4	7075-T6	QQ-A-250/4(T6)	Replace
59	66D39743-1	Stiffener	0.75 x 8.5 x 11.0	7075-T651	QQ-A-250/12(T651)	Replace
60	66J40666-14	Bracket	0.10 x 3.25 x 3.62	7075-T6	QQ-A-250/12(T6)	Replace
61	66J40666-18	Clip	0.09 x 1.5 x 2.56	7075-T6	QQ-A-250/12(T6)	Replace
62	66C40707-1	Channel	0.1 x 5.9 x 23.3	7075-T6	QQ-A-250/4(T6)	10-8
63	66D39778-2	Bracket	0.063 x 3.5 x 5.0	4130 Cond A	MIL-S-18729	Replace
64	66D39742-1	Support	0.08 x 8.0 x 9.0	7075-T6	QQ-A-250/12(T6)	10-10A
65	66E39754-1	Skin (Weldment)		7075-T6	QQ-A-250/12(T6)	4-16G
66	66D39741-1	Support	0.08 x 6.5 x 8.5	7075-T6	QQ-A-250/12(T6)	10-10A
67	66C40706-2	Angle	0.063 x 2.7 x 4.0	4130	MIL-S-18729 Cond A	Replace
68	66C40705-4	Angle	0.063 x 2.4 x 2.4	4130	MIL-S-18729 Cond A	Replace
69	66D39768-1	Support	0.08 x 5.5 x 11.0	7075-T6	QQ-A-250/12(T6)	Replace
70	66D39759-1	Support	0.08 x 4.7 x 10.7	7075-T6	QQ-A-250/12(T6)	Replace
71	66D39760-1	Support	0.08 x 4.8 x 10.7	7075-T6	QQ-A-250/12(T6)	Replace
72	66D39748-1	Support	0.08 x 7.0 x 10.0	7075-T6	QQ-A-250/12(T6)	10-10A
73	66D39767-1	Support	0.08 x 5.5 x 11.0	7075-T6	QQ-A-250/12(T6)	Replace
74	66C39757-1	Angle	0.08 x 2.3 x 3.3	7075-T6	QQ-A-250/12(T6)	Replace
75	66C39756-1	Angle	0.08 x 2.3 x 3.3	7075-T6	QQ-A-250/12(T6)	Replace
76	66J40666-12	Plate	0.025 x 0.75 x 1.2	7075-T651	QQ-A-250/12(T651)	Replace

Figure 4-22C. Aerial Refueling Dorsal Sections, F-106B, Stations 325.0 to 377.5 (Sheet 3 of 3)

Applicable after incorporation of TCTO 1F-106-986

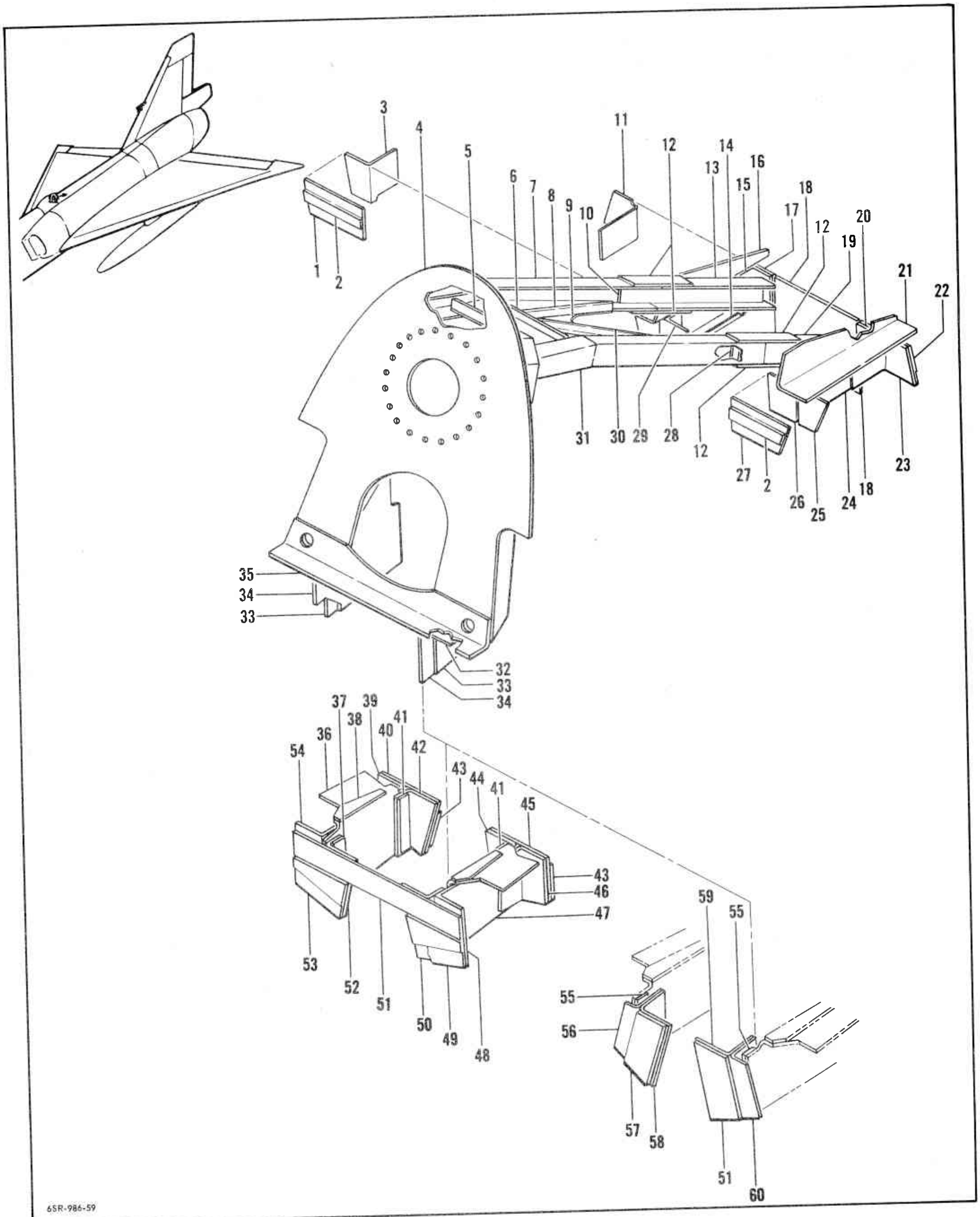


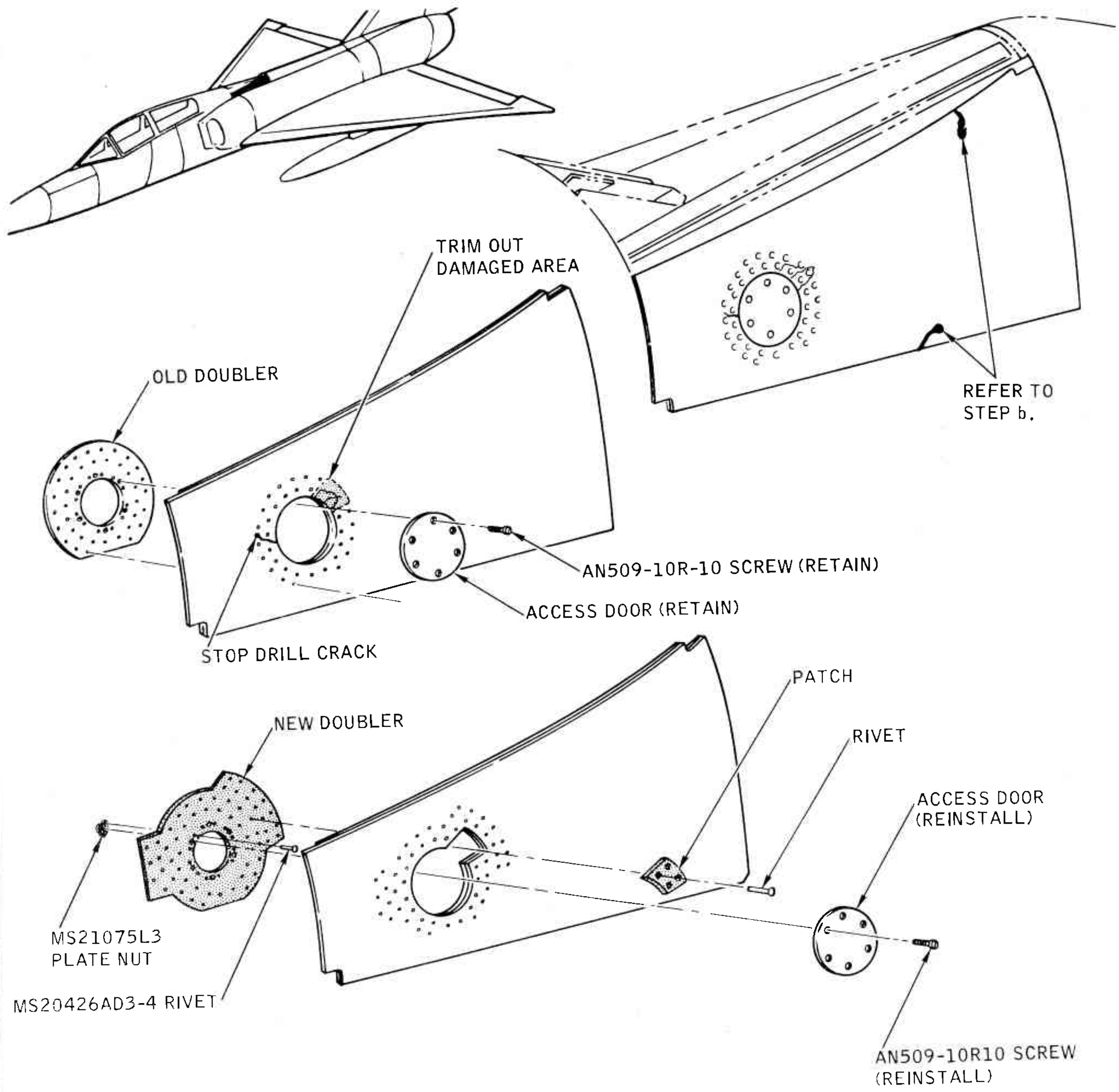
Figure 4-22D. Aerial Refueling Receptacle Support, F-106B, (Sheet 1 of 3)
Applicable after incorporation of TCTO 1F-106-986

KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
1	66J40257-21	Doubler	0.125 x 2.3 x 4.0	7075-T6	QQ-A-250/12(T6)	Replace
2	66J40257-2	Filler	0.071 x 0.9 x 3.7	7075-T6	QQ-A-250/12(T6)	Replace
3	66J40257-12	Angle	0.08 x 2.25 x 3.65	7075-T6	QQ-A-250/12(T6)	Replace
4	66J40405-1	Support (Weldment)		6061-T651	QQ-A-250/11(T651)	10-10E
5	66J40257-22	Angle	0.05 x 1.2 x 7.6	7075-T6	QQ-A-250/12(T6)	Replace
6	66J40257-33	Angle	0.05 x 1.3 x 7.6	7075-T6	QQ-A-250/12(T6)	Replace
7	66E39850-2	Support (Weldment)		4130 Cond A	MIL-S-18729	10-10E
8	66J40257-25	Angle	0.05 x 1.6 x 11.6	7075-T6	QQ-A-250/12(T6)	10-10A
9	66J40257-26	Web	0.032 x 7.6 x 14.9	7075-T6	QQ-A-250/12(T6)	10-2, 10-10B
10	66D39849-2	Spacer	0.19 x 1.3 x 12.2	4130 Cond A	MIL-S-18729 Cond A	Replace
11	66J40257-10	Angle	0.08 x 2.35 x 4.07	7075-T6	QQ-A-250/12(T6)	Replace
12	66J40257-41	Plate	0.019 x 1.5 x 6.6	4130 Cond A	MIL-S-18729 Cond A	Replace
13	66J40257-43	Channel	0.125 x 3.5 x 9.1	4130 Cond A	MIL-S-18729 Cond A	Replace
14	66J40257-4	Intercostal	0.1 x 4.9 x 6.3	7075-T6	QQ-A-250/12(T6)	Replace
15	66J40257-8	Angle	0.08 x 2.2 x 4.78	7075-T6	QQ-A-250/12(T6)	Replace
16	66J40257-18	Angle	0.125 x 8.5 x 11.5	7075-T6	QQ-A-250/12(T6)	Replace
17	66C39830-2	Plate	0.08 x 2.25 x 4.0	7075-T6	QQ-A-250/12(T6)	Replace
18	66J40257-40	Doubler	0.063 x 4.5 x 4.6	7075-T6	QQ-A-250/12(T6)	Replace
19	66J40257-42	Channel	0.125 x 3.5 x 9.1	4130 Cond A	MIL-S-18729 Cond A	Replace
20	66J40257-7	Angle	0.08 x 3.0 x 5.0	7075-T6	QQ-A-250/12(T6)	Replace
21	66J40257-17	Angle	0.125 x 8.5 x 11.5	7075-T6	QQ-A-250/12(T6)	Replace
22	66C39830-1	Plate	0.08 x 2.25 x 3.875	7075-T6	QQ-A-250/12(T6)	Replace
23	66J40257-9	Angle	0.08 x 2.35 x 4.07	7075-T6	QQ-A-250/12(T6)	Replace
24	66J40257-3	Intercostal	0.1 x 6.0 x 6.3	7075-T6	QQ-A-250/12(T6)	Replace
25	66J40257-11	Angle	0.08 x 2.25 x 3.65	7075-T6	QQ-A-250/12(T6)	Replace
26	66J40257-5	Angle	0.08 x 2.3 x 3.5	7075-T6	QQ-A-250/12(T6)	Replace
27	66J40257-20	Doubler	0.125 x 2.3 x 4.0	7075-T6	QQ-A-250/12(T6)	Replace
28	66D39849-1	Spacer	0.19 x 1.3 x 12.2	4130 Cond A	MIL-S-18729 Cond A	Replace
29	66J40257-6	Angle	0.08 x 2.3 x 3.5	7075-T6	QQ-A-250/12(T6)	Replace
30	66J40257-24	Angle	0.05 x 1.6 x 11.6	7075-T6	QQ-A-250/12(T6)	10-10A
31	66E39850-1	Support (Weldment)		4130 Cond A	MIL-S-18729	10-10E
32	66J40257-37	Filler	0.188 x 0.8 x 12.2	7075-T6	QQ-A-250/12(T6)	Replace
33	66J40257-44	Plate	0.25 x 4.5 x 4.8	6061-T6	QQ-A-250/11(T6)	Replace
34	66J40257-45	Plate	0.25 x 6.0 x 9.0	7075-T6	QQ-A-250/12(T6)	Replace
35	66J40257-38	Angle	0.125 x 3.5 x 14.5	7075-T6	QQ-A-250/12(T6)	10-10A
36	66J40257-28	Intercostal	0.08 x 6.0 x 6.9	7075-T6	QQ-A-250/12(T6)	Replace
37	66J40257-39	Filler	0.16 x 0.8 x 4.2	7075-T6	QQ-A-250/12(T6)	Replace
38	66C39832-1	Plate	0.04 x 0.7 x 3.6	7075-T6	QQ-A-250/12(T6)	Replace
39	66J40257-16	Angle	0.08 x 3.3 x 3.7	7075-T6	QQ-A-250/12(T6)	Replace
40	66J40257-34	Doubler	0.063 x 3.7 x 3.9	7075-T6	QQ-A-250/12(T6)	Replace
41	66J40257-19	Filler	0.16 x 0.7 x 3.6	7075-T6	QQ-A-250/12(T6)	Replace
42	66J40257-32	Angle	0.08 x 2.8 x 3.9	7075-T6	QQ-A-250/12(T6)	Replace
43	66J40257-35	Filler	0.05 x 3.6 x 3.7	7075-T6	QQ-A-250/12(T6)	Replace
44	66J40257-31	Angle	0.08 x 2.8 x 3.9	7075-T6	QQ-A-250/12(T6)	Replace
45	66J40257-33	Doubler	0.063 x 3.7 x 3.9	7075-T6	QQ-A-250/12(T6)	Replace
46	66J40257-15	Angle	0.08 x 3.3 x 3.7	7075-T6	QQ-A-250/12(T6)	Replace
47	66J40257-27	Intercostal	0.08 x 6.0 x 6.9	7075-T6	QQ-A-250/12(T6)	Replace
48	66J40257-13	Angle	0.08 x 3.8 x 4.1	7075-T6	QQ-A-250/12(T6)	Replace
49	66C39829-1	Plate	0.19 x 2.4 x 3.6	7075-T6	QQ-A-250/12(T6)	Replace
50	66J40257-29	Angle	0.08 x 3.0 x 4.3	7075-T6	QQ-A-250/12(T6)	Replace
51	66J40257-36	Filler	0.125 x 1.7 x 12.8	7075-T6	QQ-A-250/12(T6)	Replace
52	66J40257-30	Angle	0.08 x 3.0 x 4.3	7075-T6	QQ-A-250/12(T6)	Replace
53	66C39829-2	Plate	0.19 x 2.4 x 3.6	7075-T6	QQ-A-250/12(T6)	Replace
54	66J40257-14	Angle	0.08 x 3.8 x 4.1	7075-T6	QQ-A-250/12(T6)	Replace
55	66J40257-52	Filler	0.25 x 0.8 x 3.94	7075-T6	QQ-A-250/12(T6)	Replace
56	66J40257-48	Angle	0.08 x 4.0 x 5.0	7075-T6	QQ-A-250/12(T6)	Replace
57	66J40257-51	Filler	0.1 x 2.05 x 5.0	7075-T6	QQ-A-250/13(T6)	Replace
58	66J40257-50	Angle	0.08 x 4.5 x 5.0	7075-T6	QQ-A-250/12(T6)	Replace

Figure 4-22D. Aerial Refueling Receptacle Support, F-106B, (Sheet 2 of 3)
Applicable after incorporation of TCTO 1F-106-986

KEY NO.	DRAWING NUMBER	NAME	STOCK SIZE	MATERIAL	MATERIAL SPECIFICATION	REPAIR FIGURE
59	66J40257-49	Angle	0.08 x 4.5 x 5.0	7075-T6	QQ-A-250/12(T6)	Replace
60	66J40257-47	Angle	0.08 x 4.0 x 5.0	7075-T6	QQ-A-250/12(T6)	Replace

Figure 4-22D. Aerial Refueling Receptacle Support, F-106B, (Sheet 3 of 3)
Applicable after incorporation of TCTO 1F-106-986



REPAIR PROCEDURE

a. Skin areas not covering structure doublers may be repaired using flush patch skin repair in Section X.

b. Skin upper edge cracks or cuts not exceeding 1.2 inches length and lower edge cracks not exceeding 0.75 inch (1-1/2D edge distance minimum from nearest attachment), stop drill and smooth out surface and edges.

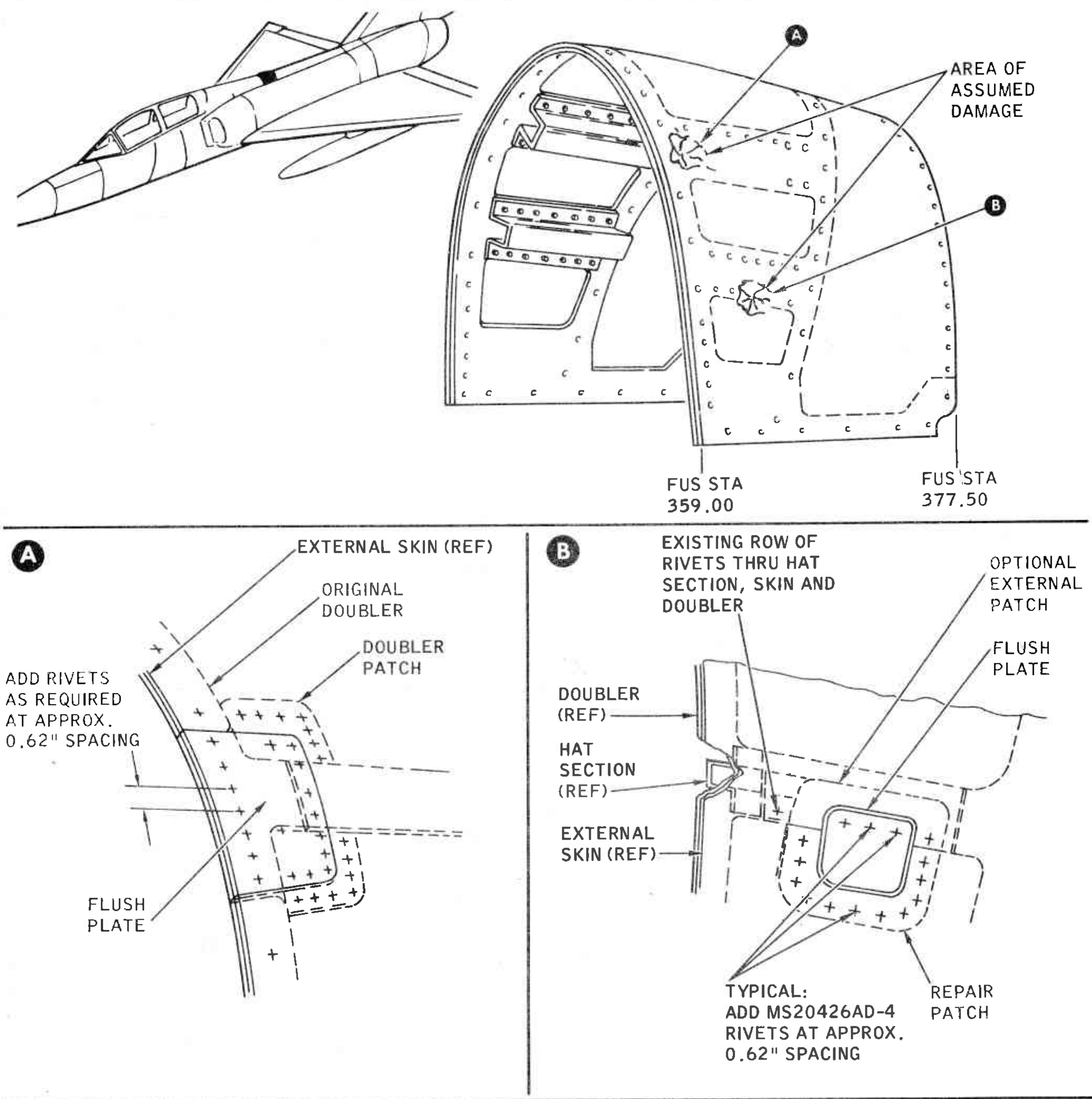
c. For access hole repair, remove old doubler. Trim out damaged area or stop drill crack. Fabricate doubler one gage heavier and patch of same material and gage as original parts. Doubler rivet pattern to be two rivets beyond damaged area in all directions.

d. Refer to Section I for surface treatment of repair parts and sealing.

e. Skin replacement may be accomplished by installing a 0.100 inch thick 7075-T7 skin in place of present skin and doubler assembly. This removes skin and doubler spot weld requirement.

65R-986-69

Figure 4-22E. Dorsal Skin Repair, F-106B, Stations 325.0 to 359.0
Applicable after incorporation of TCTO 1F-106-986



REPAIR PROCEDURE

a. Skin areas not covering structure or doublers may be repaired using flush patch skin repair in this section, except use single row of MS20426AD-4 rivets at approximately 0.62 spacing.

b. Skin edge cracks not exceeding 0.62 inch in length (1.0 edge distance minimum from nearest attachment) may be stop drilled with surfaces and edges smoothed out.

c. Trim out damaged area of skin and doubler. Fabricate patch and plate of same material and gage as original parts.

d. Refer to Section I for surface treatment of repair parts.

NOTE

EXTERNAL PATCH, IN LIEU OF INTERNAL PATCH, MAY BE USED WHERE INSTALLATION DIFFICULTY IS ENCOUNTERED. OVERLAP THE EXTERNAL PATCH ALL AROUND CUTOUT EXCEPT FOR EDGE OF SKIN; CHAMFER EDGES OF PATCH 45 DEGREES. INSTALL FLUSH PATE WHERE NEEDED AS FILLER.

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Figure 4-22F. Dorsal Skin Repair, F-106B, Station 359.0 to 377.5

Applicable after incorporation of TCTO 1F-106-986

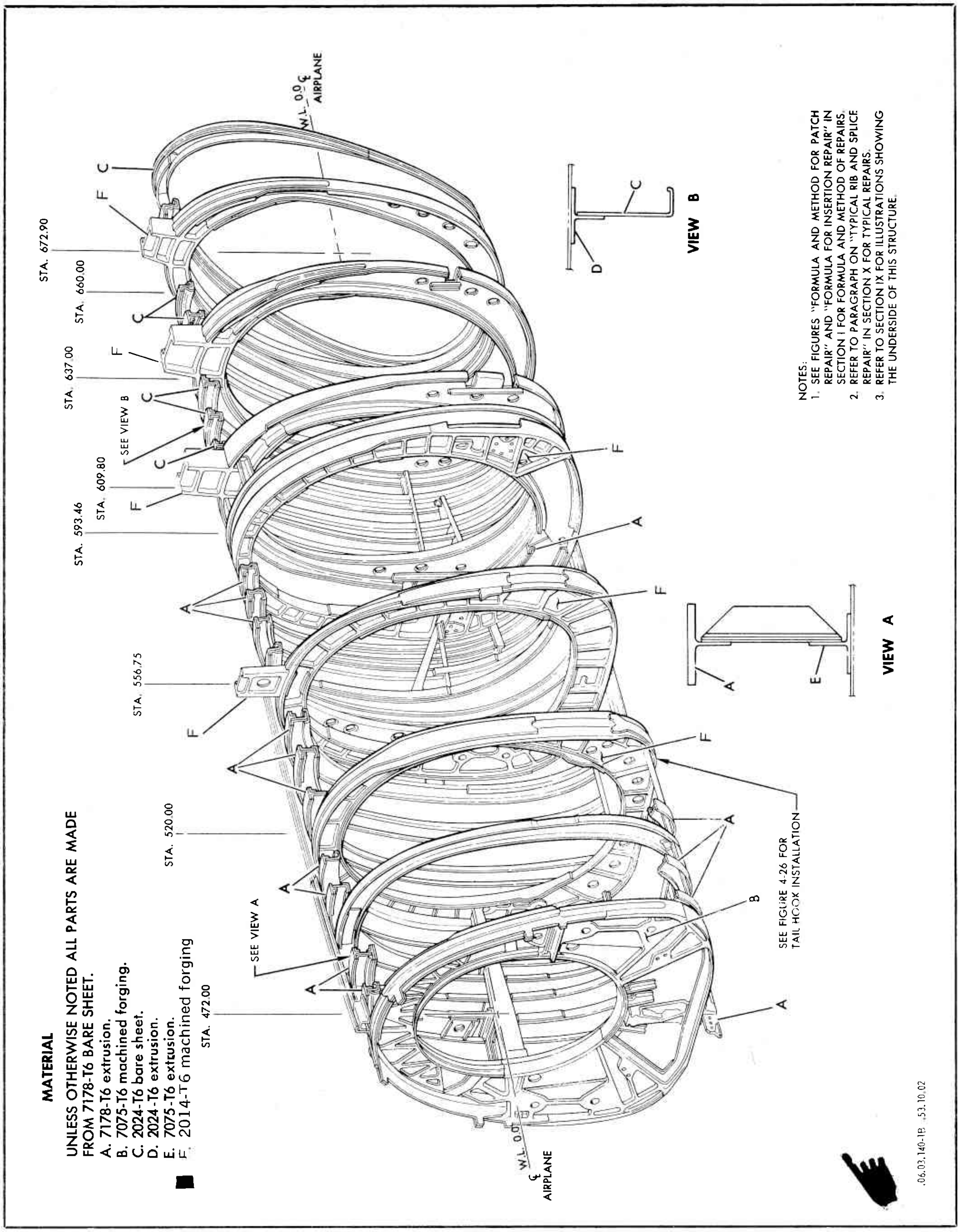
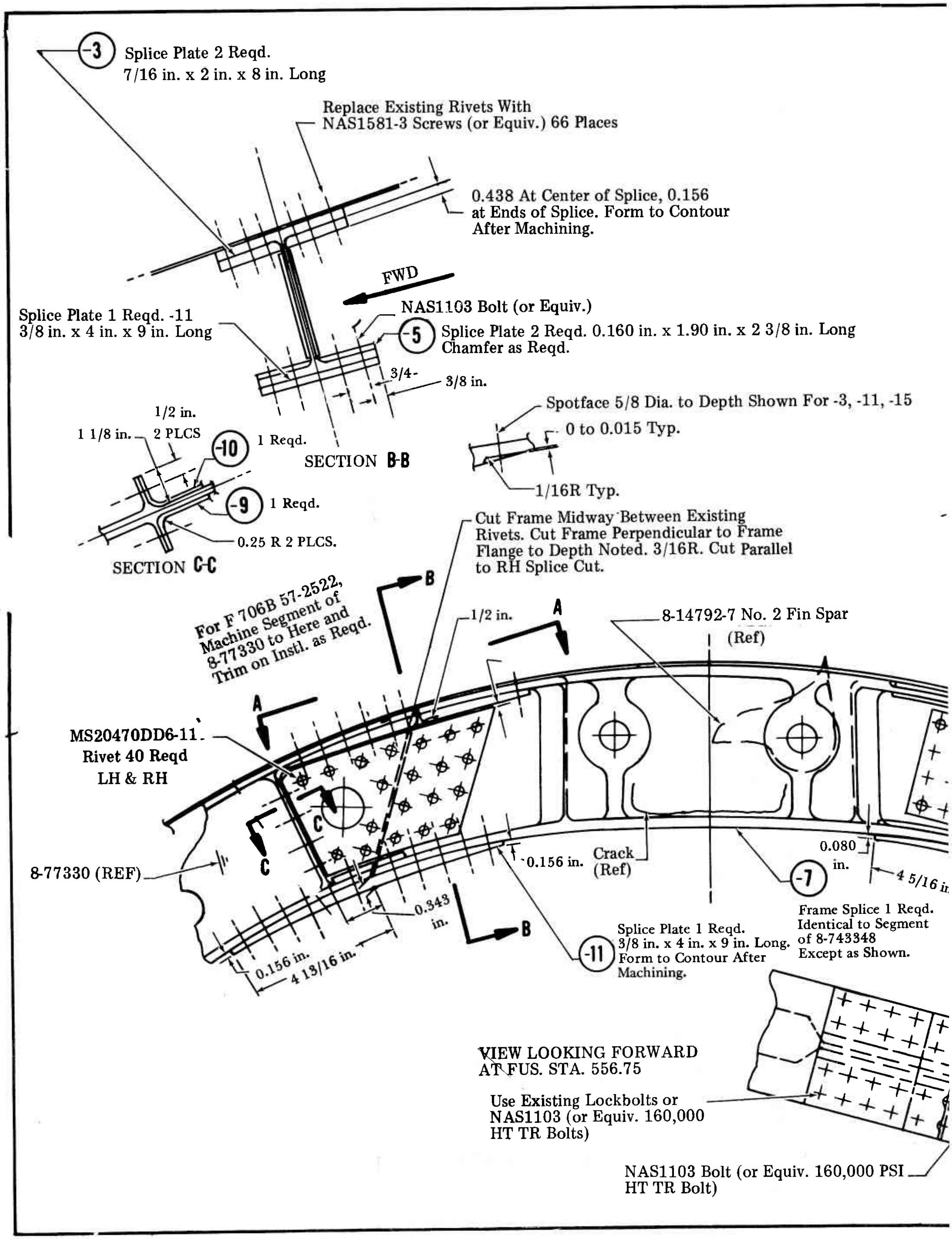


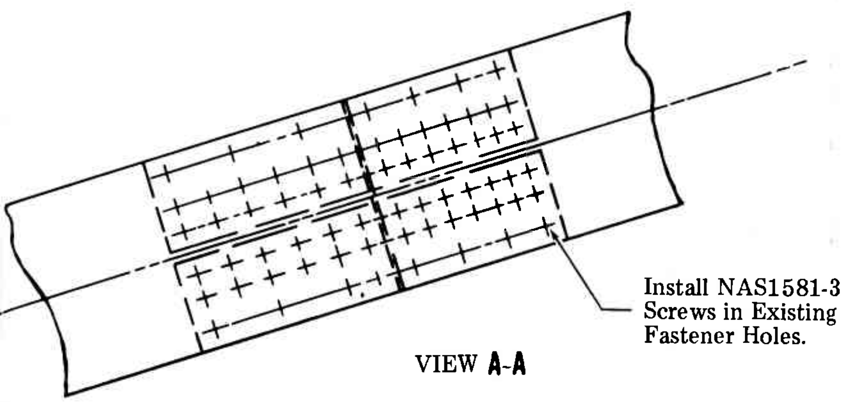
Figure 4-23. Fuselage Structure — Stations 472.00 to 672.00

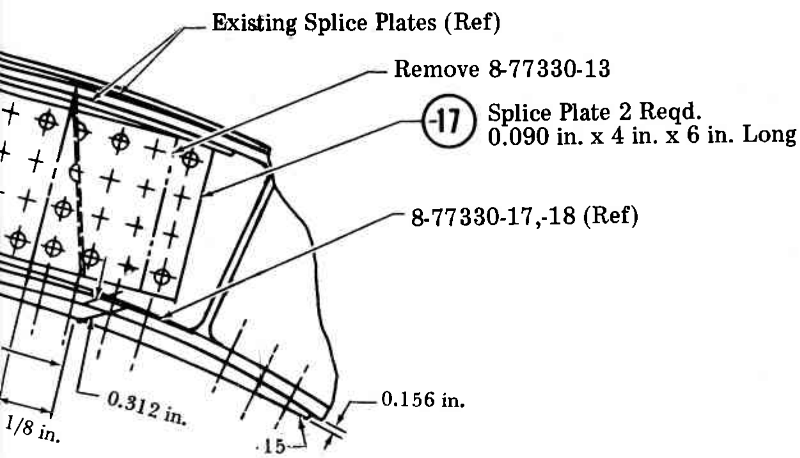


-3 Splice Plate 2 Reqd.
7/16 in. x 2 in. x 8 in. Long
Replace Existing Rivets With NAS1581-3 Screws (or Equiv.) 66 Places
0.438 At Center of Splice, 0.156 at Ends of Splice. Form to Contour After Machining.
FWD
Splice Plate 1 Reqd. -11
3/8 in. x 4 in. x 9 in. Long
NAS1103 Bolt (or Equiv.)
-5 Splice Plate 2 Reqd. 0.160 in. x 1.90 in. x 2 3/8 in. Long
Chamfer as Reqd.
3/4
3/8 in.
Spotface 5/8 Dia. to Depth Shown For -3, -11, -15
0 to 0.015 Typ.
1/16R Typ.
SECTION B-B
1/2 in.
1 1/8 in.
2 PLCS
-10 1 Reqd.
-9 1 Reqd.
0.25 R 2 PLCS.
SECTION C-C
Cut Frame Midway Between Existing Rivets. Cut Frame Perpendicular to Frame Flange to Depth Noted. 3/16R. Cut Parallel to RH Splice Cut.
For F 706B 57-2522, Machine Segment of 8-77330 to Here and Trim on Instl. as Reqd.
B
A
1/2 in.
8-14792-7 No. 2 Fin Spar (Ref)
MS20470DD6-11 Rivet 40 Reqd LH & RH
C
8-77330 (REF)
0.156 in.
Crack (Ref)
0.080 in.
4 5/16 in.
-7
Frame Splice 1 Reqd. Identical to Segment of 8-743348 Except as Shown.
0.343 in.
0.156 in.
4 13/16 in.
-11 Splice Plate 1 Reqd. 3/8 in. x 4 in. x 9 in. Long. Form to Contour After Machining.
VIEW LOOKING FORWARD AT FUS. STA. 556.75
Use Existing Lockbolts or NAS1103 (or Equiv. 160,000 HT TR Bolts)
NAS1103 Bolt (or Equiv. 160,000 PSI HT TR Bolt)

NOTES

1. 1/4 Corner Radius for all Holes Near Edge of Part.
2. Finish for -7, -9, -10, -17, -3, -5, -11 Per T.O. 1F-106A-23.
3. Dip all Fasteners in MIL-P-23377 & Install While Wet.
4. Scale Dimensions not Shown.
5. 3/8 E.D. & 3/4 Between Fasteners in -9, -10, -17.
6. Install .063 2024-T3 Shim Under Upper Part of -11 to Fill Gap Caused by Mismatch of 2 Frames.
7. Increase Length of Insulation Blanket Standoffs as Reqd.
8. -7 Frame Splice May be Made From 2014-T6 Hand Forging Per Spec MIL-A-22771 or 2014-T6 Bar Per Spec QQ-A-200/2.
9. -3, -5, -11, -15 & -17 Splice Plates May be Made From 7075-T73 AL Alloy Per Spec QQ-A-250/12 or 24.
10. -9 and -10 Angles May be Made From 2024-T^{O} AL Alloy (Heat Treat to T-81) Per Spec QQ-A-250/5.





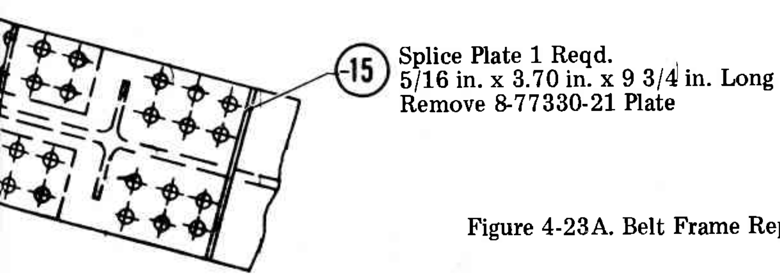
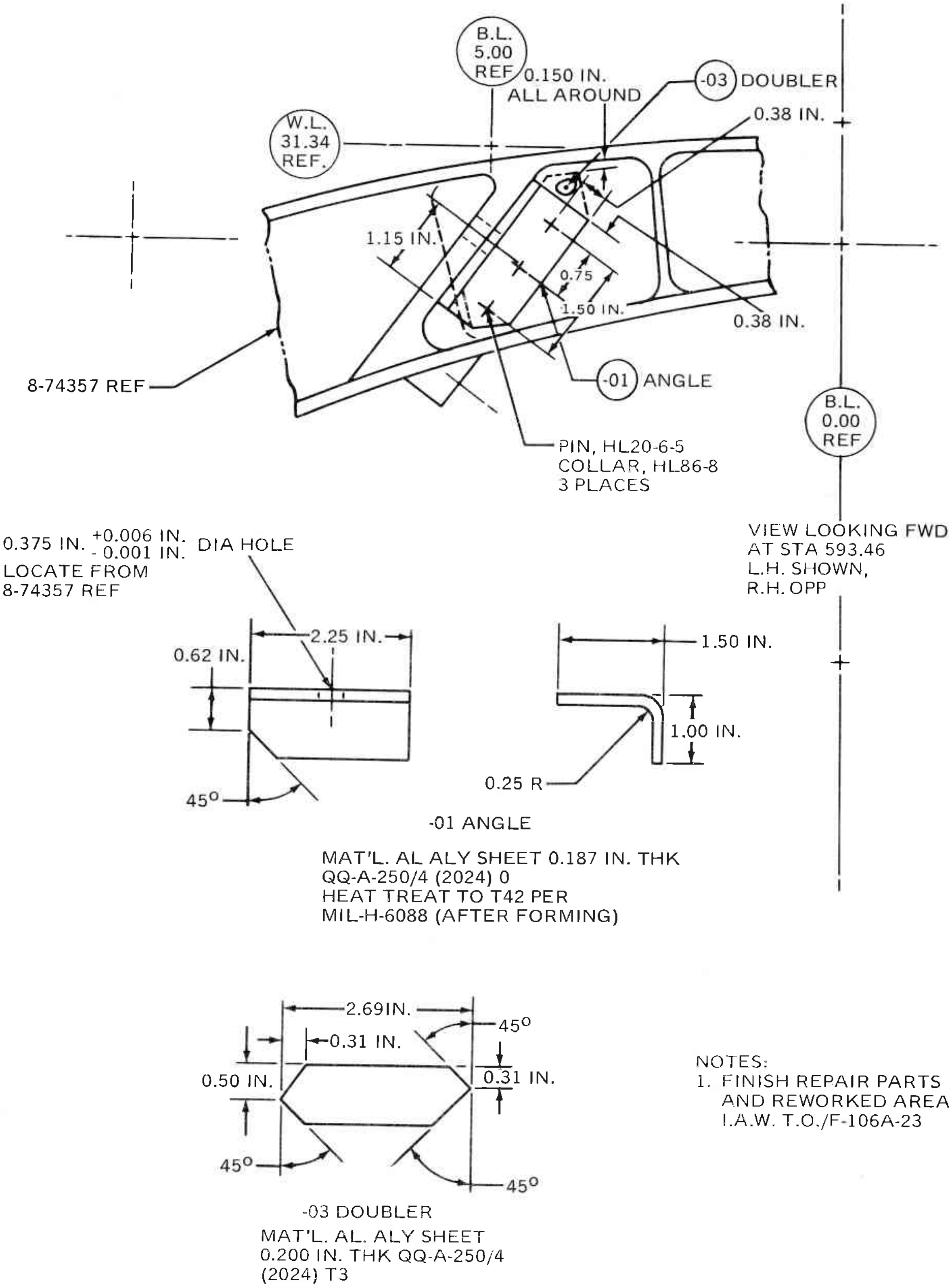


Figure 4-23A. Belt Frame Repair (Depot) Fuselage Station 556.75



NOTES:
1. FINISH REPAIR PARTS AND REWORKED AREA I.A.W. T.O./F-106A-23

Figure 4-23B. Repair of Fuselage Bulkhead Station 593.46

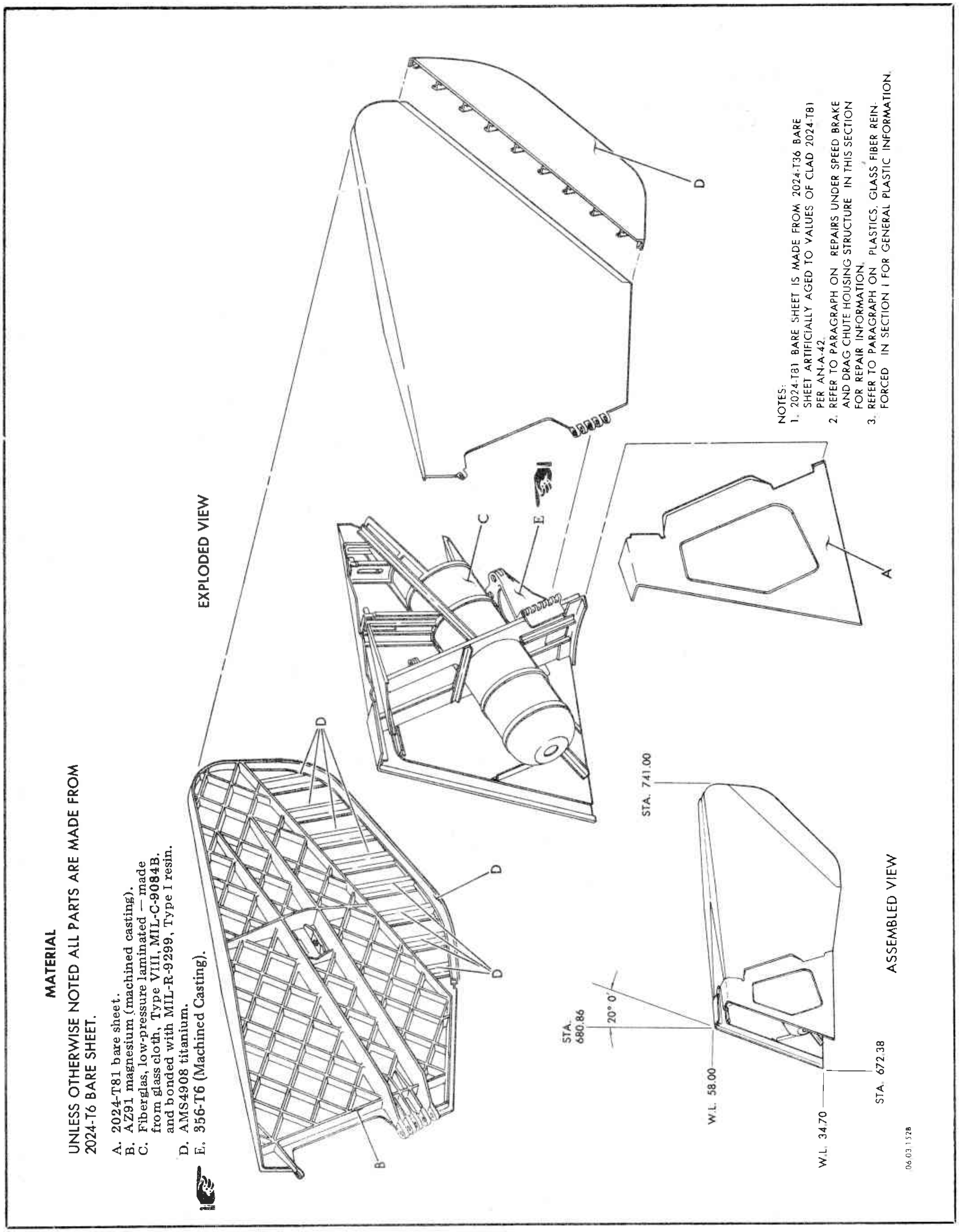
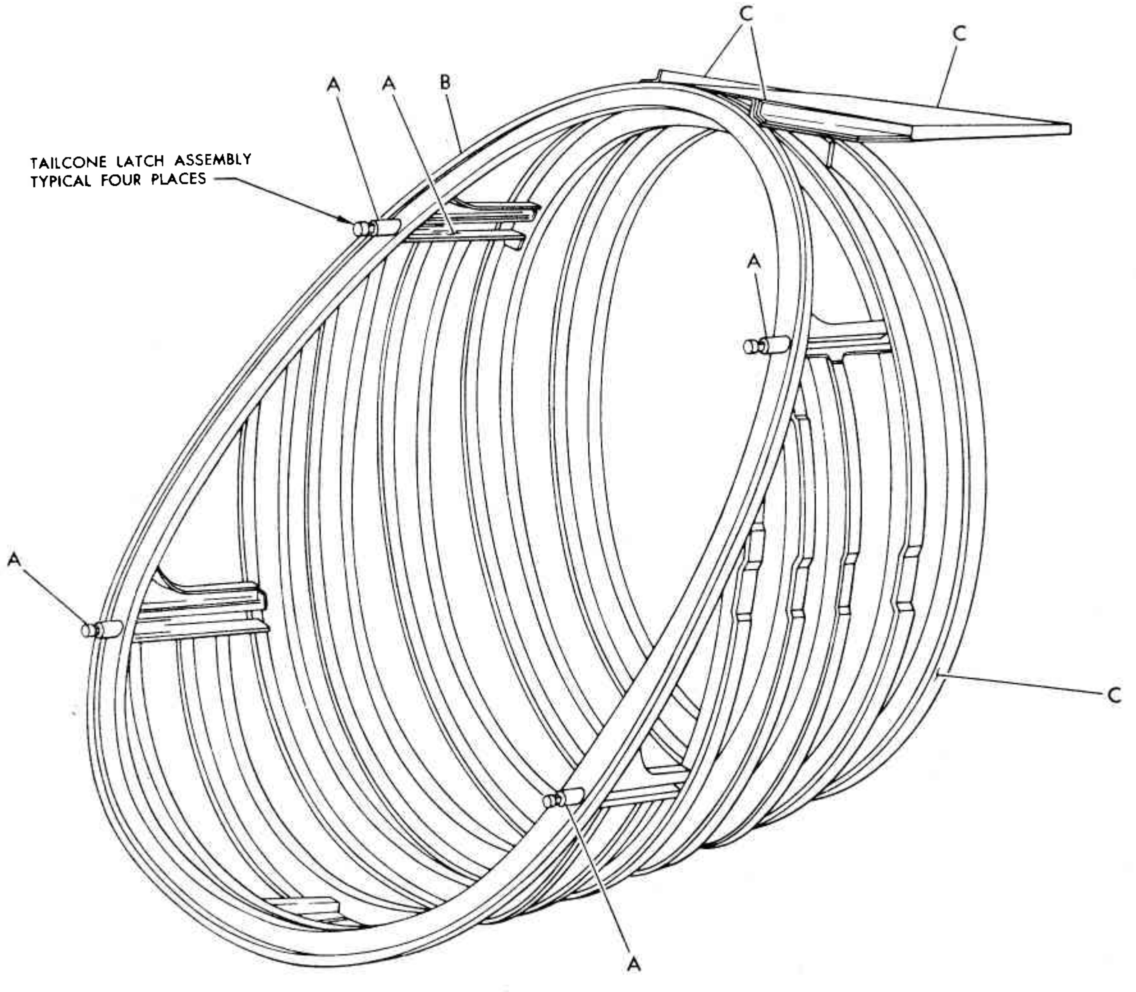


Figure 4-24. Speed Brake Structure and Drag Chute Housing

MATERIAL

UNLESS OTHERWISE NOTED ALL PARTS ARE MADE FROM 2024-T6 BARE SHEET.

A. 4130 steel, machine bar.
B. 2024-T6 extrusion.
C. AMS4908 titanium.



NOTES:
1. SEE FIGURES TITLED "FORMULA AND METHOD FOR PATCH REPAIR" AND "FORMULA FOR INSERTION REPAIR" IN SECTION I FOR FORMULA AND METHOD OF REPAIRS.
2. REFER TO SECTION X FOR TYPICAL REPAIRS.

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Figure 4-25. Tail Cone Structure

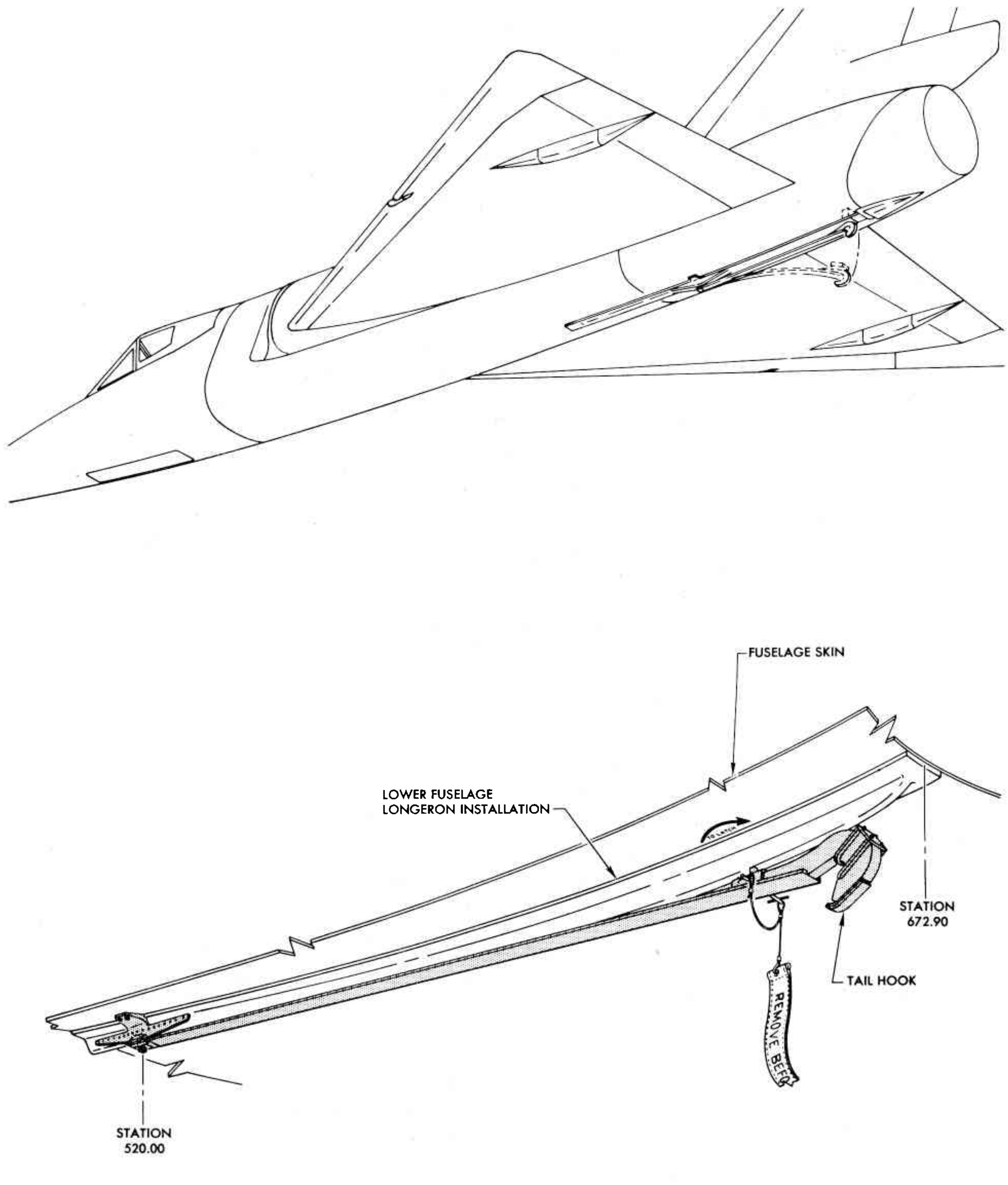
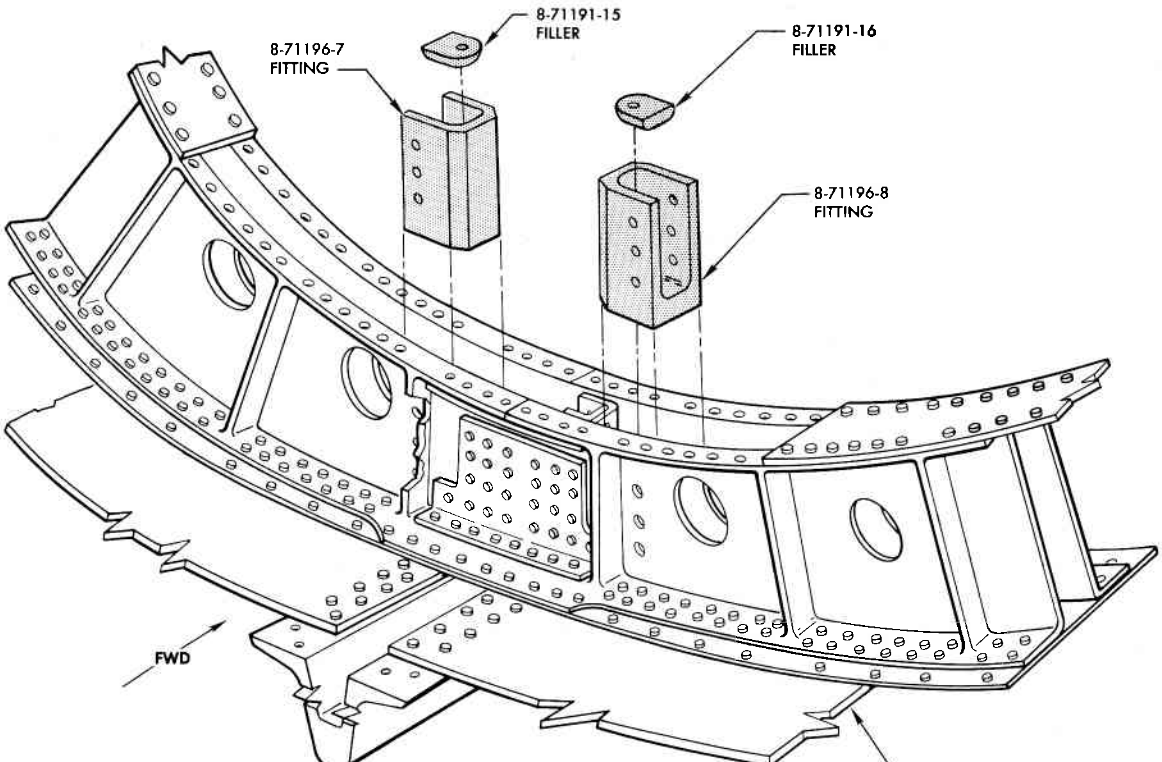
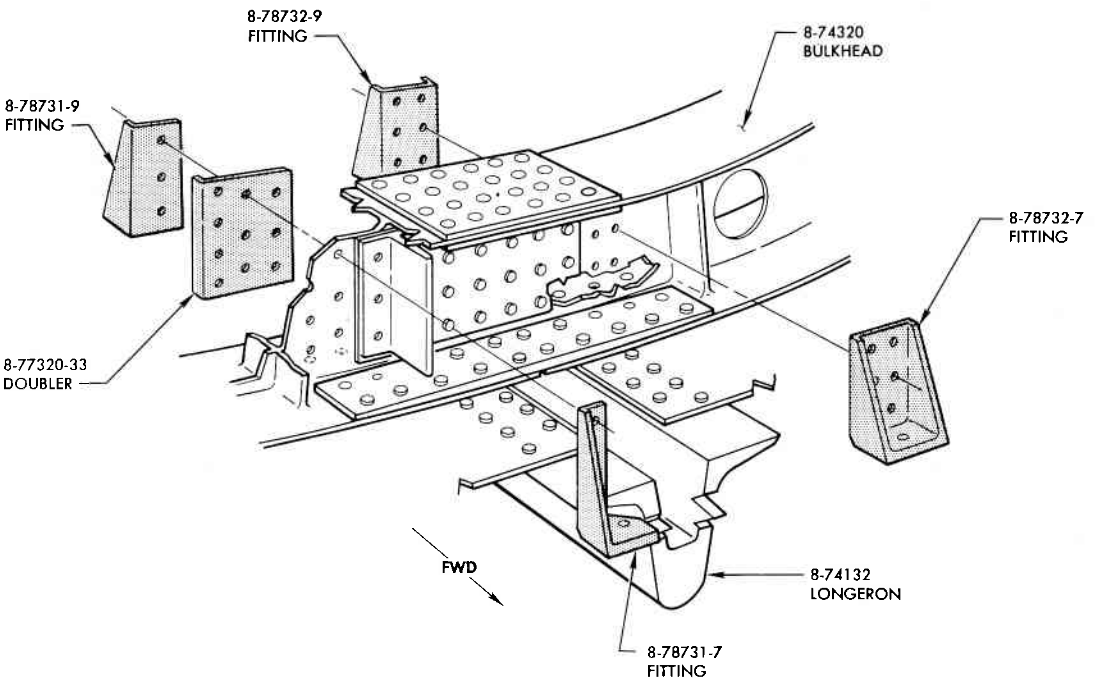


Figure 4-26. Tail Hook Installation (Sheet 1 of 2)



BULKHEAD—STATION 520.00

APPLICABLE TO F-106A AIRPLANES 57-246 THRU 57-2464, AND F-106B AIRPLANES 57-2515 THRU 57-2521.

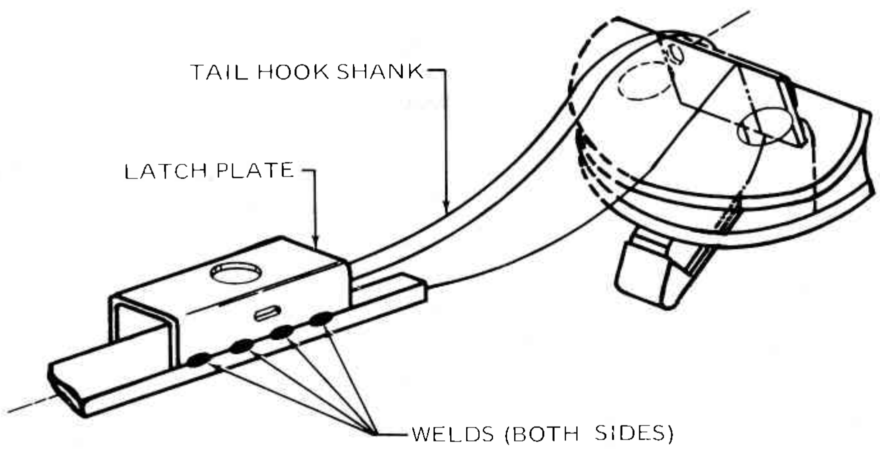


BULKHEAD—STATION 520.00

APPLICABLE TO F-106A AIRPLANES 56-453, -454, 56-456 THRU 57-245, AND 57-2465 AND SUBSEQUENT; AND F-106B AIRPLANES 57-2508 THRU 57-2514, AND 57-2522 AND SUBSEQUENT.

106.03.284-2A

Figure 4-26. Tail Hook Installation (Sheet 2 of 2)



LATCH PLATE - TO - SHANK CRACKED WELDS SHALL BE REPAIRED AS FOLLOWS:

A. CLEAN AND STRIP PER T.O. 1-1-1 AND T.O. 1-1-2.

B. FLUORESCENT PENETRANT INSPECT PER T.O. 33B-1-1.

C. GRIND OUT CRACKED WELDS WITHOUT DAMAGING HOOK SHANK.

D. REWELD IN ACCORDANCE WITH MIL-W-8611 USING 17-4 FILLER ROD (TACK WELDS TO BE OF SAME TYPE AND IN SAME LOCATION AS THOSE REMOVED).

CAUTION

EXTREME CARE SHALL BE EXERCISED DURING THE WELDING OPERATION TO PREVENT EXCESSIVE HEAT APPLICATION TO THE TAIL HOOK SHANK.

E. PENETRANT INSPECT FOR CRACKED WELDS.

F. PRIME AND PAINT PER T.O. 1F-106A-23.

Figure 4-26A. Tail Hook Shank Latch Plate Repair

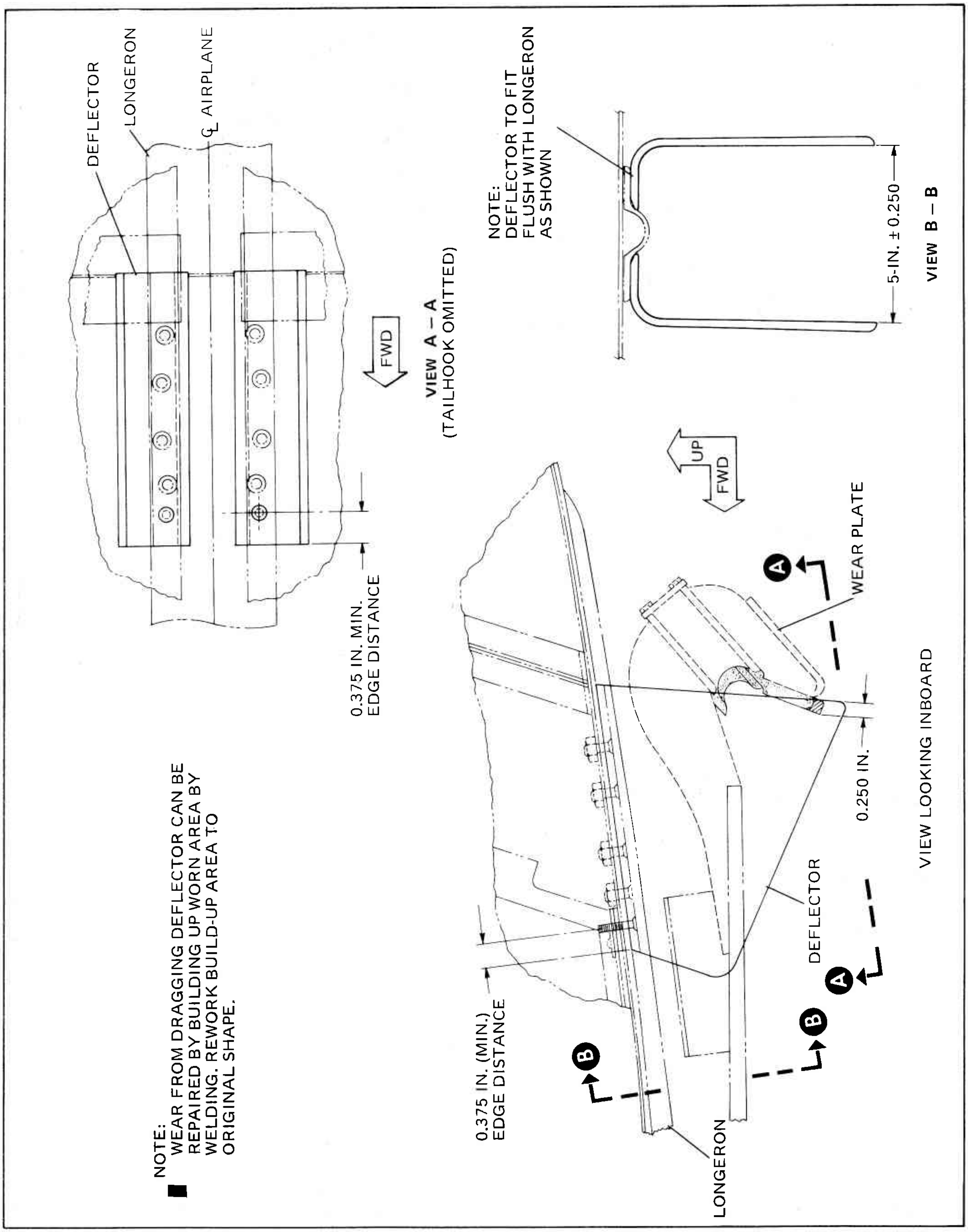


Figure 4-26B. Tail Hook Cable Deflector Installation.

NOTES:

1. TO CORRECT TAIL HOOK VIBRATION PROBLEMS INSTALL BUMPER PLATE AND PAD AS FOLLOWS:

2. EC1653 COMPOUND ON HOOK "HUMP" SHOULD BE PER CONVAIR DWG 8-05359.

3. MAKE BUMPER PLATE OF SUFFICIENT THICKNESS TO FILL GAP "A" FROM 7075-T6 AL ALLOY SHEET.

4. GAP "B" IS 0.125 INCH (MINIMUM) PLUS GAP "A"

5. FORM THE BUMPER PLATE TO CONFORM TO THE CONTOUR OF THE KEEL; FIT PLATE BETWEEN DEFLECTORS AND HOOK FAIRING.

6. PAINT PLATE WITH ZINC CHROMATE PRIMER AND FINISH WITH ENAMEL TO MATCH TAIL AREA.

7. ATTACH PLATE WITH MS20470AD6 RIVETS USING EXISTING HOLES OR BY DRILLING NEW HOLES.

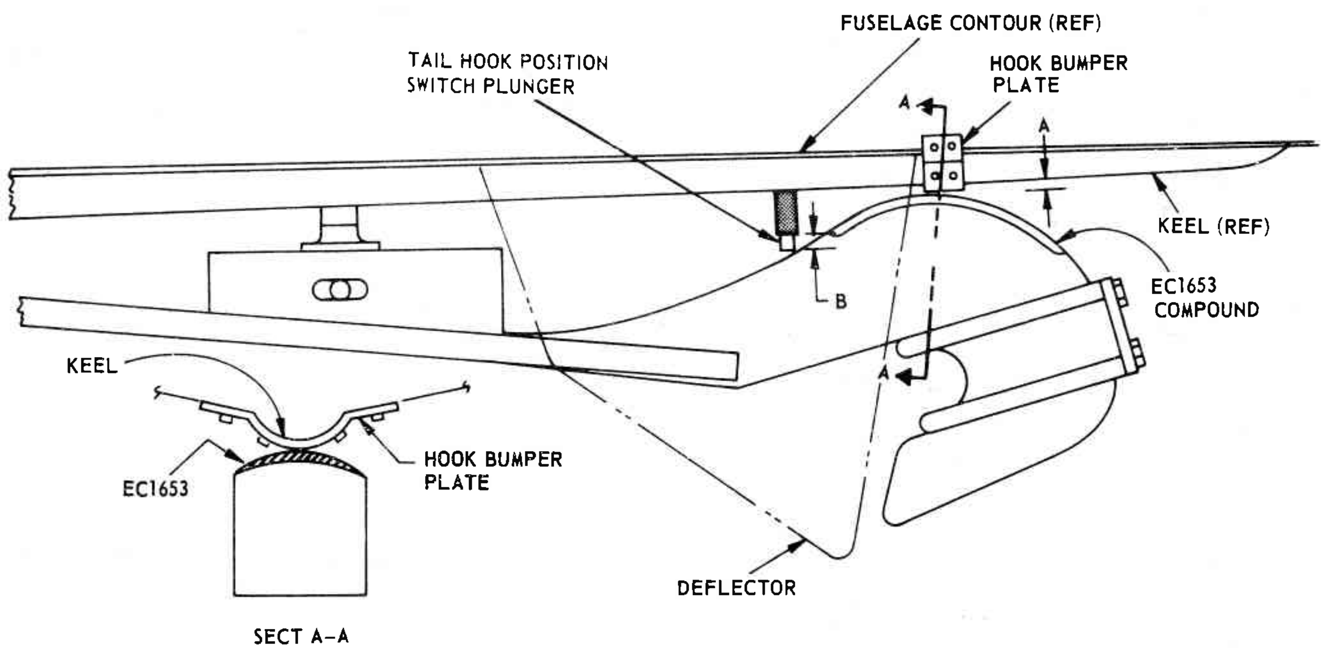
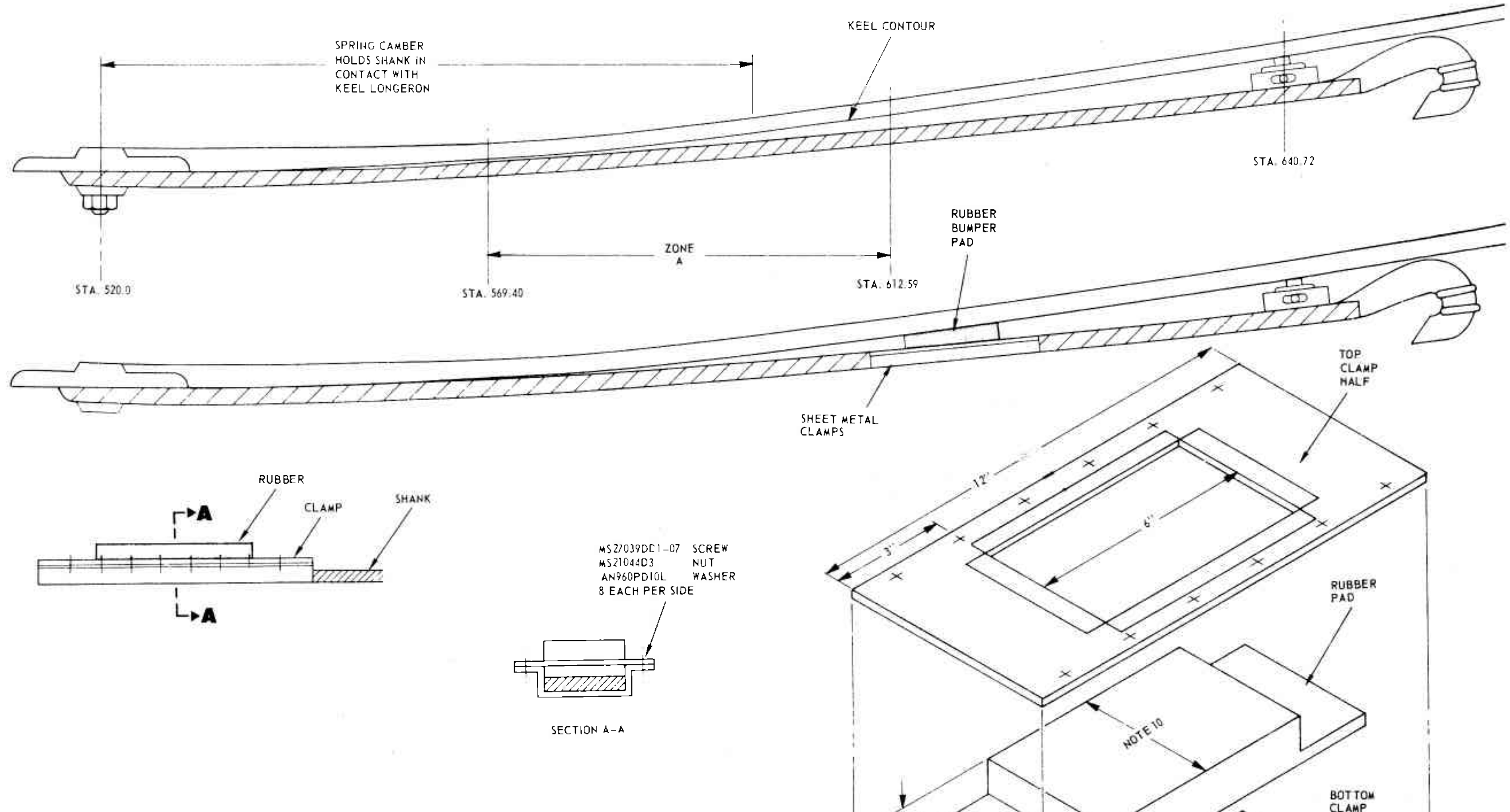
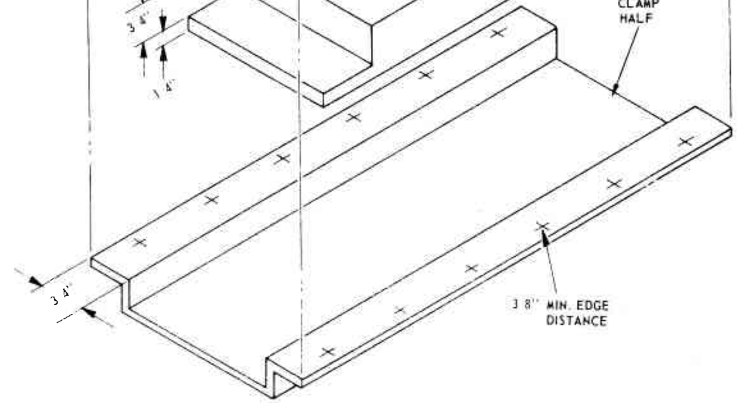


Figure 4-26C. Tail Hook Bumper Pad and Bumper Plate Installation (Sheet 1 of 2)





NOTES (CONT'D):

8. MAKE PAD CLAMPS FROM 7075-T6 AL ALLOY SHEET, 0.031-INCH THICKNESS (MIN) FOR BOTTOM CLAMP HALF, AND 0.062-INCH THICKNESS (MIN) FOR TOP CLAMP HALF.

9. MAKE RUBBER PAD FROM RUBBER SHEET, 1-INCH THICK MIL-S-6855, DUROMETER RATING 60 (FSN 9320-261-5807).

10. MAKE RUBBER PAD AND CLAMPS MATCHING WIDTHS PER WIDTH OF TAIL HOOK SHANK WHERE BUMPER IS MOUNTED.

11. FORM BOTTOM CLAMP HALF TO A HAT SECTION AS SHOWN, WITH HEIGHT SUCH THAT RUBBER PAD IS SANDWICHED AND COMPRESSED SUFFICIENTLY TO HOLD FASTENED ASSEMBLY FIRMLY ON HOOK SHANK.

12. ROUND OFF CORNERS AND SHARP EDGES OF CLAMPS.

13. PAINT ALL CLAMP SURFACES PRIOR TO ASSEMBLY WITH ZINC CHROMATE PRIMER AND FINISH WITH ENAMEL TO MATCH TAIL HOOK.

14. POSITION SCREW HEADS ON BOTTOM SIDE OF BUMPER PAD ASSEMBLY.

15. ATTACH BUMPER PAD ASSEMBLY IN ZONE "A" SO AS NOT TO INTERFER WITH TAIL HOOK REPLACEMENT TOOL P/N 8-96574.

Figure 4-26C. Tail Hook Bumper Pad and Bumper Plate Installation (Sheet 2 of 2)

4-31. Separation of Fuselage Bonded Components.

4-32. Metalbond 4021 structural adhesive is utilized on surfaces faying with the skin from station 102.00 to the canted bulkhead at station 171.50, and from the canopy longeron down to the nose wheel well opening. This adhesive process forms a strong metal-to-metal bond and the components cannot readily be separated. The following method, however, can be used to separate these bonded components:

a. Remove all fasteners from the parts to be separated.

b. Fabricate a wedge from micarta or any non-metallic material that will not nick, dent or gouge the fuselage structure.

c. Apply a layer of dry ice over the horizontal part to be removed. If the particular area in which the components are located is somewhat on a vertical plane, tape bags of dry ice over the part to be removed.

d. The dry ice should be left in contact with the part to be removed until a layer of frost appears on the opposite side of the components.

Note

Gaseous CO_2 can be substituted for dry ice in the preceding steps (c and d). The gaseous CO_2 should be applied along the bonded seam.

e. Place the sharp edge of the wedge against the faying surface of the components to be separated.

f. Using a leather or plastic mallet, strike the wedge with a single, sharp blow.

g. Repeat steps "e" and "f" at spaced intervals along the part to be removed to assure positive breaking of the bond.

CAUTION

Do not attempt to drive the wedge completely between the components being separated.

After the bonded components have been separated, the remaining tape and adhesive may be removed with a stripper (Cee-Bee "A" 202 manufactured by the Cee-Bee Chemical Co., Inc., 9250 Cee-Bee Drive, Downey, Calif.) by the following procedure:

WARNING

A face shield and/or goggles must be worn to protect the face and eyes and a rubber apron and rubber gloves worn to protect the hands and other portions of the body. In case the stripper material accidentally contacts the skin, wash the affected skin area with soap and water and rinse with alcohol. For protection of eyes, flush with large amounts of water and report to the dispensary immediately. For over-inhalation, move the affected person to an area where fresh air may be obtained. For ingestion, administer castor oil or olive oil with milk of magnesia.

a. Apply stripper to remaining adhesive with a brush.

b. After approximately ten minutes, use a plastic or micarta scraper to remove the loosened material.

c. Reapply stripper with brush.

d. Again, after a ten-minute period, scrape all loosened material.

e. Repeat stripper applications and scraping until metal bond is removed.

CAUTION

Do not use Cee-Bee "A" 202 chemical to clean titanium.

4-33. Fuselage Nose Section Repairs.

4-34. Damage to the fiberglass reinforced plastic portion of the radome structure must, in all cases, be evaluated from a radar scanning performance standpoint in addition to the loss of structural strength. Cracks, holes or any major damage to the fiberglass reinforced plastic are difficult to repair because of the strict requirements for accuracy in processing techniques and the exacting quality standards of the completed repair; refer to Specification MIL-P-8013A, type III. Completed repairs must be tested for deflections or interruptions of radar transmission. For these reasons, advanced base repairs to the radome plastic section are not advisable, and damage, as described above, shall be referred to a qualified antenna engineer for disposition. For advance base repair limitations see figure 4-6. See figure 4-6A for overhaul facilities limitations and repair procedure. Repair and maintenance of the rain erosion protective coating of the radome is described in Section I. Replacement procedure for the tubing and harness nylon retaining blocks, bonded to the inside of the radome, is contained in paragraph 4-35. For method of repair to the aluminum components of the radome see figure 1-23. The temper designations and physical characteristics of the materials used in the fuselage nose section structure are varied; therefore, each part should be given individual consideration when making repairs. For repairs

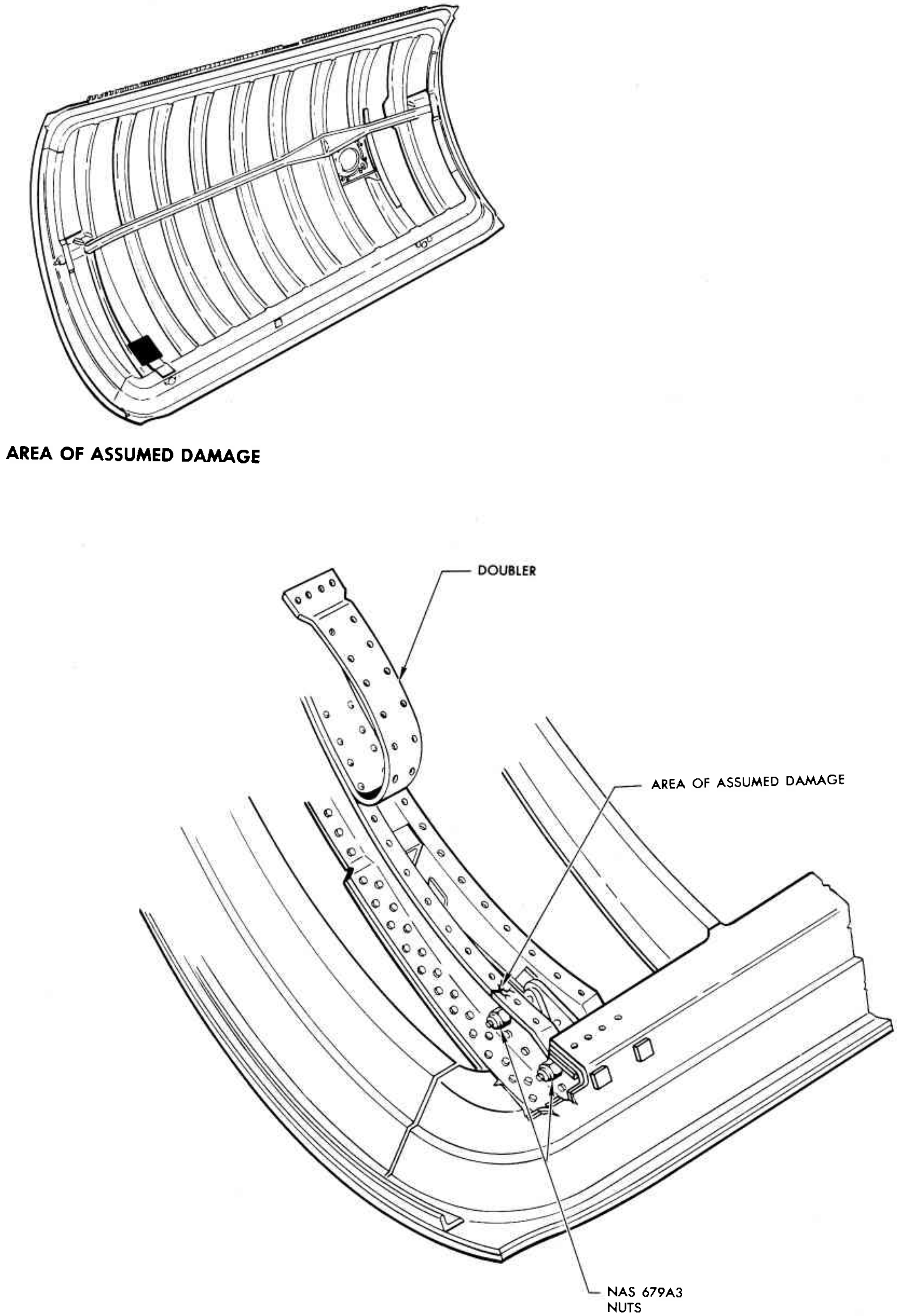


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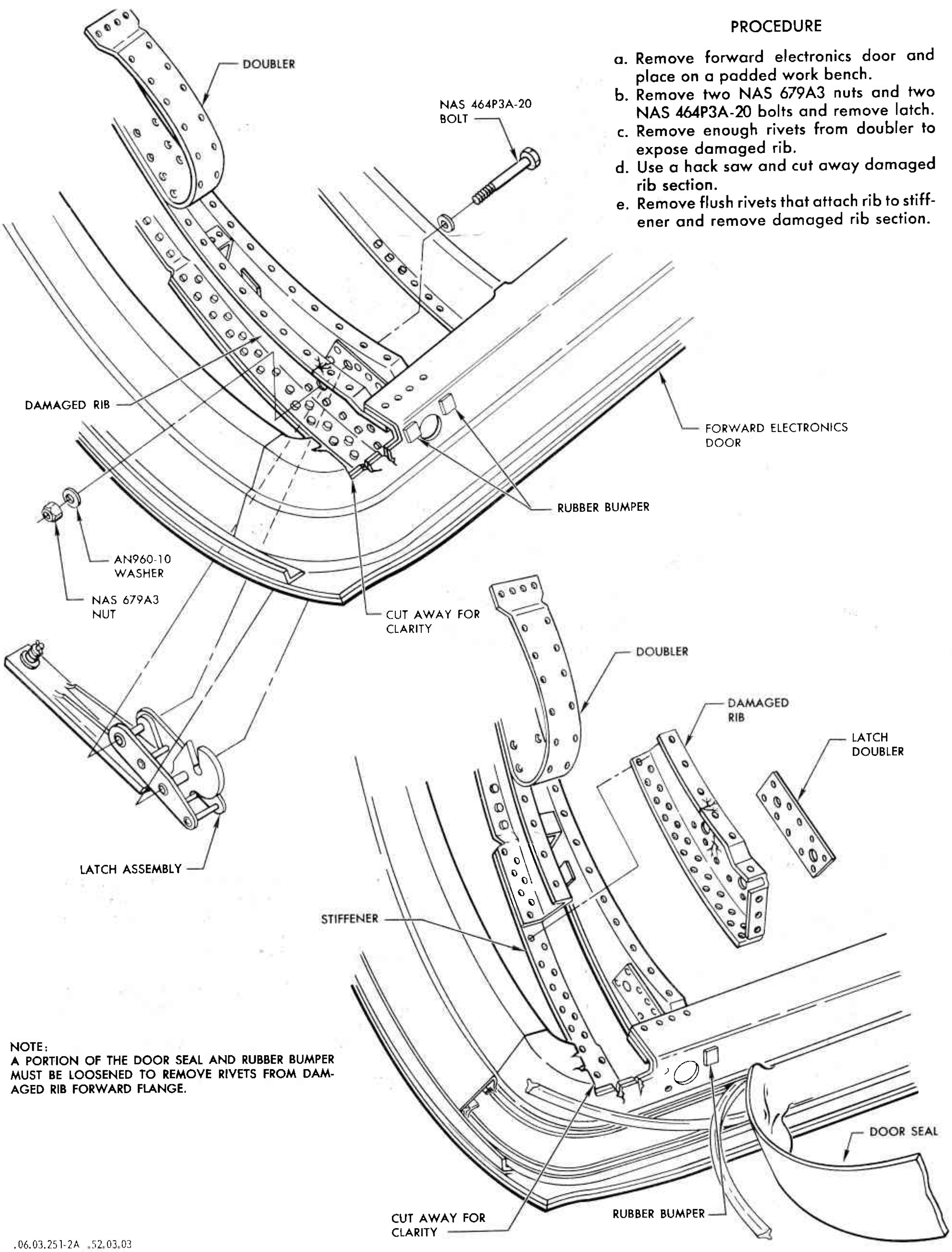
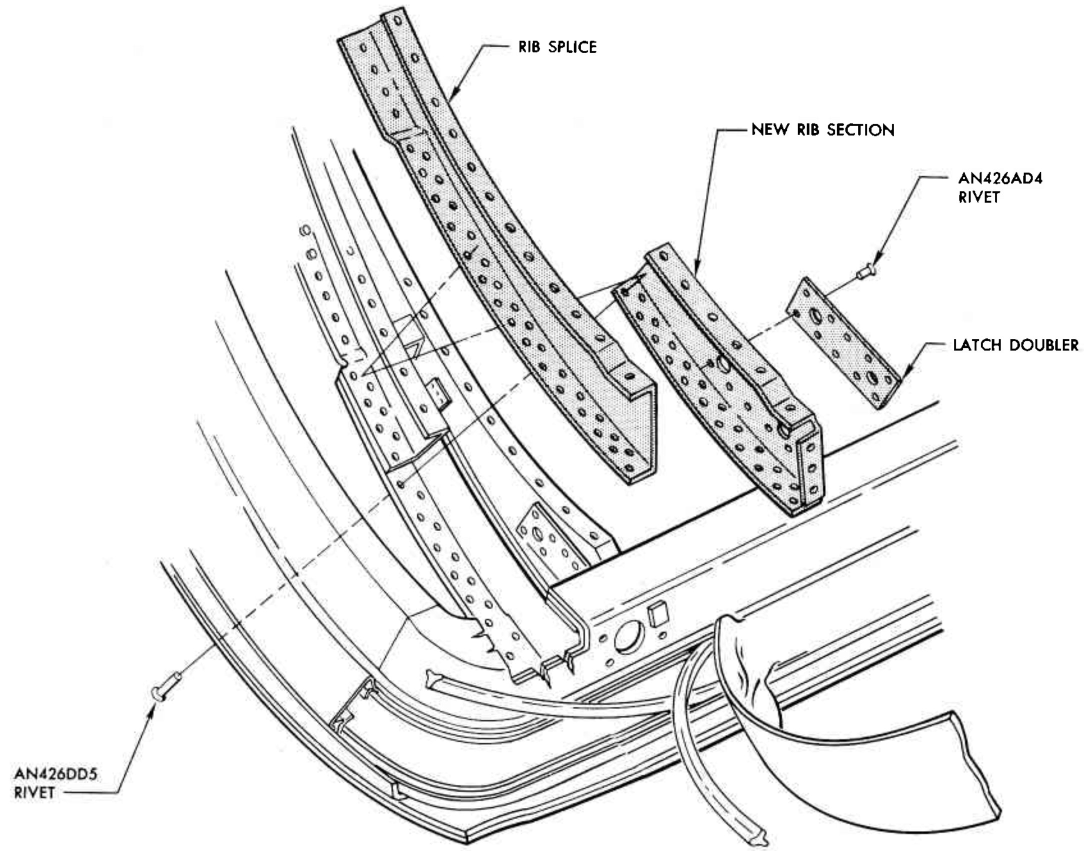
Figure 4-27. Repair of Forward Electronics Compartment Door (Sheet 1 of 4)



PROCEDURE

a. Remove forward electronics door and place on a padded work bench.
b. Remove two NAS 679A3 nuts and two NAS 464P3A-20 bolts and remove latch.
c. Remove enough rivets from doubler to expose damaged rib.
d. Use a hack saw and cut away damaged rib section.
e. Remove flush rivets that attach rib to stiffener and remove damaged rib section.

Figure 4-27. Repair of Forward Electronics Compartment Door (Sheet 2 of 4)



PROCEDURE (CONT)

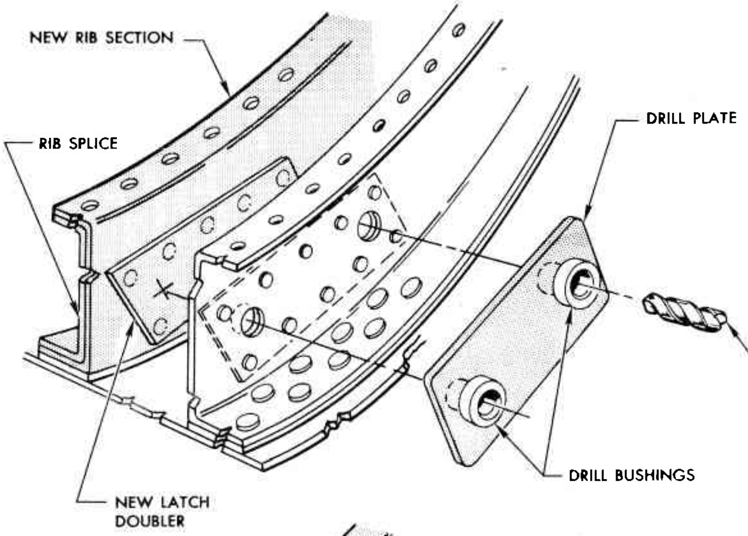
f. Fabricate new rib section from .045" 7178-0 aluminum alloy.
g. Fabricate new latch doubler from .40" 7178-0 aluminum alloy.
h. Heat treat new rib section and latch doubler to T6 conditions.
i. Drill rivet holes through latch doubler and rib. Use damaged rib as pattern for holes and latch doubler location.
j. Countersink holes and rivet latch doubler to rib. Use AN426AD4 rivets.
k. Clamp rib in place and pick up rivet holes from existing holes in stiffener and door angle.
l. Rivet forward section of rib to stiffener. (Leave holes to install rib splice.)
m. Fabricate rib splice from .051" 7178T6 aluminum alloy.
n. Rivet rib splice in place. Use AN426DD5 rivets through splice and skin and AN470DD5 rivets through splice and rib.
o. Pick up existing holes from doubler and rivet doubler in place. Use AN470DD5 rivets through rib splice AN470AD5 through rib.

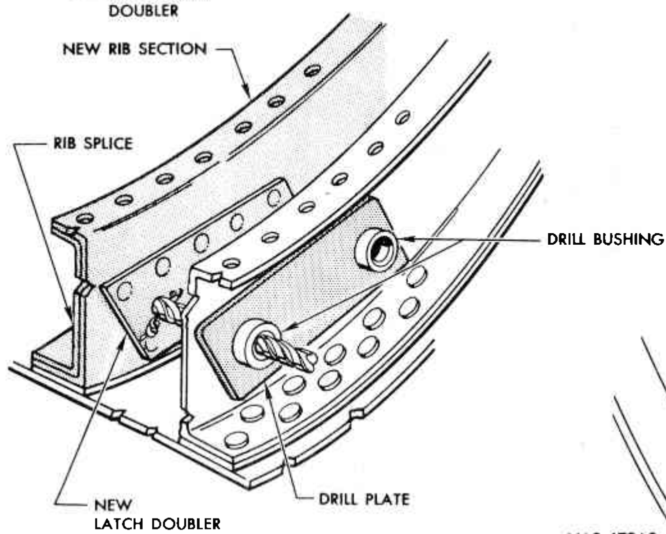
.06.03.251-3A .52.03.03

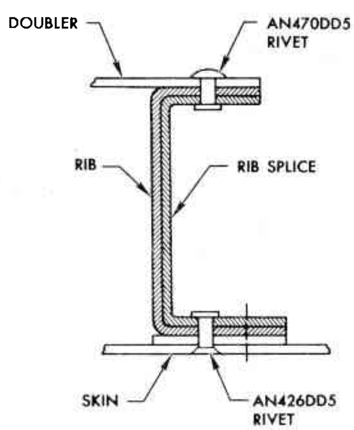
Figure 4-27. Repair of Forward Electronics Compartment Door (Sheet 3 of 4)

PROCEDURE (CONT)

p. Fabricate drill plate as shown.
q. Pick up latch mounting holes with pilot drill.
r. Ream latch mounting holes to 0.190" +.0015" −.0000".
s. Assemble latch assembly and replace rubber cushion. Use NAS 464PA-21 bolts to mount latch.
t. Reinstall door seal, and rubber bumper.
u. Reinstall door.







SECTION A-A

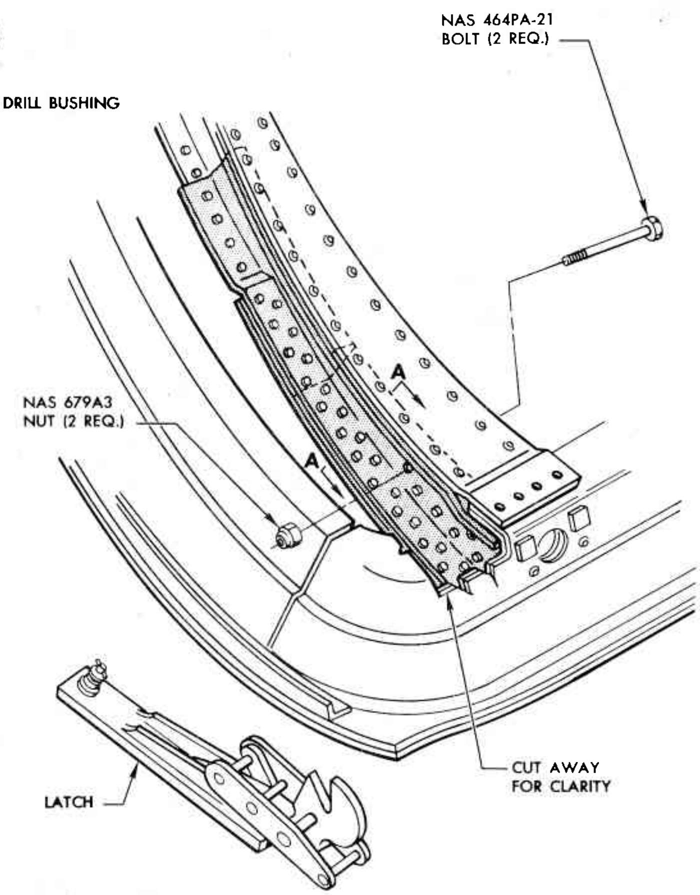


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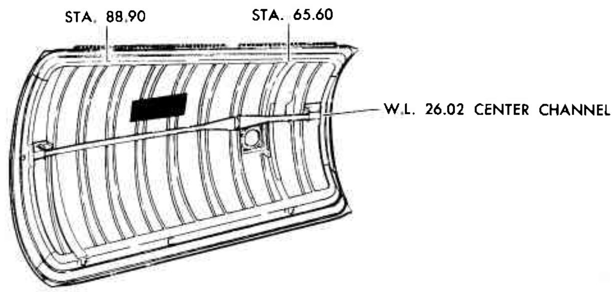
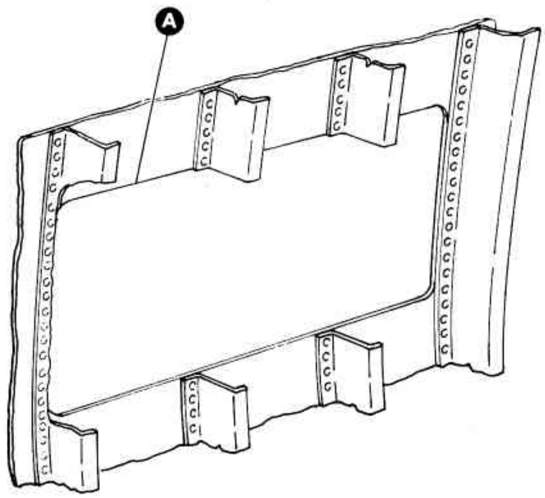
Figure 4-27. Repair of Forward Electronics Compartment Door (Sheet 4 of 4)



AREA OF ASSUMED DAMAGE

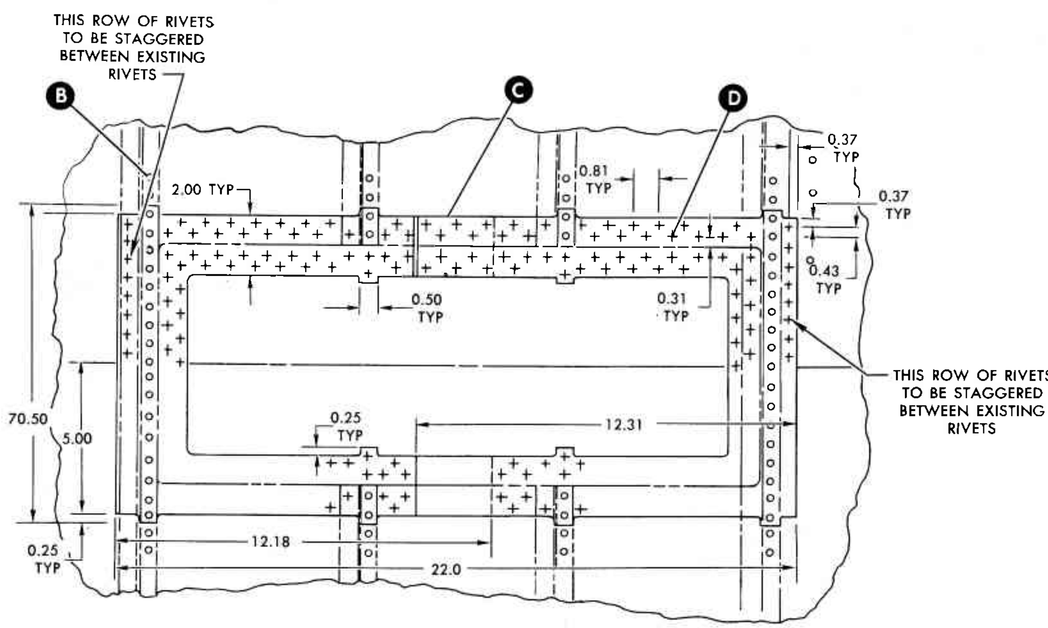


A Clean up damaged area to regular shape. Cut away damaged skin and zees as shown, and remove. For required strength, extend skin cut to nearest undamaged zee each side as shown. Round all corners to 0.38 inch radius and break all sharp edges.

B Using a No. 30 (0.128) drill remove existing rivets as indicated by (0).

C Fabricate doubler in two sections from 0.040 gage bare 7178-T6 material as shown.

D Layout rivet pattern on doubler as dimensioned and drill No. 30 (0.128) holes. Pattern typical both sides except slight variation around cutoff zees as shown.



VIEW LOOKING OUTBOARD

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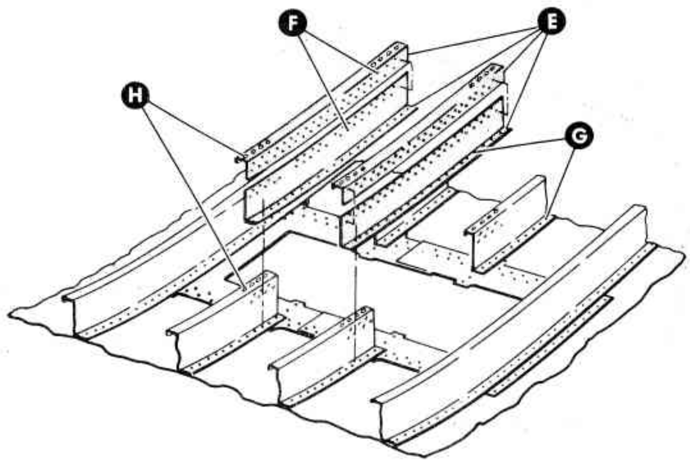
Figure 4-28. Forward Electronics Compartment Door Repair (Sheet 1 of 2)

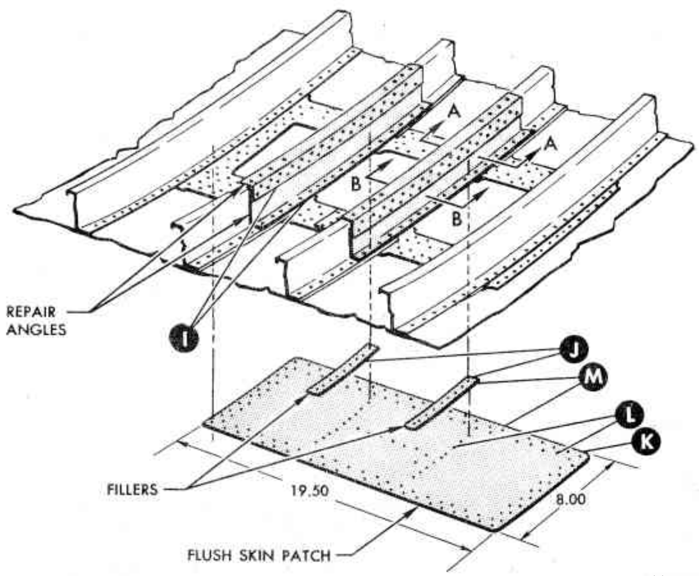
E Fabricate splice angles from 0.040 gage bare 7178-T6 material of sufficient length to pick up four rivets on zees as shown.

F Layout and drill No. 30 (0.128) holes for staggered rivet row at 0.70 inch typical on legs of splice angles. Continue pattern on legs of zees as indicated by (+).

G Pick up rivet pattern on zee bottom flange and continue across bottom flange of splice angle. Drill No. 30 (0.128) holes.

H Layout and drill No. 30 (0.128) holes on top flange of zees and top flange of splice angles as indicated by (0).





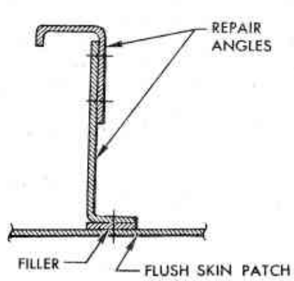
I Position splice angles and install with AN-470AD4 rivets. Use AN426AD4 flush head rivets where parts are attached through outside skin.

J Fabricate filler strips of 0.040 gage bare 7178-T6 material. Pick up rivet pattern on splice angles and use as guide to drill No. 30 (0.128) holes in filler strips.

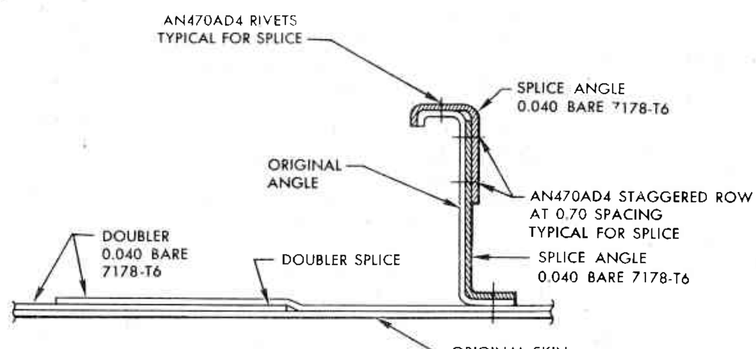
K Fabricate flush patch of 0.036 gage material as dimensioned.

L Pick up existing rivet patterns in doubler sections and splice angles and drill No. 30 (0.128) holes in flush patch. Countersink holes 100°x0.235 far inside.

M Position fillers and flush patch and install with AN426AD4 rivets.



SECTION B-B



SECTION A-A

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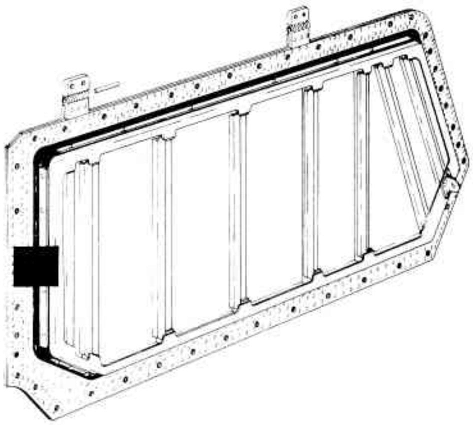
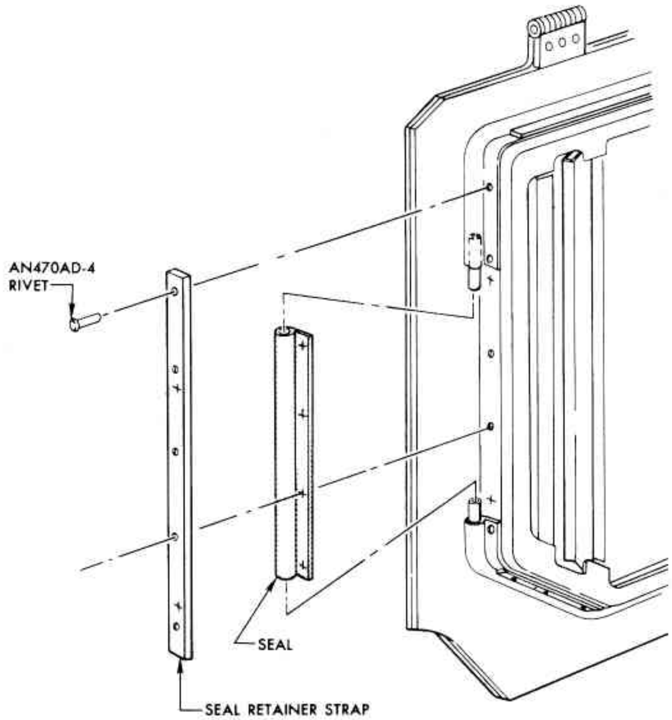
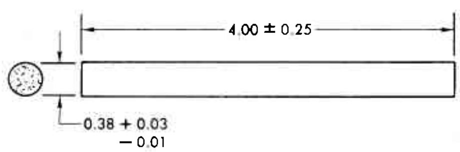
Figure 4-28. Forward Electronics Compartment Door Repair (Sheet 2 of 2)



AREA OF ASSUMED DAMAGE





SEAL SPLICE
(SEE NOTE)

NOTE:
SEAL SPLICE TO BE MADE FROM SILICONE SPONGE RUBBER ROD CONFORMING TO SPECIFICATION MIL-R-5847.

REPAIR PROCEDURE

a. Remove door and place on a padded work bench. Refer to T.O. 1F-106A-2-2-2-2 for removal procedure.
b. Use a No. 30 (0.128) drill and remove rivets that attach seal to door.
c. Remove strap and hold for reinstallation.
d. Cut out damaged section of seal as shown.
e. Clean seal faying surface of door. Use a cloth dampened with Aliphatic Naphtha, Federal Specification TT-N-95A, for cleaning.
f. Cut a section of new seal to fit cutout area.
g. Cut two seal splices as shown from Silicone sponge rubber rod conforming to Specification MIL-R-5847.
h. Roughen surface of splices with rough sand paper and wipe clean with a cloth dampened with Aliphatic Naphtha, Federal Specification TT-N-95A.
i. Clean inside surface of existing seal at each end with a cloth dampened with Aliphatic Naphtha, Federal Specification TT-N-95A.
j. Thoroughly mix four parts, by weight, of Cohrlastic C-319A Catalyst with one hundred parts, by weight, of Cohrlastic C-319 Cement. The work life of mixed cement is thirty to sixty minutes.

NOTE

C-319 CEMENT AND C-319A CATALYST IS MANUFACTURED BY THE CONNECTICUT HARD RUBBER COMPANY, 407 EAST STREET, NEW HAVEN 9, CONNECTICUT.

j. Paint seal splices with a uniform coat of prepared cement and insert splices into ends of existing seal as shown.
k. Hold seal splices in place by squeezing existing seal with fingers and sliding new seal section into place. Allow cement to cure for 24 hours at room temperature.
l. Place retainer strap over new seal section flange and punch a hole at each end of new seal section. Use an ice pick or similar pointed tool to punch hole.
m. Fasten retainer strap and new seal section to door with cleco fasteners.
n. Drill holes through new seal, using holes retainer strap as a guide. Use a No. 30 (0.128) drill.
o. Rivet retainer strap and seal to door with AN470AD-4 rivets.
p. Reinstall door. Refer to T.O. 1F-106A-2-2-2-2 for installation procedure.

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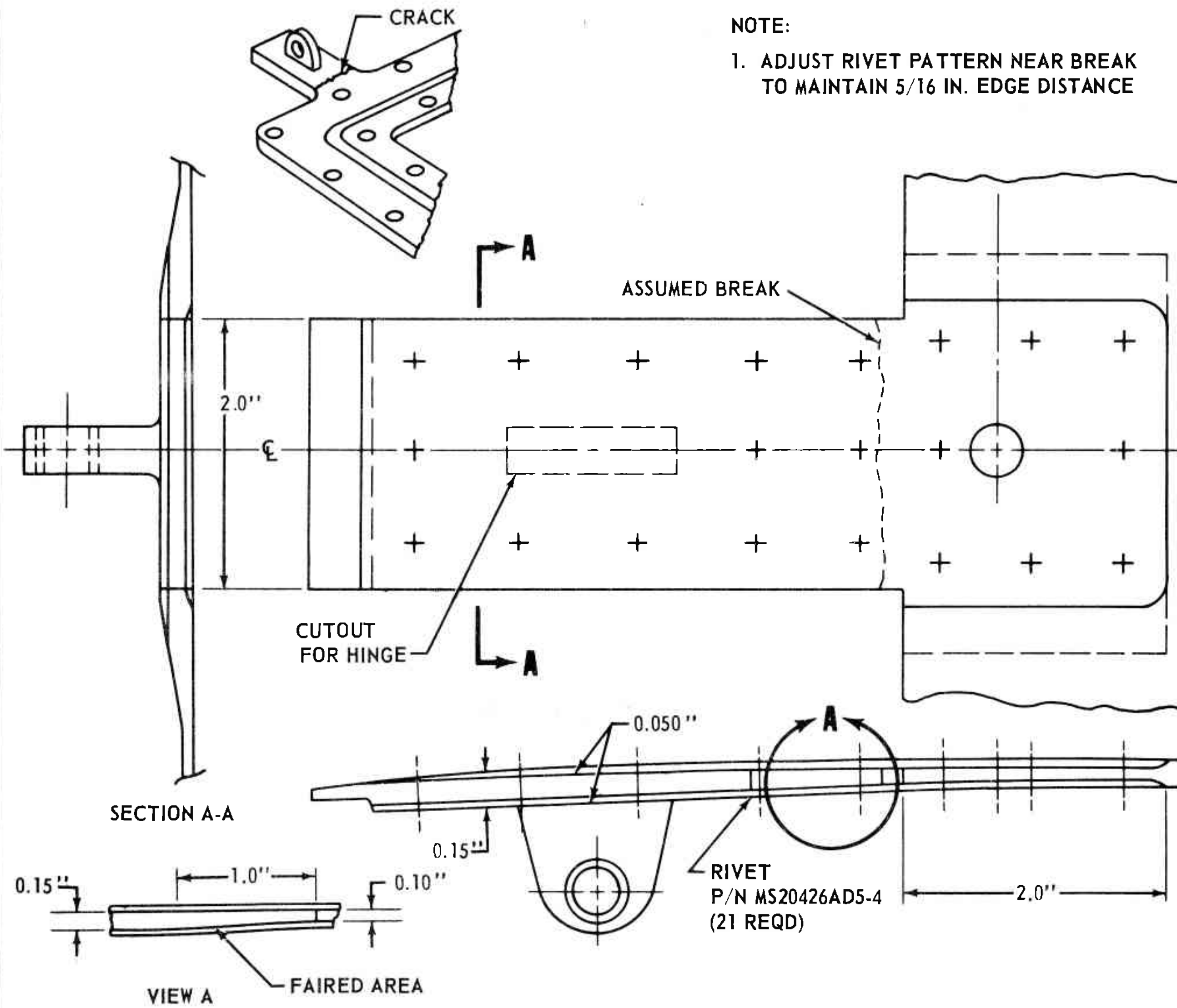
Figure 4-29. Aft Electronics Door Seal Repair



REPAIR PROCEDURE

1. Reuse of broken tab.

a. If broken tab is intact, smooth inner and outer surfaces along break line for reinstallation.

b. Fabricate repair doubler plates from 0.050 inch thick SAE-4130 steel sheet to cover full length of tab.

c. Machine outside of tab and door to allow plate to rest flush as shown.

d. Machine inside of door to allow plate to rest flush. Fair inside of tab for smooth transition of plate between door and tab. Transition to be approximately one inch as shown in View A.

e. Locate and drill 21 each (0.159 inch) holes as indicated with 5/16 inch edge distance.

f. Locate and drill one hole for Camloc fastener as shown.

g. Heat treat doubler to 140,000-160,000 psi. Cadmium plate and prime, using zinc chromate primer.

h. Install repair doubler as shown.

2. If broken tab is unserviceable.

a. Fabricate new tab (0.100 inch thick) and new lug of 6061T-4 aluminum alloy plate. Extend lug base 0.050 inch to allow for thickness of inside doubler plate.

b. Locate and weld lug onto tab.

c. Fabricate doubler plates and proceed with repair as applicable in paragraph 1 above.

Figure 4–29A. Repair of Midbay Electronic Access Door Hinge

FIG. 4–29B DELETED

applicable to the various components, see figures 1-22 through 1-24, and applicable figures in this section. Also refer to Section X for typical repair information and to Section XI for repair materials. See figures 4-27, 4-28, and 4-29 for repair of the forward and aft electronics compartment doors. Refer to Table 4-I for disposition of negligible damage to the components of the nose section. The pilot's compartment is pressurized and any damage to the components in this area must be repaired to restore complete airtightness of the compartment. Damaged castings, fittings and glass panels must be replaced. Damaged extrusions of irregular shapes that present complex repairs should be replaced. When making repairs that involve resealing for pressurization, thoroughly clean all faying surfaces before applying sealant. The sealant must be applied at the inboard edge of all faying surfaces in a continuous bead or fillet at least 1/16 inch thick where cabin air leakage is likely to occur. The sealant should be continuous and extend approximately 3/16 inch to each side of a crevice or surface edge. In areas where it is difficult to apply sealant after assembly, it is permissible to apply the sealant to the faying surface prior to riveting, provided riveting or clamping pressure is applied immediately after spreading sealant. Use sealer, Military Specification MIL-S-8802, for pressure sealing, except at faying surfaces and joints of the windshield where RTV Silastic sealer shall be used. A rubber plug may be used for holes and openings up to 7/16 inch wide. Select an appropriate size plug and coat with sealer, Military Specification MIL-S-8802, prior to installation; trim the excess rubber and complete the final seal. Where air pressure leaks occur at rivets, the rivets in question should be removed and replaced. Holes should be filled with sealer, Military Specification MIL-S-8802, and new rivets installed immediately. Remove excess sealant from surface as soon as the rivet is driven, as partially cured sealant material is difficult to remove. Refer to paragraph 1-38 for cockpit pressure leak test procedures after repair. All repairs to the fuselage fuel tank must be made to conform with fuel-tight requirements given in Section II for wing repairs. Refer to paragraph 4-43 for fuselage integral fuel tank pressure test procedures. All repairs to the fuselage plating shall be of the flush-type patch. These repairs shall be designed to restore the full strength of the damaged part and preserve the aerodynamic contour of the fuselage. See figure 1-32 for rivet spacing requirements and refer to Section X for typical repair information. Refer to Table 4-1 for disposition of negligible damage to the components of this section. For repair of the upper aft electronics compartment doors, see figure 4-15 for material specifications and gage. Cracks in the F-106A upper aft electronic compartment door frame (seal 0.040 aluminum alloy) that extend up to six-inches in length and are located between the two radii may be repaired by stop drilling. Stop drill cracks with a 1/8 inch drill being careful not to drill parts behind or adjacent to affected part. Cracks in excess of six-inches and cracks extending beyond either radius are to be repaired in accordance with figure 10-4.

4-35. Radome Assembly Nylon Retaining Block Replacement.

4-36. The nylon retaining blocks installed inside the radome assembly, along the top and bottom, may be replaced by bonding. With the radome assembly removed and radome electrical harness and tubing free and clear of the retaining block installation area, rebond the blocks to the inside surface of the radome assembly as follows:

a. Thoroughly clean retaining block and radome faying surfaces with a cheesecloth dampened in aliphatic naphtha (6810-238-2118).

b. Lightly sand faying surfaces with 320-grit sandpaper or equivalent. Wipe off all loose particles after sanding with a clean cheesecloth dampened in aliphatic naphtha.

c. Apply one coat of phenol-alcohol with a clean acid brush to radome and retaining block faying surfaces. The phenol-alcohol solution shall consist of one part commercial phenol to ten parts denatured alcohol (measured by weight). The denatured alcohol shall conform to Specification MIL-A-6091. One manufacturer of phenol is the Braun Chemical Company of Los Angeles, Calif.

d. Apply Narmco 3135 adhesive to faying surfaces of block and radome, and assemble immediately for bonding. Use just sufficient pressure to assure contact over all portions of the bonding area.

e. Allow a 24-hour curing time at room temperature before reinstalling electrical harness and tubing. Curing time may be reduced to approximately two hours by exposing the bonded area to temperatures of 93.3°C (200°F) to 121.1°C (250°F) for 45 to 90 minutes.

CAUTION

Allow adhesive to thoroughly set (approximately 20 minutes) before moving radome or applying any heat.

4-37. Airspeed Tubing Drain Trap Reinforcement — Station 102.00 Bulkhead.

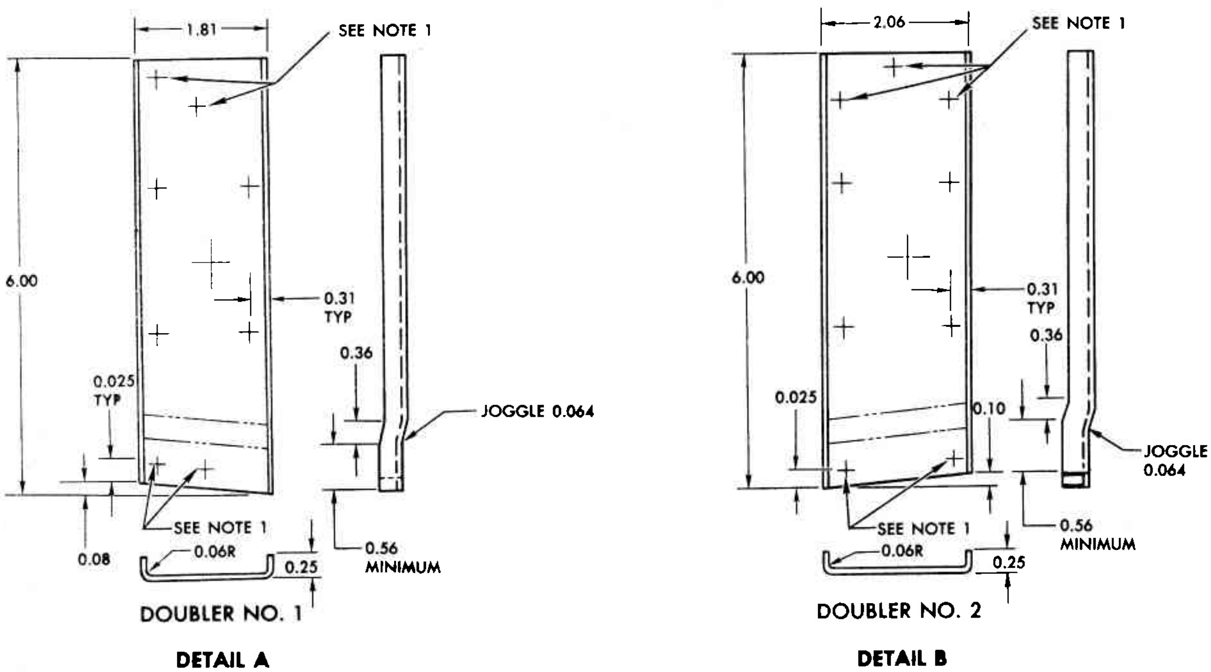
4-38. See figure 4-30 for airspeed tubing drain trap reinforcement procedure.

4-39. Airspeed Tubing Drain Trap Installation Repair — Station 102.00 Bulkhead.

4-40. The airspeed tubing drain trap installations on the aft side of station 102.00 bulkhead are accessible from the airplane's nose wheel well. On airplanes not incorporating reinforcing doublers, cracks may develop in the bulkhead, and will run from the airspeed tubing holes out. See figure 4-31 for airspeed tubing drain trap installation crack repair procedure.

4-41. Fuselage Side Panel Splice Angle Crack Repair.

4-42. See figures 4-32 and 4-33 for repair procedures on cracks located in the forward end of splice angle which is attached to the intake duct lip casting.



NOTES:
1. LOCATE THESE HOLES FROM AN470AD-4 HOLES IN STATION 102.00 BULKHEAD (SEE INSTALLATION VIEW).
2. REFER TO SECTION FOR STANDARD MINIMUM RIVET SPACING.

REINFORCEMENT PROCEDURE

a. Remove ADF antenna cover and antenna located between station 77.00 and 94.75. Gain access from underside of airplane.
b. Enter area vacated by antenna. Remove NAS221 screws from left and right frame covers at station 94.75. Disconnect airspeed tubing at forward side of station 102.00 bulkhead and at nearest accessible connection forward of bulkhead. Remove tubing and frame covers from the airplane.
c. Enter nose wheel well, disconnect and remove drain traps from aft side of station 102.00 bulkhead.
d. Fabricate two doublers from 0.040 gage 7075-0 bare sheet as shown in details A and B. Heat treat doublers to -T6 condition in accordance with specification MIL-H-6088 after forming.
e. Remove AN470AD-4 rivets in aft side of station 102.00 bulkhead as indicated in installation view. Use a No. 30 (0.128) drill.
f. Lay out and predrill all rivet holes in doublers as indicated in details A and B. Use a No. 39 (0.099) drill.
g. Cleco doublers to aft side of station 102.00 bulkhead as indicated in installation view.
h. Enter area vacated by antenna, reach through station 94.75 frame holes, and scribe airspeed tubing holes on the backs of both attached doublers.
i. Drill all rivet holes through doublers and station 102.00 bulkhead from bulkhead aft side. Use a No. 30 (0.128) drill.
j. Remove doublers. Drill airspeed tubing holes through doublers to sizes obtained from scribing in step "h."

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k. Remove burrs from doublers and station 102.00 bulkhead.
l. Apply protective coating to doublers and station 102.00 bulkhead disturbed areas. See figure "Primer and Paint Coatings" in Section I.
m. Apply a coat of sealer, Military Specification MIL-S-81733 to faying surfaces of doublers.
n. Reinstall doublers with clecos and install AN470AD-4 rivets.
o. Remove clecos and finish riveting.
p. Enter area vacated by antenna and clean out frame area extending from station 94.75 to 102.00.
q. Reinstall drain traps and airspeed tubing.
r. Remove old sealer, Military Specification MIL-S-81733 from station 94.75 left and right frame cover faying surfaces.
s. Apply a new coat of sealer, Military Specification MIL-S-81733 to faying surfaces of station 94.75 left and right frame covers.
t. Reinstall station 94.75 frame covers with NAS221 screws.
u. Reinstall ADF antenna and cover.
v. Perform pitot-static system operational check and leak test in accordance with applicable maintenance manual.
w. Perform operational check of systems affected by removal and reinstallation of ADF antenna in accordance with applicable maintenance manual.

NOTE

CHECK ANTENNA COVER GASKET FOR SERVICEABILITY. IF A NEW GASKET IS REQUIRED, APPLY SEALER, MILITARY SPECIFICATION MIL-S-8802 BETWEEN NEW GASKET AND COVER.

*Figure 4-30. Airspeed Tubing Drain Trap Reinforcement-Station **102.00 Bulkhead** (Sheet 1 of 2)*

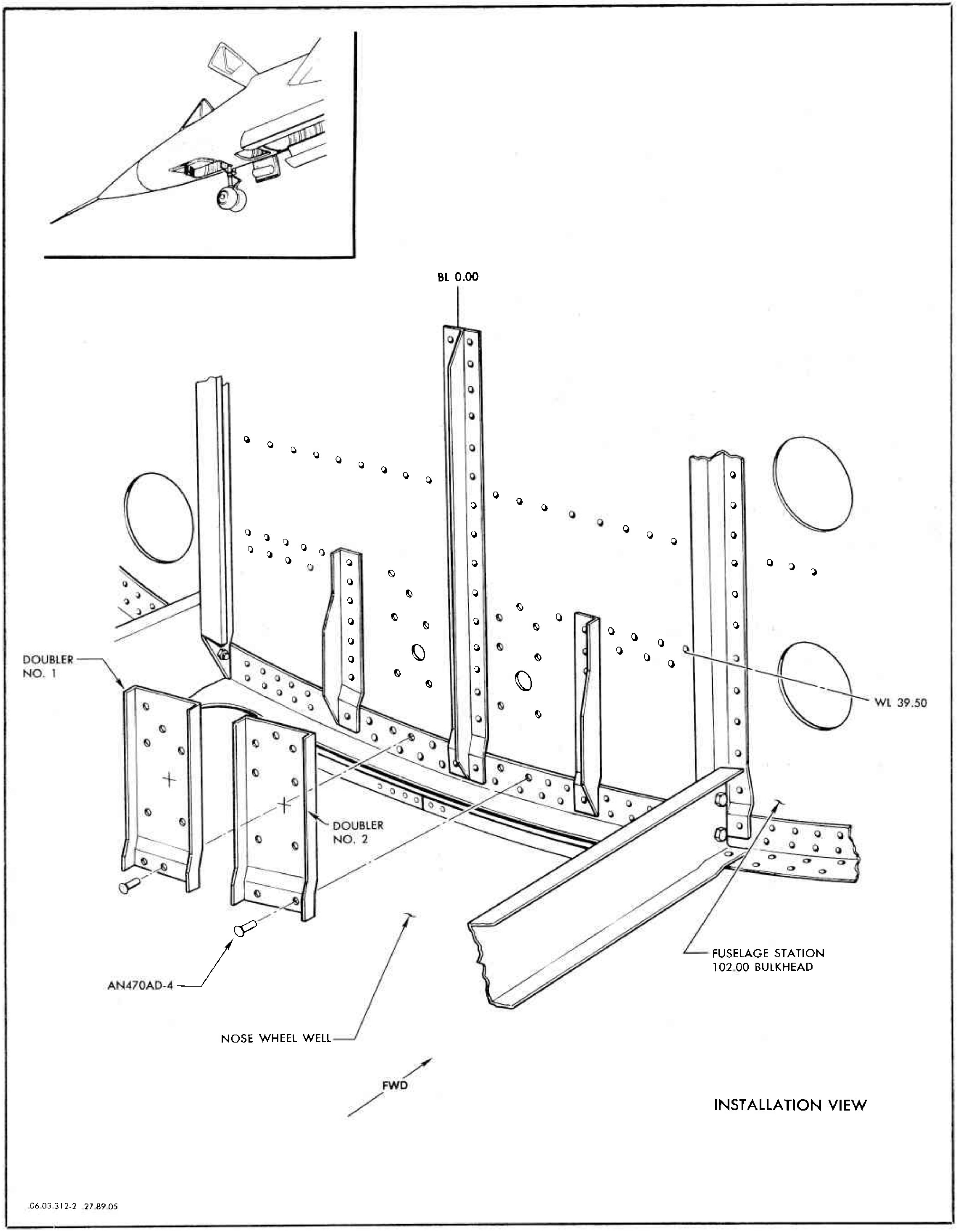


Figure 4-30. Airspeed Tubing Drain Trap Reinforcement — Station 102.00 Bulkhead (Sheet 2 of 2)

DAMAGE REMOVAL PROCEDURE

a. Remove ADF antenna cover and antenna located between station 77.00 and 94.75. Gain access from underside of airplane.
b. Enter area vacated by antenna. Remove NAS221 screws from left and right frame covers at station 94.75. Disconnect airspeed tubing at forward side of station 102.00 bulkhead and at nearest accessible connection forward of bulkhead. Remove tubing and frame covers from the airplane.
c. Enter nose wheel well, disconnect and remove drain traps from aft side of station 102.00 bulkhead.
d. Using a cloth dampened with methyl ethyl ketone, Specification TT-M-261, remove all protective coating from station 102.00 bulkhead cracked area.
e. Perform a fluorescent penetrant inspection to determine extent of bulkhead crack. Refer to Section I for fluorescent penetrant inspection procedures.

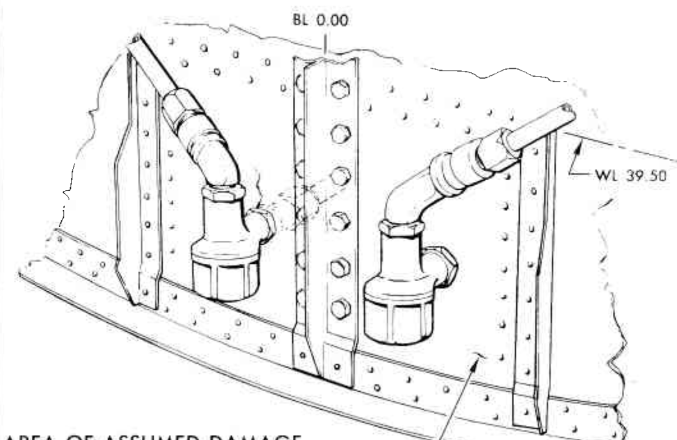
NOTE

IF OVERALL LENGTH OF ANY SINGLE CRACK EXCEEDS 1.25 INCH, CONSULT AN AERONAUTICAL STRUCTURES ENGINEER.

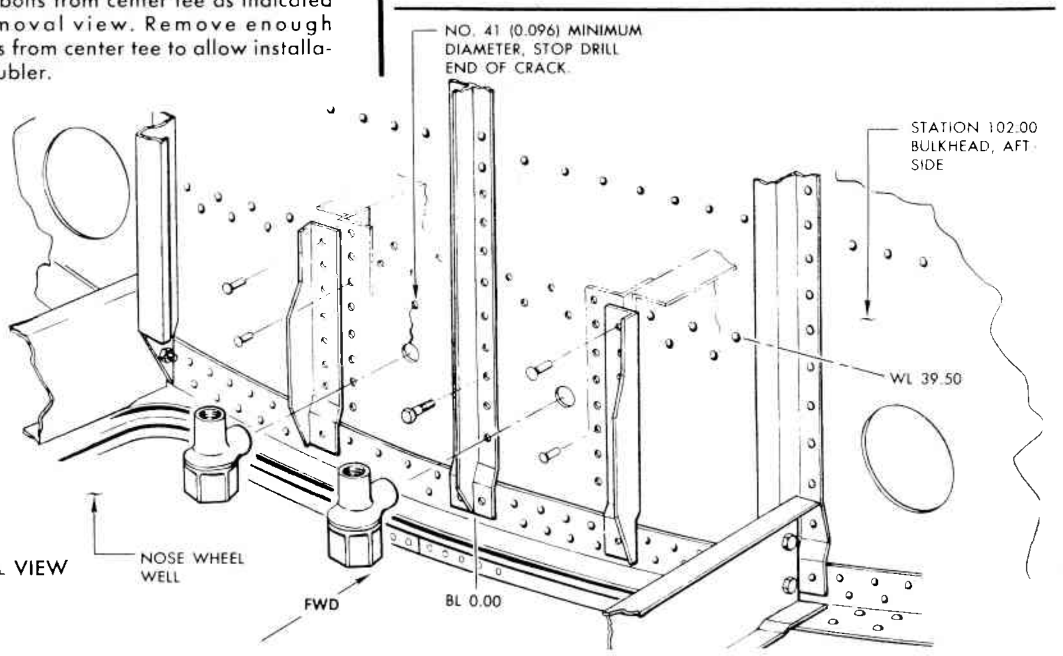
f. Stop drill crack and clean up edges.
g. Remove rivets indicated in damage removal view. Remove stiffener angles on aft side of station 102.00 bulkhead. Use a No. 30 (0.128) drill for 1/8-inch diameter rivets and a No. 20 (0.161) drill for 5/32-inch diameter rivets.
h. Remove NAS464 bolts from center tee as indicated in damage removal view. Remove enough AN470AD-5 rivets from center tee to allow installation of repair doubler.

REPAIR INSTALLATION PROCEDURE

a. Fabricate a doubler from 0.040 gage 7075-T6 bare sheet as shown in doubler fabrication detail.
b. Pick up rivet pattern from stiffener angles removed from aft side of station 102.00 bulkhead and transfer pattern to doubler as indicated in doubler fabrication detail and repair installation view. Lay out a two-row staggered rivet pattern at bottom of doubler as indicated in doubler fabrication detail. Predrill rivet holes. Use a No. 39 (0.099) drill to predrill 1/8-inch diameter rivet holes and a No. 30 (0.128) drill to predrill 5/32-inch diameter rivet holes.

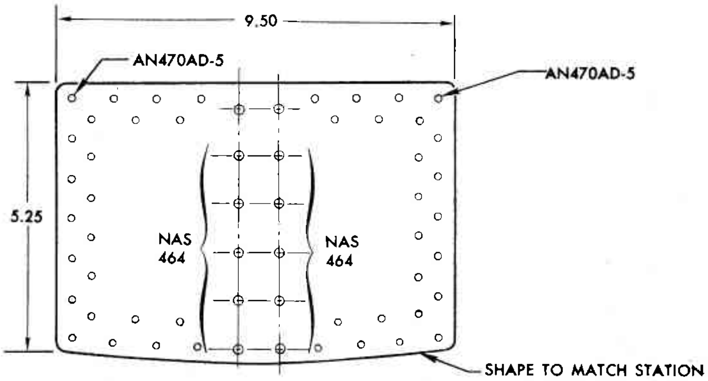


AREA OF ASSUMED DAMAGE, ENLARGED VIEW



DAMAGE REMOVAL VIEW

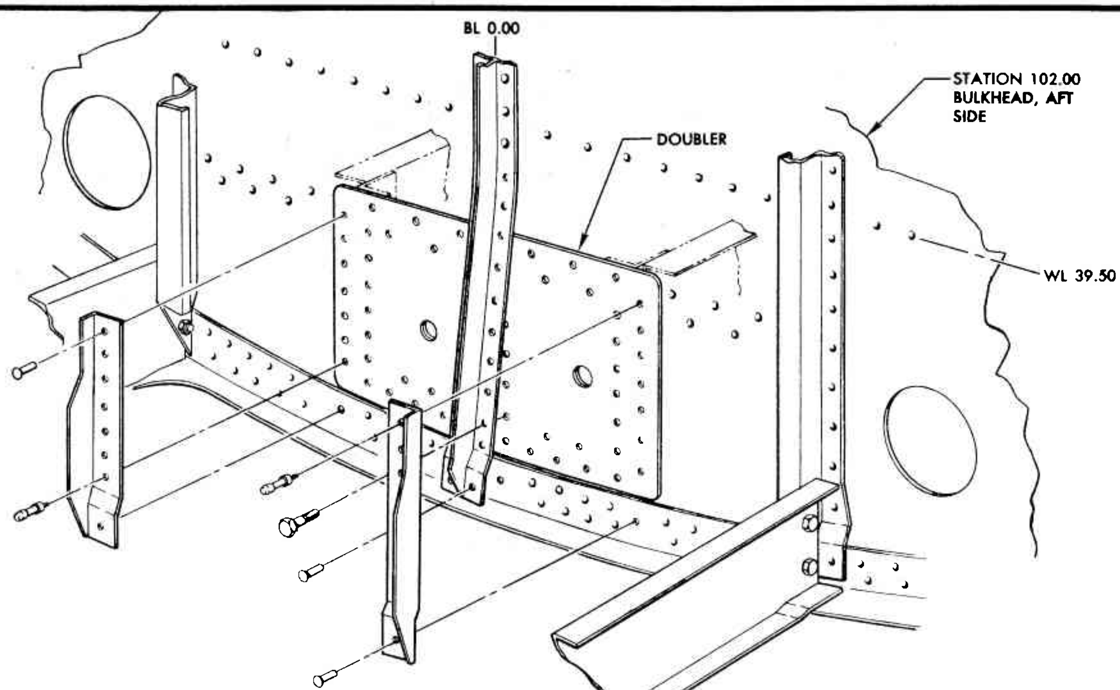
Figure 4-31. Airspeed Tubing Drain Trap Installation Repair — Station 102.00 Bulkhead (Sheet 1 of 3)



DOUBLER FABRICATION DETAIL.

NOTES:

1. FASTENER HOLES NOT INDICATED ON DOUBLER ARE FOR AN470AD-4 RIVETS.
2. NAS464 BOLT HOLES ON DOUBLER ARE LOCATED FROM EXISTING HOLES IN STATION 102.00 BULKHEAD CENTER TEE (SEE DAMAGE REMOVAL VIEW).
3. THE TWO ROWS OF AN470AD-4 RIVET HOLES AT THE TOP OF DOUBLER ARE LOCATED FROM EXISTING HOLES IN STATION 102.00 BULKHEAD AT WL 39.50 (SEE DAMAGE REMOVAL VIEW). REFER TO SECTION I FOR LOCATING BLIND HOLES.
4. THE TWO ROWS OF RIVETS AT THE LEFT AND RIGHT SIDES OF THE DOUBLER ARE LOCATED FROM EXISTING HOLES IN STATION 102.00 BULKHEAD AND FROM ANGLE STIFFENERS REMOVED FROM AFT SIDE OF BULKHEAD (SEE DAMAGE REMOVAL VIEW). REFER TO SECTION I FOR LOCATING BLIND HOLES.
5. THE TWO-ROW STAGGERED RIVET HOLE PATTERN AT THE BOTTOM OF DOUBLER SHALL BE IN ACCORDANCE WITH STANDARD MINIMUM RIVET SPACING IN SECTION I.



INSTALLATION VIEW

c. Install doubler and angles on the aft side of station 102.00 bulkhead with clecos, and predrill all rivet and NAS464 bolt holes in doubler as indicated in doubler fabrication detail and repair installation view. Use a No. 39 (0.099) drill to predrill 1/8-inch diameter rivet holes, and a No. 30 (0.128) drill for 5/32-inch diameter rivet holes, and a No. 15 (0.180) drill for NAS464 bolt holes.
d. Drill all rivet holes as shown in repair installation view. Use a No. 30 (0.128) drill for 1/8-inch diameter rivets and a No. 20 (0.161) drill for 5/32-inch diameter rivets. Ream 464 bolt holes in doubler to 0.1900 inch (+0.0015/-0.0000).
e. Enter area vacated by antenna, reach through station 94.75 frame holes, and scribe airspeed tubing holes on the back of attached doubler.
f. Remove doubler and angles from aft side of station 102.00 bulkhead. Drill airspeed tubing holes in doubler to sizes obtained from scribing in step "e".
g. Remove burrs from doubler and station 102.00 bulkhead.
h. Apply protective coating to doubler and station 102.00 bulkhead disturbed areas. See figure "Primer and paint Coatings" in Section I.
i. Apply a coat of sealer, Military Specification MIL-S-81733 to doubler faying surface.
j. Reinstall doubler as shown in installation view with clecos.
k. Install AN470AD-4 and -5 rivets as indicated in repair installation view.

NOTE

FOR ELONGATED OR OVERSIZED HOLES, RIVETS OF THE NEXT LARGER SIZE MAY BE USED PROVIDED MINIMUM RIVET EDGE DISTANCE AND SPACING REQUIREMENTS ARE MET.

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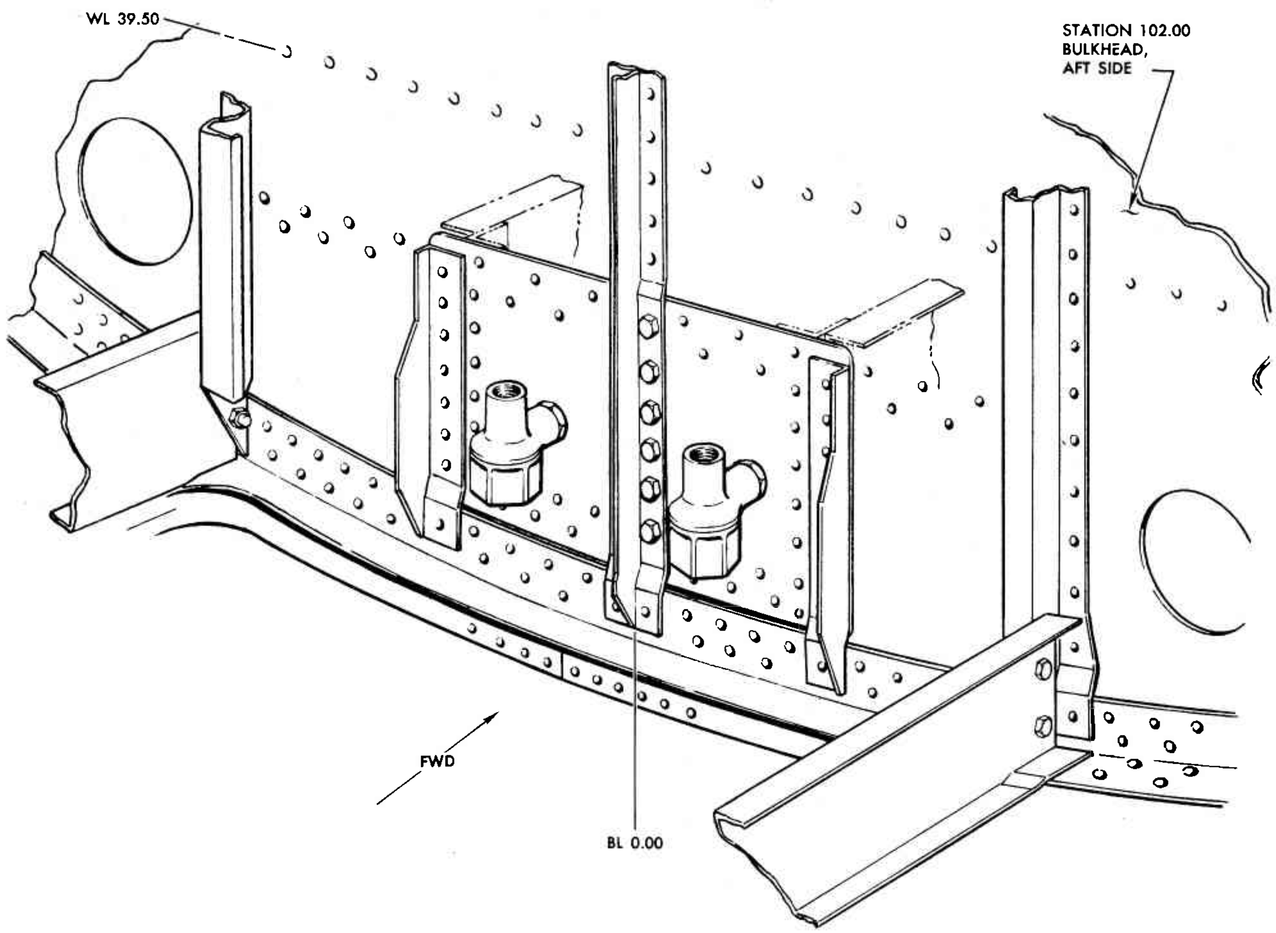
Figure 4–31. Airspeed Tubing Drain Trap Installation Repair–Station 102.00 Bulkhead (Sheet 2 of 3)



COMPLETED REPAIR

l. Install NAS464 bolts as indicated and repair installation view.

NOTE

NAS464 BOLTS SHALL BE REPLACED WITH NAS464 BOLTS HAVING ONE GRIP LENGTH LONGER THAN THOSE REMOVED.

m. Enter area vacated by antenna and clean out frame area extending from station 94.75 to 102.00.

n. Reinstall drain traps and airspeed tubing.

o. Remove old sealer from station 94.75 left and right cover faying surfaces.

p. Apply a new coat of sealer, Military Specification MIL-S-81733 to faying surfaces of station 94.75 left and right frame covers.

q. Reinstall ADF antenna and cover.

NOTE

CHECK ANTENNA COVER GASKET FOR SERVICEABILITY. IF A NEW GASKET IS REQUIRED, APPLY SEALER MILITARY SPECIFICATION MIL-S-8802 BETWEEN NEW GASKET AND COVER.

r. Perform pitot-static system operational check and leak test in accordance with applicable maintenance manual.

s. Perform operational check of systems affected by removal and reinstallation of ADF antenna in accordance with applicable maintenance manual.

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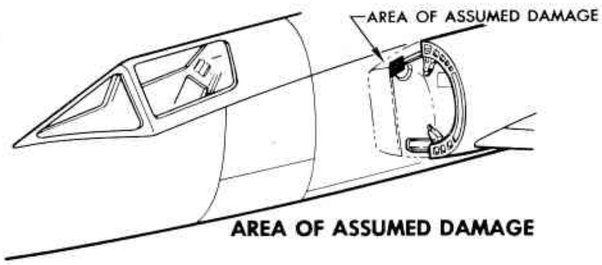
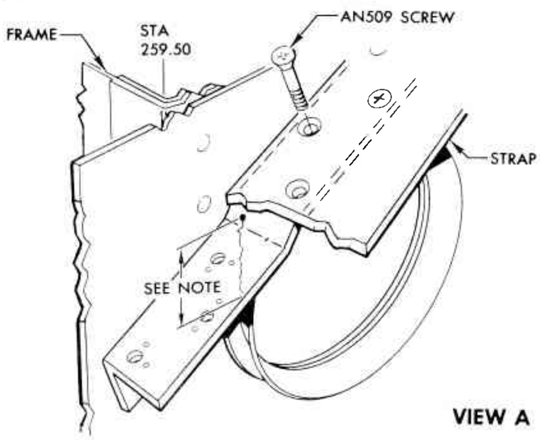
Figure 4–31. Airspeed Tubing Drain Trap Installation Repair–Station 102.00 Bulkhead (Sheet 3 of 3)





VIEW A

DAMAGE REMOVAL PROCEDURE

NOTE

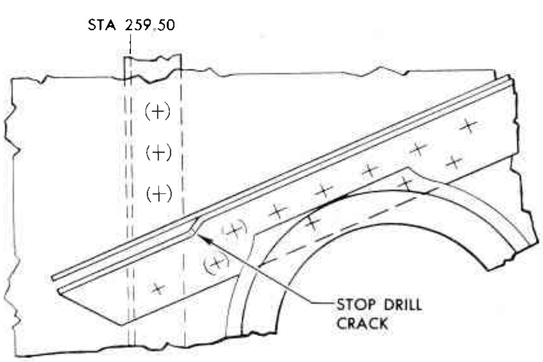
THIS REPAIR IS *APPLICABLE TO F-106A AIRPLANES* WHEN CRACKS EXIST AT FORWARD END OF SPLICE ANGLES.

a. Remove duct lip structure; place on padded bench.
b. Remove two AN509 screws immediately aft of splice angle joggle. See View A.
c. Remove all paint from damaged area; use a cloth dampened with methyl ethyl ketone, Specification TT-M-261.
d. Perform a fluorescent penetrant inspection to determine length of crack.

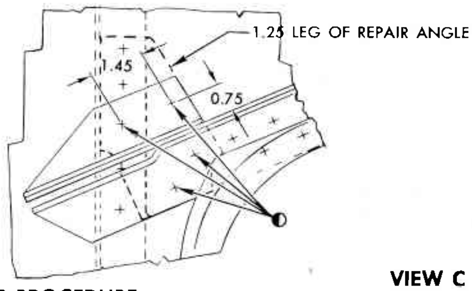
NOTE

IF LENGTH OF CRACK EXCEEDS 1.50 INCHES, FROM FORWARD END, USE REPAIR NO. 2. IF LENGTH OF CRACK EXCEEDS 5.00 INCHES, FROM FORWARD END, CONSULT AN AERONAUTICAL STRUCTURE ENGINEER.

e. Insert scrap material between angle and strap. Stop drill crack with a No. 15 (0.180) drill. See View B.



VIEW B



VIEW C

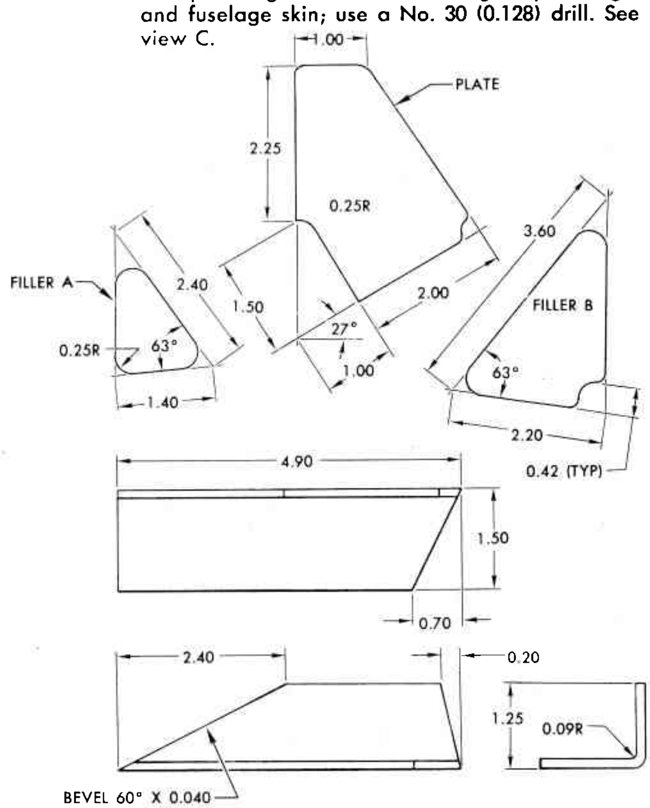
REPAIR PROCEDURE

a. Fabricate filler A from 0.030 gage 7075-T6 bare sheet material.

NOTE

2024-T4 SHEET MATERIAL MAY BE USED AS A SUBSTITUTE MATERIAL FOR FILLER A, FILLER B, AND PLATE.

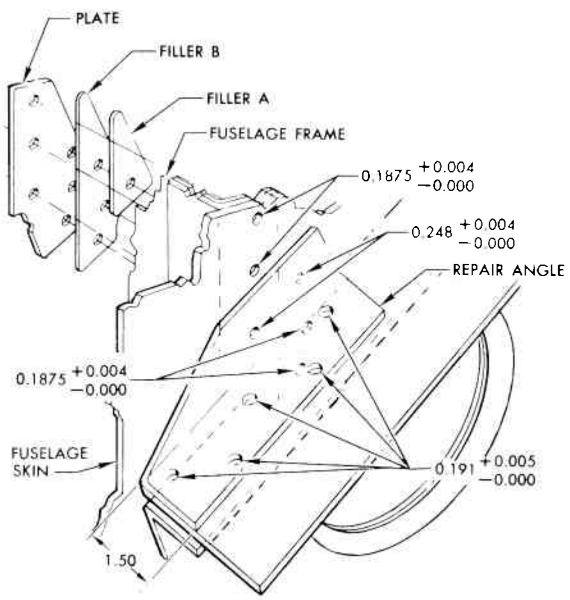
b. Fabricate filler B from 0.100 gage 7075-T6 bare sheet material.
c. Fabricate plate from 0.160 gage 7075-T6 bare sheet material.
d. Fabricate angle from 0.50 gage 302 corrosion resistant steel.
e. Drill out five existing rivets (indicated by (+); use a No. 30 (0.128) drill. See View B.
f. Position repair angle, as shown, and clamp in place. See View C.
g. Drill through repair angle to pick up existing rivet hole in fuselage skin and frame; use a No. 30 (0.128) drill.
h. Lay out pattern for second and third hole in 1.25 leg of repair angle. Drill holes through repair angle and fuselage skin; use a No. 30 (0.128) drill. See view C.



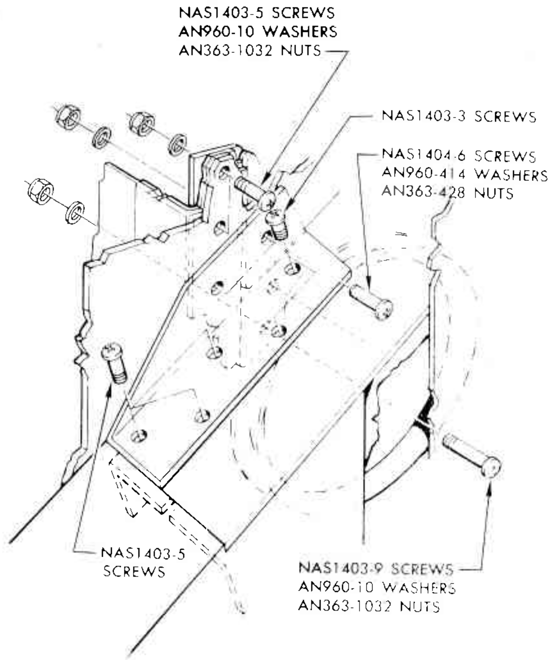
FABRICATED PARTS

.06.03.316-1

Figure 4-32. Fuselage Side Panel Splice Angle Crack Repair No. 1 (Sheet 1 of 2)



VIEW D



VIEW E

i. Position filler A, as shown in view D, and secure in place with masking tape.

NOTE

THE 63° ANGLE EDGE OF FILLER A SHALL BUTT AGAINST AFT EDGE OF FUSELAGE FRAME AND LOWER EDGE SHALL COINCIDE WITH TOP EXTERNAL STRAP.

j. Position filler B as shown in view D; route aft lower edge minimum amount to clear existing rivet.

NOTE

THE 63° ANGLE EDGE OF FILLER B SHALL BUTT AGAINST AFT EDGE OF FUSELAGE FRAME AND TOP, AND AFT EDGES SHALL COINCIDE WITH TOP AND AFT EDGES OF FILLER A.

k. With filler A and filler B positioned per steps "i" and "j," use No. 30 (0.128) drill to drill holes through filler indicated by ◐ in view C. Remove fillers.

l. Clamp filler B to plate so that aft and lower edges of plate and filler coincide. Using filler B as a jig; drill 4 holes in plate with a No. 30 (0.128) drill.

m. Position filler A, filler B, and plate; secure in place with clecos.

n. Using the forward hole in the repair angle and existing rivet holes in skin and fuselage frame as a jig, drill 3 holes in plate with a No. 30 (0.128) drill.

o. Drill and line ream 2 existing holes in repair angle, filler B, and plate to 0.248 (+0.004, −0.000) inch. See View D. Install two NAS1404-6 screws, with AN960-414 washer, and AN363-428 nut.

06.03.316-2

p. Drill and line ream remaining holes to 0.1875 (+0.004, −0.000) inch. See view D.

q. On 1.50 leg of repair angle, pick up existing holes in strap and splice angle, final drill to 0.191 (+0.005, −0.000) inch. See view D.

r. Fabricate fillers for strap countersinks by drilling off heads from AN509-10R screws; use a No. 10 (0.1935) drill.

s. Remove repair parts; remove burrs.

t. Clean all repair parts and damage area of splice angle with aliphatic naphtha, Federal Specification TT-N-95.

u. Paint repair parts and damage area of splice angle as indicated in T.O. 1F-106A-23.

v. Remove paint from close tolerance holes in repair parts.

w. Install fillers (heads of AN509-10R screws) in strap countersinks with wet prime.

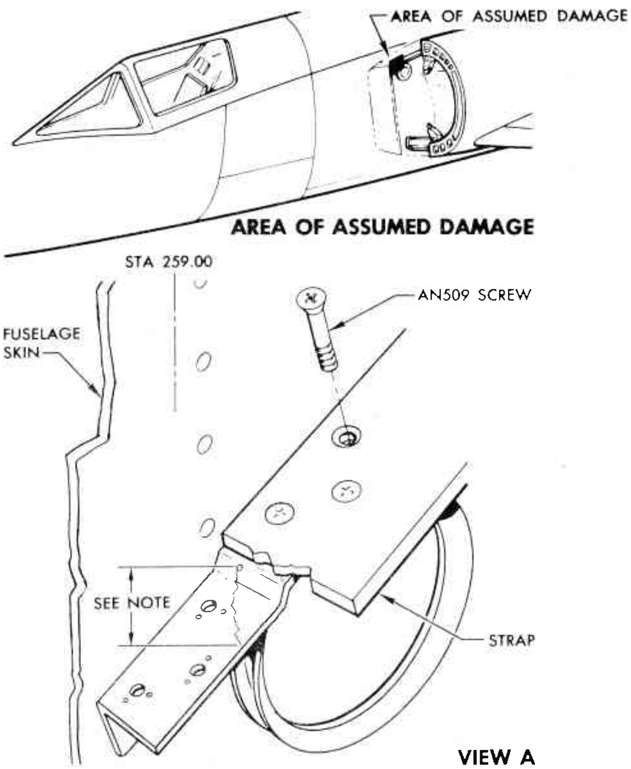
x. Install repair parts with screws, washers, and nuts, see view E.

NOTE

NAS221 and NAS222 SCREWS MAY BE SUBSTITUTED FOR NAS1403 AND NAS1404 SCREWS RESPECTIVELY.

y. Install duct lip structure.

Figure 4-32. Fuselage Side Panel Splice Angle Crack Repair No. 1 (Sheet 2 of 2)



DAMAGE REMOVAL PROCEDURE

NOTE

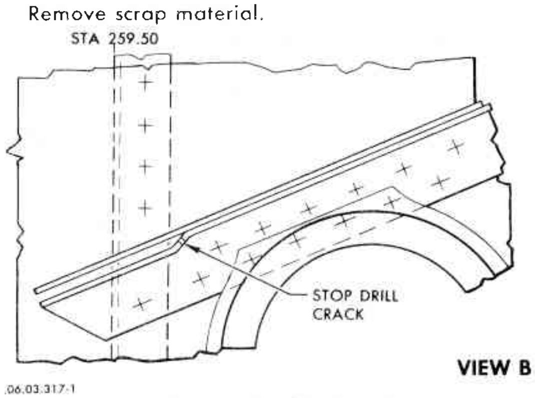
THIS REPAIR IS APPLICABLE TO F-106A AIRPLANES WHEN CRACKS EXIST AT FORWARD END OF SPLICE ANGLES.

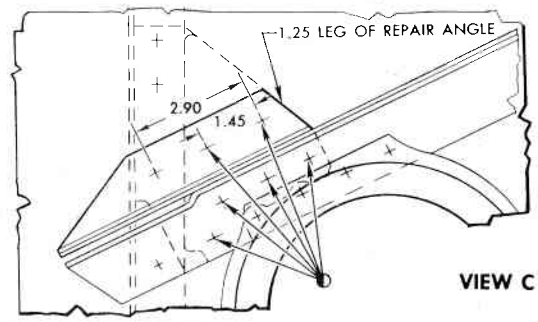
a. Remove duct lip structure; place on padded bench.
b. Remove four AN509 screws immedately aft of splice angle joggle. See view A.
c. Remove all paint from damaged area; use a cloth dampened with methyl ethyl ketone, Specification TT-M-261.
d. Perform a fluorescent penetrant inspection to determine length of crack.

NOTE

IF LENGTH OF CRACK IS LESS THAN 1.50 INCHES FROM FORWARD END, USE REPAIR NO. 1. IF LENGTH OF CRACK EXCEEDS 5.00 INCHES FROM FORWARD END, CONSULT AN AERONAUTICAL STRUCTURE ENGINEER.

e. Insert scrap material between angle and strap. Stop drill crack with a No. 15 (0.180) drill. See view B. Remove scrap material.





REPAIR PROCEDURE

a. Fabricate filler A from 0.300 gage 7075-T6 bare sheet material.

NOTE

2024-T4 SHEET MATERIAL MAY BE USED AS A SUBSTITUTE MATERIAL FOR FILLER A, FILLER B, AND PLATE.

b. Fabricate filler B from 0.100 gage 7075-T6 bare sheet material.
c. Fabricate plate from 0.160 gage 7075-T6 bare sheet material.
d. Fabricate angle from 0.50 gage 302 corrosion resistant steel.
e. Drill out eight existing rivets (indicated by (+); use a No. 30 (0.128) drill. See view B.
f. Position repair angle, as shown, and clamp in place. See view C.
g. Drill through repair angle to pick up existing rivet hole in fuselage skin and frame; use a No. 30 (0.128) drill.
h. Lay out pattern for second and third hole in 1.25 leg of repair angle. Drill holes through repair angle and fuselage skin; use a No. 30 (0.128) drill. See view C.

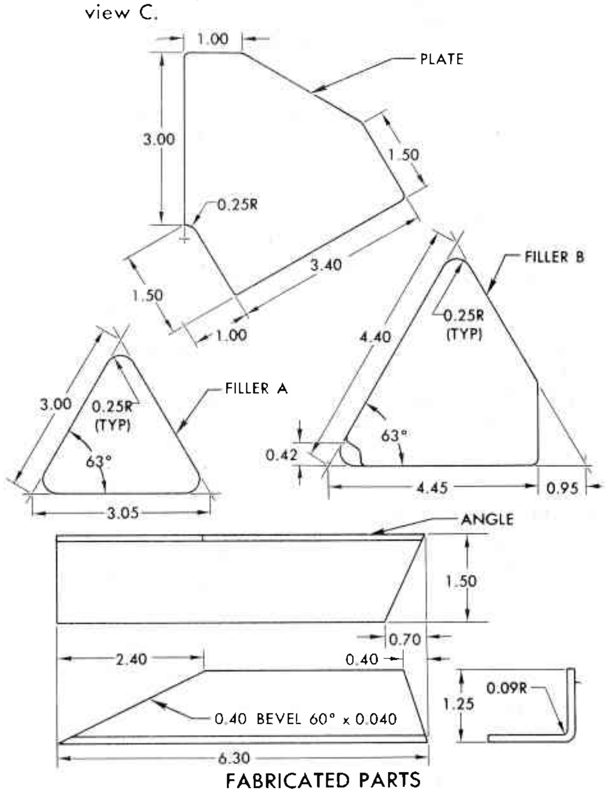
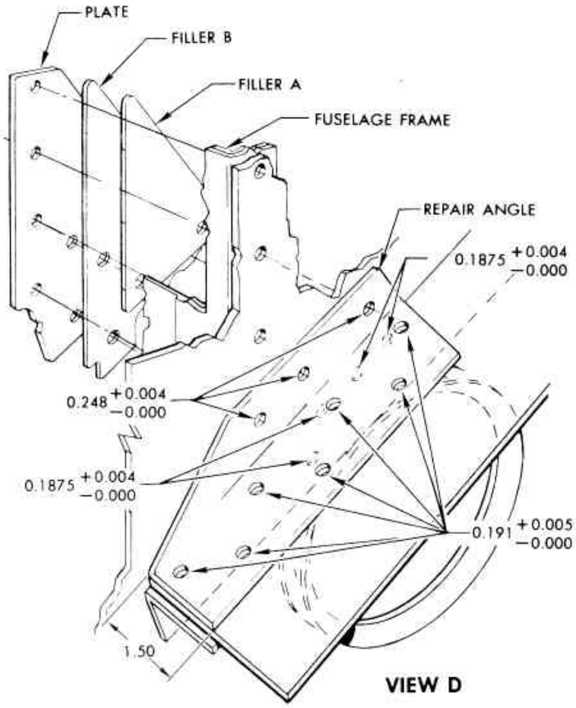
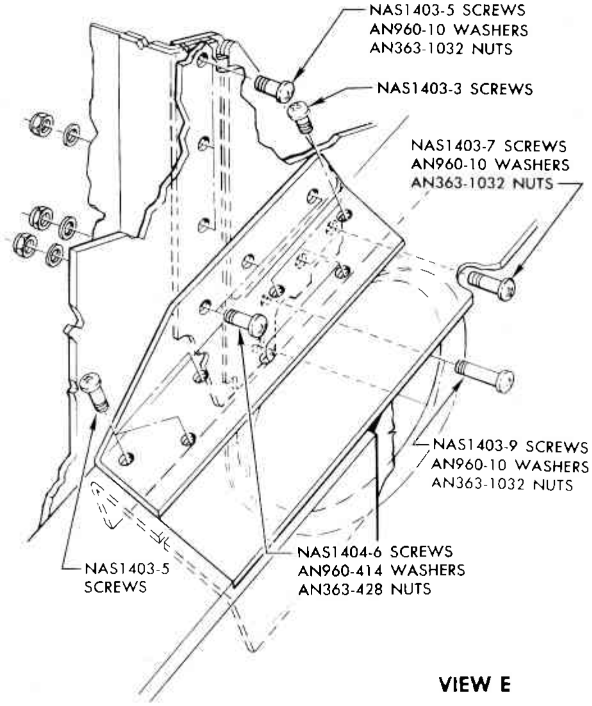


Figure 4-33. Fuselage Side Panel Splice Angle Crack Repair No. 2 (Sheet 1 of 2)



VIEW D



VIEW E

i. Position filler B as shown in view D; route aft lower place with masking tape.

NOTE
THE 63° ANGLE EDGE OF FILLER A SHALL BUTT AGAINST AFT EDGE OF FUSELAGE FRAME AND LOWER EDGE SHALL COINCIDE WITH TOP OF EXTERNAL STRAP.

j. Positionfiller B as shown in view D; route aft lower edge minimum amount to clear existing rivets.

NOTE
THE 63° ANGLE EDGE OF FILLER B SHALL BUTT AGAINST AFT EDGE OF FUSELAGE FRAME AND THE LOWER EDGE SHALL COINCIDE WITH THE LOWER EDGE OF SPLICE ANGLE.

k. With filler A and filler B positioned per steps "i" and "j" use No. 30 (0.128) drill to drill holes indicated by ◐ in view C. Remove fillers.
l. Position plate on fuselage frame, as shown in detail C. Use a pencil to mark location of aft edge of fuselage frame on plate. Remove plate and draw line between marks.
m. Clamp filler B to plate so that lower edge of plate and filler coincide and that forward edge of filler coincides with pencil mark on plate. Using filler B as a jig, drill 6 holes in plate with a No. 30 (0.128) drill.
n. Position filler A, filler B, and plate; secure in place with clecos.
o. Using the forward hole in the repair angle and existing rivet holes in skin and fuselage frame as a jig, drill 4 holes in plate with a No. 30 (0.128) drill.

06 03 317 2

p. Drill and line ream 3 existing holes in repair angle, filler B, and plate to 0.248 (+0.004, −0.000) inch. See view D. Install two NAS1404-6 screws, with AN960-414 washer, and AN363-428 nut.
q. Drill and line ream remaining holes to 0.1875, (+0.004, −0.000) inch.
r. On 1.50 leg of repair angle, pick up existing holes in strap and splice angle, final drill to 0.191 (+0.005, −0.000) inch. See view D.
s. Fabricate fillers for strap countersinks by drilling off heads from AN509-10R screws; use a No. 10 (0.1935) drill.
t. Remove repair parts; remove burrs.
u. Clean all repair parts and damage area of splice angle with aliphatic naphtha, Federal Specification TT-N-95.
v. Paint repair parts and damage area of splice angle as indicated in T.O. 1F-106A-23.

w. Remove paint from close tolerance holes in repair parts.
x. Install fillers (heads of AN509-10R screws) in strap countersinks with wet prime.
y. Install repair parts with screws, washers, and nuts, see view E.

NOTE
NAS221 AND NAS222 SCREWS MAY BE SUBSTITUTED FOR NAS1403 AND NAS1404 SCREWS RESPECTIVELY.

z. Install duct lip structure.

Figure 4-33. Fuselage Side Panel Splice Angle Crack Repair No. 2 (Sheet 2 of 2)

4-43. Fuselage Integral Fuel Tank Pressure Test.

4-44. The fuselage integral fuel tank ("F" tank) shall be pressure tested as follows after completion of repairs involving fuel-tightness.

a. Open the air-conditioning compartment access doors. See figure 1-2.

b. Disconnect and cap the following lines which are located on the aft, upper left-hand side of the "F" tank fuel-tight bulkhead at station 253.00. (Refer to Section II for equipment required, and to T.O. 1F-106A-2-5-2-1 for the fuel system schematic.)

1. One 0.25-inch pressure sense line leading to the "F" tank air pressure regulator.
2. Two 0.25-inch ambient sense lines leading from the "F" tank to the ambient sense ports.
3. One 0.25-inch fuel pressure line leading to the refueling pilot valve.
4. One 0.38-inch air pressure line to the left and right pilot float valves.
5. One 0.38-inch air pressure line to the check valve manifold and the high-level pilot float valve.
6. One 0.38-inch air pressure line from the high level pilot float valve.
7. One 0.38-inch emergency air pressure line to the left and right CG fuel shutoff valves.
8. One 4.50-inch fuel transfer line (disconnect and cap in left-hand main landing gear wheel well).
9. One 0.38-inch air pressure line leading to forward and aft vent valves.
10. One 1.50-inch air pressure vent line.

c. Remove the "F" tank access door.

d. Install an air test dummy door on the "F" tank opening in place of the access door removed in the previous step. See figure 2-33 for general requirements for door fabrication.

e. Deleted.

f. Connect a regulated air pressure line not exceeding 20 psi from the source to the inlet fitting on the air test dummy door.

g. Deleted.

CAUTION

Close all valves tightly on mercury manometer test stand before connecting air pressure line from air pressure source.

h. Deleted.

i. Deleted.

j. Slowly open the valve on the 20-psi air pressure line and pressurize the "F" tank to 7.0 (±0.5) psi.

k. Apply bubble fluid to the repair area and check for leaks.

l. If a leak is detected, relieve all air pressure from "F" tank, remove the air test dummy door and repair the leak. Replace the air test dummy door and recheck for leaks by repeating steps "j" and "k."

m. Increase pressure to 10.0 (±0.5) psi.

n. Apply bubble fluid and check for leaks. If a leak is detected, repair as directed in step "l."

WARNING

High pressures are used in the following steps. Before proceeding with remainder of test, rope off the pressure test area, and hang warning signs indicating that a pressure test is in progress. Remove personnel from area not actually engaged in the pressure test. Place a steel cargo net over the "F" tank and tie down with ropes under the fuselage. Personnel performing this test shall work from behind safety screens or equivalent while accomplishing steps "p" through "s."

o. Introduce a small amount of Freon 12 gas into the "F" tank.

NOTE

Check for escaping Freon 12 gas around air inlet fittings and pressure relief valves. Escaping gas will contaminate air around tank and make test invalid.

p. Increase air pressure in "F" tank to 15 (±0.5) ps .

q. Check for leaks using a General Electric H-2 medium scale leak detector.

r. If a leak is detected, reduce the air pressure in the "F" tank to 0 psi, remove air test dummy door and repair all leaks. Replace the air test dummy door and recheck for leaks by repeating steps "p" and "q." Repeat this process until no leaks are detected.

s. If no leaks were detected in step "q," increase pressure to 20 (±0.5) psi and hold for three minutes.

t. Reduce pressure to 10 (±0.5) psi and check for leaks by application of bubble fluid.

u. If a leak is detected while performing step "t," relieve the "F" tank of all air pressure, remove the steel cargo net and air test dummy door and thoroughly check the repair for a structural failure. If it is determined that a structural failure has occurred, completely rework the repair and perform a new pressure test by repeating steps "o" through "t." If a leak is detected in the sealant, apply a fillet of sealant, Military Specification MIL-S-8802, over leak on fuel side of tank and retest by repeating steps "m" and "n." In no case shall leaks be permitted upon completion of pressure testing.

v. If no leaks were detected after step "t," remove the steel cargo net, air test dummy door, and all test equipment from area.

w. Remove all pressure caps from lines that were capped in step "b."

CAUTION

Failure to remove all caps and plugs used during test could result in structural damage to fuel cell or failure of fuel system components during subsequent fueling, defueling and fuel system check out operations.

x. Reconnect all lines that were disconnected in step "b."

y. Replace "F" tank access door removed in step "c."

z. A fuel system operational check must be performed upon completion of step "y". Refer to T.O. 1F-106A-2-5-2-1 for procedure.

4-45. Fuselage Forward Intermediate Section Repairs.

4-46. The temper designations and physical characteristics of the material used in the fuselage forward intermediate section are varied; therefore, each part should be given individual consideration when making repairs. For repairs applicable to the various fuselage components in this area, reference should be made to figures 1-22 through 1-24, to applicable figures in the section, and to Section X for typical repair information. Due to the stresses involved in the engine air intake ducts, any repairs made to these ducts must conform to the original strength of the component being repaired. Any repairs to the variable ramp intake panels normally will require removal of these panels. Removal and replacement instructions of the variable ramp panels are given in Technical Order (TO) 1F-106A-2-4-2-1. See figure 4-19A for variable ramp panel hinge node and pin retention repair procedures. Peeled nickel plate on the variable ramp duct leading edge may be repaired as follows:

a. Grind off the peeled nickel plate from the vari-ramp duct edge (Duct Station 246.00). The nickel is a 7 5/8 inch wide "band" around the duct. Follow the grinding procedure outlined in TO 1-1-2.

(1) Use only aluminum oxide abrasive wheel with 120 grain size abrasive.

(2) Observe grinding speed limit for the particular size of wheel used.

(3) Grind off completely the peeled nickel plate. Grind lightly the nickel plated area with good bonding to remove oxide.

CAUTION

Use goggles or face shield during grinding operation.

b. Wash the reworked area with alkaline waterbase aircraft cleaner in accordance with T.O. 1-1-1 (Section III).

c. Rinse with cold water and dry.

d. Mask the adjacent area 7 5/8 inches from edge of the duct. Use barrier paper per Specification MIL-B-121 and masking tape per Specification UU-T-106, or equivalent according to T.O. 1-1-8.

e. Treat the repair area with brush-on conversion coating for aluminum alloys per Specification MIL-C-5541.

f. Cold water rinse.

g. Deleted.

h. Apply two coats of epoxy-polymide primer per Specification MIL-P-23377 according to TO 1-1-8.

i. Apply two coats of polyurethane protective coating (gray color) per Specification MIL-C-83286.

NOTE

Damage exceeding those illustrated shall be referred to an aeronautical structures engineer.

Repairs to the outer surface of the variable ramp panels in all cases must conform to aerodynamic smoothness requirements. Refer to Section I for rivet specifications and tolerances in this area. The holes in the engine cooling air scroll are not only designed to permit the flow of bleed air, but are designed to such size as to prevent foreign material, such as nuts, bolts, etc., from entering the engine air cooling system. Therefore, when a crack is detected in the perforated area of the engine cooling air scroll, drills used for stop drilling and router bits used for routing must be held to a maximum size of 0.125-inch diameter. If a crack is detected that runs from one hole to any adjacent hole, it is permissible to rout out the crack, leaving a slotted area between the two holes. This type of repair must be held to one routed slot to every two square inches of perforation. For repair of damage in excess of this permissible repair and in excess of the negligible damage limits shown in Table 4-1, repair damage in accordance with Figures 4-20 and 4-20A. All repairs of the fuselage plating in the fuselage forward intermediate section and to the plating of the missile bay doors must be of the flush-type repair to conform to the aerodynamic smoothness requirements. See figures 4-34 and 4-35 for repair of refueling and ram air turbine access loors and figures 4-36 through 4-38 for repair of missile bay doors.

4-46A. Inspection, Dehydration, Repair and Seal Surfaces Containing Honeycomb Core Applicable to Ram Air Turbine Door will be accomplished as follows:

a. Remove ram air turbine door in accordance with instructions contained in T.O. 1F-106A-2-2-2-2.

b. X-ray ram air turbine door to determine if moisture is present and/or core is damaged.

c. Utilizing low temperature boil procedure, dehydrate assembly using set moisture removal FSN 4920-ND00062P. Vacuum shall be between 10 psig to 12 psig to assure boiling of water at as low as 160°F and temperature shall not exceed 180°F. 4 to 8 hours may be required for dehydration of core assembly, dependent on the amount of water in core.

NOTE

A vacuum chamber capable of safely maintaining 12.3 psig vacuum at 160°F to 180°F with adequate safety provisions to assure preventing excess temperature or collapsing from excess vacuum may be used as a suitable substitute.

d. Re-x-ray assembly to determine that all moisture has been removed.

e. When all moisture has been removed from assembly or x-ray reveals no moisture present, repair any damage in accordance with applicable procedures outlined in this technical order.

f. Reinstall ram air turbine door in reverse order of removal.

4-46B. REPAIR OR REPLACEMENT OF SKIN AND CORE OF RAM AIR TURBINE DOOR - DEPOT LEVEL ONLY.

WARNING

Accomplish all work in authorized areas with proper protection devices to safeguard health. Work involving application of potentially harmful materials must be accomplished in an exhaust ventilated area.

a. Replacement of the skin.

(1) Remove the damaged skin as follows:

(a) Remove fasteners AS REQUIRED.

(b) Remove the skin using dry ice and wooden wedges.

(2) Clean any fuel contaminated honeycomb with trichloroethylene; invert and dry for 8 hours at 180°F.

(3) Repair the core as necessary.

(4) Vapor degrease the assembly.

(5) Make a new skin from 2024-T6, clad, 0.020 inch. Use the old skin as a template.

(6) Clean the skin for bonding in the plating shop.

(7) Bond the skin as follows:

(a) Use MIL-A-25463, Type I, Class 2 film adhesive. Align the skin to the pan with tack rivets.

(b) Install the door in a bond form, vac-bag and put into the autoclave.

(c) Retain in the autoclave for 1 hour at 250°F and 15 psig net.

(d) Cool for 1 hour. Remove all the vac-pac materials.

(8) Grind off the excess adhesive.

(9) Install fasteners AS REQUIRED

(10) Inspect the door for honeycomb bond integrity using tap test. TEST SPECIMEN IN ACCORDANCE WITH PARA 4-46B c.

b. Replacement of the skin and core.

(1) REMOVE ~~Replace~~ one skin as specified above.

(2) Remove and replace the core as required. Cut the core on the Do-all saw. Use adhesive, foaming core splice in accordance with AMS 3688 to fill voids around edges of core. Clean skin, core and pan for bonding in the plating shop. Follow steps a.(4) thru a.(10) above.

c. Prepare a lap shear specimen with the same adhesive used in the honeycomb repair and cure it simultaneously with the repair piece. Use the lap shear specimen tester in the shop. A minimum value of 2000 psi must be attained.

4-46C. REPAIR OF ELECTRONIC COOLING CHECK VALVE FLAPPER HINGE HOLES. Elongated hinge holes of the electronic cooling check valves flapper will be repaired in accordance with figure 4-39B.

4-47. Fuselage Aft Intermediate Section Repairs.

4-48. The temper designations and physical characteristics of the material used in the fuselage aft intermediate section are varied; therefore, each part should be given individual consideration when making repairs. All repairs to the fuselage plating in the fuselage aft intermediate section must be of the flush-type to conform to the aerodynamic smoothness requirements. Due to the type of material used in the construction of the drag chute housing and speed brake door plating, an aeronautical structures engineer should be consulted before making a repair. In any case, all repairs must provide airtightness to the drag chute housing structure. For repairs applicable to the fuselage aft intermediate section components, see figures 1-22 through 1-24, applicable figures in this section, and refer to Section X for typical repair information. Refer to Table 4-I for disposition of negligible damage to the components of this section of the fuselage. Repairs to, or replacement of, the gaskets used in sealing

4-48A. Fin/Fuselage Drag Angles, Part nos 8-74103-151 and -152 Repair.

4-48B. Replace defective fin/fuselage drag angles with like angles fabricated from 7075-0 material with a bend radius increased to 0.12 and heat treat to T6 condition after fabrication.

4-49. Fuselage Lower Longeron Repairs.

4-50. See Figure 9-7 for crack tolerances and cross section damage limits. Because of the wide difference in cross section area and the stress loads imposed upon the longeron, repair of areas not covered by crack and damage tolerances in Figure 9-7 must be designed by an aeronautical structures engineer.

4-51. Fuselage Tail Cone Repairs.

4-52. The fuselage tail cone is constructed of aluminum and titanium materials and any repairs in this area must be made of the same material as the original part. All repairs to the plating of the tail cone must be of the flush type in order to conform to the aerodynamic smoothness requirements. Refer to Table 4-1 for the neglible damage limits in the tail cone area and for disposition of damage classified as negligible. Should crack damage occur within a spot weld, repair damage by stop drilling crack with a Number 40 or equivalent drill bit. Cracks that extend beyond spot weld circumference must be removed and patched. There shall be no more than three stop drill repairs per 12 inches of seam. Figures 1-22 through 1-24 give methods for typical repairs for the various components in the tail cone. Refer to Section X for typical repair information and to Section XI for repair materials. See figure 4-42 for tail cone cracked frame repair procedure and figure 4-43 for tail cone alignment procedures.

4-52A. DORSAL AERIAL REFUEL CAVITY LEAK CHECK.

4-52B. Equipment Required.

FIGURE	NAME	TYPE	ALTERNATE	USE
4-33A	Aerial Refuel Cavity Test Fixture Kit.	66J40057-1 (F-106A) 66J40058-1 (F-106B)		To leak check the dorsal aerial refuel cavity.
	Regulated Air Pressurize Source			To pressurize test area.
	Drainage Receptacle	20 gallons capacity		Receiver for test fluid.
	Distilled Water (20 gallons).			Leak test fluid.
	Analine Dye (red).			To color leak test fluid.

NOTES:
1. Control Value No. 20-1/8 NPT, Dragon Engr. Co., Norwalk, Calif.
2. Air Regulator No. R2025-LP, Hannifin Cylinder Div. of Parker-Hannifin Co., Des Planes, Ill.
3. Gage No. Fig. 724-4-1/2, U.S. Gage Div. of Ametek Inc., Sellersville, Pa.
4. Safety Valve No. 790RP-2.5, Halkey-Roberts Co., Paramus, N.J.

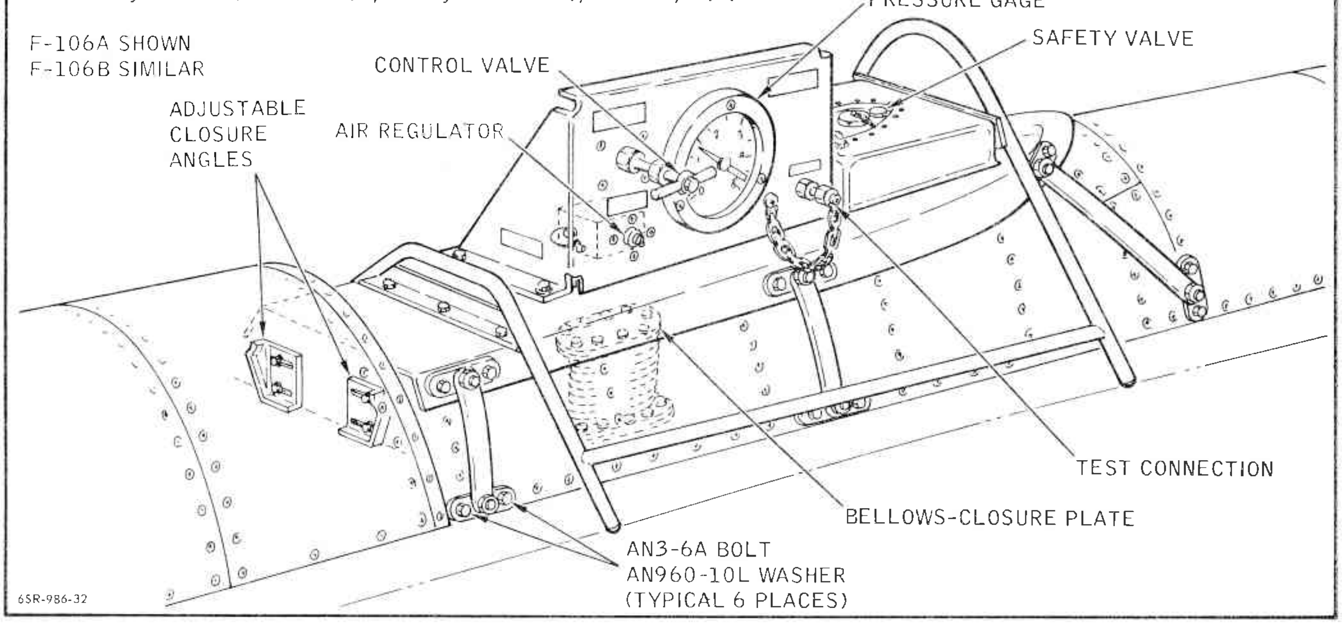


Figure 4-33A. Aerial Refueling Dorsal Cavity Leak Tester Installation.

4-52C. Preparation, F-106A.

a. Remove aerial refueling slipway door. Leave door actuator boot installed on airplane. Refer to T.O. 1F-106A-2-5-2-1 for this procedure.

b. Open refrigeration compartment access door.

c. Apply sealant MIL-S-8802 to flanged surface of slipway actuator boot. Install 66J40057-4 cover plate on boot using AN3-6A bolts (8).

d. Loosen bolts attaching 66J40057-5 and -6 bracket assemblies to forward end of test fixture; slide brackets inboard.

NOTE

If repair to the fiberglass portion of the test fixture is required, refer to Section I for fiberglass repair information.

e. Carefully position test fixture over the dorsal slipway cavity. Insure that the test fixture is bearing on the dorsal surface aft of the refueling cavity area.

f. Align test fixture hold-down straps as shown in figure 4-33A. Remove dorsal attachment screws (12) where straps attach; hold for reinstallation. Secure straps using bolts AN3-6A(12) and washers AN960-10(12). Do not torque bolts beyond 25 inch pounds.

g. Finger tighten strap nuts.

h. Slide 66J40057-5 and -6 bracket assemblies outboard until the fixture seals compress approximately 3/16 inch against sides of the dorsal pan. Tighten bracket attachment bolts.

i. Starting with the fixture aft straps, tighten each strap nut approximately eight turns. Tighten forward straps, then the center straps.

j. Close test fixture air shutoff valve and verify that the filler plug is tight.

k. Turn fixture air regulator adjustment screw several turns counterclockwise; do not remove screw.

l. Connect regulated external air (250 psi maximum) pressure source to fixture control panel.

m. Connect hose AN6270-4-60 to adapter 66J40057-3 using union MS24392-4. Install plug MS24404-4 in hose open end.

n. Install adapter assembly (step m) on fuselage overboard drain located on right side of fuselage at station 370.70 using MIL-S-8802 as attachment adhesive. Support hose weight by taping to fuselage skin. Place end of hose in 20 gallon container.

4-52D. Preparation, F-106B.

a. Remove aerial refueling slipway door. Refer to T.O. 1F-106A-2-5-2-1 for this procedure.

b. Open refrigeration compartment access doors.

c. Apply sealant MIL-S-8802 to flanged surfaces of slipway door actuator boot. Install cover plate 66J40058-3 on boot using AN3-6A bolts (9). Reinstall boot to dorsal pan assembly.

d. Loosen bolts attaching 66J40058-5 and -6 bracket assemblies to test fixture; slide brackets inboard.

NOTE

If repair to the fiberglass portion of the test fixture is required, refer to Section I for fiberglass repair information.

e. Carefully position test fixture over the dorsal slipway cavity. Insure that the test fixture is bearing on the dorsal surface aft of the refueling cavity area.

f. Align test fixture hold-down straps as shown in figure 4-33A. Remove dorsal attachment screws (12); hold for reinstallation. Secure straps using bolts AN3-6(12) and washers AN960-10(12). Do not torque bolts beyond 25 inch pounds.

g. Finger tighten strap nuts.

h. Slide 66J40058-5 and -6 bracket assemblies outboard until the fixture seals compress approximately 3/16 inch against the sides of the dorsal pan. Tighten bracket attachment bolts.

i. Starting with the fixture aft straps, tighten each strap nut approximately eight turns. Tighten forward straps, then the center straps.

j. Close test fixture air shutoff valve and verify that the filler plug is tight.

k. Turn fixture air regulator adjustment screw several turns counterclockwise; do not remove screw.

l. Connect regulated external air (250 psi maximum) pressure source to fixture control panel.

m. Connect hose AN6270-6-120 to adapter 66J40058-7 using union MS24399-8. Install plug MS24404-6 hose open end.

n. Install adapter assembly (step m) on dorsal left hand overboard drain using MIL-S-8802 as attachment adhesive. Support hose weight by taping to fuselage skin. Place end of hose in 20 gallon container. Plug dorsal left hand overboard drain using plug 66J40058-24.

4-52E. Procedure.

a. Slowly open air shutoff valve while monitoring fixture pressure gage.

NOTE

If fixture pressure gage reading exceeds 2.0 psig, immediately close air shutoff valve and turn regulator adjustment screw counter-wise; do not remove screw.

b. Slowly turn regulator adjustment screw clockwise until pressure gage indicates 2.0 psig.

c. Check test area for indications of leaks. Note areas of leakage.

d. Close air shutoff valve and slowly remove filler bleed plug to vent air pressure.

e. Remove test fixture and repair leaks. Clean area for sealant application using aliphatic naptha, Federal Specification TT-N-95; dry area. Warm area with heat lamp to approximately 100°F-130°F. Apply sealant, Specification MIL-S-8802 to areas that require sealing.

f. Repeat preparation procedure steps e through l.

g. Repeat steps a through c of this procedure until all obvious leaks have been repaired. Reduce pressure to zero.

h. Fill test fixture - dorsal cavity area with approximately 20 gallons of test fluid consisting of distilled water and enough red aniline dye to well color the water.

i. Close fixture bleed plug, open air shutoff valve and pressurize test area to 2.0 psig pressure.

j. Check outer surfaces of cavity area for leaks. No leakage is allowable through the aft bulkhead area. A quantity of 18 drops per minute maximum total is allowable through all other areas when pressurized at 2.0 psig for five minutes.

k. If leakage above specified limits is found, reduce air pressure to zero, remove plug from drain hose and drain fluid into the container.

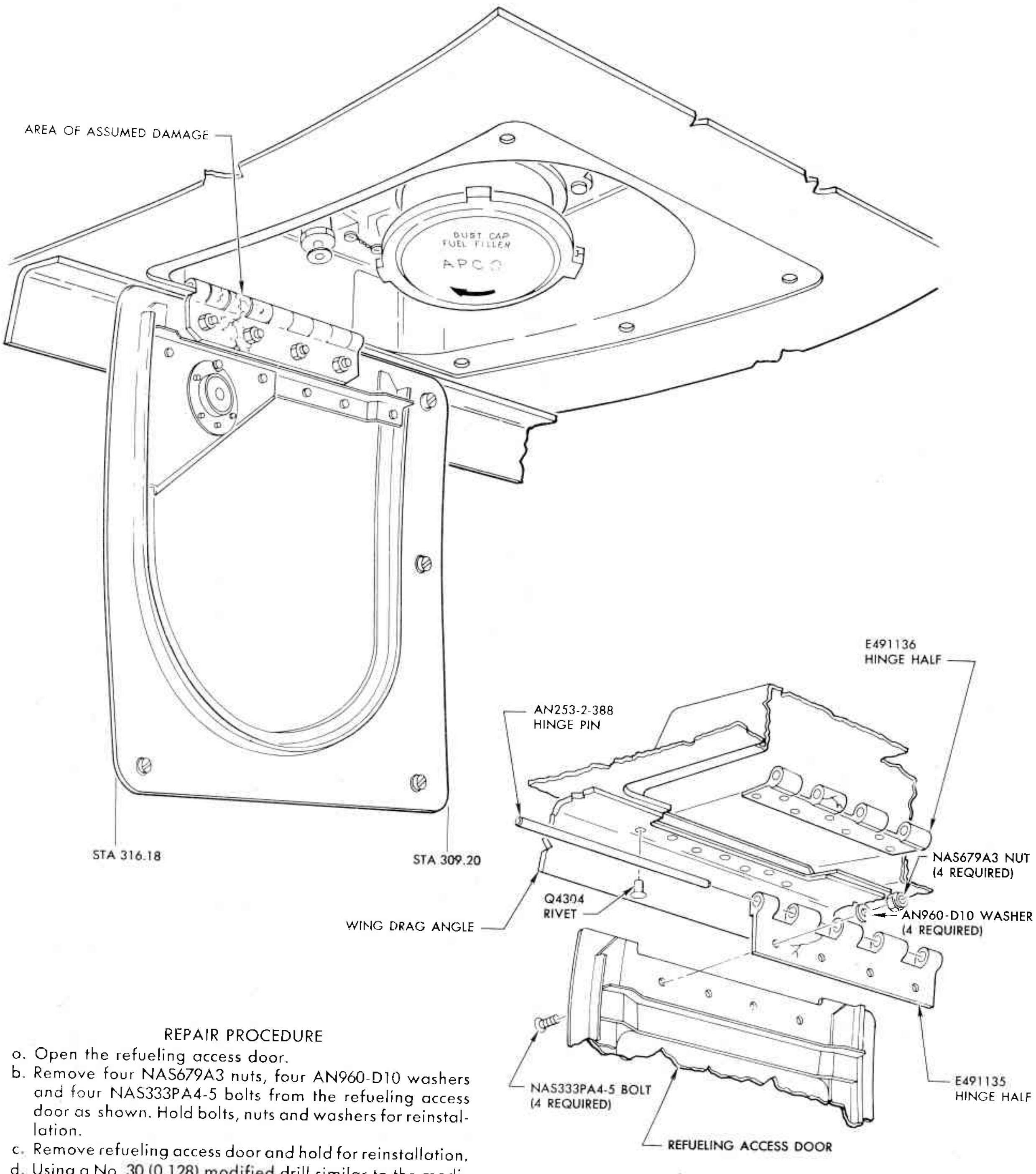
l. Conduct procedures of step e.

m. Repeat preparation procedure steps e through l.

n. Repeat steps h through j of test procedure until requirements of step j are met.

o. Reduce air pressure to zero, drain test fluid, and remove test equipment. Install screws removed in preparation step f.

p. Install slipway door. Refer to T.O. 1F-106A-2-5-2-1 for this procedure.

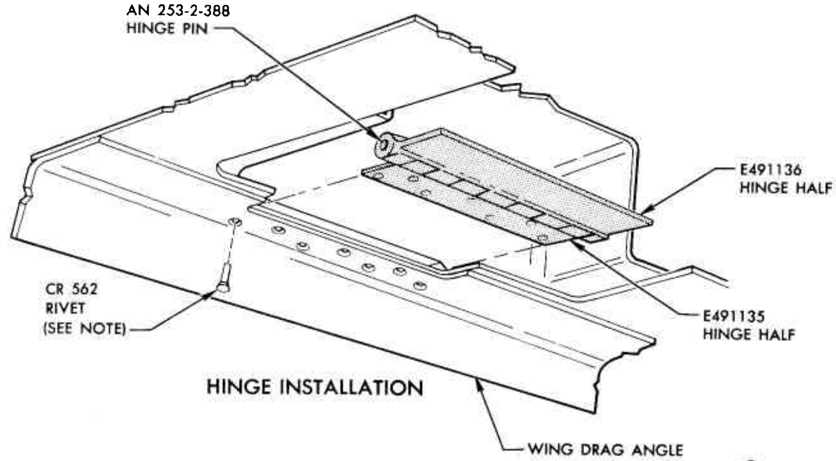


REPAIR PROCEDURE

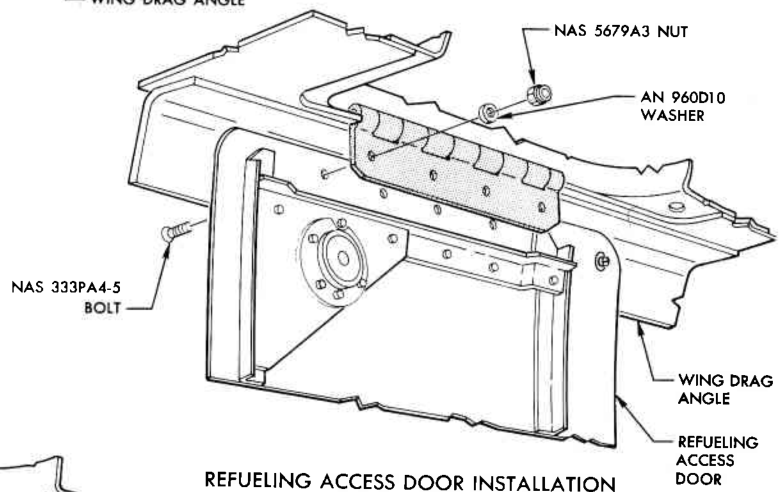
a. Open the refueling access door.
b. Remove four NAS679A3 nuts, four AN960-D10 washers and four NAS333PA4-5 bolts from the refueling access door as shown. Hold bolts, nuts and washers for reinstallation.
c. Remove refueling access door and hold for reinstallation.
d. Using a No. 30 (0.128) modified drill similar to the modified high-speed drill for titanium shown in Section I, remove the Q4304 rivets attaching the refueling access door hinge half to the fuselage.
e. Remove the hinge from the fuselage.

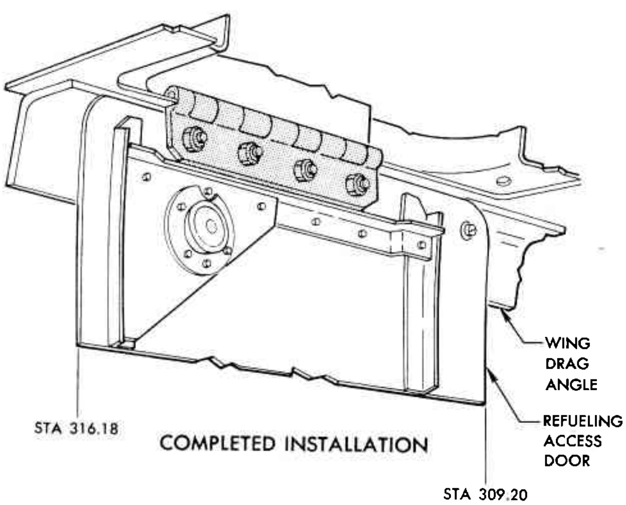
06.03.262-1B

Figure 4-34. Replacement of Refueling Door Hinge (Sheet 1 of 2)



NOTE:
REFER TO TABLE I-XXIII FOR RIVET BASIC COPE.





f. Make new hinge halves from MS 2000X14-0400 and MS 20001Y14-0400.

NOTE

MS 20001X14-0400 HINGE HALF ATTACHES TO FUSELAGE, AND MS 20001Y14-0400 HINGE HALF ATTACHES TO REFUELING ACCESS DOOR.

g. Make a new hinge pin from AN 253-5-388.

h. Mate the two hinge halves together and install hinge pin.

NOTE

CRIMP ENDS OF HINGE PIN AFTER INSTALLATION TO SECURE PIN IN PLACE.

i. Install new hinge in fuselage as shown.
j. Using existing holes in fuselage as a guide, drill holes through new hinge. Use a No. 30 (0.128) drill.
k. Ream holes drilled in step "J" with a No. 20 (0.161) drill.
l. Remove the hinge and remove all burrs from hinge and fuselage.
m. Countersink 100° all holes in exterior surface of fuselage for 5/32" diameter flush rivets.
n. Reinstall hinge and rivet in place as shown.
o. Install refueling access door as shown.
p. Using existing holes in refueling access door as a guide, drill holes through hinge. Use a 3/16" (0.1875) drill.
q. Remove the door and remove all burrs from door and hinge.
r. Reinstall door and secure in place with bolts, washers and nuts which were removed in step "B."

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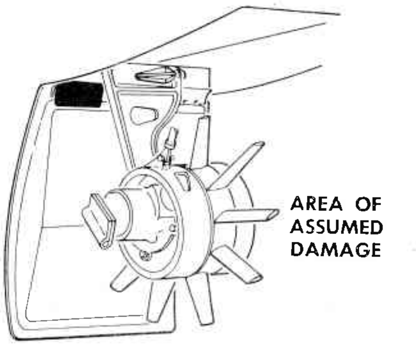
Figure 4-34. Replacement of Refueling Door Hinge (Sheet 2 of 2)



REPAIR PROCEDURE

a. Remove the ram air turbine door. Refer to T.O. 1F-106A-2-2-2-2 for removed procedure.

b. Using a cloth dampened with Methyl Ethyl Ketone, Federal Specification TT-M-261, remove all paint, primer and other foreign material from the hinge area of the ram air turbine door.

c. Using a No. 39 (0.099) drill, remove the rivets attaching the hinge to the door. Remove the hinge.

d. Using a 3/16" (0.187) drill, remove the rivets attaching the two hinge halves to the fuselage longeron. Remove the hinges.

e. Using the hinge removed in step "c" as a guide, cut a new hinge to proper size from extruded aluminum. Alcoa die number E451124.

f. Using the hinge halves removed in step "d" as a guide, cut new hinge halves to proper size from extruded aluminum. Alcoa die number E451125.

NOTE

IF EXTRUDED MATERIAL IS NOT AVAILABLE HINGES MAY BE MACHINED FROM 7075-T6 ¼-INCH PLATE ACCORDING TO DIMENSIONS GIVEN ON CONVAIR ENGINEERING BLUE PRINTS, 8-45961 AND 8-45962.

g. Apply one coat of wash primer to new hinges per Specification MIL-C-8514.

h. Apply one coat of zinc chromate primer to new hinges per Specification TT-P-1752.

i. Install new hinge in ram air turbine door as shown, and hold in place with clamps.

j. Using existing holes in door as a guide, drill holes through new hinge. Use a No. 39 (0.099) drill.

k. Remove hinge from door and remove all burrs from door and hinge.

l. Reinstall new hinge in door and rivet in place with MS20426AD-3 rivets.

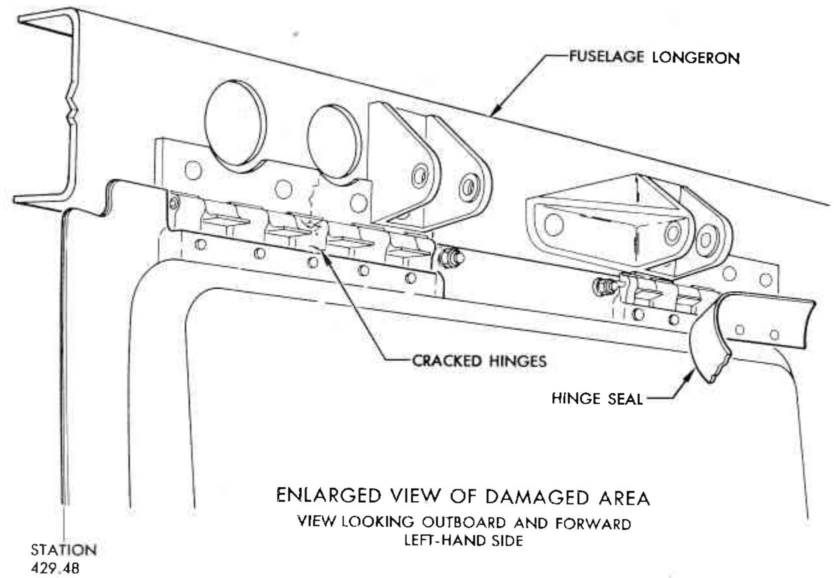
m. Install new hinge halves to fuselage longeron and hold in place with clamps.

n. Mate door hinge to fuselage hinges and install hinge pins.

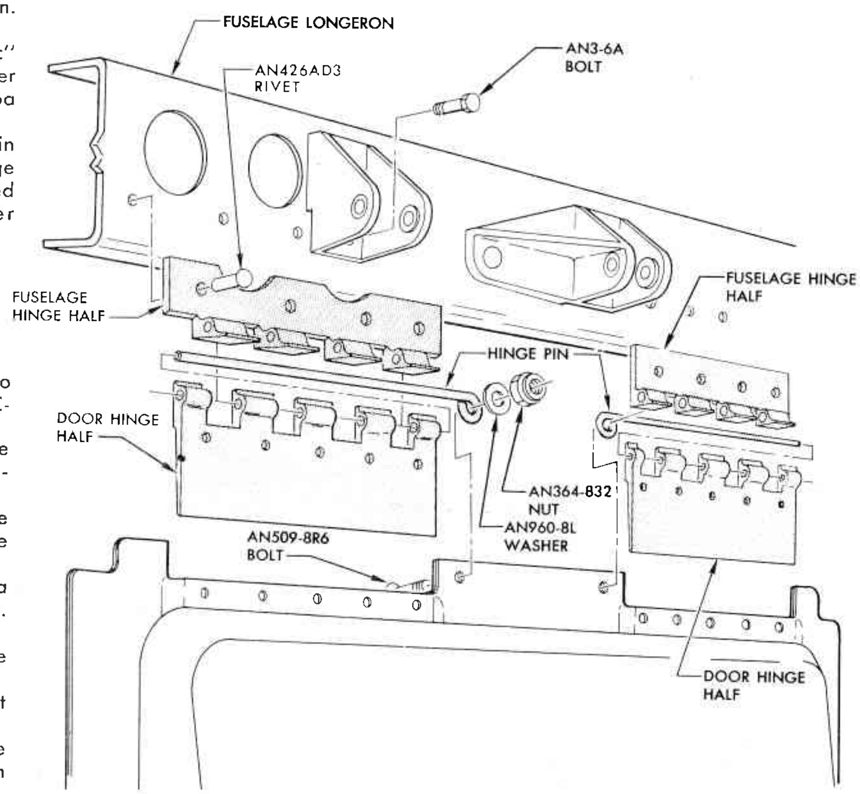
NOTE

DO NOT INSTALL HINGE PIN BOLTS AT THIS TIME.

06.03.260.1B

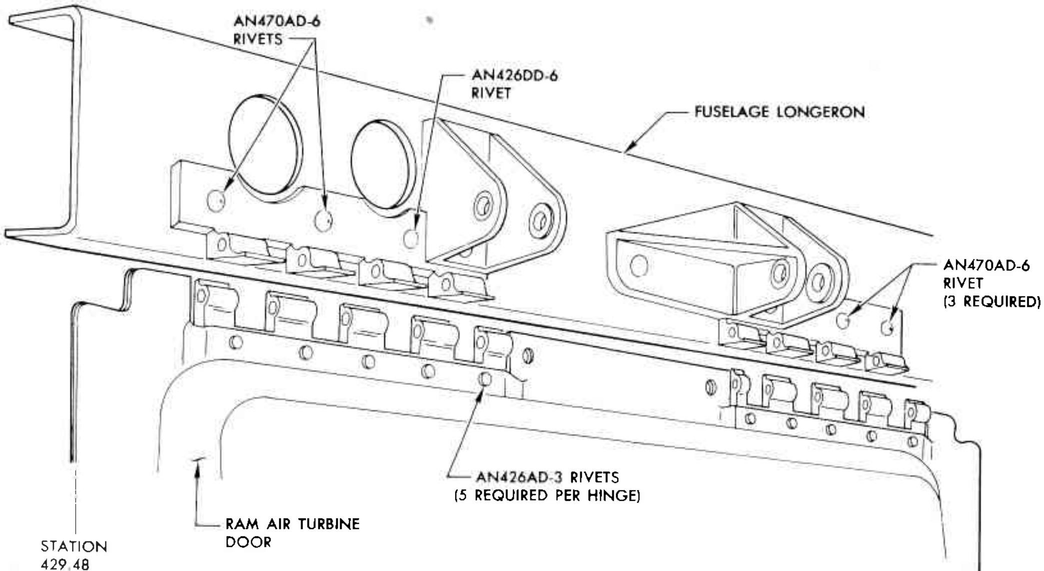


ENLARGED VIEW OF DAMAGED AREA
VIEW LOOKING OUTBOARD AND FORWARD
LEFT-HAND SIDE

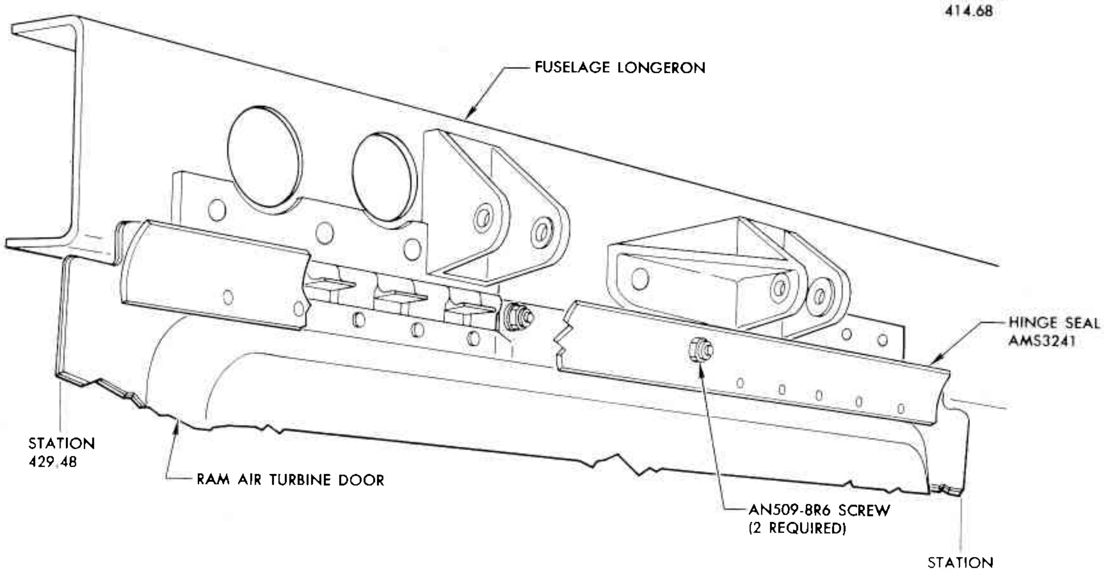


EXPLODED VIEW OF DOOR AND DOOR HINGES

Figure 4-35. Replacement of Ram Air Turbine Door Hinge (Sheet 1 of 2)



HINGE INSTALLATION VIEW



DOOR INSTALLATION VIEW

o. Close and open door manually to check position of door. Move hinge halves clamped to fuselage longeron as necessary to insure correct position of door.
p. Remove hinge pins and remove door.
q. Using existing holes in fuselage longeron as a guide, drill holes through new hinge halves.

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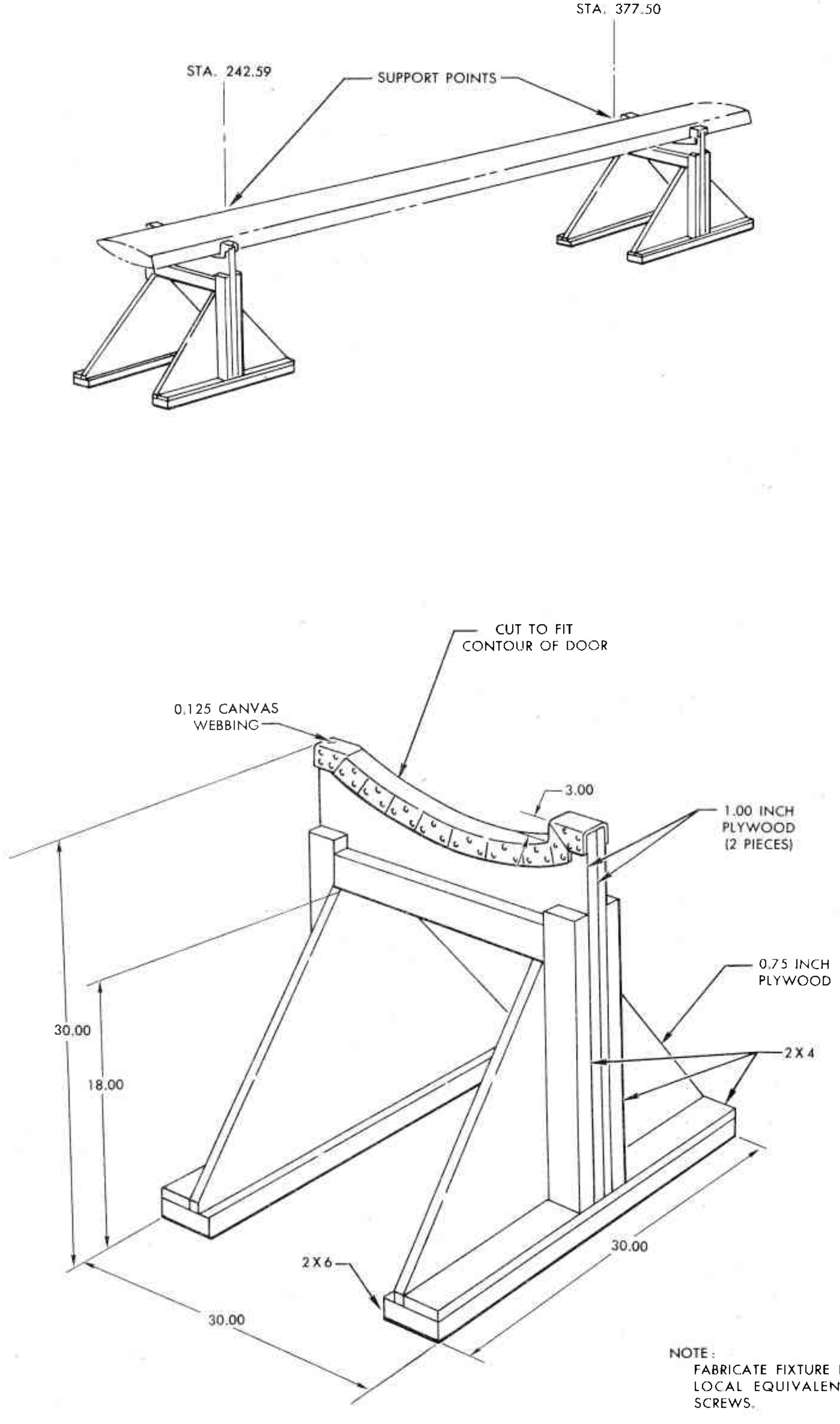
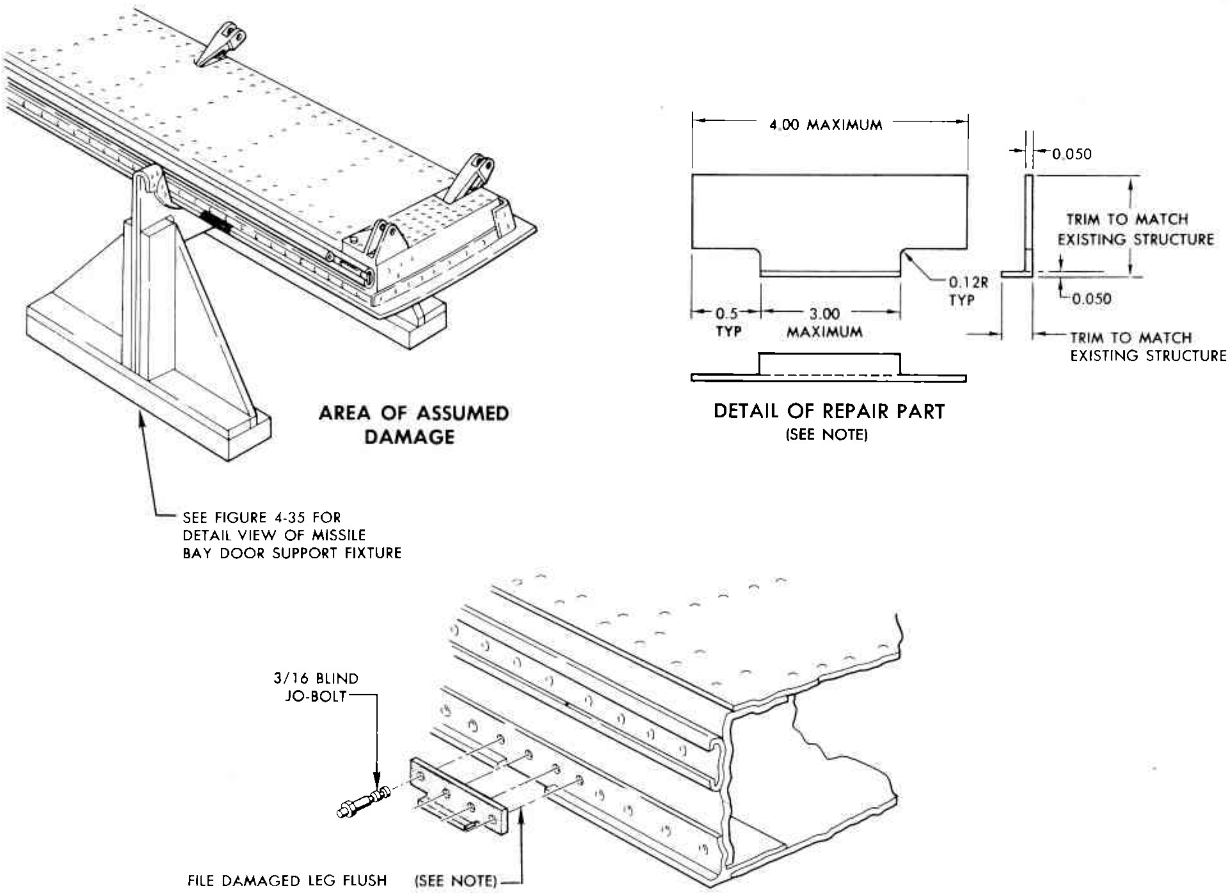
r. Rivet new hinge halves to fuselage longeron with 3/16" (0.187) diameter rivets as shown.
s. Reinstall door and perform functional checkout according to procedures given in T. O. 1F-106A-2-2-2-2.
t. Paint the exterior surface of door according to appropriate finish schedule given in Section I.

Figure 4-35. Replacement of Ram Air Turbine Door Hinge (Sheet 2 of 2)



NOTE:
FABRICATE FIXTURE FROM DOUGLAS FIR OR LOCAL EQUIVALENT, USING BOLTS AND SCREWS.

Figure 4-36. Missile Bay Door Repair



REPAIR PROCEDURE

a. Open missile bay doors and install safety locks. Refer to T.O. 1F-106A-2-12-2-1 for door opening and safety lock installation procedure. Door may be removed from airplane and supported as shown.
b. Remove damaged portion of angle with a file.

CAUTION

OBSERVE REPAIR LIMITATIONS AS SHOWN IN THIS ILLUSTRATION.

c. Fabricate a repair part as shown.

NOTE
MAKE REPAIR PART FROM 7075-T6 EXTRUSION, ALCOA DIE NO. 22526.

d. Remove rivets from damaged extrusion over cutout and one rivet each side of cutout. Use a No. 30 (0.128) drill.
e. Install repair part and pick up existing holes in missile bay door. Refer to Section I for method of locating blind holes.
f. Remove repair part and remove all drill chips and burrs from the repair part and missile bay door.
g. Reinstall repair part and rivet in place with DR rivets. Refer to Table 1-XXIII for rivet basic code.
h. Fill all gaps between repair part and missile bay door with MIL-S-38228 aerodynamic smoothing compound.
i. Lay a fillet of MIL-S-38228 aerodynamic smoothing compound around edge of repair part.
j. Refinish exterior surface of missile bay door according to applicable paint schedule given in Section I.
k. Reinstall door, if removed, or remove missile bay door safety locks and close missile bay doors. Refer to T.O. 1F-106A-2-12-2-1 for procedures.

REPAIR LIMITATIONS

a. Maximum length of repair—4 inches.
b. Minimum distance between repairs—12 inches.
c. Maximum number of repairs allowed per door is limited to replacing 40 per cent of the rivets in the original angle.

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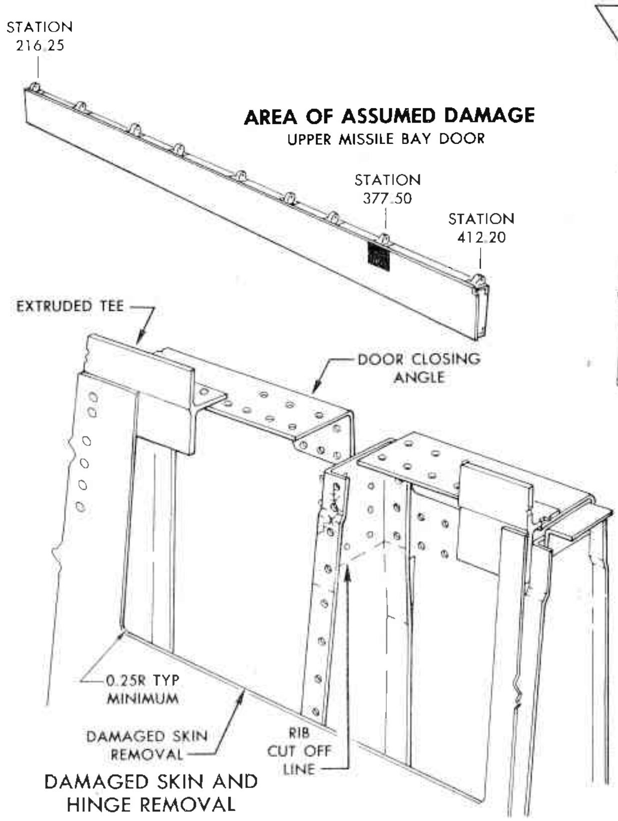
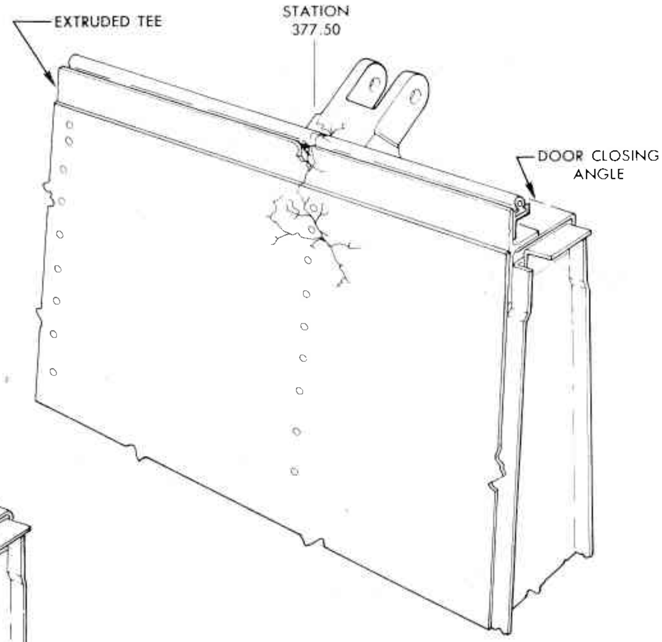
Figure 4-37. Lower Inboard Missile Bay Door Angle Repair





DAMAGE REMOVAL PROCEDURE

a. Remove missile bay door and place upon a suitably padded work bench. Refer to T.O. 1F-106A-2-12-2-1 for removal procedure.
b. Remove cotter pins, bolts and nuts attaching lower door to upper door. Hold bolts and nuts for reinstallation.
c. Perform fluorescent penetrant inspection on all bolts.

d. Separate upper door from lower door and place upper door in safe storage area.
e. Remove rubber seal from lower door. If seal has not been damaged, hold for reinstallation. Damaged seals must be replaced by new seals.
f. Using a cloth dampened with Methyl Ethyl Ketone, Federal Specification TT-M-261, remove all varnish, paint and primer from damaged area of door.
g. Perform fluorescent penetrant inspection to determine extent of damage in door skin.

h. Remove rivets from damaged area of door skin. Use a No. 30 (0.128) drill.
i. Cut out damaged skin. Use metal cutting hand shears and hand files for cutting.
j. Remove rivets attaching hinge to door. Use a No. 30 (0.128) drill to remove 1/8-inch diameter rivets, and use a No. 20 (0.161) drill to remove 5/32-inch diameter rivets.
k. Cut out damaged sections of extruded tee and extruded seal retainer.
l. Remove damaged hinge.
m. Remove primer from damaged rib. Use a cloth dampened with Methyl Ethyl Ketone, Federal Specification TT-M-261 to remove primer.

06.03.263-1B

n. Perform fluorescent penetrant inspection to determine extent of damage to rib.

o. Cut out damaged section of rib. Use metal cutting hand shears and hand files for cutting.
p. Treat all bare surfaces of missile bay door with Alodine 1200. Refer to Table 11-VII for application of Alodine 1200.

REPAIR PARTS FABRICATION PROCEDURE

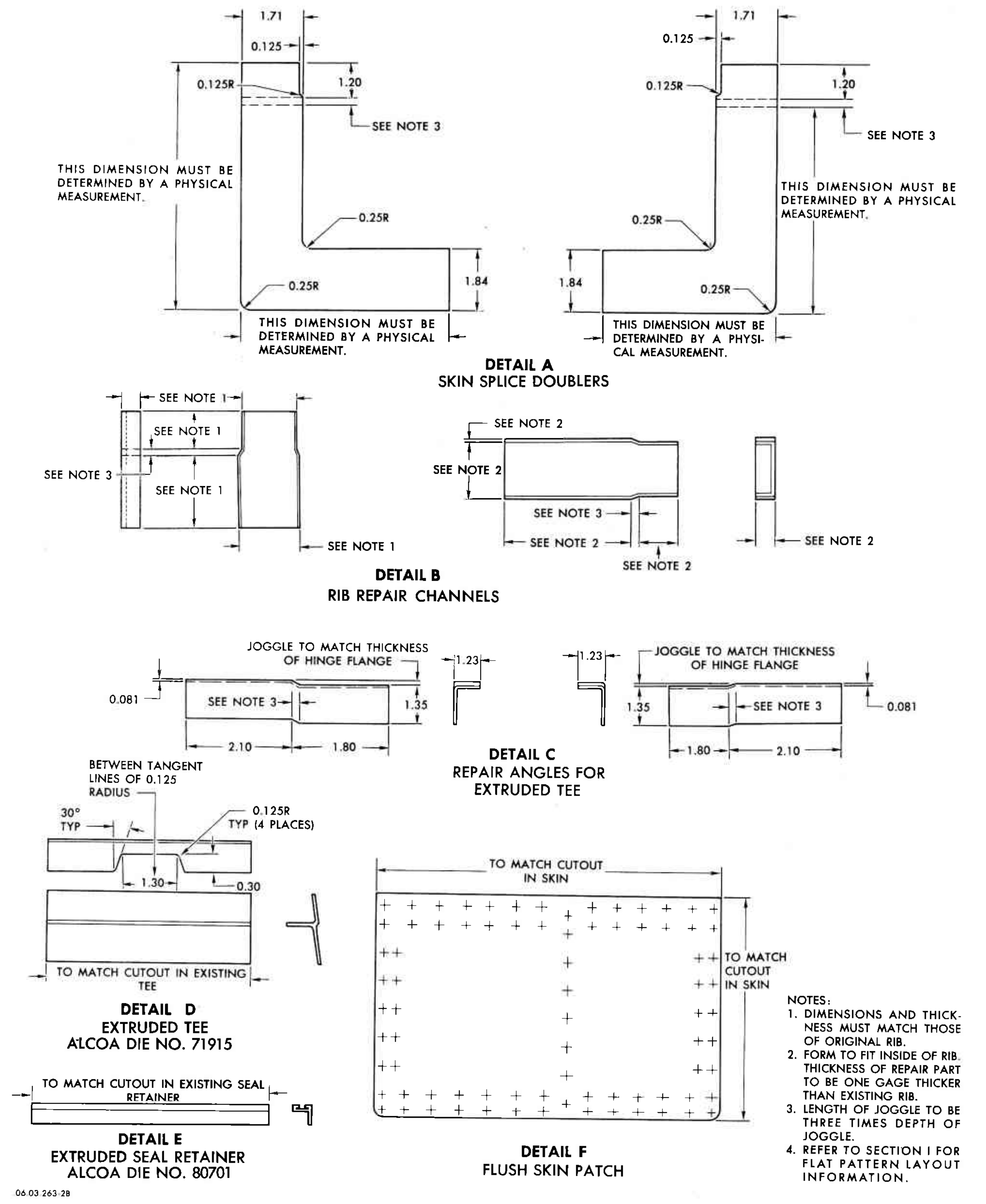
a. Fabricate skin splice doublers, as shown in Detail "A," from 7178-0 bare sheet.
b. Fabricate rib repair parts, as shown in Detail "B," from 7178-0 bare sheet.
c. Fabricate repair angles for extruded tee, as shown in Detail "C," from 7178-0 bare sheet.

NOTE

7178-0 BARE SHEET MUST BE HEAT TREATED TO -T6 CONDITION PER SPECIFICATION MIL-H-6088 AFTER FORMING. IF 7178 BARE SHEET IS NOT AVAILABLE, 7075-0 BARE SHEET ONE GAGE THICKER THAN GAGES SHOWN FOR 7178, MAY BE USED AS AN ALTERNATE. HEAT TREAT 7075-0 TO -T6 CONDITION PER SPECIFICATION MIL-H-6088.

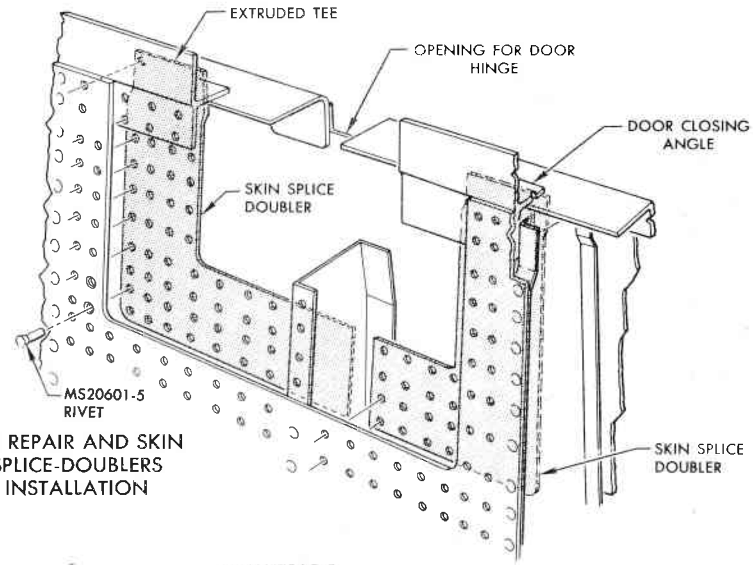
d. Tee repair part is cut from extruded aluminum alloy, Convair part No. E801202 (Alcoa Die No. 71915).
e. Seal retainer repair part is cut from extruded aluminum alloy, Convair part No. E571002 (Alcoa Die No. 80701).
f. Fabricate flush skin patch from 0.040 gage 7178-T6 bare sheet.
g. Apply one coat of wash primer to all repair parts per Specification MIL-C-8514.
h. Apply one coat of zinc chromate primer to all repair parts per Specification MIL-P-8585.

Figure 4-38. Missile Bay Door Repair (Sheet 1 of 5)

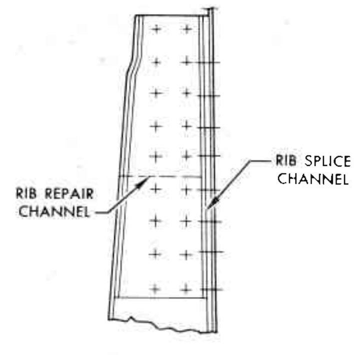


NOTES:
1. DIMENSIONS AND THICKNESS MUST MATCH THOSE OF ORIGINAL RIB.
2. FORM TO FIT INSIDE OF RIB. THICKNESS OF REPAIR PART TO BE ONE GAGE THICKER THAN EXISTING RIB.
3. LENGTH OF JOGGLE TO BE THREE TIMES DEPTH OF JOGGLE.
4. REFER TO SECTION I FOR FLAT PATTERN LAYOUT INFORMATION.

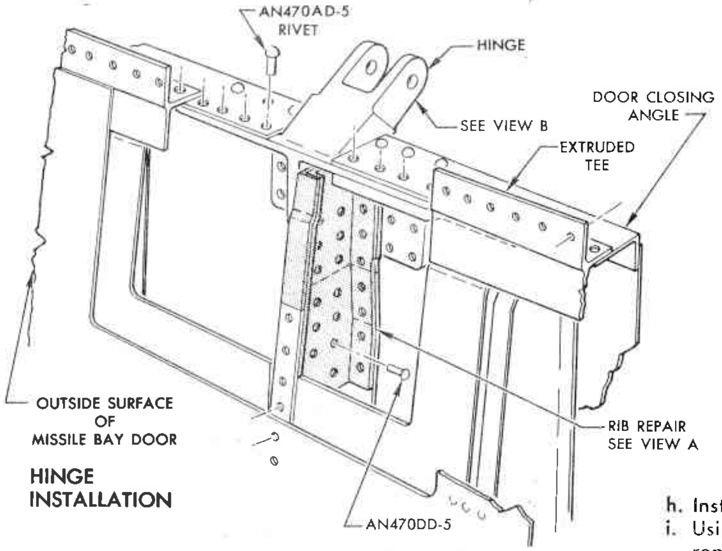
Figure 4-38. Missile Bay Door Repair (Sheet 2 of 5)



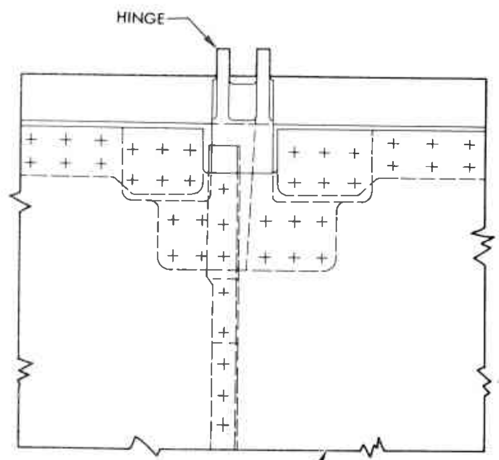
RIB REPAIR AND SKIN SPLICE-DOUBLERS INSTALLATION



VIEW A



HINGE INSTALLATION



VIEW B

REPAIR INSTALLATION PROCEDURE

a. Install rib repair channels and hold in place with clamps.
b. Using existing holes in missile bay door inside skin as a guide, drill holes through rib repair parts. Use a No. 30 (0.128) drill through skin, tapered and rib repair part. Use a No. 20 (0.161) drill through skin and rib repair part.
c. Predrill holes through remaining flanges of rib repair part with a No. 30 (0.128) drill.
d. Remove clamps and install clecos to hold rib repair parts in place.
e. Install new hinge and hold in place with clamps.
f. Using existing holes as a guide, drill holes through new hinge. Use a No. 30 (0.128) drill through skin and hinge. Use a No. 20 (0.161) drill through rib and hinge.
g. Remove clamps and install clecos to hold repair parts in place.

NOTE

CHECK ALIGNMENT OF NEW HINGE BY ALIGNING BOLT HOLES IN NEW HINGE WITH BOLT HOLES IN ADJACENT HINGES.

h. Install tee repair angles and hold in place with clamps.
i. Using existing holes as a guide, drill holes through tee repair angles. Use a No. 30 (0.128) drill through door closing angle and tee repair angles. Use a No. 20 (0.161) drill through door closing angle, hinge and tee repair angles.
j. Predrill holes through remaining flange of tee repair angles. Use a No. 30 (0.128) drill.
k. Remove clamps and install tee repair part.
l. Hold tee repair part and tee repair angles in place with clamps.
m. Using existing holes as a guide, drill holes through tee repair part. Use a No. 30 (0.128) drill through tee repair angles, door closing angle, and tee repair part. Use a No. 20 (0.161) drill through tee repair angles, door closing angle, hinge, and tee repair part.
n. Remove clamps and install clecos to hold repair parts in place.
o. Install skin splice doublers and hold in place with clamps.
p. Using predrilled holes in doublers as a guide, drill holes through skin and rib. Use a No. 30 (0.128) drill.
q. Remove doublers and remove all burrs.
r. Machine countersink 100° by 0.225 all holes in skin.

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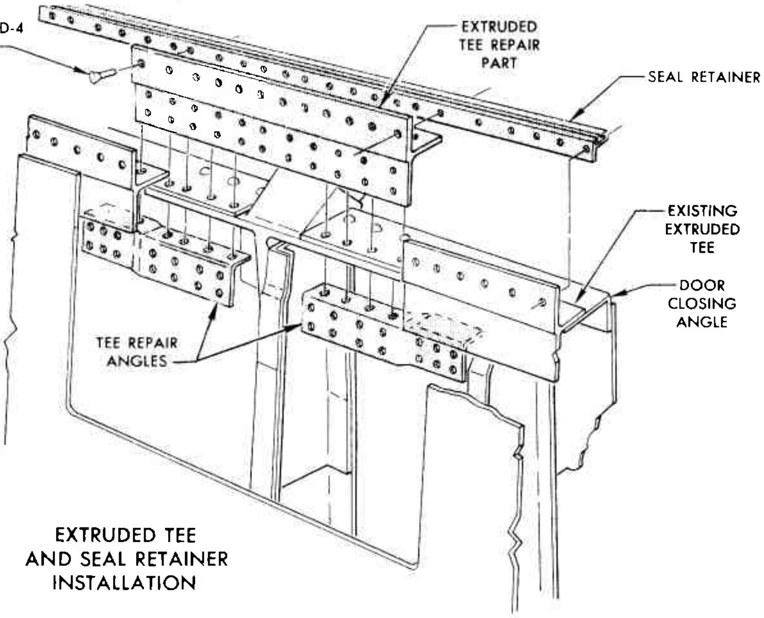
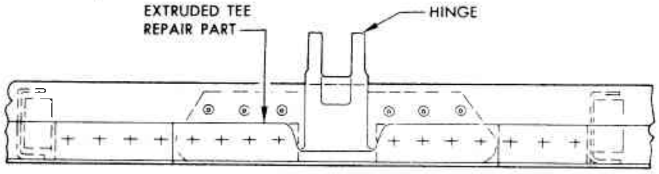
Figure 4-38. Missile Bay Door Repair (Sheet 3 of 5)



EXTRUDED TEE AND SEAL RETAINER INSTALLATION



VIEW C
(SEAL RETAINER NOT SHOWN FOR CLARITY)

s. Apply a coat of MIL-S-81733 to faying surfaces of doublers and skin.

t. Reinstall doublers and hold in place with clecos.
u. Rivet doublers to skin with AN426DD-5 rivets.
v. Remove tee repair part, tee repair angles, rib repair parts and new hinge. Remove all burrs from removed parts and from missile bay door.
w. Reinstall parts removed in step "v" and hold in place with clecos.
x. Rivet parts together with AN470DD-5 rivets.
y. Install flush skin patch.
z. Using predrilled holes in patch as a guide, drill holes through doublers. Use a No. 30 (0.128) drill.
aa. Using existing holes in tee and rib as a guide, drill holes through patch. Use a No. 30 (0.128) drill.

NOTE
REFER TO SECTION I FOR METHODS OF LOCATING BLIND HOLES

ab. Remove patch and remove burrs from patch and door.
ac. Machine countersink 100 degrees by 0.225 all holes in patch.
ad. Apply a coat of MIL-S-81733 to faying surfaces of patch, tee, doublers and rib.

ae. Reinstall patch and hold in place with clecos.
af. Rivet patch in place with MS20601AD-5 rivets.
ag. Fill gap between patch and existing skin with MIL-S-38228.

ah. Install seal retainer part and hold in place with clamps.
ai. Lay out pattern on seal retainer repair part. Use pattern in existing seal retainer as a guide for laying out rivet pattern in seal retainer repair part.
aj. Drill through seal retainer repair part and tee with a No. 30 (0.128) drill.
ak. Remove seal retainer repair part and remove all burrs.
al. Machine countersink all holes in exterior surface of tee 100 degrees by 0.225.
am. Reinstall seal retainer repair part and hold in place with clecos.
an. Rivet seal retainer repair part to tee with AN426AD-4 rivets.
ao. Refinish outside surface of missile bay door according to applicable point schedule given in Section I.
ap. Refinish inside surface of missile bay door by procedure given in Section I.
aq. Apply a coat of A-4004 primer to seal retainer. Refer to Table 11-IX for application of A-4004 Primer.
ar. Apply a coat of A-4000 cement to faying surface of seal.
as. Reinstall upper door to lower door. Use bolts and nuts removed in step "e" of Damage Removal Procedure.
at. Install new AN-381-2-12 cotter pins through bolts.
au. Reinstall missile bay doors to airplane. Refer to T.O. 1F-106A-2-12-2-1 for installation procedure.

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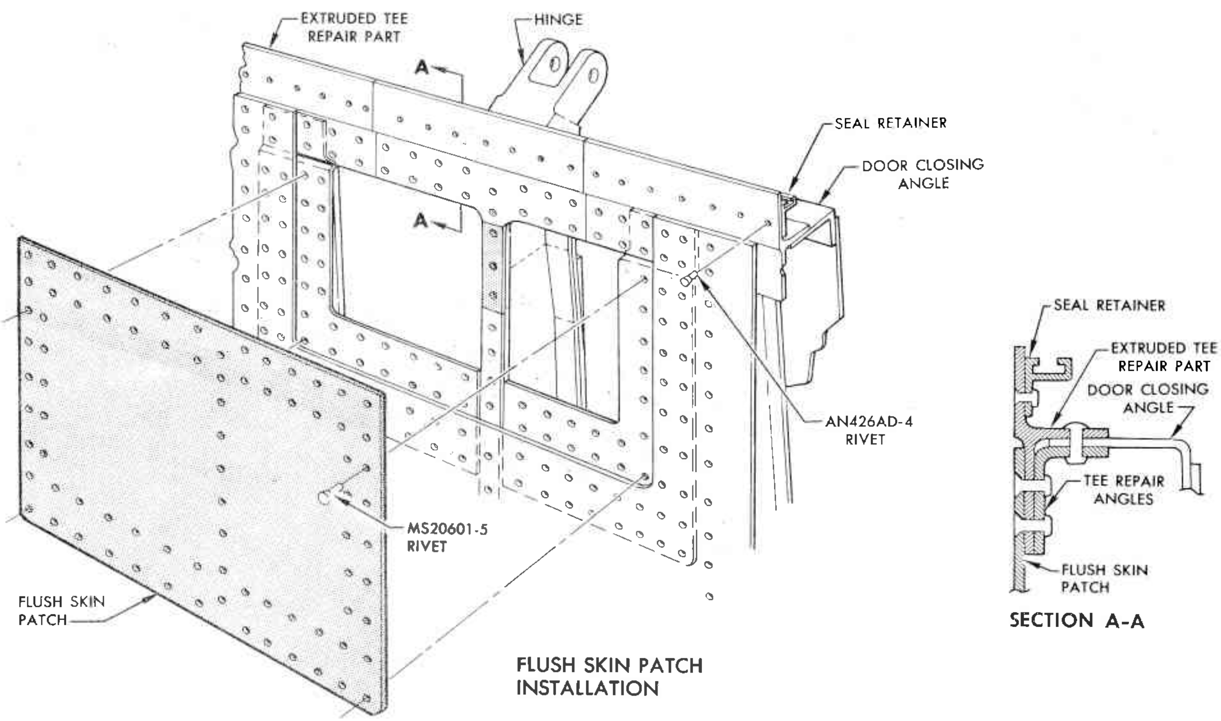
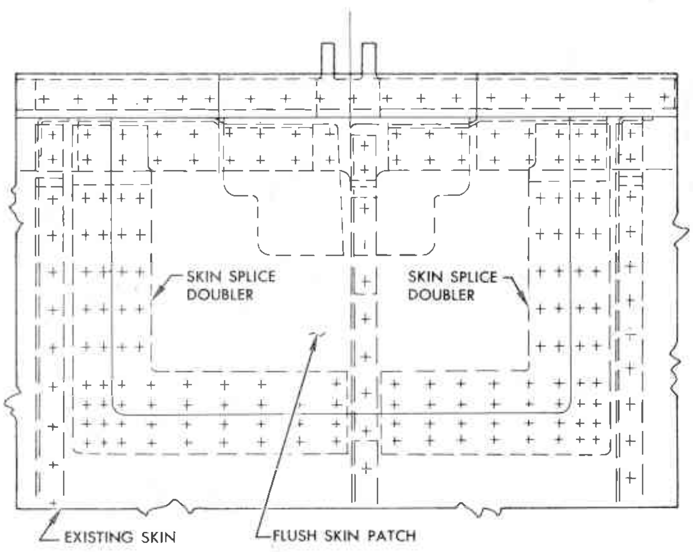
Figure 4-38. Missile Bay Door Repair (Sheet 4 of 5)



FLUSH SKIN PATCH INSTALLATION



VIEW E

PLAN VIEW—OUTSIDE SURFACE OF MISSILE BAY DOOR

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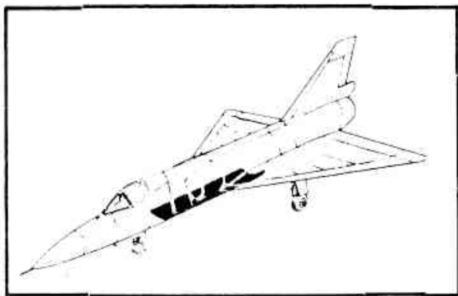
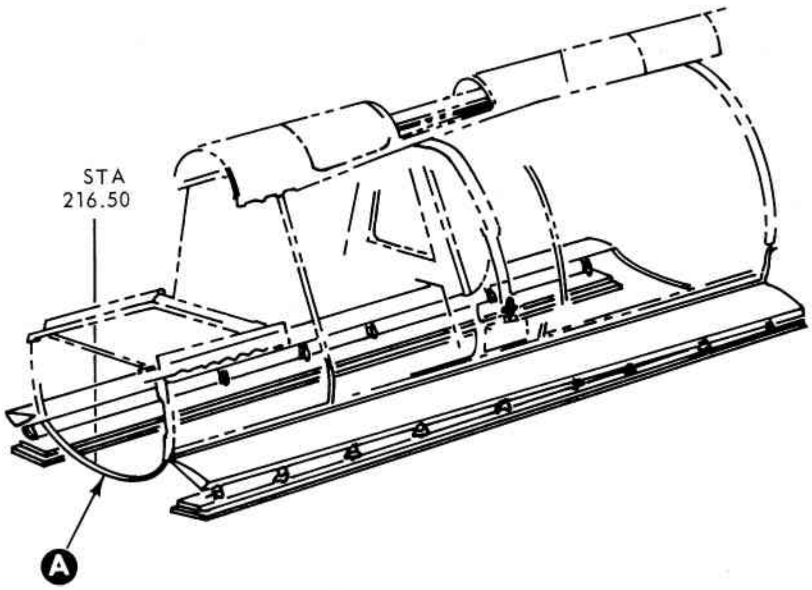
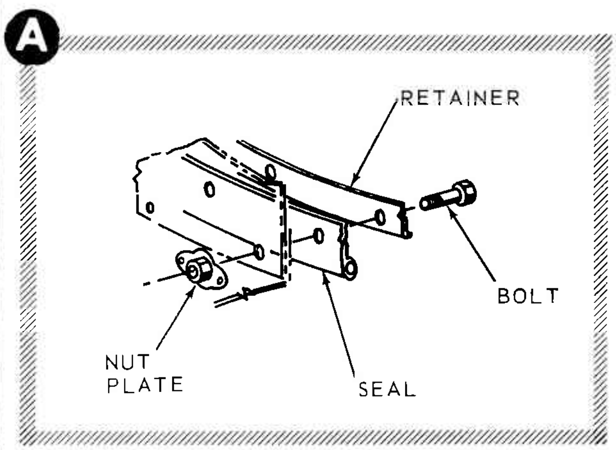
Figure 4-38. Missile Bay Door Repair (Sheet 5 of 5)







VIEW A

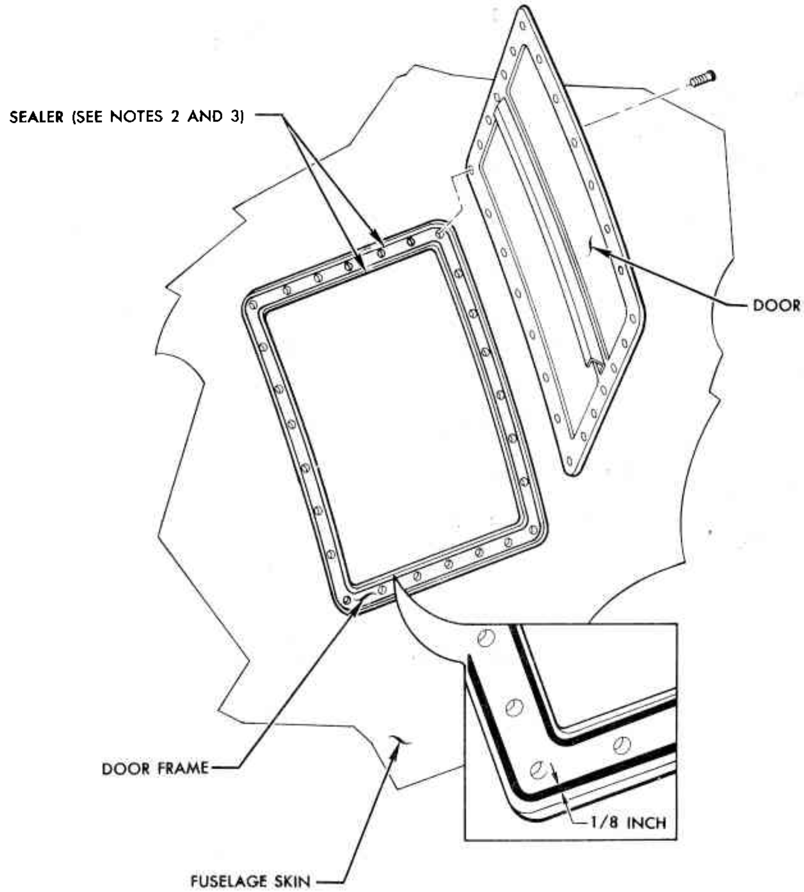
INSTALLATION OF NUT-PLATES AT TIME OF FORWARD MISSILE BAY DOOR SEAL (PART NUMBER 8-77314) REPLACEMENT.

a. Remove lower aft electronic equipment panel.
b. Remove nuts NAS679A3, bolts AN3-4A and AN3-5A, seal retainer 8-74665-21 and seal 8-77314.
c. Install nut-plates NAS696A3 (self-locking) with countersunk rivets MS40426AD as shown in View A.
d. Reinstall seal (new) and seal retainer with bolts AN3-4A and AN3-5A.
e. Reinstall electronic equipment panel.

NOTE

IF ELECTRONIC EQUIPMENT IS REMOVED TO GAIN ACCESS, CHECKOUT THE MA-1 SYSTEM IN ACCORDANCE WITH T.O. 1F-106A-2-27-2.

Figure 4-38A. Missile Bay Door Fwd Seal Nut-Plate Installation Station 216.50 - F-106A



FUSELAGE ACCESS DOORS SEALING PROCEDURE

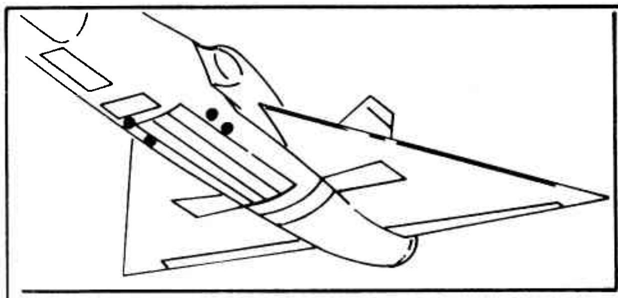
a. Scrape off defective sealer with a plastic scraper.
b. Clean surfaces thoroughly with a cloth dampened with aliphatic naphtha, Federal Specification TT-N-95.
c. Apply sealer, Military Specification MIL-S-8802 to fuselage opening in a continuous pattern, 1/8-inch wide and 1/32-inch thick. Refer to T.O. 1-1-3 for sealer mixing and curing instructions.
d. Apply a thin coat of Petrolatum, Specification VV-P-236, to the under surface of access door to prevent adhesion of door to sealer.
e. Reinstall door and tighten fasteners. Wipe off excess sealer with a cloth dampened in aliphatic naphtha, Federal Specification TT-N-95.
f. Check door for gap and mismatch in accordance with tolerances given in Section I.
g. Allow sealer to cure. Refer to Table 11-X.

NOTE

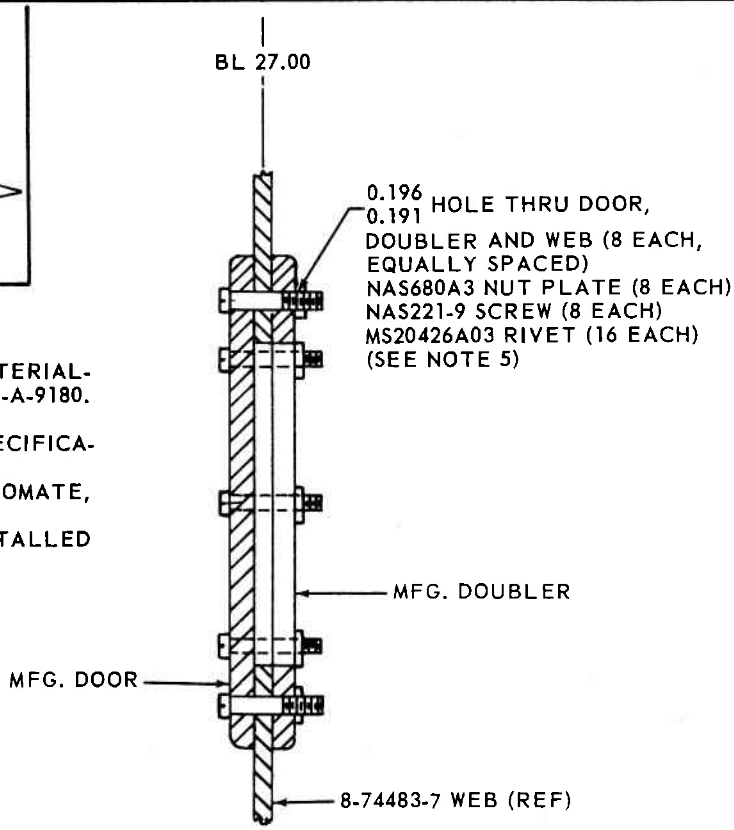
1. SPOT REPAIRS MAY BE MADE TO SLIGHTLY DAMAGED SEALS BY REMOVING THE DEFECTIVE PART OF SEAL AND APPLYING SEALER AS OUTLINED ABOVE.
2. DOORS SECURED WITH SCREWS OR BOLTS REQUIRE A SINGLE BEAD OF SEALER AROUND OUTER EDGE OF DOOR FRAME.
3. DOORS SECURED WITH SPRING TYPE FASTENERS REQUIRE A BEAD OF SEALER AROUND OUTER AND INNER EDGES OF DOOR FRAME.

106.03.229A

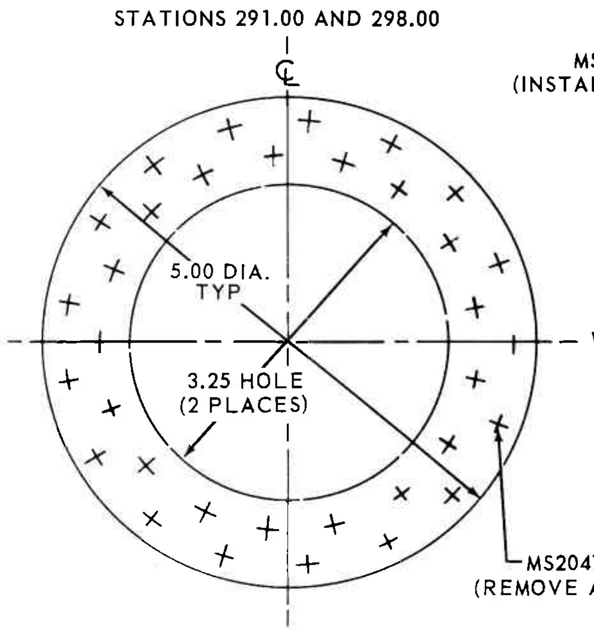
Figure 4-39. Fuselage Access Doors — Formed Gasket — Sealing Procedure



NOTES:
1. MANUFACTURE DOOR AND DOUBLER, MATERIAL-0.050, x 7178-16 AL ALY, SPECIFICATION MIL-A-9180.
2. BREAK ALL SHARP EDGES.
3. SEAL FAYING SURFACE OF DOOR WITH SPECIFICATION MIL-S-81733.
4. COAT ALL METAL SURFACES WITH ZINC CHROMATE, SPECIFICATION MIL-P-8585.
5. RIVETS ATTACHING NUT PLATES TO BE INSTALLED THRU WEB AND DOUBLER.

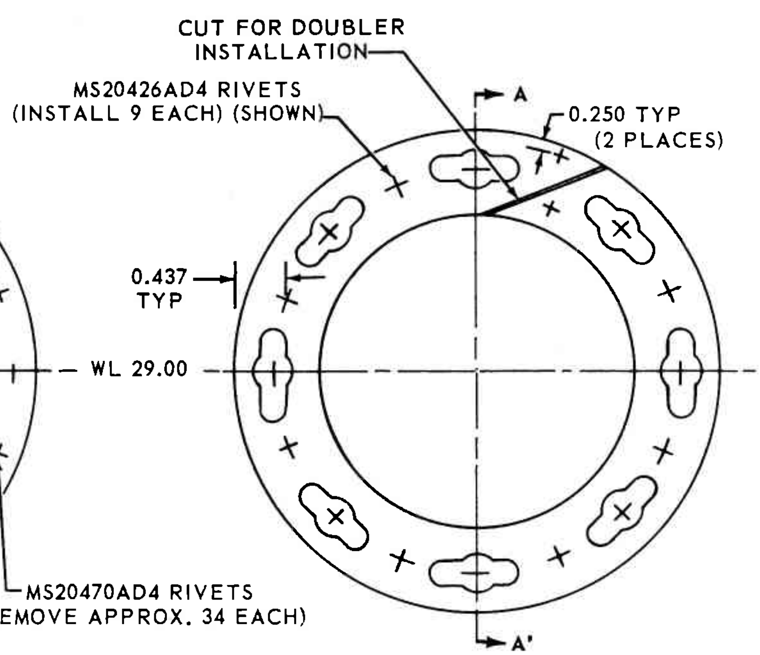


SECTION A A'



DETAIL A - EXISTING INSTALLATION

VIEW LOOKING INBD LH AND RH SIDE OF COVER PLATE, 8-74483-61, INSTALLATION.



DETAIL B - PROPOSED REPLACEMENT INSTALLATION

Figure 4-39A. Missile Bay Area Cover Plate Installation and Replacement

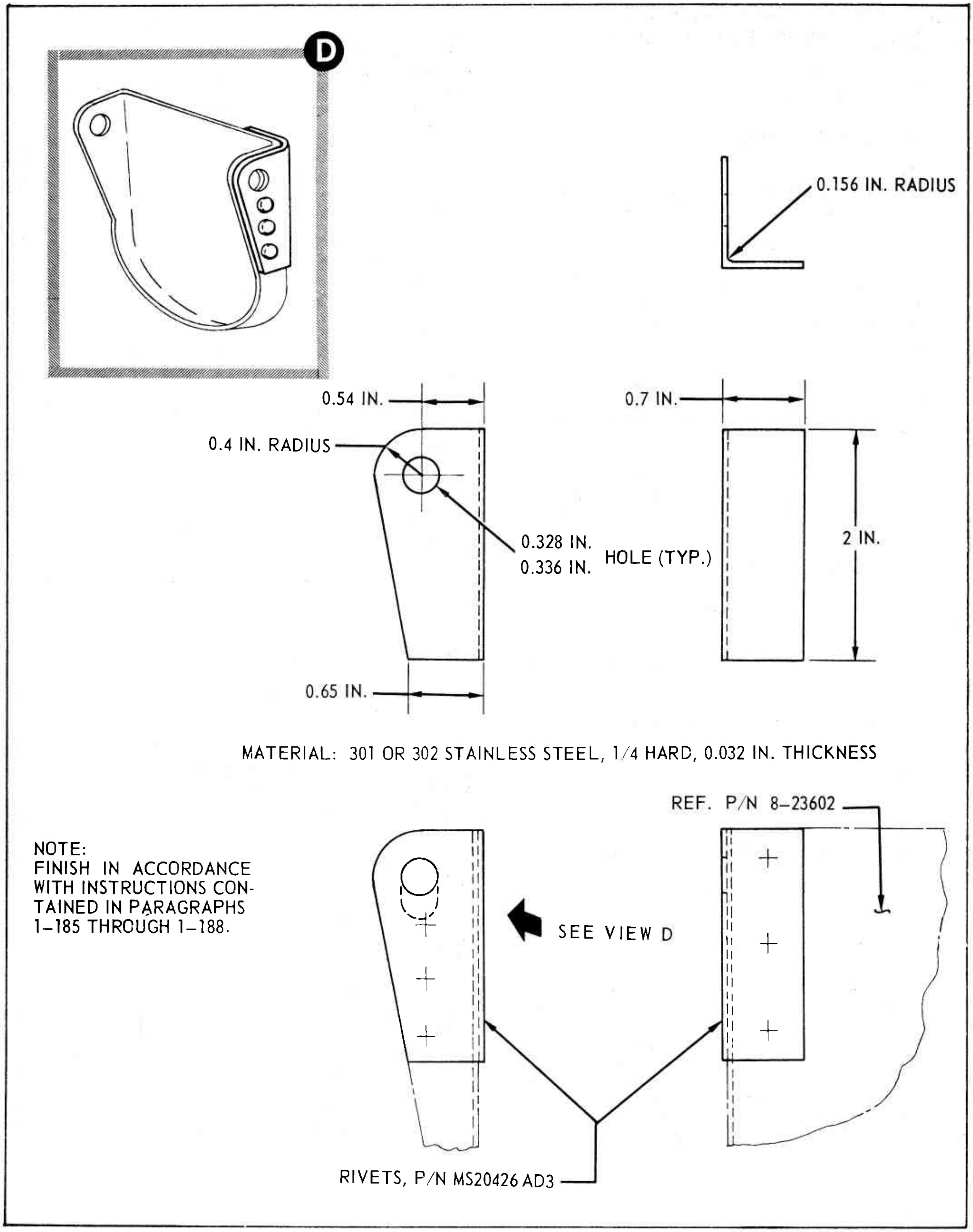


Figure 4-39B. Typical Repair for Flapper Door

4-52F. INSPECTION, REPAIR, PAINTING AND REINSULATION OF HOT BLEED AIR DUCTS (DEPOT ONLY).

a. Remove insulation.

b. Inspect duct for corrosion.

(1) If no corrosion exists, the duct will be passivated, painted and reinsulated in accordance with instructions outlined in paragraphs 4-52J, 4-52K and 4-52L.

NOTE

Stains, braze material, discoloration and grey or tan light oxide films are not to be construed as corrosion for the purpose of this inspection.

(2) If only minor corrosion exists which has not materially weakened the duct, the corrosion will be removed in accordance with instructions outlined in paragraph 4-52H.a.

(3) Corrosion pits in the duct wall which are visible to the unaided eye are not acceptable and will be reapired in accordance with instructions outlined in paragraphs 4-52H.a. and b.

4-52G. REPAIR OF STARTER DUCTS.

a. Clean with stainless steel rotary type brush in and around cracked areas.

b. Heliarc (TIG) or plasma needle arc weld entire crack or damaged area.

c. Weld rod must by type 347 MIL-R-5032 Class C15/15A AMS 5680.

d. Stress relieving not required.

e. NDI weld area for additional cracks and weld voids.

4-52H. CORROSION REWORK PROCEDURE.

a. All ducts with evidence of corrosion are to be reworked as follows:

(1) Etch part number inside one end of duct.

(2) Clean the entire external area of the duct by wet-blasting with 325 grit, or finer, non-metallic abrasive, or by dry-blasting with 200-325 grit glass beads. The glass beads must not be treated with silicones. Other anti-caking materials which will not interfere with subsequent processing of finishing of the ducts may be used. Edges of ports, flanges, etc., which form faying surfaces, must be protected to prevent entry of blasting abrasives into the faying surface. Protection of faying surfaces may be accomplished by masking to prevent entry of abrasives. Rubber bands of sufficient width and size to provide reisitance to movement by the blast of abrasive or rubber electricians' tape are suitable materials for masking. After blasting, the masking shall be removed and the areas protected by the masking shall be cleaned by stainless steel wire brushing. Avoid handling ducts with bare hands. Rinse free of abrasive.

CAUTION

Avoid excessive local blasting which may result in reducing the wall thickness or distortion of the duct.

b. Corrosion and Crack Repair: Repair of corroded areas and cracks shall be as shown in figures 4-40, 4-40A, and 4-40B. The following requirements and restrictions apply:

(1) The reinforcement strap must completely and circumferentially cover all corrosion pits or cracks in that immediate area, i.e., there must be no exposed corrosion pits or cracks within 1/4 inch of the edge of the reinforcement strap prior to welding.

(2) The duct in the area being repaired must be free of wrinkles.

(3) The repair straps are to be made of annealed AISI321 or annealed AISI347 corrosion-resistant steel.

(4) The thickness of the repair straps is to be one gage heavier than the duct wall being repaired.

(5) The form of the strap, prior to welding, must conform closely to the form of the duct.

(6) Only INERT GAS SHIELDED TUNGSTEN ARC (TIG) welding will be used, and welders are to be certified as Group II in accordance with Military Specification MIL-T-5021, or equivalent.

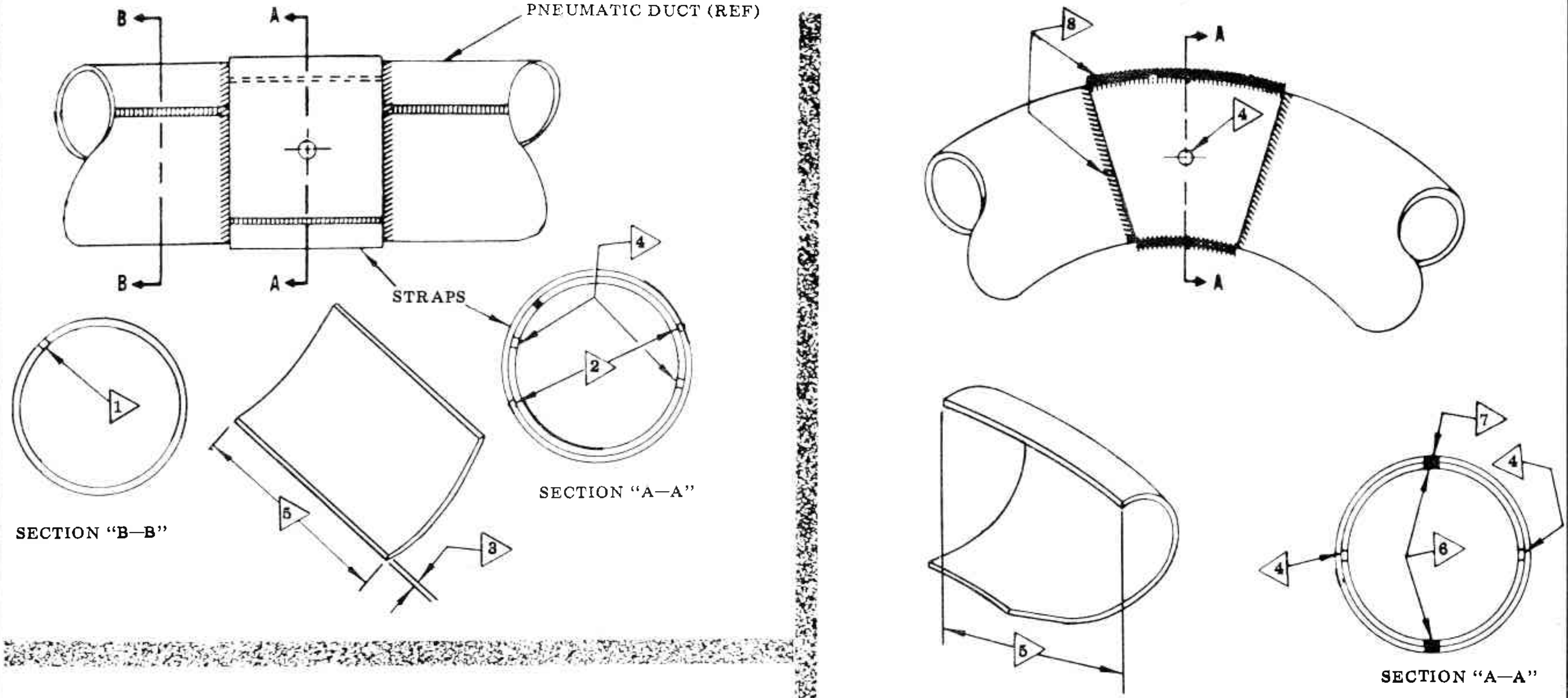
(7) The longitudinal welds in the doublers are to extend to and be fused with the circumferential fusion welds.

(8) Repair of cracks in end flanges or bellows is not permitted. Unserviceable flanges or bellows must be replaced.

(9) A maximum of three repairs per duct is allowed on ducts 4 feet or less in length. Repairs on ducts longer than 4 feet may be increased proportionately.

(10) Deleted.

(11) After being repaired, ducts must be hydrostatically-tested to the room temperature proof pressure



CORROSION REPAIR IN CURVED LENGTHS OF HALF SHELL DUCTS

LEGEND

1. THE LONGITUDINAL MACHINE WELDED AND SIZED BEAD IS THE SAME THICKNESS AS THE DUCT WALL AND IN SOME CASES MAY BE DIFFICULT TO DISCERN IN DUCTS WHICH HAVE BEEN IN SERVICE.

2. THE STRAP ENDS ARE TO BE BUTT WELDED AND WELDED TO THE DUCTS AFTER THE CIRCUMFERENTIAL WELDS ARE COMPLETED. THE LOCATION OF THE BUTT WELDS RELATIVE TO THE LONGITUDINAL MACHINE WELD IS OPTIONAL. THE BUTT WELDS ARE TO EXTEND TO AND BE FUSED WITH THE CIRCUMFERENTIAL FUSION WELDS.

3. THICKNESS OF REPAIR STRAPS TO BE ONE GAGE HEAVIER THAN DUCT WALL.

4. PRIOR TO STRAP INSTALLATION, DRILL PRESSURE RELIEF HOLES IN DUCT WALL AS FOLLOWS:

   1. DRILL SIZE NO. 30 (0.1285).
   2. LOCATE PRESSURE RELIEF HOLE(S) UNDER EACH STRAP HALF.
   3. MAINTAIN 1/4 INCH MINIMUM DISTANCE BETWEEN RELIEF HOLES AND ANY EXISTING OR SUBSEQUENT WELD LOCATIONS.
   4. DEBURR ALL PRESSURE RELIEF HOLES PRIOR TO STRAP INSTALLATION.

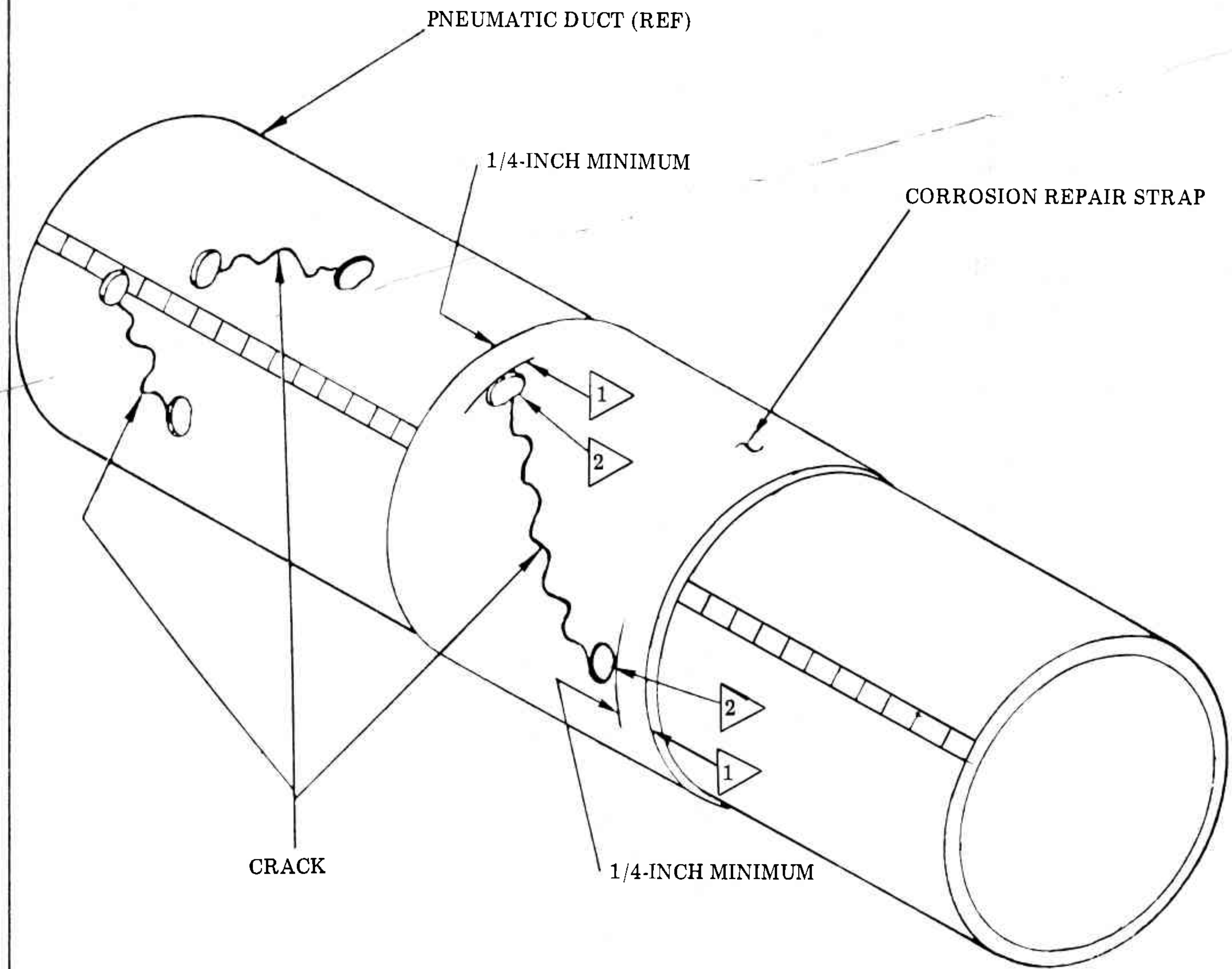
5. THIS DIMENSION IS TO BE FROM A MINIMUM OF 1 INCH TO A MAXIMUM OF 6 INCHES WITHIN THESE LIMITS.

6. THESE LONGITUDINAL WELD BEADS ARE THICKER THAN THE DUCT WALL AND ARE QUITE OBVIOUS FOR HALF SHELL DUCTS.

7. THE TWO STRAP HALVES ARE TO BE BUTT WELDED AND WELDED TO THE PRIOR LONGITUDINAL FUSION WELDS AFTER THE CIRCUMFERENTIAL WELDS ARE COMPLETED. THE BUTT WELDS ARE TO EXTEND TO AND BE FUSED WITH THE CIRCUMFERENTIAL FUSION WELDS.

8. WELD WITH INERT-GAS SHIELDED TUNGSTEN ARC PROCESS.

Figure 4-40. Strap Installation Corrosion Repair — Bleed Air Duct



LEGEND:

1 — 1/4- INCH MINIMUM.

2 — STOP DRILL ENDS OF CRACK WITH NO. 30 (0.1285) DRILL PRIOR TO APPLYING STRAP.

Figure 4-40A. Strap Installation — Crack Repair — Bleed Air Duct

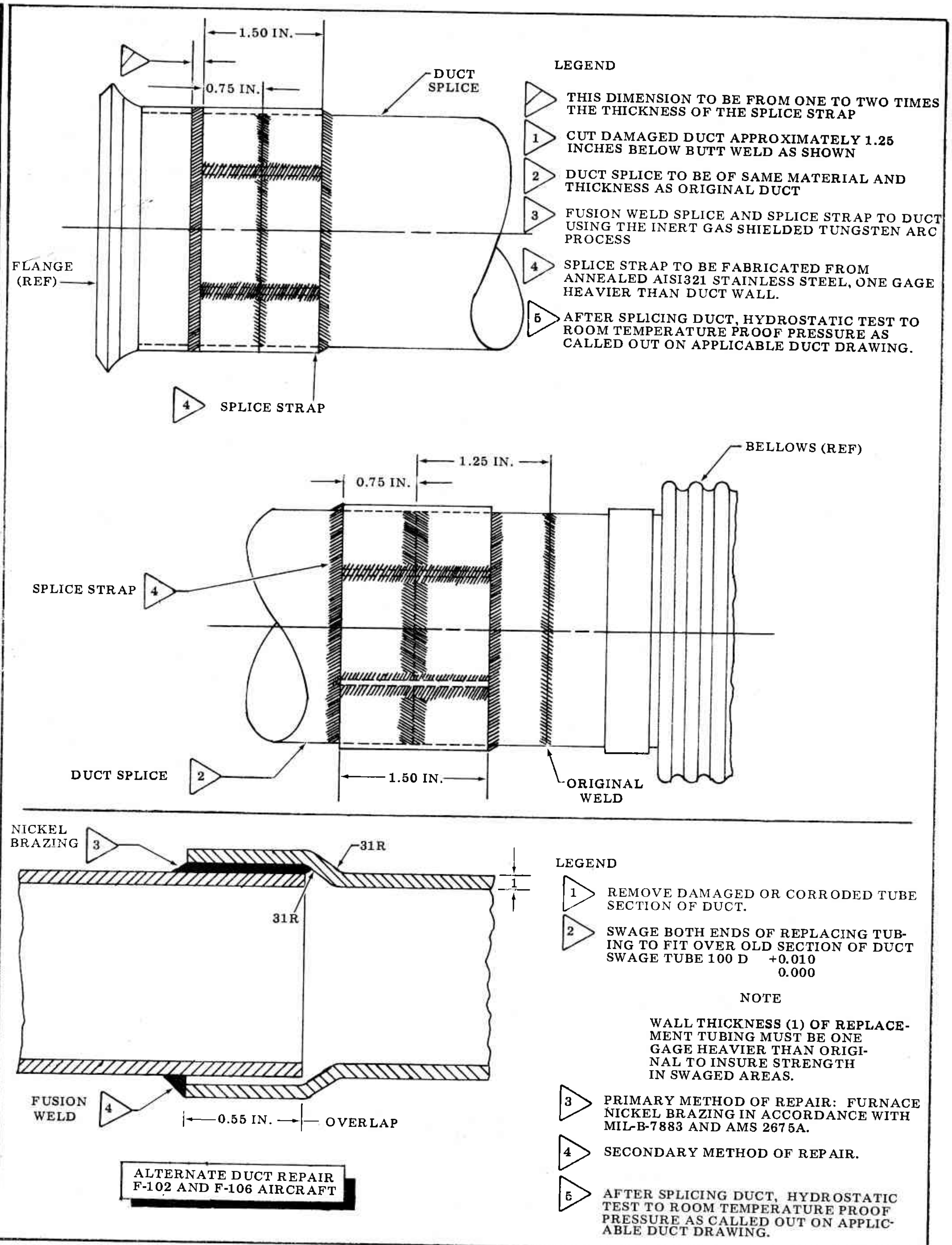


Figure 4-40B. Splice — Corrosion Repair — Bleed Air Duct (optional product test) which is called out on the applicable duct drawing.

WARNING

Do not use air to pressurize or test ducts.

(12) The corrosion repair straps and crack repair strap applied to areas of ducts made from machine welded and sized tubing may be fabricated from one piece.

4-52J. PASSIVATION:

After final blast cleaning or wire brushing, the ducts are to be passivated as follows:

a. Clean the ducts in an alkaline cleaner solution as follows:

(1) Alkaline cleaner shall be the following, or equivalent. The cleaner make-up and control are to be as follows:

Material	Make-Up (lbs-100 gal)	Operating Temp. °F
Kelite Anodyne Kelite Corp.	40	160-190
Wyandotte Max. Amp. Wyandotte Chem. Corp.	40	160-190
Alkaline Rust Turco Products, Inc. Remover	200	190-210
Rustripper Oakite Products, Inc.	200	190-210
Ferlon Wyandotte Chem. Corp.	200	190-210
Oakite 90 Oakite Products, Inc.	4400	160-190

(2) The recommended make-up procedure for the alkaline cleaner is as follows:

(a) Fill tank one-half full with water.

(b) Heat to within 20 degrees of the maximum operating temperature.

(c) Distribute cleaner uniformly over the surface, while agitating, until the required amount has been added. Add amount proportional to volume of tank.

(d) Add water to about two inches below operating level.

(e) Heat to operating temperature.

(f) Add water to operating level and agitate until the concentration is uniform throughout tank.

(3) The alkaline cleaner shall be maintained free of surface contamination, such as floating debris or oily films.

(4) If the cleaning efficiency decreases, as evidenced by increased cleaning time required for soil removal, cleaner additions should be made in amounts equal to ten per cent of the original make-up quantity. These additions may be repeated, as required, until the total of additions equals the original make-up.

(5) Rinse free of cleaning material with water.

b. Immerse duct for one-half hour in a nitric acid solution made up as follows:

(1) The nitric acid solution is to be made up by mixing equal volumes of water and concentrated (technical grade 12° Baume) nitric acid.

WARNING

Do not pour water in concentrated nitric acid. Always dilute by pouring acid in water.

(2) The solution must be maintained to prevent the concentration of silver ion from exceeding 12 grams per liter by solution withdrawal and replacement.

c. Rinse thoroughly with water and dry.

4-52K. Re-insulate the ducts in accordance with the procedures outlined in convair drawing No. 8-02293, type B. Stencil the following adjacent to the stenciled part number with 1/4-inch letters: INSP & REWK DATA / SM-ALC

4-52L. BOUNDARY LAYER INTAKE DUCT LIP REPAIR. Allowable limits and repair instructions for dents and cracks in boundary layer intake duct lip area. The following limits and repair instructions apply:

a. Limits: Dents with no cracks that are within the limits shown in class 1, figure 1-19 (page 1-18) need not be repaired.

b. Repair:

(1) All dents with cracks and dents with no cracks exceeding two inches in length should be repaired by removing damaged area and welding (Heli-Arc) in a piece (flush insert) of 6061-0 aluminum alloy material of like thickness.

(2) Cracks that are not dented may be welded (Heli-Arc) without removing damaged area.

(3) Dents with no cracks exceeding limits shown in class 1, figure 1-19, but exceeding two inches in length, may be repaired by filling with weld bead (Heli-Arc).

(4) All welding beads must be dressed down to original contour (see figure 1-25 Aerodynamics Smoothness Requirements).

(5) All repairs requiring welding will be accomplished as follows:

WARNING

Use goggles or face shield during grinding operation.

(a) First grind off the plating from the duct lip one inch beyond the area to be repaired. Follow the grinding procedures outlined in TO 1-1-2.

(b) Use only aluminum oxide abrasive wheel with 120 grain size abrasive.

(c) Observe grinding speed for the particular size of wheel used.

(d) Wash the groundout and adjacent area with alkaline waterbase aircraft cleaner in accordance with TO 1-1-1, section III.

(e) Cold water rinse and dry.

WARNING

Welding procedures on aircraft must be accomplished by welders certified in accordance with MIL-T-5021D. Prior to welding on aircraft, approval must be obtained from the Aircraft Maintenance Officer, Fire Chief, and Ground Safety Officer and proper precautions taken to prevent the possibility of fire or other mishap.

(f) Repair by welding as required and dress welding beads down to original contour.

(g) Repeat steps (5) (d) and (e) above.

(h) Mask the adjacent area using barrier paper per specification MIL-B-121 and masking tape per specification UU-T-106 or equivalent, according to TO 1-1-8.

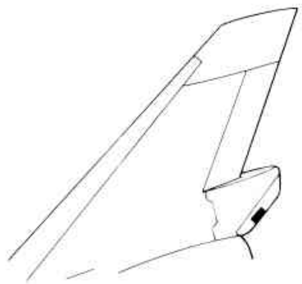
(i) Treat the repair area with brush-on conversion coating for aluminum alloy per specification MIL-C-5541.

(j) Cold water rinse and dry.

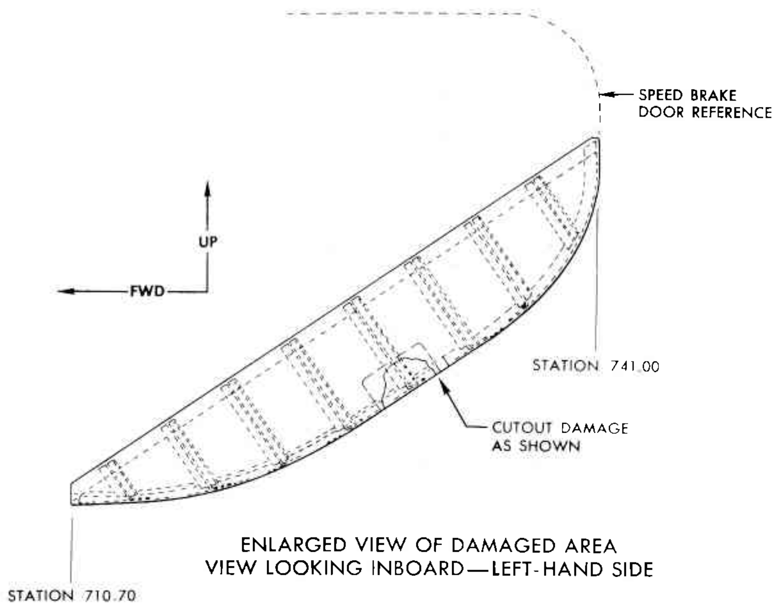
(k) Apply two coats of epoxy-polymide primer per specification MIL-P-23377 according to TO 1-1-8.

(l) Apply two coats of polyurethane coating MIL-C-83286 (gray color number 16473) in accordance with TO 1-1-8.

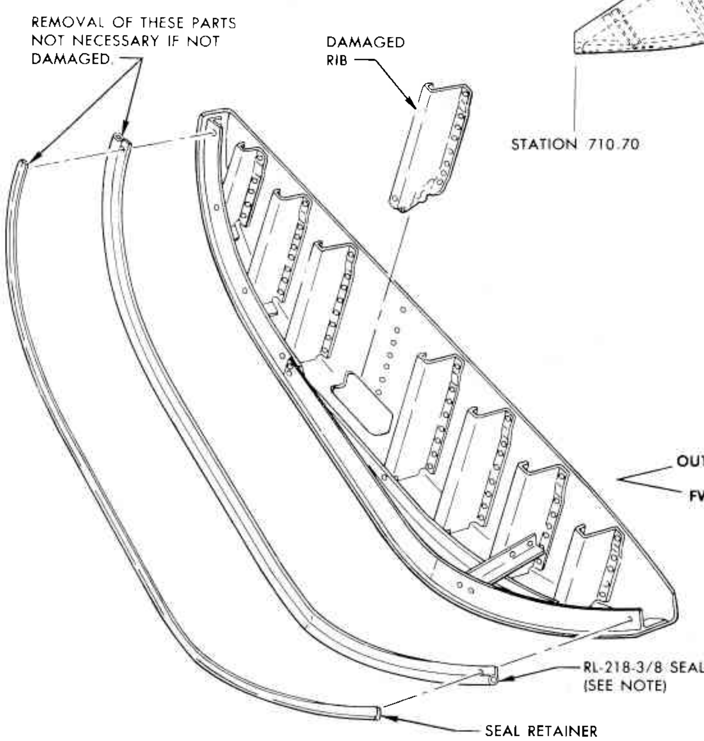
4-52M. Deleted.



AREA OF ASSUMED DAMAGE



ENLARGED VIEW OF DAMAGED AREA
VIEW LOOKING INBOARD—LEFT-HAND SIDE



NOTE:
RL-218-3/9 SEAL IS MANUFACTURED BY RAYBESTOS-MANHATTAN INC., PASSAIC, NEW JERSEY.

DAMAGE REMOVAL PROCEDURE
VIEW LOOKING OUTBOARD—LEFT-HAND SIDE

REPAIR PROCEDURE

a. Open the speed brake doors and install safety locks. Refer to T.O. 1F-106A-2-7-2-1 for opening of speed brake doors and safety lock installation procedure.
b. Perform fluorescent penetrant inspection to determine extent of damage. Refer to Section I for fluorescent penetrant inspection procedure and materials.
c. Remove damage as shown.
d. Fabricate repair parts as shown.

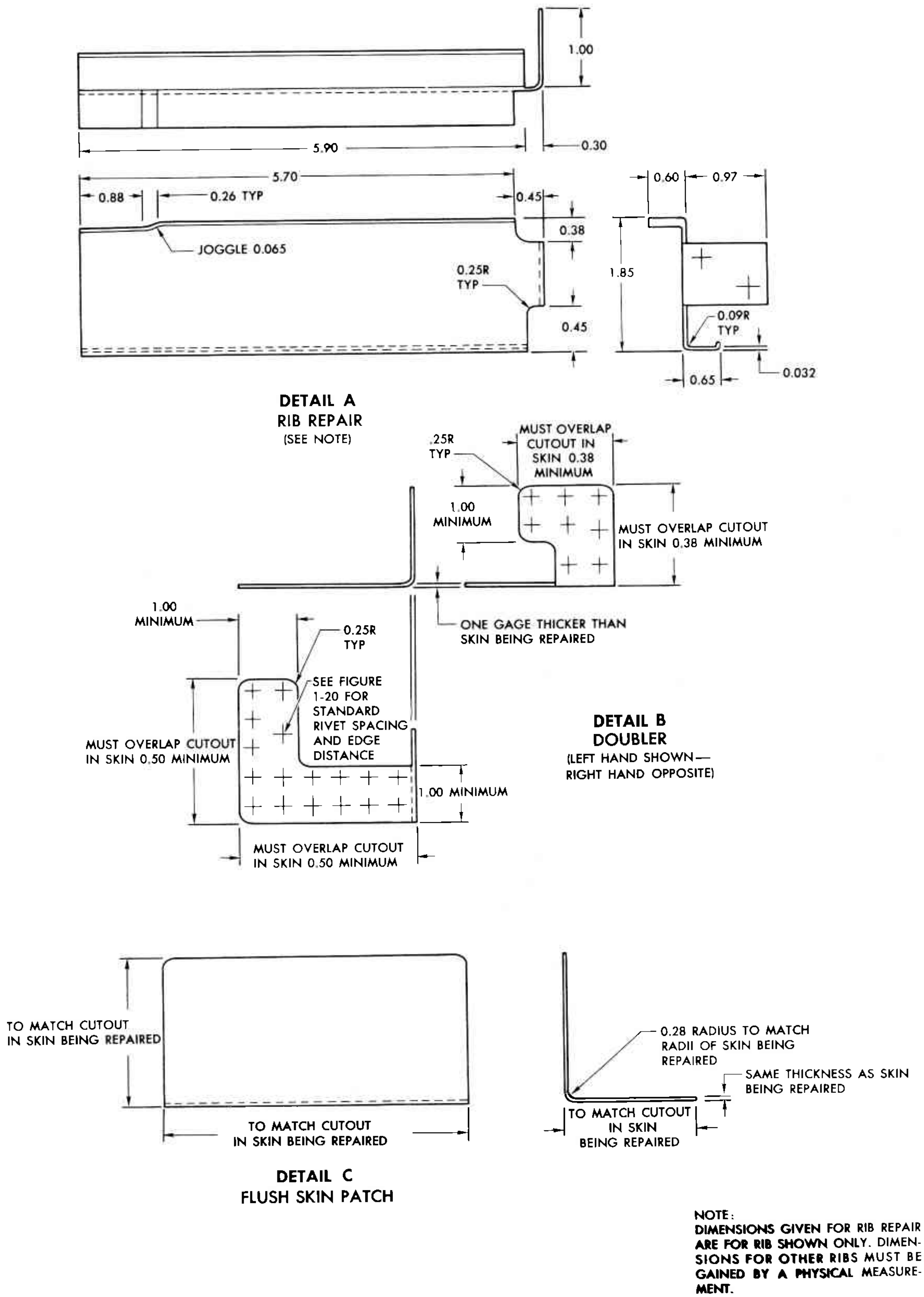
NOTE

REPAIR PARTS MAY BE MADE FROM STAINLESS STEEL IF TITANIUM IS NOT AVAILABLE.

e. Predrill holes in doublers as shown. Use a No. 40 (0.099) drill.
f. Install repair rib and hold in place with clamps.
g. Using the existing holes in speed brake door cap as a guide, drill holes through repair rib. Use a No. 30 (0.128) drill.
h. Remove repair rib and remove all burrs from repair rib and speed brake door cap.
i. Layout and drill holes through remaining portion of repair flange. Use a No. 40 (0.099) drill.
j. Reinstall rib and hold in place with clecos.
k. Install doublers and hold in place with clamps.
l. Using predrilled holes in doublers as a guide, drill holes through existing skin of speed brake door cap. Use a No. 40 (0.099) drill.

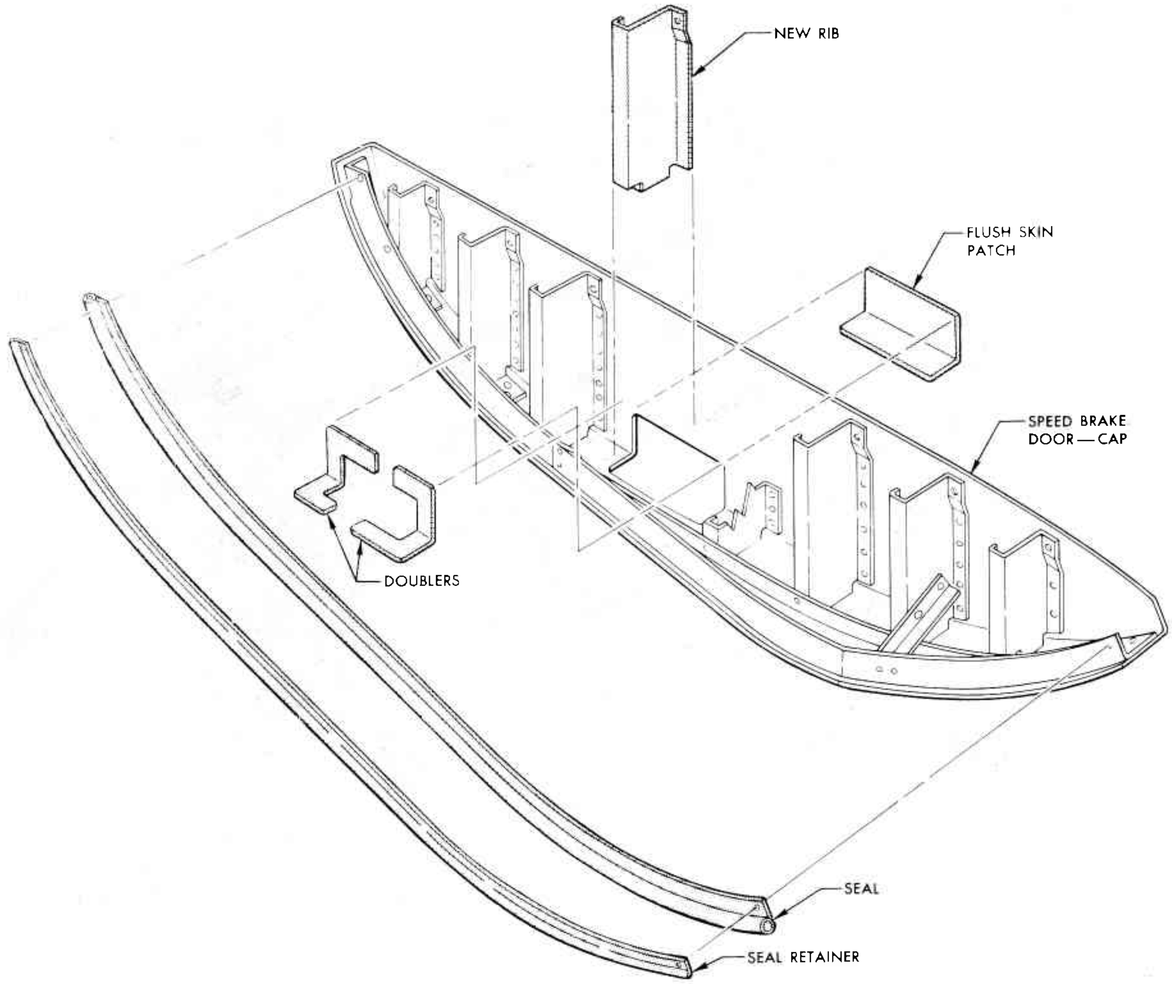
06 03 256 1B

Figure 4-41. Speed Brake Door Cap Repair (Sheet 1 of 3)



NOTE:
DIMENSIONS GIVEN FOR RIB REPAIR ARE FOR RIB SHOWN ONLY. DIMENSIONS FOR OTHER RIBS MUST BE GAINED BY A PHYSICAL MEASUREMENT.

Figure 4-41. Speed Brake Door Cap Repair (Sheet 2 of 3)



REPAIR INSTALLATION PROCEDURE

m. Insert clecos through holes drilled in step "l" and remove clamps.
n. Install flush skin patch.
o. Using predrilled holes in doublers and repair rib as a guide, drill holes through flush skin patch. Use a No. 40 (0.099) drill.

NOTE

INSERT CLECOS THROUGH HOLES IN FLUSH SKIN PATCH AS THEY ARE DRILLED TO KEEP FLUSH PATCH FROM SLIPPING OUT OF ALIGNMENT.

p. Ream holes drilled in steps "l" and "o" with a No. 30 (0.128) drill.
q. Countersink 100° all holes in existing skin and flush skin patch.
r. Remove repair parts and remove all burrs from repair parts and speed brake door cap.
s. Apply a coat of Sta-Bond C-875 to faying surface of repair parts and speed brakes door cap. Refer to Table 11-VII for application of Sta-Bond C-875.
t. Reinstall all repair parts and hold in place with clecos.
u. Rivet repair parts to speed brake door cap, as shown, with AN427M-4 rivets.
v. Fill any gaps between flush skin patch and existing speed brake door cap skin with Sta-Bond C-875.
w. Remove speed brake door safety locks and close speed brake door. Refer to T.O. 1F-106A-2-7-2-1 for procedure on removal of safety locks and closing of door.

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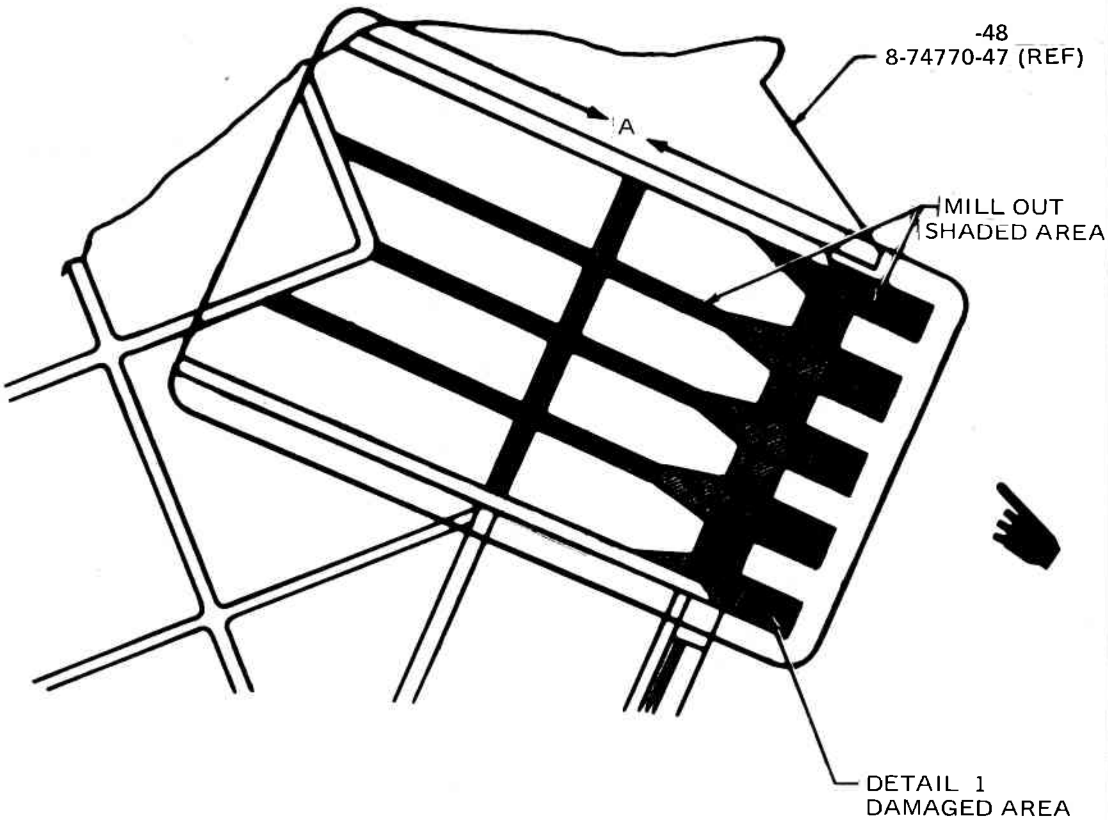
Figure 4-41. Speed Brake Door Cap Repair (Sheet 3 of 3)

REPAIR PROCEDURE (Figure 4-41A)



a. Mill out the webs in damaged area of the hinge as noted in detail 1.
b. Dress up damaged edge of door as noted in detail 1.
c. Remove the -12 retainer and the -20 seal from door. Retain for reinstallation.
d. Remove all paint from reworked area.
e. Fabricate one each hinge assy from 3 1/2 x 5 x 10 in, 7075-T6 aluminum alloy. See detail 2. Hinge will be milled out similar to original item including webs and hinge nodes.
f. Temporarily secure hinge in door while drilling No. 11 (0.191 inch dia) holes. Holes will be countersunk 100 degree on outer surface of skin.
g. Treat reworked magnesium areas with chrome pickle MIL-M-3171 (Type 1), treat aluminum hinge with alodine 1200 MIL-C-5541. Mate and bolt surfaces while wet with zinc chromate primer.
h. Finish reworked area with aircraft gray acry-lic nitrocellulose lacquer MIL-L-19537, Color 16473 per Federal Standard 595.

Figure 4-41A. Speed Brake Door Hinge – Typical Repair (Sheet 1 of 2)

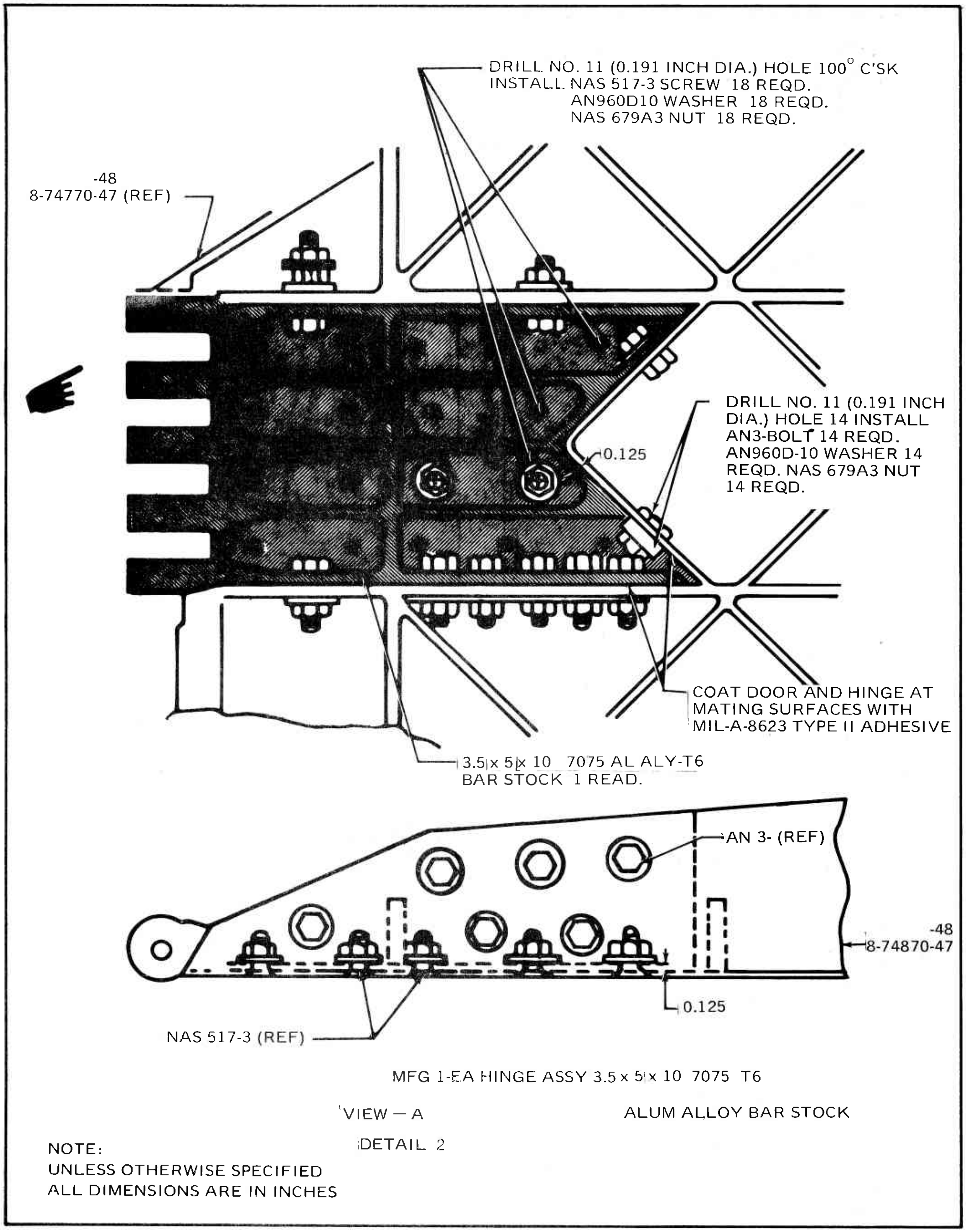
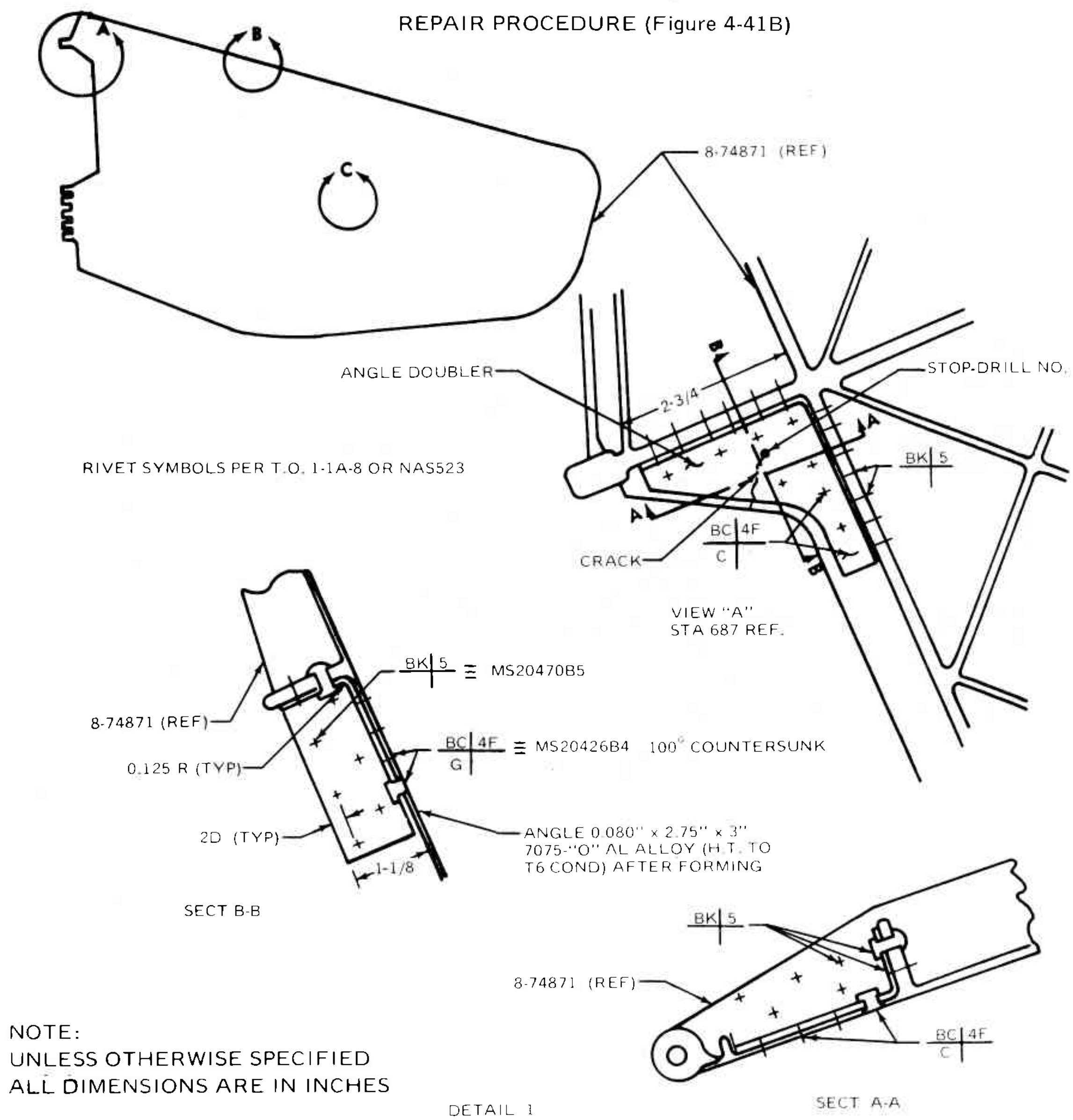


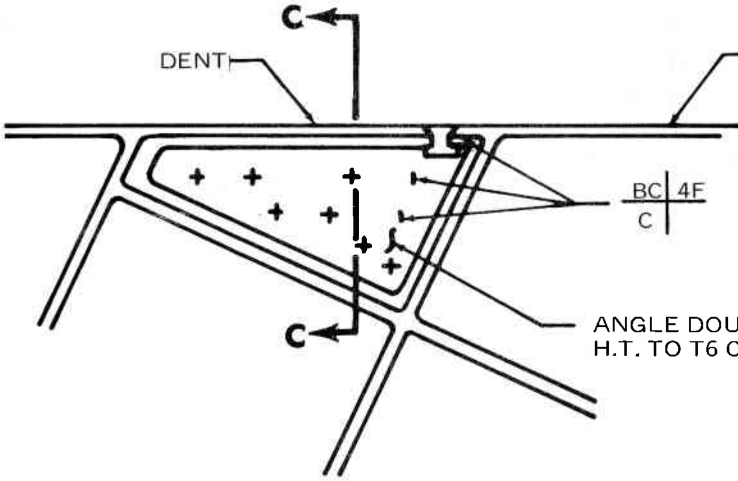
Figure 4-41A. Speed Brake Door Hinge — Typical Repair (Sheet 2 of 2)



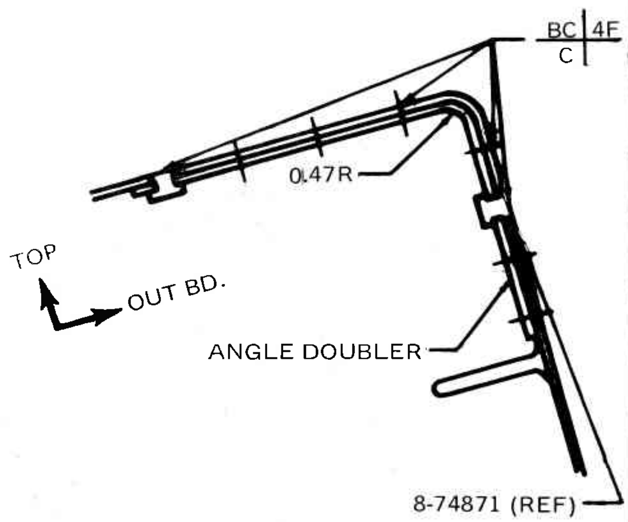
a. Remove all paint from damaged areas.
b. Fluorescent penetrant inspect the area to determine extent of damage.
c. Stop-drill the cracks with No. 30 size (0.128 inch dia.) drill (See detail 1, View A).
d. Fabricate angle doubler from 0.080 7075-0 aluminum alloy (after forming, H.T. to T6 condition).
e. Blend out all scratches and gouges around dented area (See detail 2, View B).
f. Fabricate angle doubler from 0.080, 7075-0 aluminum alloy (after forming, H.T. to T6 condition).
g. Clean up hole and make filler to fit. (See detail 3 and 4, View C). Use 7075-T6 aluminum alloy for filler. Thickness and diameter to be determined at time of fabrication.
h. Fabricate angle doubler from 0.050 7075-0 aluminum alloy (H.T. to T6 after forming).
i. Treat exposed magnesium surfaces with Chemical Film, Specification MIL-M-3171, Type 1.
j. Temporarily secure doubler angles in place and drill No. 30 size (0.128 inch dia) holes through doubler and casting. Maintain 2D edge distance on rivets and install approximately as shown.
k. Prime coat fabricated parts and reworked area with zinc primer MIL-P-8585.
l. Coat mating surfaces of doubler angles and casting with MIL-A-8623 Type II, before final assembly.

Figure 4-41B. Speed Brake Door – Typical Repair (Sheet 1 of 4)

NOTE:
UNLESS OTHERWISE SPECIFIED
ALL DIMENSIONS ARE IN INCHES



VIEW — B
STA. 703 REF
ROTATED APPROX. 20° CCW

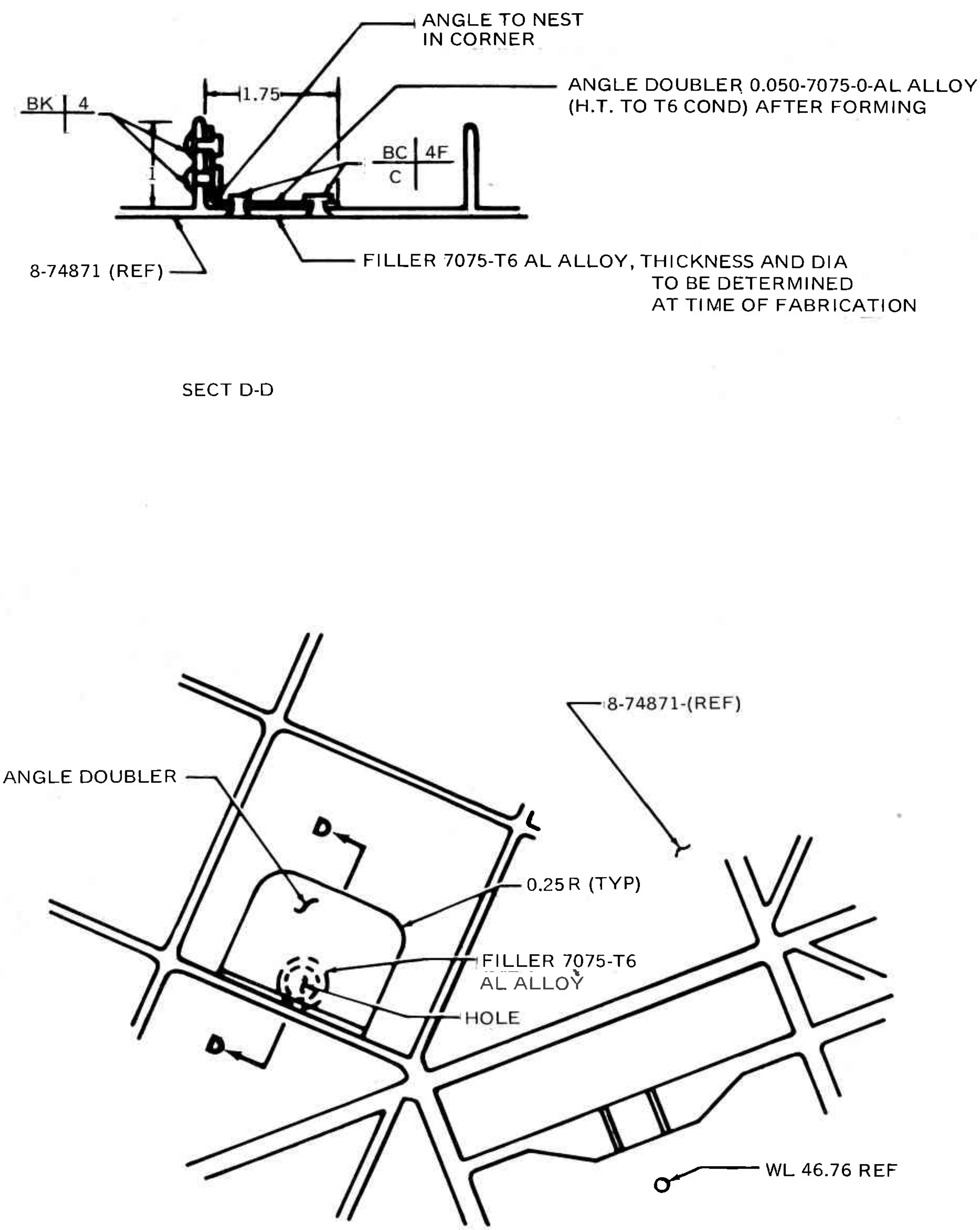


NOTE: SHAPE OF DOUBLER ANGLE AND RIVET PATTERN TO BE DETERMINED AT TIME OF REWORK.

SECT C-C
ROTATED APPROX. 15° CCW

DETAIL 2

Figure 4-41B. Speed Brake Door — Typical Repair (Sheet 2 of 4)



NOTE:
UNLESS OTHERWISE SPECIFIED
ALL DIMENSIONS ARE IN INCHES

DETAIL 3

Figure 4-41B. Speed Brake Door — Typical Repair (Sheet 3 of 4)

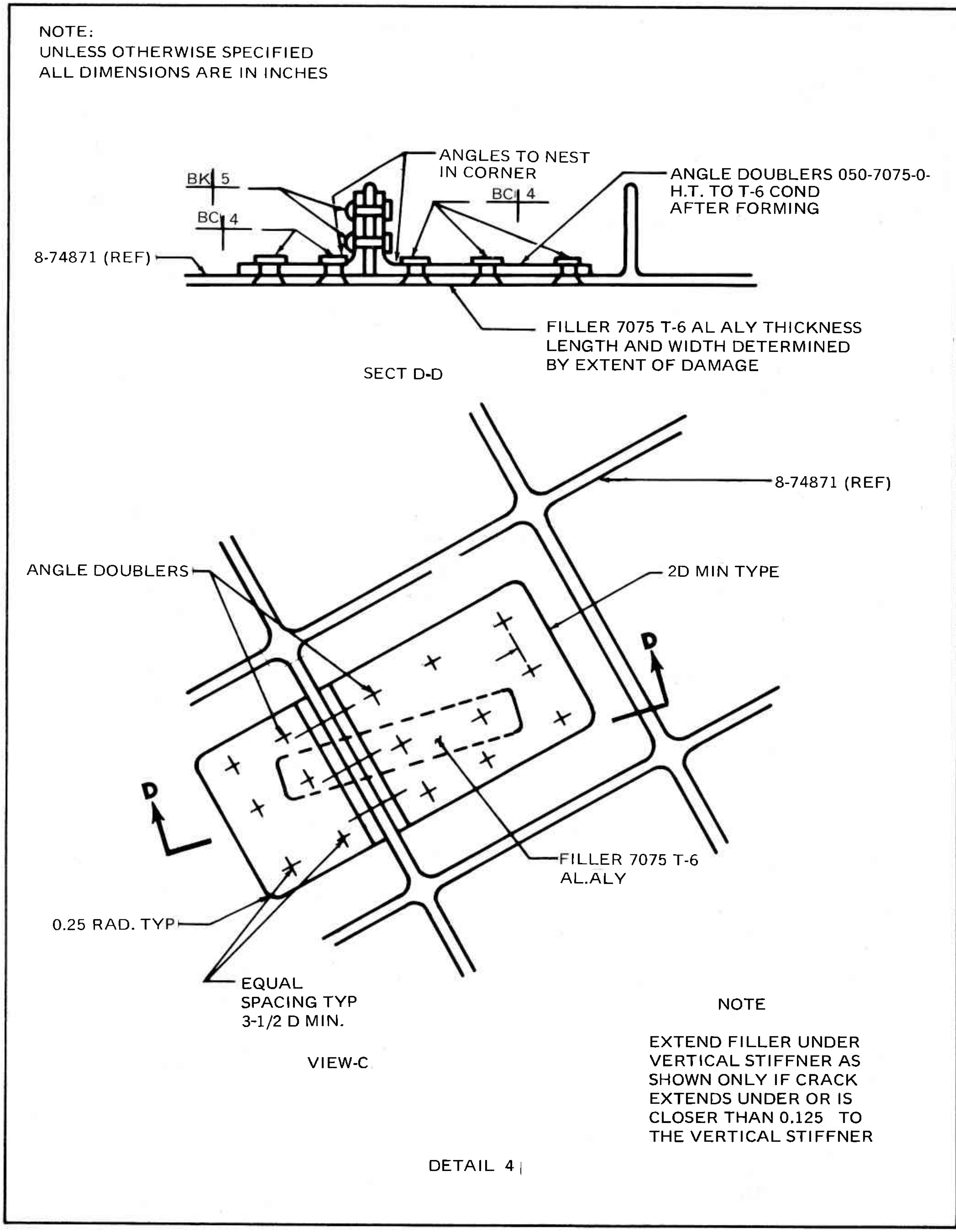
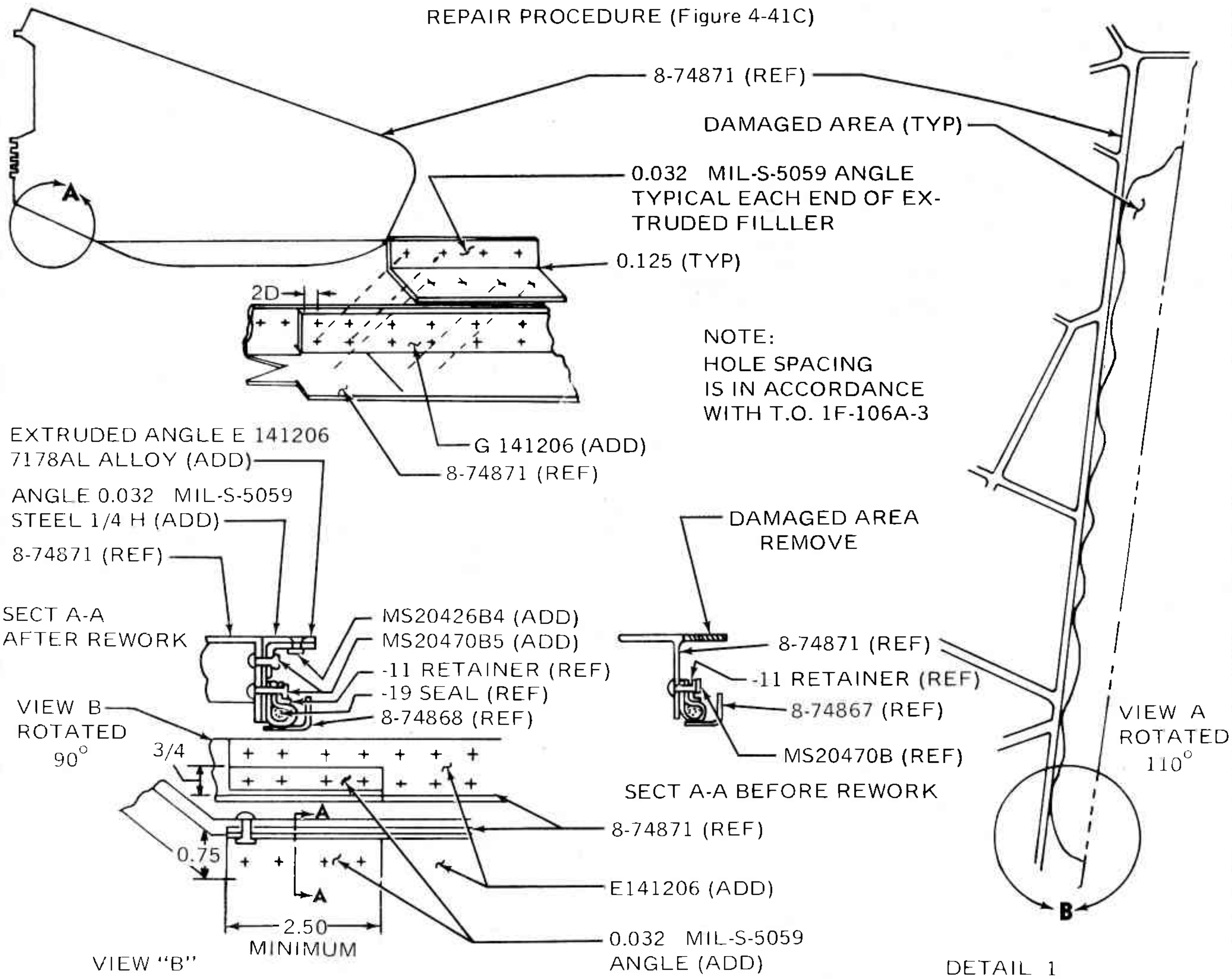


Figure 4-41B. Speed Brake Door — Typical Repair (Sheet 4 of 4)

REPAIR PROCEDURE (Figure 4-41C)

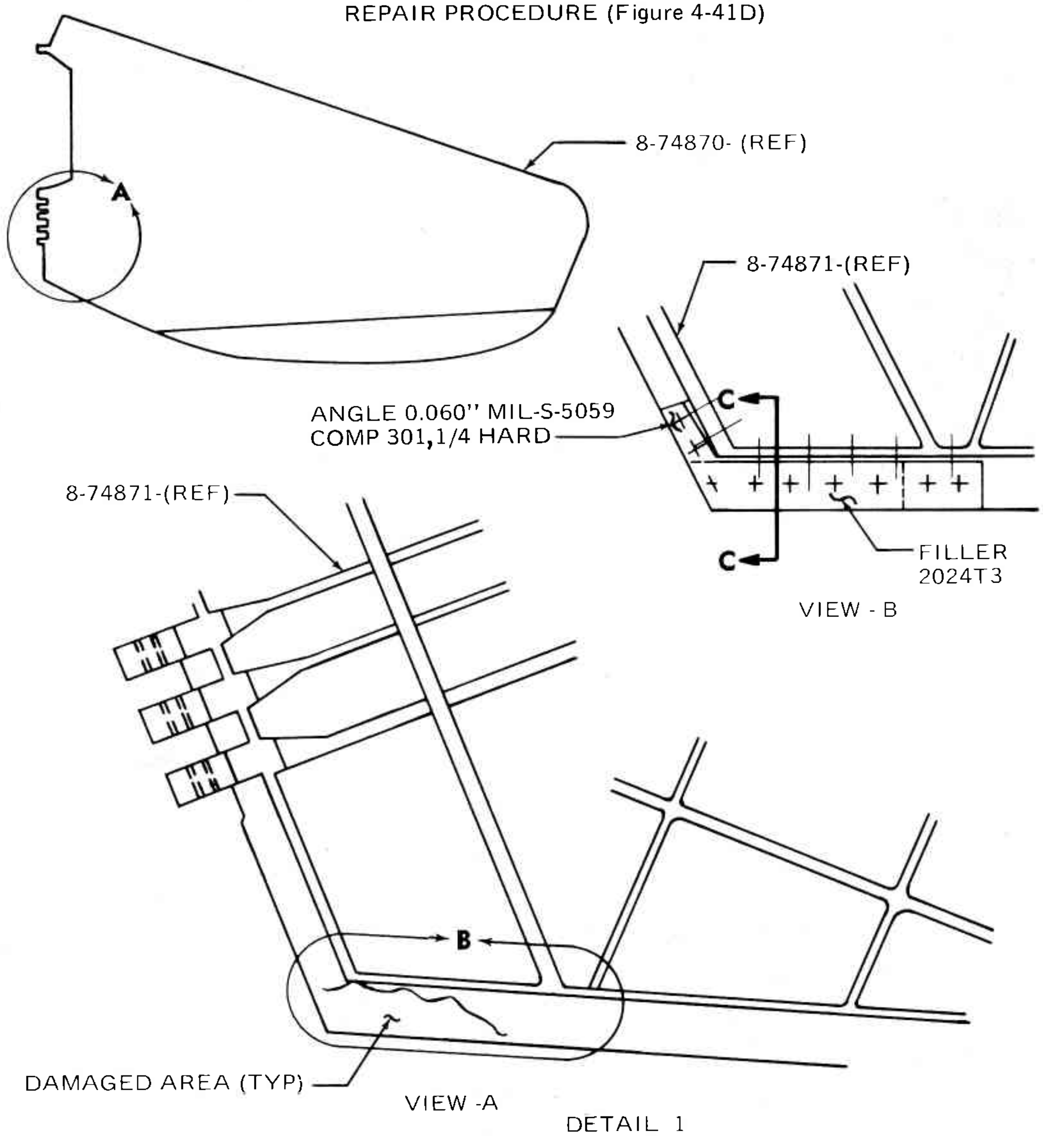


a. Remove the -11 retainer and -19 seal, retain for reinstallation.
b. Remove paint from rework area.
c. Fluorescent penetrant inspect the area to determine extent of damage.
d. Remove damaged area indicated in detail 1.
e. Treat exposed magnesium surface with Chemical Film, Specification MIL-M-3171, Type 1.
f. Fabricate filler from extruded angle E 141206, 7178 aluminum alloy. Trim extrusion to required size.
g. Temporarily secure filler in place. Use the -19 seal attaching holes as template for locating upper row of holes in extruded angle filler. Drill holes to No. 20 (0.161 inch dia).
h. Drill lower row of No. 20 drill size (0.161 inch dia) holes in filler and web of door.
i. Fabricate two 0.032 angles from MIL-S-5059A, Comp, 301, 1/4 hard material. Drill four No. 20 drill size (0.161 dia) holes in upper face of angle, using lower row of holes (reference paragraph h) in filler and web as a template.
j. Drill four each No. 30 drill size (0.128 inch dia) holes in lower edge of angle. Use the holes as template for locating holes in door and extruded angle filler.
k. Extruded angle shall be anodized or treated per MIL-C-5541 after fitting and drilling.
l. After anodizing extruded angle, apply two coats of Zinc Chromate Primer, MIL-P-8585A in accordance with MIL-P-6808.
m. Apply two coats of Zinc Chromate Primer, MIL-P-8585A in accordance with MIL-P-6808 to the 0.032 angles.
n. Secure extrusion angles, -19 seal and -11 retainer to door with MS20426B4 and MS20470B5 rivets, in accordance with View B, detail 1.
o. Finish repaired area with aircraft gray as indicated in T.O. 1F-106A-23.

NOTE:
UNLESS OTHERWISE SPECIFIED
ALL DIMENSIONS ARE IN INCHES

Figure 4-41C. Speed Brake ***Flange Repair***

REPAIR PROCEDURE (Figure 4-41D)



a. Remove the -11 retainer and -19 seal, retain for reinstallation.
b. Remove paint from rework area.
c. Fluorescent penetrant inspect the area to determine extent of damage.
d. Remove damaged area as necessary to assure a satisfactory repair. (See detail 1).
e. Treat exposed magnesium surface with chemical film, Specification MIL-M-3171, Type 1.
f. Fabricate filler from 0.080 2024-T3 aluminum alloy.
g. Fabricate two 0.060 angles from MIL-S-5059A, Comp 301, 1/4 Hard Material. (See detail 2).
h. Drill No. 30 drill size (0.128 inch dia.) holes in lower edge of angle. Use these holes as template for locating holes in door and filler. Countersink holes in lower side of filler and door to 100 degrees. (See detail 2).
i. Drill No. 20 drill size (0.161 inch dia) holes in upper surface of angle (See detail 2).
j. Use angle as template for locating holes.
k. Apply two coats of Zinc Chromate Primer, MIL-P-8585A per MIL-A-6808 to 0.080 filler.
l. Apply two coats of Zinc Chromate Primer, MIL-P-8585A, per MIL-A-6808 to 0.060 angle.
m. Secure filler to angle using MS20426B4 rivets as shown in detail 2.
n. Secure repair angle, -19 seal and -11 retainer to door with MS20470B5 rivets as shown in detail 2.
o. Finish repaired area with Aircraft Gray acrylic nitrocellulose lacquer MIL-L-19537, color 16473 per Federal Standard 595.

Figure 4-41D. Speed Brake Door Flange Repair (Sheet 1 of 2)

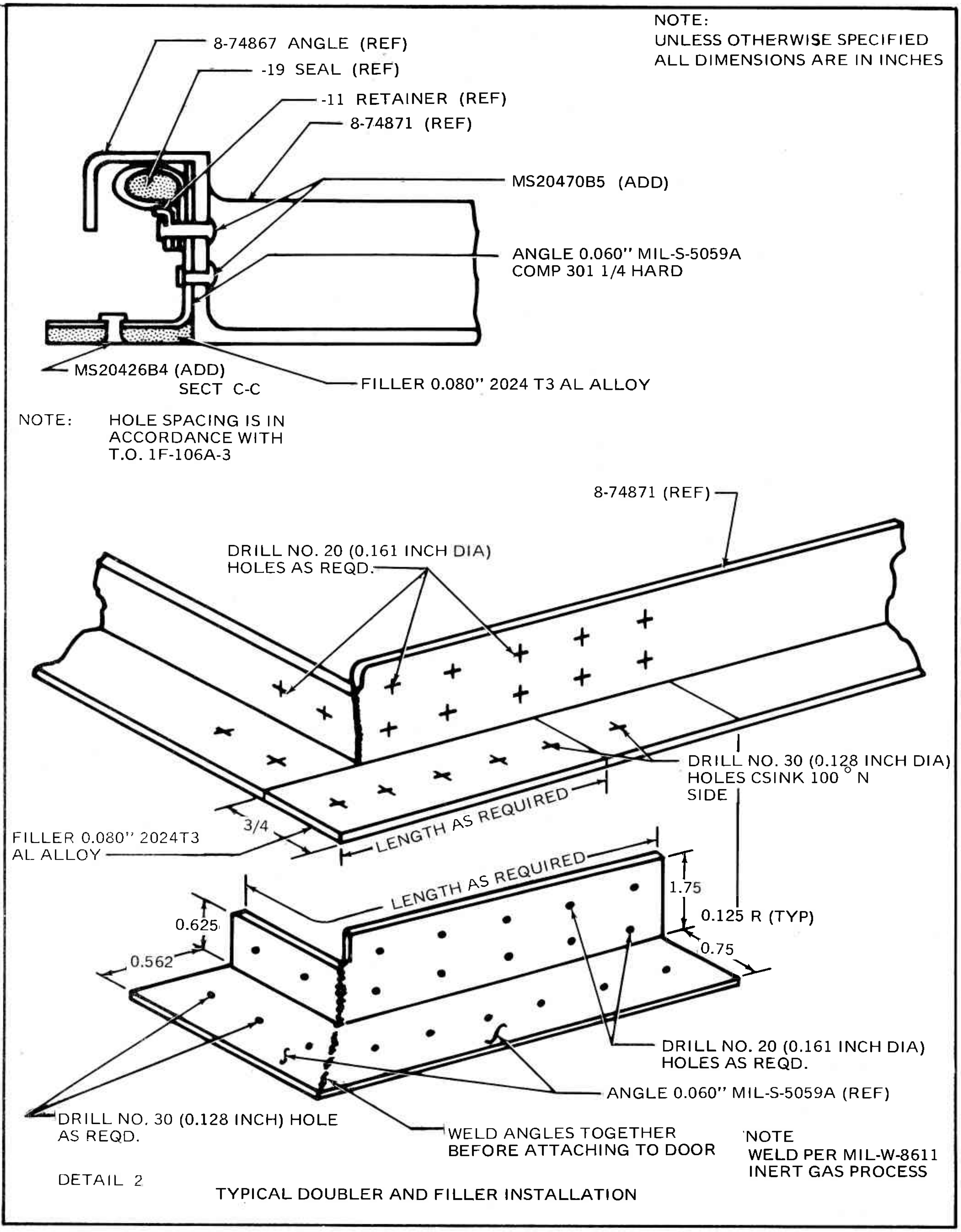
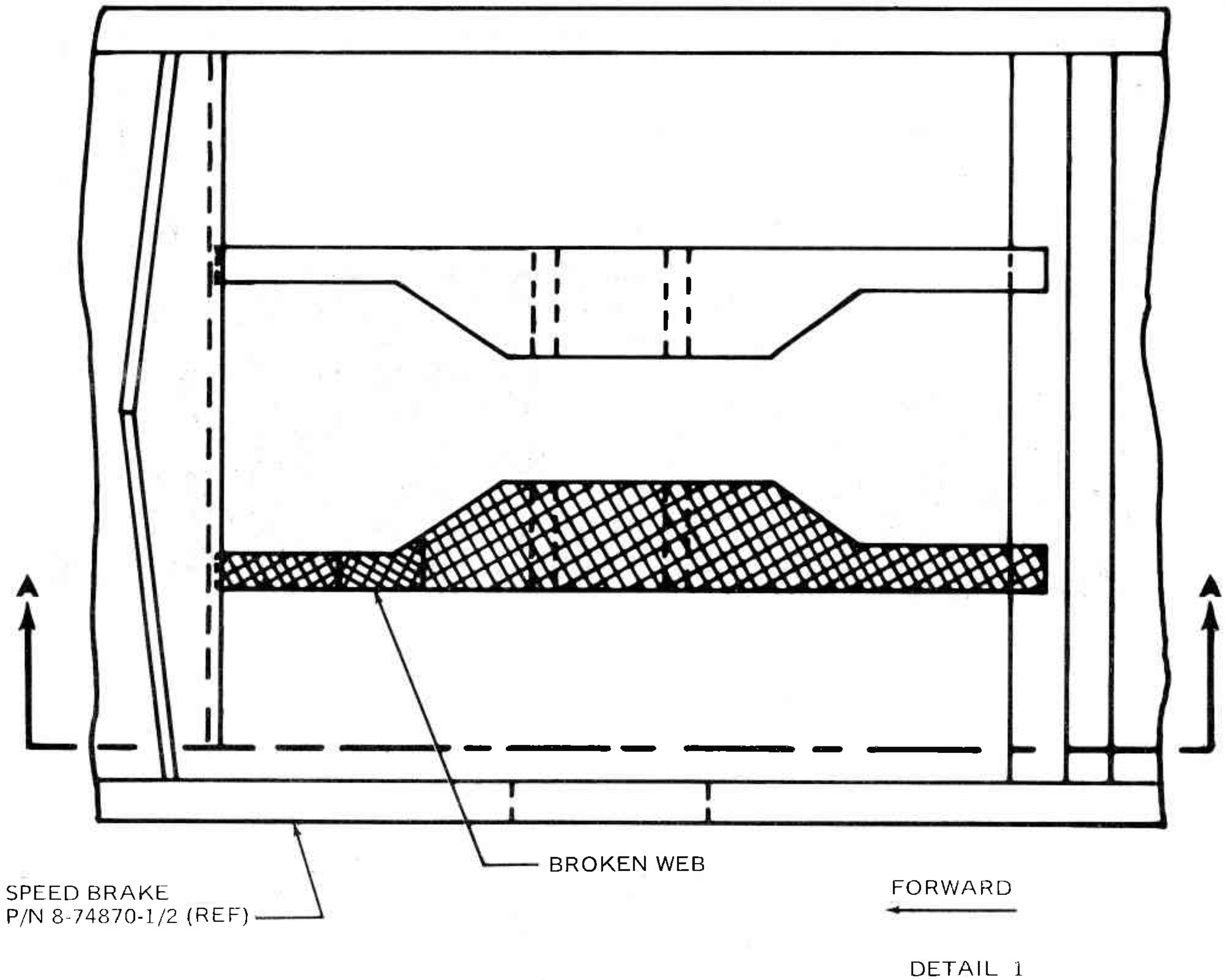


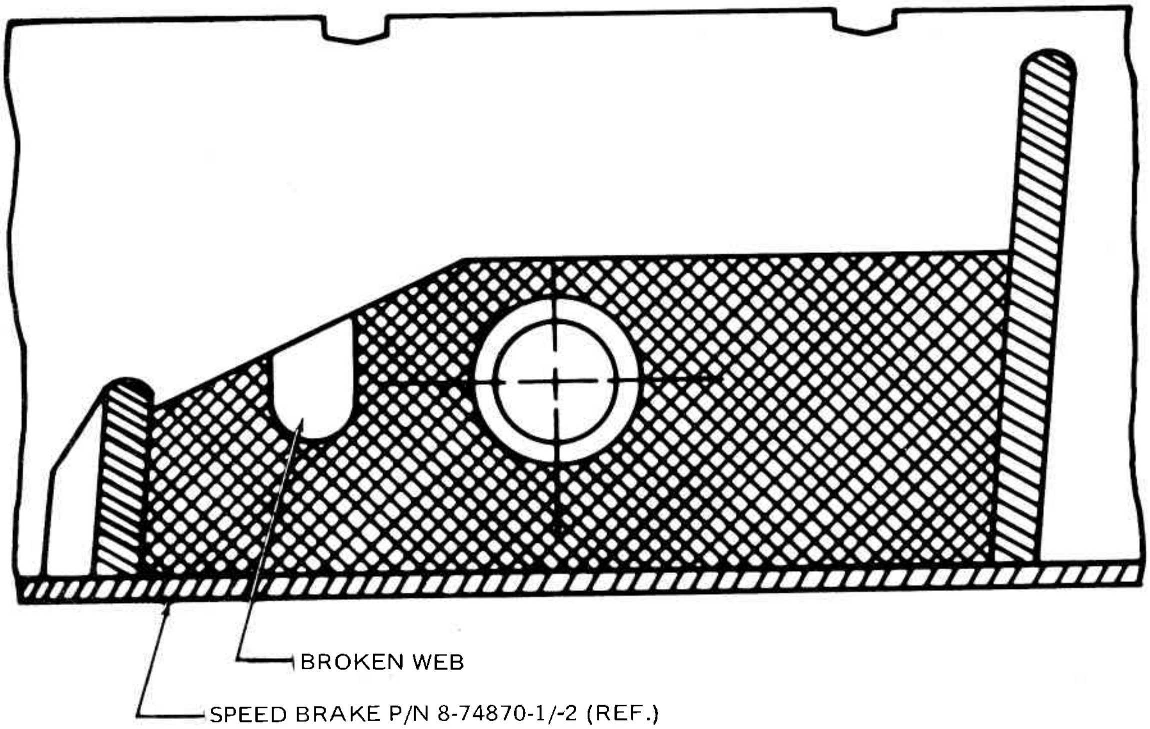
Figure 4-41D. Speed Brake Door Flange Repair (Sheet 2 of 2)

REPAIR PROCEDURE (Figure 4-41E)



a. Remove double cross hatched portion by milling as shown in detail 1 and Section A-A, detail 2.
b. Remove paint from rework area; treat exposed magnesium surface with chemical film, Specification MIL-M-3171, Type 1.
c. Manufacture new actuator fitting identical to milled portion with flanges so that fitting can be bolted to skin and webs. Fitting will be made blank and actuator bolt hole will be line-drilled after installation of fitting. Install bushing same as original bushing. (See detail 3 and instructions on detail 2.)
d. Temporarily install fitting and drill eight number 11 drill size holes (0.191 dia), five holes through skin and three holes through webs. The five holes through the skin are to be counter sunk 100 degree on the outside of the skin. (See detail 3 and detail 2 instructions.)
e. Coat mating surfaces with MIL-A-8623 Type II, at final assembly.
f. Hardware required: Five NAS 517-3 screws, three AN3 bolts, eight NAS 679A3 nuts, and eight AN960D-10 washers.
g. Apply zinc chromate primer, MIL-P-8585A per MIL-P-6808A, to exposed surfaces on fitting and door.

Figure 4-41E. Speed Brake Door Web Repair (Sheet 1 of 3)



SECTION A - A

REMOVE DOUBLE CROSSHATCHED PORTION BY MILLING (DETAIL 1 AND SECT. A - A DETAIL 2)
REMOVE PAINT FROM REWORK AREA, TREAT EXPOSED MAGNESIUM SURFACE WITH CHEMICAL FILM SPEC MIL-M-3171, TYPE 1.
MANUFACTURE NEW ACTUATOR FITTING FROM 2 x 2 x 6 INCH 7075-T6 ALUMINUM ALLOY (DETAIL 3).
DRILL NO. 11 DRILL SIZE HOLES (0.191 DIA.) (5) THRU OUTER SKIN 100° C'SINK OUTSIDE (3) THRU WEBS. ON ASSY INSTALL 5 NAS 517-3 SCREWS, 3 AN3 BOLTS, 8 AN960D-10 WASHERS AND 8 NAS 679A3 NUTS. USE MIL-A-8673 TYPE II ADHESIVE ON ALL MATING SURFACES AT FINAL ASSY.
FITTING WILL BE MADE BLANK AND ACTUATOR BOLT HOLE WILL BE LINE-DRILLED AFTER ASSY.
APPLY ALODINE 1200 MIL-C-5541 TO FITTING BEFORE ASSY.
APPLY ALODINE TO INSIDE OF BOLT HOLE AFTER LINE DRILLING.

DETAIL 2

Figure 4-41E. Speed Brake Door Web Repair (Sheet 2 of 3)

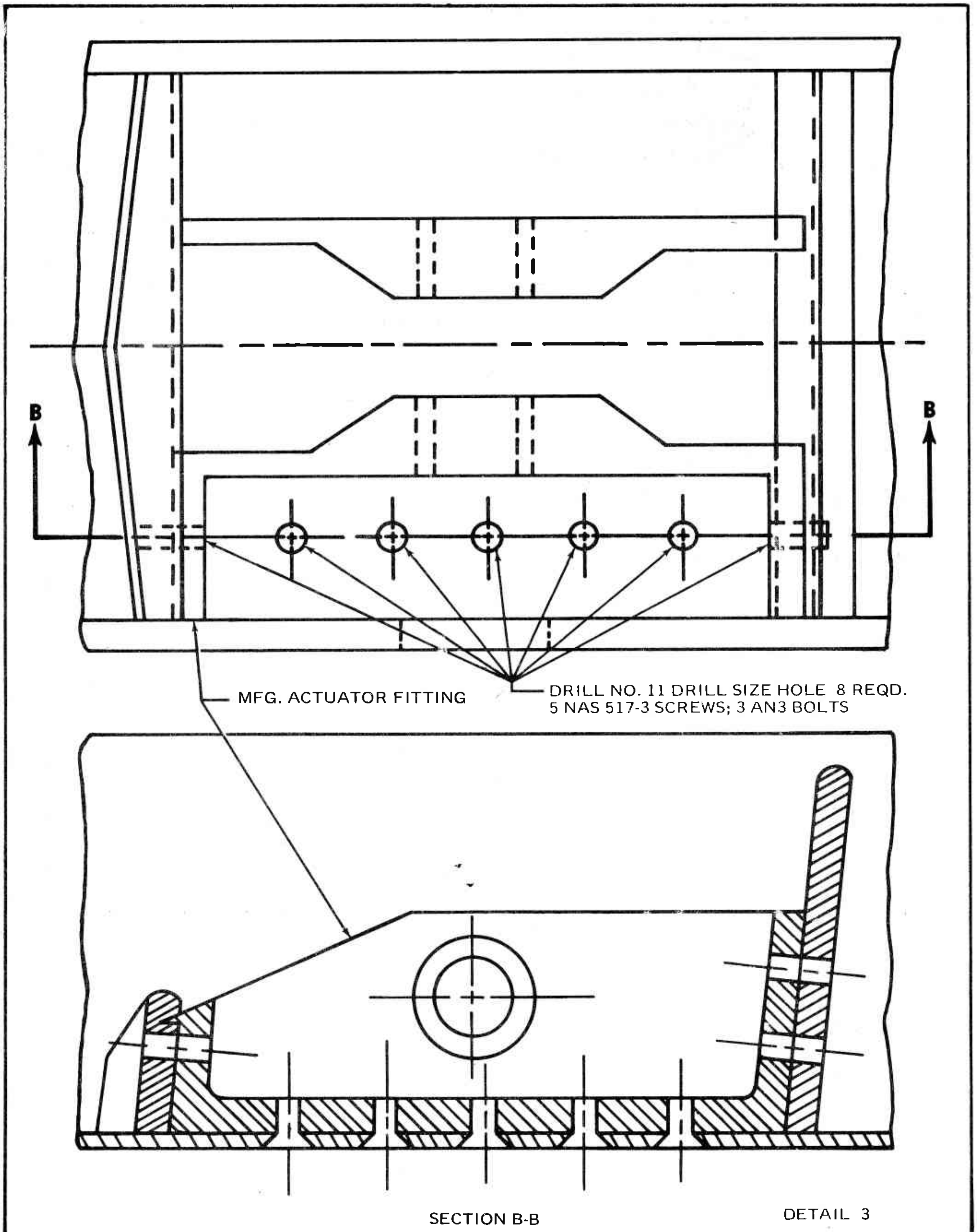
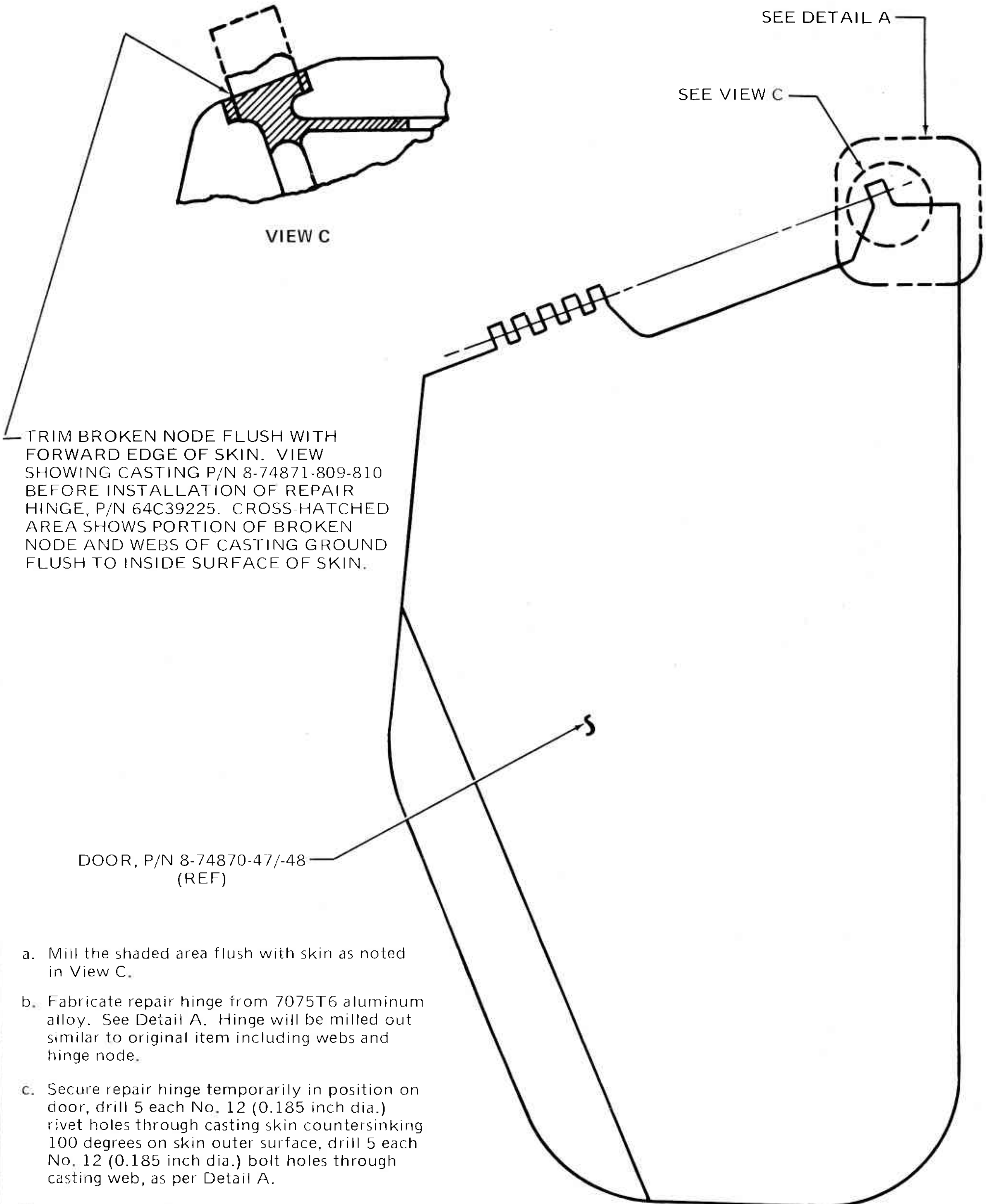


Figure 4-41E. Speed Brake Door Web Repair (Sheet 3 of 3)



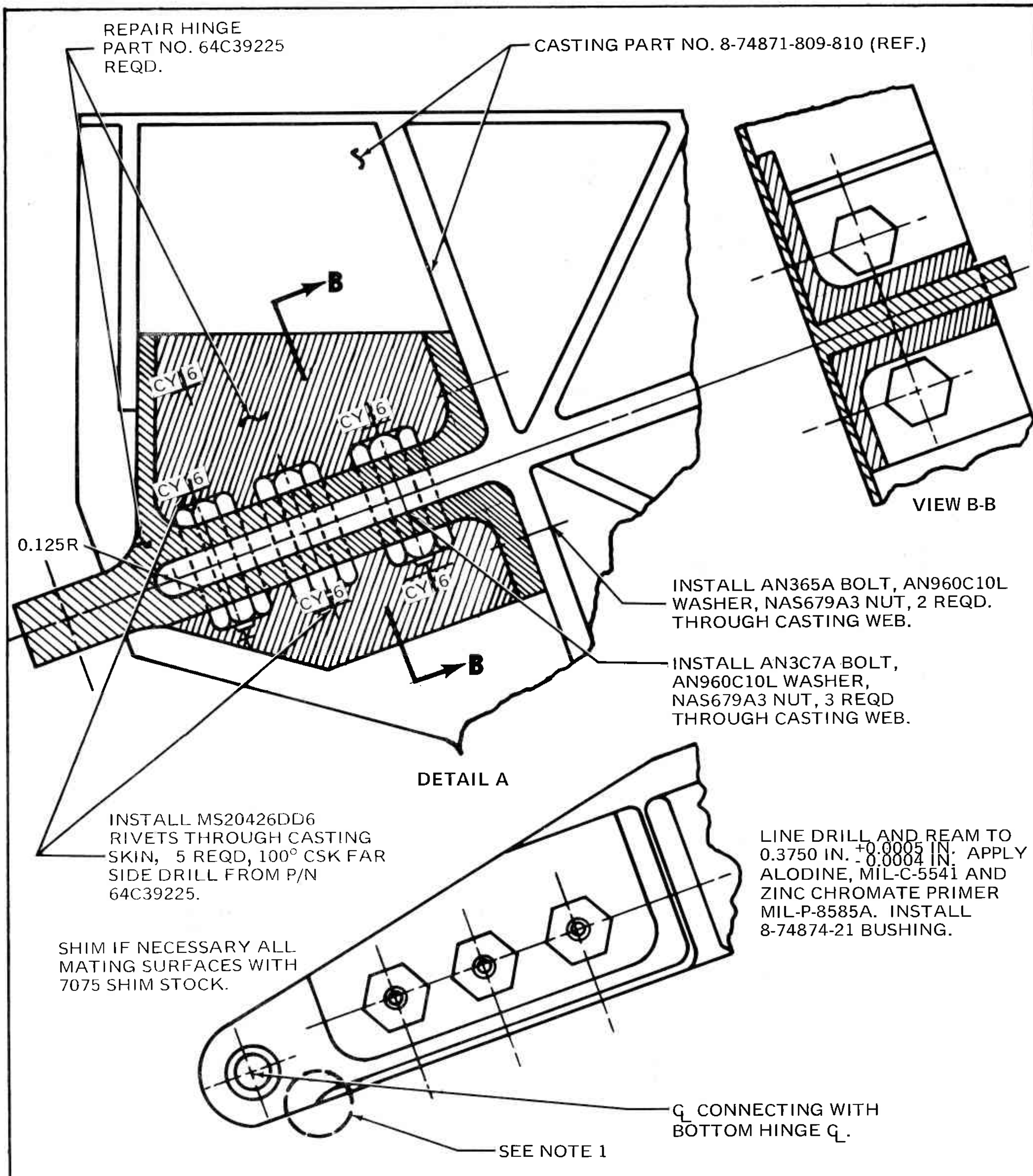
a. Mill the shaded area flush with skin as noted in View C.

b. Fabricate repair hinge from 7075T6 aluminum alloy. See Detail A. Hinge will be milled out similar to original item including webs and hinge node.

c. Secure repair hinge temporarily in position on door, drill 5 each No. 12 (0.185 inch dia.) rivet holes through casting skin countersinking 100 degrees on skin outer surface, drill 5 each No. 12 (0.185 inch dia.) bolt holes through casting web, as per Detail A.

d. Treat reworked magnesium and aluminum as indicated in T.O. 1F-106A-23.

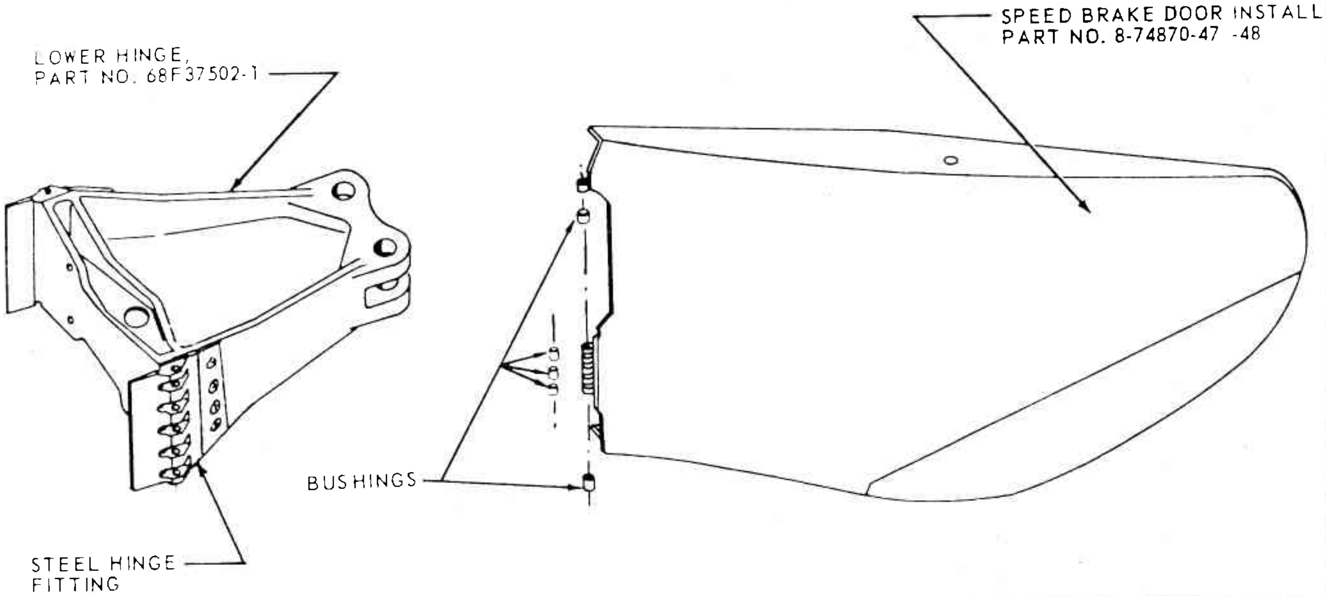
Figure 4-41F. Speed Brake Door Upper Hinge Repair (Sheet 1 of 2)



NOTES:

1. BEVEL TO AN ANGLE OF 15°, THE INSIDE FORWARD EDGE OF THE SKIN, A DISTANCE OF 0.3 INCHES ON EITHER SIDE OF THE ℄ OF THE ORIGINAL HINGE NODE.
2. COAT DOOR PART NO. 8-74870-47-4S AND MATING SURFACES OF REPAIR HINGE WITH EPOXY RESIN ADHESIVE EC-1469, MIL-A-8623, MINN MINING AND MANUFACTURING CO.

Figure 4-41F. Speed Brake Door Upper Hinge Repair (Sheet 2 of 2)



REWORK INSTRUCTIONS

A. Remove speed brake door. Refer to T.O. 1F-106A-2-7-2-1 for procedure.

B. Remove all loose/worn hinge bushings that do not have the required press fit.

C. Remove all bushings with ID exceeding 0.2500-inch plus 0.0017-inch minus 0.0000-inch service wear limits.

D. Using back-cutting reamer from the lower side, ream worn/elongated hinge node holes in the speed brake door and/or hinge fittings as required to 0.3245 inch plus 0.0005 inch minus 0.0004 inch ID.

E. Fluorescent penetrant — inspect the hinge fitting and speed brake hinge for cracks and/or defects.

F. Treat rework hole cavities in accordance with T.O. 1F-106A-23.

G. Manufacture 0.012-inch oversize bushings (reference Drawing 8-74874) and install to an interference fit of 0.0005 – 0.0015-inch. Freeze bushings in dry ice alcohol solution 5 minutes prior to installation.

H. Reinstall speed brake doors (refer to T.O. 1F-106A-2-7-2-1 for procedure).

I. Install bushings as required in worn/elongated hinge cavities in lower hinge steel fitting, in accordance with above rework instructions (see Paragraph C for wear limits).

Note

Reinstall loose bushings not exceeding the wearout limits in paragraph C with loctite, Specification MIL-R-46082, Type 1, and locquic primer, Specification MIL-S-22473, Grade T, Form R.

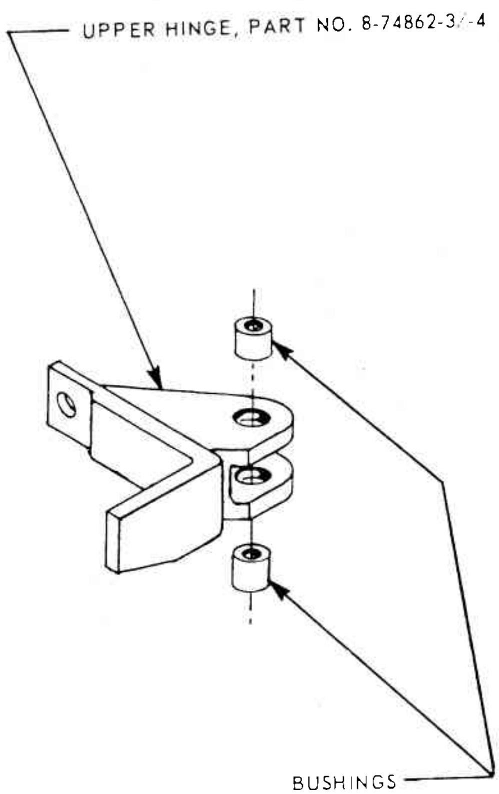


Figure 4-41G. Speed Brake Hinge **Bushing Repair** - F-**106 Aircraft**

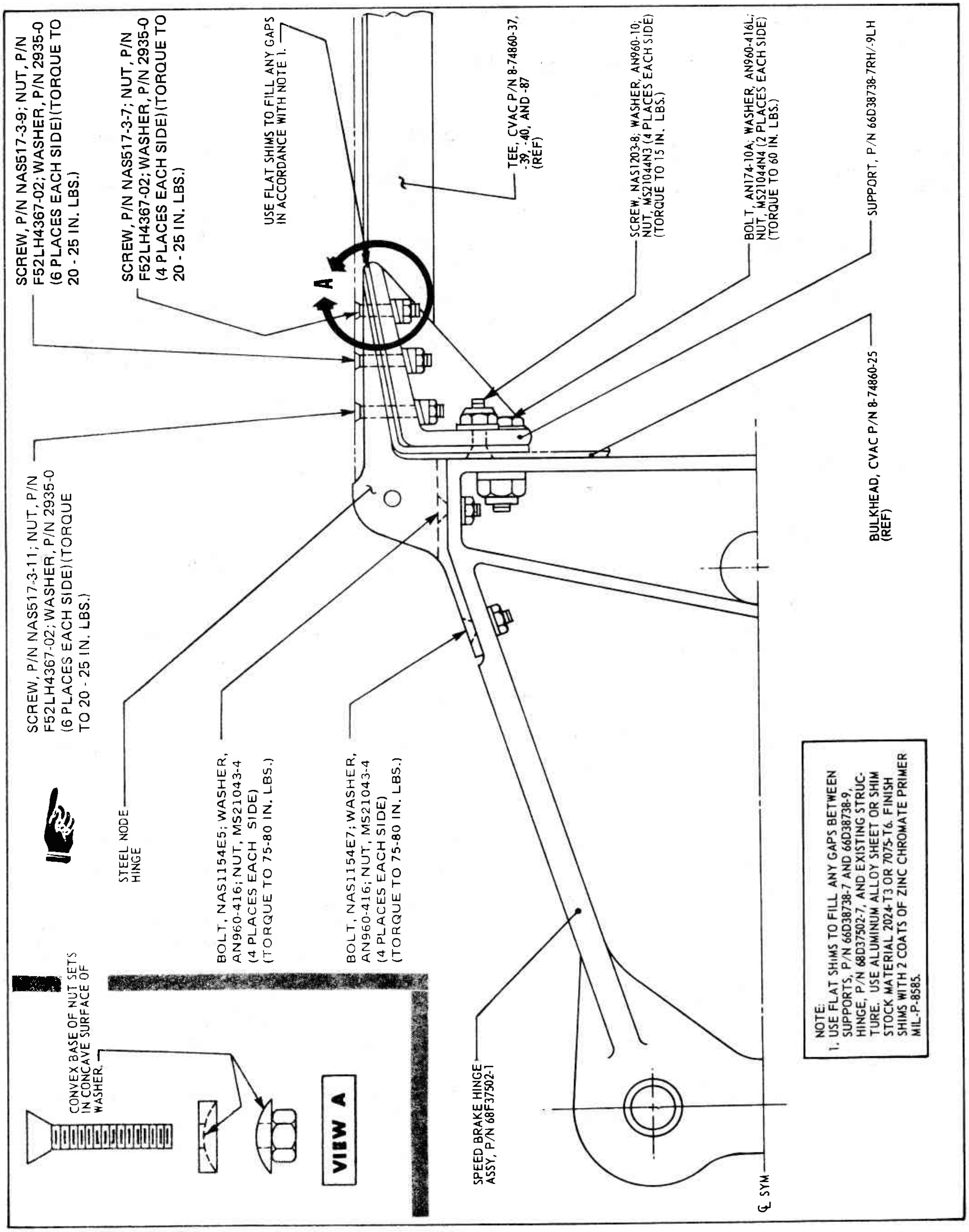
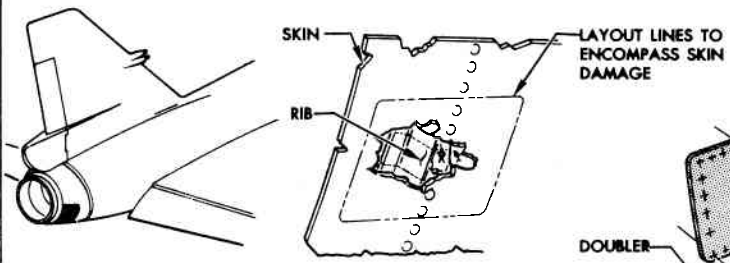
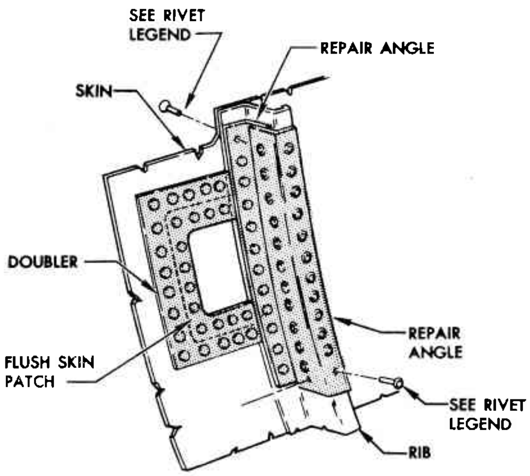


Figure 4-41H. Speed Brake Hinge Fitting Installation Hardware

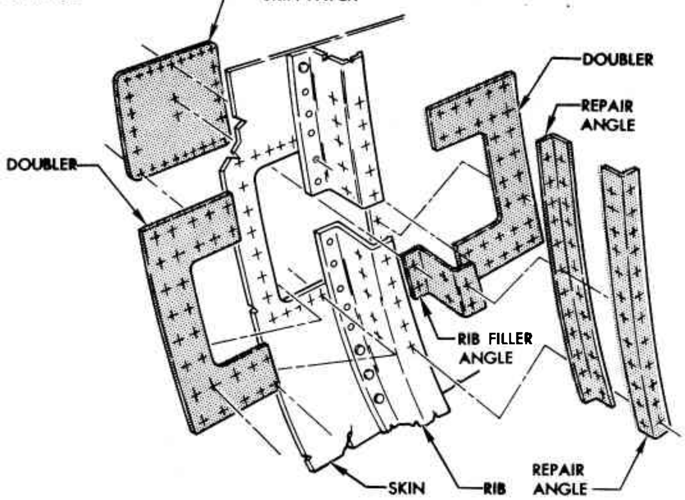
.06.03.153-1B



AREA OF ASSUMED DAMAGE



ASSEMBLED REPAIR
VIEW ROTATED 180°



EXPLODED REPAIR
VIEW ROTATED 180°

RIVET LEGEND	
INTERNAL RIVETS FWD. OF STA. 698,00	MS20470AD-4
INTERNAL RIVETS AFT. OF STA. 698,00	MS20615M-4
EXTERNAL RIVETS FWD. OF STA. 698,00	MS20426AD-4
EXTERNAL RIVETS AFT. OF STA. 698,00	MS20427M-3

REPAIR PROCEDURE

a. Remove the tail cone. Refer to T.O. 1F-106A-2-4-2-1 for tail cone removal procedure.

b. Locate the damage area and perform a fluorescent penetrant inspection to determine extent of damage. Refer to Section I for fluorescent penetrant inspection procedures and materials.

NOTE

IF DAMAGE IS IN PAINTED AREA OF TAIL CONE REMOVE PAINT. USE METHYL ETHYL KETONE, FEDERAL SPECIFICATION TT-M-261 FOR PAINT REMOVAL.

c. Using metal cutting hand shears, remove damaged portion of skin as shown.

d. Using metal cutting hand shears, remove damaged portion of rib as shown.

e. Fabricate repair parts as shown.

NOTE

REPAIR PARTS FORWARD OF STATION 698.00 ARE TO BE MADE FROM 2024-T81. REPAIR PARTS AFT OF STATION 698.00 ARE TO BE MADE FROM AMS4901 TITANIUM. ONE-QUARTER HARD 301 STAINLESS STEEL MAY BE USED AS AN ALTERNATE FOR AMS4901.

f. Install repair angles and hold in place with clamps.

g. Install rib filler angle and hold in place with clamps.

h. Using predrilled holes in repair angles and existing holes in the rib as a guide, perform drilling operation. Use a No. 30 (0.128) drill forward of station 698.00. Use a No. 39 (0.099) drill aft of station 698.00.

i. Remove repair parts and remove all burrs from repair parts and tail cone.

NOTE

IF REPAIR PARTS ARE MADE FROM ALUMINUM, REFER TO SECTION I FOR APPLICABLE PAINT FINISH.

j. Reinstall repair angles and hold in place with clecos.

k. Rivet repair angles together as shown.

l. Install doublers and hold in place with clamps.

m. Using predrilled holes in doublers as a guide, drill holes through existing skin. Use a No. 30 (0.128) drill forward of station 698.00. Use a No. 39 (0.099) drill aft of station 698.00.

n. Install flush skin patch.

o. Using predrilled holes in patch and existing holes in rib as a guide, perform drilling operation. Use a No. 30 (0.128) drill forward of station 698.00. Use a No. 39 (0.099) drill aft of station 698.00.

p. Remove doublers and patch and remove all burrs.

NOTE

SKIN REPAIRS FORWARD OF STATION 698.00 REQUIRE MACHINE COUNTERSUNK HOLES IN EXTERIOR SURFACE OF SKIN. SKIN REPAIRS AFT OF STATION 698.00 REQUIRE HOLES IN EXTERIOR SURFACE TO BE HEAT DIMPLED.

q. Reinstall doublers and patch and hold in place with clecos.

r. Rivet doublers and skin patch in place as shown.

s. Refer to applicable finish requirements in Section I if touch-up painting is required.

t. Reinstall tail cone. Refer to T.O. 1F-106A-2-4-2-1 for installation procedure.

.06.03.153-1B

Figure 4-42. Tail Cone Skin and Rib Repair (Sheet 1 of 2)

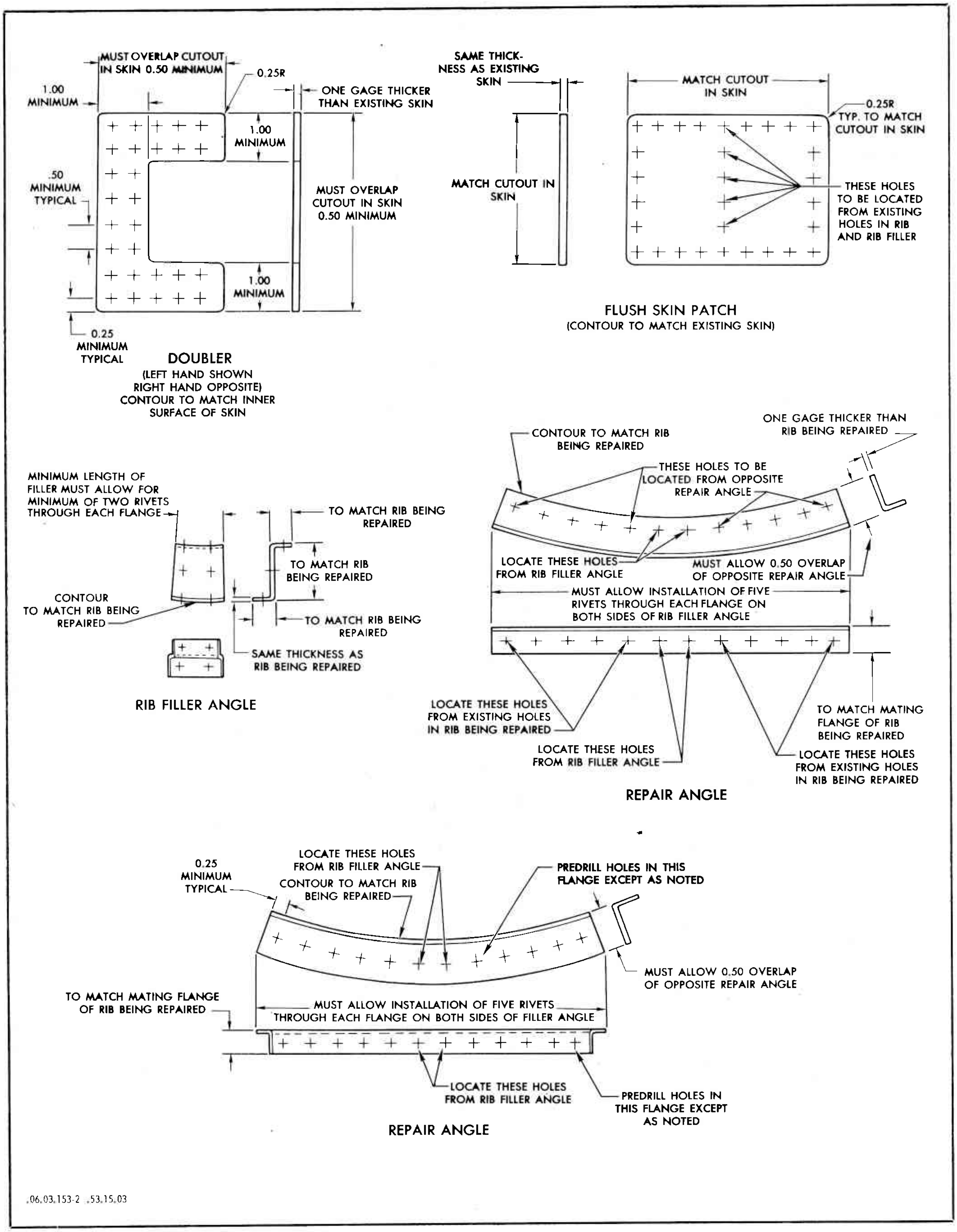
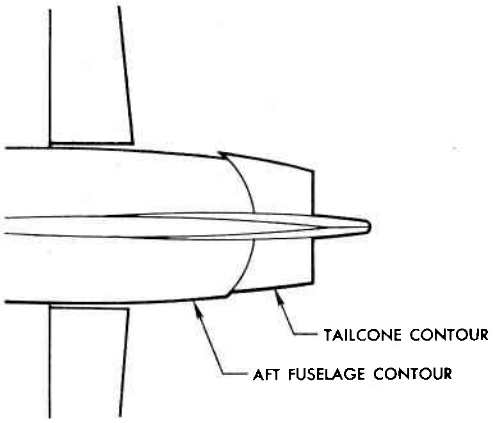


Figure 4-42. Tail Cone Skin and Rib Repair (Sheet 2 of 2)



VIEW A
(AREA OF MISMATCH)

REPAIR PROCEDURE

a. When tail cone is out of alignment with fuselage as shown in view A, shims must be added to tail cone structure as follows:

1. Drill out rivets from tail cone skin in area of misalignment.
2. Fabricate shims from 2024-T3 aluminum alloy. Taper shims as required to build up and align tail cone outer skin with fuselage contour.

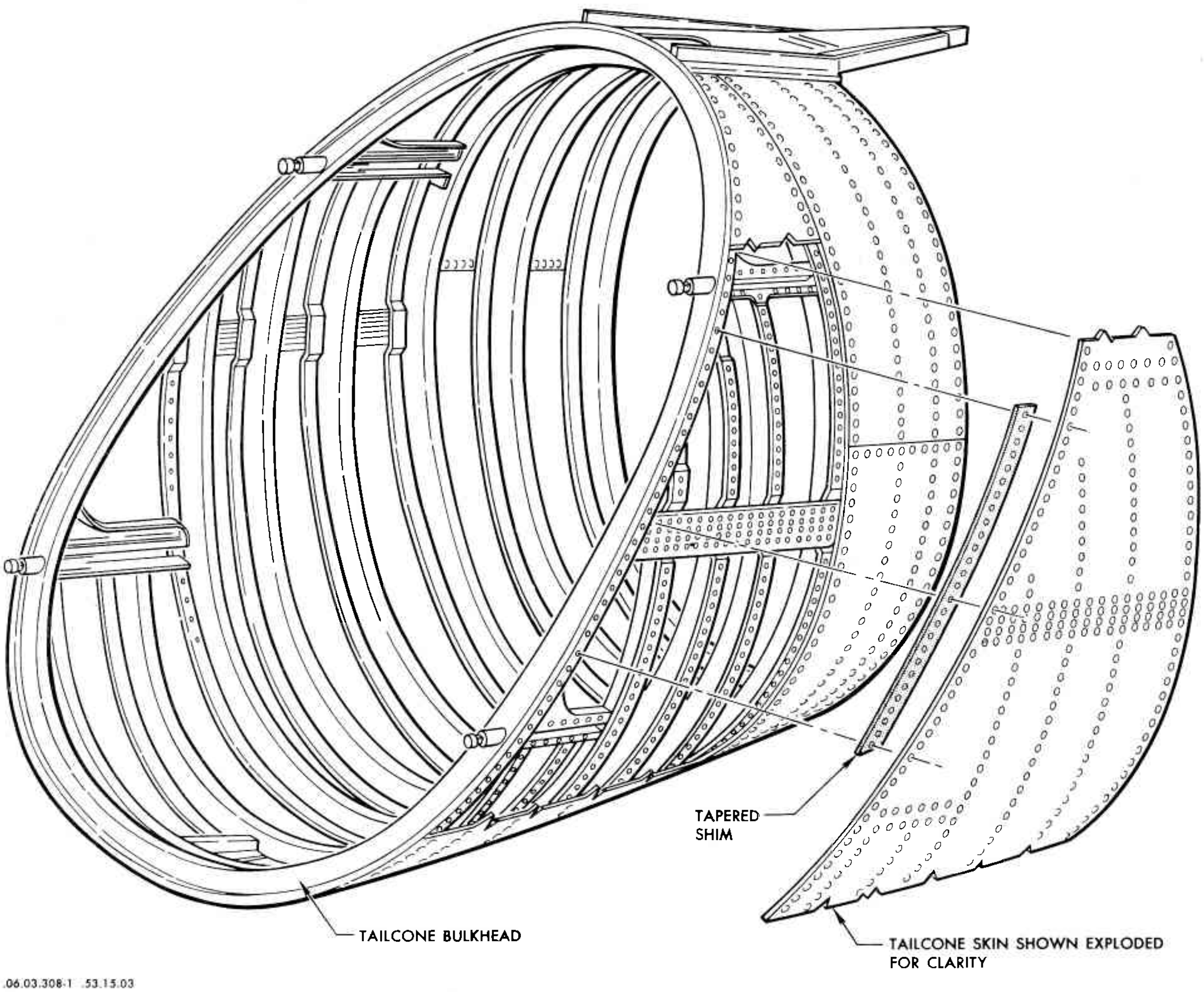
CAUTION

TO PREVENT STRETCHING SKIN, SHIMS SHOULD NOT EXCEED 0.064" IN THICKNESS.

3. Insert shims between tail cone skin and bulkhead, and pick up rivet holes through shims from existing holes in skin.

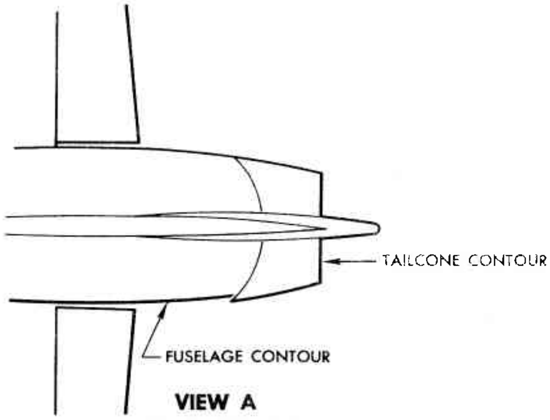
NOTE

IF ANY RIVET HOLES HAVE BEEN MISALIGNED BECAUSE THE SHIM HAS BEEN ADDED, REDRILL HOLES FOR NEXT LARGER SIZE RIVETS.



06.03.308-1 .53.15.03

Figure 4-43. Tail Cone Alignment (Sheet 1 of 2)



VIEW A
(AREA OF MISMATCH)

REPAIR PROCEDURE

a. When fuselage is out of alignment with tail cone as shown in view A, shims must be added to fuselage aft canted bulkhead as follows:

1. Drill out rivets from skin in area of misalignment.
2. Fabricate shims from 2024-T3 aluminum alloy. Taper shims as required to build up and align skin with tail cone.

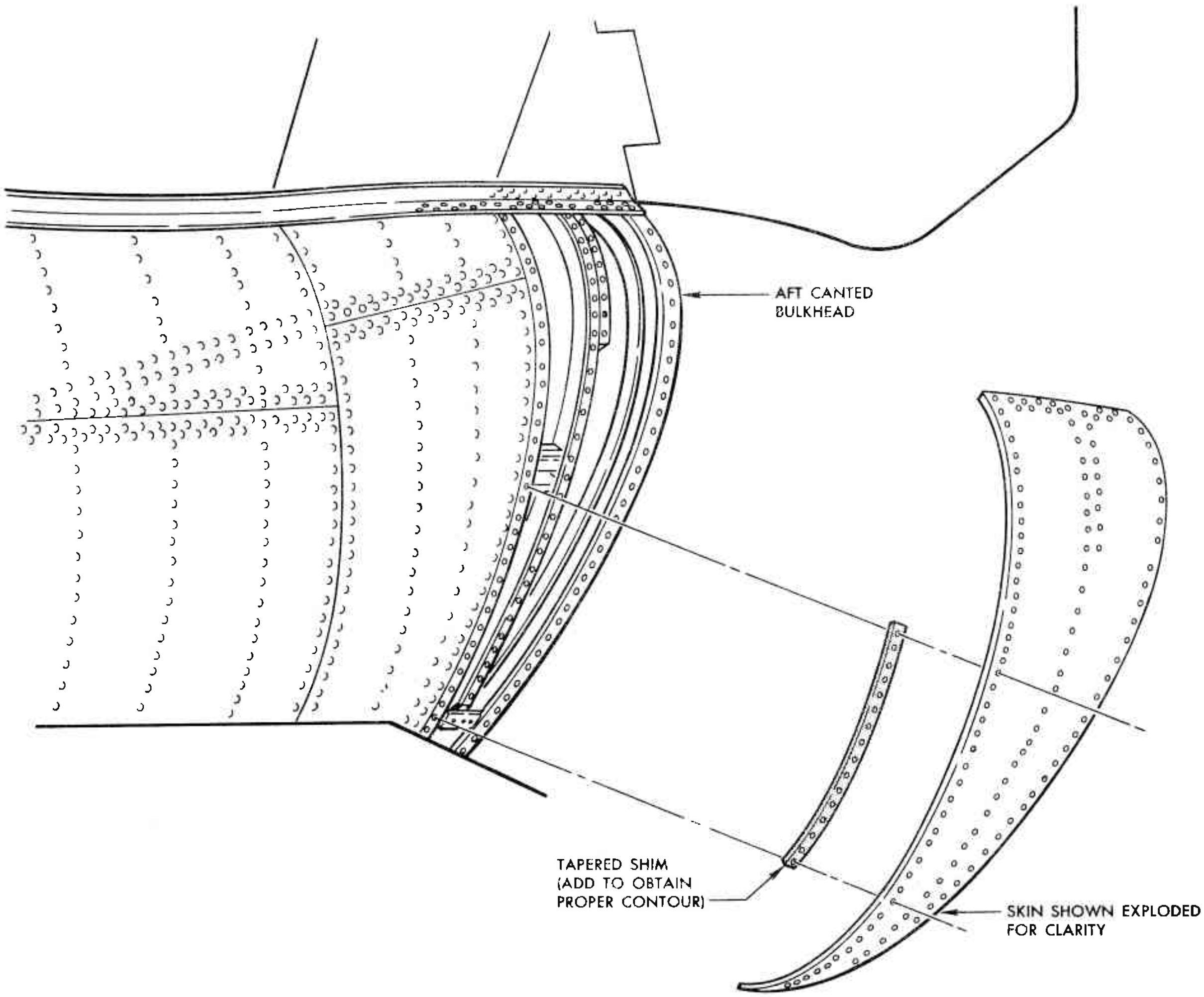
CAUTION

TO PREVENT STRETCHING SKIN, SHIMS SHOULD NOT EXCEED 0.064" IN THICKNESS.

3. Insert shims between bulkhead and skin; pick up rivet holes from existing holes in skin and canted bulkhead.

NOTE

IF SHIM THICKNESS IS GREATER THAN 0.035", USE MS20427M4C RIVETS AS REPLACEMENT RIVETS.



.06.03.308-2 .53.15.03

Figure 4-43. Tail Cone Alignment (Sheet 2 of 2)

4-53 to 4-57. Deleted.

4-58. FUSELAGE LOADING/SHIPPING STAND PROCEDURES.

4-59. SHIPPING STAND PREPARATION.

a. Loosen wood cradle support stops to allow slight side movement of the cradle during fuselage loading. (The cradles must be free to slide to either side. The cradles may have to be moved manually to position snugly to aircraft contour.)

b. Install cable assembly, Part No. 7037480-50, to shipping stand on I-beam side as shown on Drawing 7037483, sheet 1 of 2.

c. Remove anchor, Part No. 7037479-10, from stand. The anchor is located on the right side of the stand at fuselage station 472.

d. Remove NLG. Installation hardware shall be kept handy for reuse.

e. Install Adapter, Part No. 7346939-10, to the 3 NLG connecting points using same hardware (as required) removed in step 4 above. Remove cable Part No. 7037480-70, from the adapter assembly to avoid its getting in the way during fuselage loading.

4-60. FUSELAGE PREPARATION.

a. Remove all projecting tubes, lines, and other attachments that may be damaged or will interfere during fuselage loading on stand.

b. Install adapters, Part No. 7037482-10 and -30, on fuselage wing attaching slots at fuselage station 363.90 as shown on Drawing 7037483. Use the same wing bolts and nuts.

c. Install fuselage lifting adapters, Part No. 7037481-10 and -30, per T.O. 1F-106-3, figure 1-65, sheet 2 of 3.

d. The F-106 Fuselage sling, Part No. 8-96039, FSN 1730-555-4599, already installed on 15-ton crane will be connected to fuselage adapters, Part No. 7037481-10 and -30. Use aircraft T.O. lifting procedures.

4-61. FUSELAGE LOADING.

a. Raise fuselage to remove fuselage support stands.

b. Lower fuselage to an approximate height of 3 feet from the ground.

c. Place stand parallel and to the right side of the fuselage. Slide the stand under the fuselage until the left side is directly under and parallel to the fuselage belly centerline.

d. Using two 10,000 LB forklifts, one each placed approximately 3 feet from the stand ends, raise stand from its' right side to approximately 45°. (The 45° angle on the stand is reached when the lower outside I-beam flange is 65 inches from the ground. This measurement shall be taken on both ends of the stand.)

e. Secure stand to forklifts with chains to prevent stand from sliding on the ground and/or slipping from the forklifts during fuselage loading. The chains shall have a little slack to allow the stand I-beam to roll on the forklift hooks as it is raised. The chain shall have grab hooks on both ends, to allow locking chain loops to hold stand to forklift.

CAUTION

Forklift hook ends must not extend more than 8 inches beyond I-beam flange, otherwise they may damage the fuselage.

The stand must be placed such to have the wood cradle supports as close to the respective fuselage bulkheads centerlines as possible.

g. As the fuselage is lowered to the stand, it must constantly be watched for bulkhead alignment, and fuselage contour fit to the stand wood support cutouts. One each alignment operator will be placed on the rear, front, and on the left side of the stand. The front and rear alignment operators will be responsible for seeing that the fuselage is placed snugly on the wood supports. The alignment operator on the left of the stand will be responsible for the placement of fuselage bulkhead centerline as close to the center of the wood supports.

h. Coordination of the three alignment operators is essential while lowering fuselage onto the stand.

All data on pages 4-74A thru 4-74D, deleted.

(5) Cleaning supplies: clean cotton rags (lint and oil free) or clean cheese cloth, wiping tissues, solvent dispensing cans (safety type), small plastic wash bottles, etc.

d. Process.

WARNING

- MEK is hazardous. Do not inhale the vapors from this volatile material and avoid skin contact. Wear protective gloves and face protection when applying this material.
- Pasa-Jell 105 is an acidic corrosive chemical. Any spill of this material should be immediately washed away with water. Residues may be treated with sodium bicarbonate to neutralize any remaining acid.
- The adhesive materials listed in paragraph b(3),, b.(4), and b.(8) contain chemicals and volatile ingredients known to be toxic and irritating to the skin and eyes. These adhesives can cause dermatitis if skin contact occurs. Avoid skin contact with these adhesives.
- Adequate local ventilation should be provided where the volatile material discussed in paragraph f.(1) is handled on work benches. If possible this material should not be used in the layup room. The ventilation system shall provide lateral exhaust ventilation of 100 feet/minute at the work bench. The type of ventilation system shall be coordinated with bioenvironmental Engineering prior to planning and installation.
- At the end of the day or after completion of the adhesive or solvent handling operations for the day, wash your hands thoroughly and then apply skin conditioner. Remember, cleanliness is the best defense against dermatitis.
- Accomplish all work in authorized/approved area with proper protection devices to safeguard health and safety. Environmental assessment has been made of this process.

NOTE

This process is abbreviated. Shop workers are assumed to be trained in structural adhesive bonding techniques.

(1) Remove the curved skin from the door.

(2) Mechanically abrade off the old adhesive from the faying surface around the honeycomb.

(3) Determine if hydraulic oil is present in the honeycomb by placing the stripped door, honeycomb side down, on absorbent paper in an oven at 200°F. Oil present, rupture the adhesive blocking off each cell with a scribe. Soak in clean Freon and drain. Repeat soak-drain cycle with new solvent until all trace of oil is gone (as indicated by oven test).

(4) Repair dents in honeycomb with EA-934.

(5) Sand the honeycomb surface so that the curved skin will conform.

(6) Cut and fit the aluminum skin and drill 5/32" rivet holes for alignment during assembly.

(7) Prepare doubler, Z-members, and spacer strips as required for repair of the door.

(8) Degrease and Pasa-Jell treat the inside of the skin and other aluminum parts. Do not Pasa-Jell pan and honeycomb.

(9) Move the pan with honeycomb and all cleaned aluminum parts to the lay-up room.

(10) Remove roll of expandable core adhesive from the cold box. Allow to come to room temperature and cut strips to fit into the grooves around the edges of the honeycomb. Wearing cotton gloves, remove the polyethylene protector from both sides of the film and place into the grooves.

(11) Remove roll of FMS-1013 Film Adhesive from the cold box and allow to warm to room temperature. Cut piece to fit the door. Remove the green protector from one side of adhesive and press adhesive into place. Cut off excess adhesive from the edges. Remove the other green protector.

(12) Punch through the film adhesive at 5/32" holes with a scribe to allow easy placement of alignment rivets. Insert the liner strip with its layer of film adhesive if required.

(13) Place the skin on the exposed film adhesive. Align the assembly with 5/32" rivets.

(14) Tape the curved metal plate to the bond form with Mystic tape. Place the door on the form and secure with hinge pins. Place the necessary doublers, patches, Z-members, etc.

(15) Proceed with the standard vacuum pack according to paragraph f.(4).

(16) Cure in the autoclave as follows:

Apply a uniformly distributed load of 45 $\pm$ 5 psi over the facing area during the cure cycle. Heat from room temperature to 350° $\pm$ 10°F at a uniform rate in 35 $\pm$ 5 minutes. Hold at 350° $\pm$ 10°F for 60 $\pm$ 10 minutes.

(17) Allow to cool, release vacuum when the temperature of the piece reaches 180°F and remove from the autoclave. Strip off the vacuum pack. Sand off the excess epoxy. Seal off the bleed holes with EA-934.

(18) Install the necessary brackets, strips, plates, hinges, and other hardware.

(19) Treat for corrosion using Alodine. Route to inspection and paint shop.

e. Inspection or Testing:

(1) Inspect the unit after bonding for voids using the auditory "coin tapping" method.

f. References.

(1) Standard Pasa-Jell Process for Treating Aluminum, Aluminum Alloys, and Titanium Prior to Adhesive Bonding. Ref T.O. 1F-111A-3.

(2) Quality Standard for Honeycomb Core Defects. (Ref General Dynamics Specification FPS-1017 and FPS-1009D.)

(3) Standard Process for Storing, Handling and Applying Film Adhesives and Preimpregs. (Ref General Dynamics Specification FMS-1009D, and FPS-1016B).

(4) Standard Vacuum Bag Process (Ref General Dynamics Specifications FPS-1009D, FPS-1016B and T.O. 1F-111A-2-2-1).

(5) Ram Air Door Assembly.

(a) Door assembly, Part No. 8-45958-31.

(b) Outer skin, Part No. 8-45958-7.

(c) Strip, Part No. 8-45958-11, -23.

(d) Pan, Part No. 8-45958-9.

(e) Plate, Part No. 8-45958-713.

(f) Core, Part No. 8-45958-17.

(g) Hinge, Part No. 8-45961.

4-58. FUSELAGE LOADING/SHIPPING STAND PROCEDURES.

4-59. SHIPPING STAND PREPARATION.

a. Loosen wood cradle support stops to allow slight side movement of the cradle during fuselage loading. (The cradles must be free to slide to either side. The cradles may have to be moved manually to position snugly to aircraft contour.)

b. Install cable assembly, Part No. 7037480-50, to shipping stand on I-beam side as shown on Drawing 7037483, sheet 1 of 2.

c. Remove anchor, Part No. 7037479-10, from stand. The anchor is located on the right side of the stand at fuselage station 472.

d. Remove NLG. Installation hardware shall be kept handy for reuse.

e. Install Adapter, Part No. 7346939-10, to the 3 NLG connecting points using same hardware (as required) removed in step 4 above. Remove cable Part No. 7037480-70, from the adapter assembly to avoid its getting in the way during fuselage loading.

4-60. FUSELAGE PREPARATION.

a. Remove all projecting tubes, lines, and other attachments that may be damaged or will interfere during fuselage loading on stand.

b. Install adapters, Part No. 7037482-10 and -30, on fuselage wing attaching slots at fuselage station 363.90 as shown on Drawing 7037483. Use the same wing bolts and nuts.

c. Install fuselage lifting adapters, Part No. 7037481-10 and -30, per T.O. 1F-106-3, figure 1-65, sheet 2 of 3.

d. The F-106 Fuselage sling, Part No. 8-96039, FSN 1730-555-4599, already installed on 15-ton crane will be connected to fuselage adapters, Part No. 7037481-10 and -30. Use aircraft T.O. lifting procedures.

4-61. FUSELAGE LOADING.

a. Raise fuselage to remove fuselage support stands.

b. Lower fuselage to an approximate height of 3 feet from the ground.

c. Place stand parallel and to the right side of the fuselage. Slide the stand under the fuselage until the left side is directly under and parallel to the fuselage belly centerline.

d. Using two 10,000 LB forklifts, one each placed approximately 3 feet from the stand ends, raise stand from its' right side to approximately 45°. (The 45° angle on the stand is reached when the lower outside I-beam flange is 65 inches from the ground. This measurement shall be taken on both ends of the stand.)

e. Secure stand to forklifts with chains to prevent stand from sliding on the ground and/or slipping from the forklifts during fuselage loading. The chains shall have a little slack to allow the stand I-beam to roll on the forklift hooks as it is raised. The chain shall have grab hooks on both ends, to allow locking chain loops to hold stand to forklift.

CAUTION

Forklift hook ends must not extend more than 8 inches beyond I-beam flange, otherwise they may damage the fuselage.

The stand must be placed such to have the wood cradle supports as close to the respective fuselage bulkheads centerlines as possible.

g. As the fuselage is lowered to the stand, it must constantly be watched for bulkhead alignment, and fuselage contour fit to the stand wood support cutouts. One each alignment operator will be placed on the rear, front, and on the left side of the stand. The front and rear alignment operators will be responsible for seeing that the fuselage is placed snugly on the wood supports. The alignment operator on the left of the stand will be responsible for the placement of fuselage bulkhead centerline as close to the center of the wood supports.

h. Coordination of the three alignment operators is essential while lowering fuselage onto the stand.

i. The wood supports shall be watched very closely so that they may be shifted to either side as required.

j. Once the fuselage mates the stand, connect two cables, Part No. 7037480-30, to adapter, Part No. 7037482-10, and to the two lugs welded at a 45° angle on the stand left side I-beam. (Reference drawing 7037483 sheet 1 of 2.) Leave cable with one inch slack.

k. Connect 2 each cable assembly, Part No. 7037480-50, to adapter, Part No. 7037482-30.

l. Connect 2 each cable assembly, Part No. 7037480-10, between adapter, Part No. 7037481-10, and I-beam lug at fuselage station 472. Leave cable with one inch slack.

m. The fuselage will then be lowered slightly loading fuselage on the cradle. The stand will then slowly be lowered by the forklifts. As the stand is lowered, the crane operator will coordinate lowering of the fuselage and at the same time adding a little side load by moving the crane forward to help the fuselage roll with the stand and to keep the fuselage sling legs from contacting the fuselage.

NOTE

The crane forward movement must be controlled to prevent aircraft sling from coming in contact with the fuselage.

n. Remove forklifts when stand sits flat on the ground.

o. Release (slowly) all tension from the fuselage sling.

p. Remove fuselage sling.

q. Check tilt indicator on adapter, Part No. 7037482-10, located at fuselage station 363 left side. The tilt should be very close to 45°.

r. Install anchor, Part No. 7037479-10, to adapter, Part No. 7037481-30 in accordance with Drawing 7037483, sheet 3 of 3. Mount adapter, Part No. 7037479-10, to I-beam in accordance with Drawing 7037483, sheet 1 of 2. Washers, Part No. MS122044, will be used as shims as required.

s. Install cable Part No. 7037480-70 to adapter Part No. 7346939-10. To connect cable to stand drill a 3/4 diameter hole to stand I-beam or cross channel. The hole shall be located such to keep the cable in-line with the adapter and to allow installation with a little room for cable final adjustment.

t. Remove tail hook at station 520. Save hardware. Install adapter, Part No. 7346952-10, on tail hook bolt. The adapter ears shall be positioned to face aft and to project away from fuselage. Add cable assembly, Part No. 7037480-90, to the adapter. Connect cables to the stand by drilling 3/4 diameter holes to the I-beam outer top flange. Angle of holes shall be as required. The location of the holes shall be such to keep the cables as close aligned to adapter ear angle and to allow connection to the I-beam and have slack for final adjustment.

u. Tighten all cables tautly.

v. Tighten all nuts.

w. Tighten all wood cradle support stops.

i. The wood supports shall be watched very closely so that they may be shifted to either side as required.

j. Once the fuselage mates the stand, connect two cables, Part No. 7037480-30, to adapter, Part No. 7037482-10, and to the two lugs welded at a 45° angle on the stand left side I-beam. (Reference drawing 7037483 sheet 1 of 2.) Leave cable with one inch slack.

k. Connect 2 each cable assembly, Part No. 7037480-50, to adapter, Part No. 7037482-30.

l. Connect 2 each cable assembly, Part No. 7037480-10, between adapter, Part No. 7037481-10, and I-beam lug at fuselage station 472. Leave cable with one inch slack.

m. The fuselage will then be lowered slightly loading fuselage on the cradle. The stand will then slowly be lowered by the forklifts. As the stand is lowered, the crane operator will coordinate lowering of the fuselage and at the same time adding a little side load by moving the crane forward to help the fuselage roll with the stand and to keep the fuselage sling legs from contacting the fuselage.

NOTE

The crane forward movement must be controlled to prevent aircraft sling from coming in contact with the fuselage.

n. Remove forklifts when stand sits flat on the ground.

o. Release (slowly) all tension from the fuselage sling.

p. Remove fuselage sling.

q. Check tilt indicator on adapter, Part No. 7037482-10, located at fuselage station 363 left side. The tilt should be very close to 45°.

r. Install anchor, Part No. 7037479-10, to adapter, Part No. 7037481-30 in accordance with Drawing 7037483, sheet 3 of 3. Mount adapter, Part No. 7037479-10, to I-beam in accordance with Drawing 7037483, sheet 1 of 2. Washers, Part No. MS122044, will be used as shims as required.

s. Install cable Part No. 7037480-70 to adapter Part No. 7346939-10. To connect cable to stand drill a 3/4 diameter hole to stand I-beam or cross channel. The hole shall be located such to keep the cable in-line with the adapter and to allow installation with a little room for cable final adjustment.

t. Remove tail hook at station 520. Save hardware. Install adapter, Part No. 7346952-10, on tail hook bolt. The adapter ears shall be positioned to face aft and to project away from fuselage. Add cable assembly, Part No. 7037480-90, to the adapter. Connect cables to the stand by drilling 3/4 diameter holes to the I-beam outer top flange. Angle of holes shall be as required. The location of the holes shall be such to keep the cables as close aligned to adapter ear angle and to allow connection to the I-beam and have slack for final adjustment.

u. Tighten all cables tautly.

v. Tighten all nuts.

w. Tighten all wood cradle support stops.

KEPT FOR REF ONLY WLT 30 OCT 84

All data on pages 4-76 thru 4-79, including figures 4-44 thru 4-46, deleted.

TABLE 4-I

Negligible Damage Limits — Fuselage Group

COMPONENT	TYPE AND CLASS OF DAMAGE ALLOWED AFTER REWORK					REMARKS
	Scratch	Nick	Dent	Hole	Crack	
FUSELAGE PLATING						
A-110AT—Titanium	I	I	I	*	*	
AZ31A—Magnesium	I	I	I	*	*	
2024-T6 Clad	I	I	I	*	*	
2024-T81 Bare	I	I	I	*	*	
7178-T6 Bare	I	I	I	*	*	
MAIN BULKHEADS—STATIONS 40.89 TO 711.40 FOR F-106A						
Station 40.89 Channel	II	II	*	I	I	
Station 59.31 Web	III	III	III	II	II	
Stiffeners	III	III	I	III	III	
Station 102.00 Web	II	III	III	II	II	
Zees	II	II	I	I	I	
Angles	II	II	*	I	I	
Station 162.90 Doublers	II	III	III	II	II	
Webs	II	III	III	II	II	
Zees	II	II	I	I	I	
Angles	II	II	*	I	I	
Channels	II	II	I	I	I	
Station 216.50 Webs	II	III	III	II	II	
Doublers	II	III	III	II	II	
Tees	II	II	*	I	I	
Zees	II	II	I	I	I	
Angles	II	II	*	I	I	
Station 253.00 Web	II	III	III	II	II	
Angles	II	II	*	I	I	
Stiffeners	III	III	I	III	III	
Station 316.00 Webs	II	II	III	II	II	
Doubler	II	II	III	II	II	
Rails	I	I	I	I	I	
Inner Strap	I	I	I	I	I	
Angles	II	II	I	I	I	
Stiffeners	III	III	I	III	III	

*Component must be repaired or replaced.

*U.S. GOVERNMENT PRINTING OFFICE: 1982-669-062/4334 Region 7.

TOOL FUNCTION

To mate the fuselage central barrel section to the fuselage nose section and fuselage aft barrel section.

OPTICAL TOOLING EQUIPMENT REQUIRED

a. One Jig Transit
b. One Sight Level.
c. Two Optical Squares.
d. Two M8J 172 TOAC.
e. Two Alignment Telescopes.
f. Two 2¼-inch diameter targets with light source.
g. Four 3½-inch diameter Spherical Balls.
h. One 1½-inch diameter Plastic Target on Pin.
i. Two 1½-inch diameter Plastic Targets.
j. One precision white face scale (Keuffel & Esser No. 7092 or equivalent).
k. One 12-inch Master Spirit Level.
l. One set of 30-foot inside Micrometers.

l. One set of 30-inch inside Micrometers.

MASTERING PROCEDURE

a. Level fixture and establish alternate line of sight as follows:
   (1) Set up sight level to one side of fixture.
   (2) Refer to Section I for leveling procedures using the sight level.
   (3) Establish a horizontal plane between points A, B, C and D so as to establish the alternate line of sight (ALOS) at WL -48.00 and BL 465.00, left-and right-hand sides.

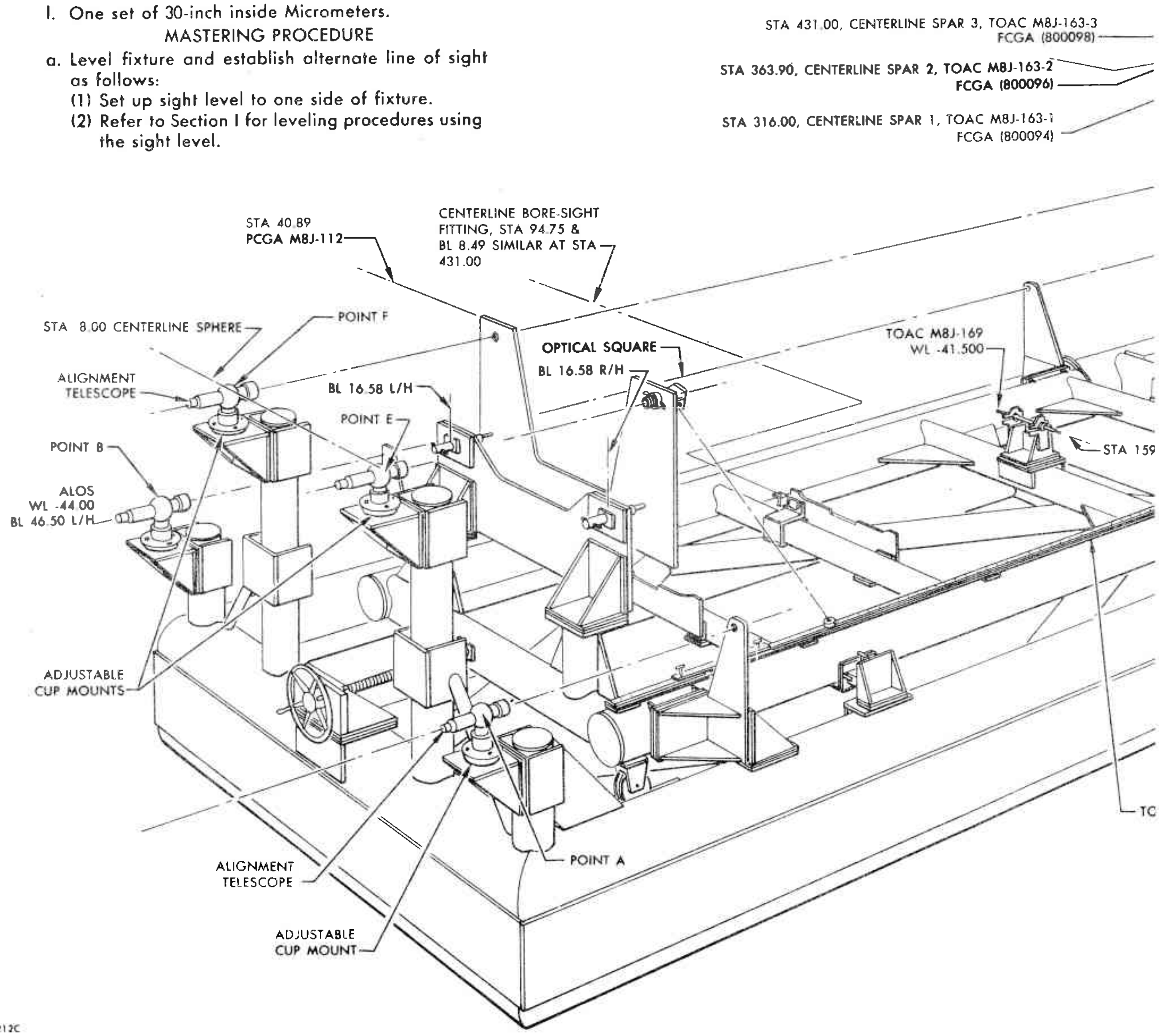
NOTE

THE ALOS WILL BE USED TO MAINTAIN FIXTURE ALIGNMENT.

BASIC LINE OF SIGHT

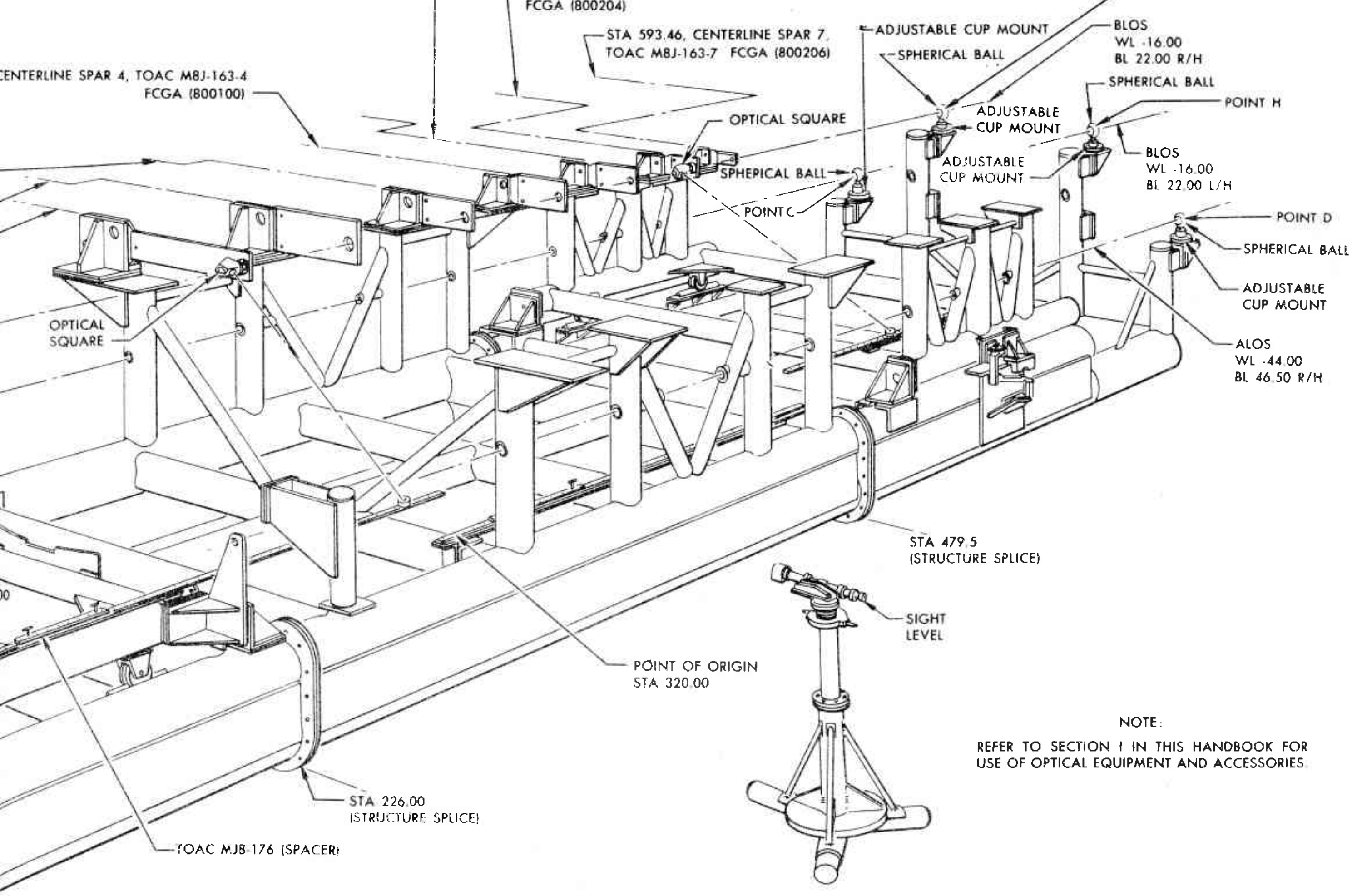
b. Establish a basic line of sight (BLOS) as follows:
   (1) Install adjustable cup mounts at points E, F, G and H.
   (2) Install alignment telescopes at points E and F with spherical balls.
   (3) Install targets with light source in spherical balls at points G and H.

Center l
on static
measure
c. Use
spre
poin
d. Posi
plat
e. Use
brin
f. Usin
stati



ne of spherical balls at points E and F to be
n 8.00. This establishes the basis for all linear
ment.
acility gage M8J-112, to control 44.000 inch
d between points E and F, and between
s G and H.
on tooling accessory M8J 163-1 and angle
at station 316.00.
acility gage, 800094, with optical square to
tooling accessory, M8J 163-1, into line of sight.
the 30-foot inside micrometer, measure from
n 8.0 to establish station 316.00.

LEGEND	
TOAC	TOOLING ACCESSORY
FCGA	FACILITY GAGE
TOGA	TOOLING GAGE
ALOS	ALTERNATE LINE OF SIGHT
BLOS	BASIC LINE OF SIGHT



g. Using the 12-inch master spirit level, check the equipment at station 316.000 for rotation.

h. With optical square attached to locating surface of facility gage, 800206, mount facility gage on forward surface of tooling accessory, M8J 163-7.

(1) Mount tooling accessory M8J 163-7, at station 593.46.

(2) Bring optical square into the basic line of sight and check for rotation with the 12-inch master spirit level.

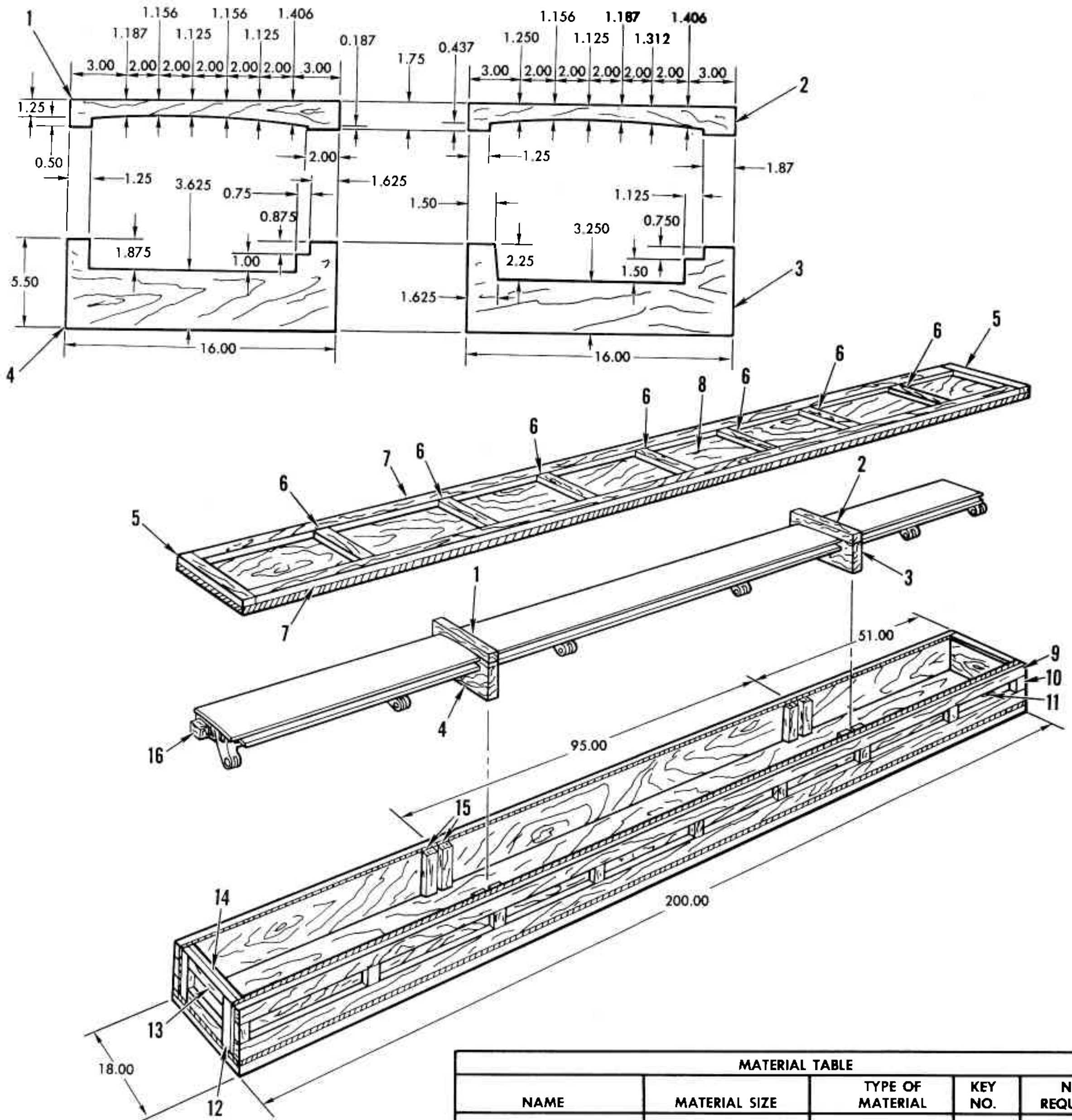
(3) Position facility gage for station location with optical square directed at tooling bar. Use white face scale for accurate measurement.

i. Repeat procedure outlined in step "h.(3)" to position tooling accessories and facility gages at stations 363.900, 431.000, 472.000, 520.000 and 556.750.

Recheck station 316.000.

NOTE
ACCOMPLISHMENT OF STEPS "d" THROUGH "g" ESTABLISHES THE WING TO FUSELAGE ATTACH POINTS.

Figure 4-44. Master Tooling — Fuselage Mating



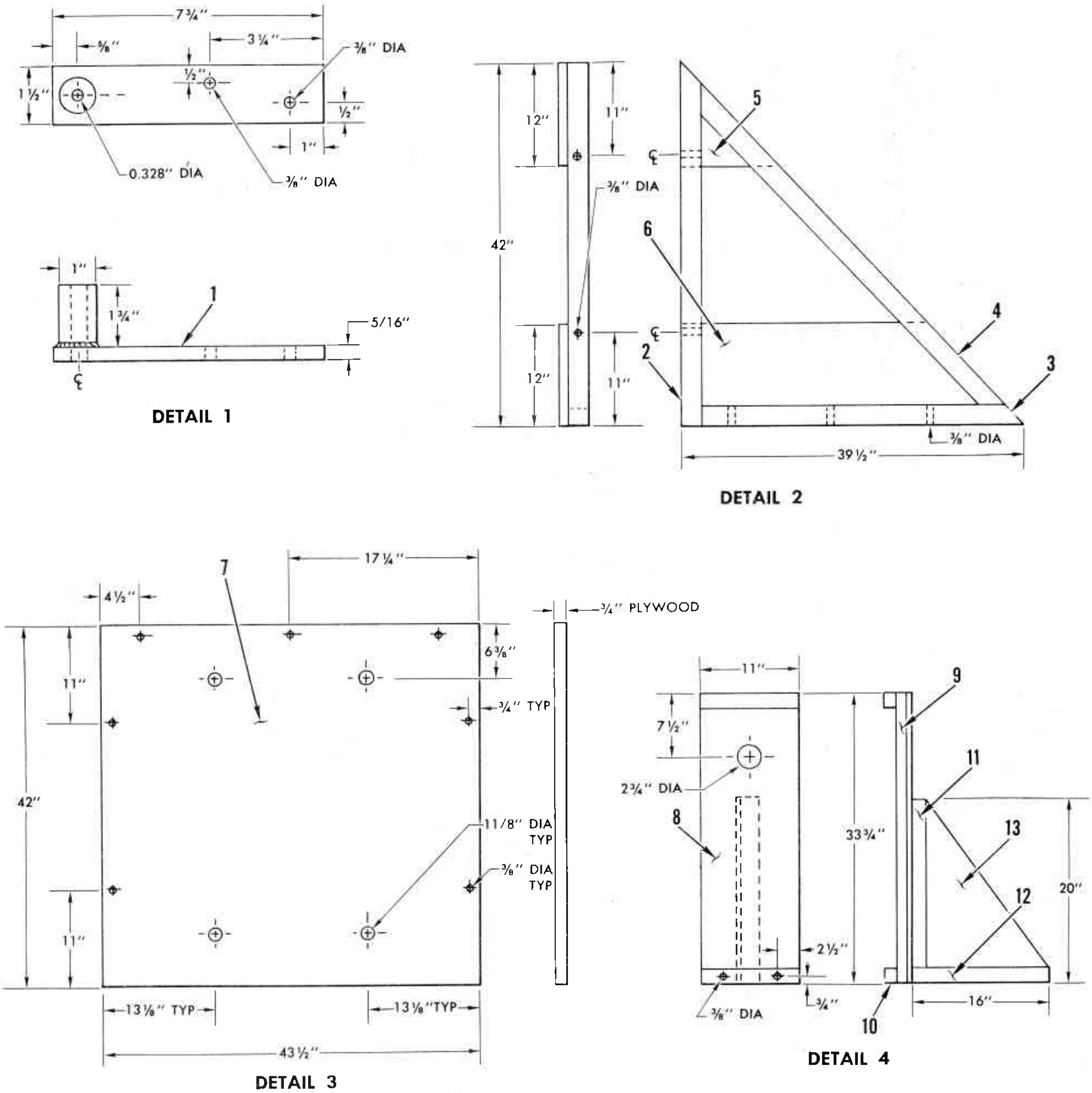
NOTE:

PAD ALL POINTS OF CONTACT OF PART AND CRATE WITH PAPER, SPECIFICATION MIL-B-121, GRADE A, AND DUNNAGE, FEDERAL SPECIFICATION PPP-C-843, TYPE III.

MATERIAL TABLE				
NAME	MATERIAL SIZE	TYPE OF MATERIAL	KEY NO.	NO. REQUIRED
HOLDING FIXTURE	2x1¾x16	BOUND FIBER	1	1
HOLDING FIXTURE	2x1¾x16	BOUND FIBER	2	1
HOLDING FIXTURE	2x5½x16	BOUND FIBER	3	1
HOLDING FIXTURE	2x5½x16	BOUND FIBER	4	1
CROSS CLEAT	¾x1¾x18	PLYWOOD	5	4
FILLER CLEAT	¾x1¾x14½	LUMBER	6	14
LONGERON CLEAT	¾x1¾x196½	LUMBER	7	4
PANEL	¼x18x200	PLYWOOD	8	2
LONGERON CLEAT	¾x1¾x200	LUMBER	9	4
VERTICAL CLEAT	1⅝x1¾x3¾	LUMBER	10	18
PANEL	¼x7¼x200	PLYWOOD	11	2
VERTICAL CLEAT	¾x1¾x7¼	LUMBER	12	4
PANEL	¼x7¼x200	PLYWOOD	13	2
HORIZONTAL CLEAT	¾x1¾x12½	LUMBER	14	4
BLOCK	1⅝x2½x5½	LUMBER	15	2
CLEAT	¾x1¾x7¼	LUMBER	16	8

06.03.254B

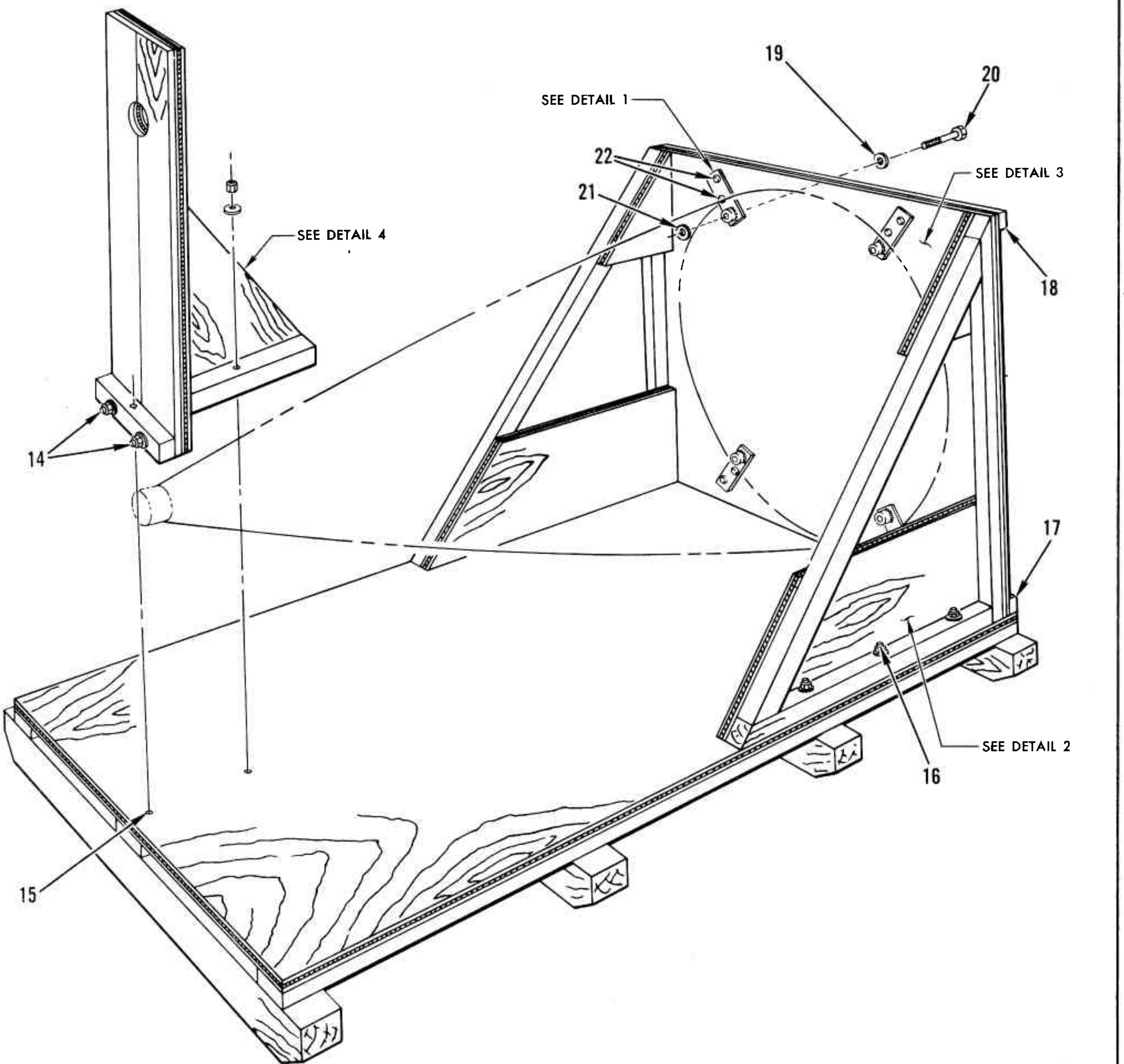
Figure 4-45. Packing and Crating — Missile Bay Door



MATERIALS				
NAME	DESCRIPTION	TYPE	PART NO.	NO. REQUIRED
MOUNTING BRACKET		STEEL	1	4
VERTICAL SUPPORT MEMBER	1-⅝" x 1-⅝" x 42"	LUMBER	2	2
HORIZONTAL SUPPORT MEMBER	1-⅝" x 1-⅝" x 39-½"	LUMBER	3	3
DIAGONAL SUPPORT MEMBER	1-⅝" x 1-⅝" x 54"	LUMBER	4	2
REINFORCING PLATE	⅜" x 12" x 12"	PLYWOOD	5	2
REINFORCING PLATE	⅜" x 12" x 39-½"	PLYWOOD	6	2
MOUNTING PLATE	¾" x 42" x 43½"	PLYWOOD	7	1
FRONT SUPPORT MEMBER	¾" x 5-½" x 33-¾"	LUMBER	8	2
FRONT SUPPORT PLATE	5/16" x 11" x 33-¾"	PLYWOOD	9	2
FRONT SUPPORT CLEAT	1-⅝" x 1-⅝" x 11"	LUMBER	10	2
VERTICAL MEMBER	1-⅝" x 1-⅝" x 18-¾"	LUMBER	11	1
HORIZONTAL MEMBER	1-⅝" x 1-⅝" x 16"	LUMBER	12	1
REINFORCING PLATE	5/16" x 16" x 20"	PLYWOOD	13	1

06.03.280-1 .53.01.04

Figure 4-46. Packing and Crating — Radome (Sheet 1 of 3)



MATERIALS				
NAME	DESCRIPTION	TYPE	PART NO.	NO. REQUIRED
CARRIAGE BOLT	3/8" x 3"	STEEL	14	9
CARRIAGE BOLT	3/8" x 6"	STEEL	15	6
CARRIAGE BOLT	3/8" x 4-1/4"	STEEL	16	5
LOAD BEARING MEMBER	1-5/8" x 1-5/8" x 43-1/2"	LUMBER	17	1
STIFFENER CLEAT	1-5/8" x 1-5/8" x 43-1/2"	LUMBER	18	1
WASHER	1" OD x 11/32" ID x 1/16"	STEEL	19	4
AIRCRAFT BOLT	5/16" x 24"	STEEL	20	4
WASHER	1" OD x 11/32" ID x 1/16"	STEEL	21	4
CARRIAGE BOLT	3/8" x 1-3/4"	STEEL	22	8

06.03.280-2A .53.01.04

Figure 4-46. Packing and Crating — Radome (Sheet 2 of 3)

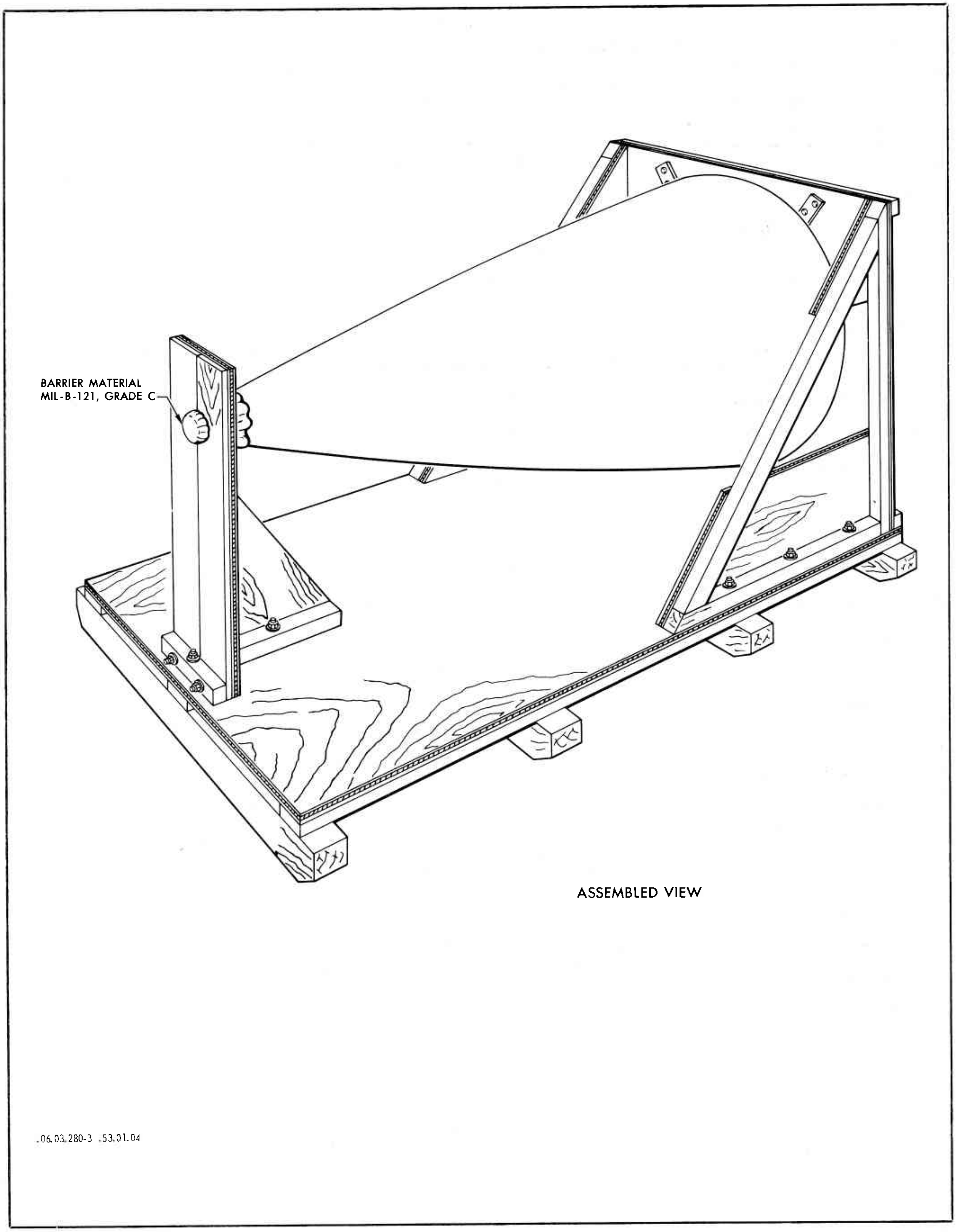


Figure 4-46. Packing and Crating — Radome (Sheet 3 of 3)

TABLE 4-I

Negligible Damage Limits — Fuselage Group

COMPONENT	TYPE AND CLASS OF DAMAGE ALLOWED AFTER REWORK					REMARKS
	Scratch	Nick	Dent	Hole	Crack	
FUSELAGE PLATING						
A-110AT—Titanium	I	I	I	*	*	
AZ31A—Magnesium	I	I	I	*	*	
2024-T6 Clad	I	I	I	*	*	
2024-T81 Bare	I	I	I	*	*	
7178-T6 Bare	I	I	I	*	*	
MAIN BULKHEADS—STATIONS 40.89 TO 711.40 FOR F-106A						
Station 40.89 Channel	II	II	*	I	I	
Station 59.31 Web	III	III	III	II	II	
Stiffeners	III	III	I	III	III	
Station 102.00 Web	II	III	III	II	II	
Zees	II	II	I	I	I	
Angles	II	II	*	I	I	
Station 162.90 Doublers	II	III	III	II	II	
Webs	II	III	III	II	II	
Zees	II	II	I	I	I	
Angles	II	II	*	I	I	
Channels	II	II	I	I	I	
Station 216.50 Webs	II	III	III	II	II	
Doublers	II	III	III	II	II	
Tees	II	II	*	I	I	
Zees	II	II	I	I	I	
Angles	II	II	*	I	I	
Station 253.00 Web	II	III	III	II	II	
Angles	II	II	*	I	I	
Stiffeners	III	III	I	III	III	
Station 316.00 Webs	II	II	III	II	II	
Doubler	II	II	III	II	II	
Rails	I	I	I	I	I	
Inner Strap	I	I	I	I	I	
Angles	II	II	I	I	I	
Stiffeners	III	III	I	III	III	

*Component must be repaired or replaced.

TABLE 4-I

Negligible Damage Limits — Fuselage Group (Cont)

COMPONENT	TYPE AND CLASS OF DAMAGE ALLOWED AFTER REWORK					REMARKS
	Scratch	Nick	Dent	Hole	Crack	
MAIN BULKHEADS—STATIONS 40.89 TO 711.40 FOR F-106A (cont)						
Channels	II	II	I	I	I	
Station 365.90 Webs	II	II	III	II	II	
Rails	I	I	I	I	I	
Angles	II	II	I	I	I	
Stiffeners	III	III	I	III	III	
Channels	II	II	I	I	I	
Station 412.00 Webs	II	II	III	II	II	
Doublers	II	II	III	II	II	
Inner Strap	I	I	*	I	I	
Angles	I	I	*	I	I	
Zees	I	I	I	I	I	
Station 431.00 and 472.00 Webs	II	II	III	II	II	See Figure 9-8
Splice Angles	I	I	*	*	I	
Splice Strap	I	I	*	*	I	
Intercostal	II	II	II	II	II	
Station 520.00 Splice Angles	I	I Refer to an Aeronautical Structural Engineer				
Plates	I	I	I	I	I	
Straps	I	I	*	I	I	
Station 593.46 Webs	II	II	III	II	II	
Rails	I	I	*	I	I	
Stiffeners	III	III	I	III	III	
Straps	I	I	*	I	I	
Station 614.80 Rails	I	I	*	I	I	
Splice Plates	I	I	*	I	I	
Webs	II	II	III	II	II	
Splice Angles	I	I	*	I	I	
Stiffeners	III	III	I	III	III	
Station 645.38 Rails	I	I	*	I	I	
Splice Straps	I	I	*	I	I	
Splice Plates	I	I	*	I	I	

*Component must be repaired or replaced.

TABLE 4-I

Negligible Damage Limits — Fuselage Group (Cont)

COMPONENT	TYPE AND CLASS OF DAMAGE ALLOWED AFTER REWORK					REMARKS
	Scratch	Nick	Dent	Hole	Crack	
MAIN BULKHEADS—STATIONS 40.89 TO 711.40 FOR F-106A (cont)						
Splice Angles	I	I	*	I	I	
Webs	II	II	III	II	II	
Stiffeners	III	III	I	III	III	
Station 672.38 Rails	I	I	*	I	I	
Splice Straps	I	I	*	I	I	
Splice Plates	I	I	*	I	I	
Splice Angles	I	I	*	I	I	
Webs	II	II	III	II	II	
Stiffeners	III	III	I	III	III	
Station 672.90 Rails	I	I	*	I	I	
Splice Angles	II	II	*	II	II	
Straps	II	II	*	II	II	
FRAMES AND BULKHEADS—STATIONS 169.00 TO 316.00 FOR F-106B						
Bulkheads Station 169.00 280.50 and 316.00						
Channels	II	II	*	II	II	
Webs	II	II	III	II	II	
Doublers	II	II	III	II	II	
Angles	II	II	*	II	II	
Zees	II	II	I	II	II	
Tees	II	II	*	II	II	
Frames Station 216.50 to 308.50						
Angles	II	II	*	II	II	
Frame	I	I	*	I	I	
Channels	II	II	I	II	II	
Zee	II	II	I	II	II	
BELTFRAMES—STATIONS 102.00 TO 711.40						
Station 102.00 to 357.10						
Angles	II	II	*	I	I	
Stiffeners	III	III	I	III	III	
Channels	II	II	I	I	I	
Station 357.20 to 464.20 Webs	II	II	III	II	II	

*Component must be repaired or replaced.

TABLE 4-I

Negligible Damage Limits — Fuselage Group (Cont)

COMPONENT	TYPE AND CLASS OF DAMAGE ALLOWED AFTER REWORK					REMARKS
	Scratch	Nick	Dent	Hole	Crack	
BELTFRAMES—STATIONS 102.00 TO 711.40 (cont)						
Angles	II	II	*	I	I	
Zees	II	II	I	II	II	
Channels	II	II	I	II	II	
Station 464.20 to 672.90 Webs	II	II	III	II	II	
Angles	II	II	*	I	I	
Stiffeners	III	III	I	III	III	
Tees	II	II	I	II	II	
Straps	I	I	*	I	I	
Channels	II	II	I	II	II	
Station 672.90 to 711.40 Zees	II	II	I	II	II	
Angles	II	II	*	I	I	
Doublers	II	III	III	II	II	
MAIN LONGERONS						
Station 40.89 to 216.50 Upper Longerons	I	I	*	I	I	
Lower Longerons	I	I	*	I	I	
Station 216.50 to 556.75 Upper Longerons	I	I	*	I	I	
Station 450.00 to 657.00 Lower Longeron	I	I	*	I	I	See Figure 9-7
Station 216.50 to 412.00 Missile Bay Longerons.	I	I	*	I	I	
PILOTS COMPARTMENT STRUCTURE FOR F-106A AND F-106B						
Panel Frames	I	I	*	II	II	
Floor Webs	II	II	III	II	II	
Floor Zees	II	II	I	II	II	
Floor Tees	II	II	I	II	II	
Floor Channels	II	II	I	II	II	
MAIN AIR INDUCTION DUCTS FOR F-106A AND F-106B						
Frames	I	I	I	I	*	
Leading Edge	I	I Refer to an Aeronautical Structural Engineer				
Inner Skin	I		*	*	*	See Figures 4-18A & 4-18B
Variable Ramp Slip Joint Adapters	II (a)	I (a)	*	*	*	

*Component must be repaired or replaced. (a) Refinish by Touch-up Plating

TABLE 4-I

Negligible Damage Limits — Fuselage Group (Cont)

COMPONENT	TYPE AND CLASS OF DAMAGE ALLOWED AFTER REWORK					REMARKS
	Scratch	Nick	Dent	Hole	Crack	
RADAR COMPARTMENT DOORS FOR F-106A AND F-106B						
Skin	II	II	*	*	*	
Pan	II	II	III	II	II	
Frames	II	II	II	II	II	
Hinge Fittings	II	II	Refer to an Aeronautical Structural Engineer			
MISSILE BAY AREA STRUCTURES						
Webs, Roof & Sides	II	II	III	II	II	
Stiffeners, Roof & Sides	III	III	I	III	III	
Drag Arms (Except 8-57803)	III	III	Refer to an Aeronautical Structural Engineer			See Note 1
MISSILE BAY DOORS STRUCTURES						
Inner Skins	II	II	III	II	II	
Outer Skins	II	II	*	*	*	
Doublers	II	II	II	II	II	
Ribs	II	II	I	II	II	
Stiffeners	III	III	I	III	III	
Hinge Fittings	I	I	Refer to an Aeronautical Structural Engineer			
ENGINE COOLING AIR SCROLL—STATION 431.00 TO 445.00						
Skins, Perforated	II	II	*	*	*	
Attachment Flanges	I	I	Refer to an Aeronautical Structural Engineer			
Formers	II	II	I	I	I	
Stiffeners	III	III	I	II	II	
DRAG CHUTE HOUSING STRUCTURE						
Skin, Ti. AMS 4908	II	II	*	II	II	
Zees, Ti. AMS 4908	II	II	I	II	II	
Angles, Ti. AMS 4908	II	II	*	II	II	
Machined Casting Magnesium	I	I	Refer to an Aeronautical Structural Engineer			

*Component must be repaired or replaced.

TABLE 4-I

Negligible Damage Limits — Fuselage Group (Cont)

COMPONENT	TYPE AND CLASS OF DAMAGE ALLOWED AFTER REWORK					REMARKS
	Scratch	Nick	Dent	Hole	Crack	
TAIL CONE STRUCTURE						
Skin	III	III	*	*	*	
Beltframes	III	III	III	II	II	
Splice Angles	III	III	II	I	I	
AERIAL REFUELING SEALING MATERIALS						
Skin	II	II	*	*	*	
Formers	II	II	I	*	*	
Pan	III	III	III	*	I	Seal Cracks with putty, Specification MIL-P-20628.
*Component must be repaired or replaced.						

NOTES:
1. Location or edge magine restrictions not required for corrosion clean-out (REF. FIG. 1-17). Damage to 8-57803 refer to Aeronautical Structural Engineer.

Section V

LANDING GEAR

5-1. NOSE AND MAIN LANDING GEAR.

5-2. The F-106A and F-106B airplanes are equipped with hydraulically operated retractable tricycle landing gear consisting of two main landing gear assemblies and one steerable nose landing gear assembly, as shown on figure 5-1. When airborne, the main landing gear assemblies retract inboard and up into the wheel wells in the wing and fuselage area. In the retracted position, each main landing gear assembly is housed in its respective wheel well by a hydraulically actuated fuselage door and wing-mounted fairing actuated by arms attached to the gear strut. The right and left main landing gear assemblies are interchangeable, except for the wing fairing plates, which are hand trimmed to fit each individual opening. The nose landing gear assembly retracts forward and up into the fuselage nose wheel well, and when retracted is housed in its wheel well by the nose landing gear door.

5-3. DESCRIPTION OF LANDING GEAR AND COMPONENTS.

5-4. Nose Landing Gear.

5-5. The nose landing gear assembly shown on figure 5-2 consists of a shock strut assembly, steerable dual nose wheels mounted on a single axle, a steer-damp unit, and a gear retraction mechanism. The shock strut is supported in trunnion fittings riveted to the aircraft structure in the nose wheel well. The shock strut attaches to these trunnion fittings by means of a pivot beam integral with the strut outer cylinder housing. The outer cylinder housing and integral pivot beam is a 7075-T6 aluminum alloy forging with attached steel trunnion pins. The piston assembly is made of SAE 4340 heat-treated steel and is integral with the single axle. The steering collar assembly keys to the lower end of the outer cylinder housing and turns in a set of upper and lower bushings to permit the hydraulically powered steer-damp unit operating lever and link assembly to turn the dual nose wheels. The drag brace is a 7075-T6 aluminum alloy forging; it connects at its upper end to the gear retracting mechanism and at its lower end to the fixed truss on the forward side of the strut.

5-6. Nose Landing Gear Door.

5-7. The nose landing gear door consists of a rectangular outer frame longitudinally bisected by a central beam. Four stringers parallel the central beam, two through the upper half of the door and two through the lower half of the door. The stringers are spaced equally between the door outer frame and the central beam. Eight cross frame members are perpendicular to the central beam and stringers. The cross frame members are attached to the outer frame, central beam, and stringers with clips and rivets. The entire door structure is covered by an inner and outer skin. The nose landing gear door mechanism consists of a bellcrank assembly and door link operated by a hydraulic actuating cylinder. The door mechanism has no locking action since the "door-closed" locks are integral parts of the actuating cylinder. The nose landing gear door structure and plating details are shown on figure 5-3.

5-8. Main Landing Gear.

5-9. The main landing gear components are shown on figure 5-4. Each of the two main landing gear assemblies consists of a shock strut with integral axle, an actuating cylinder, a side brace, a scissors assembly or torque arms, and two drag braces. The shock strut is supported in a trunnion mounted pivot beam attached to forged recesses in number three and four wing spars. On one end the pivot beams are hinged to the shock strut mountings and on the other end they are pinned in the recessed trunnion mountings. The shock strut outer cylinder housing is made of 7075-T6 aluminum alloy forging. The forward and aft drag braces on each main landing gear are cylindrical in shape and are pressure-welded at each end to forged end fittings. The drag braces are hollow and serve as air storage accumulators for the high-pressure pneumatic system. The drag braces are made of SAE 4340 steel. The side brace assembly of each main landing gear consists of two subassemblies. The subassemblies hinge as a knee at a fork connection. The lower assembly is a forged I-section. The upper assembly is cylindrical and incorporates an internal mechanical lock linkage. The torque arm assembly, or scissors, includes an upper arm attached to the shock strut outer cylinder housing, and a lower arm linked to the piston tube. The arms hinge at their apex. Both sections of the scissors are made of 7075-T6 aluminum alloy.

5-10. Main Landing Gear Fuselage Doors.

5-11. The main landing gear fuselage door structure consists of a zee-shaped pan forming the outer frame, a

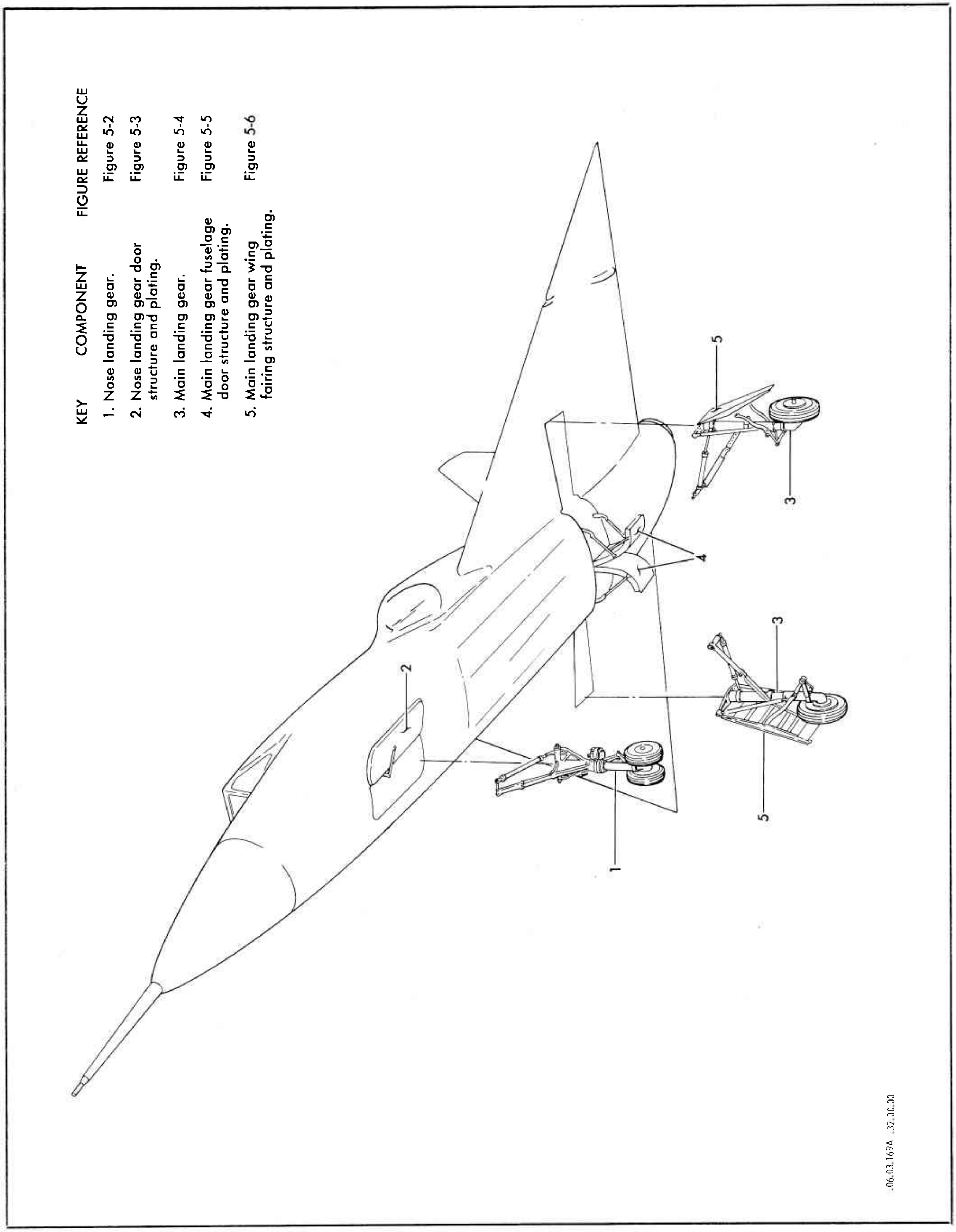


Figure 5-1. Landing Gear Components and Figure Index

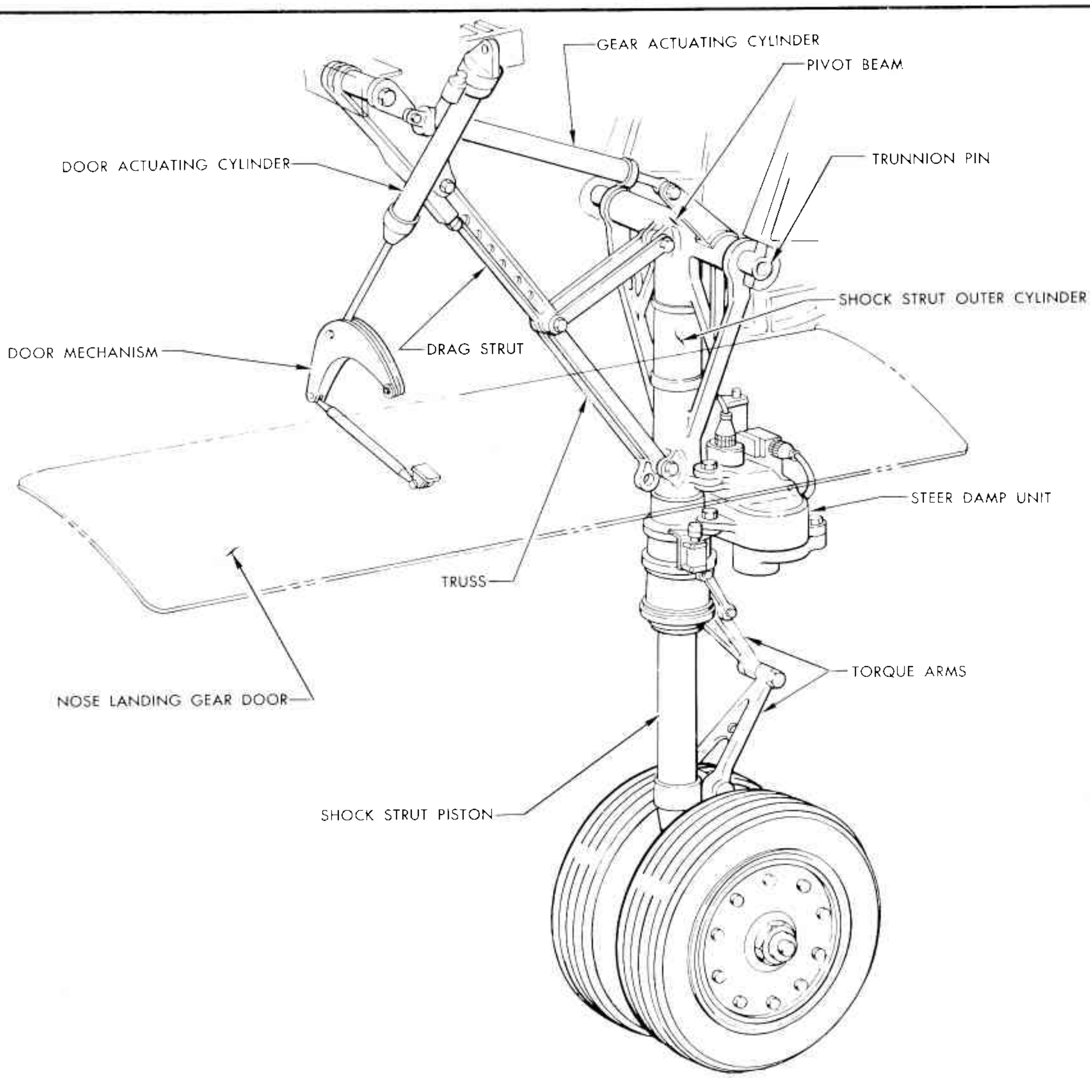


Figure 5-2. Nose Landing Gear

1. TO PREVENT STRESS CONCENTRATION, REMOVAL OF MATERIAL IN RADII IS NOT PERMITTED.
2. ON THE SURFACES INDICATED BY ARROWS AND THE SHADED SURFACES, 20% OF THE ORIGINAL THICKNESS MAY BE REMOVED OVER THE ENTIRE SURFACE FOR CLEAN UP OF CORROSION.
3. IF THE SURFACE TO BE REWORKED REQUIRES REMOVAL OF MORE THAN 20% OF THE ORIGINAL THICKNESS, CONSULT AN AEROSPACE ENGINEER.
4. IF CORROSION DAMAGE EXISTS ON BOTH SIDES OF A SURFACE, THE TOTAL REMOVAL FROM BOTH SIDES SHALL NOT EXCEED 20% OF THE ORIGINAL THICKNESS.
5. THE REWORKED AREAS MUST BE BLENDED INTO THE SURROUNDING AREAS WITH NO SHARP DISCONTINUITIES IN THE MATERIAL SURFACE.
6. TOUCH UP REWORKED AREAS IN ACCORDANCE WITH THE CORROSION PREVENTION TECHNIQUES CONTAINED IN SECTION I OF THIS T.O.

NOTE

THE SAME AREA OF THE FITTING WILL NOT BE REWORKED IF 20% OF ORIGINAL THICKNESS HAS PREVIOUSLY BEEN REMOVED.

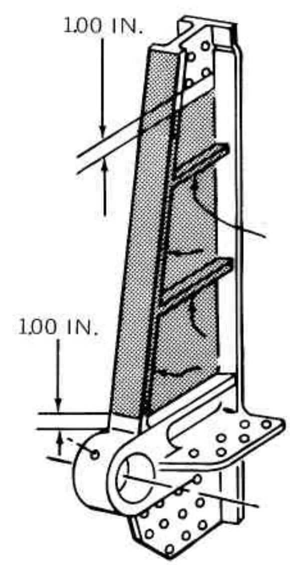


Figure 5-2A. Nose Landing Gear Trunnion Fitting Repair

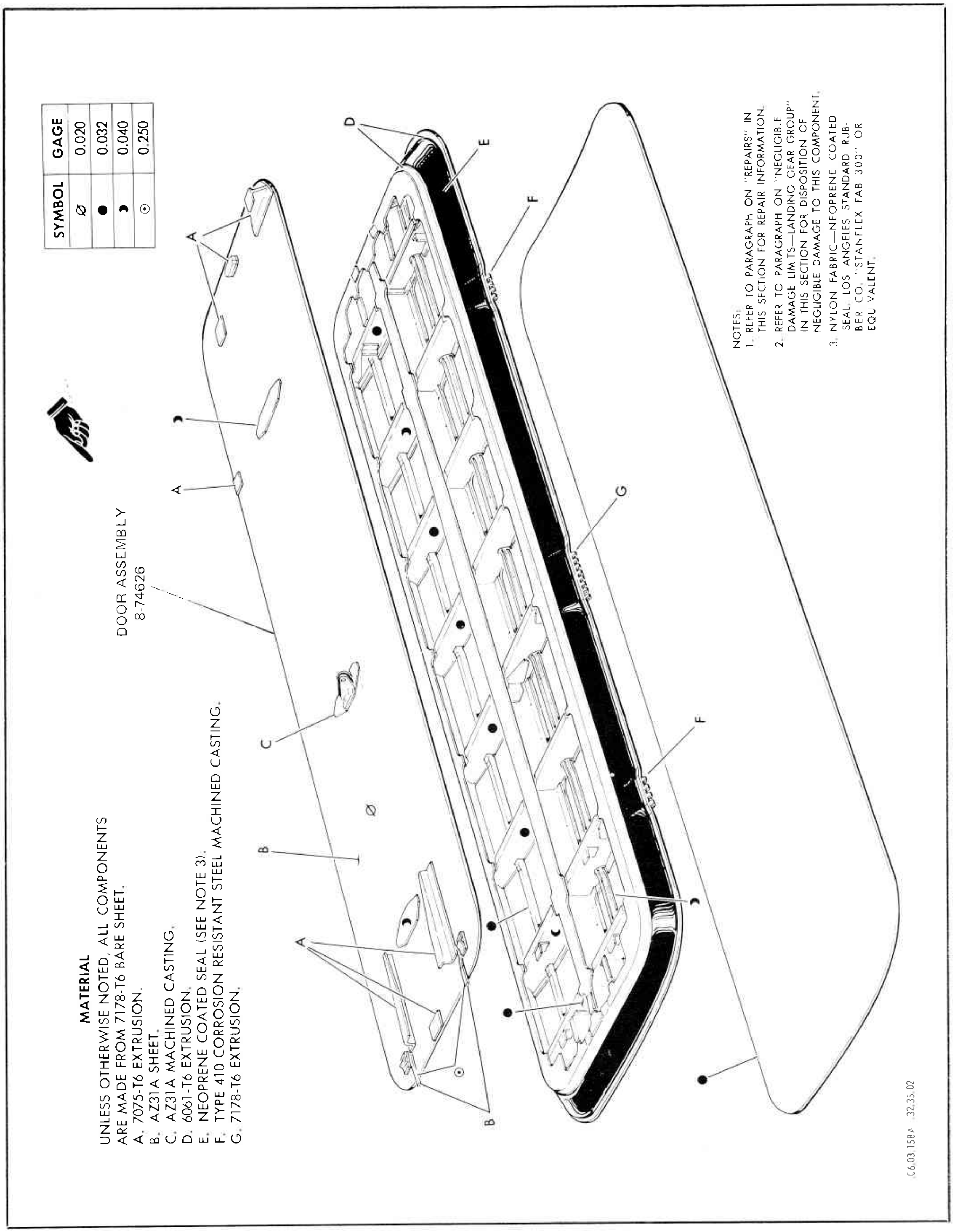


Figure 5-3. Nose Landing Gear Door Structure and Plating

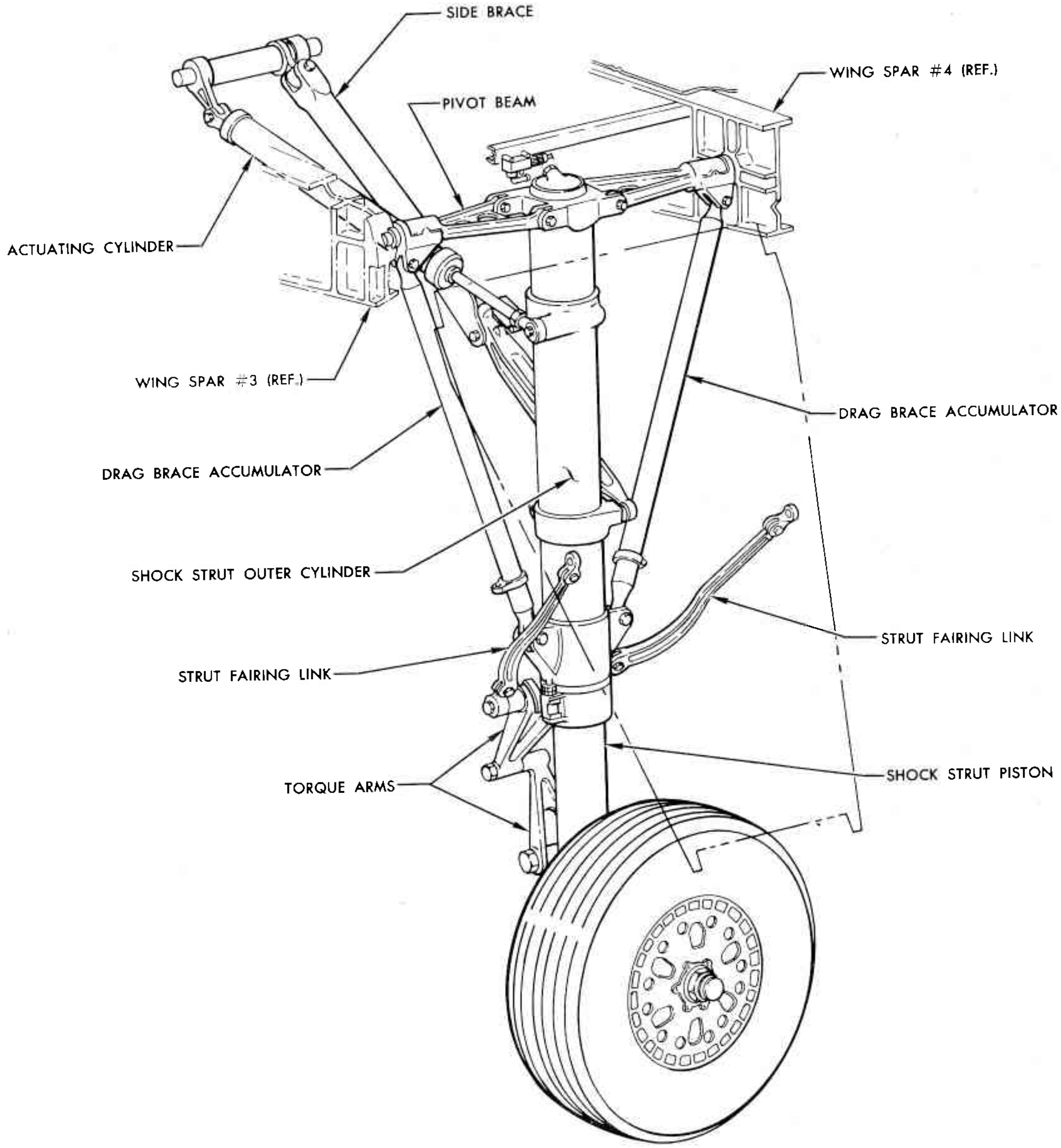


Figure 5-4. Main Landing Gear

longitudinal beam, three transverse ribs, and an inner and outer skin. As shown on figure 5-5, the outer frame is divided longitudinally by a beam formed of angles and tees. Three ribs perpendicular to the longitudinal beam are attached to the outer frame with clips and rivets. The beam and ribs reinforcing the outer frame are made of 7075-T6 clad aluminum alloy. A heavy gage outer skin made of 7075-T6 clad aluminum alloy sheet is spotwelded to the inner door structure and a lighter gage inner skin of the same material is riveted to the outer frame, beam, and ribs. The door is actuated by a hydraulic door mechanism consisting of an idler type bellcrank and an actuating cylinder. The door mechanism is spring-loaded overcenter in both the open and closed positions. The main landing gear fuselage door attaches to the fuselage with a piano-type hinge along the inboard edge of the door structure. See figures 5-9 and 5-10 for typical door repair.

5-12. Main Landing Gear Wing Fairings.

5-13. The main landing gear wing fairing structure consists of a forward and aft vertical beam, a series of horizontal stiffener ribs, an outer skin, and a partial inner skin. As shown on figure 5-6, the horizontal stiffeners are attached to the forward and aft vertical beams with clips and rivets. The beams, stiffener ribs, and the skins are made of 7075-T6 clad aluminum alloy of various gages. The outer skin is spotwelded and riveted to the beams and stiffeners, and completely covers the outer surface of the fairing structure. The inner skin is in two parts and covers the inner portion near the upper and lower extremities of the fairing. Brackets are bolted near the center of the forward and aft beams to provide the attaching points for the link assemblies fastened to the main landing gear shock struts. The top of the fairing is hinged from the wing structure at buttock line 88.70 by means of two fitting assemblies and removable pins. The fairing is locked in the up position by means of striker plates which mesh with rollers on the main landing gear fuselage door. See figure 5-11 for repair of fairing.

5-14. REPAIRS.

5-15. Negligible Damage.

5-16. Negligible damage to the landing gear components will be restricted to nicks and dents which, after being cleaned up to a regular shape, do not exceed 1/32 inch in depth and do not occur within 3/4 inch of terminal or lug holes. Any damage other than negligible damage will require engineering disposition. Table 5-I indicates the maximum allowable classifications of five types of negligible damage (damage allowed to remain "as is," after minor rework such as stop drilling cracks and fairing in nicks or scratches). The maximum allowable damage classification will be found to the right of the component name in the vertical column under the "type of damage" heading. After classification is determined, see figures 1-17 through 1-19 for the damage limits allowed for each class: I, II, or III. The limits given on figures 1-17 through 1-19 apply only for a damaged area after rework, as shown on figures 1-20 and 1-21. An aeronautical structural engineer must be consulted for damages exceeding the limits given on figures 1-17 through 1-19, and for damage to components not listed in the table.

5-17. Repair of Nose Wheel Well Curtains.

5-18. Rips, tears, and holes in the nose wheel well curtains are to be repaired as follows:

a. Roughen surface of curtain to be repaired with rough sandpaper.

b. Mix enough EPON 828 to complete repair. Follow the manufacturer's mixing instruction shown on the container.

c. Brush on one coat of mixed adhesive to repair area of curtain.

d. Apply a fiberglass cloth patch over damage. Patch should overlap damaged area by 1/2 inch on all sides.

e. Brush a coat of mixed adhesive over patch and allow adhesive to cure.

NOTE

Use a heat lamp to speed cure time.

5-19. Rivet Substitution.

5-20. All rivets used in the landing gear fairing and in the fuselage landing gear doors may be substituted by different types which have either the same or greater strength values than the original rivet, except for the following restriction: No blind rivets may be used in the nose landing gear door.

5-21. Landing Gear Alignment.

5-22. The landing gear is the nonadjustable fixed type manufactured by the Menasco Manufacturing Company. These gears are aligned by a precision fixture at time of airplane manufacture and are held in alignment by their attachment to the airplane structure. For further information concerning landing gear alignment, refer to paragraph 1-270.

5-23. Repair to Doors.

5-24. All repairs to the nose landing gear door must be the flush-type and will require the addition of a sealer to faying surfaces of the parts involved to provide airtightness. Figure 5-8 illustrates and describes a typical nose landing gear door structure and plating repair. See figures 5-5 and 5-6 for reference information regarding repairs to the main landing gear fairing and fuselage door.

5-25. Main Landing Gear Fairing Gap Repair.

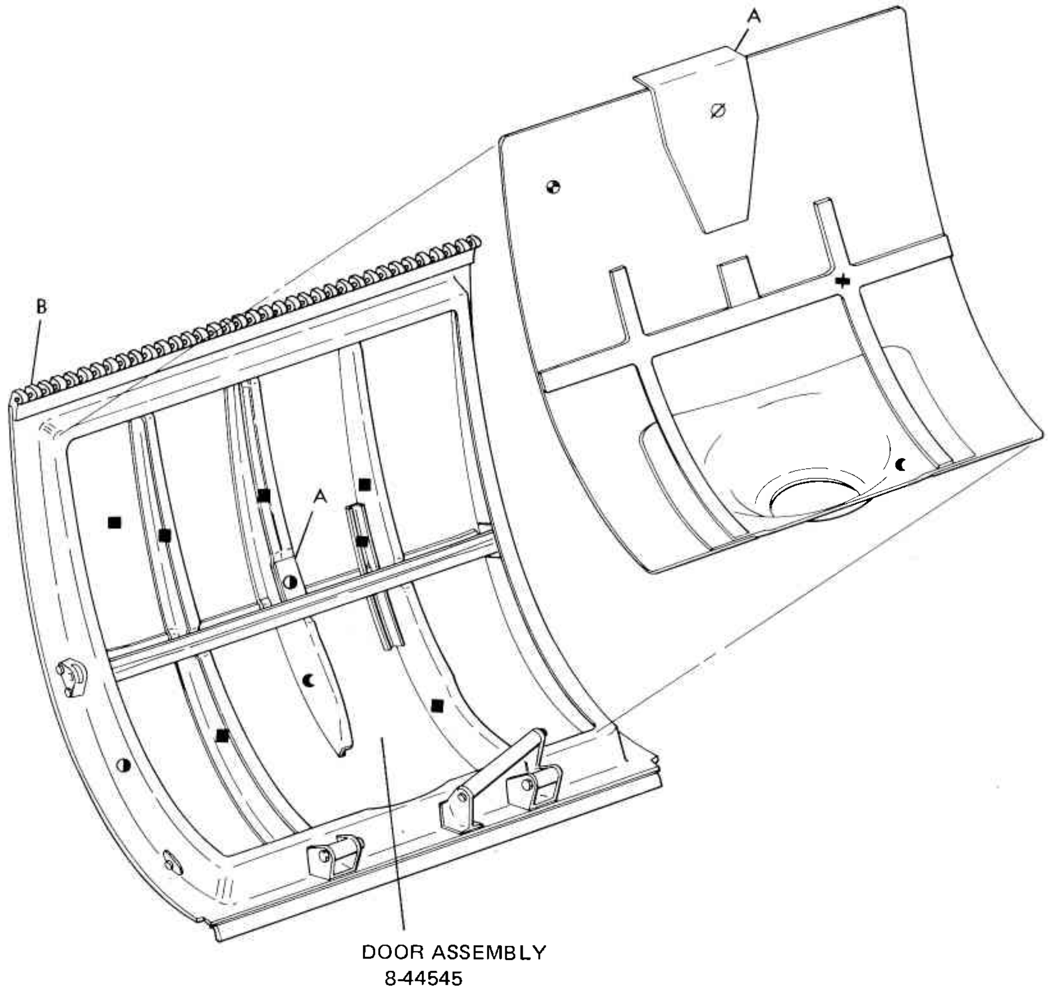
5-26. Transverse gaps greater than 0.080 inch may be reduced to within tolerance by the use of aerodynamic smoothing compound as illustrated on figure 5-7.

MATERIAL

UNLESS OTHERWISE NOTED, ALL COMPONENTS ARE MADE FROM 7075-T6 CLAD SHEET

A. Type 301 corrosion resistant steel.
B. 7178-T6 extrusion.

SYMBOL	GAGE
∅	0.020
◕	0.025
☾	0.040
■	0.050
◑	0.063
✚	0.090



NOTES:
1. REFER TO PARAGRAPH ON "REPAIRS" IN THIS SECTION FOR REPAIR INFORMATION.
2. REFER TO PARAGRAPH ON "NEGLIGIBLE DAMAGE LIMITS— LANDING GEAR GROUP" IN THIS SECTION FOR DISPOSITION OF NEGLIGIBLE DAMAGE TO THIS COMPONENT.

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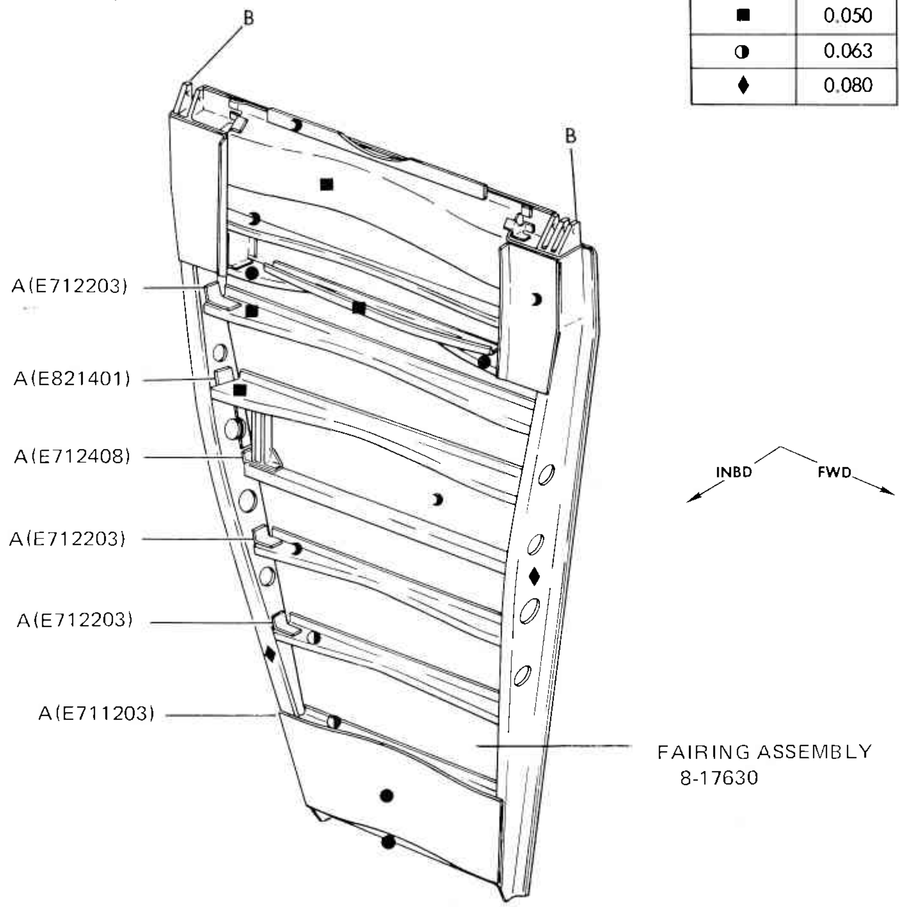
Figure 5-5. Main Landing Gear Fuselage Door Structure and Plating

MATERIAL

UNLESS OTHERWISE NOTED, ALL COMPONENTS ARE MADE FROM 7075-T6 CLAD SHEET

A. 7075-T6 extrusion with Ident No. (See Sec VIII).
B. 7075-T6 machined bar.

SYMBOL	GAGE
●	0.032
◗	0.040
■	0.050
◑	0.063
♦	0.080



NOTES:
1. REFER TO PARAGRAPH ON "REPAIRS" IN THIS SECTION FOR REPAIR INFORMATION.
2. REFER TO PARAGRAPH ON "NEGLIGIBLE DAMAGE LIMITS—LANDING GEAR GROUP" IN THIS SECTION FOR DISPOSITION OF NEGLIGIBLE DAMAGE TO THIS COMPONENT.

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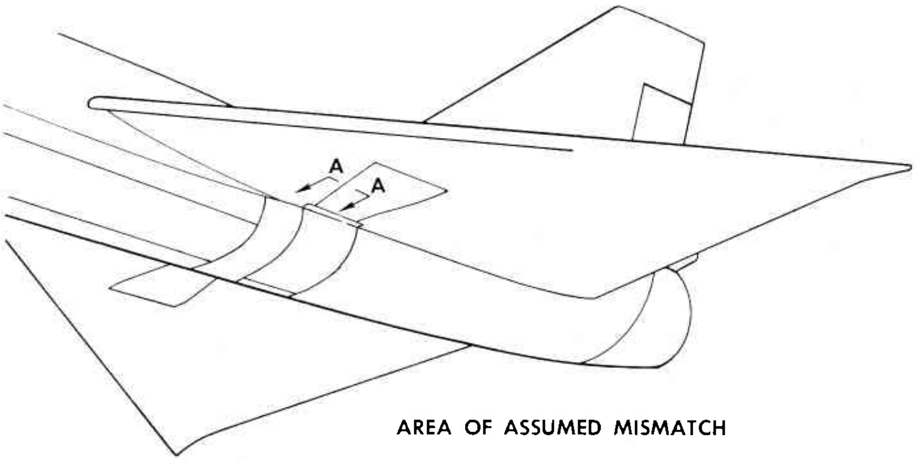
Figure 5-6. Main Landing Gear Wing Fairing Structure and Plating



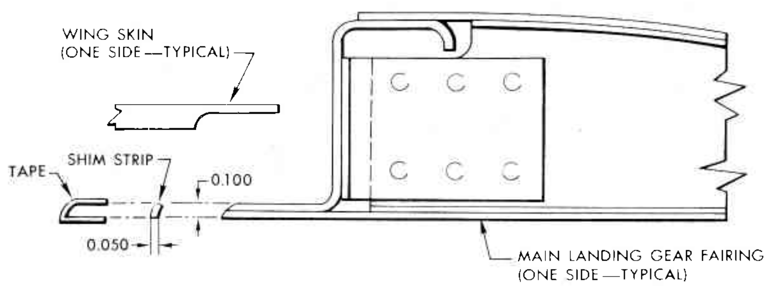
TRANSVERSE GAPS (GAPS ALONG BOTH SIDES OF MAIN LANDING GEAR FAIRING WHERE IT MATES WITH WING SKIN) IN EXCESS OF 0.080 INCH BETWEEN WING SKINS AND RETRACTED MAIN LANDING GEAR FAIRING DOOR MAY BE CORRECTED BY THE FOLLOWING PROCEDURE:

a. Clean painted mating flange of wing skin with cheese cloth dampened with aliphatic naphtha, Specification TT-N-95.
b. Fabricate a shim from a strip of metal or plastic to required length. Dimensions 0.050 inch by 0.10 inch.
c. Attach shim strip to edge of door with a pressure sensitive tape which is non-adhesive on the back side. Use Teflon tape (1.0 inch wide), Minnesota Mining and Mfg. Co., No. 529 or equivalent.
d. Retract gear and fairing to the fully closed position.

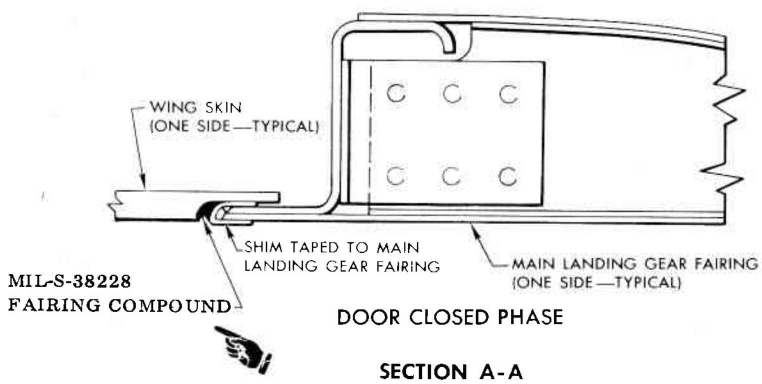
NOTE

INSPECT SHIM STRIP TO MAKE CERTAIN IT IS STILL TAPED SECURELY IN PLACE AND DOES NOT EXTEND BEYOND OUTER CONTOUR OF DOOR AND WING SKIN.

e. Mix a suitable quantity of MIL-S-38228 fairing compound.
f. Apply a fillet of mixed MIL-S-38228 to radius of door opening machined in wing skin. Apply fairing compound in a band wide enough to fill gap and thick enough to match outside contour.
g. Remove excess compound and allow compound to cure.
h. Sand smooth, open door, remove tape and shim strip, and touch up with applicable paint schedule contained in Section I.



DOOR OPEN PHASE

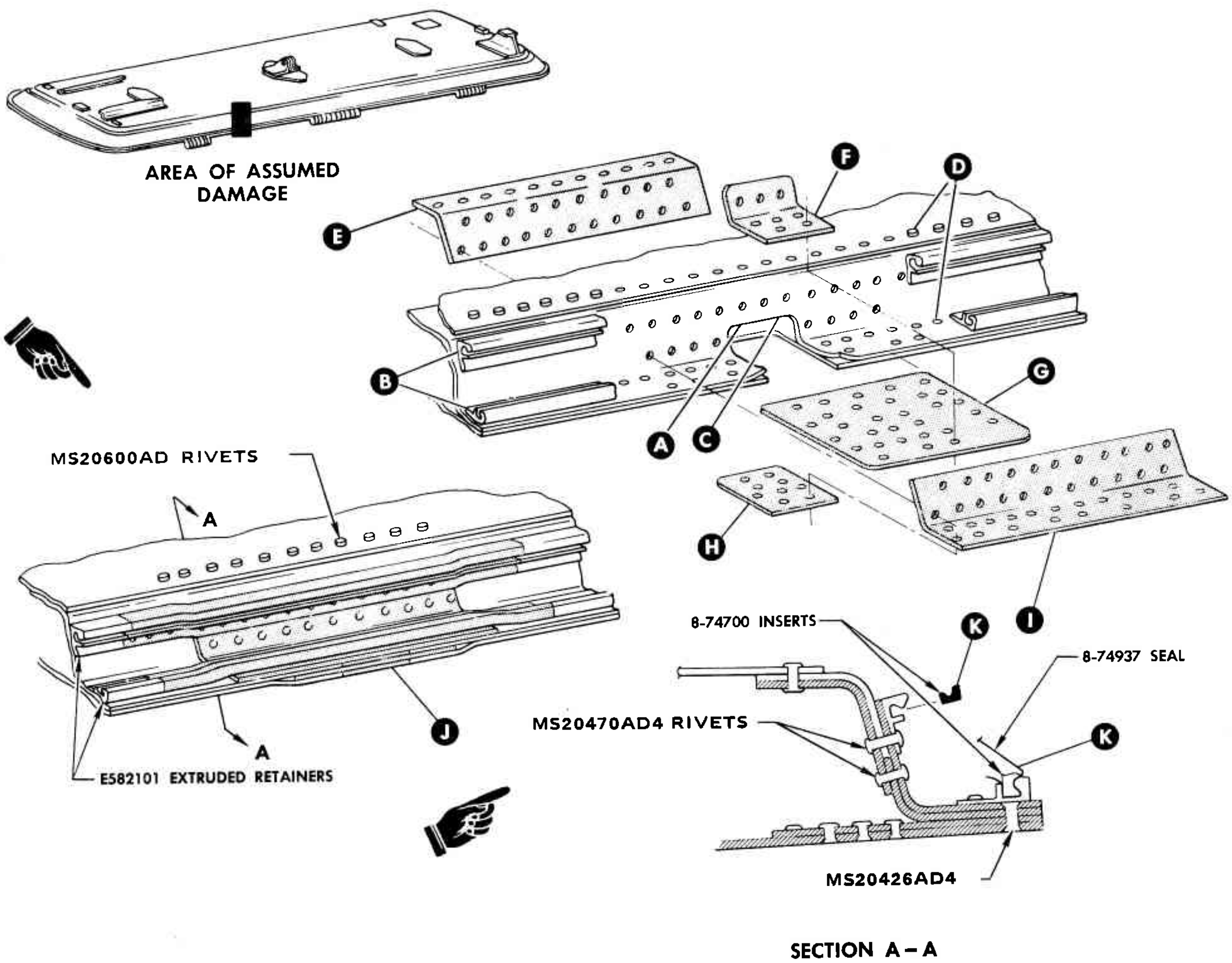


DOOR CLOSED PHASE

SECTION A-A

106 03 292B

Figure 5-7. Main Landing Gear Fairing Repair



A Clean up damaged area to regular shape. Round all corners and break all sharp edges.

B Remove damaged sections of E582101 extruded retainers, 8-74700 inserts and 8-74937 seal.

C Cutback sufficiently to allow for installation of repair materials.

D Using a No. 30 (0.128) drill, remove existing MS20600AD rivets sufficiently to peel back web for accessibility. Drill No. 30 (0.128) holes at spot welds. Countersink 100°x0.235 far side and both sides outer row.

E Fabricate inside splice angle of same type material and gage as door pan. Drill No. 30 (0.128) holes to match existing rivet holes in pan.

F Fabricate filler plate of same type material and gage as door pan. Drill existing holes.

G Fabricate doubler from 0.020 gage type 301 ¼ hard corrosion resistant steel or suitable substitute. Drill No. 30 (0.128) holes to match existing patterns. Insert between skin and flange of pan. (Refer to Section I for minimum rivet spacing.)

H Fabricate flush patch of same type material and gage as outer skin. Drill No. 30 (0.128) to match existing patterns. (Refer to Section I for minimum rivet spacing.)

I Fabricate outside splice angle from 0.025 gage, type 301 ¼ hard corrosion resistant steel or suitable substitute. Drill No. 30 (0.128) holes to match existing rivet patterns.

J Install inside splice angle, filler plate, flush patch, doubler, outside splice angle and section of E582101 extruded retainers with MS20600AD, MS20470AD4 and MS20426AD4 rivets as shown.

K Install 8-74700 inserts and 8-74937 seal.

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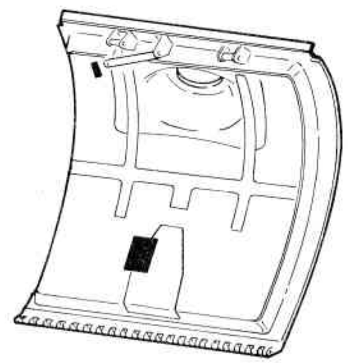
Figure 5-8. Nose Landing Gear Door Repair



AREA OF ASSUMED DAMAGE

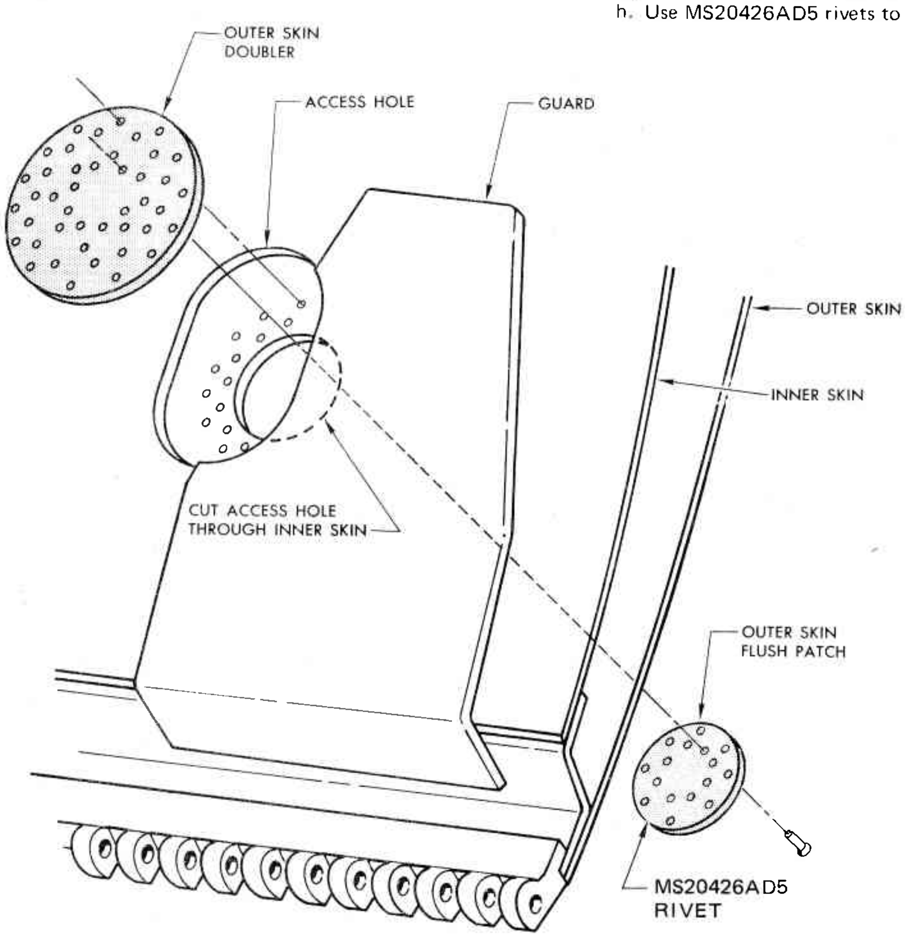
REPAIR PROCEDURE
MAIN LANDING GEAR DOOR

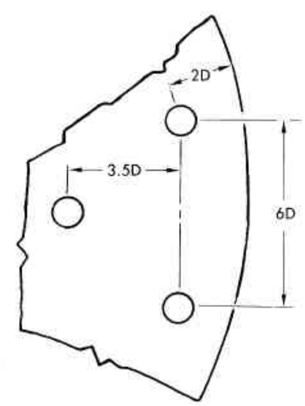
a. Remove main landing gear door and place on a padded work bench.
b. To determine extent of damage make a fluorescent penetrant inspection. Refer to Section I for fluorescent penetrant procedures.
c. Rout out damaged area to a regular shape.
d. Cut an access hole through inner skin. Access hole must be large enough to install a doubler to the outer (damaged) skin.
e. Fabricate doubler from 0.063" 7075-T6 bare aluminum alloy sheet.

NOTE

APPLY FINISH PROTECTION TO REPAIR MATERIALS. REFER TO T.O. 1F-106A-23 FOR APPLICABLE FINISH REQUIREMENTS.

f. Hold doubler in place and drill pilot holes through skin and doubler. Use a No. 24 (0.152) pilot drill. Refer to figure 1-32 for rivet spacing. Ream holes with a No. 20 (0.161) drill.
g. Countersink rivet holes in outer skin.
h. Use MS20426AD5 rivets to attach doubler to skin.





TYPICAL COUNTERSUNK RIVET PATTERN

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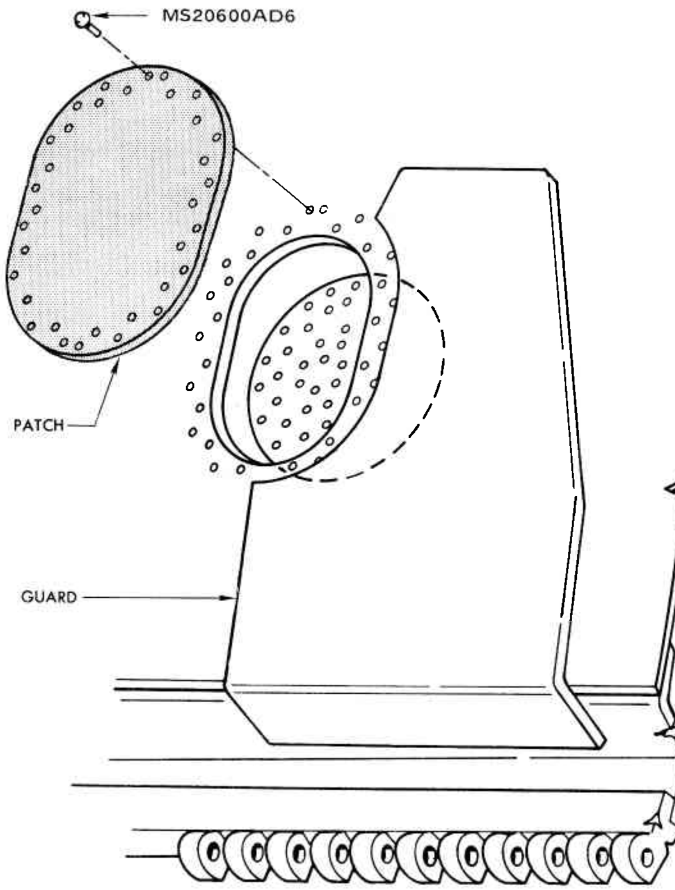
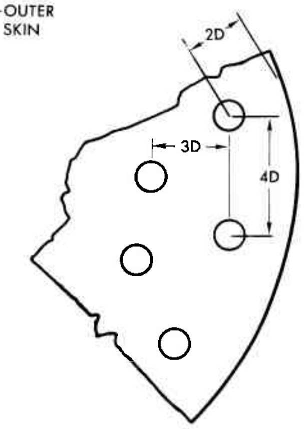
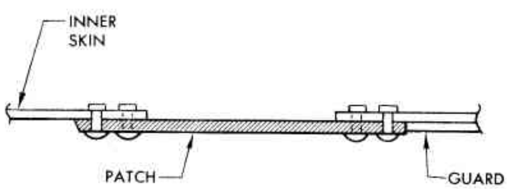
Figure 5-9. Main Landing Gear Door Repair (Sheet 1 of 2)



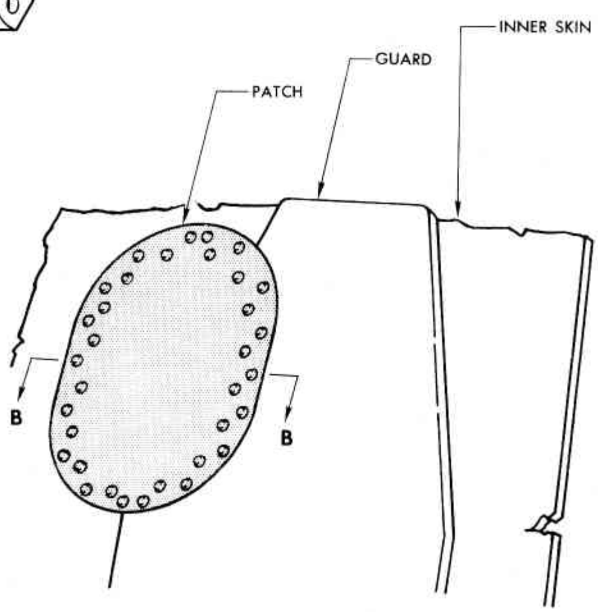
EXPLODED REPAIR



TYPICAL PROTRUDING HEAD
RIVET PATTERN



SECTION B-B



ASSEMBLED REPAIR

REPAIR PROCEDURE
MAIN LANDING GEAR DOOR
(CONT)

i. Fabricate a patch from 0.032" 7075ST clad aluminum alloy.
j. Hold patch in place and drill rivet holes. Use a No. 29 (0.136") drill. Refer to Tables 1-VI and 1-VII for rivet spacing.
k. Hold patch in place with cleco fasteners and rivet to inner skin with MS20600AD6 Rivets. Install rivets in accordance with instructions contained in Section I.
l. Install door and adjust in accordance with T.O. 1F-106A-2-8-2-1.

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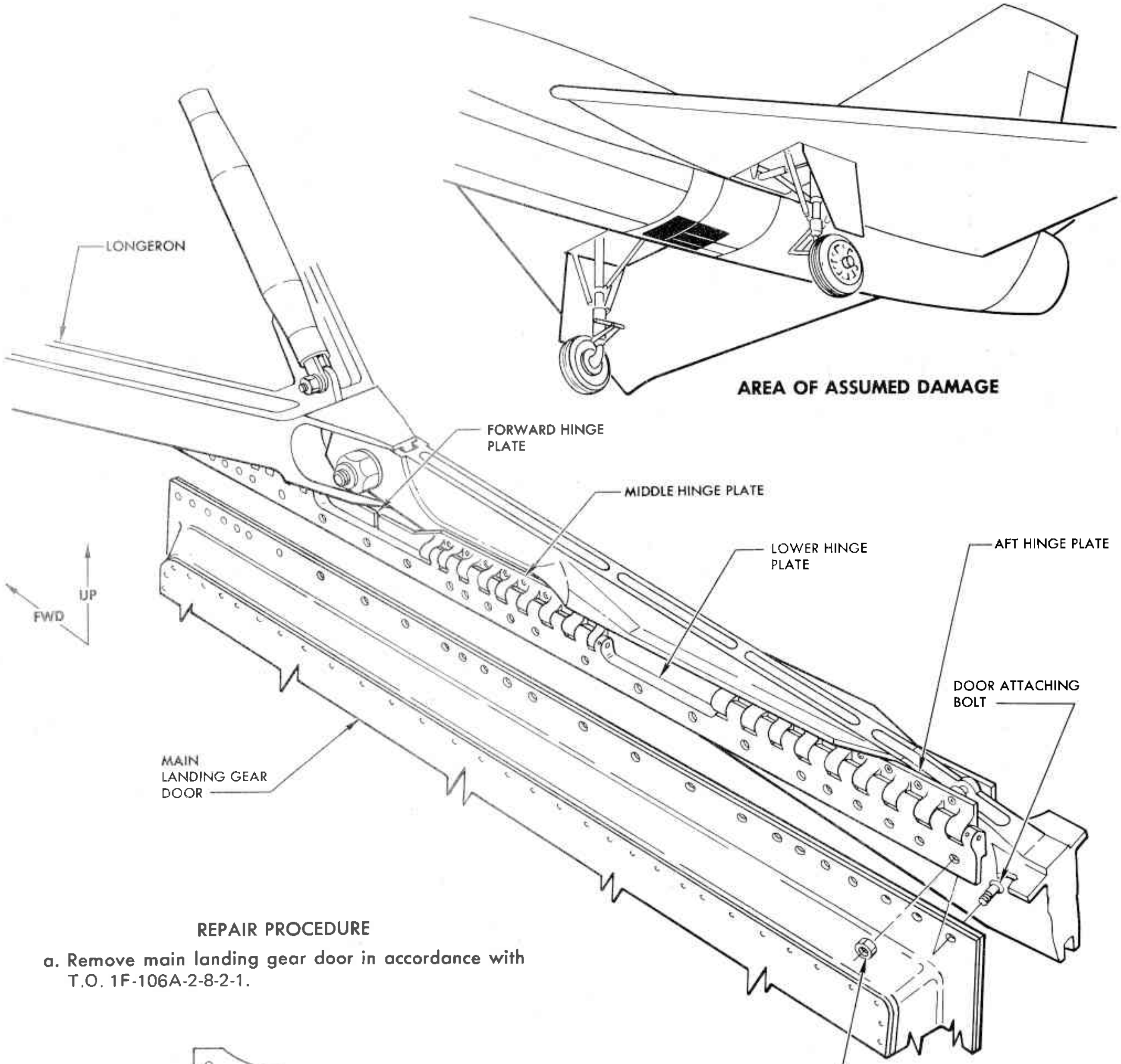
Figure 5-9. Main Landing Gear Door Repair (Sheet 2 of 2)



REPAIR PROCEDURE

a. Remove main landing gear door in accordance with T.O. 1F-106A-2-8-2-1.

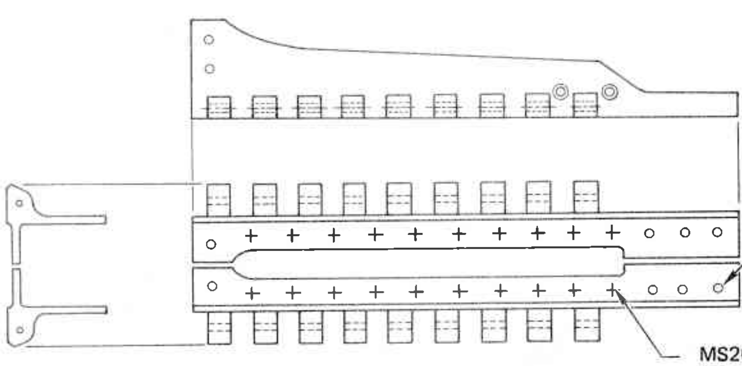


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Figure 5-10. Main Landing Gear Door Attachment Repair (Sheet 1 of 2)

REPAIR PROCEDURE
(CONT)

b. Remove AN3-6 bolts (2) at forward end of damaged forward hinge plate.
c. Remove 10-32 screws (2) from lower forward end of damaged forward hinge plate.
d. Remove 10-32 screws (2) from aft end of damaged forward hinge plate.
e. Drill out MS20426AD5 rivets and 3/16 inch blind bolts (core bolts) from filler and forward hinge plate.
f. Remove attaching screws from aft hinge plate and remove entire door attachment assembly (forward hinge plate, aft hinge plate and lower hinge plate).
g. Remove hinge pin lock from lower hinge plate and remove forward hinge pin.
h. Replace damaged forward hinge plate and reassemble attachment assembly.
i. Installation of door attachment assembly is essentially the reverse of removal.

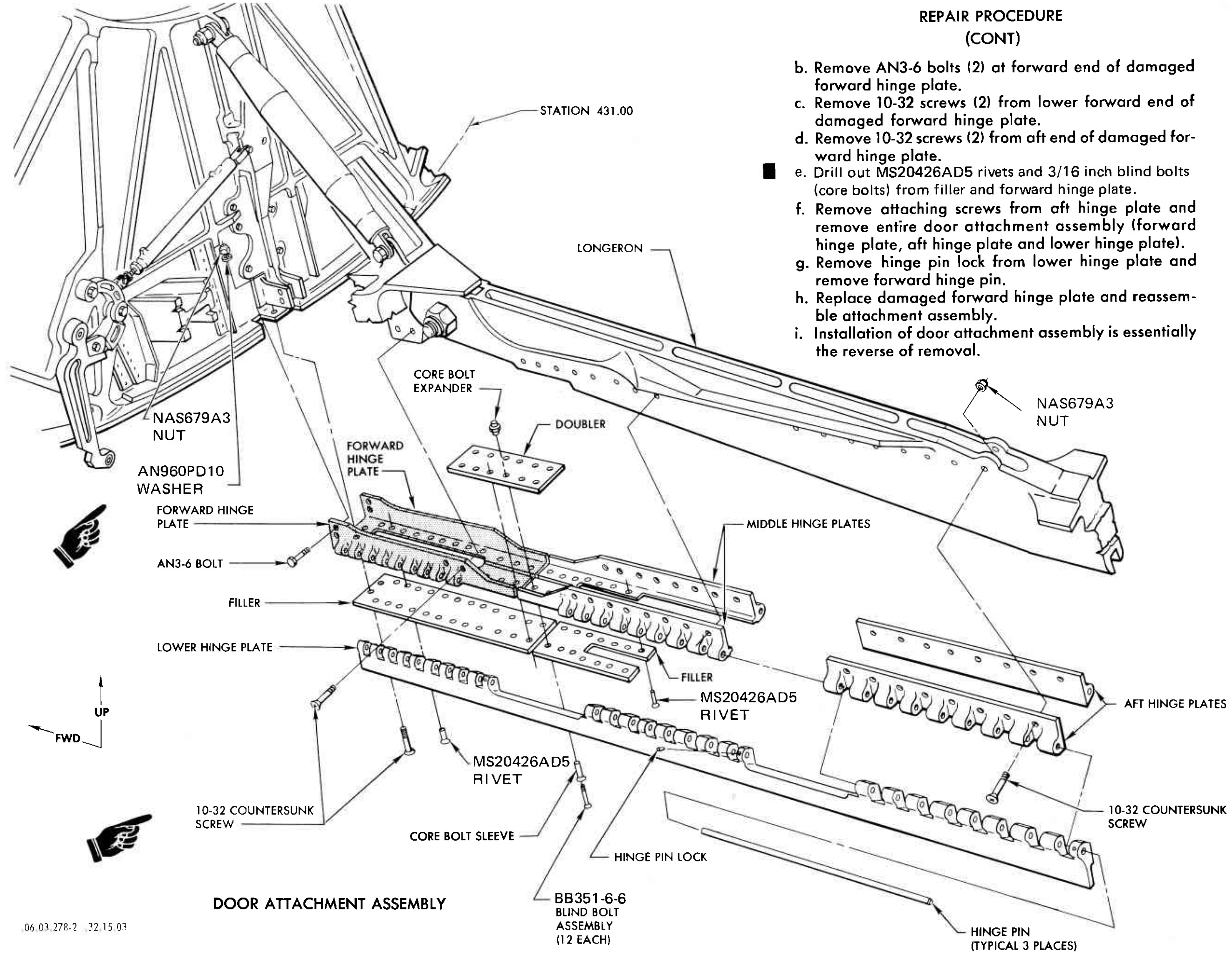
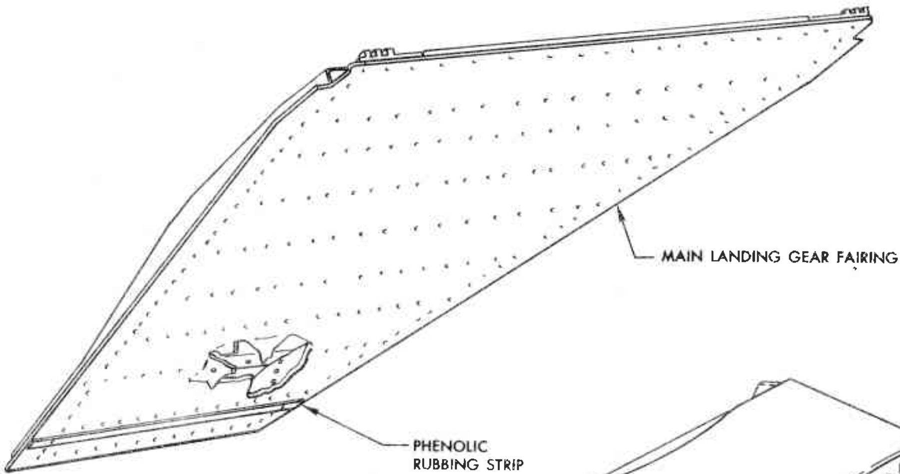
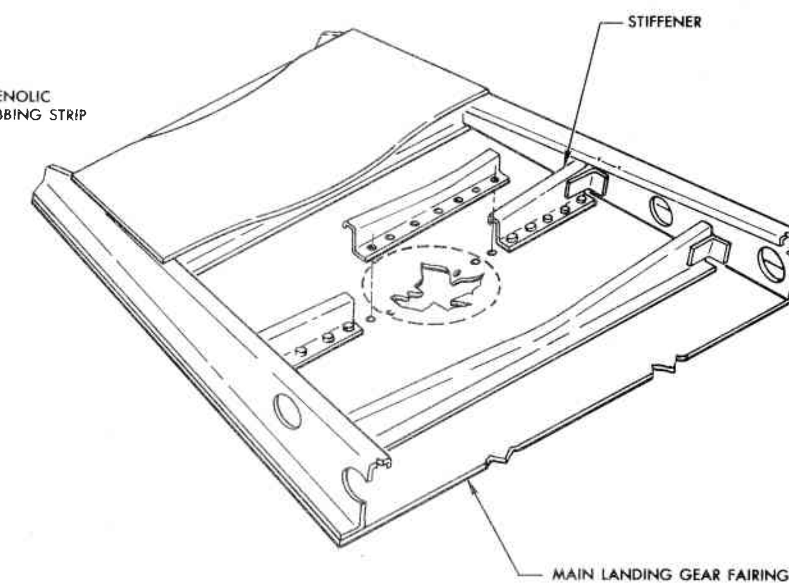
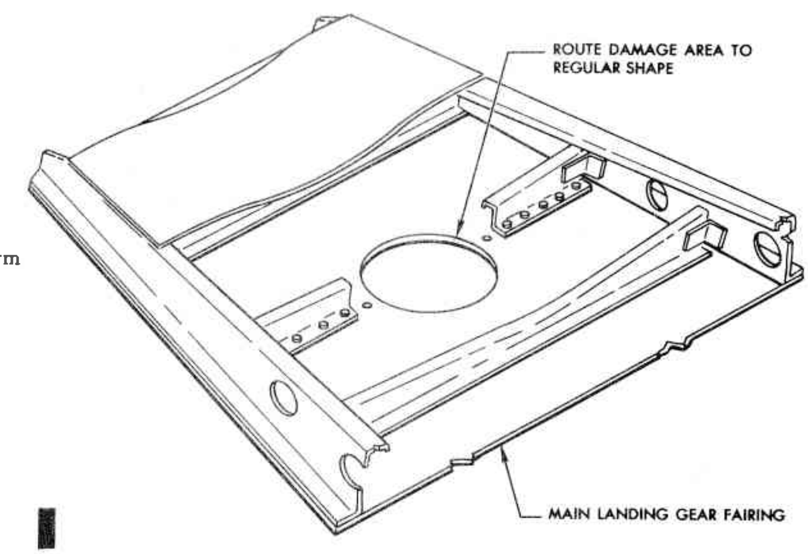


Figure 5-10. Main Landing Gear Door Attachment Repair (Sheet 2 of 2)



AREA OF ASSUMED DAMAGE





REPAIR PROCEDURE MAIN LANDING GEAR FAIRING

a. Remove main landing gear fairing and place on a padded work bench.
b. To determine extent of damage, perform a fluorescent penetrant inspection.
c. Cut out damaged area of stiffener.
d. Rout out damaged skin to a regular shape.
e. Fabricate a doubler from 0.050" 7075-T6 bare sheet.

NOTE

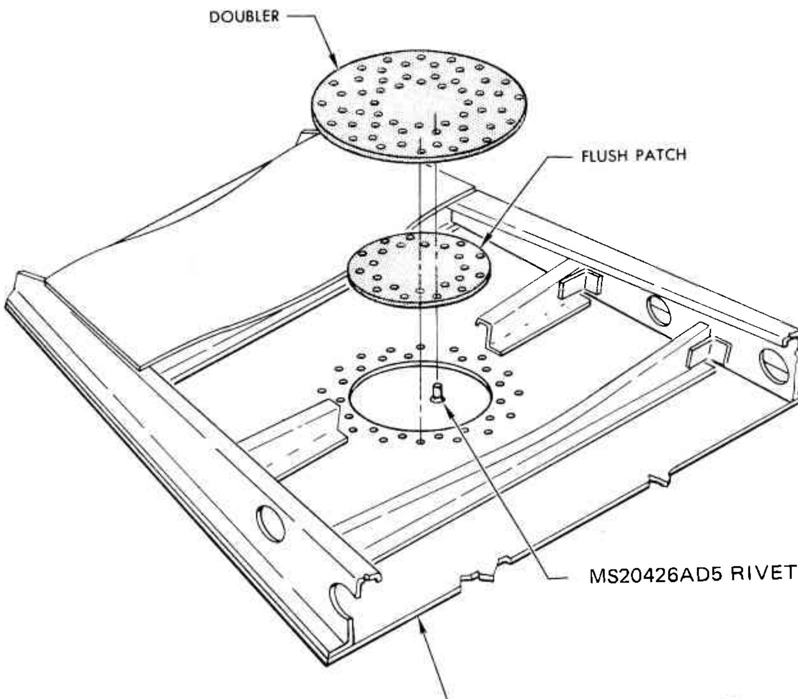
APPLY FINISH PROTECTION TO REPAIR MATERIALS PER T.O. 1F-106A-23.

06.03.268-1B .32.27.03

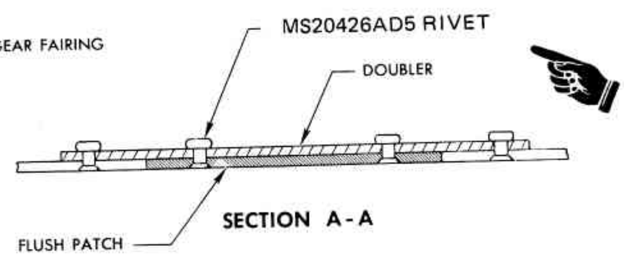
Figure 5-11. Main Landing Gear Fairing Repair (Sheet 1 of 3)

REPAIR PROCEDURE
MAIN LANDING GEAR FAIRING
(CONT)

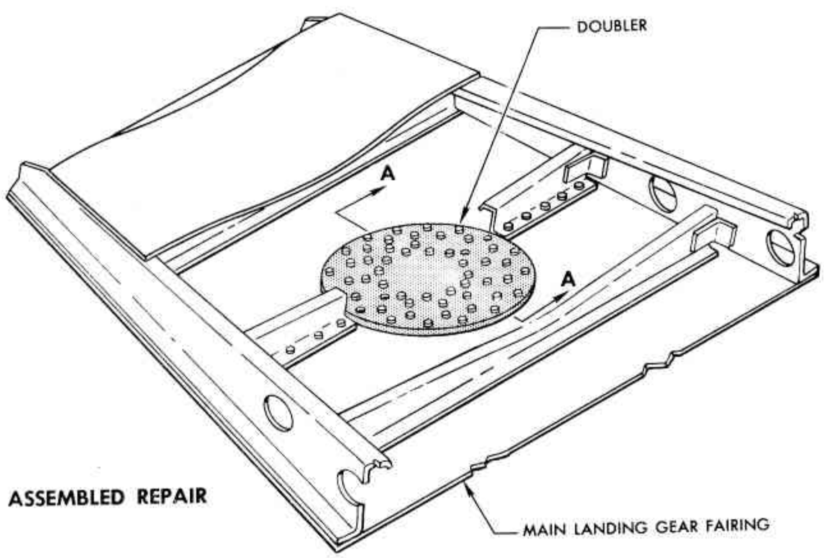
f. Hold doubler in place and drill rivet holes. Use a No. 21 (0.0160'') pilot drill. Ream with a No. 20 (0.161'') drill. Refer to Section I for standard rivet spacing.
g. Countersink rivet holes in skin.
h. Use MS20426AD5 100 degree countersink rivets to fasten doubler to skin.
i. Fabricate flush patch from 0.040'' 7075-T6 clad aluminum alloy sheet.
j. Use a No. 21 (0.160'') drill to drill rivet holes through flush patch and doubler. Ream with a No. 20 (0.161'') drill. Refer to Section I for standard rivet spacing.
k. Countersink rivet holes in patch and rivet to doubler.



EXPLODED REPAIR



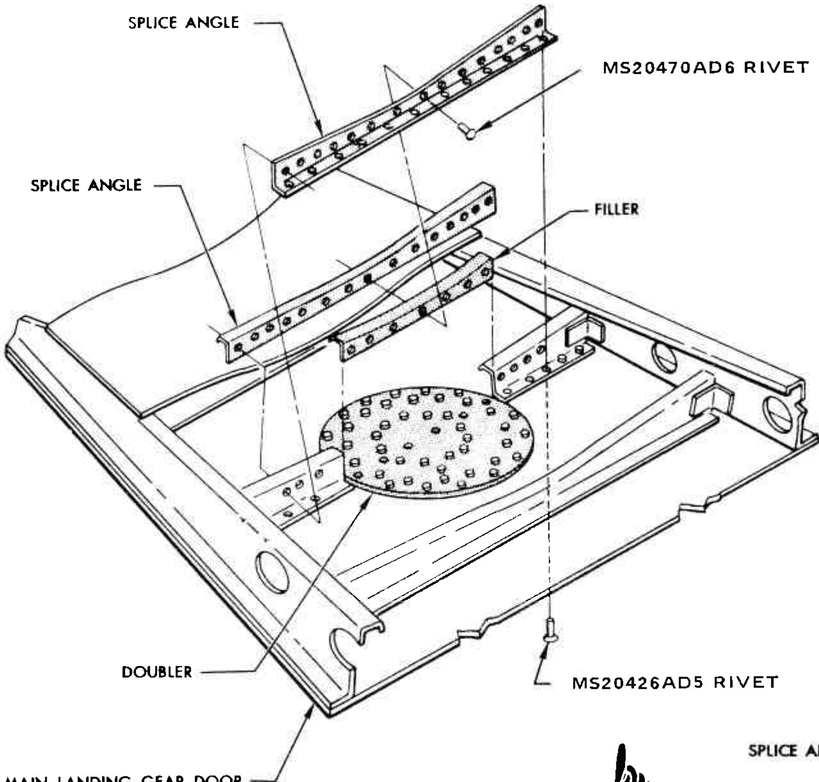
SECTION A-A



ASSEMBLED REPAIR

06.03.268-2B

Figure 5-11. Main Landing Gear Fairing Repair (Sheet 2 of 3)

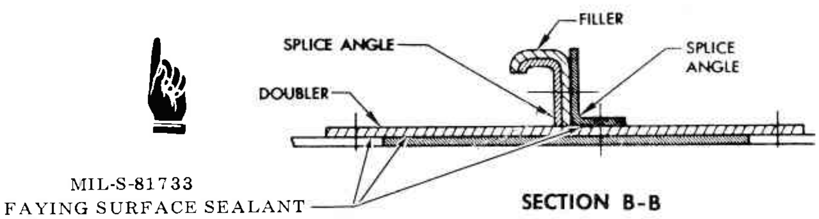


EXPLODED REPAIR

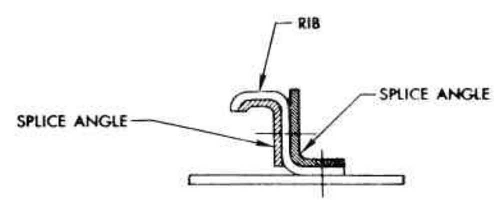
REPAIR PROCEDURE MAIN LANDING GEAR FAIRING (CONT)

l. Fabricate filler from 0.063" 7075-0 aluminum alloy.
m. Shape filler to conform to contour of damaged stiffener.
n. Fabricate two splice angles from 0.063" 7075-0 aluminum alloy.
o. Heat treat filler and repair angles to T6 condition.
p. Drill rivet holes for splice angles as shown.
q. Rivet repair in place. Use MS20470AD5 rivets to attach filler and splice angles; use MS20426AD5 rivets to attach skin to splice angles.

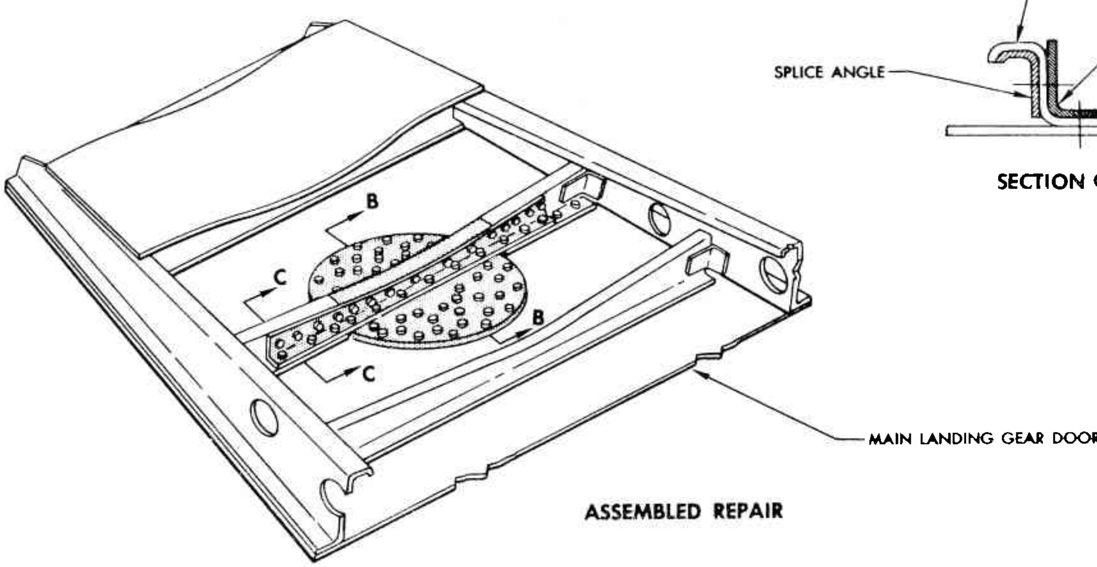
NOTE
IF PHENOLIC RUBBING STRIP SHOWN IN AREA OF ASSUMED DAMAGE VIEW IS DAMAGED, INSTALL A NEW STRIP OF THE SAME DIMENSIONS. INSTALL NEW STRIP WITH MIL-S-81733.



SECTION B-B



SECTION C-C



ASSEMBLED REPAIR

06.03.268-3A .32.27.03

Figure 5-11. Main Landing Gear Fairing Repair (Sheet 3 of 3)

TABLE 5-I

Negligible Damage Limits — Landing Gear Section

COMPONENT	TYPE AND CLASS OF DAMAGE ALLOWED AFTER REWORK					REMARKS
	Scratch	Nick	Dent	Hole	Crack	
NOSE LANDING GEAR FITTINGS						
Support Fitting	Refer to an Aeronautical Structural Engineer					
Drag Brace Fitting	Refer to an Aeronautical Structural Engineer					
NOSE LANDING GEAR DOOR STRUCTURE						
Inner Skin	II	II	III	II	II	
Outer Skin	I	I	*	*	*	
Pan	II	II	III	II	I	
Ribs	II	II	III	II	I	
Hinge Fittings	I	I	*	*	*	
MAIN LANDING GEAR FITTING						
Side Brace Fitting	Refer to an Aeronautical Structural Engineer					
MAIN LANDING GEAR DOOR STRUCTURES						
Inner Skin	II	II	III	II	II	
Outer Skin	II	II	*	*	*	
Pan	II	II	III	II	II	
Ribs	II	II	II	II	I	
Hinge Fittings	I	I Refer to an Aeronautical Structural Engineer				
MAIN LANDING GEAR FAIRING						
Inner Skin	II	II	III	II	II	
Outer Skin	II	II	*	*	*	
Channels	II	II	I	I	I	
Ribs	II	II	I	II	II	

*Component must be repaired or replaced.

5-27. Main Landing Gear Fairing Vertical Beams Repair.

5-28. The main landing gear fairing is pulled securely into wing contour when the landing gear door closes. For this reason the stiffness of the fairing vertical beams is critical. The repairs shown in Figure 5-12 can be made between BL47.50 and BL72.40 and are limited to one repair to the inner flange and one repair to the outer flange of each beam. If damage to a beam is more extensive, the beam should be replaced. After a beam repair. fairing contour should be checked; see paragraph 5-29.

5-29. Main Landing Gear Fairing Contour Check.

5-30. The contour of the main landing gear fairing must fall within specified limits. A poorly contoured fairing might not provide the necessary security when it is clamped in place by action of the gear door. Contour should be checked after a repair to a broken beam. A contour check template, Part No. 8-17630-93, may be manufactured locally if not available (see Figure 5-13). Figure 5-14 illustrates and describes the contour offset limits in graph form; fairings that do not meet these limits must be replaced.

5-31. Rebushing of Main Landing Gear Trunnion Cavity Hole.

5-32. If wing cavity hole, located in Spar Number 3 and Number 4, exceeds standard 1.9995-2.0001 diameter, rebore hole to minimum ID necessary for clean-up and install oversize bushing with 0.0015-0.0026 interference fit.

a. Recommended hole size for first rebore is 0.016 oversize.

b. Hole cavity may be rebored up to a maximum of 2.125 inches diameter where necessary to repair a defective hole. Maximum hole cavity surface roughness is 63 RMS.

WARNING

Personnel handling frozen parts must be adequately protected to prevent injury resulting from contact with parts.

c. Manufacture one piece oversize bushing similar to GD/Convair Part Number 8-16512-8 except the O.D. would graduate in size as per hole cavity diameter. Maximum bushing surface roughness is 40RMS. Thermally insert bushing by freezing the bushing with liquid nitrogen or dry ice and methylalcohol.

CAUTION

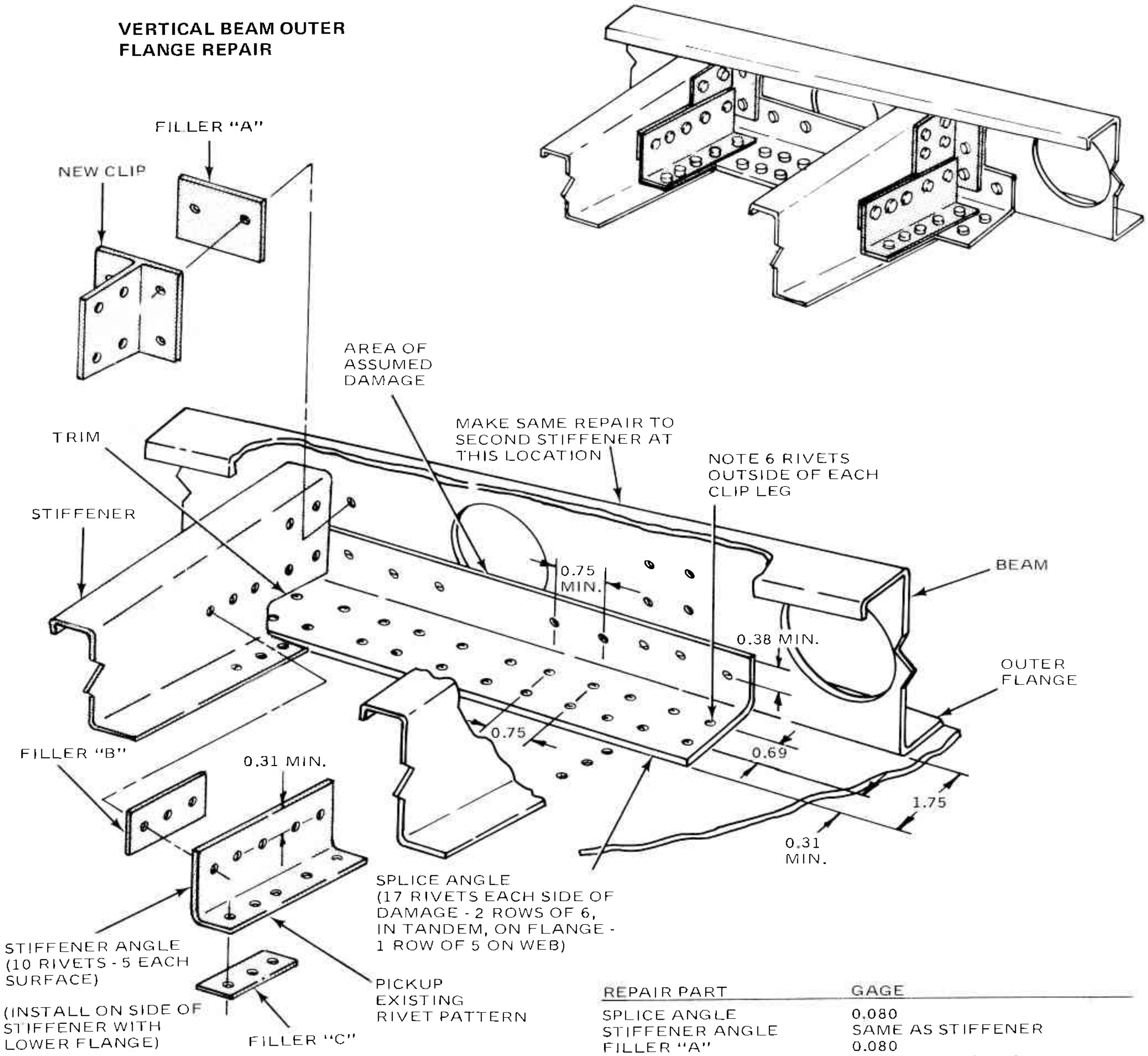
Methylalcohol shall not be used on magnesium materials due to corrosive action.

NOTE

Applicable to all F-106A/B aircraft.

5-33. Repair of Main Landing Gear Side Brace Bushing, Part Number 8-16511.

5-34. Rework/replace Main Landing Gear Side Brace Bushing as per instructions outlined in figure 5-16.

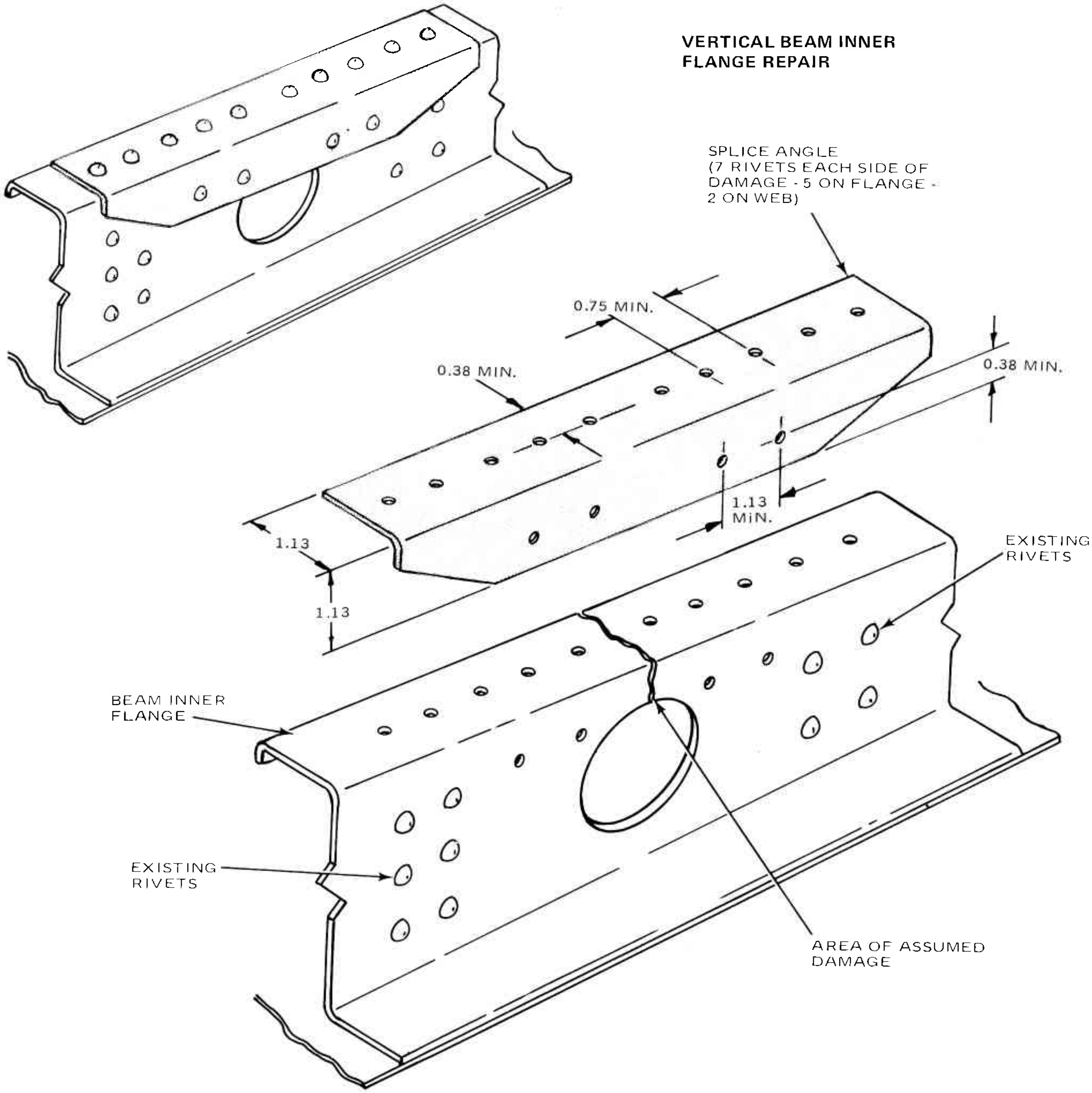


REPAIR PART	GAGE
SPLICE ANGLE	0.080
STIFFENER ANGLE	SAME AS STIFFENER
FILLER "A"	0.080
FILLER "B"	SAME AS CLIP LEG
FILLER "C"	0.080 MINUS STIFFENER GAGE

REPAIR PROCEDURE:

A. REMOVE MAIN LANDING GEAR DOOR FAIRING AND PLACE ON PADDED WORKBENCH.
B. REMOVE CLIPS FROM DAMAGED AREA USING A NO. 10 (0.193) DRILL TO REMOVE BEAM RIVETS AND A NO. 20 (0.161) DRILL FOR STIFFENER RIVETS.
C. USE NO. 20 (0.161) DRILL TO REMOVE RIVETS FROM ENDS OF STIFFENER FLANGES.
D. TRIM STIFFENERS AS NECESSARY TO ACCOMMODATE SPLICE ANGLE.
E. FABRICATE NEW CLIPS FROM 7075-T6 EXTRUSIONS. SEE FIGURE 5-6, MLG FAIRING STRUCTURE & PLATING, FOR EXTRUSION IDENTIFICATIONS. REHEAT TREAT TO W (TEMPORARILY UNSTABLE) CONDITION BEFORE FORMING.
F. FABRICATE SPLICE ANGLE, STIFFENER ANGLES, AND FILLERS FROM 7075-T6 CLAD. SEE CHART FOR GAGES. REHEAT TREAT TO W (TEMPORARILY UNSTABLE) CONDITION BEFORE FORMING ANGLES.
G. ARTIFICIALLY AGE CLIPS & ANGLES TO RETURN THEM TO T6 CONDITION.
H. POSITION SPLICE ANGLE AND DRILL NO. 10 (0.193) HOLES THROUGH BEAM WEB AND ANGLE. PICK UP EXISTING HOLES AT CLIP LOCATIONS. DRILL NO. 20 (0.161) HOLES THROUGH SKIN AND LOWER FLANGE OF ANGLE.
I. COUNTERSINK RIVET HOLES IN SKIN.
J. INSTALL SPLICE ANGLE USING MS20470DD6 AND MS20426AD5 RIVETS. DO NOT RIVET AT CLIP HOLES.
K. POSITION NEW CLIPS AND FILLERS AND DRILL USING NO. 10 (0.193) DRILL ON BEAM FACE AND NO. 20 (0.161) DRILL ON STIFFENER LEG. PICK UP ALL EXISTING CLIP HOLES.
L. RIVET CLIPS AND FILLERS IN PLACE USING MS20470DD6 AND MS20470AD5 RIVETS. DO NOT RIVET AT STIFFENER ANGLE HOLES.
M. POSITION STIFFENER ANGLES AND FILLERS AND DRILL NO. 20 (0.161) HOLES. PICK UP EXISTING HOLES IN SKIN. USE IDENTICAL PATTERN IN STIFFENER WEB.
N. RIVET STIFFENER ANGLES AND FILLERS IN PLACE USING MS20426AD5 AND MS20470AD5 RIVETS.

Figure 5-12. Main Landing Gear Fairing Vertical Beam Repair. (Sheet 1 of 3)



REPAIR PROCEDURE:

A. REMOVE MAIN LANDING GEAR FAIRING AND PLACE ON PADDED WORK BENCH.
B. FABRICATE SPLICE ANGLE FROM 0.90 GAGE 7075-T6 CLAD. REHEAT TREAT TO W (TEMPORARILY UNSTABLE) CONDITION BEFORE FORMING.
C. ARTIFICIALLY AGE TO RETURN SPLICE ANGLE TO T6 CONDITION.
D. POSITION SPLICE ANGLE OVER DAMAGE AREA AND DRILL NO. 10 (0.193) HOLES THROUGH ANGLE AND BEAM. COUNTERSINK 4 HOLES ON TRIMMED SIDE OF ANGLE.

NOTE

IF EXISTING RIVET HEADS INTERFERE WITH PLACEMENT OF REPAIR ANGLE, REMOVE 1 OR 2, AS NECESSARY, AND REPLACE WITH MS20426DD6 COUNTERSUNK RIVETS.

E. RIVET SPLICE ANGLE IN PLACE USING MS20426DD6 AND MS20470DD6 RIVETS.

Figure 5-12. Main Landing Gear Fairing Vertical Beam Repair. (Sheet 2 of 3)

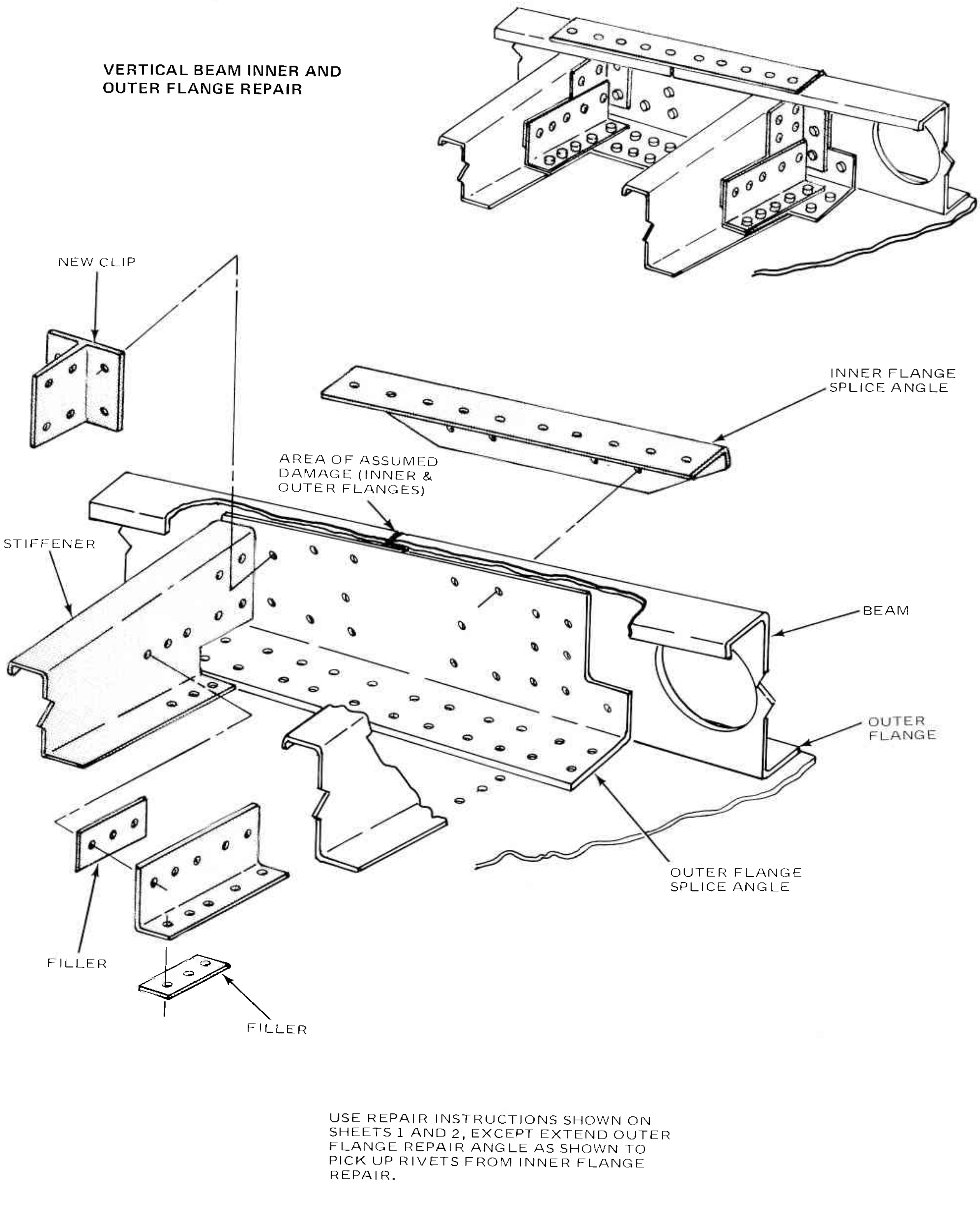
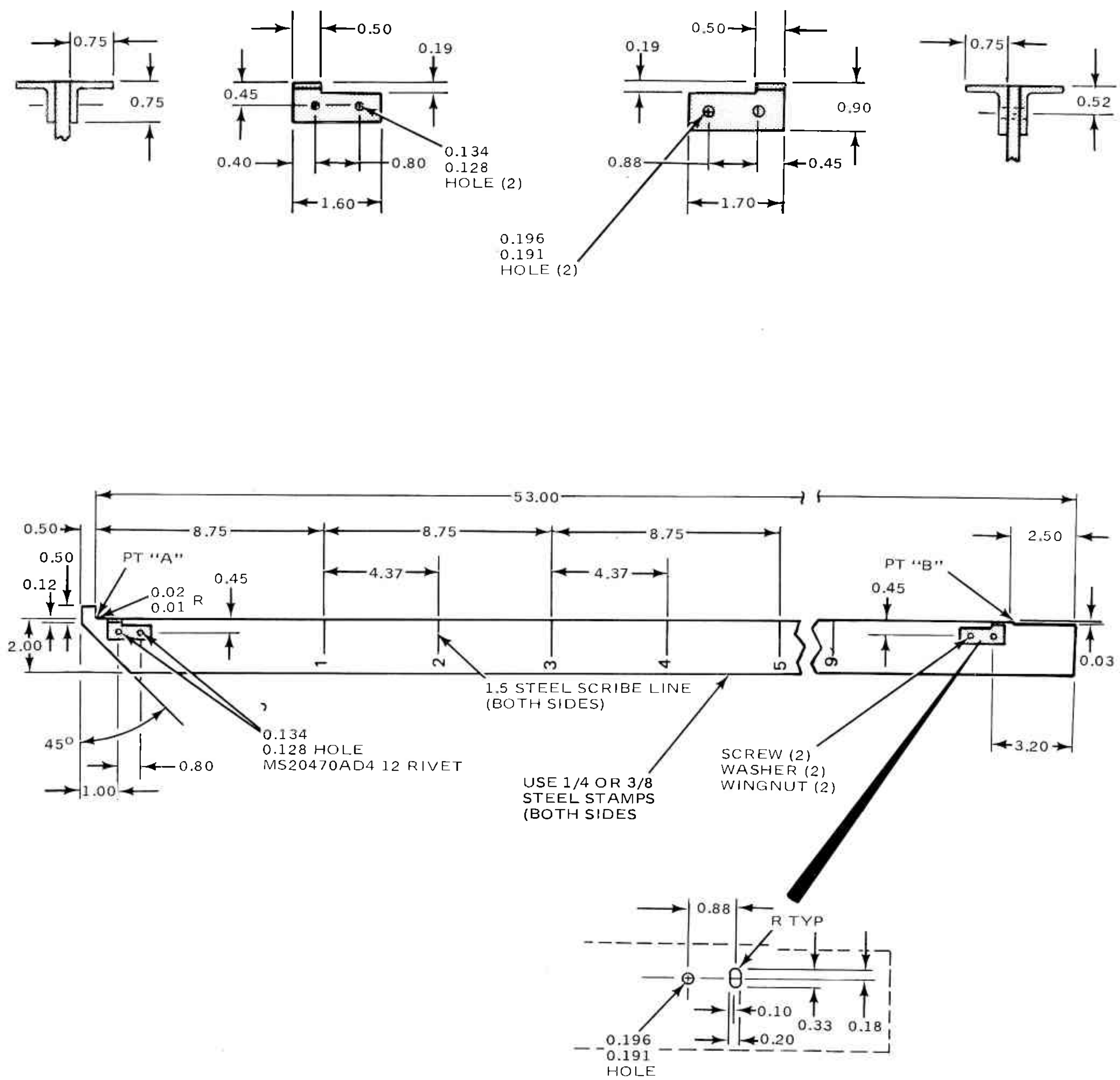


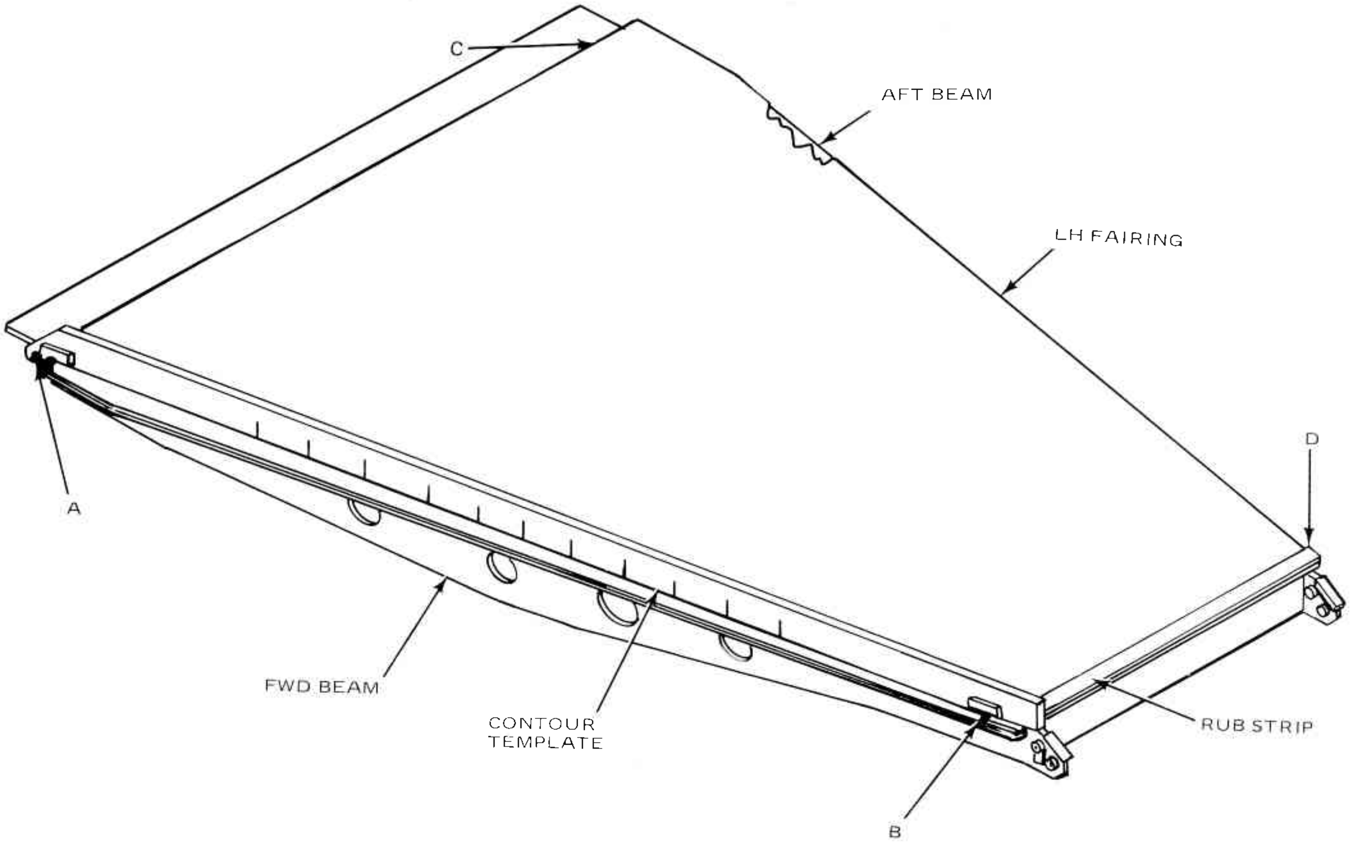
Figure 5-12. Main Landing Gear Fairing Vertical Beam Repair. (Sheet 3 of 3)

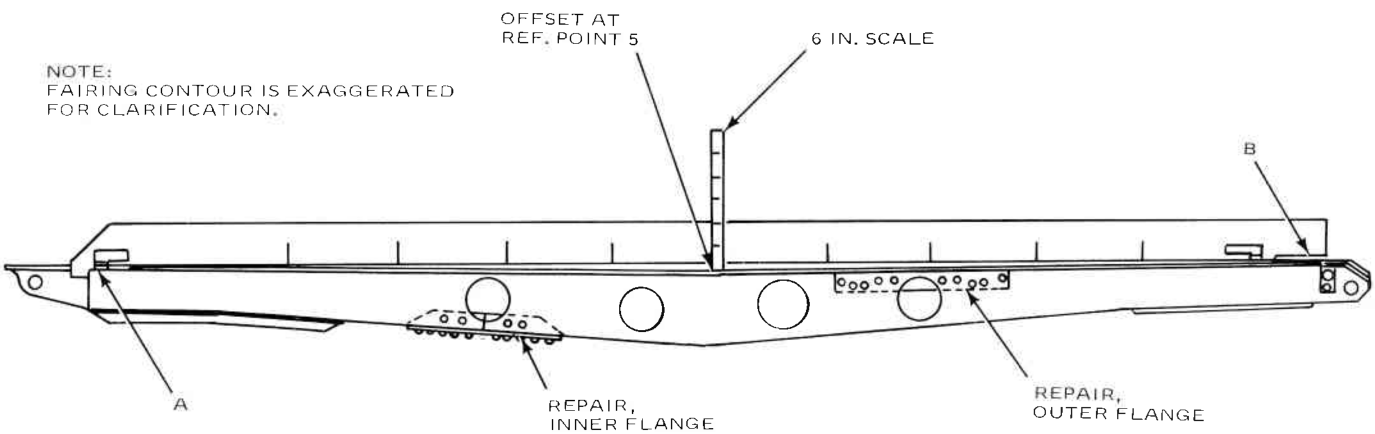


THIS CONTOUR CHECK TEMPLATE MAY BE MANUFACTURED LOCALLY.
MANUFACTURING INSTRUCTIONS:

1. MAKE FROM ALUMINUM ALLOY.
2. HARDWARE SHOWN IS SUGGESTED ONLY, USE EQUIVALENT IF NECESSARY.
3. MINIMUM THICKNESS OF TEMPLATE MATERIAL IS 0.18 IN.
4. EDGE SURFACE BETWEEN POINTS "A" AND "B" IS TO BE STRAIGHT WITHIN ± 0.01 IN. DO NOT ROUND OFF CORNERS.
5. IDENTIFY TEMPLATE WITH RUBBER STAMP OR STEEL STAMP. SIZE OF LETTERS AND NUMBERS AND LOCATION IS OPTIONAL. "MLG FAIRING CONTOUR CHECK TEMPLATE, P/N 8-17630-93."

Figure 5-13. Main Landing Gear Fairing Beams Contour Check Template





AFTER MAIN LANDING GEAR FAIRING BEAM REPAIR, CHECK FAIRING CONTOUR USING CONTOUR CHECK TEMPLATE, P/N 8-17630-93. IF ALLOWABLE OFFSETS ARE EXCEEDED, FAIRING MUST BE REPLACED.

1. POSITION CHECK TEMPLATE OVER FORWARD BEAM. HOOK LIP OF TEMPLATE OVER UPPER EDGE OF BEAM OUTER FLANGE (POINT A); THE OPPOSITE END OF TEMPLATE WILL REST ON RUB STRIP (POINT B). ADJUST STABILIZING ANGLE, IF NECESSARY, TO ASSURE FIRM, VERTICAL POSITION.
2. USING A 6-INCH STEEL MACHINIST SCALE, OR SIMILAR, MEASURE OFFSET DISTANCE BETWEEN FAIRING SURFACE AND CONTOUR TEMPLATE AT EACH NUMBERED POSITION. RECORD VALUES ON OFFSET GRAPH.
3. POSITION CONTOUR TEMPLATE OVER AFT BEAM (POINTS C AND D) AND REPEAT OFFSET CHECKS.

Figure 5-14. Main Landing Gear Fairing Contour Check

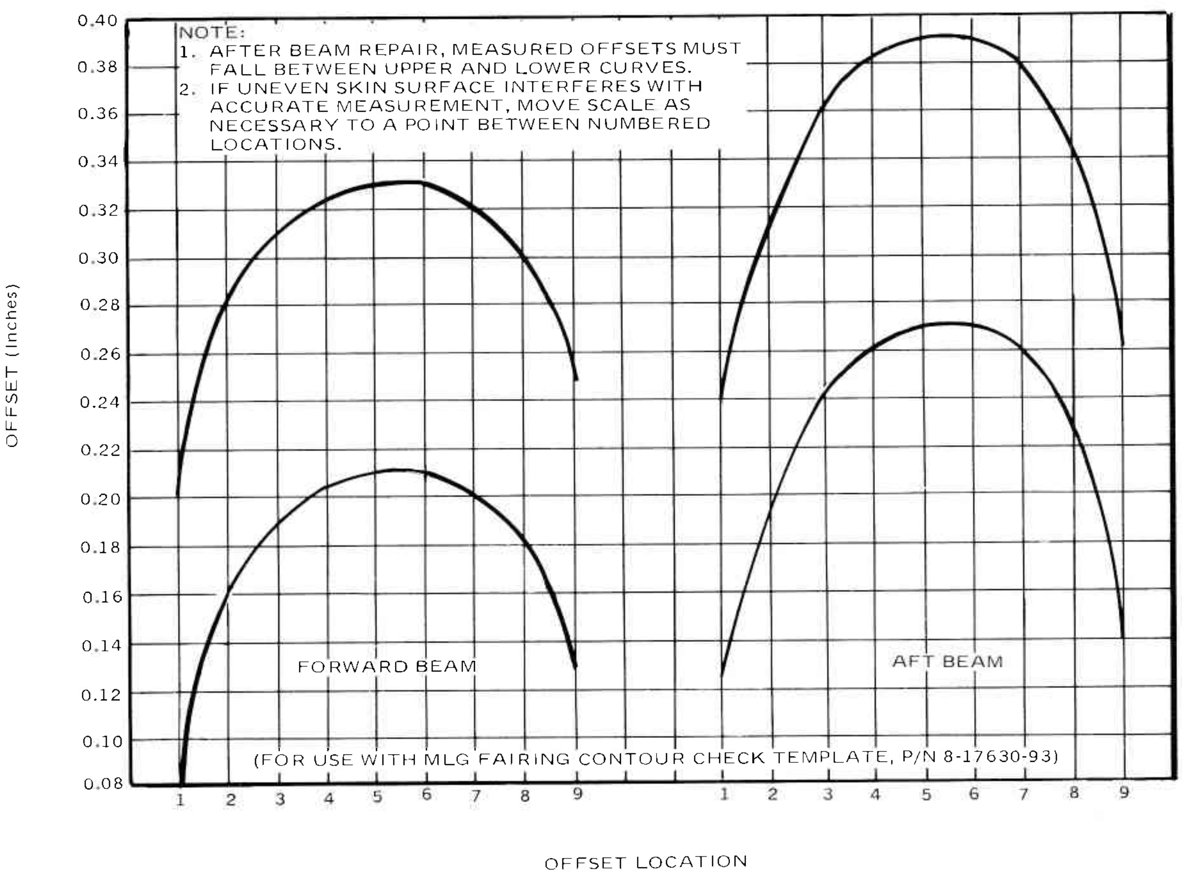
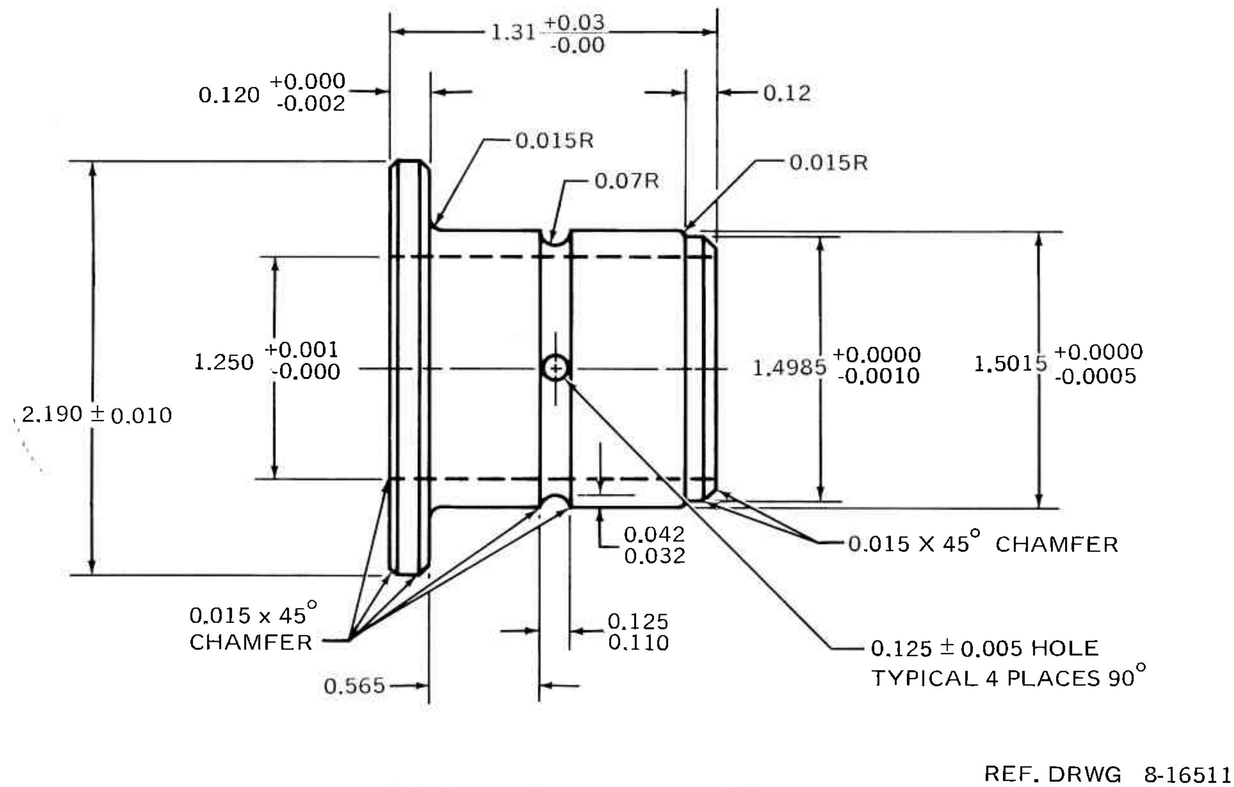


Figure 5-15. Main Landing Gear Fairing Beams Allowable Offset Graph



BUSHING REPLACEMENT DATA

ATTACHMENT BUSHING	ORIGINAL HOLE SIZE IN ATTACHMENT FITTING	MAX. ALLOWABLE HOLE OVERSIZE	PIN, BOLT OR SHAFT DIAMETER	**INTERFERENCE FIT
LANDING GEAR SIDE BRACE	1.5000 + 0.0005 - 0.0000	1.6200 + 0.0005 - 0.0000	1.248 TO 1.249	- 0.0005 - 0.0015

NOTES:

1. MAKE BUSHING FROM SAE4130 OR SAE8630 STEEL, HEAT TREAT TO 180,000 TO 200,000 PER MIL-H-6875.

2. CADMIUM PLATE ALL BUSHING SURFACES EXCEPT THE INSIDE DIAMETER PER SPECIFICATION QQ-P-461. DIMENSIONS TO BE MET AFTER PLATING.

3. BUSHING SIDES TO BE PARALLEL AND CONCENTRIC WITHIN 0.003 INCH TOTAL INDICATOR READING.

4. BUSHING TO RECEIVE MAGNETIC INSPECTION PER MIL-I-6868. (T.O. 33B-1-1)

5. IF ORIGINAL HOLE SIZE IS WITHIN TOLERANCE, BRUSH PLATING MAY BE APPLIED TO BUSHINGS, IF AVAILABLE, TO OBTAIN PROPER INTERFERENCE FIT, PLATING IS TO BE CADMIUM.

**INTERFERENCE FIT BETWEEN OUTER DIAMETER OF BUSHING AND INNER DIAMETER OF MAIN LANDING GEAR SIDE BRACE FITTING.

Figure 5-16. Main Landing Gear Side Brace Bushing Repair

Section VI

ENGINE

6-1. ENGINE.

6-2. The aft section of the fuselage between stations 431.00 and 711.00 is designed to accommodate the engine. An engine cooling air induction system extends aft from the scroll assembly at station 431.00. The engine is enclosed by a detachable shroud and insulation blankets which provide a fireseal and air cooling space. The engine is supported in the fuselage by three support links and one thrust bearing. A nose fairing attaches to the forward face of the engine compressor. Several access doors are located in the fuselage and shroud to provide for inspection and maintenance of the engine and engine accessories. See figure 6-1 for engine components covered in this section.

6-3. Engine Cooling Shrouds.

6-4. The construction of the engine shroud consists generally of a series of circular formed frames enveloped with light gage rigidized skins which are reinforced by longitudinal longerons. Monel riveting and seam welding is used to secure the components together. The shroud slips forward over the aft portion of the engine and is held in place by means of turnbuckles attached to brackets on the engine. The engine and shroud are supported in the fuselage by three support links. The forward end of the shroud incorporates a seal around the entire forward edge which fits against the engine seal adapter to form the fireseal. Insulation blankets consisting of fiberglass batting enveloped by thin gage corrosion resistant foil are installed inside the engine shroud. The engine shroud incorporates a convergent-divergent nozzle permanently attached to the aft of the shroud except on airplanes listed on figure 6-2. Figure 6-2 shows the engine shroud.

6-5. Engine Nose Cone Fairing.

6-6. The engine nose cone fairing is connected to the center of the engine by eight bolts which pass through the extruded ring assembly at the aft end of the nose cone and into the forward face of the engine compressor. The engine nose cone fairing consists of inner and outer stressed aluminum alloy skins flush-riveted and seam-welded to an inner structure of aluminum alloy stiffeners and formers. See figure 6-3 for the material specifications and gages used in the nose cone fairing.

6-7. Engine Supports.

6-8. The engine is supported in the fuselage by two adjustable support links at the aft end of the engine, one engine thrust bearing at the forward right side, and one nonadjustable engine support link at the forward left side.

6-9. REPAIRS.

6-10. Engine Cooling Shrouds.

6-11. Damage to the engine cooling shroud, such as cracks, holes, gaps between shroud seal and firewall, or other more serious damage which is likely to create a fire hazard, will require repair or replacement of parts involved. Monel rivets shall be used in all repairs unless otherwise directed by an aeronautical engineer. Refer to T.O. 1F-106A-2-4-2-1 for removal installation procedures for engine and engine cooling shroud. See figure 6-4 for repair of shroud frames, longerons and rigidized skin. HT-1 Sealer (8030-515 2230 Local Purchase) is used as a cushioning agent and sealer when patching cracks which occur along seamwelds. See figure 6-5 for repair procedure. Gaps between the shroud seal and firewall should be closed by wedging the shroud seal against the firewall. Figure 6-6 outlines wedging procedure. Refer to Table 6-1 for negligible damage limits. Shroud insulation blanket damage such as pinholes, cracks, and tears in skin of blanket must, with few exceptions, be repaired to prevent absorption of fuel. Evaluate possible repair or replacement, observing these precautions:

a. Within an arc of 15 degrees above each side of bottom center line (blanket installed), all cracks, pinholes and tears in skin of blanket must be repaired.

b. Above lower limiting area, pinholes of under 0.032-inch diameter may be left open if not closer together than 12 inches and not more than 12 holes in each blanket.

c. The size of a repair and the number of repairs permitted are limited only by the structural integrity of the shroud blanket and good maintenance practices.

Blanket skins must be repaired by light pokewelding 0.002 inch to 0.004 inch thick 321 or 347 corrosion resistant foil patches over the damaged area. Patches must be large enough to allow a full overlapping spotweld pattern with a minimum of 0.25-inch edge distance all

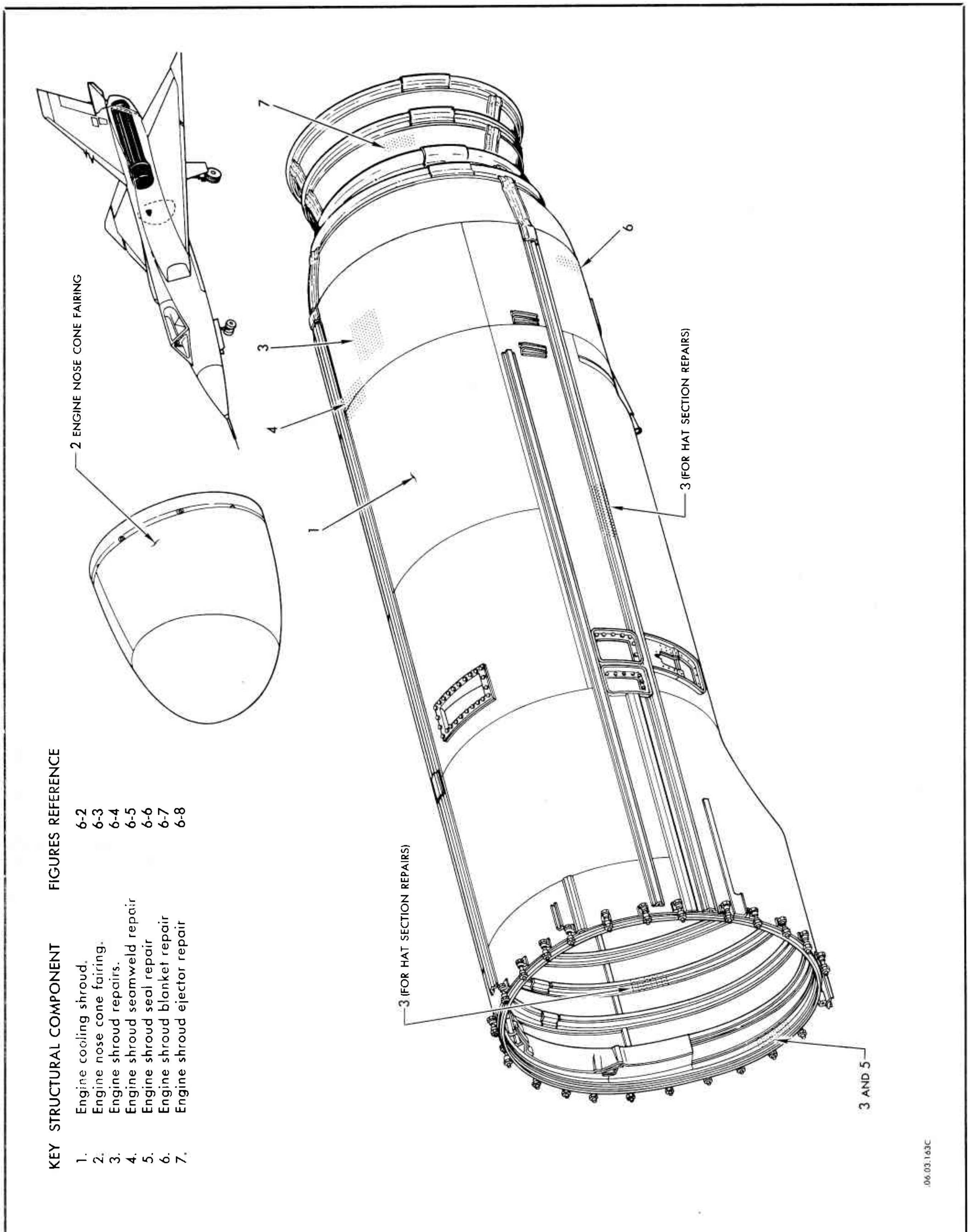
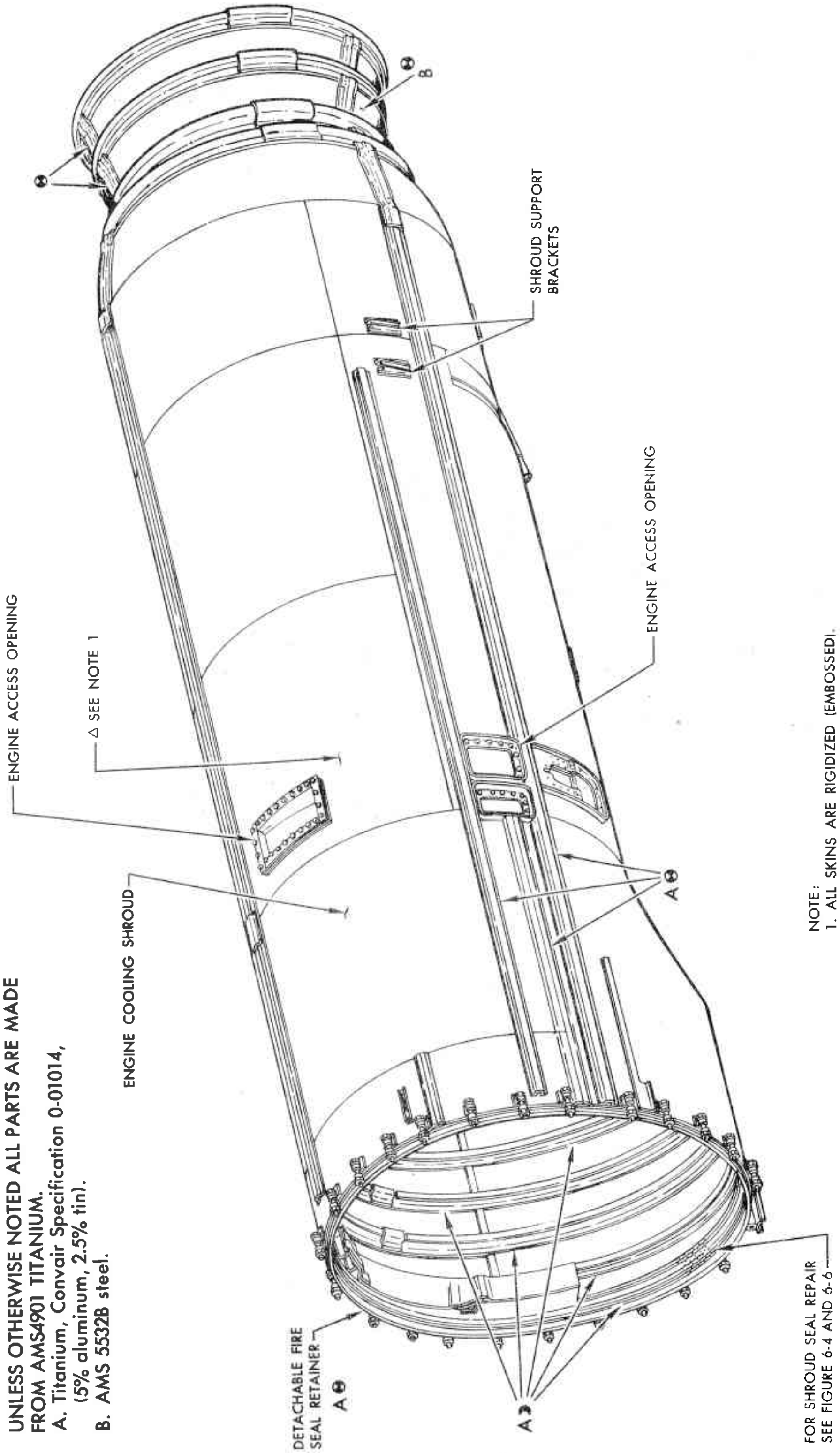


Figure 6-1. Engine Section Components and Figure Index



MATERIAL

UNLESS OTHERWISE NOTED ALL PARTS ARE MADE FROM AMS4901 TITANIUM.

A. Titanium, Convair Specification 0-01014, (5% aluminum, 2.5% tin).

B. AMS 5532B steel.

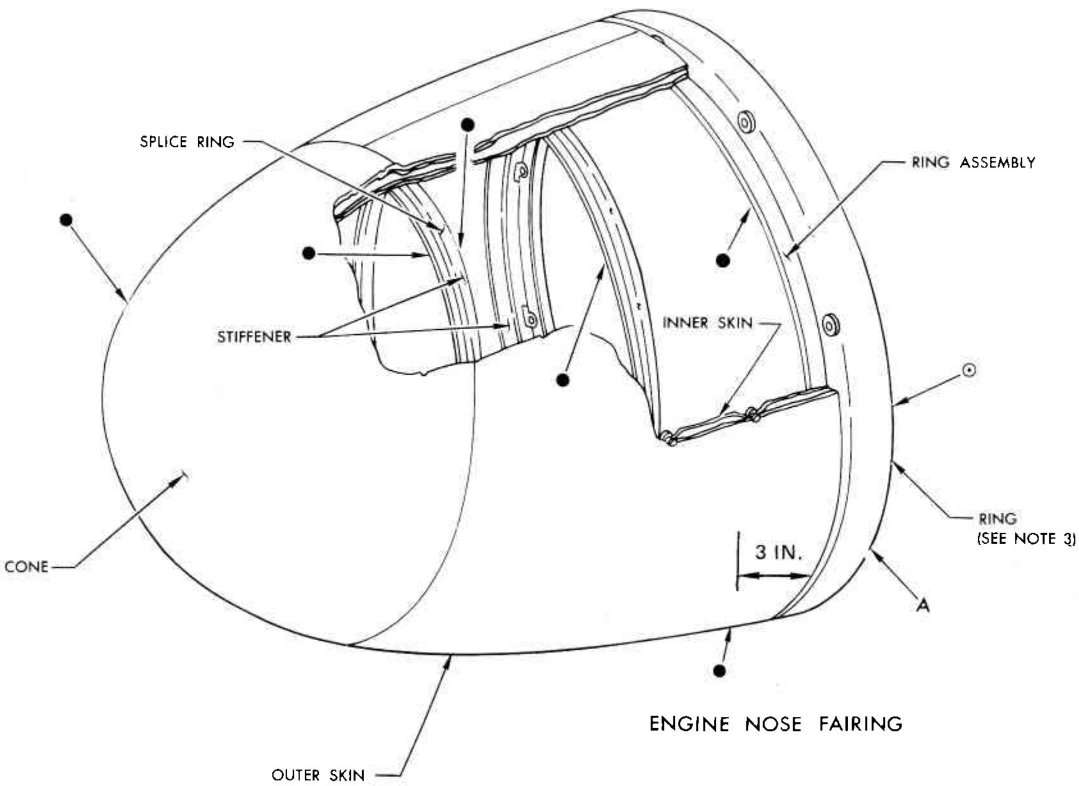
NOTE:
1. ALL SKINS ARE RIGIDIZED (EMBOSSED).

SYMBOL	GAGE
△	0.012
⊖	0.025
◗	0.040

Figure 6-2. Engine Shroud

MATERIAL
UNLESS OTHERWISE NOTED, ALL COMPONENTS ARE MADE FROM 2024-T6 UNCLAD.
A. 6061-T6 extrusion.

SYMBOL	GAGE
●	.032
⊙	.625

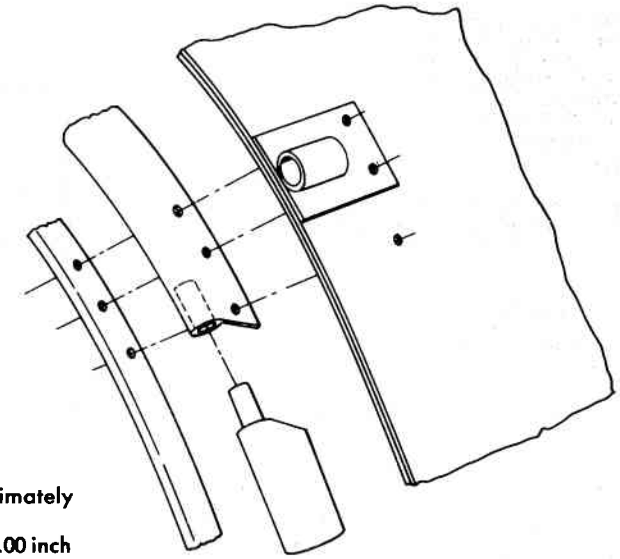


ENGINE NOSE FAIRING

NOTES:
1. REFER TO THIS SECTION FOR ENGINE SECTION NEGLIGIBLE DAMAGE LIMITS.
2. REFER TO TYPICAL SKIN REPAIRS IN SECTION X FOR FAIRING SKIN REPAIR INFORMATION.
3. REFER TO THIS SECTION FOR ENGINE NOSE CONE FAIRING REPAIR INFORMATION.
4. O & I REPAIR (FLUSH PATCH)IS LIMITED TO 3 INCHES FORWARD OF ATTACHING POINT. REPAIR IN ACCORDANCE WITH FIGURE 10-1.

06.03.187C

Figure 6-3. Engine Nose Cone Fairing



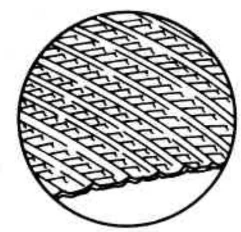
SEAL REPAIR

1. Remove stitches from cover for approximately 2-1/2 inches at ends of repair splice.
2. Cut covering material on bias leaving 1.00 inch of core material exposed. On mating seal, remove core material inside cover as illustrated.
3. Re-sew ends of material after splicing.

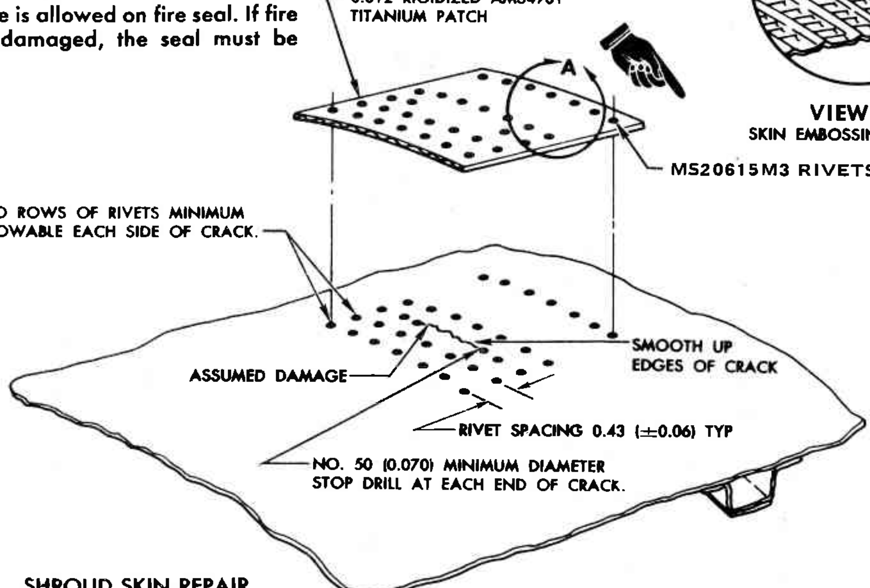
NOTE
SEAL MATERIAL IS TEFLON IMPREGNATED ASBESTOS TADPOLE SEAL WITH FULLY SINTERED INCONEL MEST CORE NO. RL285, MANUFACTURED BY RAYBESTOS MANHATTEN INC., MANHEIM, PA.

4. One repair splice is allowed on fire seal. If fire seal is further damaged, the seal must be replaced.

EMBOSSING ON PATCH SHALL NEST PROPERLY INTO EMBOSSING IN SKIN.



VIEW A
SKIN EMBOSSING DETAIL



SHROUD SKIN REPAIR

1. Stop drill ends of crack and clean up edges of crack.
2. Prepare patch of 0.012 rigidized titanium (Spec. AMS4901). Refer to Section I for formula and method for patch repair. Rivet pattern and minimum rivet edge distance to determine size of patch.
3. Install patch using MS20615M3 rivets.

NOTES:
1. TYPE 321 FLAT STAINLESS STEEL SHEET MAY BE USED AS A TEMPORARY SUBSTITUTE FOR AMS 4901 RIGIDIZED TITANIUM IN REPAIR PATCH. REFER TO PARAGRAPH ON MATERIAL SUBSTITUTIONS IN SECTION I. STAINLESS STEEL PART SHALL BE REPLACED AT NEXT DEPOT OVERHAUL.
2. IF OVERALL LENGTH OF ANY SINGLE CRACK EXCEEDS 8.0 INCHES, CONSULT AN AERONAUTICAL STRUCTURES ENGINEER FOR DISPOSITION OF DAMAGE.

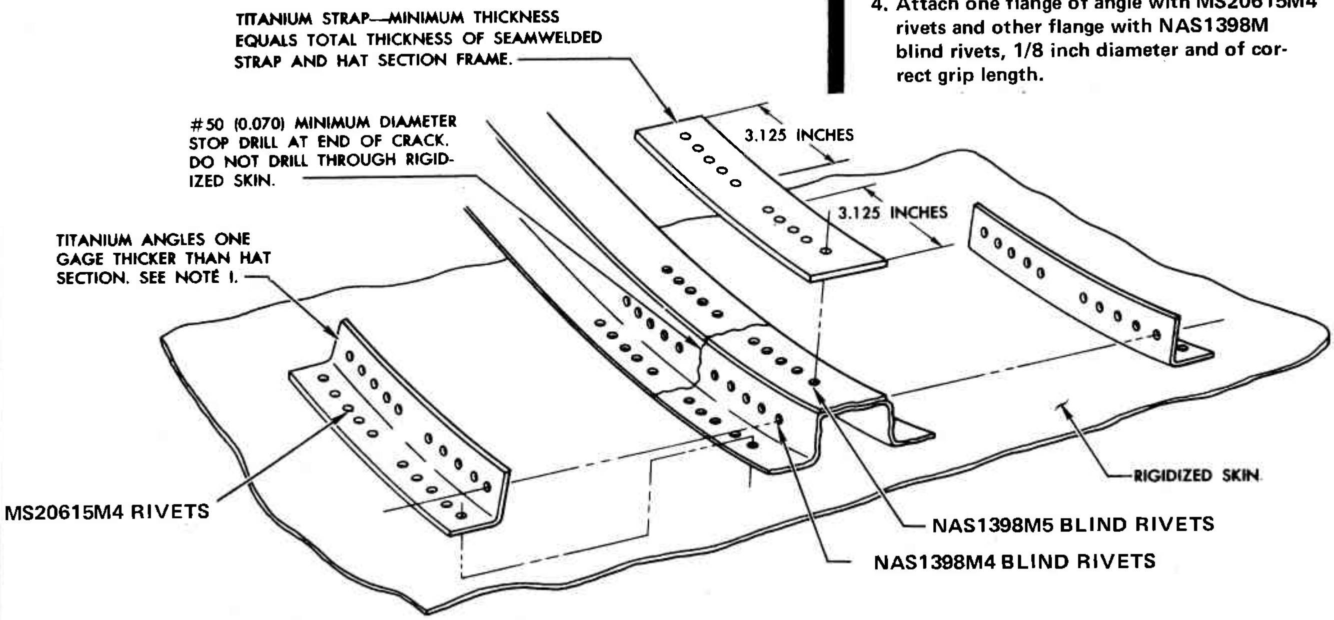
.06.03.188-1A

Figure 6-4. Engine Shroud Repairs (Sheet 1 of 2)

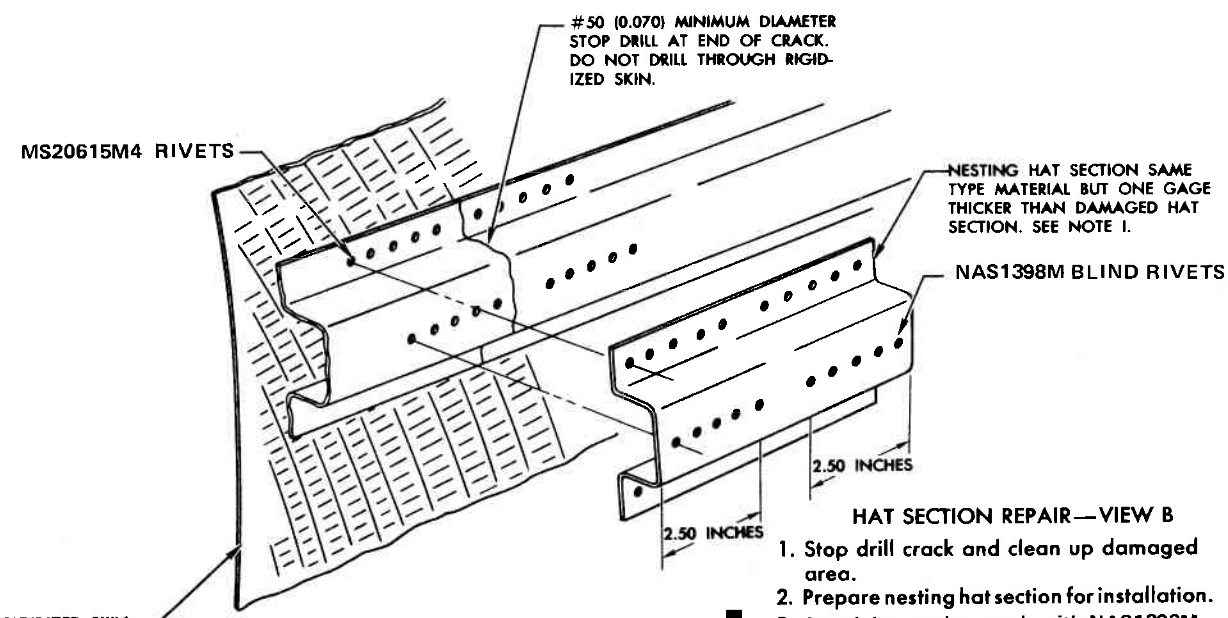
NOTES:
1. TYPE 321 STAINLESS STEEL SHEET MAY BE USED AS A TEMPORARY SUBSTITUTE FOR CONVAIR SPECIFICATION 0.01014 (5% ALUMINUM —2.5% TIN) TITANIUM IN HAT REPAIR. STAINLESS STEEL PART SHALL BE REPLACED AT NEXT DEPOT OVERHAUL.
2. ALTERNATE SUBSTITUTE FOR DUPONT BLIND RIVETS ARE NAS 1398M AND NAS 1399M.

HAT SECTION REPAIR—VIEW A

1. Stop drill crack and clean up damaged area.
2. Prepare angles and strap for installation.
3. Attach strap with NAS1398M blind rivets, 5/32 inch diameter and of correct grip length.
4. Attach one flange of angle with MS20615M4 rivets and other flange with NAS1398M blind rivets, 1/8 inch diameter and of correct grip length.



VIEW A



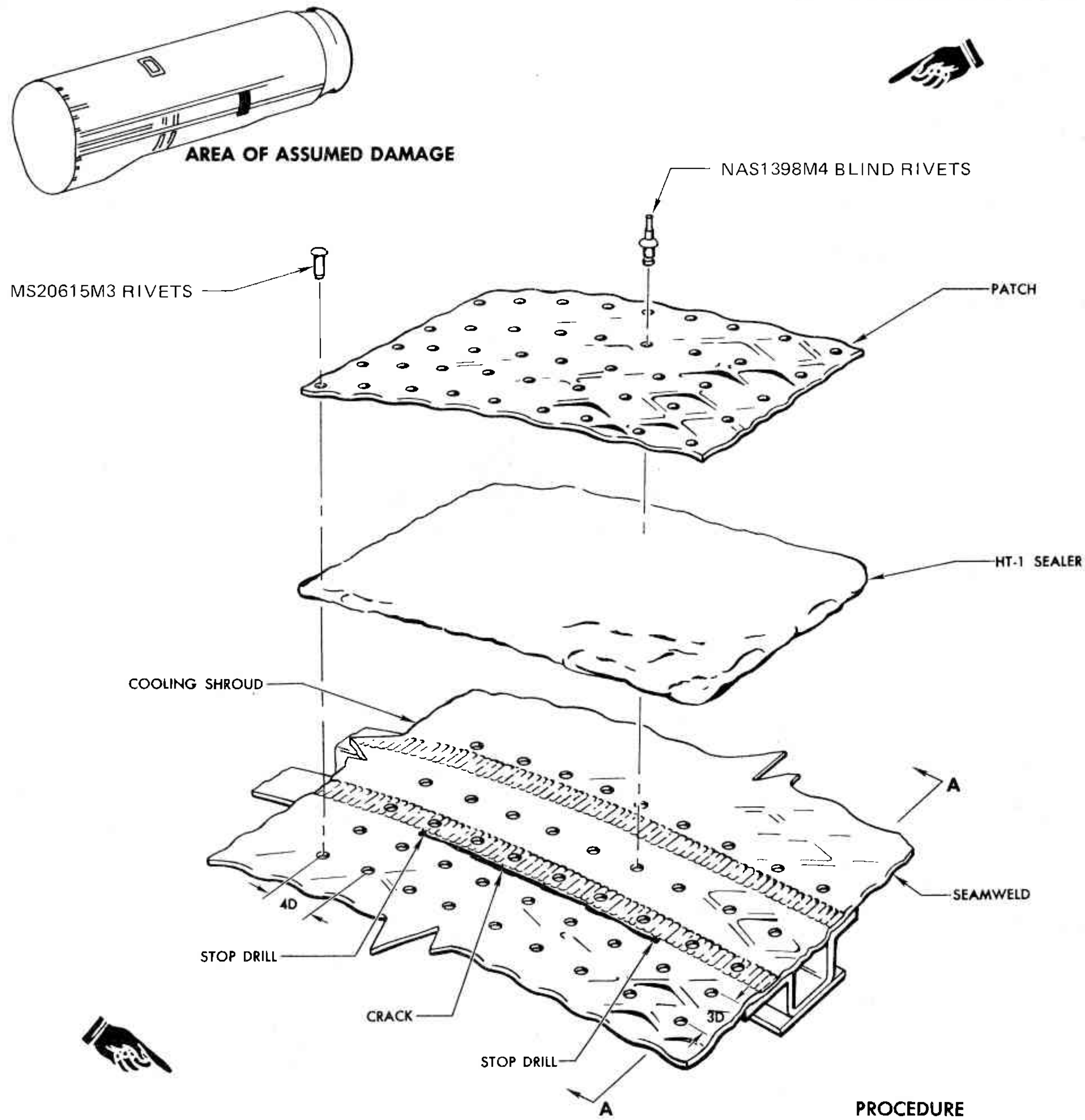
VIEW B

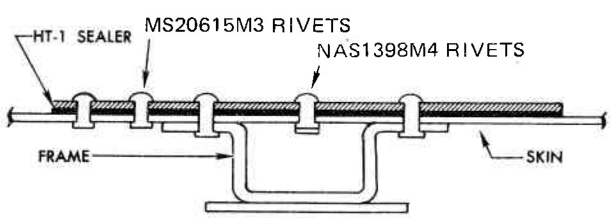
HAT SECTION REPAIR—VIEW B

1. Stop drill crack and clean up damaged area.
2. Prepare nesting hat section for installation.
3. Attach hat section repair with NAS1398M blind rivets, 5/32 inch diameter and 1/8 inch diameter NAS1398M. Attach flanges using MS20615M4 rivets.

.06.03.188-2A .72.75.17

Figure 6-4. Engine Shroud Repairs (Sheet 2 of 2)





SECTION A-A

PROCEDURE

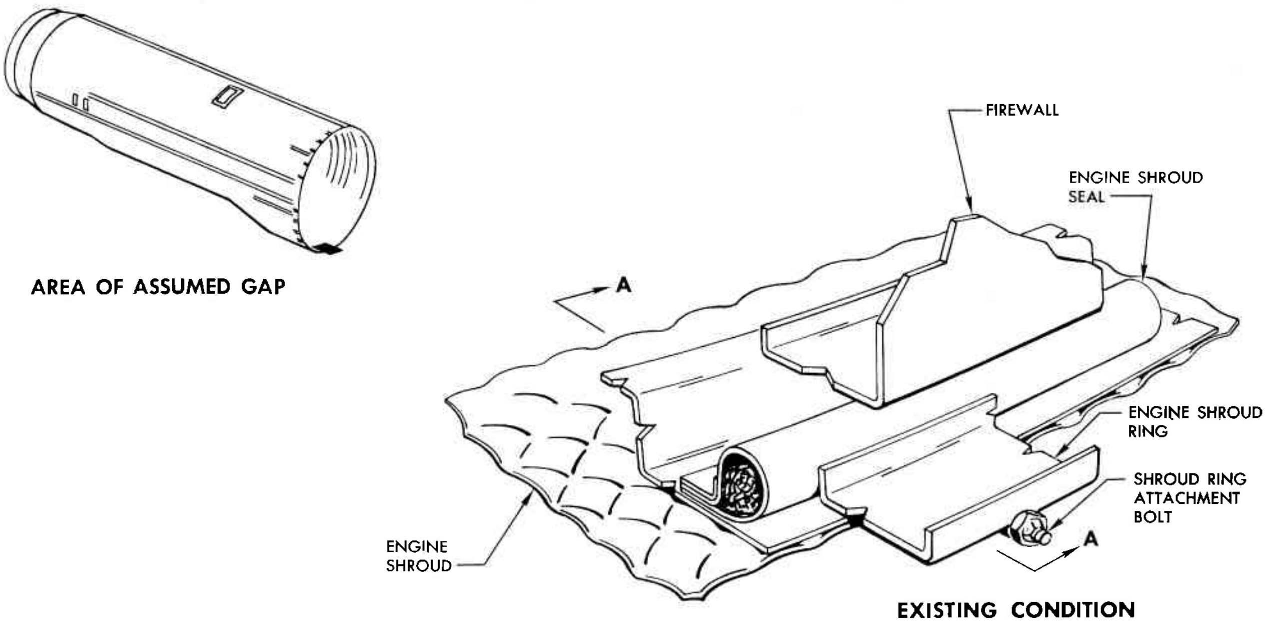
a. Stop drill ends of crack.

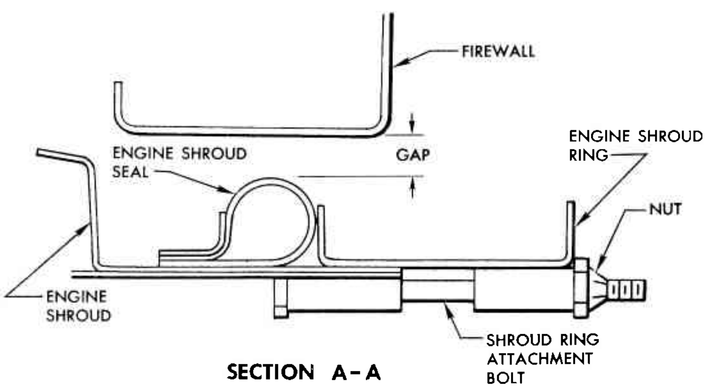
NOTE
THERE IS NO LIMIT ON LENGTH OF CRACK AS LONG AS GOOD REPAIR CAN BE ACCOMPLISHED.

b. Fabricate a patch of 0.012 titanium (AMS4901).
c. Rivet pattern and spacing to be as shown.
d. Clean area to be patched with Methyl Ethyl Ketone Federal Stock No. 6810-290-0055.
e. Shape a layer of HT-1 sealer (FSN 8030-515-2230) and place over patch area.
f. Rivet patch to shroud with NAS1398M4 rivets in blind areas and MS20615M3 rivets in accessible areas.
g. Remove excess HT-1 from patched area.

.06.03.266 .72.75.17

Figure 6-5. Engine Shroud Seamweld Repair





PROCEDURE

a. Remove shroud retainer ring retainer nuts in area where gap stops.
b. Fabricate wedges from 321 stainless steel, Federal Specifications MIL-S-6721 as shown.
c. Insert wedges between shroud ring and seal in area where gap exists.

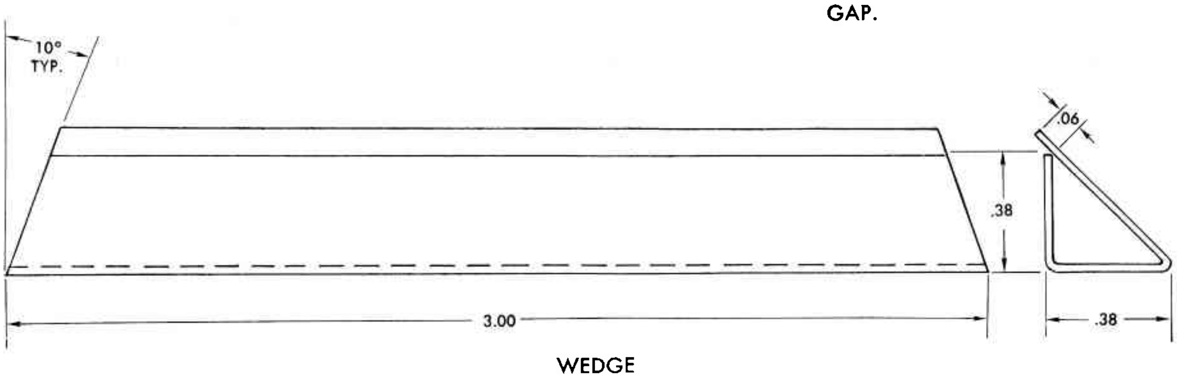
NOTE

THE NUMBER OF WEDGES USED WILL DEPEND ON THE LENGTH OF GAP TO BE CLOSED. WEDGES MUST BUTT TOGETHER TO FORM A SEAL.

d. Install retainer ring nuts and tighten enough to force seal against firewall.

CAUTION

DO NOT OVERTORQUE NUTS, SINCE THIS WILL TWIST SHROUD RING AWAY FROM WEDGES AND CAUSE ANOTHER GAP.



.06.03.275-1A

Figure 6-6. Engine Shroud Seal Repair (Sheet 1 of 2)

U.S. Government Printing Office 1981/771-060/2384

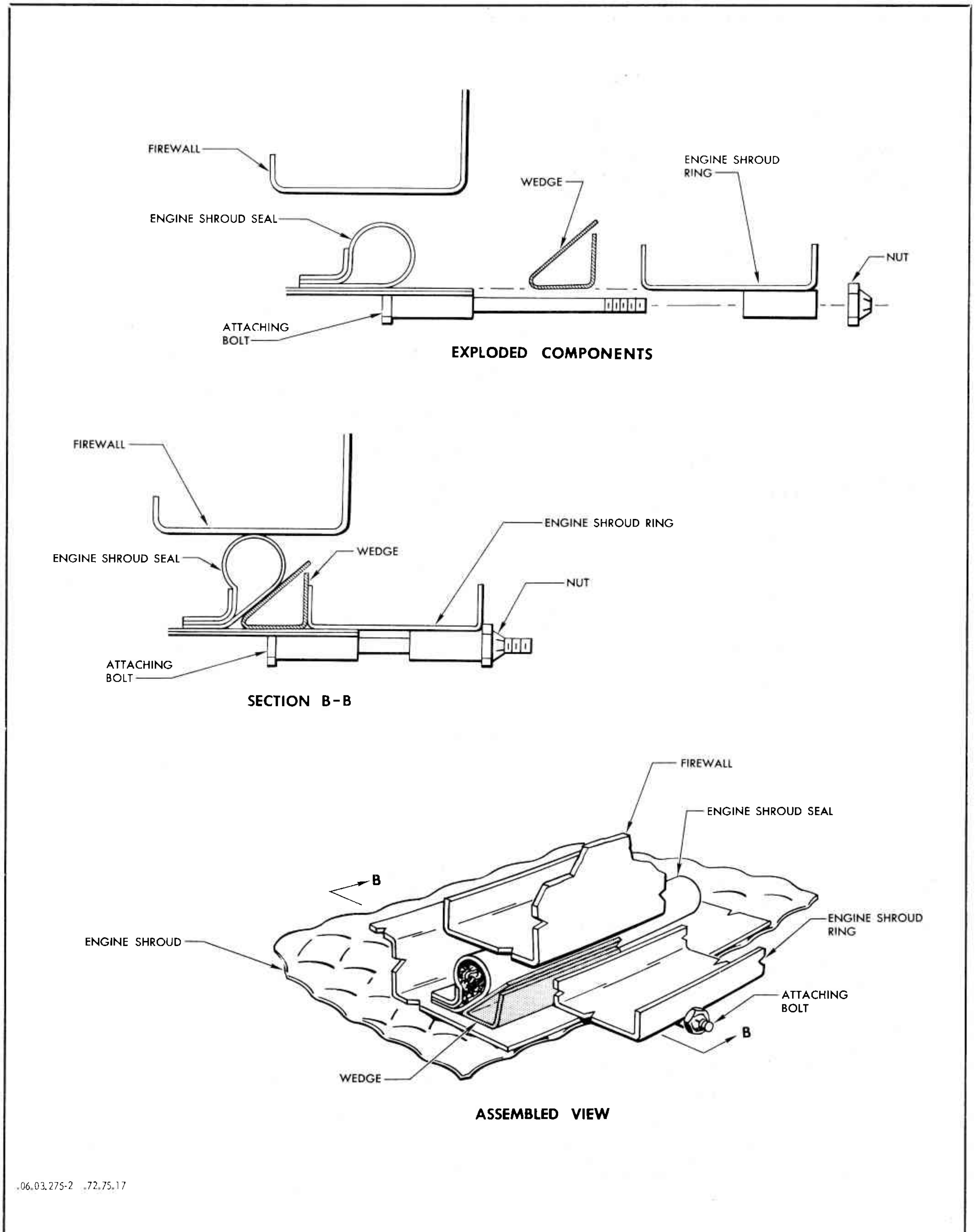
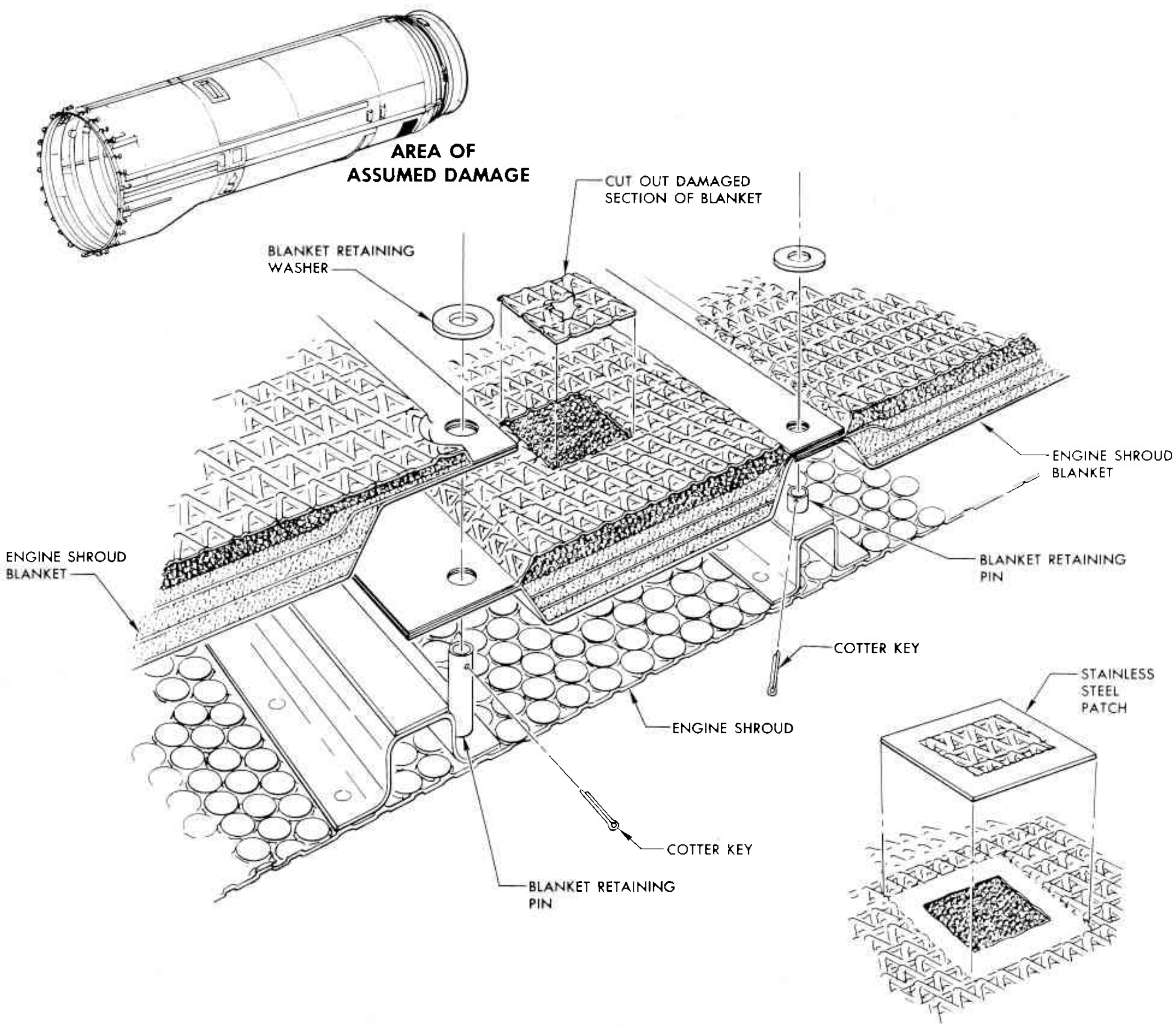


Figure 6-6. Engine Shroud Seal Repair (Sheet 2 of 2)



REPAIR PROCEDURE

a. Remove engine from airplane. Refer to T.O. 1F-106A-2-4-2-1 for engine removal procedure.
b. Remove engine shroud from engine. Refer to T.O. 1F-106A-2-4-2-1 for removal procedure.
c. Remove cotter keys and washers holding damaged shroud blanket in place. Hold washers for reinstallation.
d. Remove damaged shroud blanket.
e. Using metal cutting hand shears, cut out damaged section of shroud blanket skin.
f. Using a pair of pliers with smooth jaws, flatten edges of cutout in shroud blanket as shown.
g. Fabricate a patch of 0.004 stainless steel foil.

NOTE

"HIT KIT" SPOTWELDER IS MANUFACTURED BY H. I. THOMPSON CO., 1733 CORDOVA STREET, LOS ANGELES 7, CALIFORNIA.

h. Place patch over cutout in shroud blanket.
i. Plug "Hit Kit" spotwelder into 110-volt ac socket and place control switch in on position. "Hit Kit" power plug is the three prong twist type and must be inserted in the same type of electrical socket.
j. Place electrode with microswitch button in one hand and smaller tube shaped electrode in opposite hand.

NOTE

THE SMALL UNIT IS THE WELDING ELECTRODE. THE UNIT WITH MICROSWITCH IS THE GROUND ELECTRODE.

k. Place welding electrode firmly on corner of patch with ground electrode on opposite corner of patch.
l. Depress microswitch button while holding electrodes firmly against patch.
m. Release microswitches button; move welding electrode slightly and depress microswitch again. Proceed around edge of patch, placing welds close enough together to create a liquid tight seal.
n. Reinstall blanket in engine shroud using washers removed in step c and new cotter keys to secure blanket in place.
o. Reinstall engine shroud and reinstall engine. Refer to T.O. 1F-106A-2-4-2-1 for installation procedures.

.06.03.261-1 .72.75.03

Figure 6-7. Engine Shroud Blanket Repair (Sheet 1 of 2)

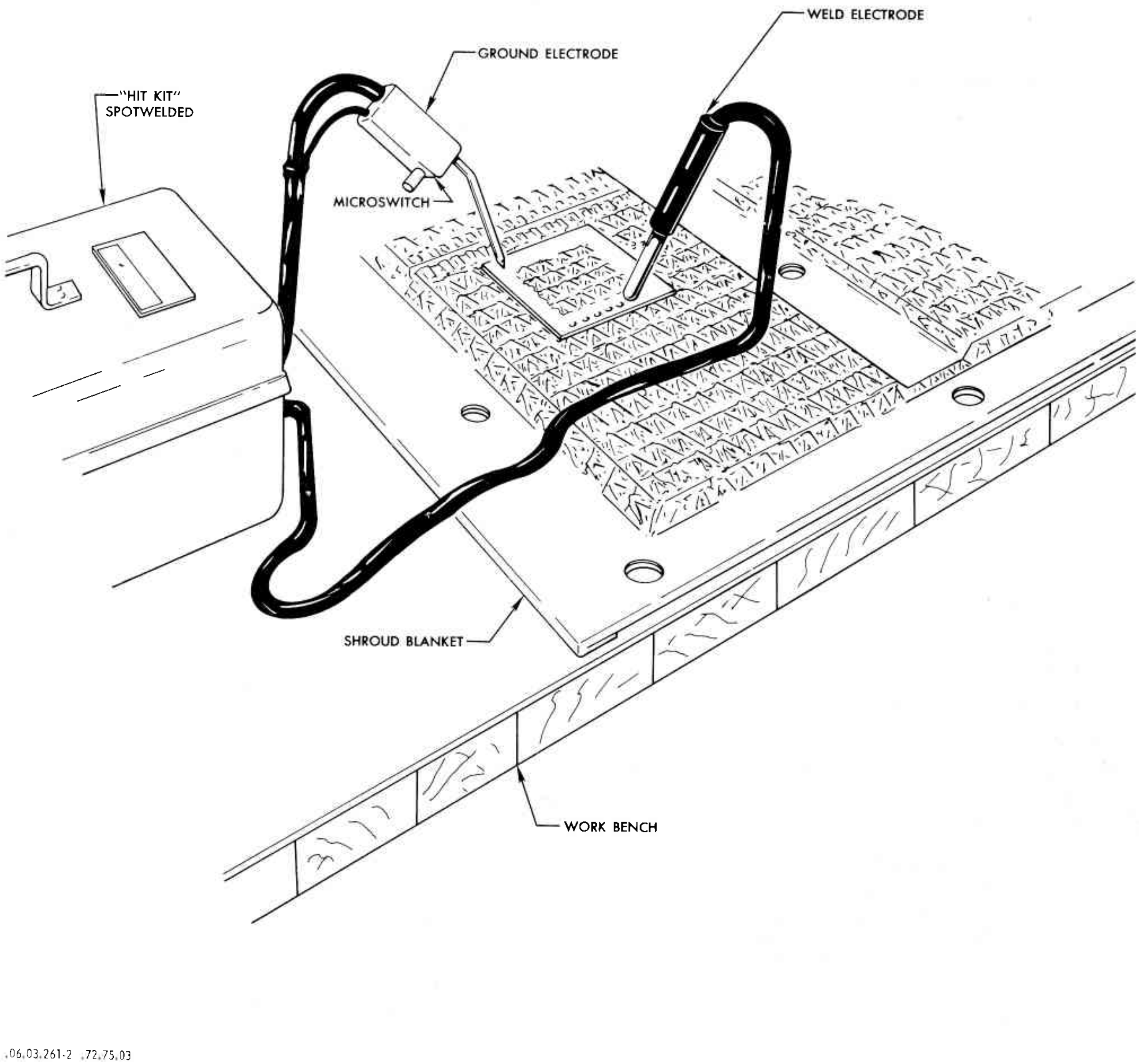
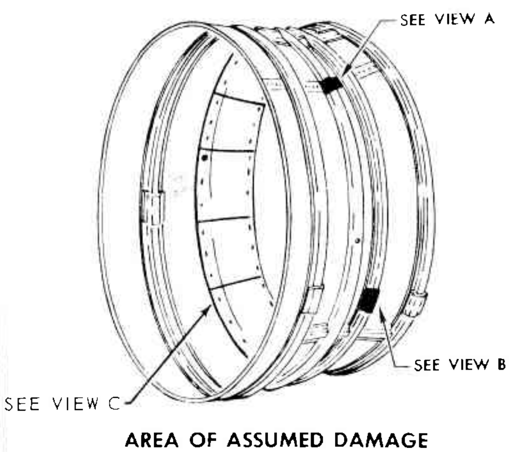
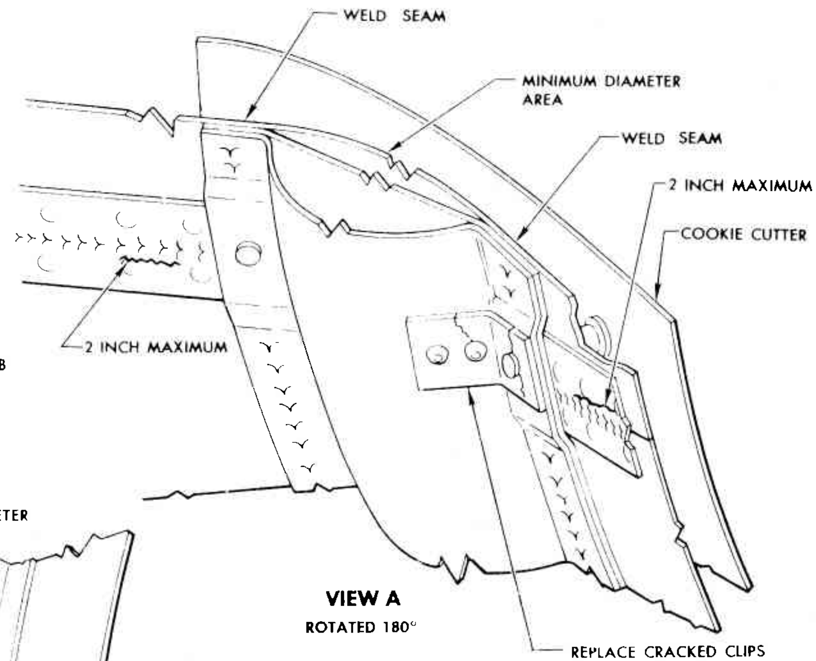
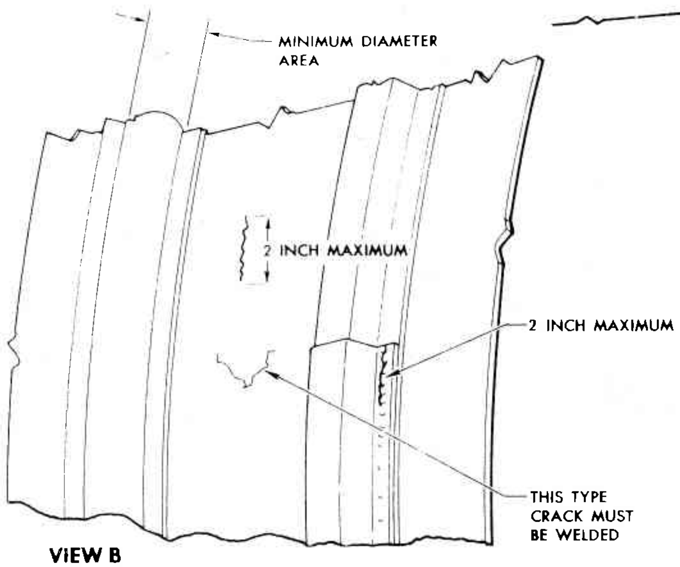


Figure 6-7. Engine Shroud Blanket Repair (Sheet 2 of 2)







CRACK LIMITS:

Cracks in Views A and B which do not exceed 2 inch in length must be stop drilled with a 0.090 inch drill. Cracked clips shown in View A must be replaced. Any crack which exceeds 2 inch in length or changes direction as shown in View B must be welded.

Cracks in the minimum diameter area, between weld seams under cookie cutter, (View A), may be stop drilled when they are parallel to the longitudinal axis and do not extend into weld seams. Cracks in this area, that are not parallel to the longitudinal axis or that extend into the weld seams must be welded.

NOTE

CRACKS RESULTING FROM POOR-QUALITY WELDS MAY BE REPAIRED BY REWELDING PROVIDED ALL OF POOR QUALITY WELD IS REMOVED AND ALL SURFACE CRACKS ARE REMOVED BY GRINDING. IF MORE THAN 50% OF ORIGINAL MATERIAL THICKNESS IS REMOVED DURING GRINDING, MAKE RADIUS CORNERED CUTOUT OF A LENGTH AND WIDTH WHICH IS ¼" GREATER THAN HEAT AFFECTED AREA AND MAKE A FLUSH PATCH INSERT WELD REPAIR.

WELDING PROCEDURE:

a. Perform a fluorescent penetrant inspection to determine length of crack.
b. Stop drill both ends of crack. Use a 0.090 inch drill.
c. Clean crack area with a stainless steel wire brush.
d. Use a Heliarc welding torch and N-155 welding wire that conforms with Federal Specification MIL-R-5031, Class 9 to weld crack.

INSERT REPLACEMENT:

Replace damaged inserts as per View C. To compensate for any variance that may be encountered, the 8-26054-8 inserts are supplied with 1/8 inch excess material for trim to maintain the specified gap between inserts. Trim excess material from the 8-26054-7 inserts to maintain the specified 0.16 inch gap; for any additional trim, use a suitable grindstone.

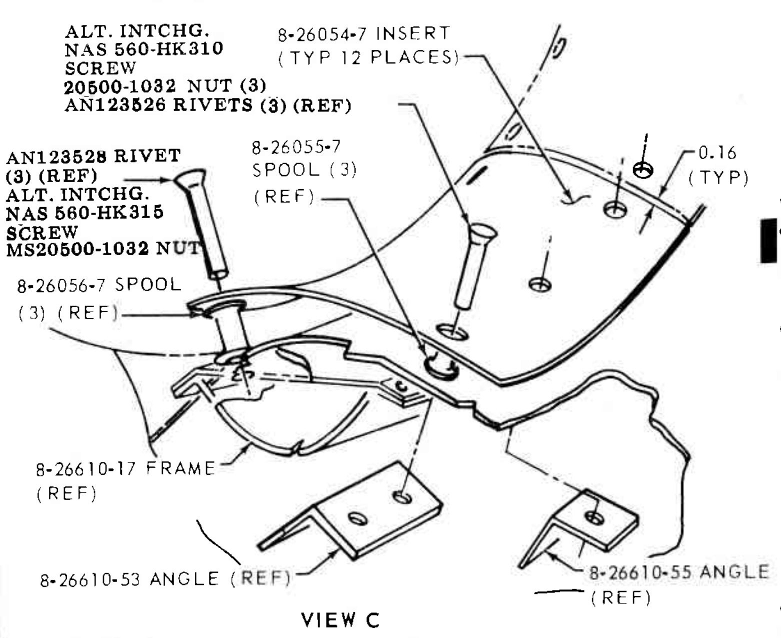


Figure 6-8. Engine Shroud Ejector Repairs and Insert Replacement

around to make a tight repair. Torn edges or cutouts must be repaired by wrapping patch around edge and pokewelding in place. Torn out grommets shall be repaired by wrapping patch around edge, pokewelding in place, punching new hole, and replacing grommet. If replacement grommets are not available, a plate made of 0.016-inch stainless steel may be manufactured locally and pokewelded in place. Drill or punch a hole in the replacement plate the same size as the original hole. Inside edges of the hole must be sealed completely with overlapping pokewelds around the hole. See figure 6-7 for repair of engine shroud blankets by pokewelding. See figure 6-8 for repairs to the engine shroud ejector.

NOTE

Excessively discolored blankets should be replaced.

6-12. Afterburner Liner.

6-13. Smooth Section of Afterburner Liner.

a. Cracks in the leading edge of perforated section may be continued in service provided cracks are no more than 2 inches in length and at least 4 inches from any other crack, and the number of cracks do not exceed ten and no buckling or lipping up is evident. Fluorescent penetrant inspect to located end of crack. Stop drill end of crack with size no. 9 drill. Fluorescent penetrant inspect after stop drilling to insure end of crack is drilled. Stop drilling and/or welding may be accomplished with engine installed in aircraft.

b. All welding in this area of duct must be in accordance with T.O. 2J-J75-6 with the exception of a gas backup. All fluorescent penetrant must be removed from area to be welded. To continue liner in service cracks up to four inches in length not closer together than four inches shall be welded.

6-14. Corrugated Section of Afterburner Liner.

a. One flat spot with 10 inches maximum diameter and/or 14 inches maximum axial length, with the center of the flat spot not to extend into the gas path more than 0.3 inch from the normal position. If two flat spots are present, each must not exceed two thirds of the above limits.

b. Liners which contain cracks may be continued in service provided cracks do not exceed 2 inches in length and are no closer together than 2 inches. Fluorescent penetrant inspect to locate end of crack and stop drill with size no. 21 drill. Fluorescent penetrant inspect after stop drilling to insure end of crack is drilled. Stop drilling and/or welding may be accomplished with engine installed in aircraft. To continue liner in service cracks up to three inches in length not closer together than two inches shall be welded in accordance with T.O. 2J-J75-6 with the exception of gas backup.

Note

A drill stop must be used during drilling operation to prevent over penetration of drill bit. Excessive penetration could result in damage to afterburner duct weldment.

c. Cracks in the flange progressing from the bolt holes outward are acceptable and need not be welded.

NOTE

Any condition found that exceeds the limits specified in paragraphs 6-12 through 6-14 will be repaired in accordance with T.O. 2J-J75-6 with afterburner duct (liner) removed.

6-15. Engine Nose Cone Fairing.

6-16. Damage to the nose cone fairing is critical in that the exterior of the cone is in the engine air intake airstream. Refer to paragraph 6-20 for the extent of negligible damage allowed to remain "as is" after minor rework. If the nose cone is damaged to the extent of the limits listed in paragraph 6-20, remove the nose cone, as outlined in T.O. 1F-106A-10, prior to performing repairs. All repairs to the nose cone exterior surface shall be aerodynamically smooth within the limits set forth in Section I. Blind-type rivets shall not be used when making repairs to the nose cone since this type of rivet may loosen and be drawn into the engine. See figure 10-1 for repairs applicable to the exterior of the nose cone. See figure 6-3 for material specifications and gages.

6-17. Engine Supports.

6-18. Any damage to the engine supports is critical due to the heavy loads and vibrations imposed upon them. Due to the type of construction of the engine supports, repairing is highly improbable and is not advisable. Damaged engine support components shall be replaced.

6-19. Negligible Damage Limits— Engine Section.

6-20. Table 6-I indicates the maximum allowable classifications of five types of negligible damage (damage allowed to remain "as is" after minor rework such as stop drilling cracks and fairing in nicks or scratches). The maximum allowable damage classification will be found to the right of the component name in the vertical column under the "Type of Damage" heading. After classification is determined, see figures 1-17 through 1-19 for the damage limits to be allowed for each class: I, II, or III. The limits given on figures 1-17 through 1-19 apply only for a damaged area after rework as shown on figures 1-20 and 1-21. An aeronautical structures engineer must be consulted for damages exceeding the limits given on figures 1-17 through 1-19 and for damage to components not listed in Table 6-I.

TABLE 6-I
Negligible Damage Limits — Engine Section

COMPONENT	TYPE AND CLASS OF DAMAGE ALLOWED AFTER REWORK					REMARKS
	Scratch	Nick	Dent	Hole	Crack	
SHROUD						
Skins	II	II	III	*	*	
Longitudinal Stiffeners	II	II	III	II	I	
Angles	II	II	III	I	I	
Frames	II	II	III	I	I	
NOSE CONE FAIRING						
Skins (outer)	I	I	*	*	*	
Skins (inner)	I	I	*	*	*	
Attachment flanges	II	II	I	II	II	
Formers	II	II	I	II	II	
Stiffeners	II	II	I	II	II	

*Component must be repaired or replaced.

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Section VII

FABRIC REPAIR AND ATTACHMENT

This section not applicable to F-106A and F-106B aircraft.

Section VIII

EXTRUSION AND ROLL-FORMED CHARTS

8-1. EXTRUSIONS.

8-2. An extrusion is a die-formed structural shape that can be made to specific cross-sectional dimensions. Extrusions are formed by forcing a metal alloy in a plastic state through a stationary die by means of a hydraulic ram. The extrusion process makes possible quantity production of structural shapes in which the metal is more efficiently disposed with relation to design loads than is generally possible in standard roll-formed shapes. By the extrusion process, practical production of structural shapes with unequal thickness of flanges, tapered, bulbed, or lipped flanges, or with square or controlled radius edges is practicable. Another advantage of extrusions is often a considerable weight reduction.

8-3. Identification of Extrusions.

8-4. Extruded shapes are identified by a die or shape number, which in nearly all cases is preceded by an identifying letter, or group of letters. Identifying letters are AND , K, and E. Extrusions without an identifying letter, or with only the letter E, are contractor's numbers. The tables contain dimensional data, cross-sectional area, and type of material. Complicated or special extruded shapes are shown in individual illustrations. Tolerances not detailed, such as corner or fillet radii, shall conform to Specification, Federal Standard No. 245.

8-5. Substitution.

8-6. Substitution of formed shapes for extruded ones is not normally recommended for these airplanes; however, substitution of a structural member by bend-forming sheet metal of the same material is permissible in some instances. When substitution of an extruded shape is necessary, an extrusion of equal or slightly larger dimensions or machined from a larger extruded shape may be used. If the only substitute available is formed sheet stock, specific approval must be obtained from an aeronautical structures engineer.

8-7. Repairs.

8-8. Repairs for some extruded shapes are illustrated in Section X. The design principles of these repairs are outlined in Section I.

8-9. ROLL-FORMED SECTIONS.

8-10. Roll-formed sections differ from extruded shapes in that roll-formed sections are of uniform thickness and are formed from sheet stock. These sections have the advantage of lower cost than extrusions, but are limited in the variety of shapes possible. In specific instances, some roll-formed sections can serve the same function as an extrusion; however, this practice is not recommended without the prior approval of an aeronautical structures engineer.

8-11. Identification of Roll-Formed Sections.

8-12. Identification numbers on roll-formed sections are contractor's designations and are preceded by the letter "Y." A typical example of identifying a roll-formed section is as follows:

> EXAMPLE: Y2-B39T. The prefix "Y2" is a Convair angle of 90 degrees. The letter "B" indicates type of material as shown in table below. The number "39" indicates dimension shown in corresponding table. The letter "T" indicates that material is in hardened condition.

In the following tables, the letters A, B, D, E, F, and H have been omitted as material identification letters. Instead, material is indicated in a separate column. The condition letter "T" has also been omitted since all repair members are in the hardened condition when installed in the airplanes. The material identification letters are as follows:

"A" is 2024 clad material.
"B" is 7075 clad material.
"D" is 2024 bare material.
"E" is 7075 bare material.
"F" is 7178 bare material.
"H" is 7178 clad material.

The various shapes available in roll-formed sections are grouped in the following index. Nominal dimensions for roll-formed "Y" sections are given in the tables for each illustration. Length and width dimensions should be held to ±0.030 inch and angular dimension to ±1 degree.

8-13. Substitution.

8-14. With proper equipment, substitution for all "Y" sections may be fabricated in the field. As noted in the preceding paragraph, all "Y" sections are made of 2024, 7075, or 7178 clad and bare materials. The 2024 material can usually be formed to the required dimensions when in the hard condition. The 7075 and 7178 material, being harder than 2024, must be in the SO or SW condition when being formed and must be in the fully heat treated (T6) condition when installed in the airplane.

8-15. Repairs.

8-16. Repairs for some roll-formed sections are illustrated in Section X. The design principles of these repairs are outlined in Section I.

8-17. EXTRUSION INDEX.

8-18. The following index lists the extrusions by their die number and gives the page number on which each extrusion is shown.

EXTRUSION INDEX

EXTRUSION DIE NUMBER	PAGE NUMBER
AND 10133	8-9
AND 10134	-10
AND 10136	- 5
AND 10137	-38
AND 10138	-47
AND 10140	-45
HM 26852	-38
K 59224	-47
60919 (29321)	-34
E 101104	-48
E 101105	-48
E 101208	-35
E 102601	-11
E 111104	- 9
E 111105	- 9
E 111106	- 9
E 111201	- 9
E 112302	- 9
E 121103	-10
E 121108	-10
E 121112	-10
E 121120	-10
E 121121	-10
E 121122	-10
E 121124	-10
E 121201	-10
E 121202	-48
E 121208	-10
E 121212	-10
E 121213	-10
E 121214	-10
E 121220	-11
E 121302	-11
E 121306	-10
E 121307	-10
E 122101	-10
E 122302	-10
E 122305	-10
E 122311	-10
E 122501	-10
E 141101	-11
E 141104	-11
E 141127	-11
E 141206	8-12
E 141207	-48
E 141220	-12
E 141304	-12
E 141305	-11
E 141404	-12
E 142213	-12
E 151101	-11
E 151301	-12
E 161102	-35
E 161202	-32
E 161203	-32
E 181201	-32
E 201107	-13
E 201301	-13
E 211105	-13
E 221105	-13
E 221202	-15
E 221205	-15
E 241203	-14
E 261206	-14
E 261207	-13
E 291201	-15
E 291202	-15
E 311106	-15
E 311107	-15
E 321107	-15
E 321108	-14
E 341102	-14
E 341202	-14
E 341501	-14
E 351402	-13
E 361502	-14
E 401105	-46
E 441001	-39
E 441005	-39
E 441006	-40
E 441007	-40
E 441008	-40
E 441009	-40
E 441010	-40
E 441011	-43
E 441034	8-40
E 441035	-39
E 441036	-39
E 441045	-39
E 441046	-39
E 441101	-43
E 441104	-41
E 441109	-40
E 441118	-42
E 441206	-41
E 441301	-43
E 441501	-43
E 451105	-44
E 451106	-44
E 451402	-44
E 451403	-44
E 461103	-46
E 461203	-46
E 461501	-41
E 461502	-41
E 462404	-46
E 471001	-39
E 471002	-40
E 471107	-39
E 471207	-40
E 471208	-39
E 471401	-46
E 471402	-46
E 471501	-39
E 481105	-42
E 481106	-40
E 481107	-40
E 481108	-42
E 481406	-42
E 491002	-32
E 491003	-32
E 491106	-33
E 491115	-31
E 491116	-32
E 491119	-32
E 491124	-31

EXTRUSION INDEX (CONT)

EXTRUSION INDEX (CONT)

EXTRUSION DIE NUMBER	PAGE NUMBER
E 731502	8-18
E 732701	-18
E 741103	-20
E 741108	-20
E 741213	-20
E 741214	-20
E 741215	-20
E 741216	-20
E 741302	-21
E 741308	-21
E 741309	-21
E 741310	-20
E 741311	-20
E 741312	-20
E 741313	-20
E 741406	-22
E 741502	-20
E 751302	-29
E 751303	-29
E 751401	-29
E 751404	-23
E 751601	-29
E 761203	-29
E 761301	-28
E 761302	8-28
E 771201	-29
E 771206	-23
E 771210	-36
E 781202	-28
E 801001	-25
E 811102	-28
E 811103	-23
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E 811203	-30
E 811304	-24
E 812305	-22
E 812309	-22
E 821203	-27
E 821204	-28
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E 821401	-23
E 821403	-22
E 821501	-30
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E 832501	8-24
E 841203	-27
E 841204	-30
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E 881201	-30
E 881202	-30
E 881501	-25
E 891101	-26
E 891102	-25
E 891103	-25
E 901101	-47
E 901202	-47
E 921401	-47
E 942304	-47
E 961501	-47
7144552	-46
7144699	-44

8-19. ROLL-FORMED SECTION INDEX.

8-20. The following index lists the roll-formed sections by the contractor (Convair) designated numbers and gives the figure number on which each roll-formed section is shown.

ROLL-FORMED SECTION INDEX

Roll-Formed Section Number	Page Number
Y 2	8-51
Y 3	-51
Y 4	-50
Y 5	-50
Y 6	-52
Y 7	-48
Y 10	-49
Y 12	-48
Y 32	-48
Y 34	-48
Y 36	-49

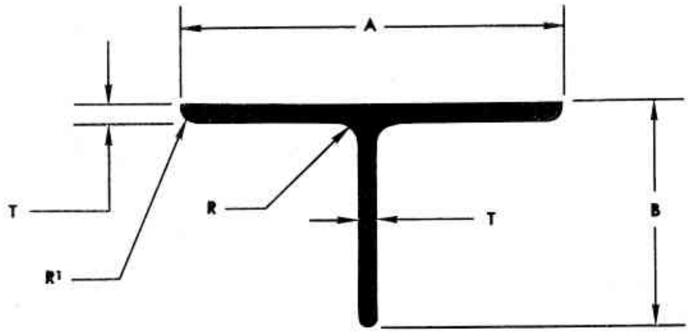
EXTRUSION INDEX (CONT)

EXTRUSION INDEX (CONT)

8-19. ROLL-FORMED SECTION INDEX.

8-20. The following index lists the roll-formed sections by the contractor (Convair) designated numbers and gives the figure number on which each roll-formed section is shown.

ROLL-FORMED SECTION INDEX



NO.	A	B	T	R	R1	AREA	2024	7075	7178
AND10136									
-1202	1.250	0.750	0.063	0.125	0.063	0.127		X	
-1303	1.375	1.000	0.078	0.125	0.078	0.183		X	
-1304	1.375	1.000	0.094	0.125	0.094	0.216		X	
-1305	1.375	1.250	0.094	0.125	0.094	0.240	X		
-1402	1.500	0.750	0.094	0.125	0.094	0.204	X	X	
-1403	1.500	1.000	0.078	0.125	0.078	0.193	X	X	X
-1405	1.500	1.250	0.078	0.125	0.078	0.212	X	X	
-1406	1.500	1.250	0.094	0.125	0.094	0.252		X	
-1407	1.500	1.250	0.125	0.125	0.125	0.326		X	
-1603	1.750	1.125	0.094	0.125	0.094	0.263		X	
-1607	1.750	1.625	0.094	0.125	0.094	0.310		X	X
-1608	1.750	1.625	0.125	0.125	0.125	0.405	X	X	
-1609	1.750	1.625	0.156	0.125	0.156	0.497	X		
-1701	1.875	1.000	0.078	0.125	0.078	0.222	X	X	X
-2001	2.000	1.000	0.078	0.125	0.078	0.231		X	
-2002	2.000	1.000	0.094	0.125	0.094	0.275	X	X	X
-2004	2.000	1.250	0.078	0.125	0.078	0.251	X	X	
-2005	2.000	1.250	0.094	0.125	0.094	0.299			X
-2006	2.000	1.250	0.125	0.125	0.125	0.389		X	
-2008	2.000	1.750	0.125	0.125	0.125	0.451		X	
-2401	2.500	1.250	0.094	0.125	0.094	0.346		X	
-2403	2.500	1.250	0.156	0.125	0.156	0.554		X	
-2405	2.500	1.625	0.125	0.125	0.125	0.498		X	
-3005	3.000	2.500	0.125	0.125	0.125	0.670			X
-3006	3.000	2.500	0.188	0.188	0.188	0.995		X	
-1404	1.500	1.000	0.094	0.125	0.094	0.228 (6061 and 7075)			
-1606	1.750	1.375	0.156	0.125	0.156	0.457 (ZK60A)			

.06.03.037-1A.58.01.00

Figure 8-1. Extruded Shapes

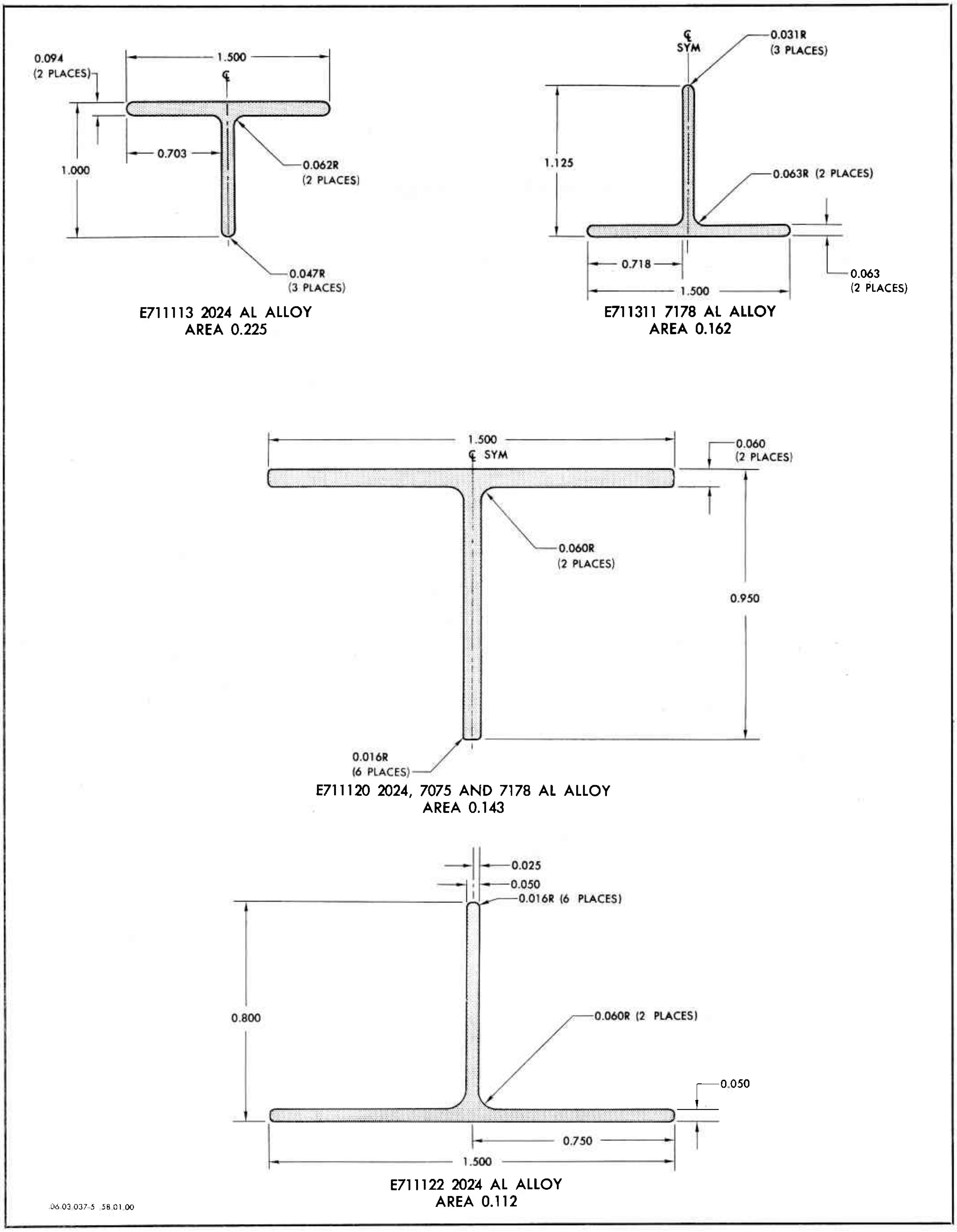
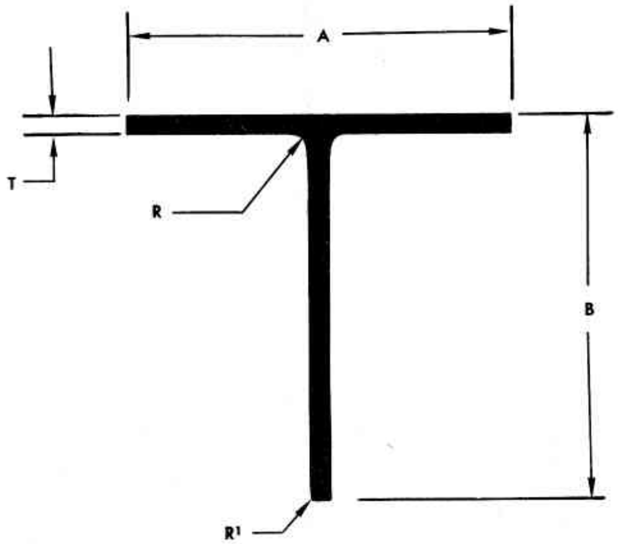


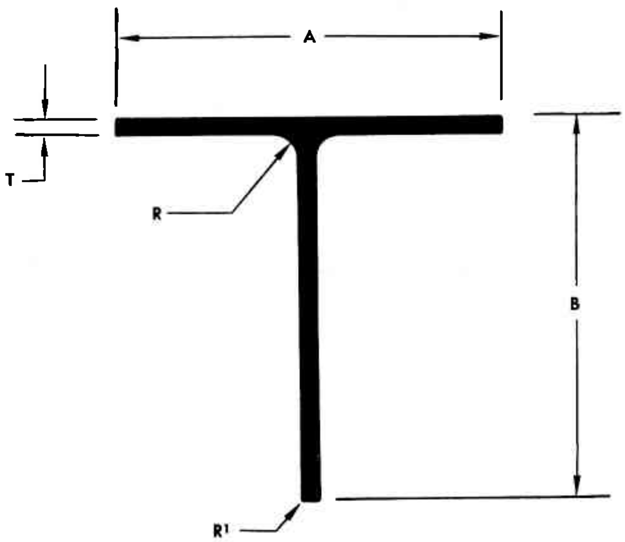
Figure 8-2. Extruded Shapes



NO.	A	B	T	R	R¹	AREA	2024	7075	7178
E711306	3.500	2.120	0.100	0.125	0.016	0.560		X	
E711308	3.000	2.500	0.072	0.125	0.016	0.394		X	
E711310	3.000	2.000	0.125	0.180	0.047	0.622		X	
E711313	3.750	2.500	0.078	0.125	0.016	0.488		X	X
E711314	2.970	2.750	0.087	0.130	0.016	0.497		X	
E711315	1.800	3.250	0.080	0.900	0.016	0.401		X	
E711402	4.100	2.120	0.100	0.130	0.031	0.618			X
E711403	4.000	2.530	0.100	0.190		0.658	X		
E711406	4.000	3.420	0.130	0.190		0.963		X	
E712202	1.500	1.250	0.062	0.062	0.016	0.170	X	X	X
E712203	1.500	1.500	0.064	0.125	0.016	0.195		X	X
E712207	1.625	1.750	0.085	0.094	0.016	0.280		X	
E712211	1.500	0.750	0.051	0.125	0.016	0.117		X	
E712303	2.310	1.400	0.064	0.940		0.240		X	X
E712305	2.400	2.830	0.072	0.125	0.016	0.380		X	
E712307	2.500	1.500	0.064	0.060	0.016	0.282	X	X	
E712401	1.937	3.250	0.125	0.156		0.630	X		
E712406	1.000	3.000	0.094	0.156	0.016	0.470		X	
E712408	1.500	3.290	0.081	0.120	0.016	0.387		X	

.06.03.037- 6 .58.01.00

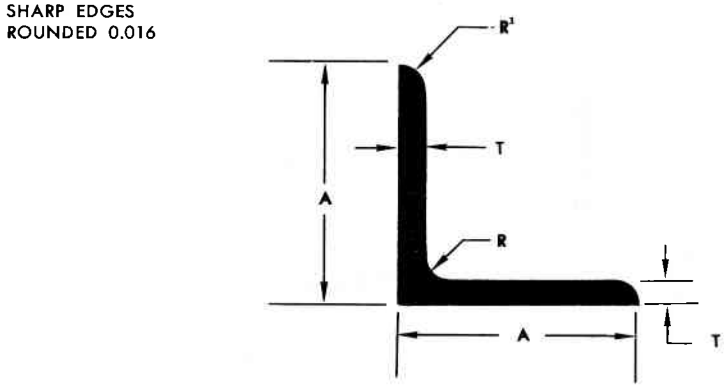
Figure 8-3. Extruded Shapes



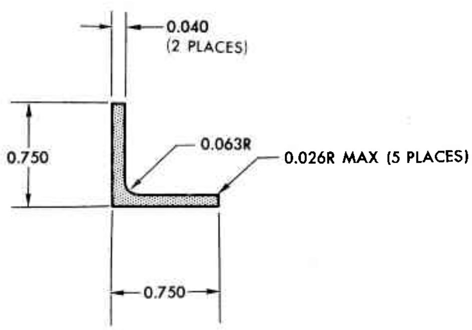
NO.	A	B	T	R	R¹	AREA	2024	7075	7178
E711110	1.250	1.250	0.050	0.094	0.026	0.126		X	
E711111	1.875	1.000	0.050	0.094	0.016	0.145		X	
E711115	1.750	1.250	0.051	0.094	0.016	0.152			X
E711119	1.750	1.750	0.050	0.125	0.016	0.179		X	
E711123	1.500	0.950	0.080	0.060	0.016	0.190	X		
E711125	1.500	0.950	0.100	0.125	0.016	0.240	X		
E711126	1.400	0.750	0.100	0.060	0.016	0.206	X		
E711127	1.500	1.200	0.070	0.070	0.016	0.160	X		
E711129	1.600	1.700	0.070	0.125	0.016	0.233			
E711132	1.500	1.125	0.050	0.094	0.016	0.132	X		X
E711133	1.600	1.600	0.060	0.125	0.030	0.195		X	
E711138	1.950	1.800	0.500	0.062	0.016	0.185		X	
E711140	1.750	1.500	0.188	0.125		0.581	X		
E711201	2.180	2.000	0.070	0.090	0.016	0.291			X
E711202	2.250	2.000	0.070	0.094	0.016	0.293			X
E711203	1.600	2.090	0.070	0.094	0.016	0.255		X	
E711204	1.700	2.040	0.125	0.125	0.031	0.467			X
E711206	1.750	1.550	0.078	0.125	0.015	0.258		X	
E711209	2.250	1.750	0.062	0.062	0.016	0.248	X	X	
E711210	1.750	1.120	0.078	0.120	0.016	0.224			X
E711211	2.900	1.120	0.064	0.125	0.016	0.2655		X	
E711212	2.000	1.500	0.078	0.125	0.016	0.273		X	
E711214	2.000	2.000	0.125	0.125	0.031	0.485			X
E711216	2.000	1.000	0.063	0.063	0.016	0.186	X	X	
E711218	1.625	1.750	0.085	0.094	0.032	0.283		X	
E711219	2.000	1.000	0.090	0.125	0.026	0.269		X	
E711220	2.000	1.250	0.072	0.125	0.010	0.235		X	X
E711223	2.000	2.000	0.094	0.156	0.026	0.378		X	
E711227	1.600	1.850	0.050	0.125	0.016	0.177		X	
E711228	1.750	2.500	0.091	0.125	0.016	0.380		X	
E711230	1.250	2.000	0.080	0.125	0.016	0.260		X	
E711232	1.810	1.950	0.125	0.125	0.016	0.477		X	
E711233	1.600	2.400	0.100	0.090	0.016	0.393			X
E711234	2.000	2.000	0.080	0.125	0.016	0.320		X	
E711235	0.472	2.220	0.072	0.125	0.015	0.195	X		
E711239	2.000	1.100	0.100	0.120	0.016	0.306			X
E711301	3.250	2.500	0.070	0.190	0.016	0.403			X
E711302	3.000	1.875	0.080	0.125	0.031	0.389		X	
E711303	3.000	1.000	0.094	0.125	0.094	0.375	X		
E711305	2.375	2.625	0.080	0.125	0.016	0.400	X	X	X

.06.03.037-2A .58.01.00

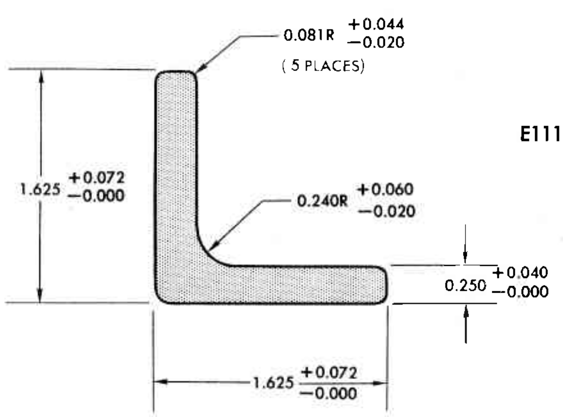
Figure 8-4. Extruded Shapes (Sheet 1 of 2)



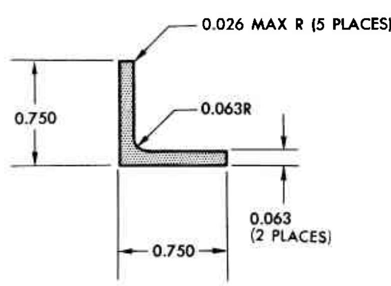
NO.	A	T	R	R1	AREA	2024	7075	7178
AND10133								
–0601	0.750	0.063	0.094	0.063	0.091	X	X	X
–0602	0.750	0.063	0.094	0.063	0.130	X	X	
–0701	0.875	0.063	0.094	0.063	0.107		X	
–0702	0.875	0.094	0.094	0.094	0.154		X	X
–0703	0.875	0.125	0.094	0.125	0.198		X	
–1002	1.000	0.094	0.188	0.094	0.183		X	
–1003	1.000	0.125	0.188	0.125	0.235		X	
–1201	1.250	0.063	0.188	0.063	0.159	X		
–1203	1.250	0.125	0.188	0.125	0.298			X
–1401	1.500	0.063	0.188	0.063	0.191	X		
–1402	1.500	0.094	0.188	0.094	0.277	X		
–1403	1.500	0.125	0.188	0.125	0.360	X	X	X
–1404	1.500	0.188	0.188	0.188	0.521		X	
–2004	2.000	0.250	0.250	0.250	0.924		X	
–2403	2.500	0.250	0.250	0.250	1.170	X		
E111106	1.500	0.094	0.094	0.047	0.273		X	
E112302	1.000	0.062	0.062	0.031	0.121		X	



E111104 2024 AL ALLOY
AREA 0.059

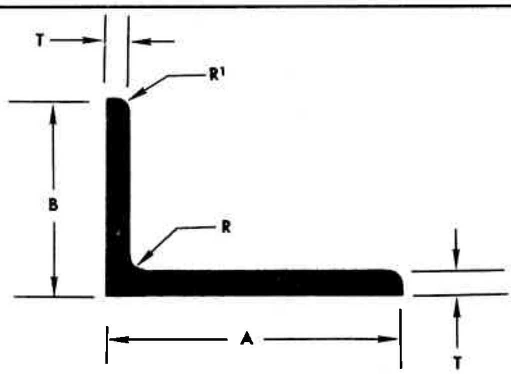


E111201 7178 AL ALLOY
AREA 0.756

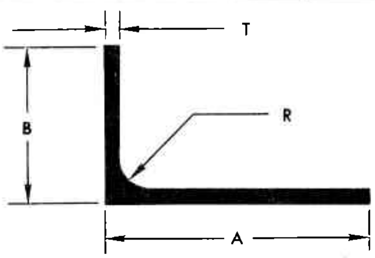


E111105 2024 AL ALLOY
AREA 0.091

Figure 8-4. Extruded Shapes (Sheet 2 of 2)

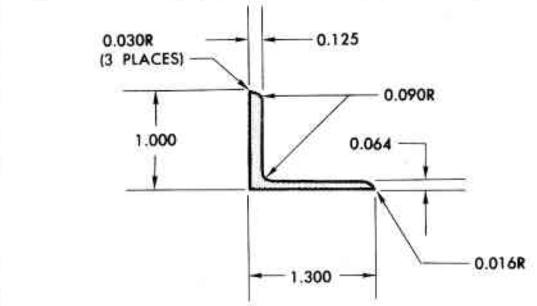


NO.	A	B	T	R	R1	AREA	2024	7075	7178
AND10134 –0602	0.750	0.625	0.063	0.063	0.063	0.082		X	
–1003	1.000	0.750	0.063	0.125	0.063	0.108		X	
–1005	1.000	0.875	0.063	0.125	0.063	0.116		X	
–1201	1.250	0.750	0.063	0.125	0.063	0.124		X	
–1202	1.250	0.750	0.094	0.125	0.094	0.179	X		
–1203	1.250	0.750	0.125	0.125	0.125	0.231	X	X	
–1204	1.250	1.000	0.063	0.125	0.063	0.139	X	X	
–1205	1.250	1.000	0.094	0.125	0.094	0.202	X	X	
–1206	1.250	1.010	0.125	0.125	0.125	0.262		X	
–1401	1.500	0.750	0.094	0.156	0.094	0.204		X	
–1402	1.500	0.750	0.125	0.156	0.125	0.264		X	
–1406	1.500	1.250	0.094	0.156	0.094	0.251		X	
–1407	1.500	1.250	0.125	0.156	0.125	0.327			X
–1601	1.750	1.000	0.125	0.156	0.125	0.327		X	
–1602	1.750	1.250	0.125	0.156	0.125	0.358	X		
–2005	2.000	1.250	0.188	0.188	0.188	0.568	X		
–2010	2.000	0.750	0.156	0.188	0.156	0.558		X	
E121108	1.000	0.750	0.063	0.063	0.031	0.106			X
E121120	0.875	0.750	0.094	0.094	0.094	0.142			X
E121121	1.500	1.250	0.072	0.090	0.070	0.198			X
E121213	2.500	1.250	0.125	0.188	0.094	0.547			X

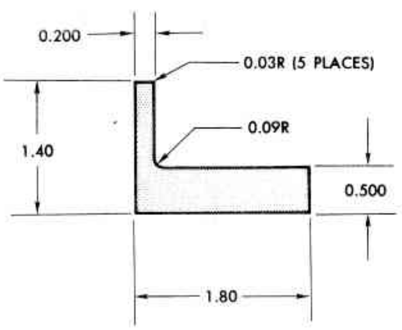


NO.	A	B	T	R	AREA	2024	7075	7178
E121103	1.200	0.880	0.090	0.090	0.189			X
E121112	1.750	0.750	0.062	0.125	0.154		X	
E121122	1.375	0.812	0.072	0.125	0.160		X	
E121124	1.350	0.700	0.080	0.090	0.158		X	
E121201	2.120	1.890	0.070	0.090	0.276		X	X
E121208	1.750	1.125	0.064	0.125	0.183		X	
E121212	2.250	1.165	0.125	0.120	0.470		X	
E121214	2.625	1.750	0.072	0.125	0.313	X		
E121306	2.750	1.200	0.080	0.125	0.3136	X		
E121307	3.000	1.250	0.125	0.125	0.517		X	
E122101	1.160	0.930	0.065	0.130	0.135			X
E122302	2.000	1.500	0.064	0.125	0.223	X		
E122305	2.625	1.750	0.072	0.125	0.3130			X
E122311	2.780	1.063	0.125	0.063	0.466	X		
E122501	3.500	1.750	0.093	0.156	0.580		X	
CORNER OR FILLET RADII NOT CALLED OUT SHALL CONFORM TO FED. STD. NO. 245								

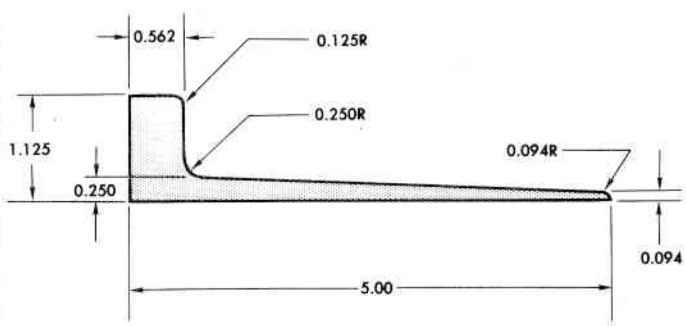
Figure 8-5. Extruded Shapes



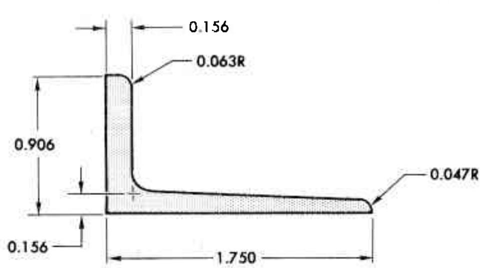
ANGLE-90°, FLANGE WIDTH AND THICKNESS UNEQUAL
E141101 2024 AL ALLOY.
AREA 0.200



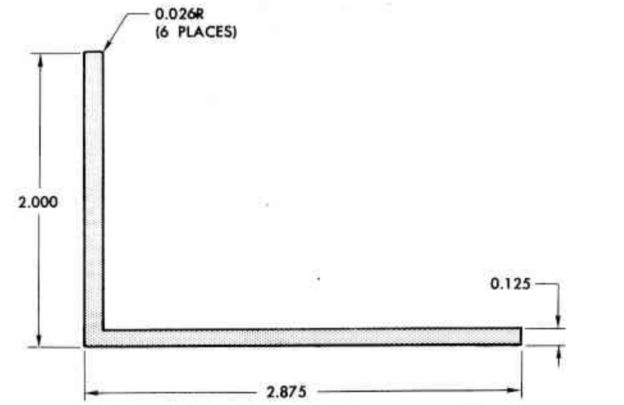
ANGLE-90°, FLANGE WIDTH AND THICKNESS UNEQUAL
E121220 7178 AL ALLOY
AREA 1.08



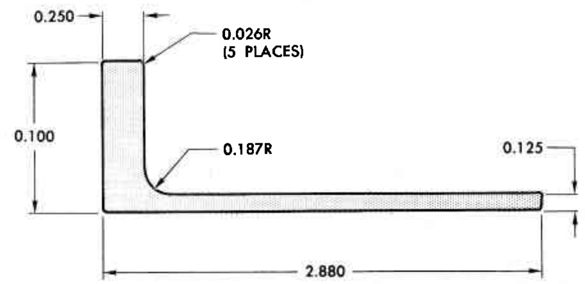
ANGLE—90°, FLANGE WIDTH UNEQUAL,
THICKNESS TAPERED
E102601 2024 AL ALLOY
AREA 1.404



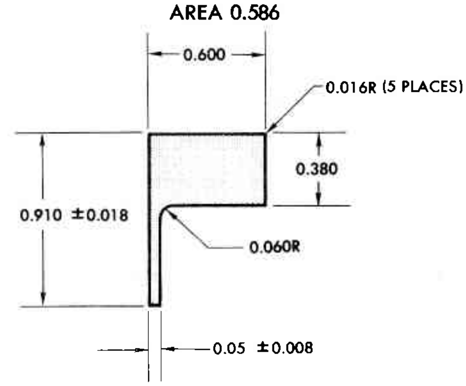
ANGLE-90°, FLANGE WIDTH UNEQUAL, THICKNESS TAPERED
E151101 7075 AL ALLOY
AREA 0.320



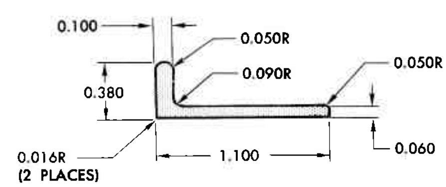
ANGLE-90°, FLANGE WIDTH UNEQUAL, THICKNESS EQUAL
E121302 2024 AL ALLOY
AREA 0.594



ANGLE-90°, FLANGE WIDTH AND THICKNESS UNEQUAL
E141305 7075 AL ALLOY
AREA 0.586

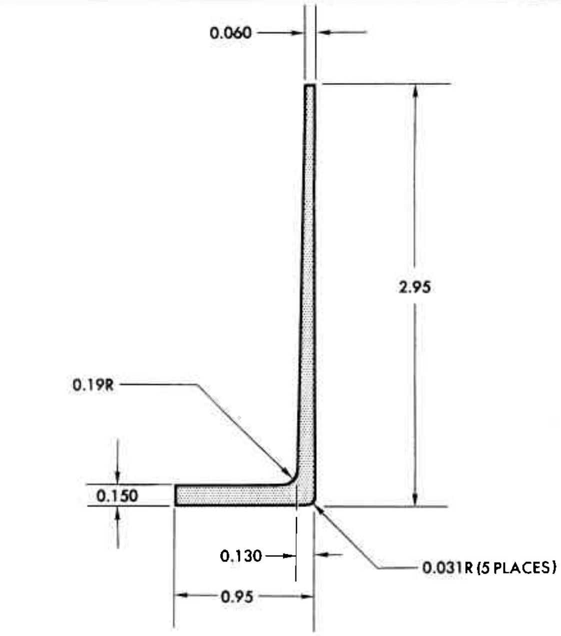


ANGLE-90°, FLANGE WIDTH AND THICKNESS UNEQUAL
E141127 7075 AL ALLOY
AREA 0.254

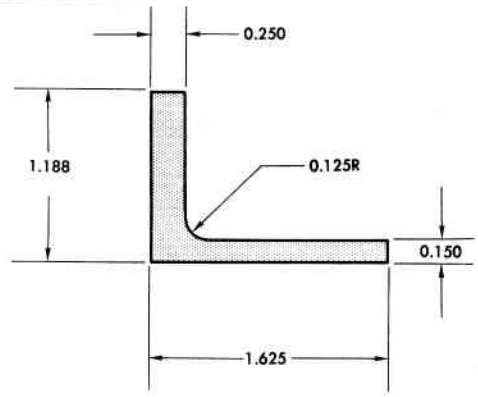


ANGLE-90°, FLANGE WIDTH AND THICKNESS UNEQUAL
E141104 ZK-60A MAG ALLOY
AREA 0.097

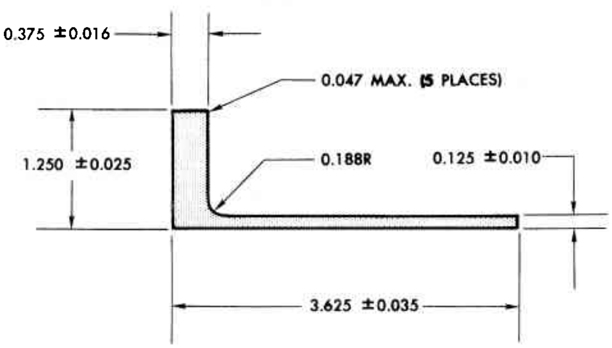
Figure 8-6. Extruded Shapes



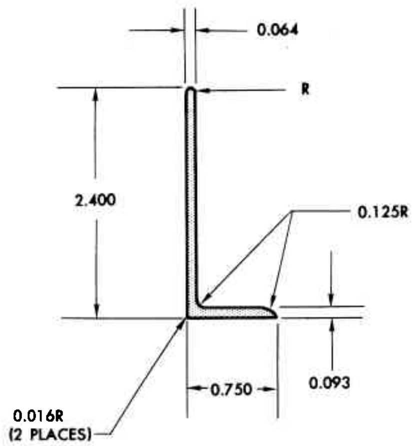
ANGLE—90°, FLANGE WIDTH UNEQUAL,
THICKNESS TAPERED
E151301 7178 AL ALLOY
AREA 0.420



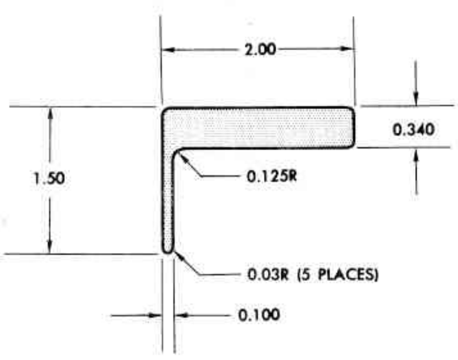
ANGLE—90°, FLANGE WIDTH AND THICKNESS UNEQUAL
E142213 7178 AL ALLOY
AREA 0.50



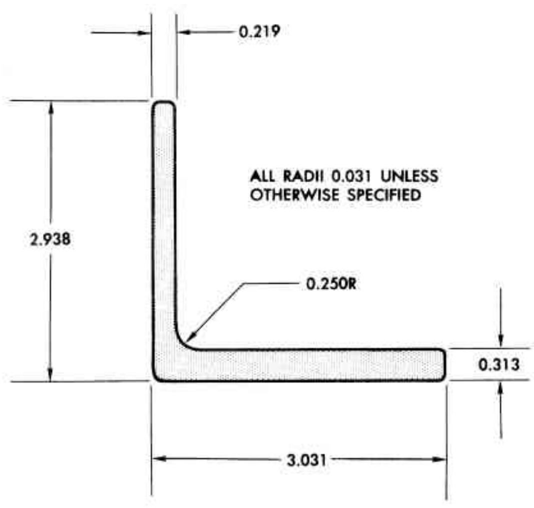
ANGLE-90°, FLANGE WIDTH AND THICKNESS UNEQUAL
E141304 7178 AL ALLOY
AREA 0.882



ANGLE-90°, FLANGE WIDTH AND THICKNESS UNEQUAL
E141206 7178 AL ALLOY
AREA 0.223

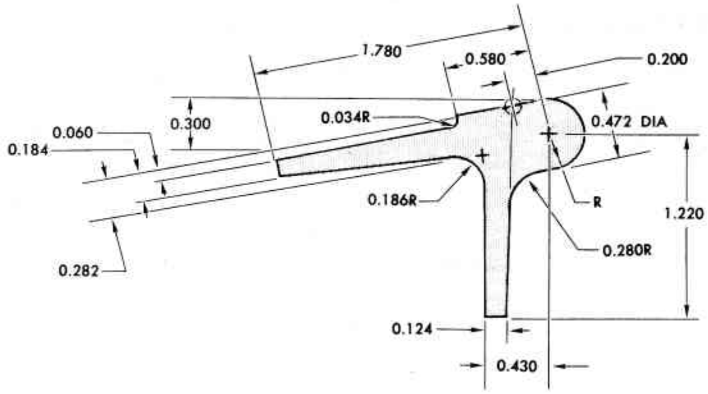


ANGLE-90°, FLANGE WIDTH AND THICKNESS UNEQUAL
E141220 7075 AL ALLOY
AREA 0.798

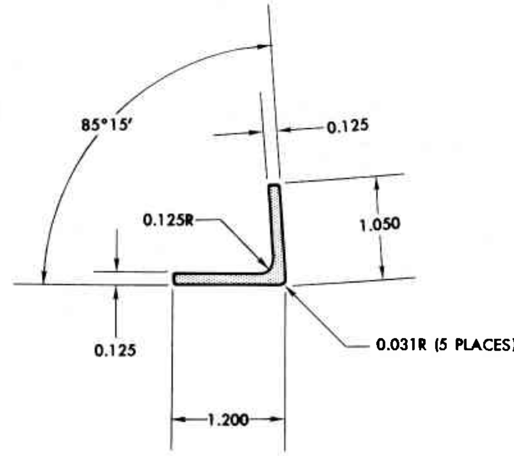


ANGLE-90°, FLANGE WIDTH AND THICKNESS UNEQUAL
E141404 7075 AL ALLOY
AREA 1.536

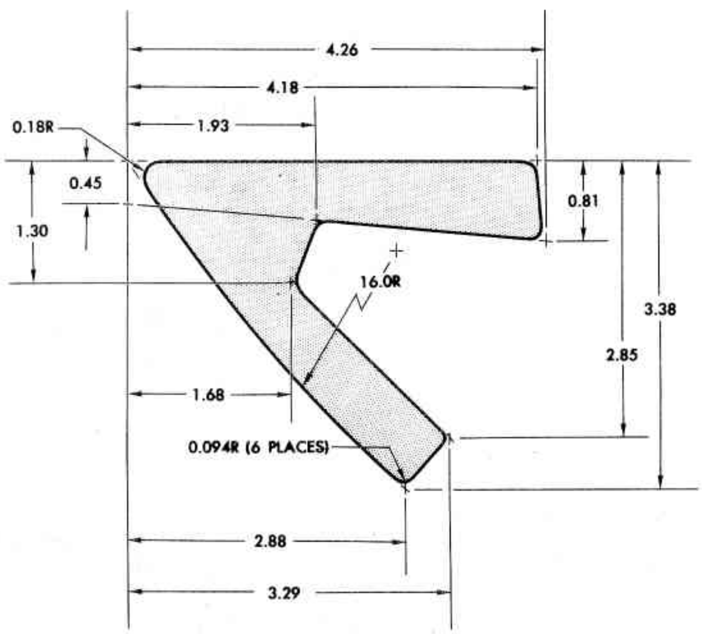
Figure 8-7. Extruded Shapes



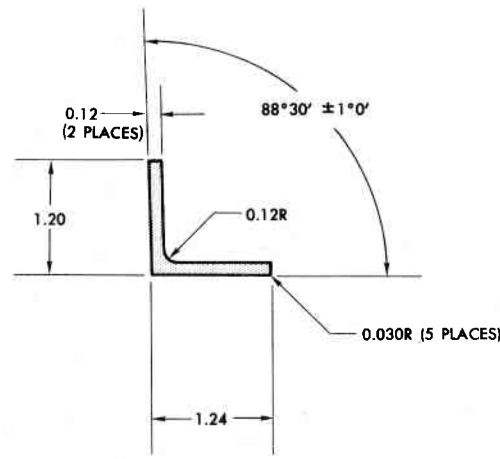
ANGLE-ACUTE, BULBED APPENDAGE
E261207 7075 AL ALLOY
AREA 0.700



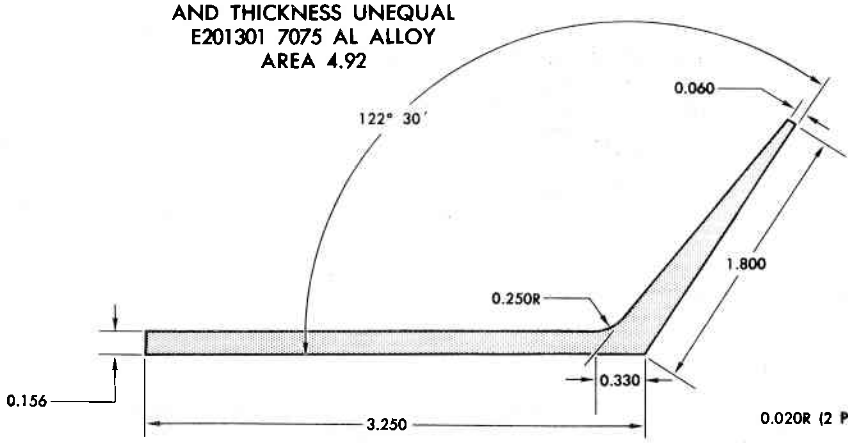
ANGLE—ACUTE, FLANGE WIDTH UNEQUAL, THICKNESS EQUAL
E221105 7075 AL ALLOY
AREA 0.269



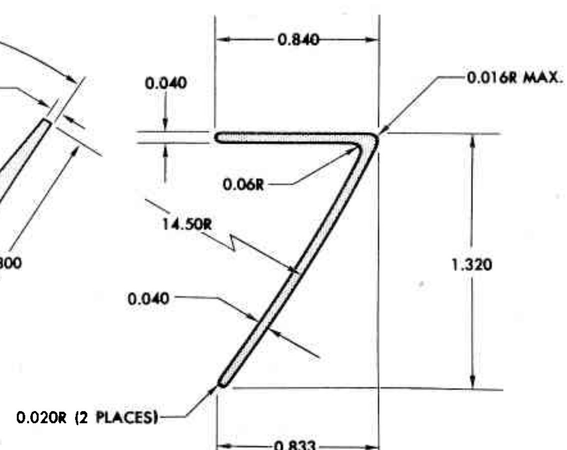
ANGLE—ACUTE, FLANGE WIDTH AND THICKNESS UNEQUAL
E201301 7075 AL ALLOY
AREA 4.92



ANGLE—ACUTE, FLANGE WIDTH UNEQUAL, THICKNESS EQUAL
E211105 7075 AL ALLOY
AREA 0.281



ANGLE-OBTUSE, TAPERED FLANGE
E351402 7178 AL ALLOY
AREA 0.777



ANGLE—ACUTE, MISC.
E201107 7178 AL ALLOY
AREA 0.094

Figure 8-8. Extruded Shapes

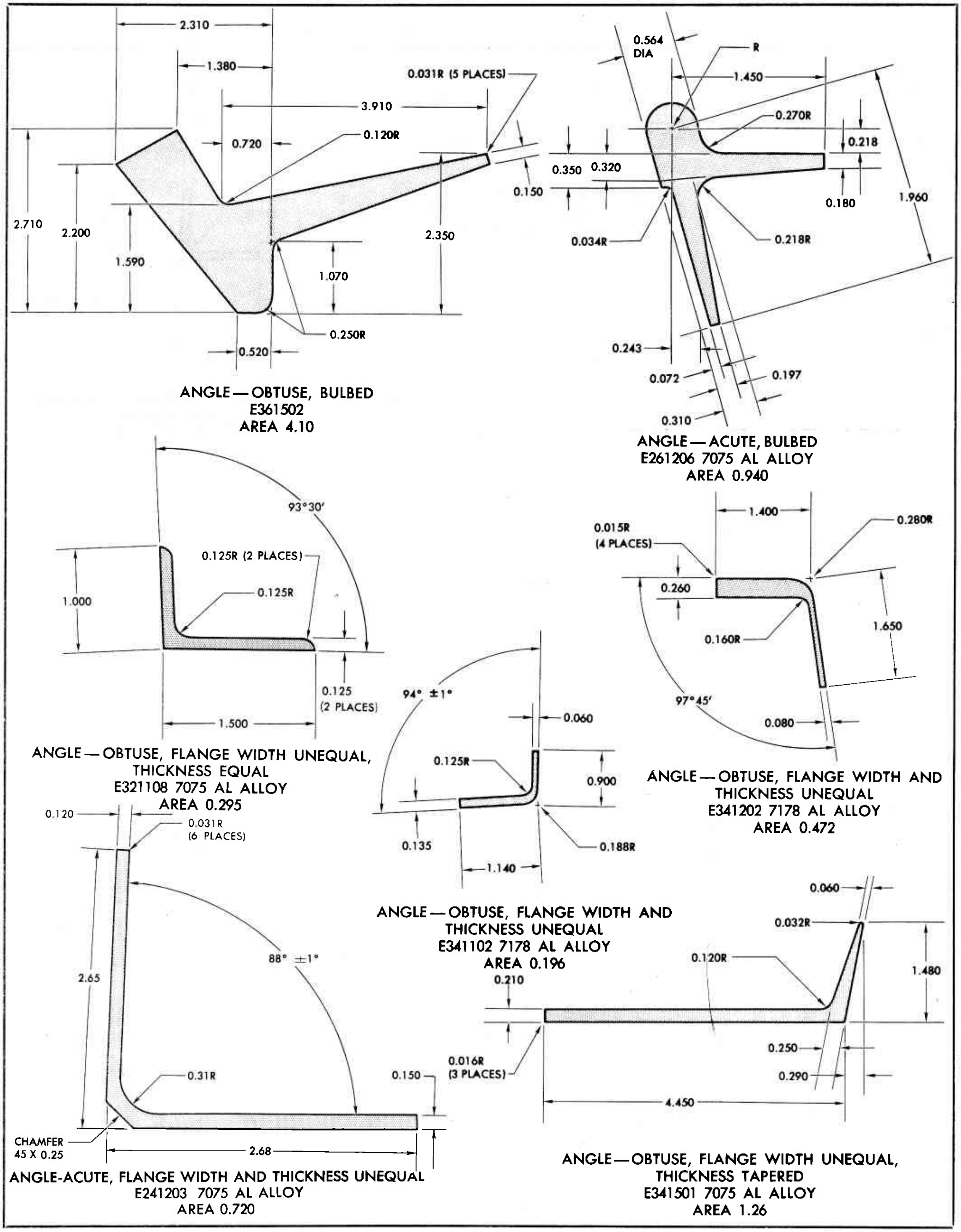
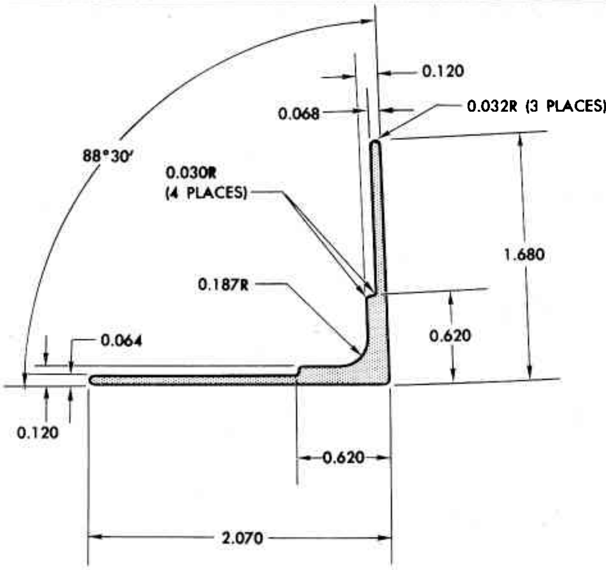
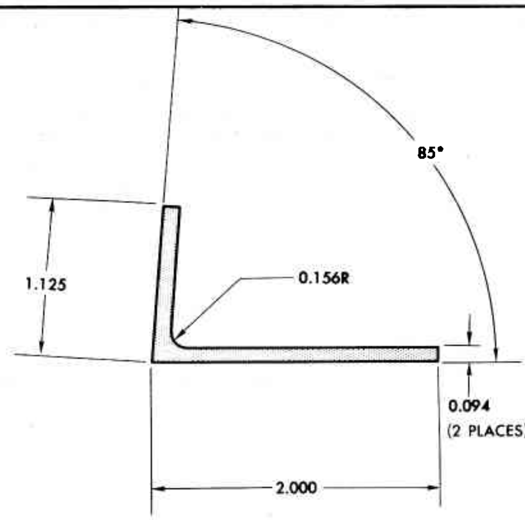


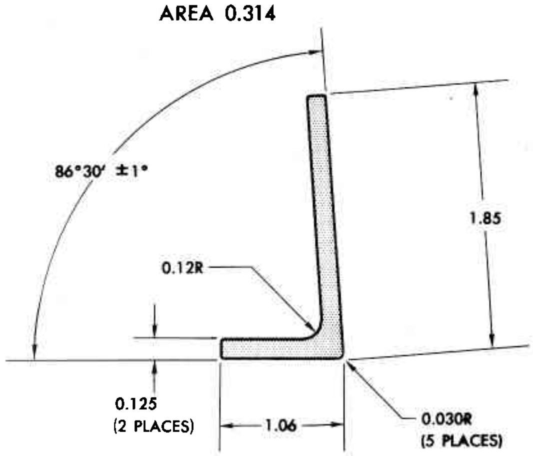
Figure 8-9. Extruded Shapes



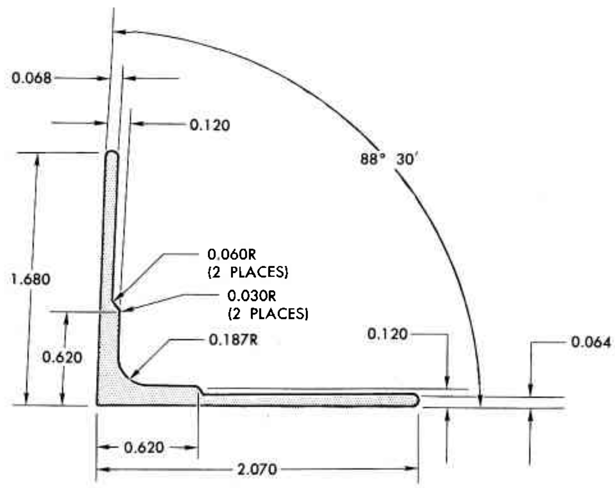
ANGLE—ACUTE, STEPPED FLANGE
E291201 7178 AL ALLOY
AREA 0.314



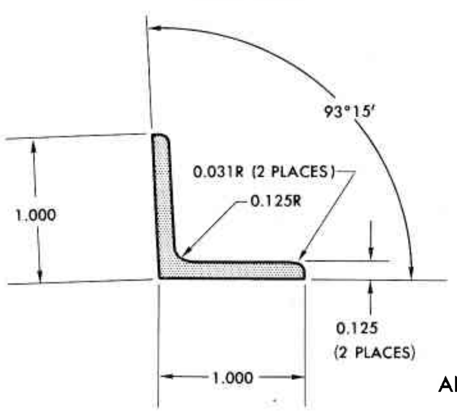
ANGLE—ACUTE, FLANGE WIDTH UNEQUAL,
THICKNESS EQUAL
E221205 7075 AL ALLOY
AREA 0.275



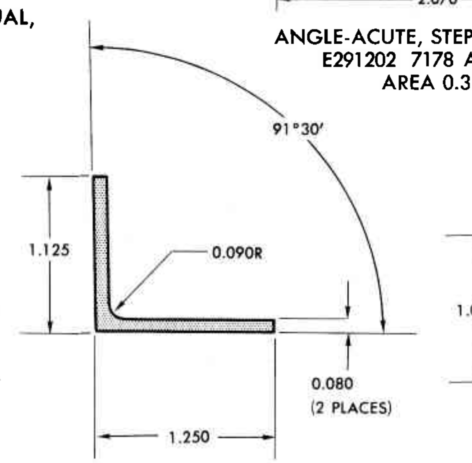
ANGLE—ACUTE, FLANGE WIDTH UNEQUAL,
THICKNESS EQUAL
E221202 7075 AL ALLOY
AREA 0.350



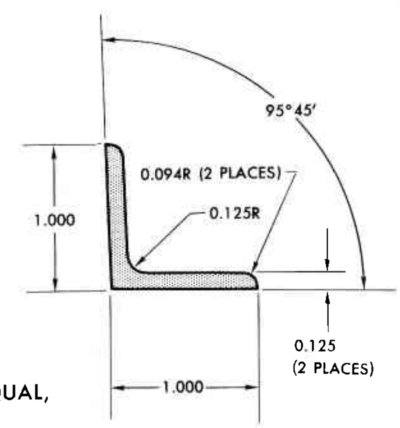
ANGLE-ACUTE, STEPPED FLANGE
E291202 7178 AL ALLOY
AREA 0.315



ANGLE—OBTUSE, FLANGE WIDTH AND
THICKNESS EQUAL
E311106 7075 AL ALLOY
AREA 0.235



ANGLE—OBTUSE, FLANGE WIDTH UNEQUAL,
THICKNESS EQUAL
E321107 7075 AL ALLOY
AREA 0.142



ANGLE—OBTUSE, FLANGE WIDTH AND
THICKNESS EQUAL
E311107 7075 AL ALLOY
AREA 0.235

Figure 8-10. Extruded Shapes

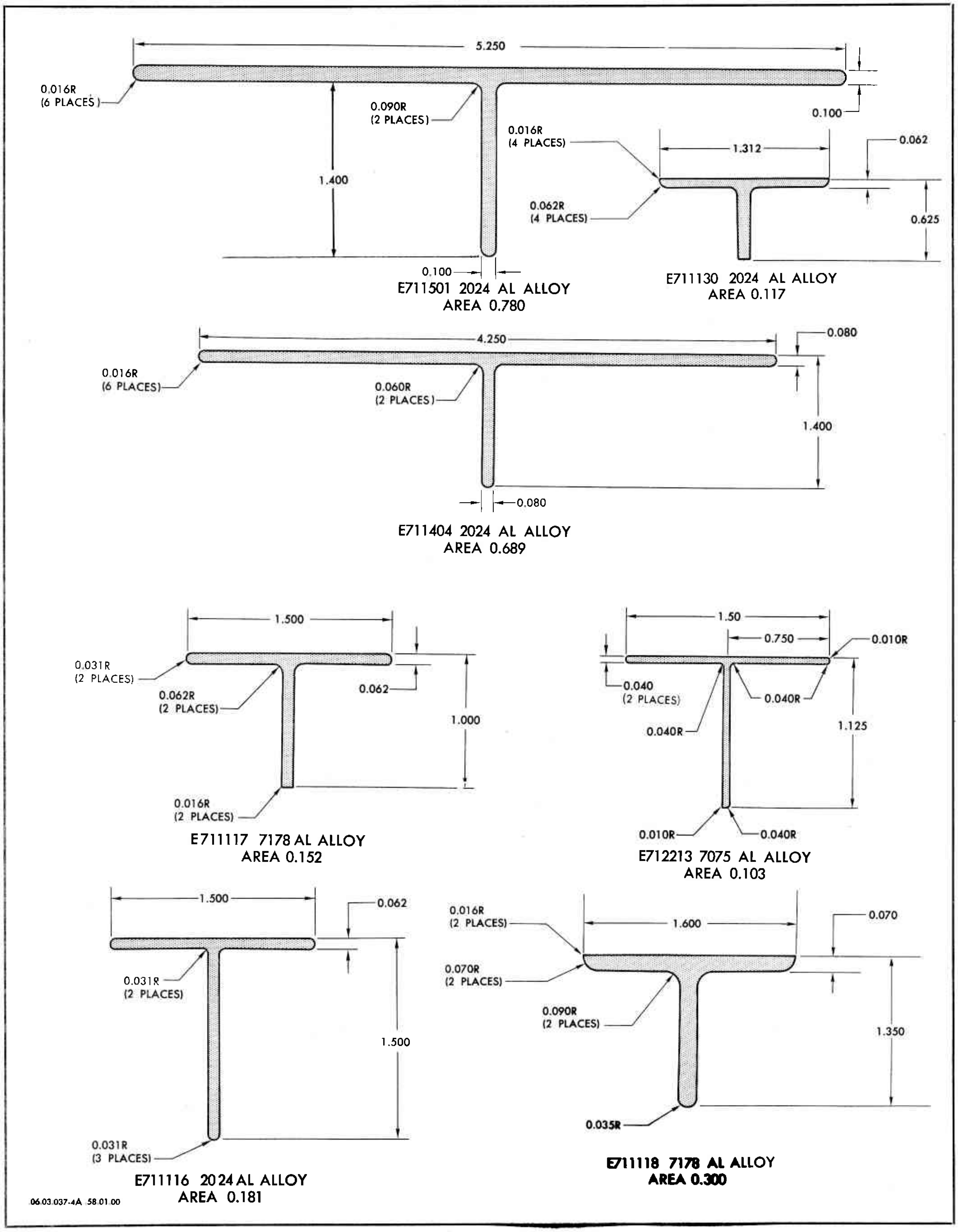


Figure 8-11. Extruded Shapes

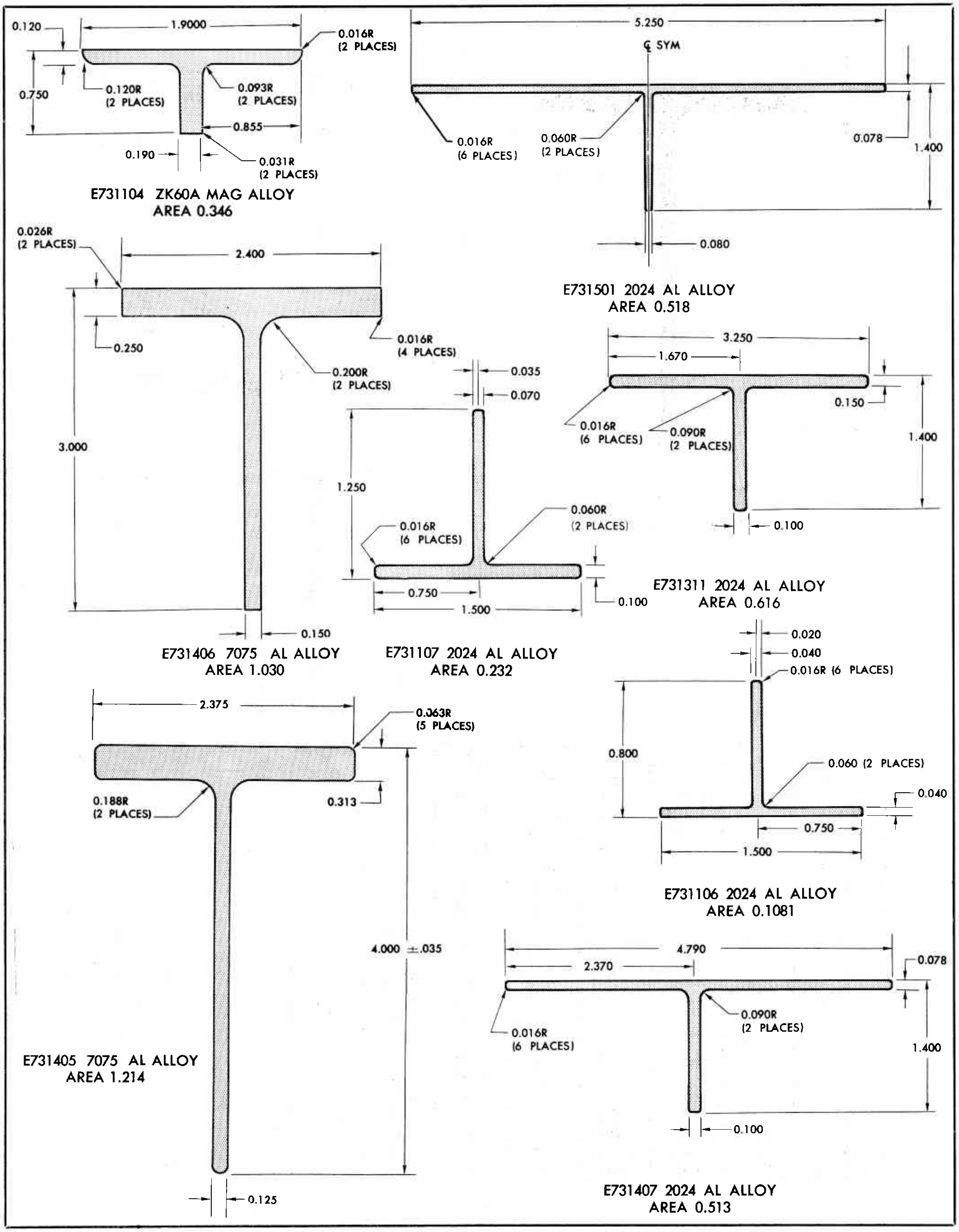
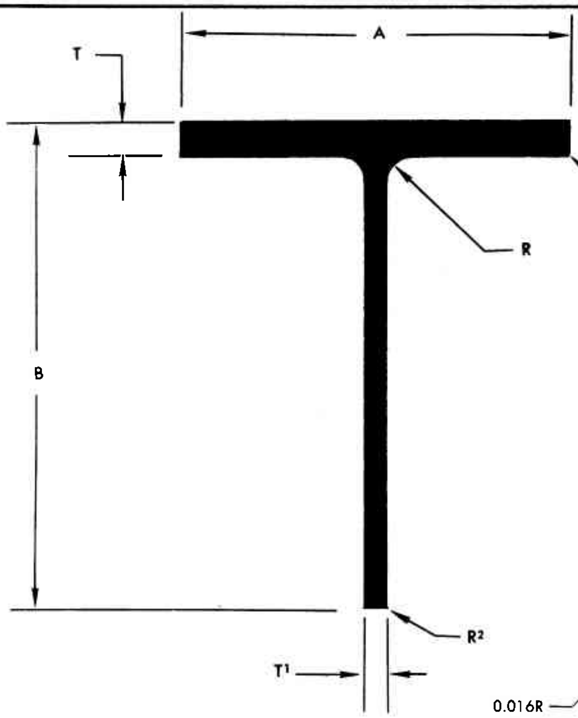
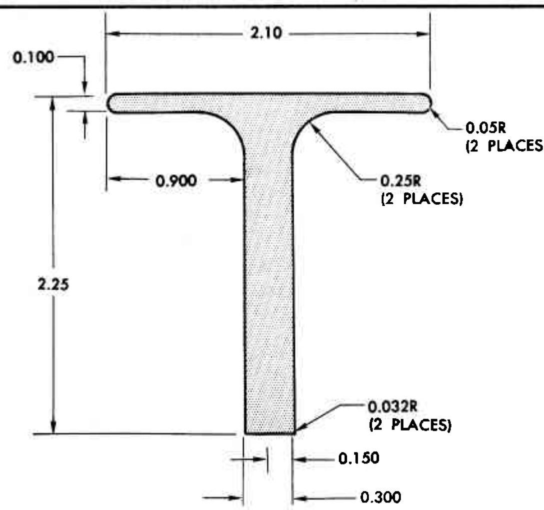


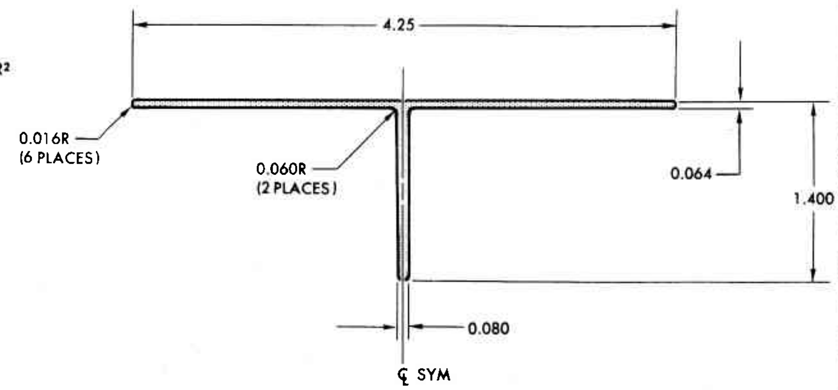
Figure 8-12. Extruded Shapes



USE CHART BELOW



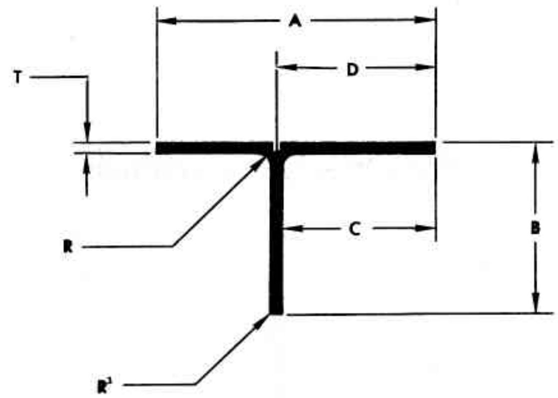
E731212 7075 AL ALLOY
AREA 0.985



E731409 2024 AL ALLOY
AREA 0.380

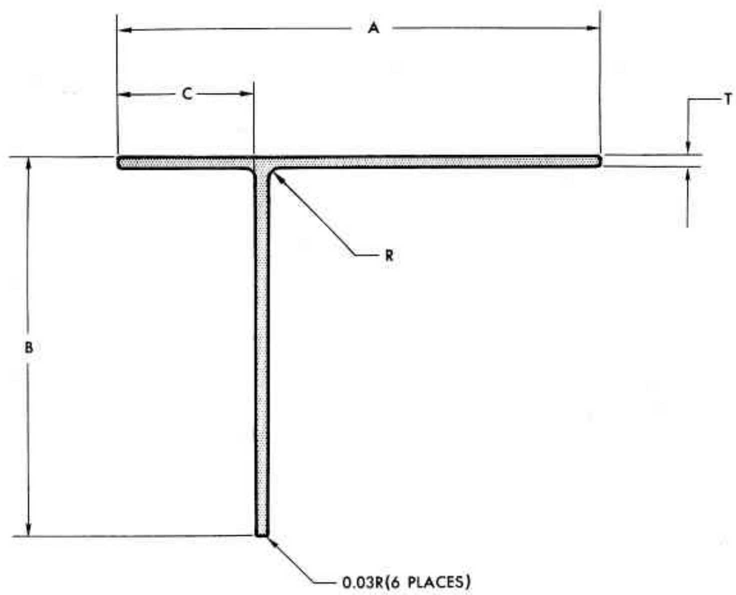
NO.	A	B	T	T^1	R	R^1	R^2	AREA	2024	7075	7178
E711124	1.500	1.350	0.125	0.156	0.125	0.030	0.030	0.429	X		
E711401	3.950	2.000	0.078	0.070	0.094	0.016	0.016	0.412		X	
E721203	2.000	2.600	0.188	0.100	0.125	0.031	0.016	0.663			X
E731101	1.875	1.000	0.100	0.051	0.125	0.016	0.016	0.838			X
E731108	1.640	1.509	0.125	0.188	0.125	0.030	0.030	0.469	X		
E731208	2.750	1.375	0.093	0.187	0.250	0.047	0.047	0.522	X		
E731216	2.500	2.000	0.150	0.060	0.125	0.016	0.030	0.440			X
E731217	2.000	1.625	0.188	0.125	0.090	0.016	0.016	0.551		X	
E731219	2.000	1.000	0.120	0.090	0.060	0.016	0.016	0.314		X	
E731220	2.000	1.250	0.125	0.188	0.062	0.016	0.016	0.461	X		
E731222	1.830	2.256	0.550	0.070	0.120	0.010	0.010	1.153			X
E731223	1.750	2.750	0.125	0.090	0.125	0.016	0.016	0.461		X	
E731303	2.200	3.000	0.188	0.100	0.125	0.031	0.016	0.695		X	
E731308	2.250	2.850	0.156	0.125	0.130	0.031	0.031	0.680			X
E731310	3.000	1.625	0.188	0.125	0.188	0.016	0.016	0.758		X	
E731312	2.000	3.000	0.750	0.500	0.090	0.030	0.030	2.630		X	
E731402	1.830	4.500	0.200	0.090	0.125	0.031	0.016	0.755		X	X
E731403	2.000	3.750	0.188	0.125	0.125	0.047	0.047	0.827			X
E731408	4.000	3.000	0.750	0.500	0.090	0.030	0.030	4.125		X	
E731410	4.630	1.030	0.094	0.110	0.045	0.016	0.016	0.539			X
E731502	5.000	2.800	0.200	0.800	0.120	0.030	0.030	2.470		X	
E732701	2.813	6.500	0.080	0.094	0.156	0.031	0.031	0.848			X

Figure 8-13. Extruded Shapes



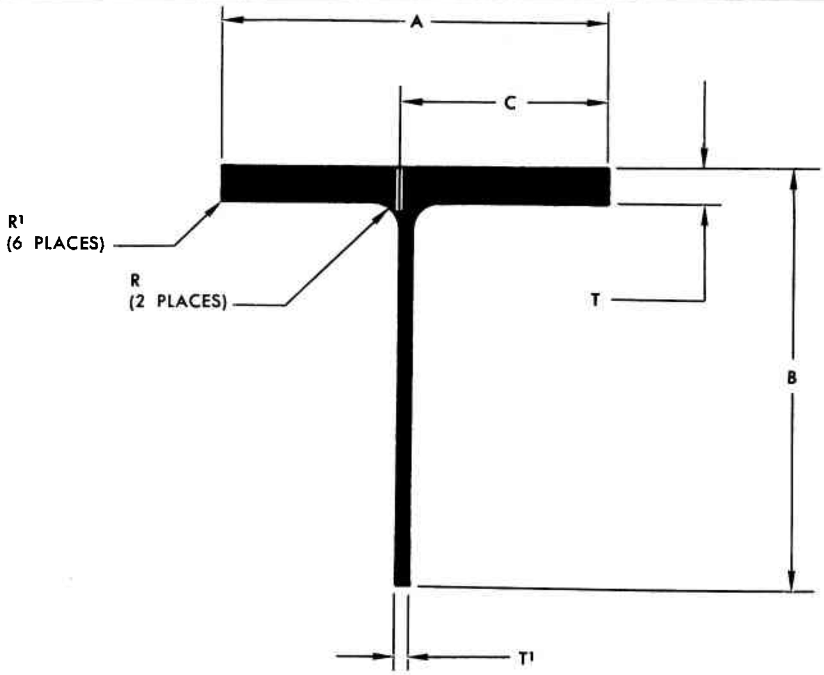
TEE—90°, FLANGE WIDTH UNEQUAL, THICKNESS EQUAL

NO.	A	B	C	D	T	R	R[1]	AREA	2024	7075	7178
E721101	1.430	0.900		0.805	0.051	0.060	0.016	0.116			X
E721102	1.750	0.800		1.000	0.051	0.090	0.016	0.125	X		
E721103	1.620	0.830	1.020		0.060	0.090	0.016	0.150		X	
E721205	1.900	1.600	1.747		0.070	0.070	0.016	0.242			X
E721206	2.670	0.950	0.980		0.080	0.090	0.016	0.218	X		
E721301	2.600	2.550	1.465		0.070	0.070	0.016	0.361		X	X
E721302	4.000	1.500	2.312		0.094	0.094	0.016	0.510			X
E721303	3.860	2.280	2.070		0.080	0.190	0.016	0.500			X
E721401	4.450	2.410	2.380		0.125	0.160	0.016	0.860		X	
E722201	2.250	1.300	1.125		0.051	0.091	0.016	0.182			X
E722601	5.625	3.938		3.750	0.250	0.188	0.031	2.343			X

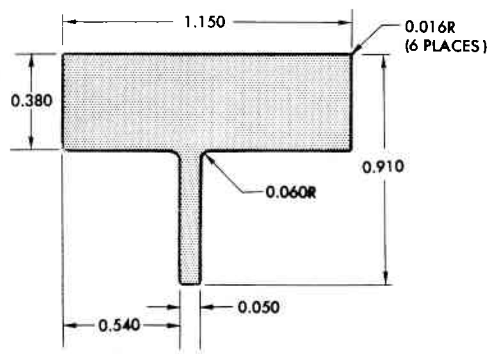


E721217 7075 AND 7178 AL ALLOY
AREA 0.271

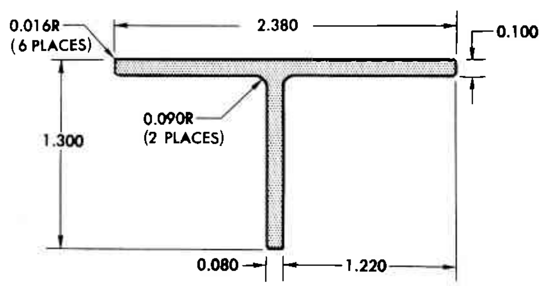
Figure 8-14. Extruded Shapes



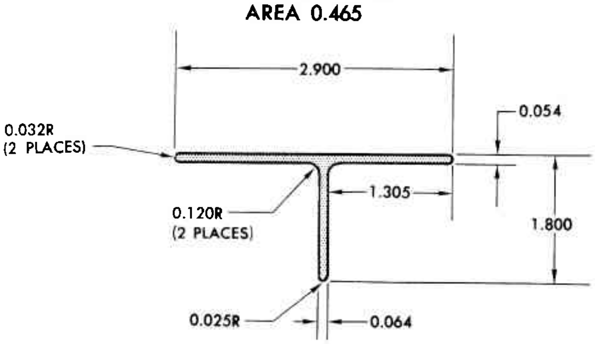
NO.	A	B	C	T	T1	R	R1	AREA	7075	7178
E741214	1.900	2.220	0.950	0.210	0.100	0.120	0.016	0.606		X
E741215	1.900	2.200	0.950	0.180	0.080	0.120	0.016	0.509		X
E741310	1.900	3.000	0.950	0.120	0.080	0.125	0.015	0.461		X
E741311	1.900	3.060	0.950	0.180	0.080	0.120	0.016	0.578		X
E741312	1.960	2.950	1.220	0.150	0.100	0.090	0.016	0.577		X
E741502	3.580	4.250	2.140	0.450	0.200	0.125	0.031	2.374	X	



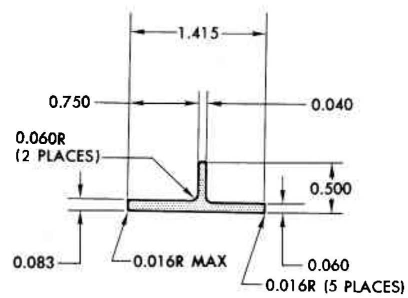
E741108 7075 AL ALLOY
AREA 0.465



E741216 2024 AL ALLOY
AREA 0.337



E741213 7178 AL ALLOY
AREA 0.272



E741103 7178 AL ALLOY
AREA 0.107

Figure 8-15. Extruded Shapes

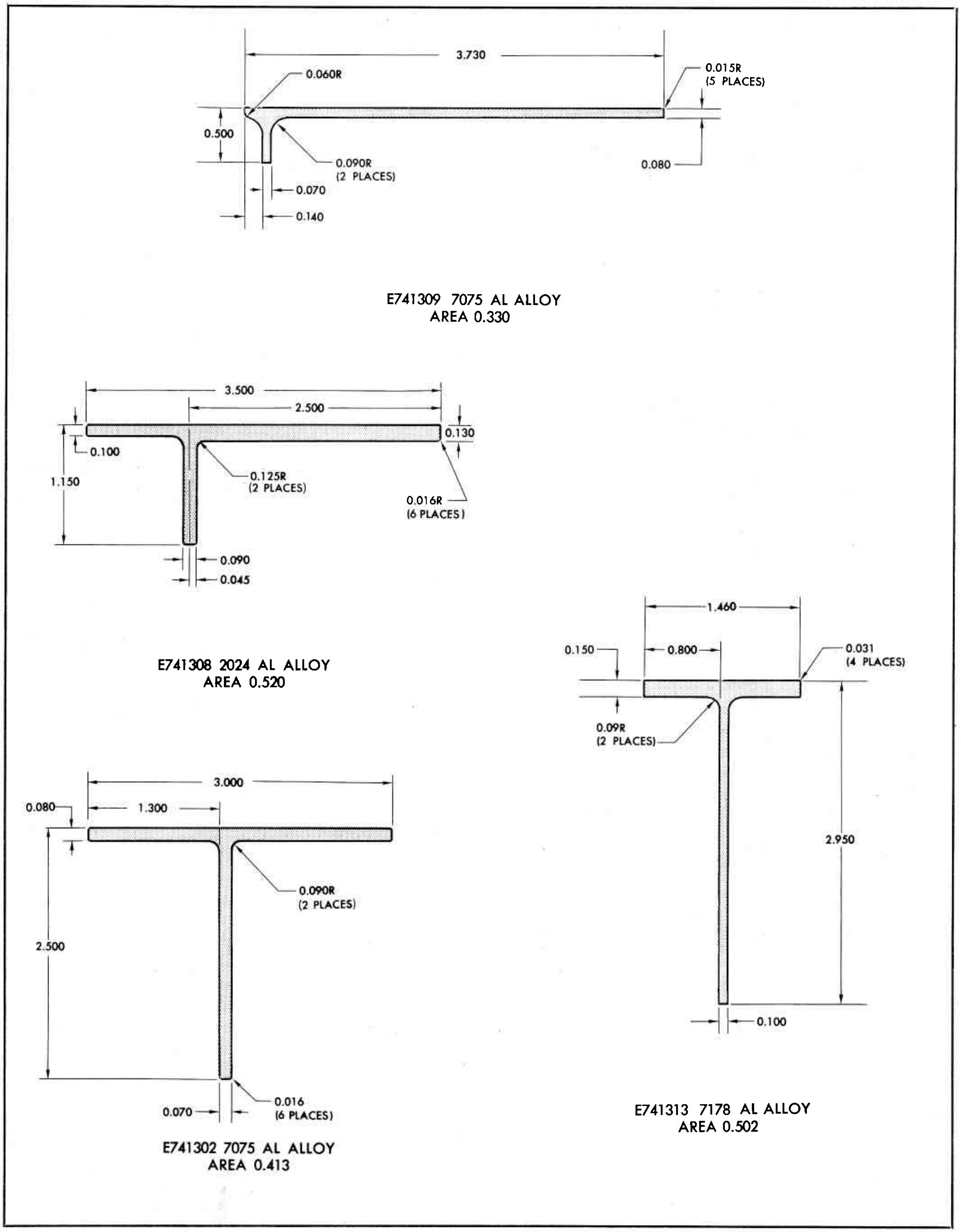
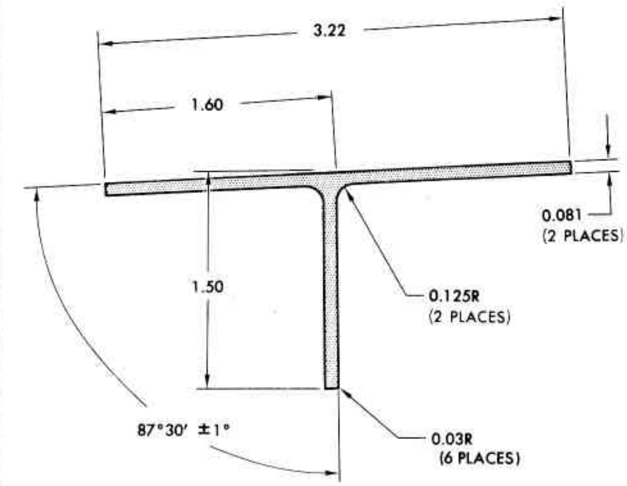
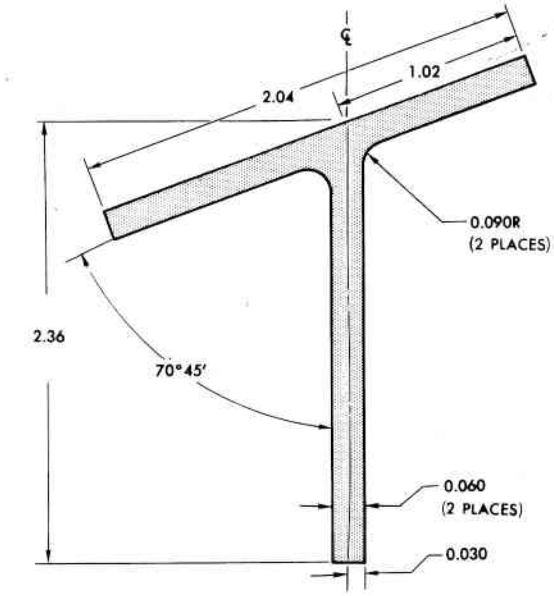


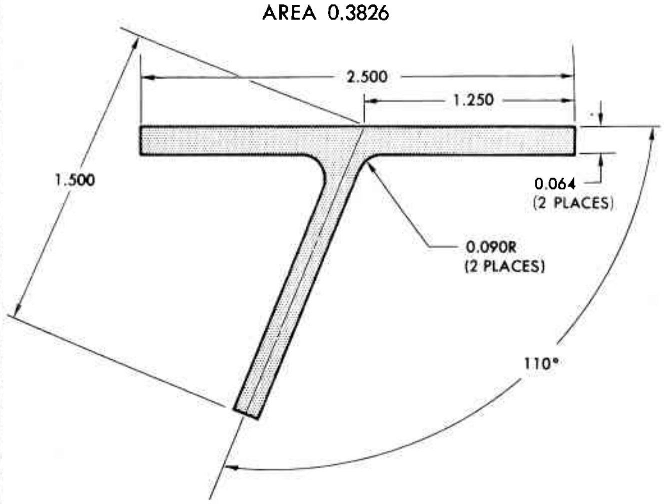
Figure 8-16. Extruded Shapes



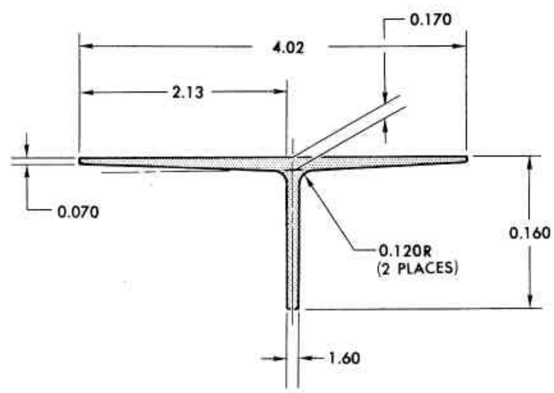
TEE—FLANGE WIDTH UNEQUAL,
THICKNESS EQUAL
E821403 7075 AL ALLOY
AREA 0.3826



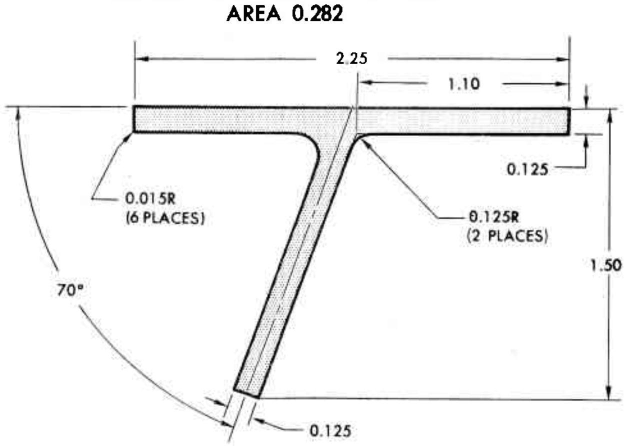
TEE—FLANGE WIDTH AND THICKNESS EQUAL
E812309 7178 AL ALLOY
AREA 0.264



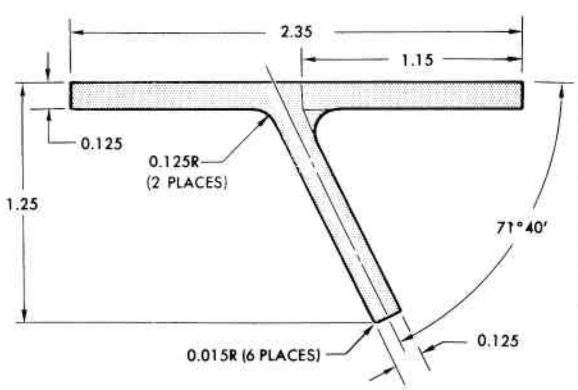
TEE—FLANGE WIDTH AND THICKNESS EQUAL,
ANGLE UNEQUAL
E812305 7075 AND 7178 AL ALLOY
AREA 0.282



TEE—FLANGE WIDTH AND THICKNESS UNEQUAL
E741406
AREA 0.716

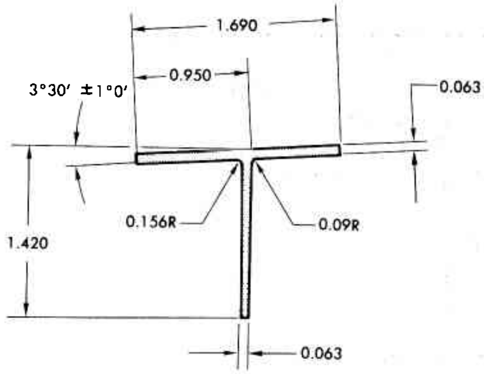


TEE—ANGLE, FLANGE WIDTH UNEQUAL,
THICKNESS EQUAL
E821207 2024 AL ALLOY
AREA 0.470

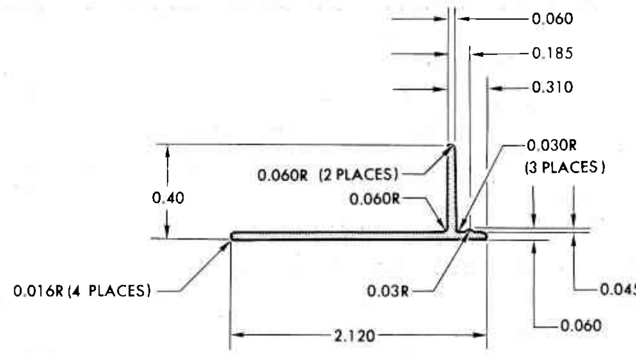


TEE—ANGLED, FLANGE WIDTH UNEQUAL,
THICKNESS EQUAL
E821208 2024 AL ALLOY
AREA 0.44

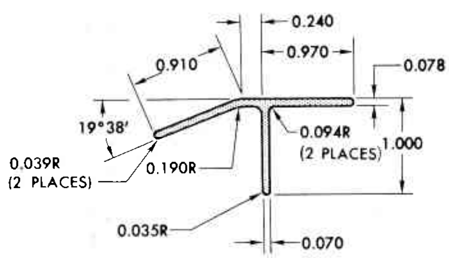
Figure 8-17. Extruded Shapes



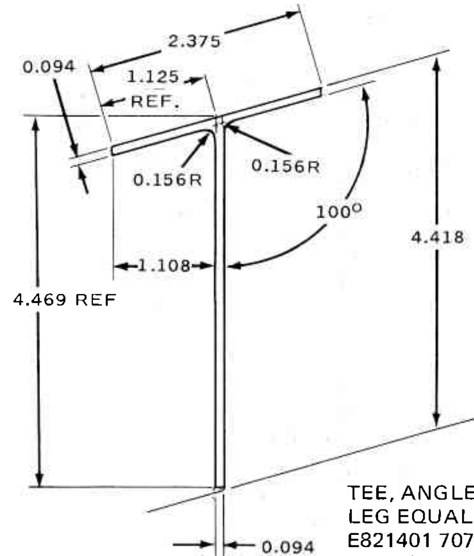
TEE—ANGLED, FLANGE WIDTH AND THICKNESS EQUAL
E811103 2024 AL ALLOY
AREA 0.21



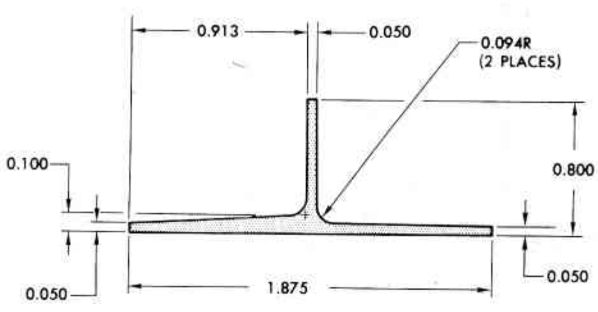
TEE—SPECIAL
E701211 2024 AL ALLOY
AREA 0.149



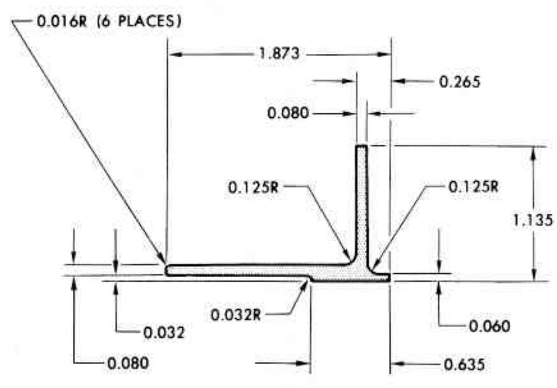
TEE—90° MISC.
E751404 7178 AL ALLOY
AREA 2.047



TEE, ANGLED UNEQUAL LEG EQUAL THICKNESS
E821401 7075T6AL ALLOY
41129(ALCOA DIE) AREA 0.646.

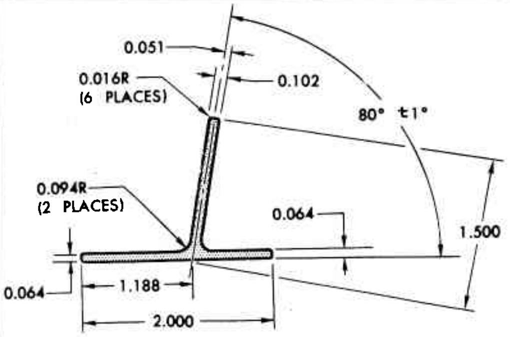


TEE—TAPERED
E701105
AREA 0.157

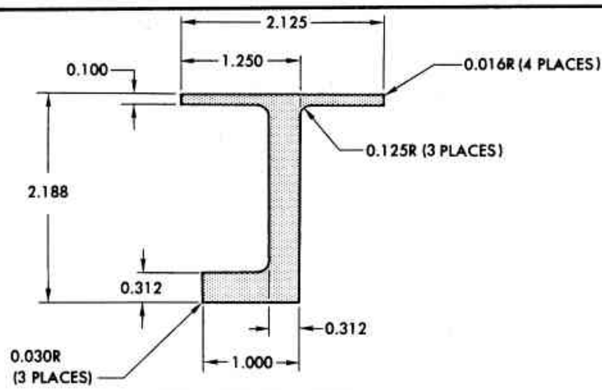


TEE—NON UNIFORM
E701202 7178 AL ALLOY
AREA 0.252

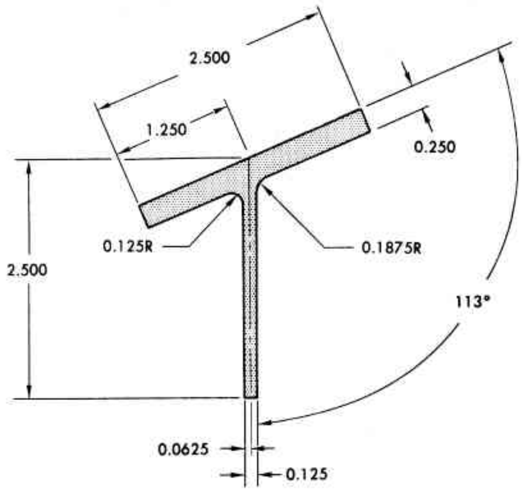
Figure 8-18. Extruded Shapes



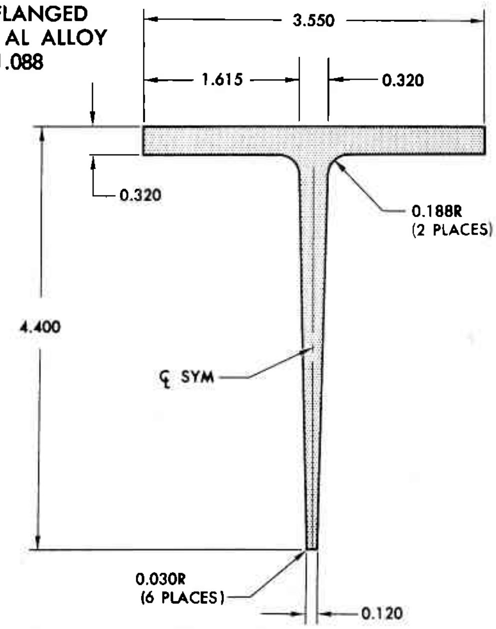
TEE—ANGLED, FLANGE WIDTH AND THICKNESS UNEQUAL
E841207 7178 AL ALLOY
AREA 0.2845



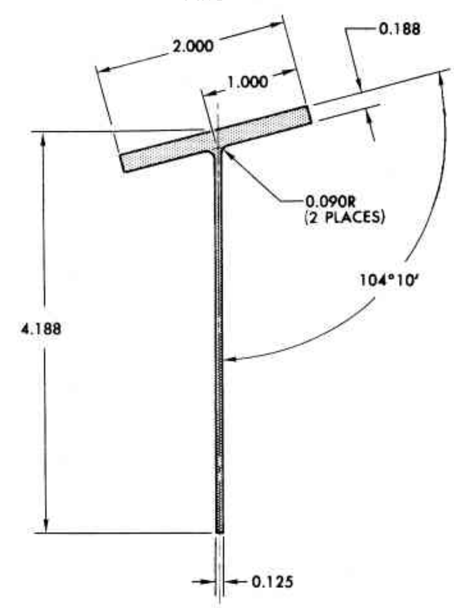
TEE—90° FLANGED
E771206 7075 AL ALLOY
AREA 1.088



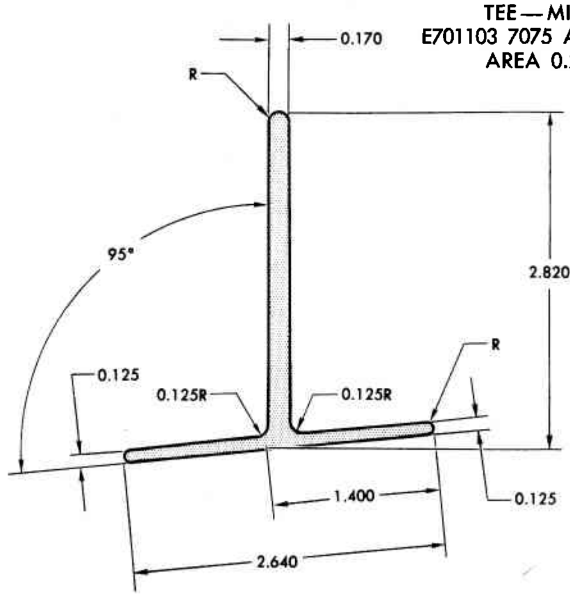
TEE—ANGLED
E831304 7178 AL ALLOY
AREA 0.915



TEE—MISC.
E701103 7075 AL ALLOY
AREA 0.231



TEE—ANGLED
E832501 7178 AL ALLOY
AREA 0.755



TEE—FLANGE WIDTH EQUAL
E811304 7075 AL ALLOY
AREA 0.789

Figure 8-19. Extruded Shapes

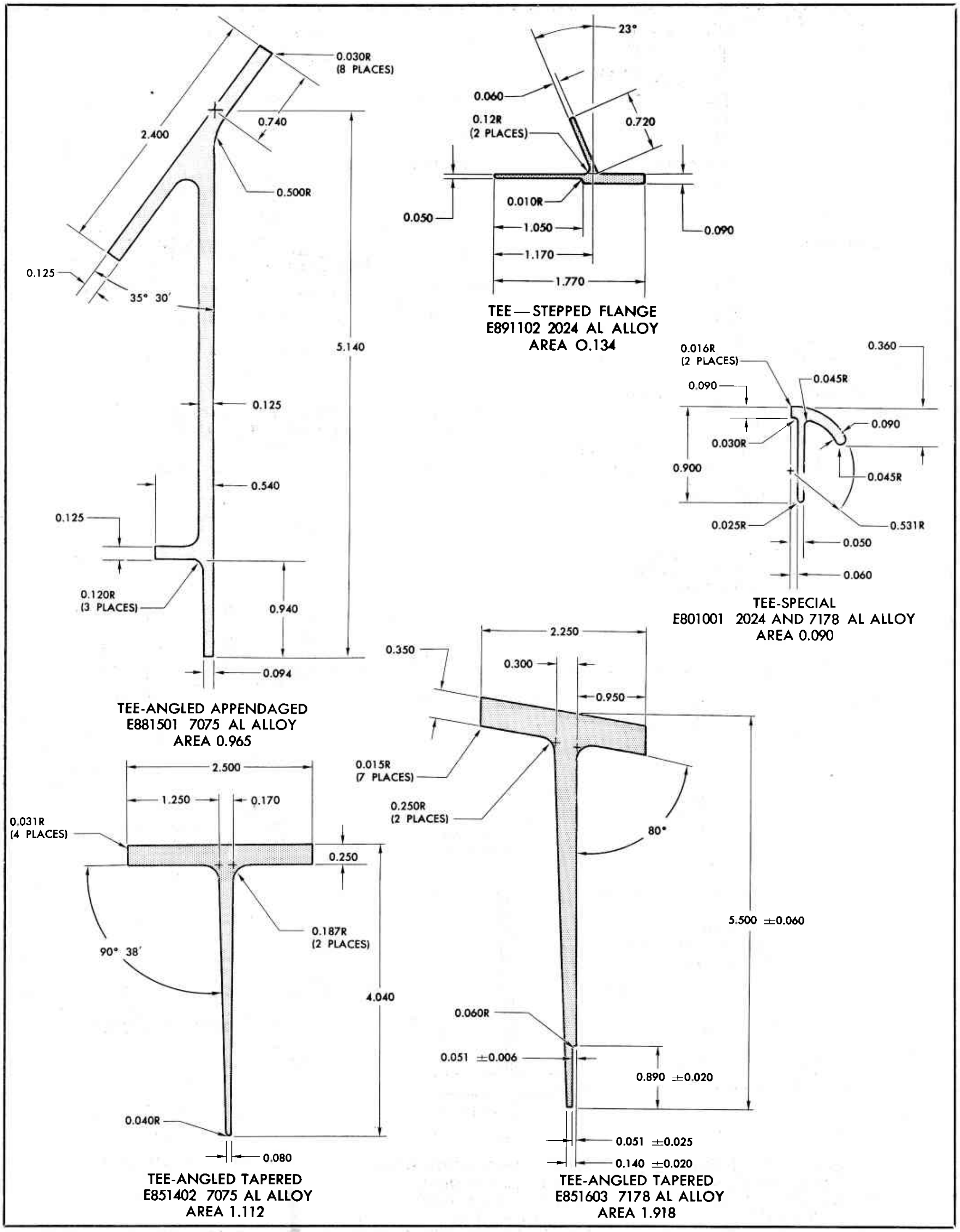


Figure 8-20. Extruded Shapes

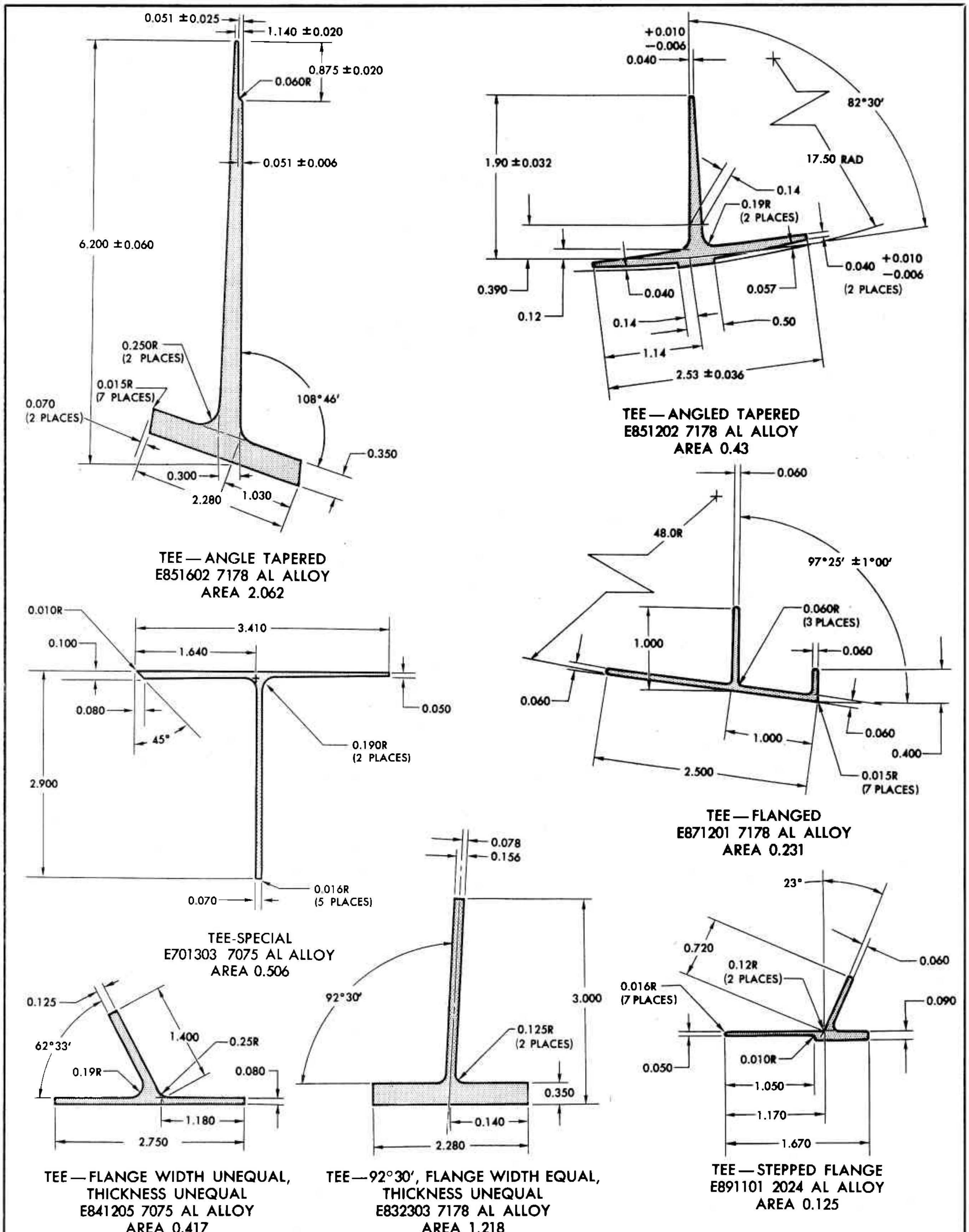
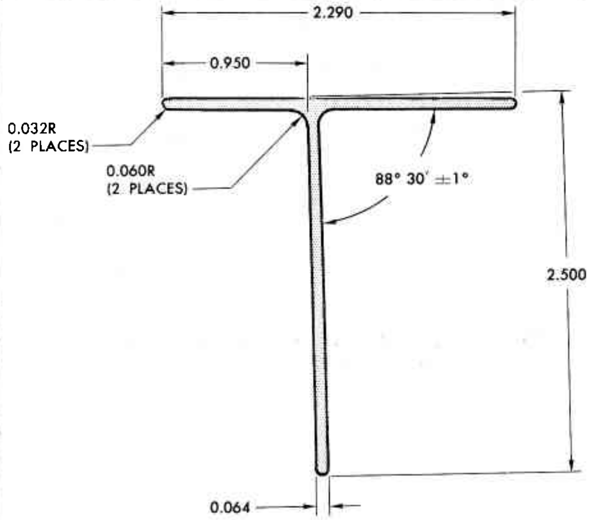
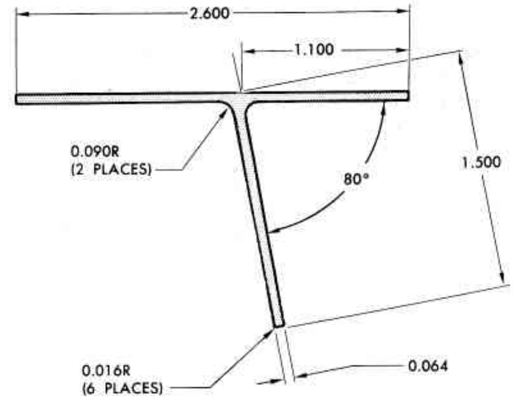


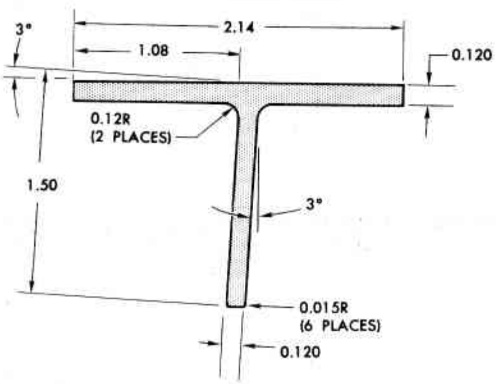
Figure 8-21. Extruded Shapes



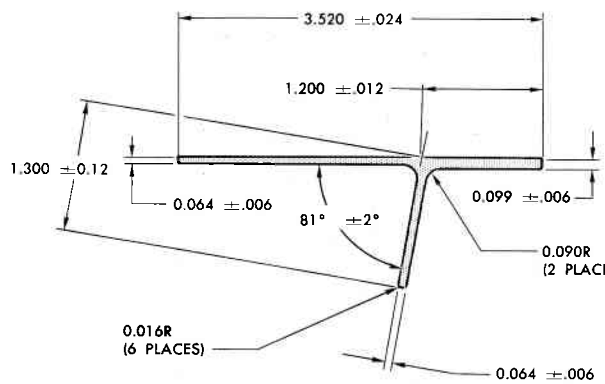
TEE-ANGLED, FLANGE WIDTH UNEQUAL, THICKNESS EQUAL
E821203 7075 AL ALLOY
AREA 0.303



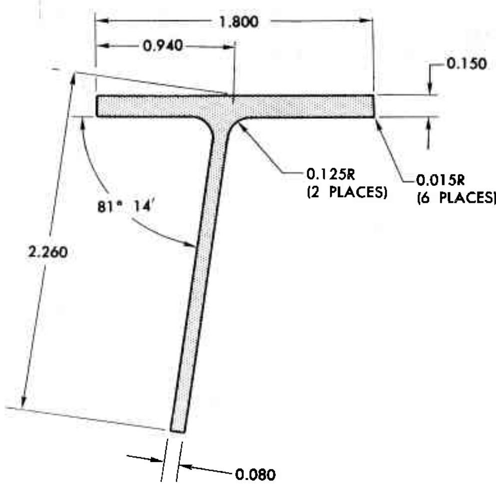
TEE-ANGLED, FLANGE WIDTH UNEQUAL, THICKNESS EQUAL
E821205 7178 AL ALLOY
AREA 0.262



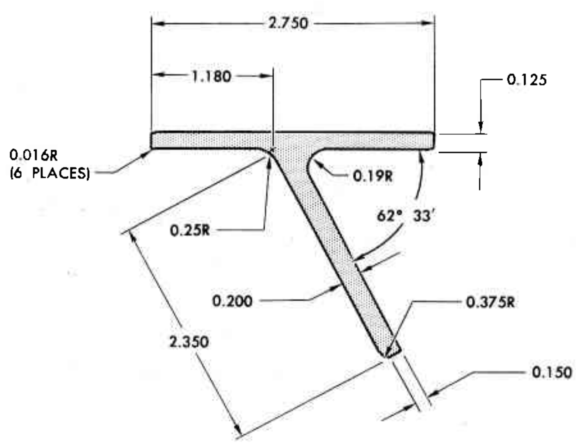
TEE-ANGLED,
FLANGE WIDTH UNEQUAL, THICKNESS EQUAL
E821206 7178 AL ALLOY
AREA 0.420



TEE-ANGLED, FLANGE WIDTH AND THICKNESS UNEQUAL
E841305 7075 AL ALLOY
AREA 0.343

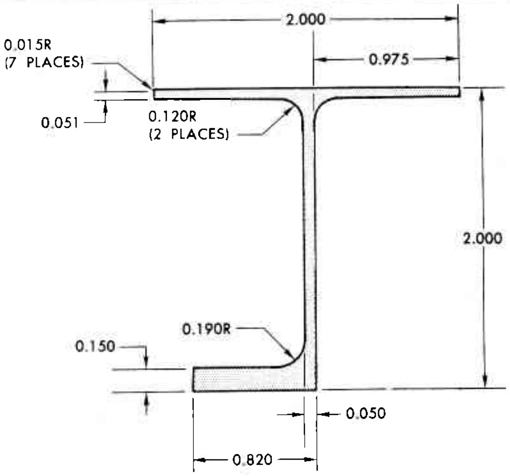


TEE-ANGLED,
FLANGE WIDTH EQUAL, THICKNESS UNEQUAL
E841203 7178 AL ALLOY
AREA 0.450

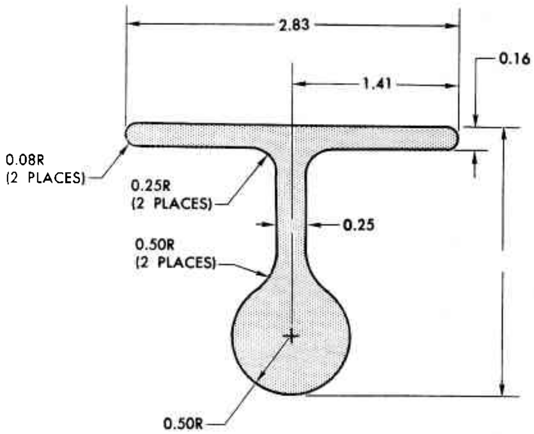


TEE-ANGLED, FLANGE WIDTH AND THICKNESS UNEQUAL
E841306 7075 AL ALLOY
AREA 0.771

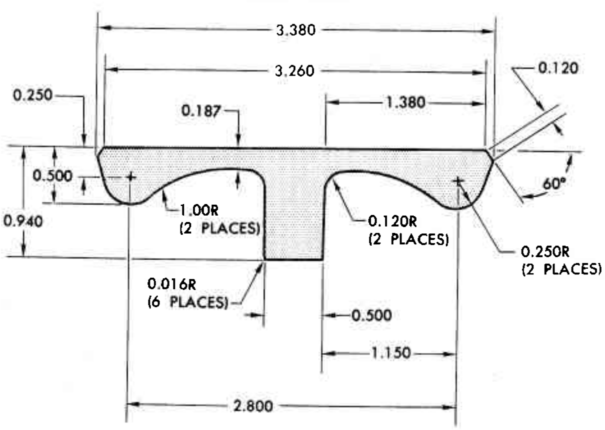
Figure 8-22. Extruded Shapes



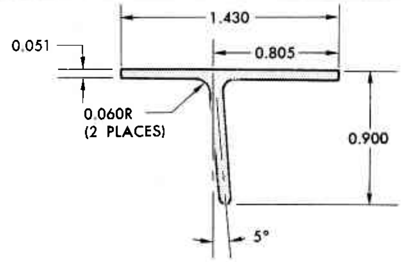
TEE—90° FLANGED
E781202 7178 AL ALLOY
AREA 0.317



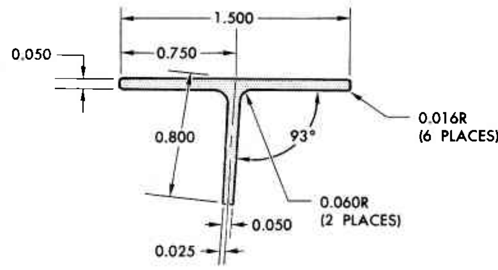
TEE - BULBED
E761301 7075 AL ALLOY
AREA 1.345



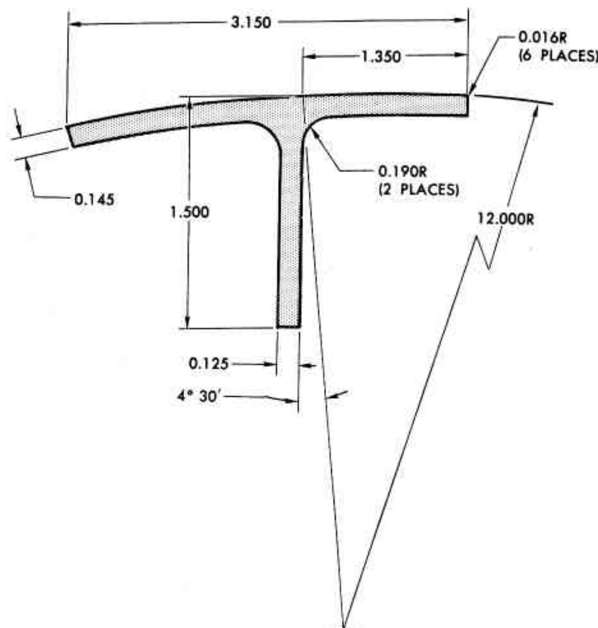
TEE - BULBED
E761302 7178 AL ALLOY
AREA 1.366



TEE-ANGLED, FLANGE WIDTH UNEQUAL, THICKNESS EQUAL
E811102 7178 AL ALLOY
AREA 0.116



TEE-ANGLED, FLANGE WIDTH UNEQUAL, THICKNESS EQUAL
E811104 7075 AL ALLOY
AREA 0.112



TEE-ANGLED, FLANGE WIDTH UNEQUAL, THICKNESS EQUAL
E821204 2024 AL ALLOY
AREA 0.490

Figure 8-23. Extruded Shapes

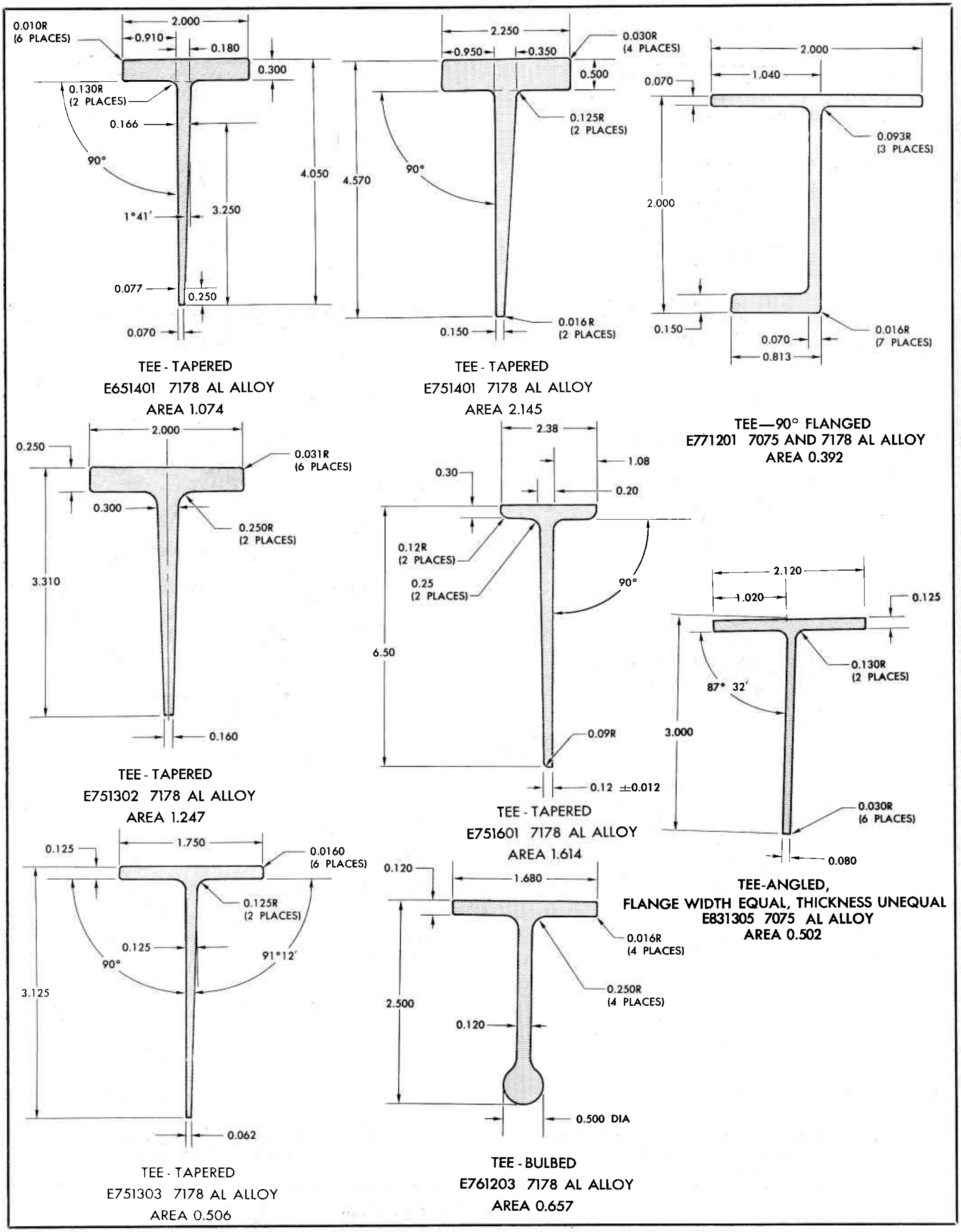
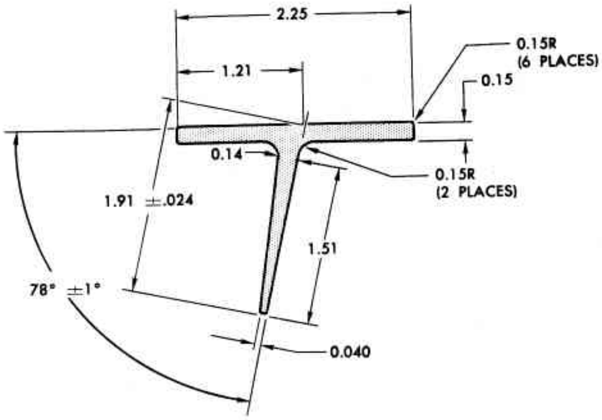
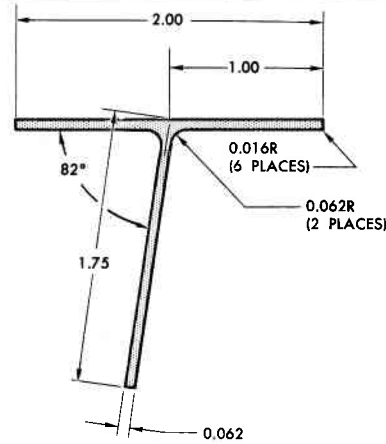


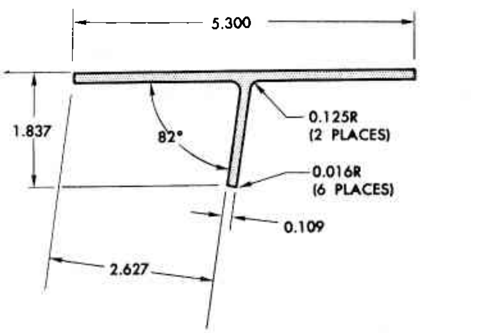
Figure 8-24. Extruded Shapes



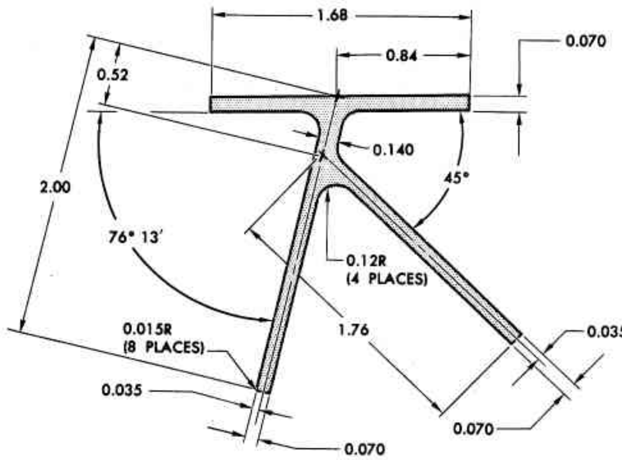
TEE-ANGLED, TAPERED
E851203 7178 AL ALLOY
AREA 0.540



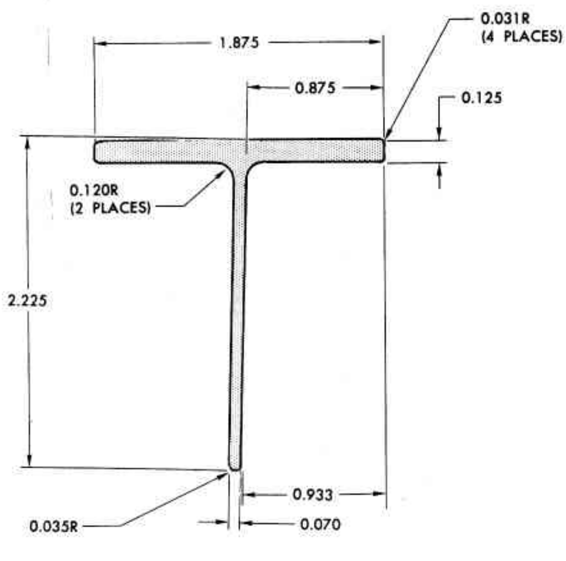
TEE-ANGLED, FLANGE WIDTH UNEQUAL, THICKNESS EQUAL
E811203 7075 AL ALLOY
AREA 0.229



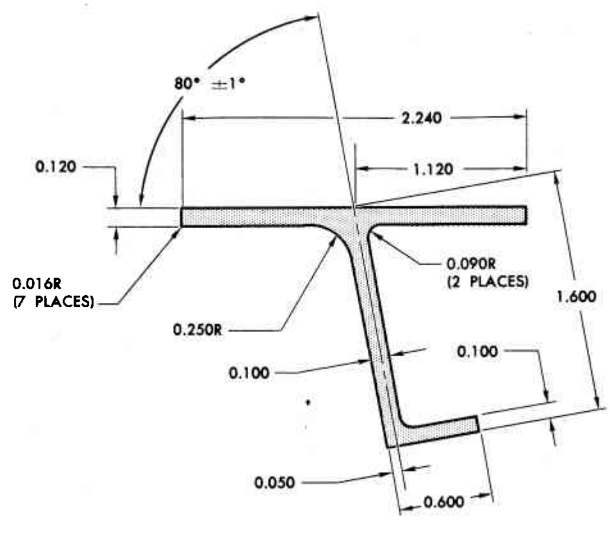
TEE-ANGLED,
FLANGE WIDTH AND THICKNESS UNEQUAL
E821501 7075 AL ALLOY
AREA 0.774



TEE-ANGLED APPENDAGED
E881202 7075 AL ALLOY
AREA 0.425

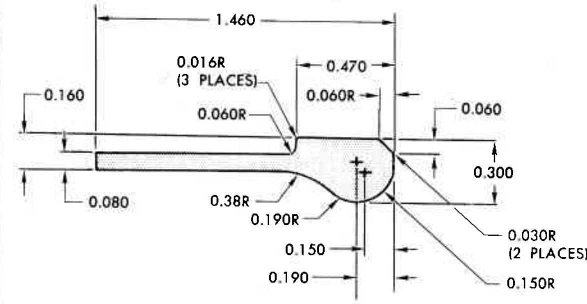


TEE-ANGLED, FLANGE WIDTH AND THICKNESS UNEQUAL
E841204 7075 AL ALLOY
AREA 0.386

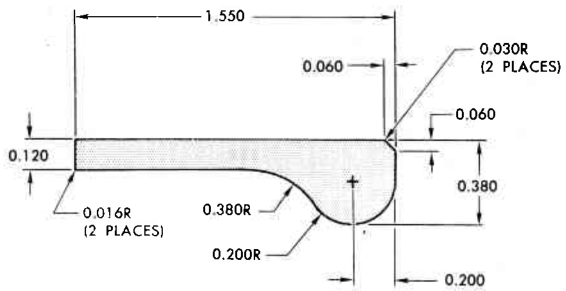


TEE-ANGLED APPENDAGED
E881201 7178 AL ALLOY
AREA 0.431

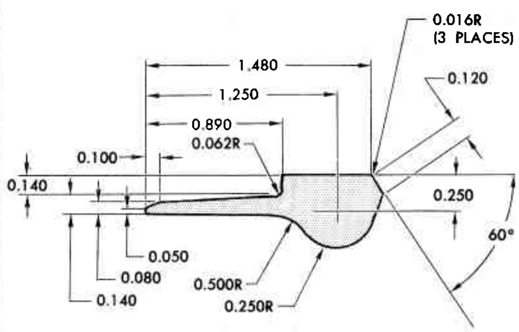
Figure 8-25. Extruded Shapes



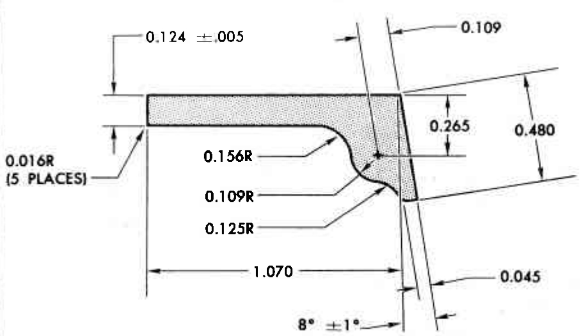
BAR-HINGE
E491133 7075 AL ALLOY
AREA 0.198



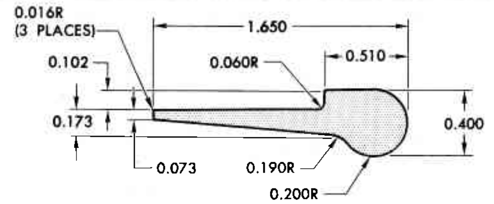
BAR-HINGE
E491132 7075 AL ALLOY
AREA 0.282



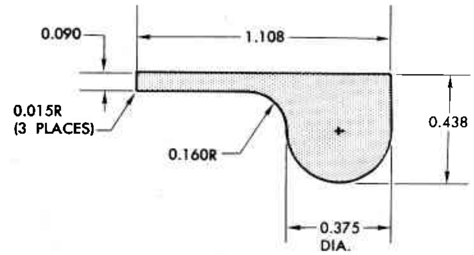
BAR—HINGE
E491131 7178 AL ALLOY
AREA 0.362



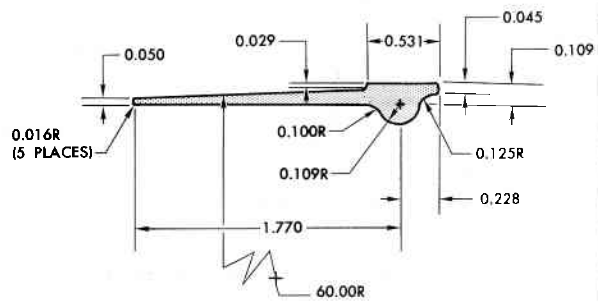
BAR—HINGE
E491125 2024 AND 7075 AL ALLOY
AREA 0.196



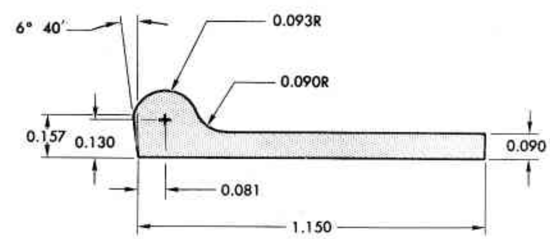
BAR—HINGE
E491130 7075 AL ALLOY
AREA 0.302



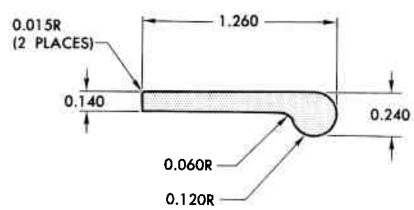
BAR—HINGE
E491127 ZK60A MAG. ALLOY
AREA 0.254



BAR—HINGE
E491124 2024 AND 7075 AL ALLOY
AREA 0.169

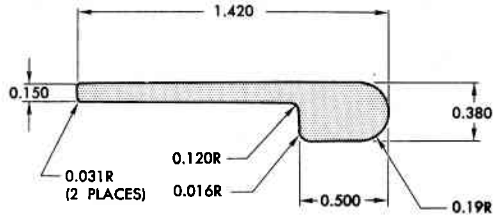


BAR—HINGE
E491115 7075 AL ALLOY
AREA 0.135

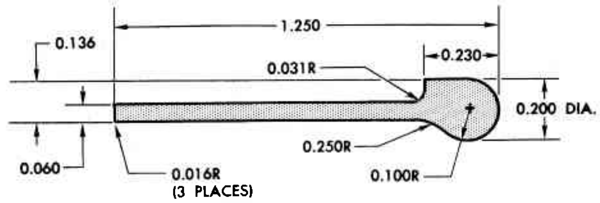


BAR—HINGE
E491128 7178 AL ALLOY
AREA 193

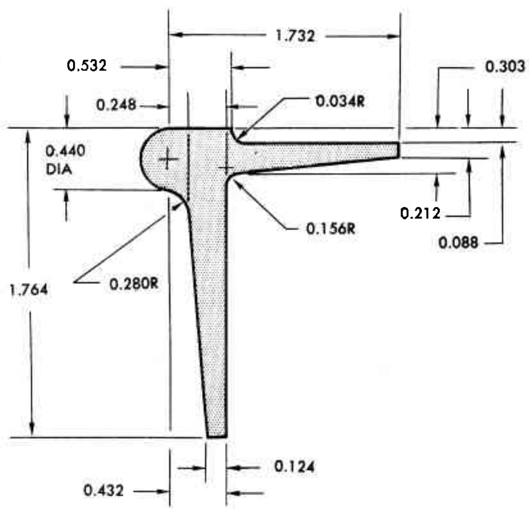
Figure 8-26. Extruded Shapes



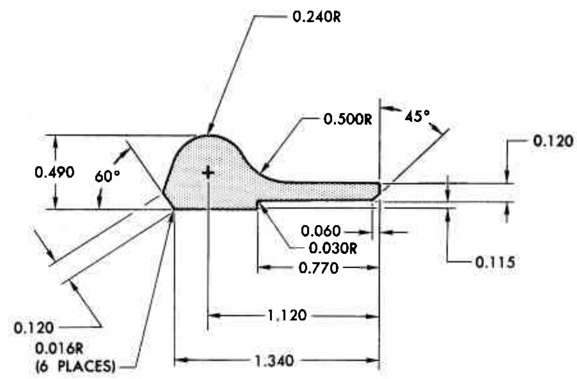
BAR—HINGE
E491129 2024 AL ALLOY
AREA 0.287



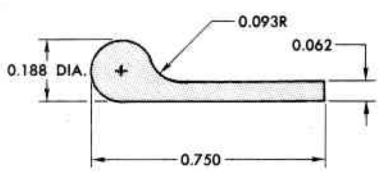
BAR—HINGE
E491119 7075 AL ALLOY
AREA 0.096



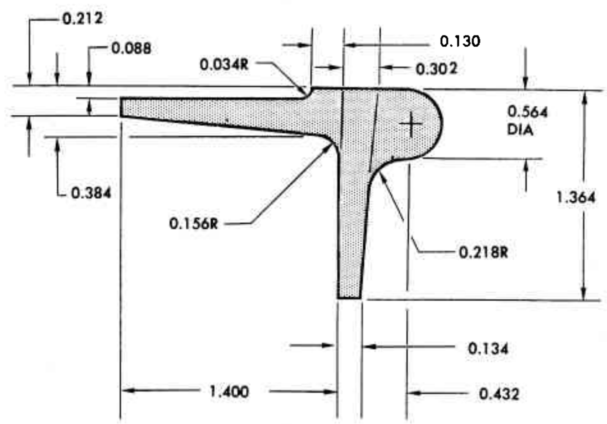
ANGLE—90°, BULBED E161203 7075 AL ALLOY AREA 0.78



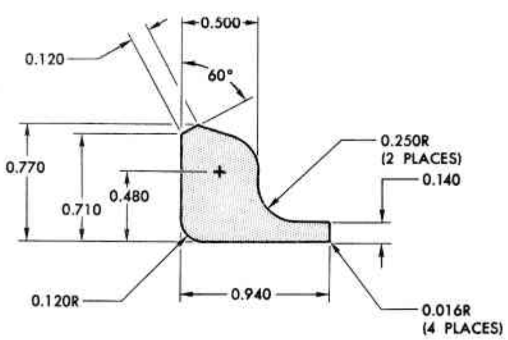
BAR—HINGE
E491116 7178 AL ALLOY
AREA 0.379



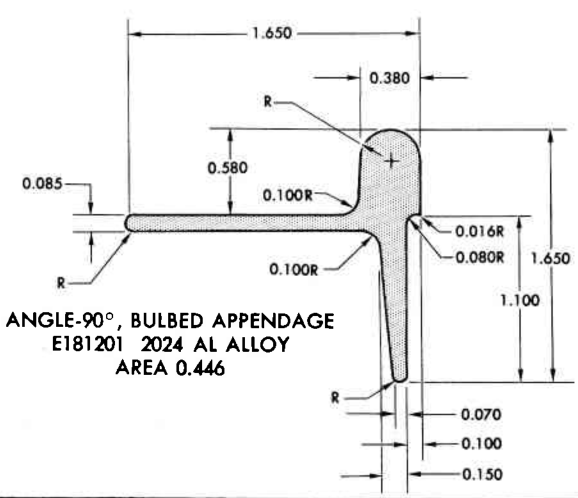
BAR—HINGE
E491002 2024 AL ALLOY
AREA 0.066



ANGLE—90°, BULBED E161202 7075 AL ALLOY AREA 0.82

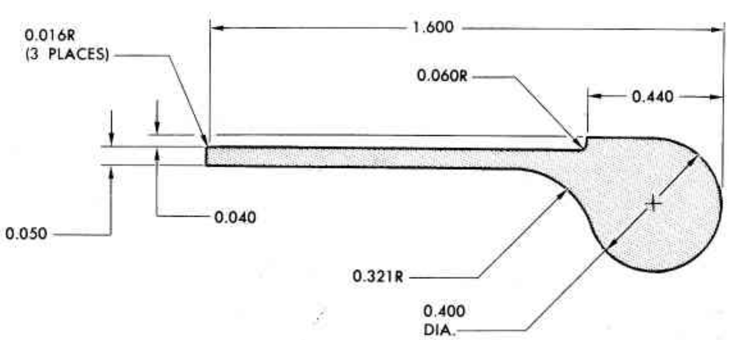


BAR—HINGE
E491003 7178 AL ALLOY
AREA 0.280

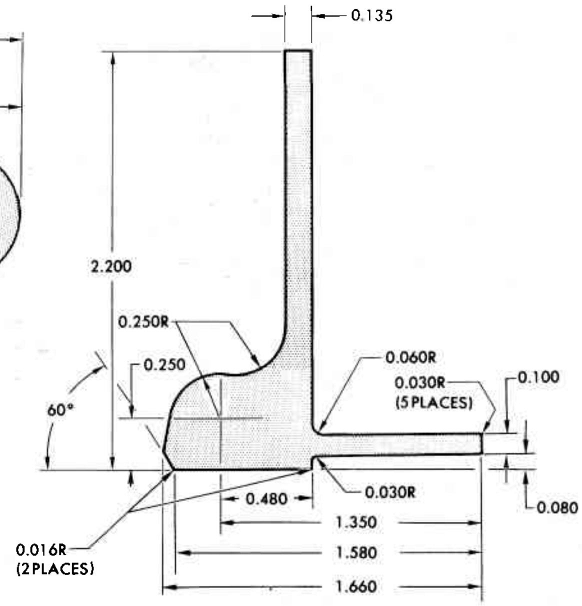


ANGLE-90°, BULBED APPENDAGE
E181201 2024 AL ALLOY
AREA 0.446

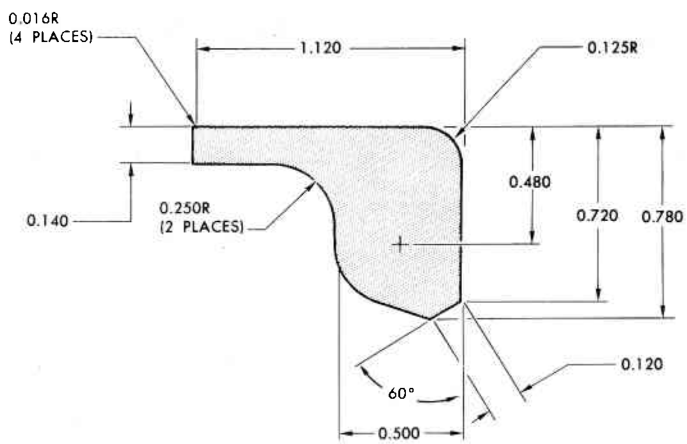
Figure 8-27. Extruded Shapes



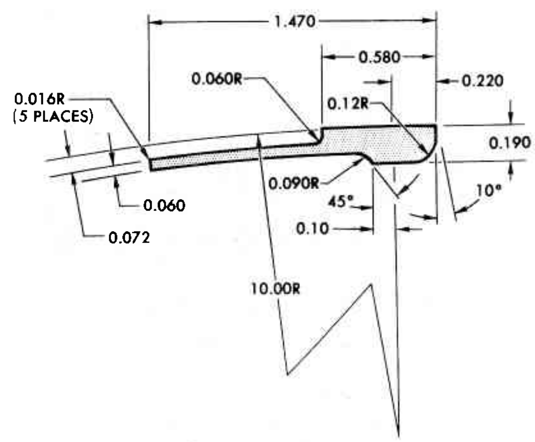
BAR—HINGE
E491136 7178 AL ALLOY
AREA 0.209



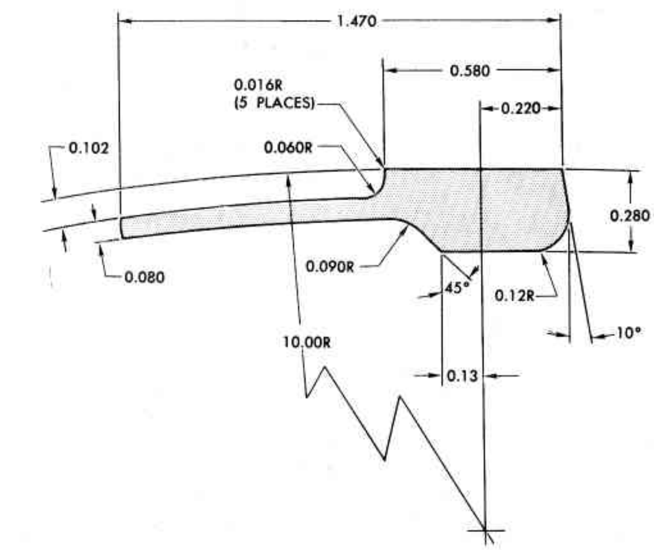
BAR—HINGE
E491203 7178 AL ALLOY
AREA 0.686



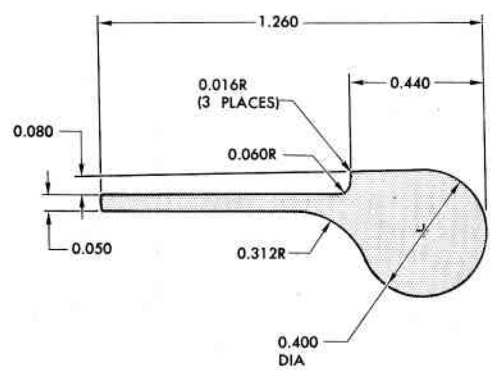
BAR—HINGE
E491138 7178 AL ALLOY
AREA 0.455



BAR—CURVED HINGE
E491106 2024 AL ALLOY
AREA 0.1473

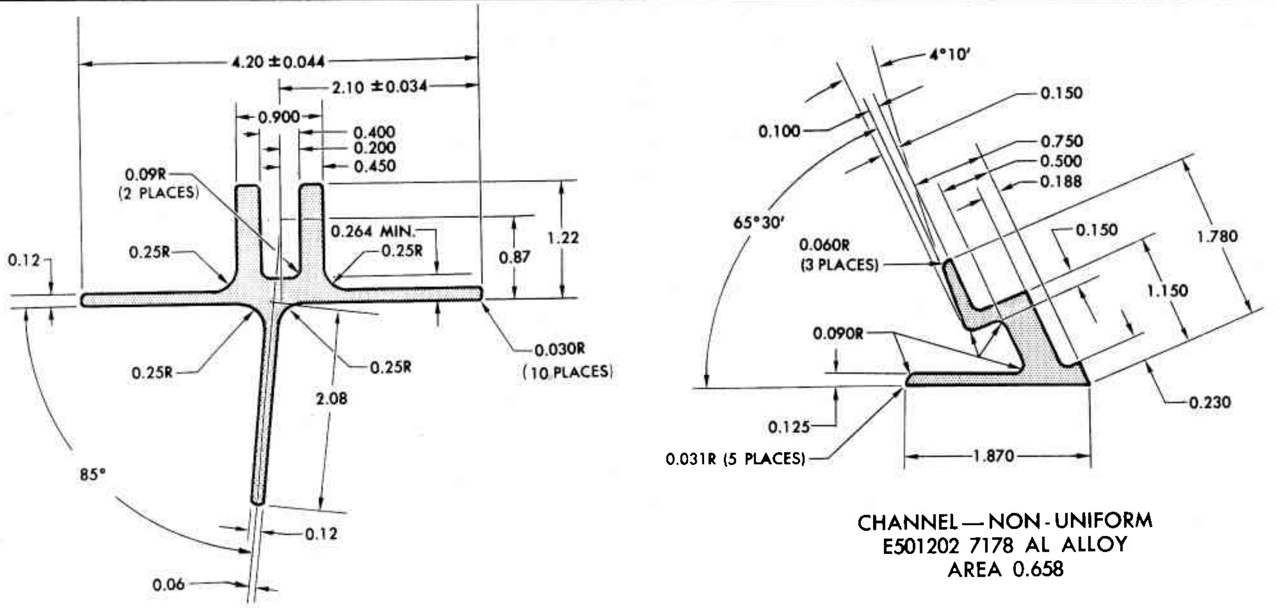


BAR-HINGE
E491134 7178 AL ALLOY
AREA 0.230



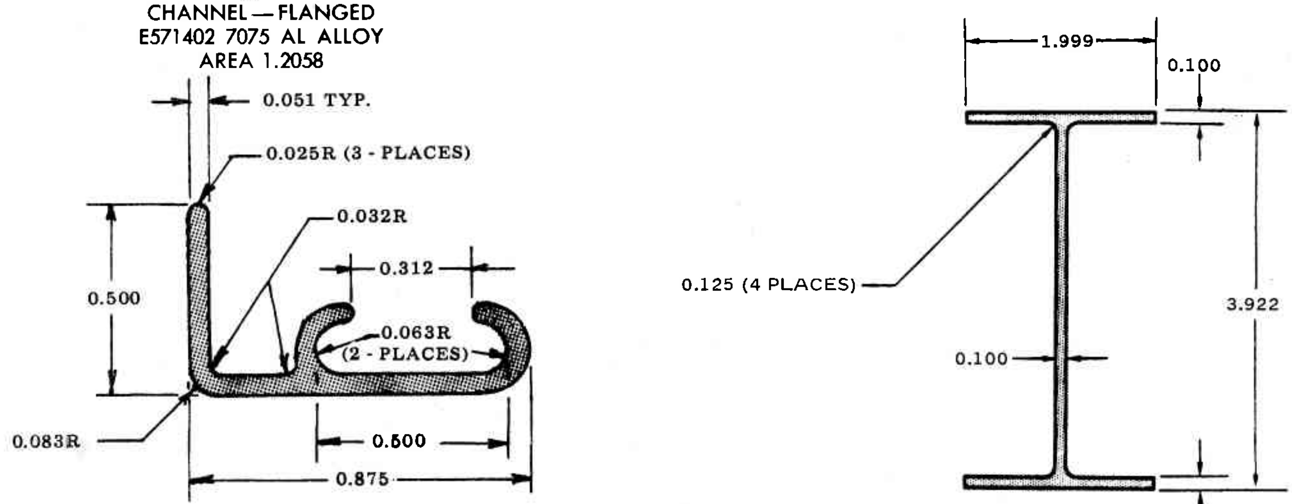
BAR-HINGE
E491135 7178 AL ALLOY
AREA 0.247

Figure 8-28. Extruded Shapes



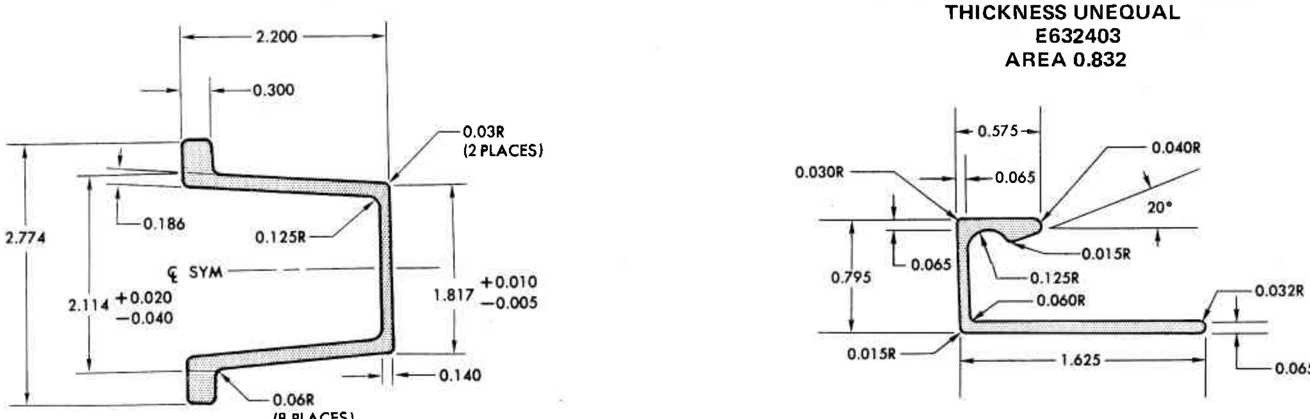
CHANNEL — FLANGED
E571402 7075 AL ALLOY
AREA 1.2058

CHANNEL — NON-UNIFORM
E501202 7178 AL ALLOY
AREA 0.658



CHANNEL – RETAINER
7075-T6511 AREA 0.088
NO. 60-919 (29321)

CHANNEL - FLANGE WIDTH EQUAL, THICKNESS UNEQUAL
E632403
AREA 0.832



CHANNEL — MISC.
E501206 7075 AL ALLOY
AREA 1.161

CHANNEL-SPECIAL
E501107 7075 AL ALLOY
AREA 0.227

Figure 8-29. Extruded Shapes

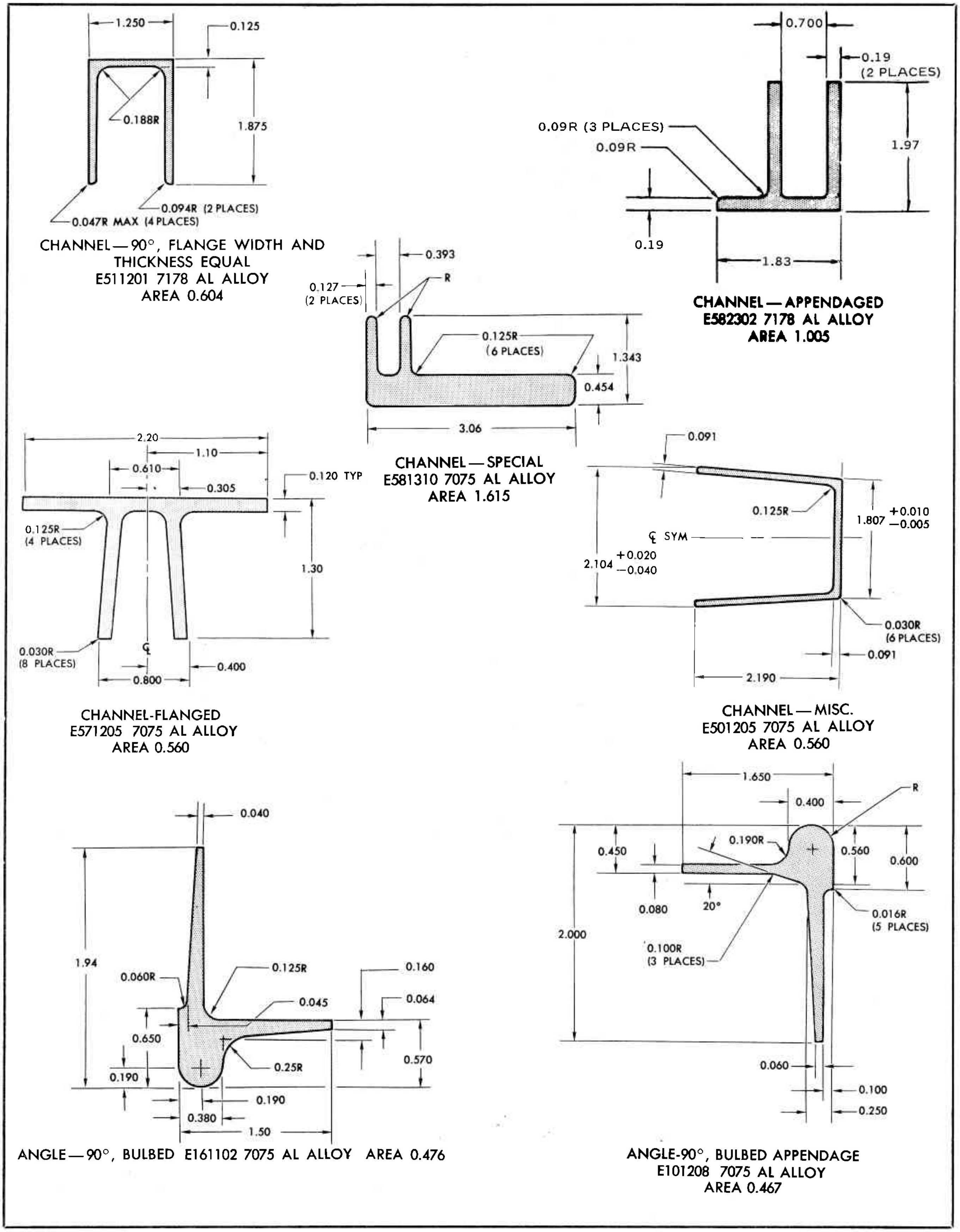
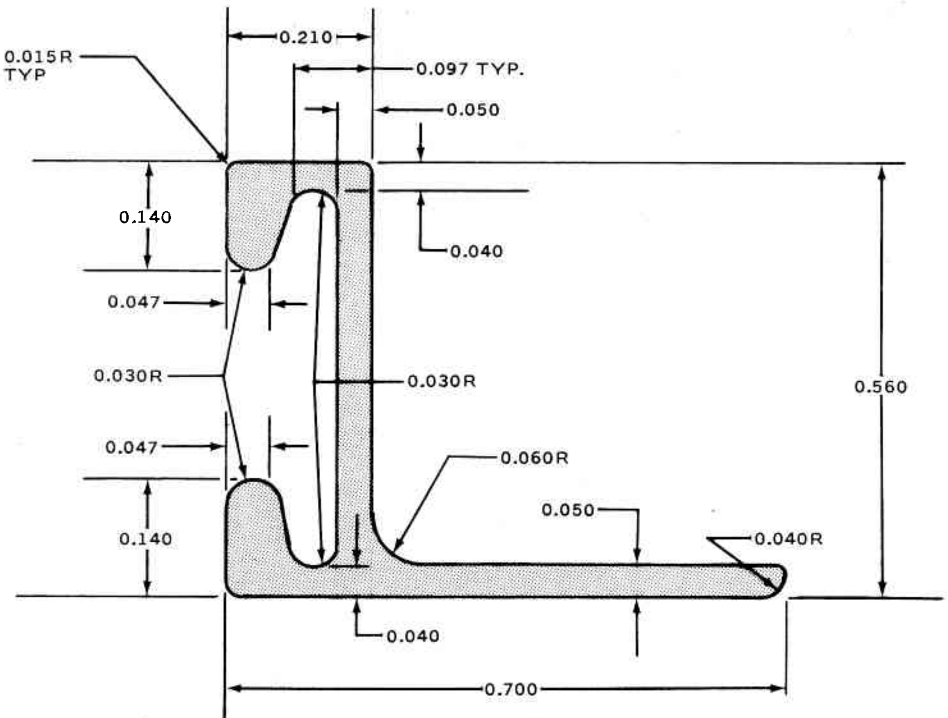
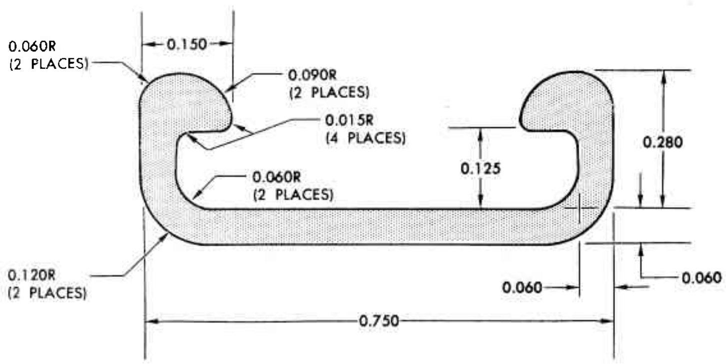


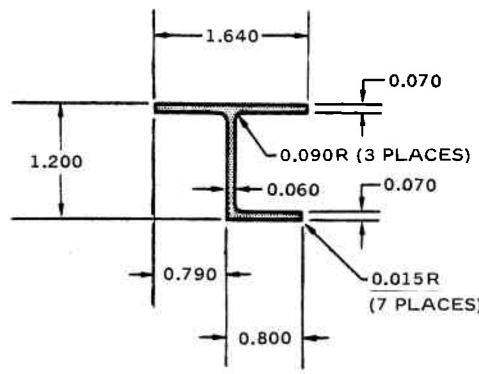
Figure 8-30. Extruded Shapes



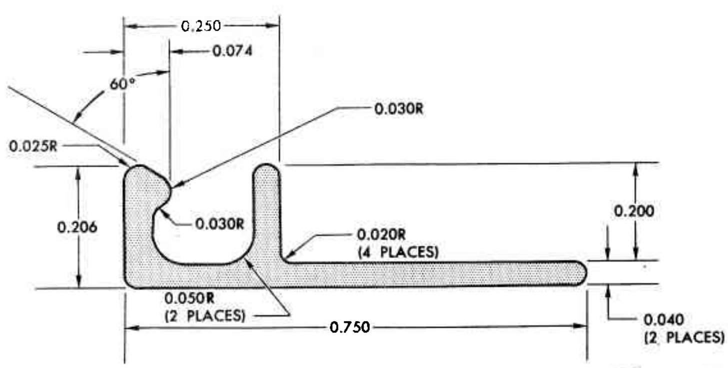
CHANNEL - FLANGER
E571002 7178 AL ALLOY
AREA 0.083



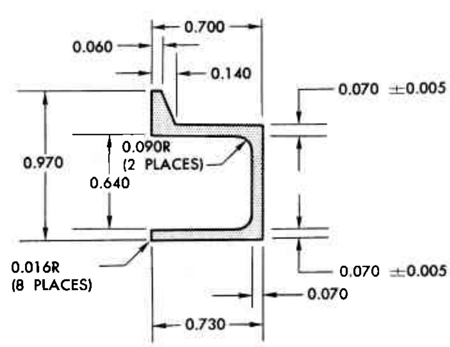
CHANNEL-BULBED
E561002 7075 AL ALLOY
AREA 0.080



CHANNEL - FLANGED
E771210 7178 AL ALLOY
AREA 0.2398



CHANNEL-APPENDAGED
E582101 6061 AL ALLOY
AREA 0.046



CHANNEL-90° APPENDAGED
E581102 2024 AL ALLOY
AREA 0.1671

Figure 8-31. Extruded Shapes

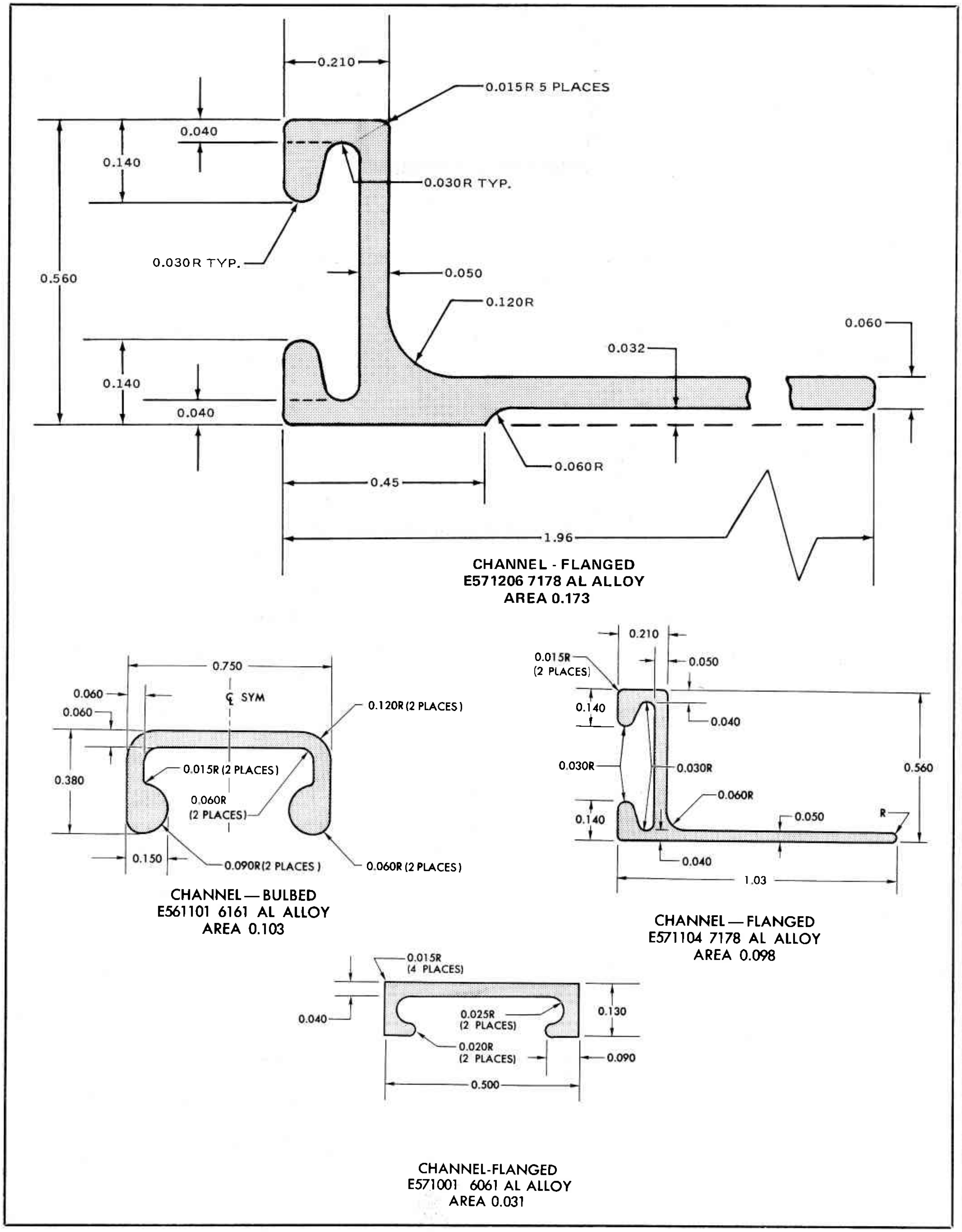
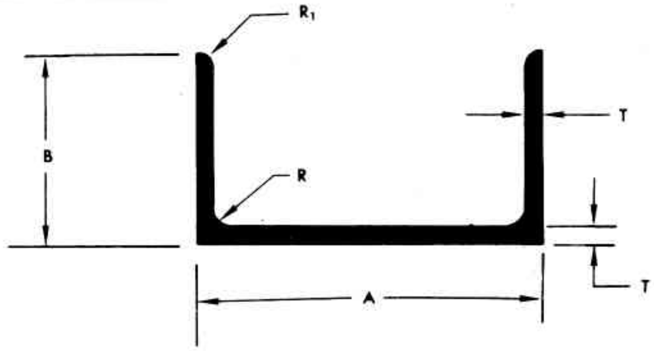
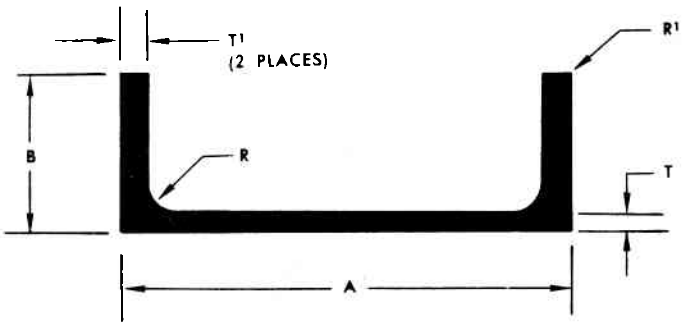


Figure 8-32. Extruded Shapes



CHANNEL—FLANGE WIDTH AND THICKNESS EQUAL

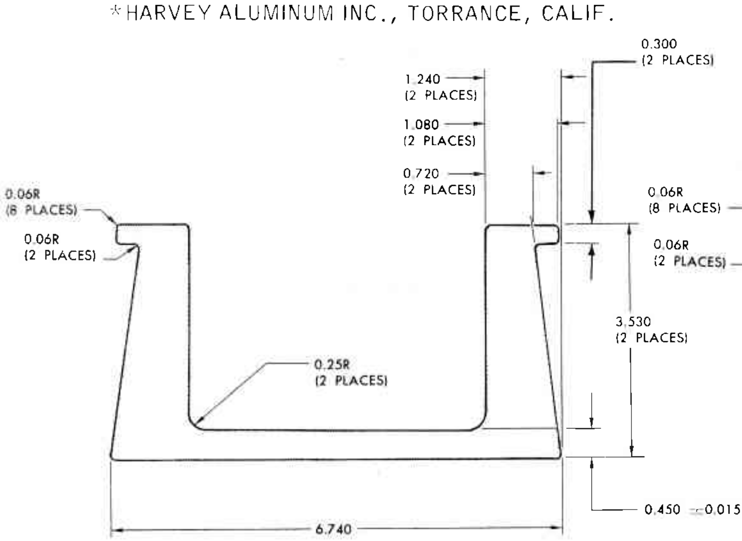
NO.	A	B	T	R	R_1	AREA	2024	7075	7178
AND10137									
-1204	1.250	1.125	0.078	0.125	0.078	0.265		X	
-1604	1.750	1.000	0.094	0.125	0.094	0.338		X	
-2001	2.000	0.750	0.094	0.125	0.094	0.314		X	



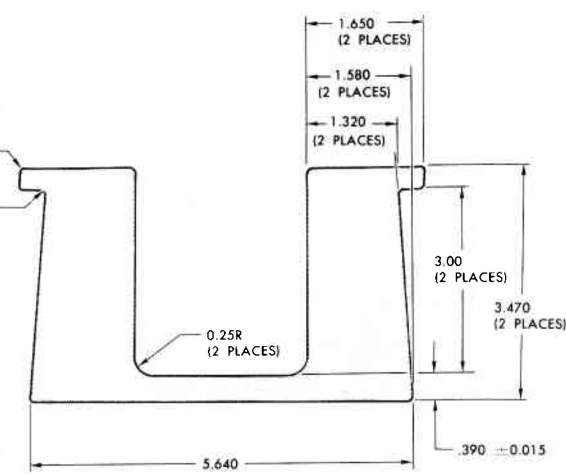
CHANNEL—FLANGE WIDTH EQUAL, THICKNESS UNEQUAL

NO.	A	B	T	T1	R	R1	AREA	2024	7075	7178
E531001	0.875	0.250	0.050	0.120	0.060	0.010	0.096	X		
E531201	0.813	2.750	0.125	0.094	0.062	0.026	0.594			X
E531204	2.300	0.840	0.100	0.140	0.120	0.016	0.443			X
*HM26852	5.000	1.500	0.188	0.250	0.190	0.031	1.610			X

*HARVEY ALUMINUM INC., TORRANCE, CALIF.

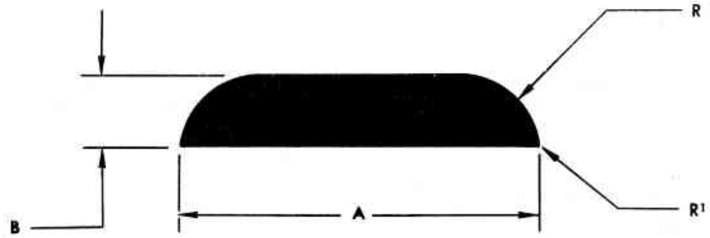


CHANNEL—NON-UNIFORM
E501701 7075 AL ALLOY
AREA 8.8



CHANNEL—NON-UNIFORM
E501601 7075 AL ALLOY
AREA 11.0

Figure 8-33. Extruded Shapes



BAR—FILLER

NO.	A	B	R	R[1]	AREA	2024	7075
E441001	0.938	0.188	0.219	0.026	0.155		X
E441035	0.600	0.125	0.125		0.068	X	
E441036	0.780	0.160	0.160	0.016	0.1138	X	

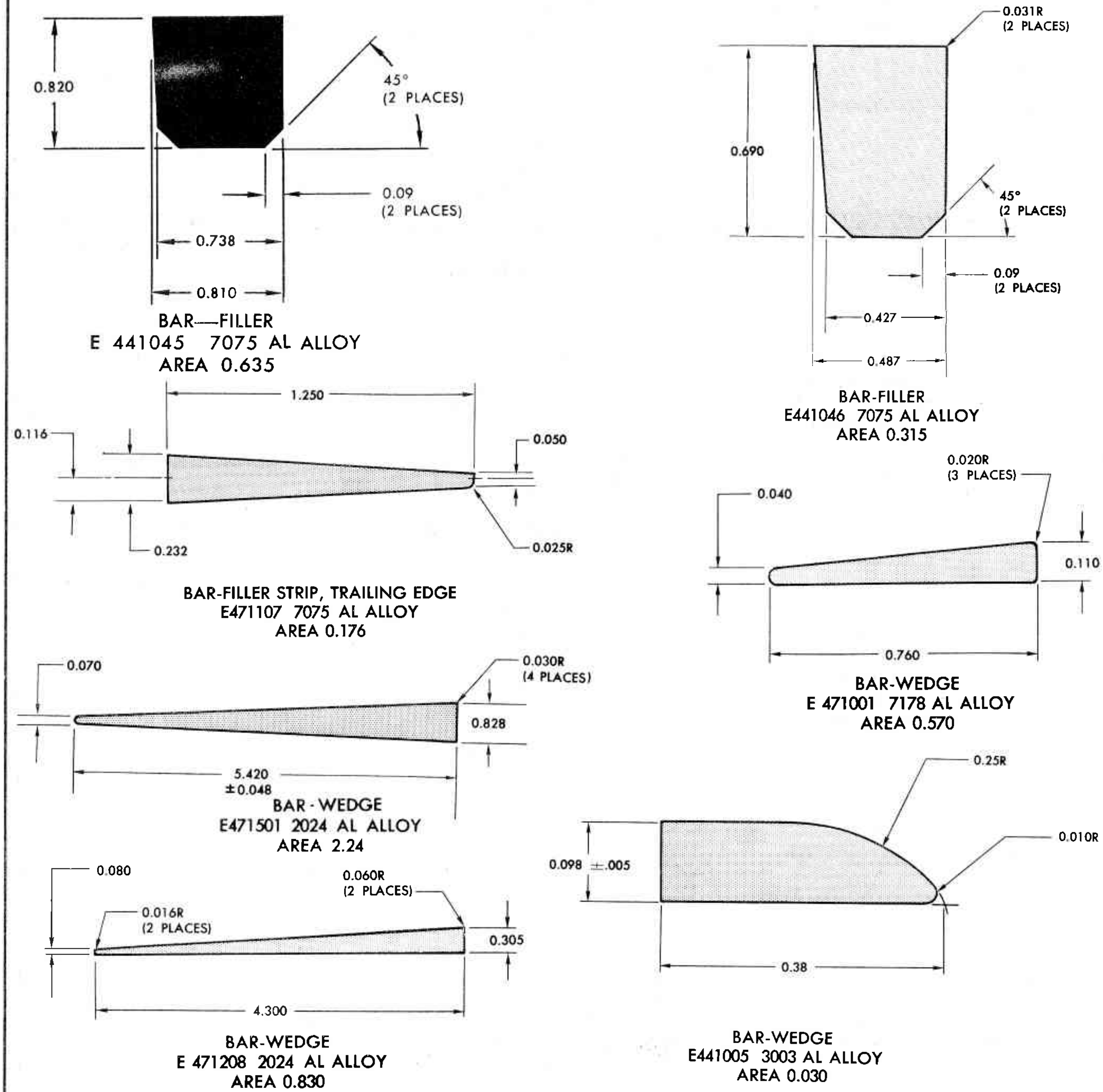


Figure 8-34. Extruded Shapes

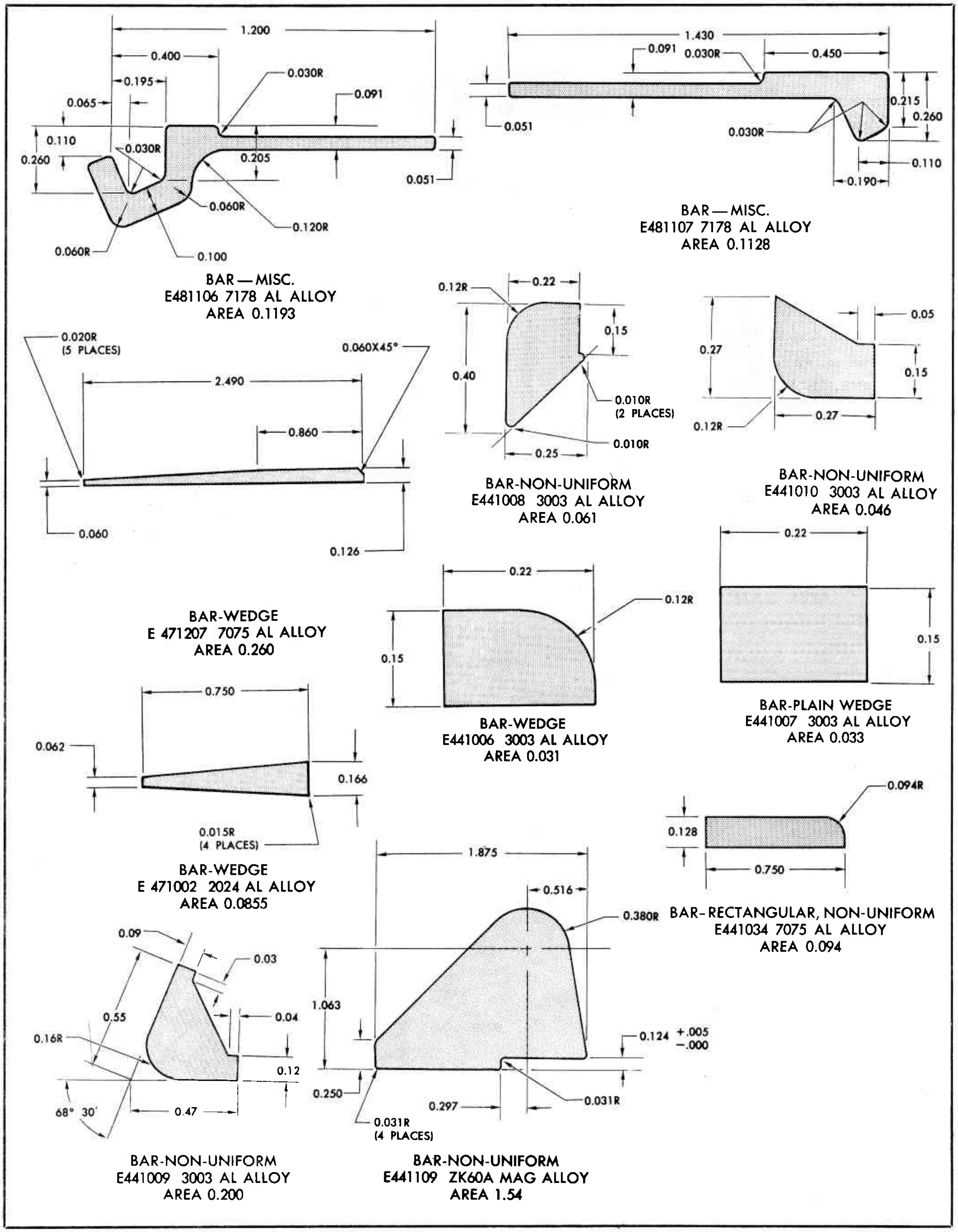
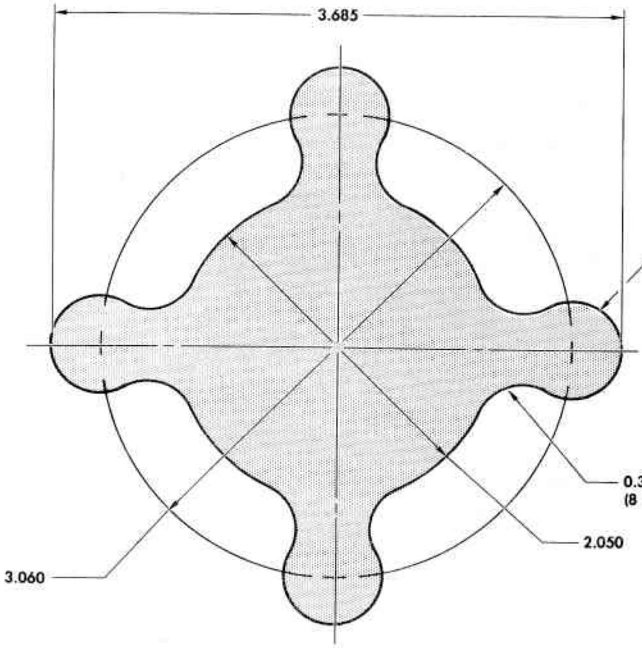
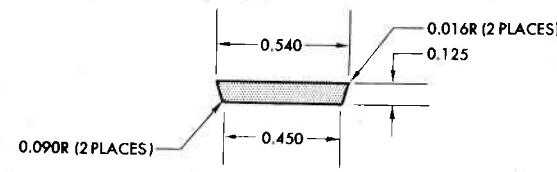


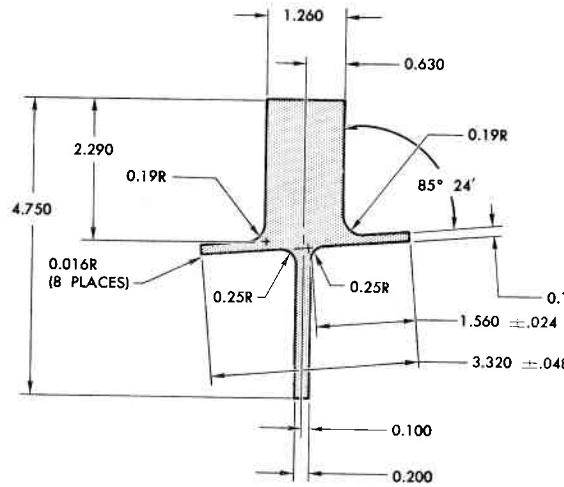
Figure 8-35. Extruded Shapes



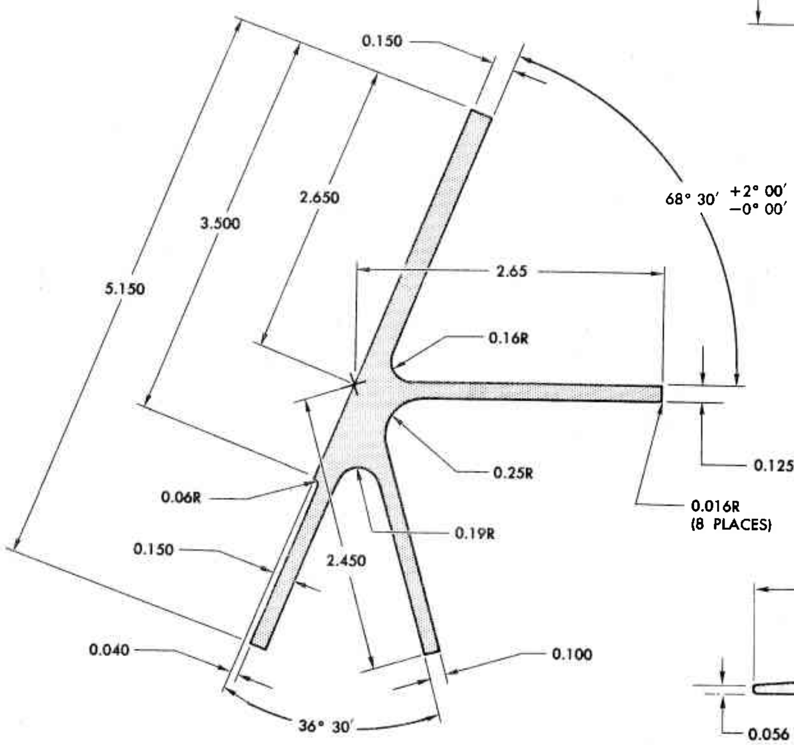
BAR-SPECIAL
E581304 ZK60A MAG ALLOY
AREA 4.250



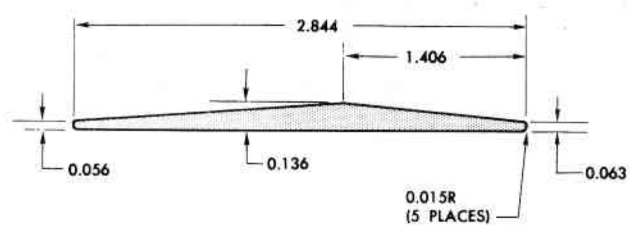
BAR—TRAPEZOID SYMMETRICAL
E441104 2024 AL ALLOY
AREA 0.065



BAR-CROSS SHAPED
E461502 7075 AL ALLOY
AREA 3.80

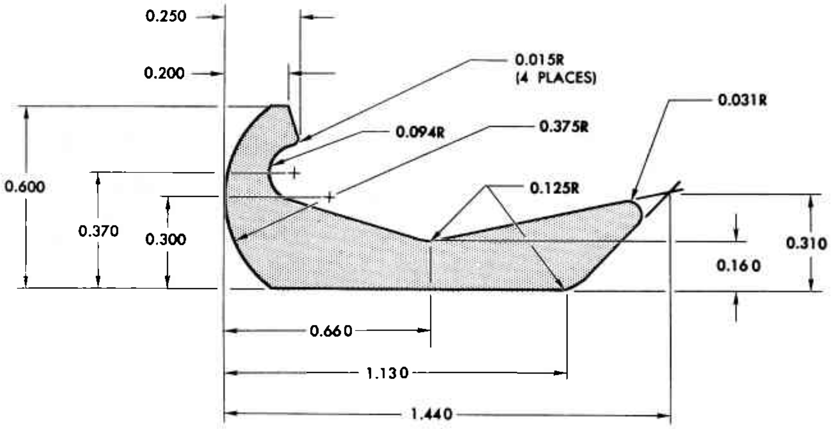


BAR-SPECIAL
E461501 7075 AL ALLOY
AREA 1.31

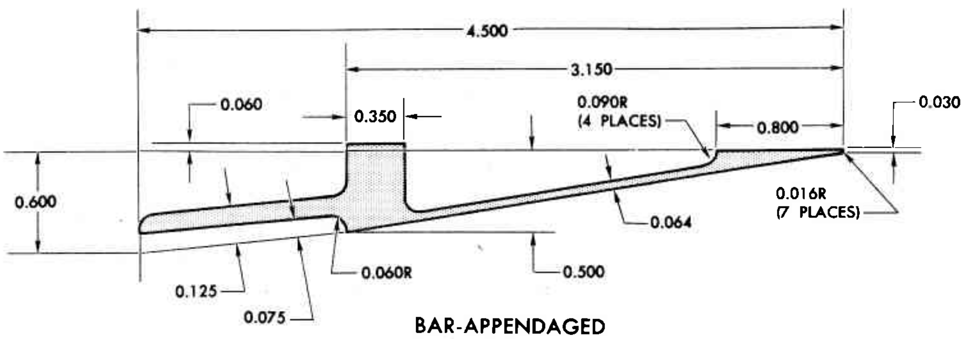


BAR-NON-UNIFORM
E441206 7075 AL ALLOY
AREA 0.284

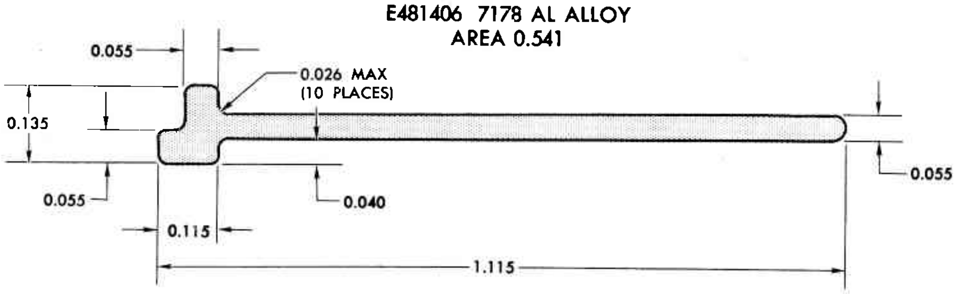
Figure 8-36. Extruded Shapes



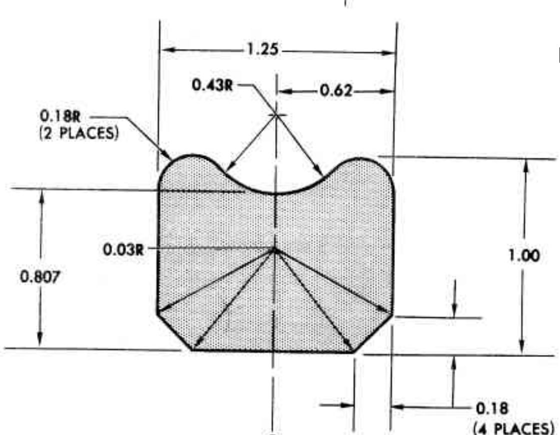
BAR-SPECIAL
E481108 7075 AL ALLOY
AREA 0.330



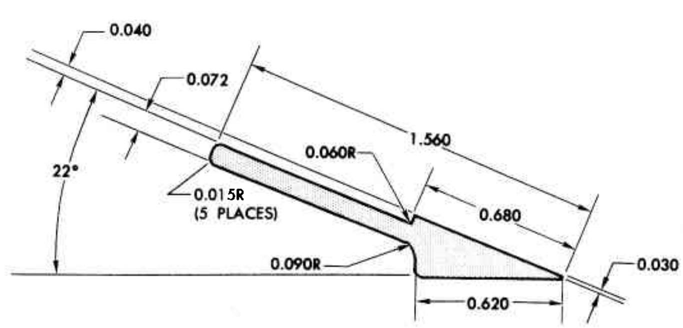
BAR-APPENDAGED
E481406 7178 AL ALLOY
AREA 0.541



BAR-APPENDAGED
E481105 2024 AL ALLOY
AREA 0.066

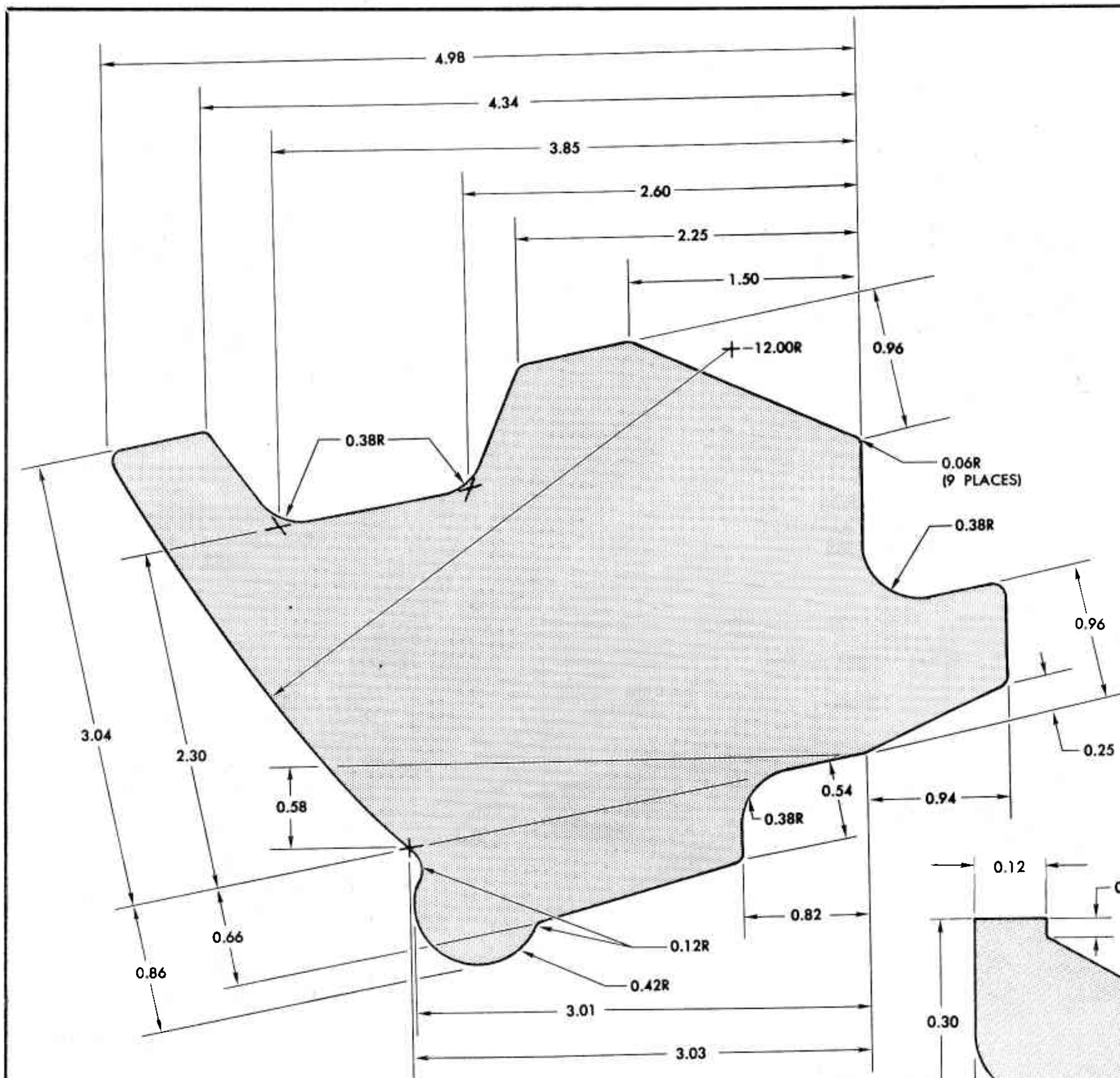


BAR—SPECIAL
E441118 7075 AL ALLOY
AREA 1.10

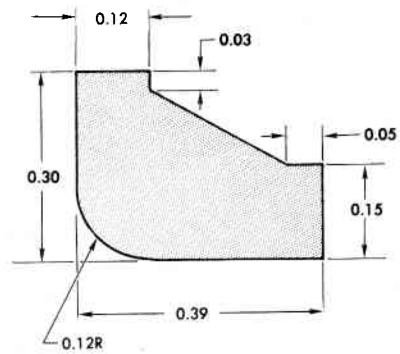


BAR-SPECIAL
E541101 7178 AL ALLOY
AREA 0.164

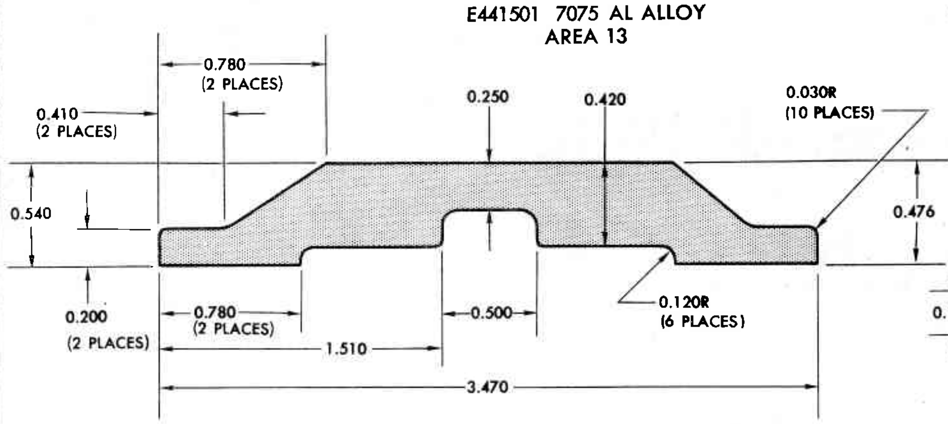
Figure 8-37. Extruded Shapes



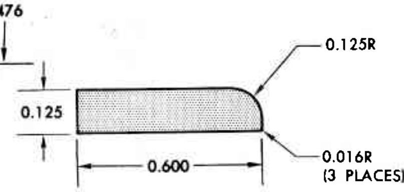
BAR-SPECIAL
E441501 7075 AL ALLOY
AREA 13



BAR-NON-UNIFORM
E441011 3003 AL ALLOY
AREA 0.091



BAR — SPECIAL
E441301 ZK-60A MAG ALLOY
AREA 1.1476



BAR-RECTANGULAR, NON-UNIFORM
E441101 2024 AL ALLOY
AREA 0.072

Figure 8-38. Extruded Shapes

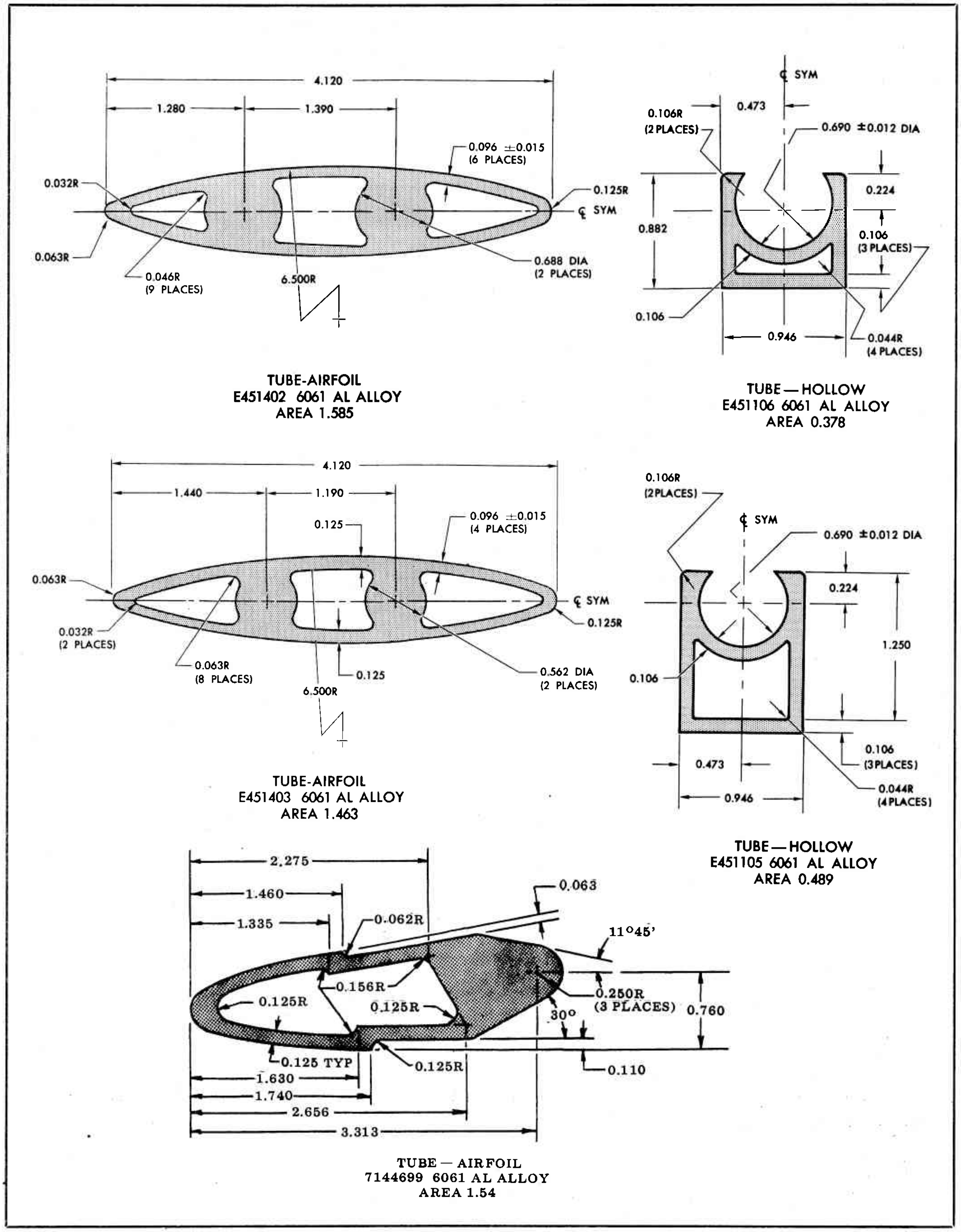


Figure 8-39. Extruded Tubes

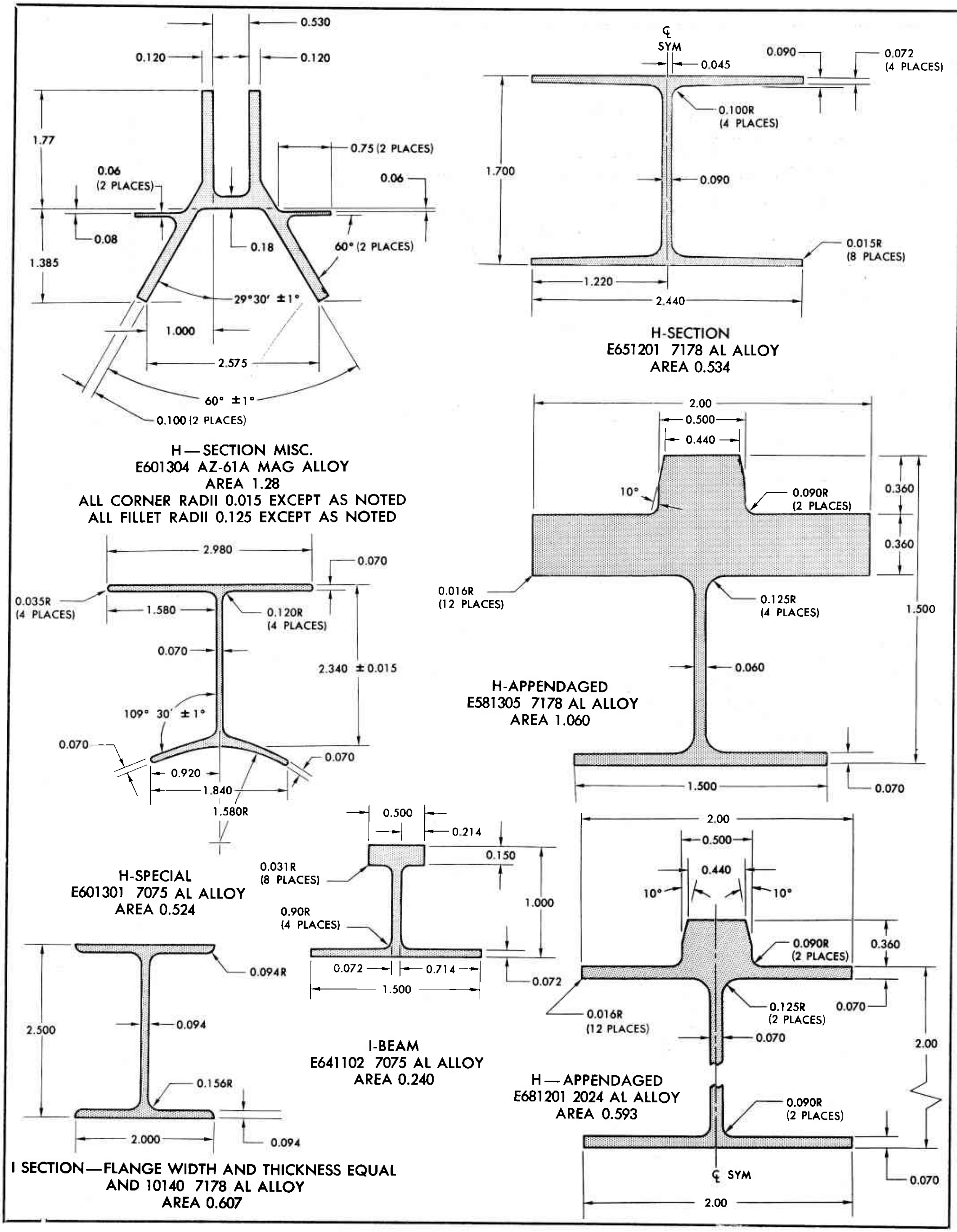


Figure 8-40. Extruded Shapes

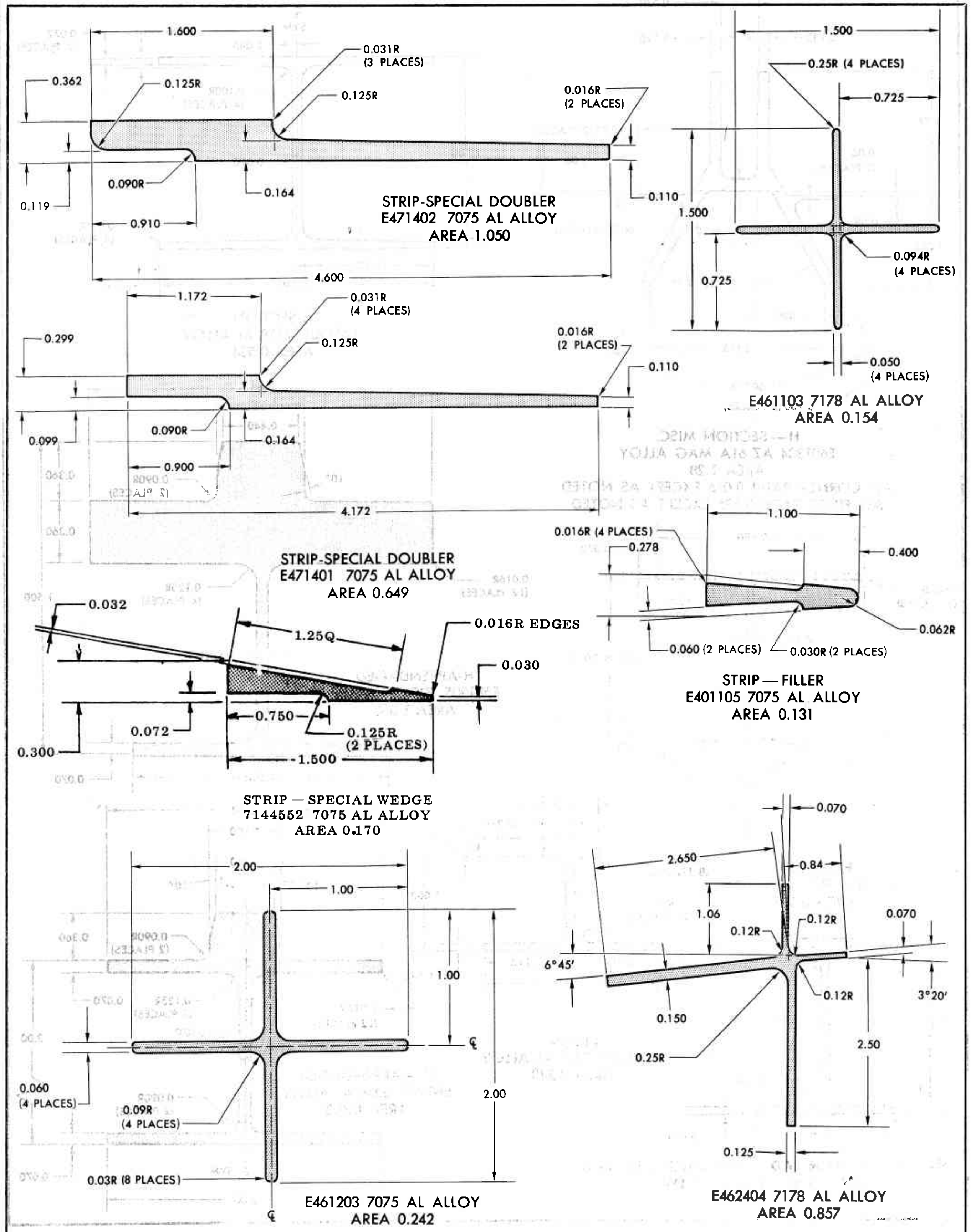
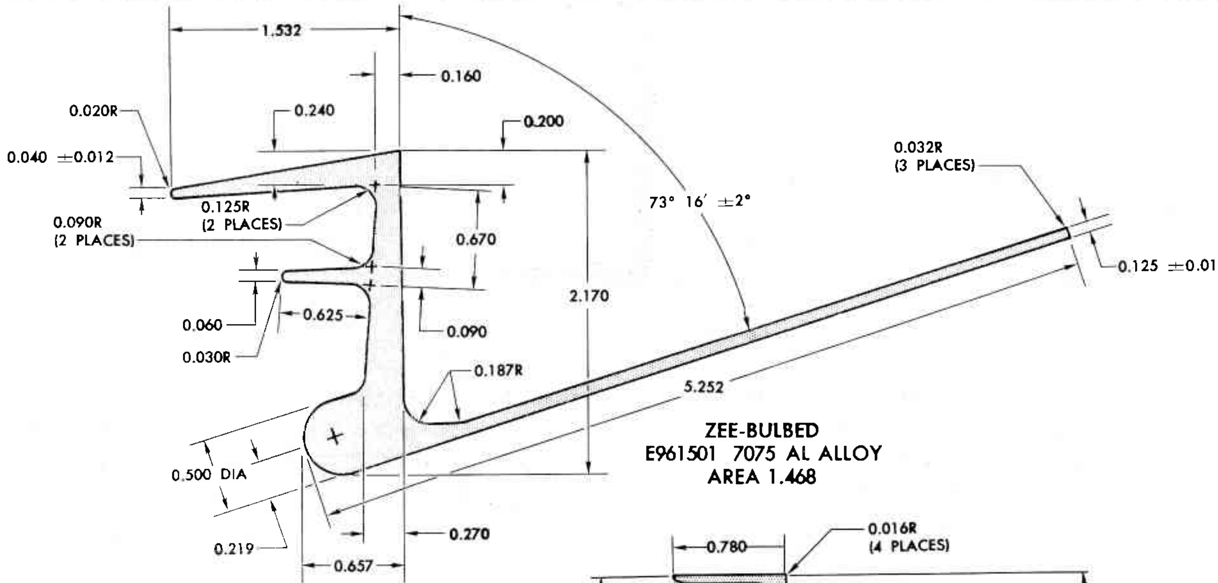
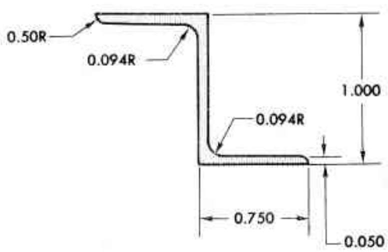


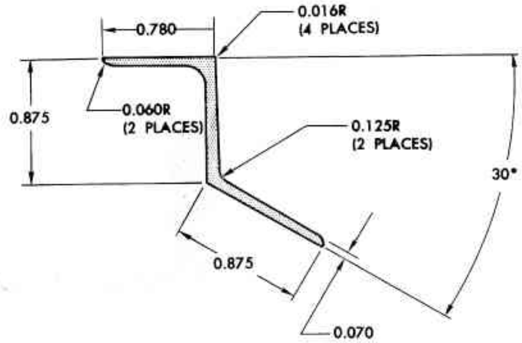
Figure 8-41. Extruded Shapes



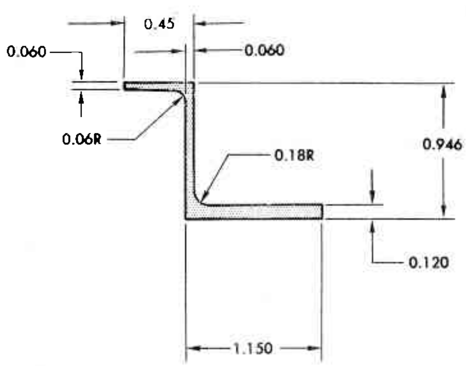
ZEE-BULBED
E961501 7075 AL ALLOY
AREA 1.468



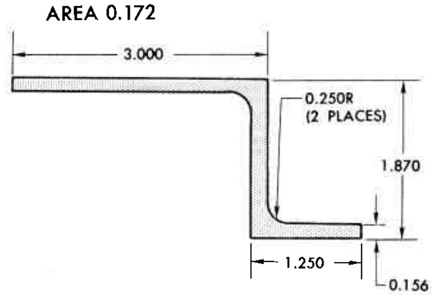
ZEE-FLANGE WIDTH AND THICKNESS EQUAL
AND 10138 7075 AL ALLOY
AREA 0.123



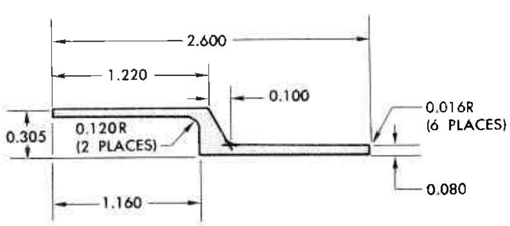
ZEE-ANGLED, FLANGE WIDTH UNEQUAL,
THICKNESS EQUAL
E901101 7075
AL ALLOY
AREA 0.172



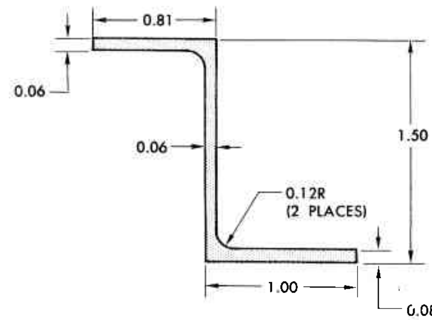
ZEE—FLANGE WIDTH AND THICKNESS UNEQUAL
E942304 7075 AL ALLOY
AREA 0.219



ZEE-FLANGE WIDTH UNEQUAL, THICKNESS EQUAL
E 921401 7178 AL ALLOY
AREA 0.932

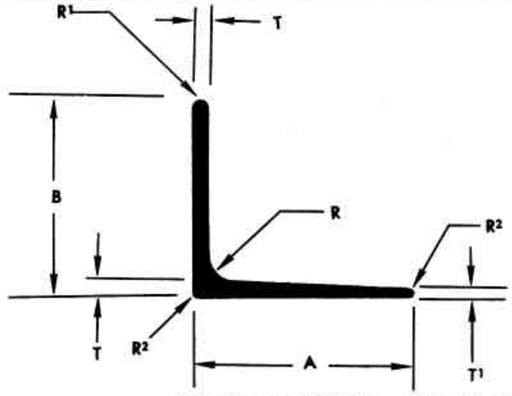


ZEE-FLANGE WIDTH UNEQUAL, THICKNESS EQUAL
E 901202 7075 AL ALLOY
AREA 0.260



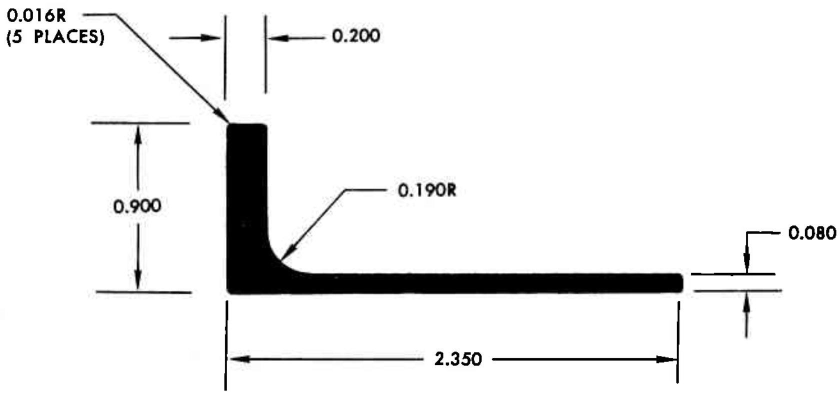
ZEE-FLANGE WIDTH AND THICKNESS UNEQUAL
K 59224 7075 AL ALLOY
AREA 0.216

Figure 8-42. Extruded Shapes

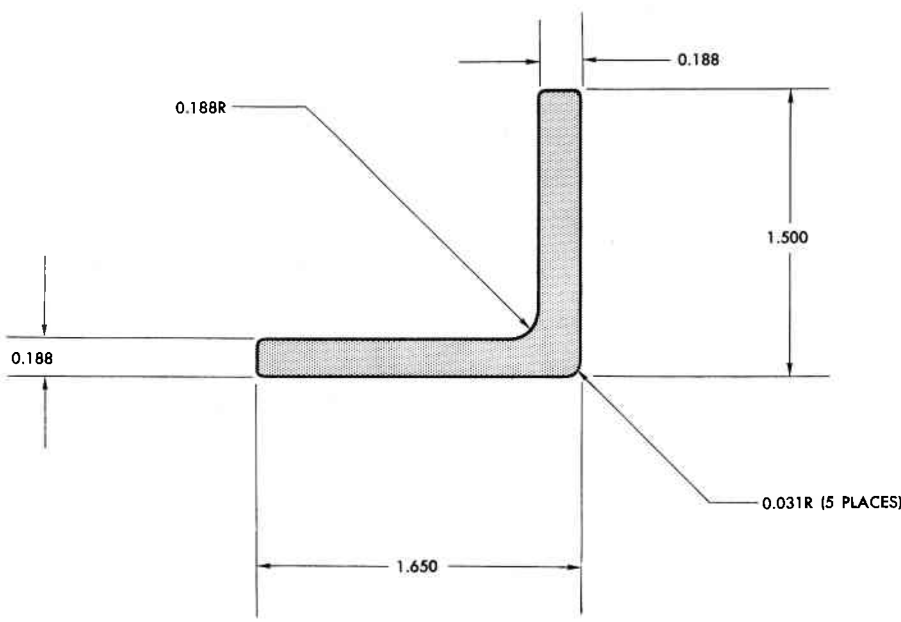


ANGLE—90° FLANGE WIDTH UNEQUAL AND FLANGE TAPERED

NO.	A	B	T	T^1	R	R^1	R^2	AREA	7075	7178
E101104	1.125	1.000	0.060	0.030	0.125	0.030	0.015	0.110	X	
E101105	1.125	0.800	0.060	0.030	0.125	0.030	0.015	0.098	X	X

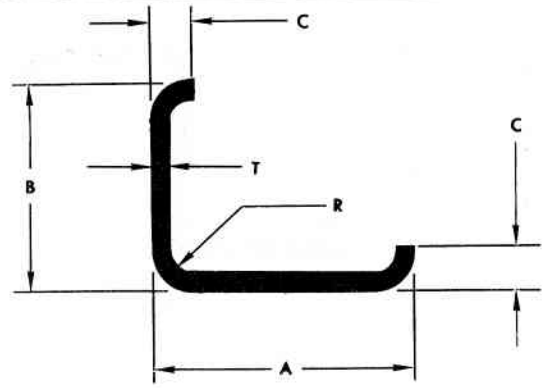


ANGLE—90°, FLANGE WIDTH AND THICKNESS UNEQUAL
E 141207 2024 AL ALLOY
AREA 0.400



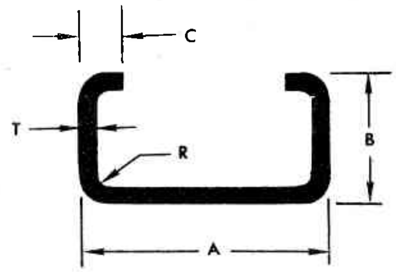
E121202 7075 AL ALLOY
AREA 0.563

Figure 8-43. Extruded Shapes



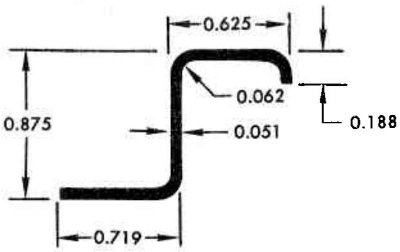
ANGLE—90°, UNEQUAL LEG, FLANGED

DASH NO.	SIZE A	SIZE B	SIZE C	SIZE R	SIZE T	AREA SQ. IN.	BARE 7075	BARE 7178	CLAD 7075
Y7-11	0.750	0.625	0.188	0.125	0.040	0.058	X		
-22	1.000	0.750	0.188	0.125	0.051	0.091	X		
-23	1.000	0.750	0.188	0.125	0.064	0.111	X	X	X
-30	1.000	0.875	0.188	0.125	0.051	0.097	X		
-42	1.250	1.000	0.188	0.125	0.051	0.116			X

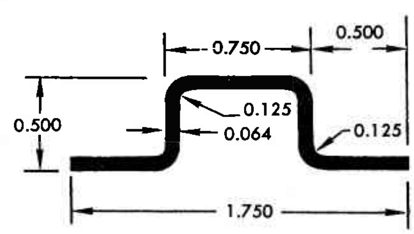


CHANNEL—UNEQUAL LEG, FLANGED

DASH NO.	SIZE A	SIZE B	SIZE C	SIZE R	SIZE T	AREA SQ. IN.	BARE 7075	BARE 7178	CLAD 2024
Y12-4	0.750	0.750	0.175	0.062	0.040	0.092	X		
-16	1.250	0.750	0.188	0.125	0.051	0.136		X	
-26	1.500	0.750	0.188	0.125	0.051	0.149		X	X

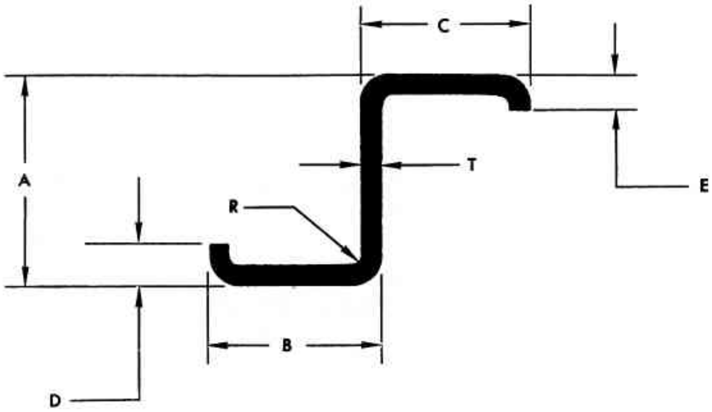


ZEE—VERTICAL, ONE LEG FLANGED
Y34-73 2024 AL ALLOY (BARE)
AREA 0.109



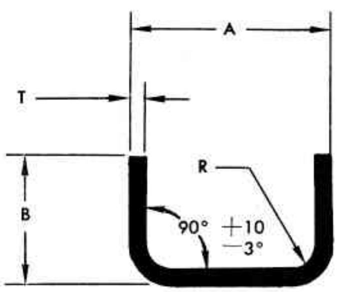
HAT—RECTANGULAR, PLAIN
Y32-10 2024 AL ALLOY (CLAD)
AREA 0.151

Figure 8-44. Standard Roll-Formed Sections



ZEE—VERTICAL, FLANGED

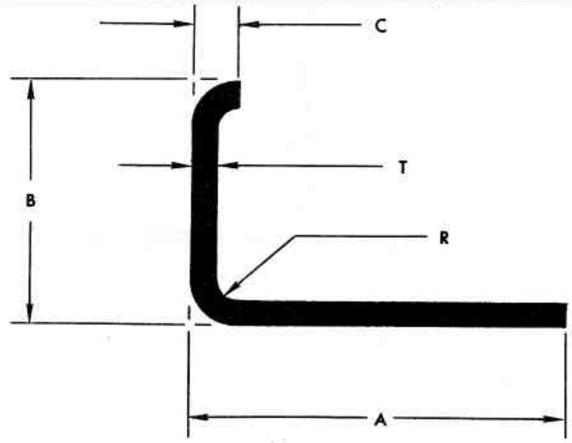
DASH NO.	SIZE A	B	C	D	E	R	T	AREA SQ. IN.	BARE 2024	7075	7178
Y36–5	0.875	0.625	0.625	0.188	0.188	0.062	0.040	0.088		X	X
–17	1.000	0.688	0.688	0.188	0.188	0.062	0.032	0.080		X	
–18	1.000	0.688	0.688	0.188	0.188	0.062	0.040	0.098		X	X
–19	1.000	0.688	0.688	0.188	0.188	0.062	0.051	0.122		X	X
–21	1.000	0.688	0.688	0.188	0.188	0.062	0.064	0.149		X	X
–27	1.000	0.875	0.875	0.188	0.188	0.094	0.072	0.188			X
–29	1.125	0.812	0.812	0.219	0.219	0.094	0.064	0.167		X	
–40	1.250	0.875	0.875	0.188	0.188	0.094	0.081	0.229		X	
–41	1.250	0.875	0.875	0.219	0.219	0.094	0.072	0.211		X	
–48	1.500	0.812	0.812	0.188	0.188	0.062	0.040	0.128			X
–49	1.500	0.812	0.812	0.188	0.188	0.062	0.051	0.160			X
–50	1.500	0.812	0.812	0.219	0.219	0.094	0.064	0.198		X	
–51	1.500	0.812	0.812	0.188	0.188	0.094	0.072	0.215		X	
–52	1.500	0.812	0.812	0.250	0.250	0.125	0.091	0.270	X		



CHANNEL—EQUAL LEG, PLAIN

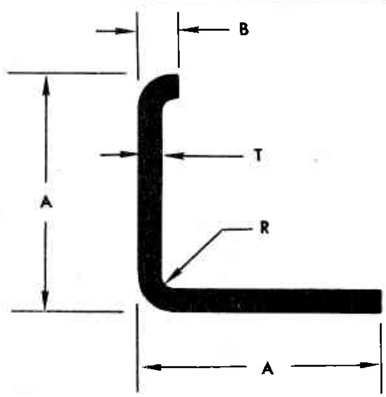
DASH NO.	SIZE A	B	R	T	AREA SQ. IN.	BARE 7075	7178	CLAD 2024	7075
Y10–27	0.875	0.625	0.062	0.064	0.123				X
–38	1.000	0.300	0.062	0.032	0.047			X	
–42	1.000	0.531	0.062	0.032	0.062			X	
–44	1.000	0.625	0.062	0.040	0.084	X			
–54	1.094	0.750	0.062	0.072	0.170		X		
–58	1.250	0.750	0.062	0.051	0.131			X	
–68	1.500	0.344	0.125	0.040	0.079			X	
–69	1.500	0.750	0.062	0.040	0.114		X		
–71	1.500	0.750	0.062	0.064	0.179				X
–80	1.750	0.750	0.062	0.051	0.157		X		
–185	0.750	0.380	0.094	0.051	0.067	X			
–186	0.750	0.380	0.094	0.064	0.082	X			

Figure 8-45. Standard Roll-Formed Sections



ANGLE—UNEQUAL LEG

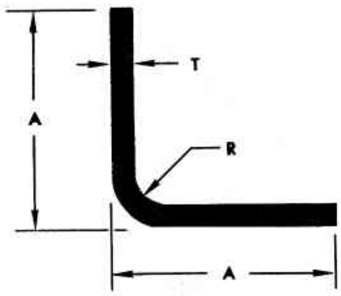
DASH NO.	SIZE					AREA SQ. IN.	BARE		CLAD
	A	B	C	R	T		7075	7178	2024
Y5–9	0.750	0.625	0.188	0.125	0.040	0.054			X
–16	0.875	0.750	0.188	0.125	0.051	0.081	X	X	
–17	0.875	0.750	0.188	0.125	0.064	0.099		X	
–21	1.000	0.750	0.188	0.125	0.040	0.069	X	X	
–22	1.000	0.750	0.188	0.125	0.051	0.087	X	X	
–24	1.000	0.750	0.219	0.125	0.072	0.121		X	
–28	1.000	0.875	0.188	0.125	0.064	0.123			X
–32	1.250	0.750	0.188	0.125	0.051	0.100			X
–33	1.250	0.750	0.188	0.125	0.064	0.123	X		
–43	1.250	1.000	0.188	0.125	0.064	0.139		X	
–45	1.250	1.000	0.219	0.125	0.081	0.175		X	
–68	1.500	0.875	0.188	0.125	0.064	0.147		X	



ANGLE—EQUAL LEG

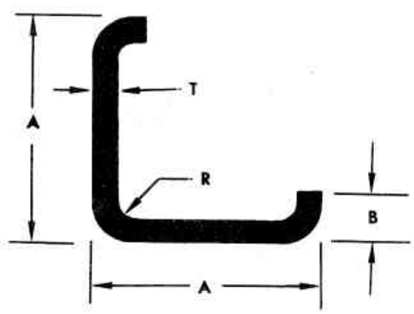
DASH NO.	SIZE				AREA SQ. IN.	BARE		CLAD
	A	B	R	T		7075	7178	7075
Y4-23	0.750	0.156	0.125	0.032	0.047		X	
-24	0.750	0.188	0.125	0.040	0.059	X	X	X
-25	0.750	0.188	0.125	0.051	0.074	X	X	X
-26	0.750	0.188	0.125	0.064	0.091		X	
-33	0.875	0.188	0.125	0.040	0.069		X	
-35	0.875	0.188	0.125	0.064	0.107		X	
-43	1.000	0.188	0.125	0.040	0.079		X	
-45	1.000	0.188	0.125	0.064	0.123	X	X	
-46	1.000	0.219	0.125	0.072	0.140		X	
-47	1.000	0.219	0.125	0.081	0.155	X		
-57	1.250	0.219	0.125	0.081	0.196		X	
-67	1.500	0.219	0.125	0.081	0.236	X		

Figure 8-46. Standard Roll-Formed Sections



ANGLE—90°, EQUAL LEG, PLAIN

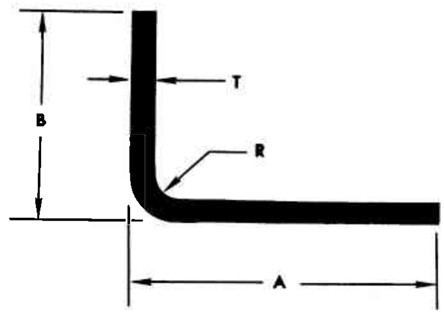
DASH NO.	SIZE			AREA SQ. IN.	BARE			CLAD	
	A	R	T		2024	7075	7178	2024	7075
Y2-2	0.625	0.094	0.051	0.059			X		X
-3	0.812	0.094	0.064	0.096			X		
-4	0.500	0.062	0.032	0.030				X	
-5	0.500	0.062	0.040	0.037			X		
-11	0.562	0.062	0.040	0.042			X		
-12	0.562	0.062	0.051	0.053			X		
-15	0.875	0.094	0.040	0.066		X			
-16	0.625	0.062	0.064	0.067		X			
-17	0.625	0.188	0.040	0.045				X	
-24	0.750	0.188	0.040	0.055				X	X
-25	0.750	0.188	0.051	0.069			X		
-26	0.750	0.188	0.064	0.086			X		
-27	0.750	0.094	0.064	0.089	X		X	X	
-28	0.750	0.094	0.072	0.099			X		
-34	0.875	0.188	0.051	0.082		X			
-35	0.875	0.188	0.064	0.102			X		
-43	1.000	0.188	0.040	0.075				X	
-46	1.000	0.094	0.072	0.135			X		
-49	1.000	0.125	0.102	0.186			X		
-69	1.500	0.094	0.102	0.289				X	
-70	1.125	0.156	0.156	0.311			X		
-90	1.000	0.094	0.064	0.120		X		X	
Y2-91	0.750	0.062	0.051	0.086		X	X		



ANGLE—90°, EQUAL LEG, FLANGED

DASH NO.	SIZE				AREA SQ. IN.	BARE		
	A	B	R	T		2024	7075	7178
Y3-16	0.625	0.156	0.125	0.032	0.041		X	X
-22	0.750	0.188	0.125	0.040	0.063		X	X
-23	0.750	0.156	0.125	0.032	0.049			X
-25	0.750	0.188	0.125	0.051	0.078		X	X
-26	0.750	0.188	0.125	0.064	0.095	X	X	X
-34	0.875	0.188	0.125	0.051	0.091		X	
-44	1.000	0.188	0.125	0.051	0.104			X

Figure 8-47. Standard Roll-Formed Sections



ANGLE—90°, UNEQUAL LEG, PLAIN

DASH NO.	SIZE				AREA SQ. IN.	BARE			CLAD	
	A	B	R	T		2024	7075	7178	2024	7075
Y6-7	0.750	0.500	0.125	0.040	0.046				X	
-8	0.750	0.625	0.094	0.051	0.065		X	X	X	X
-9	0.750	0.625	0.094	0.064	0.080			X	X	
-11	0.750	0.625	0.188	0.040	0.050				X	
-15	0.781	0.250	0.125	0.040	0.037		X			
-21	1.000	0.750	0.188	0.040	0.065		X	X		
-22	1.000	0.750	0.188	0.051	0.082			X	X	
-23	1.000	0.750	0.188	0.064	0.102				X	
-28	1.250	0.750	0.125	0.051	0.096	X				
-31	1.250	0.750	0.188	0.040	0.075		X		X	
-38	1.250	0.750	0.062	0.064	0.121				X	
-43	1.250	1.000	0.188	0.064	0.134				X	
-45	1.250	1.000	0.188	0.081	0.168				X	
-52	0.875	0.625	0.094	0.072	0.099	X				
-53	0.875	0.625	0.125	0.051	0.071		X		X	
-61	1.500	0.750	0.188	0.051	0.108				X	
-82	1.500	1.250	0.188	0.064	0.166		X			
-122	1.750	1.500	0.188	0.064	0.198				X	

.06.03.054 A .58.02.00

Figure 8-48. Standard Roll-Formed Sections

Section IX

DAMAGE DUE TO LANDING GEAR FAILURE

9-1. REPAIR OF DAMAGE DUE TO LANDING GEAR FAILURE.

9-2. Extensive damage to the airplane structure will result from landing gear failure. Repair of the airplane will require the replacing of structural components such as skin panels, longerons, bulkheads, beltframes, landing gears and their supporting structure. Care must be exercised when making repairs or replacements to maintain proper alignment of components and restore the airplane to as near its original configuration as possible.

NOTE

Investigation and repair of damage shall be supervised by an aeronautical structures engineer.

This section covers the following conditions of landing gear failure: nose gear up and main gears down; main gears up and nose gear down; and all gears up. Along with each condition of landing gear failure is information and illustrations pertaining to areas of probable damage, inspection and repair procedures. Refer to paragraph 1-221 for preliminary and general inspection information on the airplane exterior and substructure.

9-3. NOSE GEAR UP.

9-4. Extent of Damage.

9-5. A landing made with the nose landing gear retracted or collapsed will incur extensive damage to the following structural components: pitot-static boom; radome; forward lower longerons; lower portion of bulkheads, beltframes, skin panels and doors from the radome to approximately fuselage station 216.50; and the nose landing gear and attaching structure. The airplane wings and engine may incur damage from flying debris. The main landing gear may also incur damage due to abnormal stresses caused by the failure of the nose landing gear. See figures 9-1 through 9-3 for a survey of the damage applicable to nose landing gear failure.

9-6. Inspection of Airplane.

9-7. A thorough detailed examination of the damaged area and a visual inspection of the entire airplane is required. Remove the engine for an inspection of the compressor for damage due to flying debris. Inspect all primary structure such as longerons, bulkheads, beltframes, and nose landing gear attach point supporting structure for hidden damage or distortion. Check airframe alignment to insure against any distortion or deformation that may not be apparent in visual inspection. Check landing gear for alignment and operation. Refer to paragraph 1-268 for alignment procedures. Refer to T.O. 1F-106A-2-8-2-1 for landing gear operation procedure. Any parts suspected of being damaged shall be removed and inspected by the magnetic particle inspection or dye penetrant inspection method, whichever is applicable.

9-8. Removal of Damaged Area.

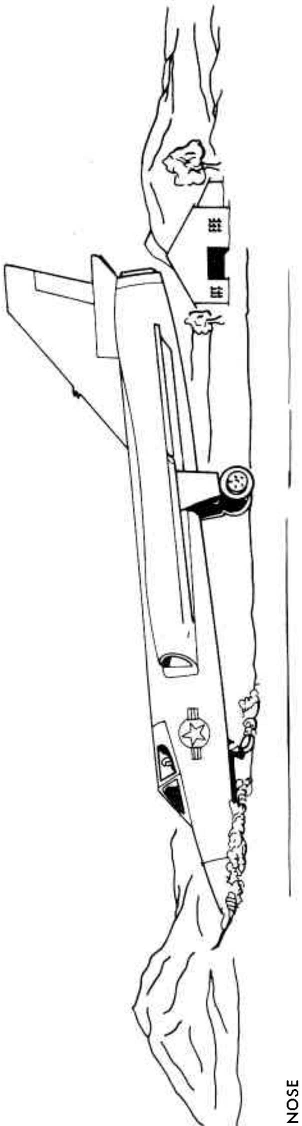
9-9. Support the airplane in such a manner as not to interfere with the repair or replacement of damaged structure; refer to paragraph 1-333 for instructions. Take care not to damage sound structure adjacent to damaged areas when making replacements. In reworking damaged structure, trim out the damaged or distorted areas and leave ample fastener edge margins in the original structure to facilitate repairs. Trimmed edges should be smooth and should allow generous radii in the corners. Large structural components to be replaced should be disassembled at some convenient point, such as manufacturing splice at fuselage station 102 or 216.50 for example, to facilitate the installation of new structure. Bolt-on components, such as the nose landing gear or the radome, should be removed to facilitate repairs.

9-10. Procurement of Parts.

9-11. Replacement parts or assemblies required for structural repair may be obtained from salvage, fabricated, or procured from spares sources. For identification of components and engineering drawing numbers see figures 1-92 and 1-93, and refer to the Illustrated Parts Breakdown, T.O. 1F-106A-4 or T.O. 1F-106B-4.

9-12. Installation of Parts.

9-13. Repairs and replacement of large structural components should be accomplished first. Installation of the



NOTES:
1. FIGURE ILLUSTRATES DAMAGE ZONE TYPICAL OF NOSE LANDING GEAR FAILURE. SEE FIGURE 9-2 AND 9-3 FOR COMPLETION OF NOSE LANDING GEAR FAILURE SURVEY.
2. REFER TO THIS SECTION FOR INSPECTION OF AIRFRAME, PROCUREMENT OF PARTS, AND REPAIR INFORMATION.
3. A DETAILED EXAMINATION OF STRUCTURE IN AREA OF LANDING GEAR ATTACH FITTINGS IS REQUIRED.
4. REPLACE COMPONENTS THAT ARE DAMAGED EXTENSIVELY.
5. EXAMINE NOSE SECTION STRUCTURE IN DETAIL FOR DAMAGE AND DISTORTION.
6. FOR SURVEY OF DAMAGE RESULTING FROM BOTH NOSE AND MAIN LANDING GEAR FAILURE, SEE FIGURES 9-1 THROUGH 9-8.

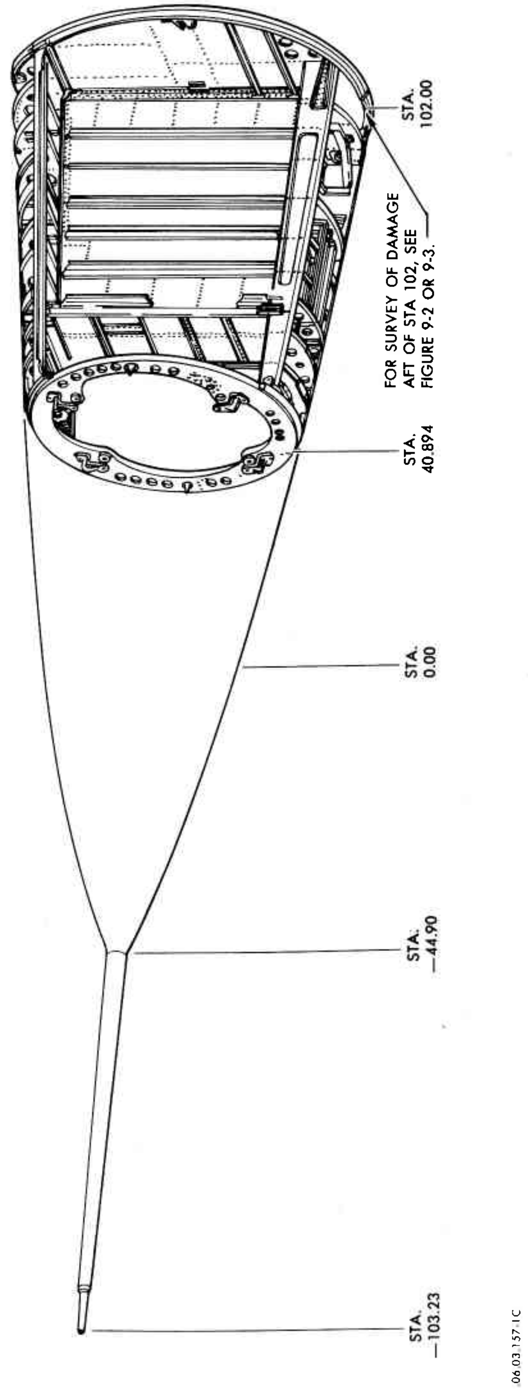


*Figure 9-1. Damage Due to Landing Gear **Failure — Nose Gear Up***

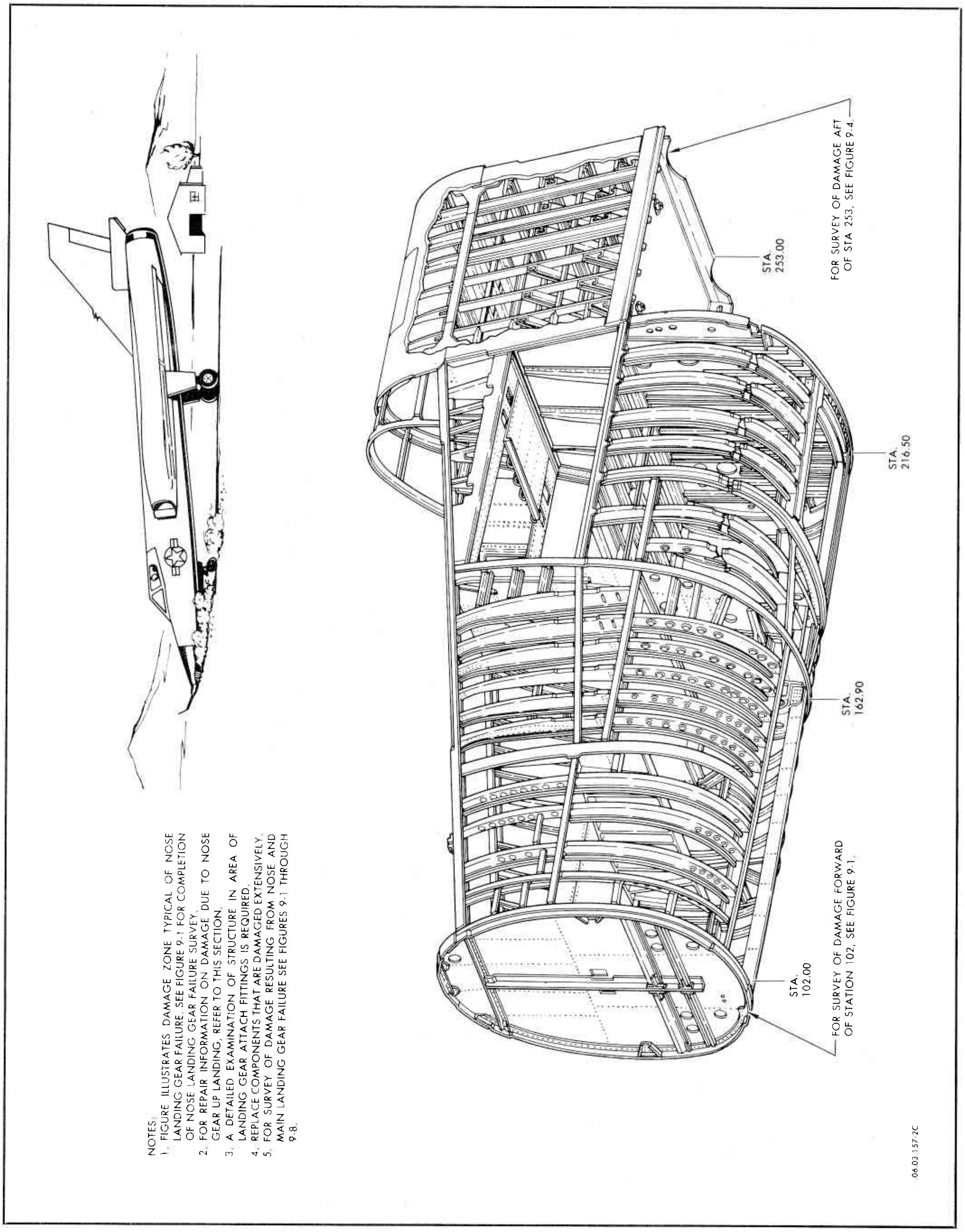


Figure 9-2. Damage Due to Landing Gear Failure — Nose Gear Up, F-106A

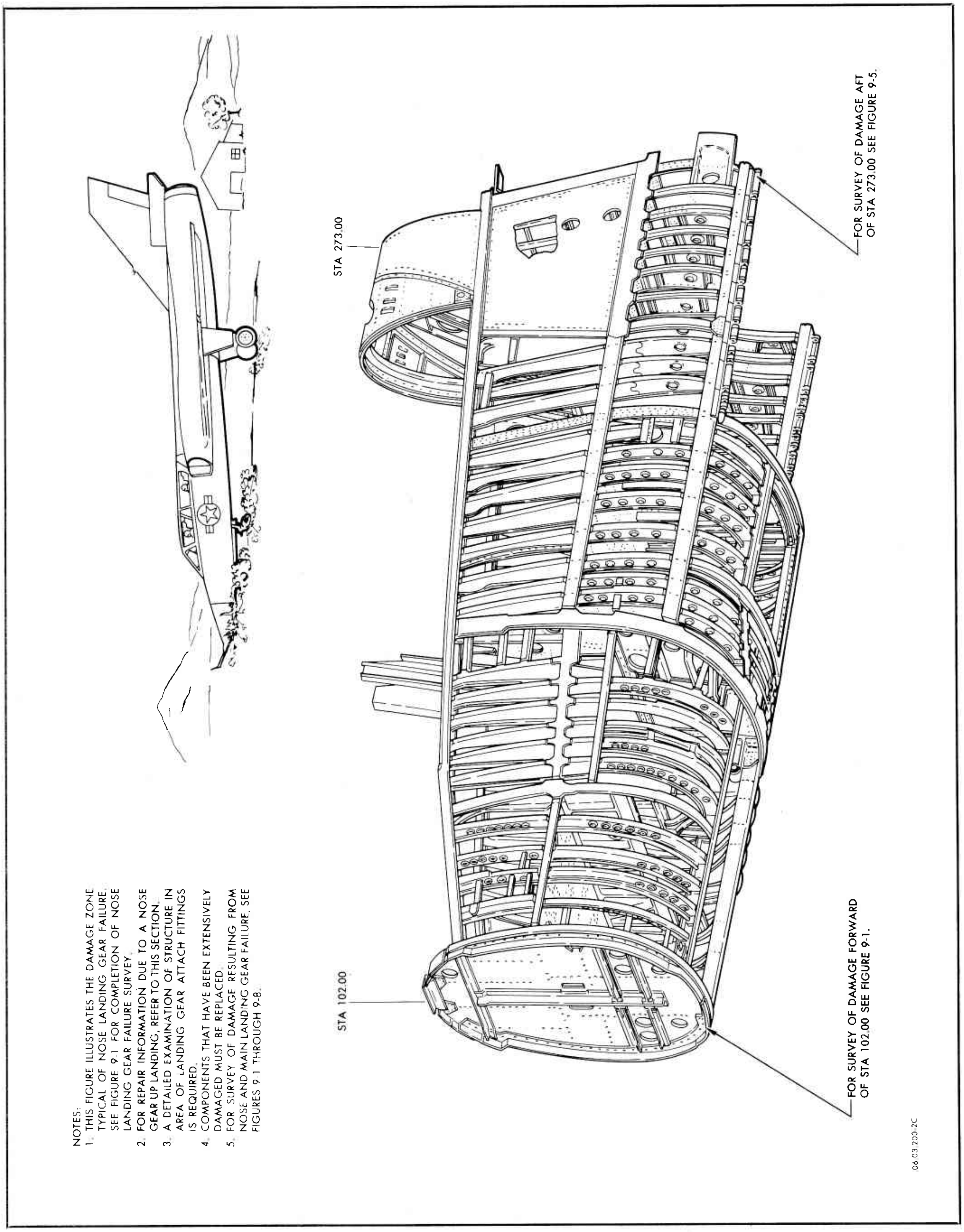


Figure 9-3. Damage Due to Landing Gear Failure — Nose Gear Up, F-106B

large components should be done with the aid of jigs and fixtures to maintain proper alignment and to facilitate repairs. Replacement parts should be checked for any mismatch with the original structure before installation. Some replacement parts may require rework before installation can be accomplished. Refer to Sections II and IV for a complete breakdown of the wing and fuselage structure and the types of material used in their construction.

9-14. MAIN GEAR UP.

9-15. Extent of Damage.

9-16. A landing made with the main landing gear retracted or collapsed will incur extensive damage to the following airplane structural components: main landing gear and doors; aft lower longerons; lower portion of bulkheads, beltframes, skin panels and doors from approximately fuselage station 472 to station 711.40 including the lower portion of the afterburner shroud; wings and elevons; and the main landing gear attach point supporting structure. The airplane engine may incur damage from flying debris. The nose landing gear may also incur damage due to the abnormal stresses caused by the failure of the main landing gear. See figures 9-4 through 9-8 for a survey of the damage applicable to main landing gear failure.

9-17. Inspection of Airplane.

9-18. A thorough detailed examination of the damaged area and a visual inspection of the entire airplane is required. Remove the engine for an inspection of the compressor and afterburner shroud for damage. Inspect all primary structure such as longerons, bulkheads, beltframes, and main landing gear attach point supporting structure for hidden damage or distortion. Check airframe alignment to insure against any distortion or deformation that may not be apparent in visual inspection. Check landing gear for alignment and operation. Refer to paragraph 1-268 for alignment procedures. Refer to T.O. 1F-106A-2-8-2-1 for landing gear operation procedure. Any parts that are suspected of being damaged shall be removed and inspected by the magnetic particle inspection or fluorescent penetrant inspection method, whichever is applicable.

9-19. Removal of Damaged Area.

9-20. Support the airplane in such a manner as not to interfere with the repair or replacement of damaged structure; refer to paragraph 1-333 for instructions. Exercise care not to damage sound structure adjacent to damaged areas when making replacements. In reworking damaged structure, trim out the damaged or distorted areas leaving ample fastener edge margins in the original structure to facilitate repairs. Trimmed edges should be smooth and should allow generous radii in the corners. Large structural components to be replaced may be disassembled at some convenient point, such as manufacturing splice at fuselage station 472 for example, to facilitate the installation of new structure. Bolt-on components, such as the landing gear, wings, or elevons, should be removed to facilitate repairs.

9-21. Procurement of Parts.

9-22. Replacement parts or assemblies required for structural repair may be obtained from salvage, fabricated, or procured from spares sources. For identification of components and engineering drawing numbers, see figures 1-92 and 1-93 and refer to the Illustrated Parts Breakdown, T. O. 1F-106A-4 or T. O. 1F-106B-4.

9-23. Installation of Parts.

9-24. Repairs and replacement of large structural components should be accomplished first. Installation of the large components should be done with the aid of jigs and fixtures to maintain proper alignment and facilitate repairs. Replacement parts should be checked for any mismatch with the original structure before installation. Some replacement parts may require rework before installation can be accomplished. Refer to Sections II and IV for a complete breakdown of the wing and fuselage structure and the types of material used in their construction.

9-25. ALL GEAR UP.

9-26. Extent of Damage.

9-27. A landing made with all landing gear retracted or collapsed will incur extensive damage to the following airplane structural components: nose and main landing gears and their attach point supporting structure; pitot-static boom and radome; lower longerons; lower portion of bulkheads, beltframes and skin panels; doors; and the wings and elevons. Should the landing occur at a location not equipped to make major repairs, inspect critical structural components such as the fuselage lower longeron and the machined bulkheads at stations 431.00 and 472.00 for damage (see figures 9-7 and 9-8) to determine the feasibility of making temporary repairs for a "one-time" flight to a repair depot. This procedure would preclude disassembling the airplane and consequent costly shipment to repair facilities. Refer to paragraph 1-81 for aerodynamic limitations on "one-time" flights. The airplane's engine may incur damage from flying debris. See figures 9-1 through 9-8 for a complete fuselage damage survey applicable to nose and main landing gear failure.

9-28. Inspection of Airplane.

9-29. A thorough detailed examination of the damaged area and a visual inspection of the entire airplane is required. The engine will have to be removed from the airframe for its inspection and to facilitate the inspection and repair of the structure. Inspect all primary structure such as longerons, bulkheads, beltframes, and

NOTES:
1. FIGURE ILLUSTRATES INTERMEDIATE DAMAGE ZONE TYPICAL OF BOTH NOSE AND MAIN LANDING GEAR FAILURE. SEE FIGURE 9-1 THROUGH 9-8 FOR COMPLETE DAMAGE SURVEY OF AIRFRAME.
2. FOR REPAIR INFORMATION ON DAMAGE DUE TO AN ALL GEAR UP LANDING, REFER TO THIS SECTION.
3. DETAILED EXAMINATION OF STRUCTURE IN AREA OF LANDING GEAR ATTACH FITTINGS IS REQUIRED.
4. REPLACE COMPONENTS THAT ARE DAMAGED EXTENSIVELY.

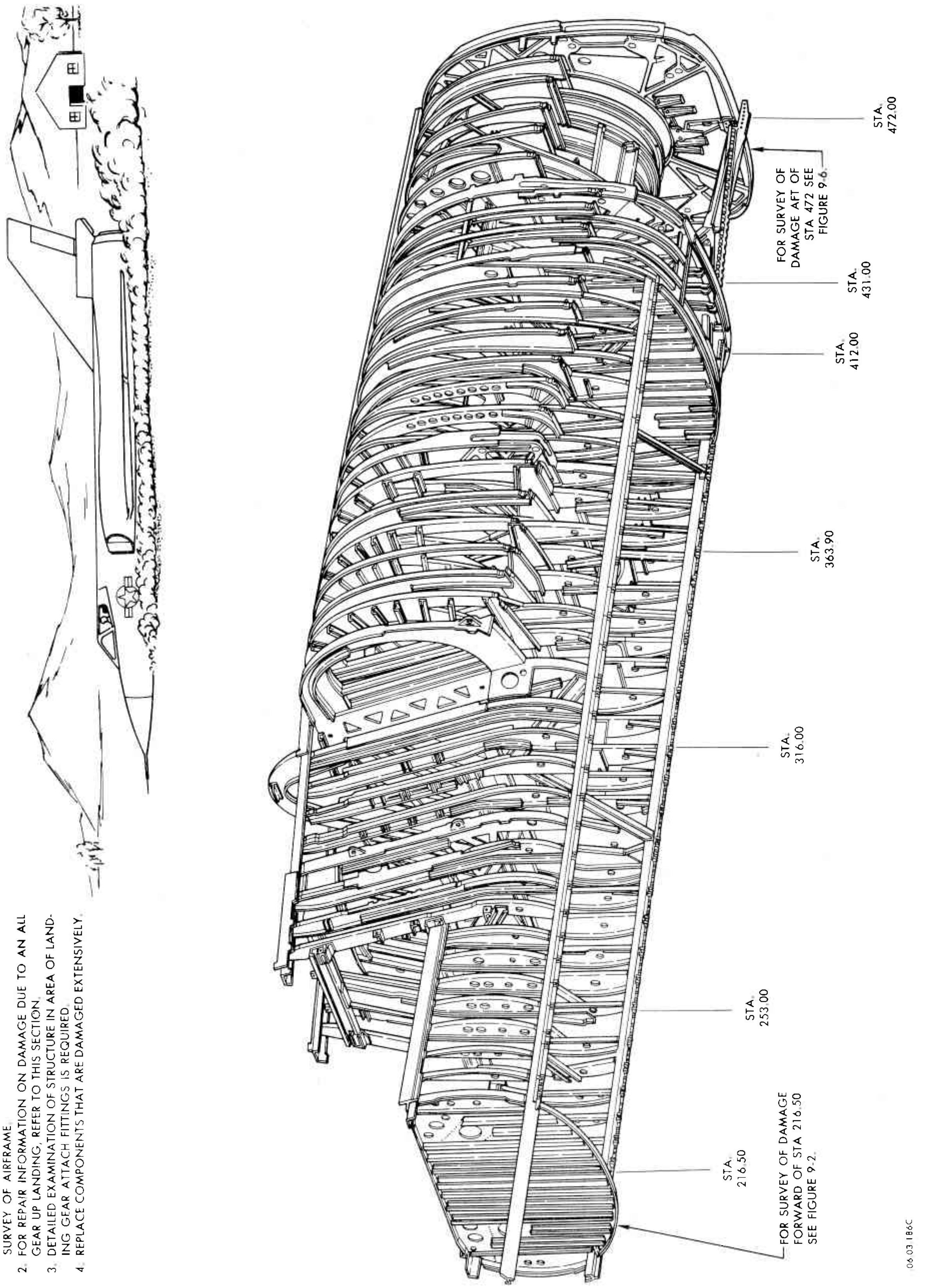


Figure 9-4. Damage Due to Landing Gear Failure — All Gear Up, F-106A

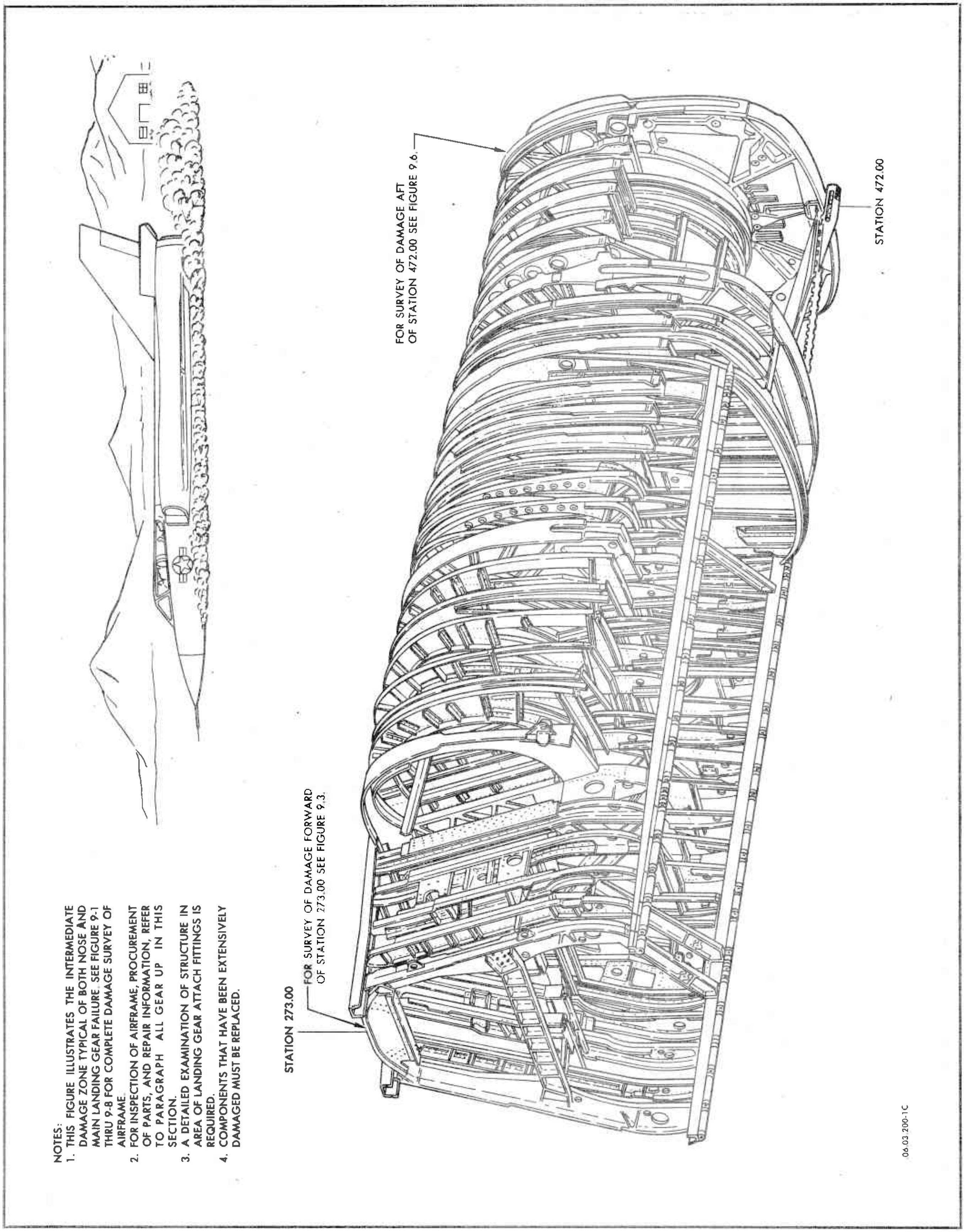


Figure 9-5. Damage Due to Landing Gear Failure — All Gear Up, F-106B

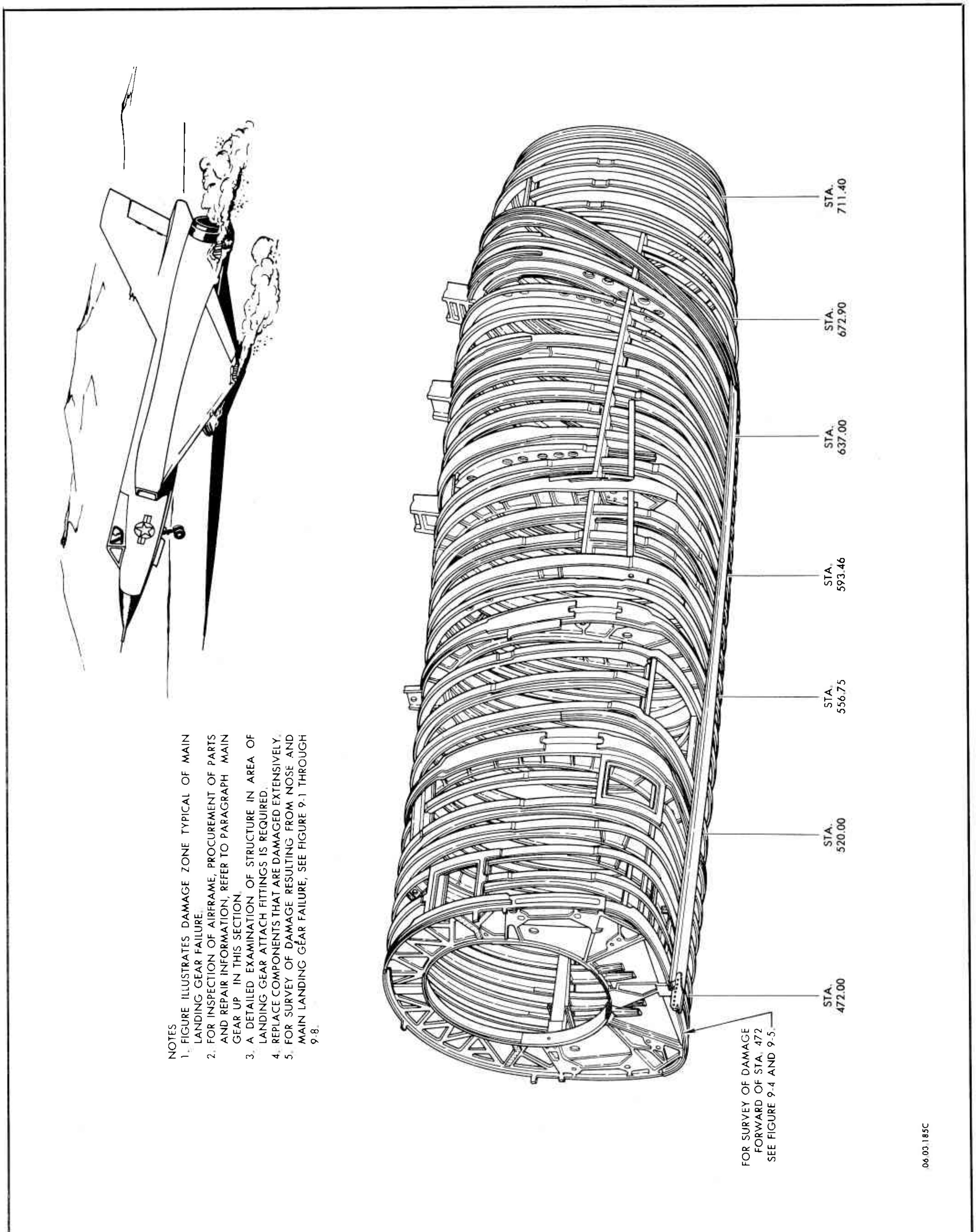
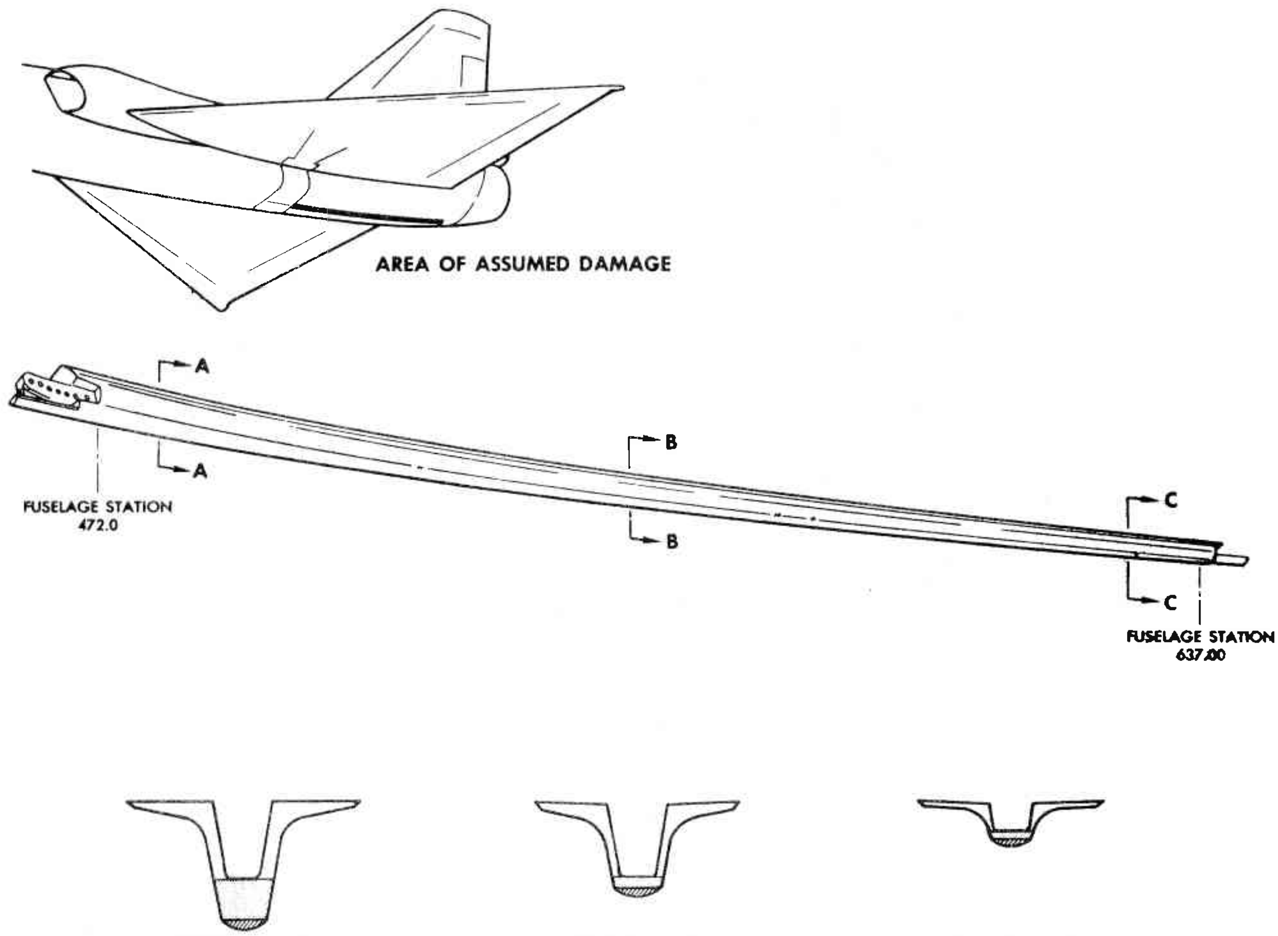


Figure 9-6. Damage Due to Landing Gear Failure — Main Gear Up



TYPICAL CROSS SECTION AREAS

DAMAGE LIMITS

MAX 10% OF LONGERON CROSS SECTION AREA MAY BE WORN AWAY BETWEEN FUSELAGE STATIONS 472.00 AND 637.00—AIRCRAFT REMAIN "IN SERVICE."

MAX OF 40% OF LONGERON CROSS SECTION WORN AWAY—AIRCRAFT MAY BE RELEASED FOR "ONE TIME FLIGHT" PROVIDED DAMAGE DOES NOT EXCEED CROSS SECTION AREA AS SHOWN.

CRACK TOLERANCES

a. REPAIR OF THE LONGERON HAS PROVEN INEFFECTIVE AND IS UNNECESSARY.

b. LONGERON WILL BE REPLACED WHEN CRACK(S) PROGRESS BETWEEN ALL THE HORIZONTAL BOLT HOLES ATTACHING TAIL HOOK TO LONGERON.

c. LONGERON WILL BE REPLACED WHEN A CRACK OR SERIES OF ABUTTED CRACKS, OTHER THAN INDICATED IN PARAGRAPH b., EXCEEDS 24 INCHES IN LENGTH.

NOTE

IF A SPARE LONGERON IS UNAVAILABLE, CONTACT THE F-106 SYSTEM SUPPORT MANAGEMENT DIVISION FOR INSTRUCTIONS CONCERNING CONTINUED USE.

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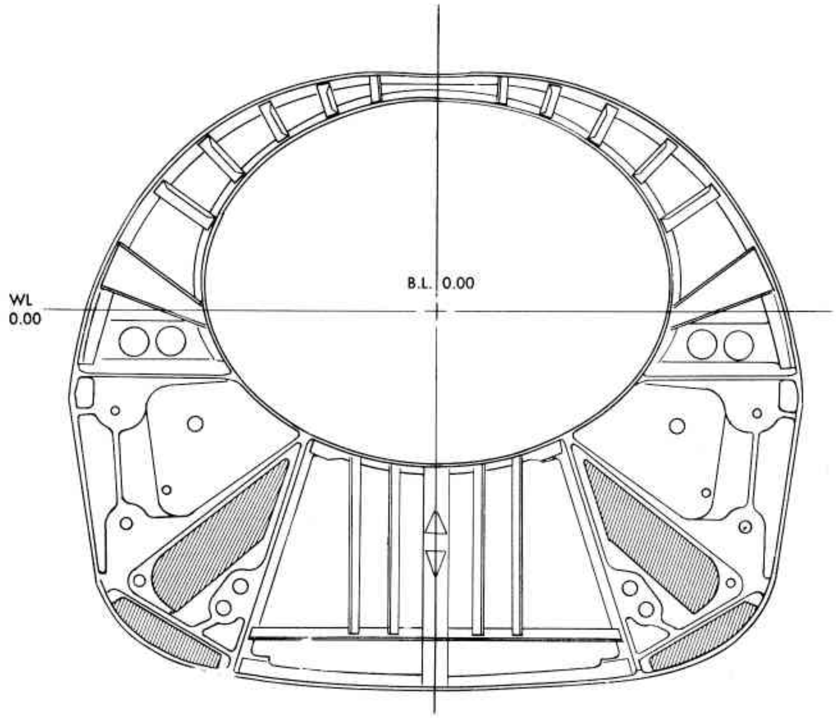
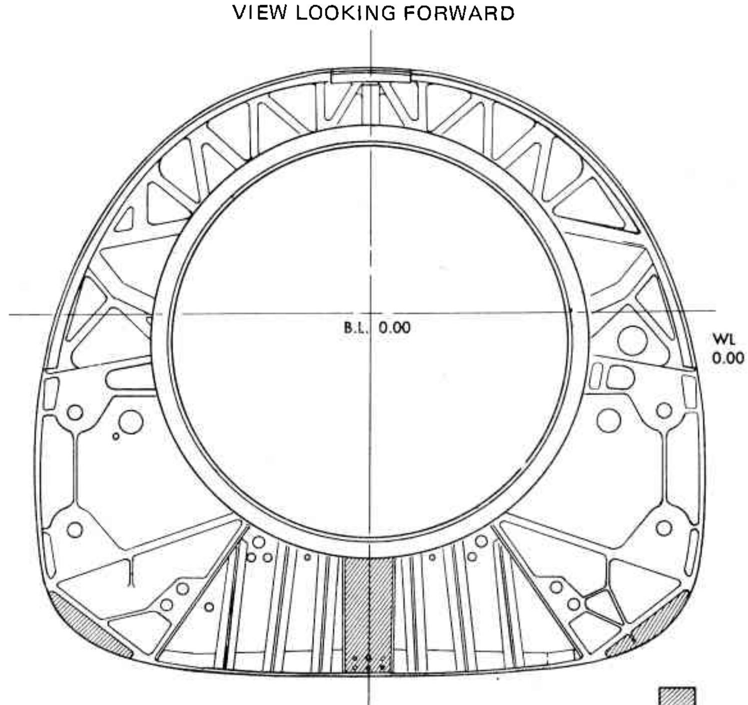
Figure 9-7. Damage Limits — Fuselage Lower Longeron



MACHINED BULKHEAD—STATION 431.00
VIEW LOOKING FORWARD



MACHINED BULKHEAD – STATION 472.00
VIEW LOOKING AFT

CROSS-LINED AREA INDICATES LIGHT METAL, NON-MACHINED AREAS WHICH MAY BE REPAIRED AS ILLUSTRATED IN SECTION X.

MACHINED AREA. WILL REQUIRE REPAIRS DESIGNED BY A STRUCTURES ENGINEER.

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Figure 9-8. Damage Limits — Machined Bulkheads, Stations 431.00 and 472.00

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landing gear attach point supporting structure for hidden damage or distortion. Check airframe alignment to insure against any distortion or deformation that may not be apparent in visual inspection. Check landing gear for alignment and operation. Refer to paragraph 1-268 for alignment procedures. Refer to T.O. 1F-106A-2-8-2-1 for landing gear operation procedure. Any parts suspected of being damaged shall be removed and inspected by the magnetic particle inspection or dye penetrant inspection method, whichever is applicable.

9-30. Removal of Damaged Area.

9-31. Support the airplane in such a manner as not to interfere with the repair or replacement of damaged structure; refer to paragraph 1-333 for instructions. Exercise care not to damage sound structure adjacent to damaged areas when making replacements. In reworking damaged structure, trim out damaged or distorted areas leaving ample fastener edge margins in the original structure to facilitate repairs. Trimmed edges should be smooth and should allow generous radii in the corners. Large structural components to be replaced may be disassembled at some convenient point, such as manufacturing splices for example, to facilitate the installation of new structure. Bolt-on components, such as the landing gear, wings, or elevons, should be removed to facilitate repairs.

9-32. Procurement of Parts.

9-33. Replacement parts or assemblies required for structural repair may be obtained from salvage, fabricated, or procured from spares sources. For identification of components and engineering drawing numbers, see figures 1-92 and 1-93 and refer to the Illustrated Parts Breakdown, T.O. 1F-106A-4 or T.O. 1F-106B-4.

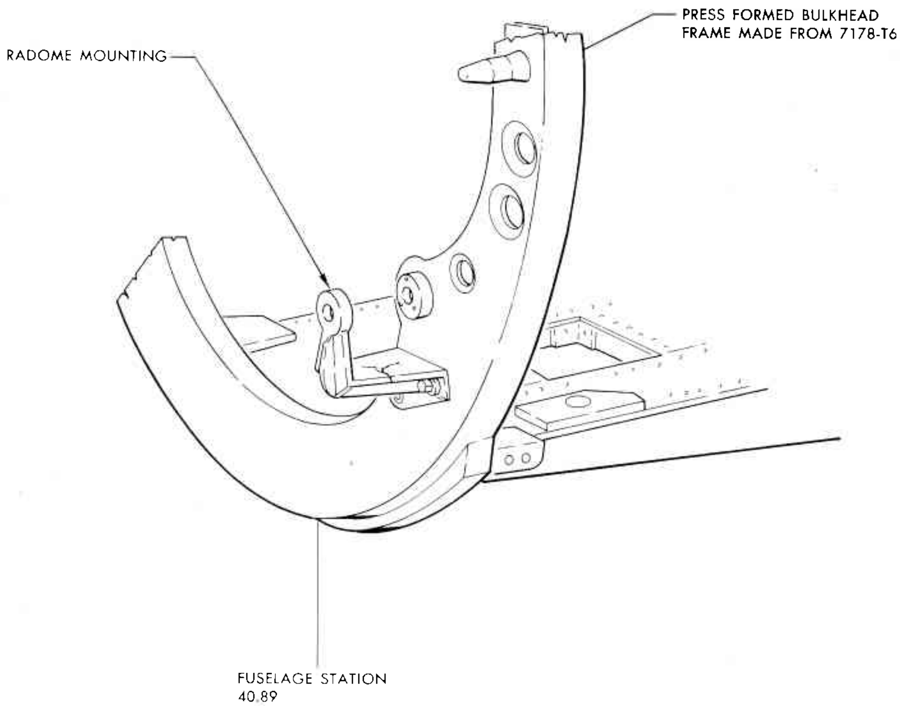
9-34. Installation of Parts.

9-35. Repairs and replacement of large structural components should be accomplished first. Installation of the large components should be done with the aid of jigs and fixtures to maintain proper alignment and facilitate repairs. Replacement parts should be checked for any mismatch with the original structure before installation. Some replacement parts may require rework before installation can be accomplished. Refer to Sections II and IV for a complete breakdown of the wing and fuselage structure and the types of material used in their construction.

9-36. SPECIFIC REPAIRS.

9-37. Figures 9-9 through 9-13 show specific repair procedures for built-up fuselage frames and bulkheads used in the F-106A and F-106B airplanes. The damage due to a landing gear failure will vary in each accident, and these illustrations are meant to be used as a guide for designing repairs within the different sections of the airplane. Refer to Section I for additional repair information.

a. Remove the fiberglas radome. Refer to T.O. 1F-106A-2-2-2-2 for radome removal procedure.
b. Open the left and right hand forward electronic compartment doors, and install an 8-96013 (1730-522-2745) door lock assembly in each door to hold the doors in an open position.
c. Remove two AN365-428 nuts, two AN960-416 washers and two AN4-10A bolts and remove the damaged radome mounting.



NOTES:
1. REFER TO SECTION IV FOR PERMISSIBLE REPAIRS TO THE FIBERGLAS RADOME.
2. REFER TO PARAGRAPH INVESTIGATION OF DAMAGE IN SECTION 1 FOR DAMAGE EVALUATION PROCEDURE.

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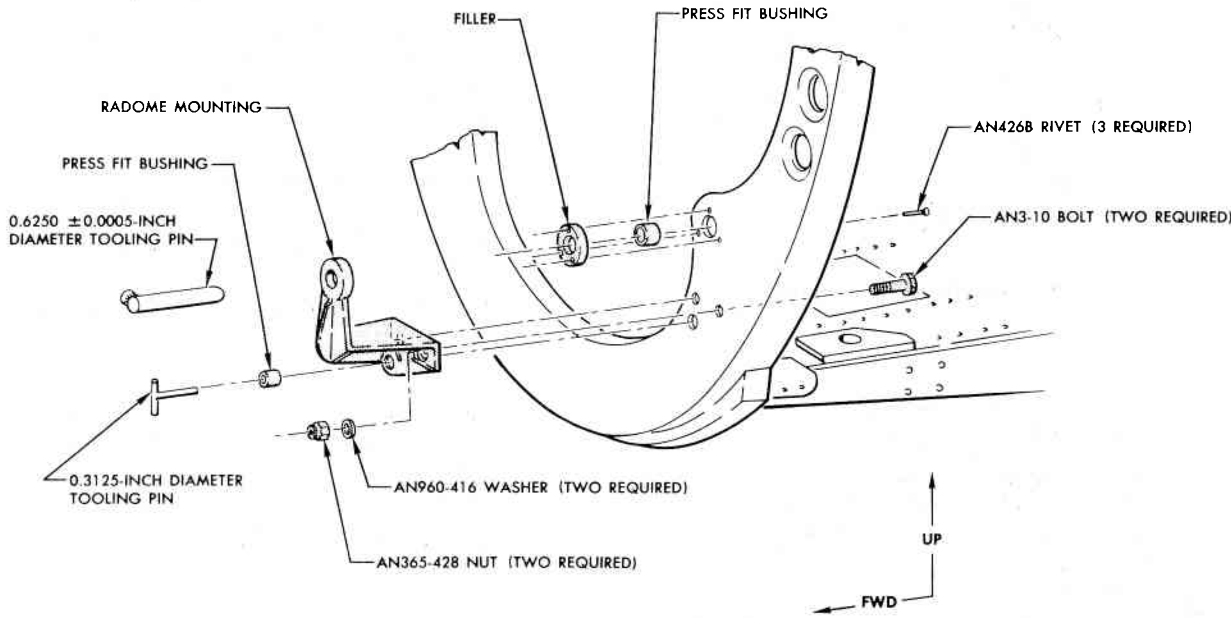
Figure 9-9. Replacement of Radome Mounting (Sheet 1 of 2)



d. Using a No. 30 (0.128) drill, remove the three AN426B rivets connecting the filler to the bulkhead frame.
e. Remove the damaged filler.
f. Remove the press fit bushing from the bulkhead if it has been damaged.
g. Install a new press fit bushing in the bulkhead if the original bushing was damaged.
h. Install a new filler. Filler may be ordered from spare stock or may be fabricated locally from 7075-T6 bar stock. Dimensions for fabricating a new filler may be obtained from a similar filler on the opposite side of the airplane.
i. Install a new radome mounting.
j. Place 0.6250 (± 0.005)-inch tooling pin through the radome mounting and the fuselage bulkhead as shown.
k. Place a 0.3125 (± 0.0015)-inch tooling pin through the radome mounting and the fuselage bulkhead as shown.
l. Locate and drill three No. 30 (0.128) holes in the new filler. Locate holes from existing holes in the fuselage bulkhead.
m. Secure the new filler to the fuselage bulkhead with AN426B rivets.
n. Install two new AN 4-10A bolts through the new radome mounting and the fuselage bulkhead. Install the bolts from the aft side of the bulkhead.
o. Place an AN960-416 washer over each bolt and secure with new AN365-428 nuts.
p. Tighten the nuts according to the standard torque values given on figure "Standard Bolt Torque Values" in Section 1.
q. Remove the tooling pins.
r. Reinstall the radome. Refer to T.O. 1F-106A-2-2-2 for radome installation procedure.

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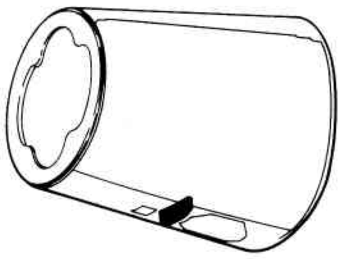
Figure 9-9. Replacement of Radome Mounting (Sheet 2 of 2)



ASSUMED AREA OF DAMAGE

REPAIR PROCEDURE

a. Support the airplane on support fixtures to allow access to the damaged area. Refer to Airplane Handling Equipment in Section I.
b. Open the left and right hand forward electronic compartment doors and install an 8-96013 (1730-522-2745) door lock assembly in each door to hold the doors in an open position.
c. Remove the electronic equipment from the opened compartment. Refer to T.O. 1F-106A-2-27-2 for the electronic removal procedure.
d. Remove the paint and any foreign material from the outer surface of the lower panel between stations 40.89 and 102.00.

CAUTION

REFER TO PAINTING PRECAUTIONS IN AREA OF CANOPY PLEXIGLASS WINDOWS IN SECTION I IF AN AROMATIC TYPE SOLVENT IS USED TO REMOVE PAINT AND FOREIGN MATERIAL IN THIS AREA.

e. Using a No. 30 (0.128) drill, remove all rivets which attach the skin to the fuselage structure between stations 40.89 and 102.00.

NOTE

A DRILL STOP SHOULD BE USED WHEN REMOVING RIVETS FROM THE SKIN, TO PREVENT DAMAGE TO ANY UNDERLYING STRUCTURE.

f. Remove the skin and hold for comparison when fabricating a new skin.
g. Remove a sufficient number of vertical stiffener angles from the damaged frame to allow removal of damaged portion of the frame. Use a No. 30 (0.128) drill to remove rivets.

NOTE

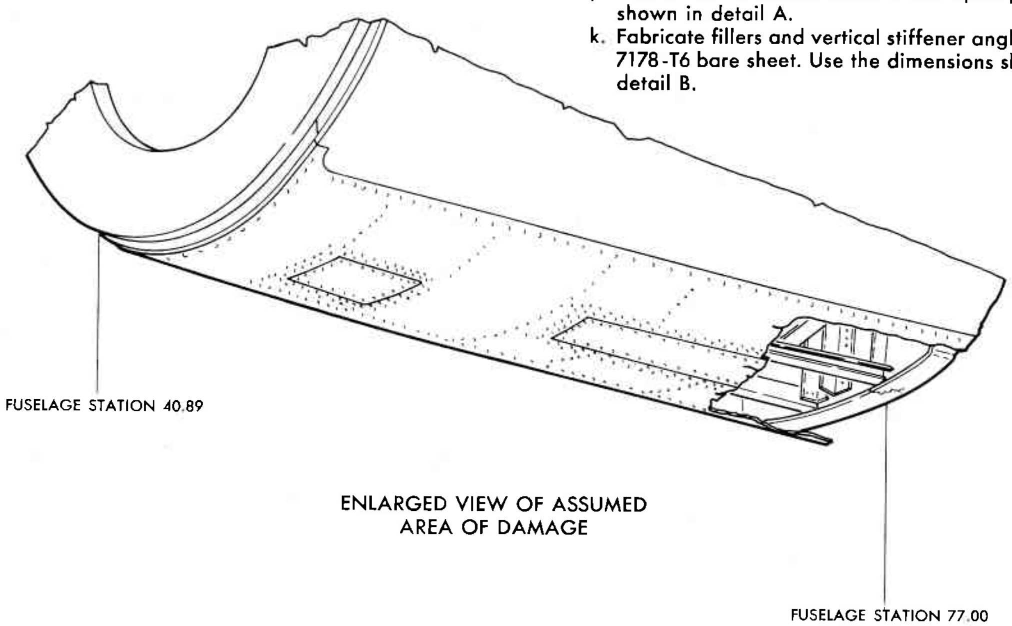
DUE TO THE PROXIMITY OF THE FRAME WORK IN THIS AREA, IT MAY BE NECESSARY TO USE A RIGHT ANGLE OR SNAKE DRILL TO REMOVE THE RIVETS ATTACHING THE VERTICAL STIFFENER ANGLES TO THE FRAME.

h. Remove the damaged portion of the frame. Use metal cutting hand shears to cut out the damaged portion of the frame. Finish cutting out the damaged area with hand files until a smooth shape is obtained.

NOTE

IF FORTY PER CENT OR MORE OF THE FRAME HAS BEEN DAMAGED, IT IS RECOMMENDED THAT THE FRAME BE REPLACED.

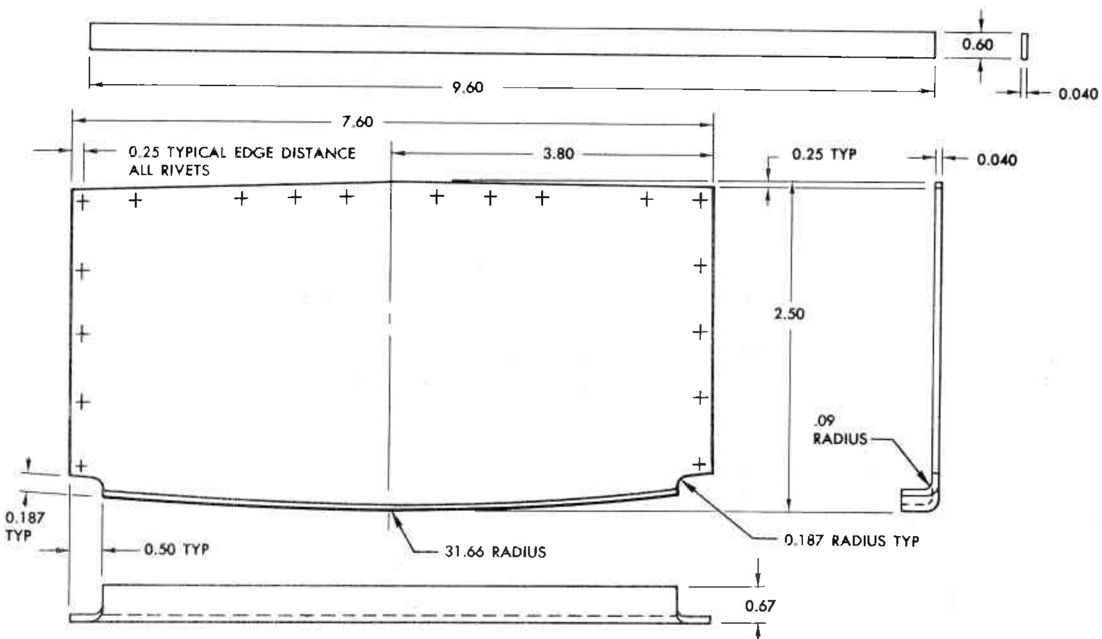
i. Fabricate a repair plate to cover the cut out area in the frame. Fabricate the repair plate from 7178-T6 bare sheet one gage thicker than the frame being repaired.
j. Predrill No. 39 (0.099) holes in the repair plate as shown in detail A.
k. Fabricate fillers and vertical stiffener angles from 7178-T6 bare sheet. Use the dimensions shown in detail B.



ENLARGED VIEW OF ASSUMED AREA OF DAMAGE

06.03.217-1A

Figure 9-10. Lower Panel Damaged Frame Repair— Fuselage Stations 40.89 to 102.00 (Sheet 1 of 5)



DETAIL A

NOTE
APPLY PROTECTIVE COATING TO ALL REPAIR PARTS PRIOR TO INSTALLATION. SEE FIGURE "PRIMER AND PAINT COATING" IN SECTION I FOR TYPE AND AMOUNT OF PROTECTIVE COATING TO BE APPLIED TO PARTS IN THIS AREA.

l. Secure the repair plate to the frame with clamps. Align the repair plate so that the contour of the repair plate matches the contour of the frame being repaired.

m. Using the predrilled holes in the repair plate as a guide, drill holes through the frame. Use a No. 30 (0.128) drill.

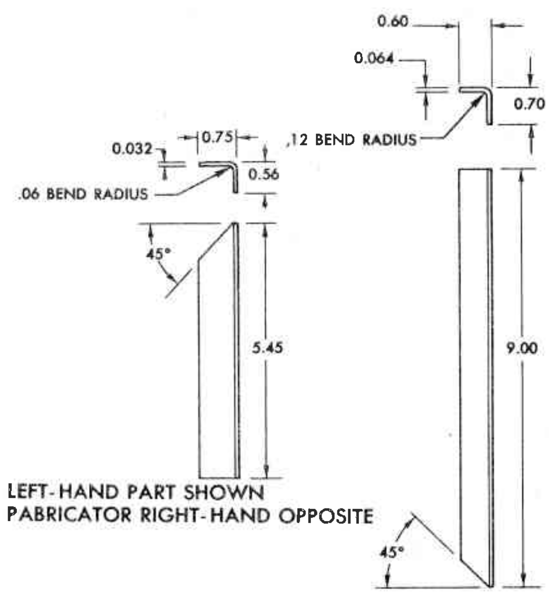
NOTE
WHEN MATING REPAIR PARTS TO THE ORIGINAL FRAME WORK, REMOVE THE REPAIR PARTS UPON COMPLETION OF DRILLING, IF NECESSARY, TO REMOVE BURRS AND DRILL CHIPS FROM BETWEEN THE FAYING SURFACES.

n. Rivet the repair plate to the frame with AN470-AD-4 rivets.

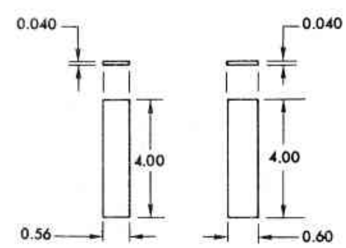
o. Secure the vertical stiffener angles and fillers in place with clamps.

p. Using the existing holes in the upper portion of the original frame as a guide, drill holes through the fillers and angles. Use a No. 30 (0.128) drill.

q. Layout and drill holes through the lower portion of the repair angles, and repair plate. Use a No. 39 (0.099) drill. See figure "Standard Minimum Rivet Spacing" in Section 1 for rivet spacing and edge distance information.



DETAIL B



DETAIL C

NOTE:
ALL RIVETS ARE TO BE EQUALLY SPACED BETWEEN LOCATED END RIVETS. SEE FIGURE "STANDARD MINIMUM RIVET SPACING" IN SECTION 1.

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Figure 9-10. Lower Panel Damaged Frame Repair — Fuselage Stations 40.89 to 102.00 (Sheet 2 of 5)

r. Ream the 0.099 holes with a No. 20 (0.161) drill.
s. Rivet the vertical stiffener angles and fillers in place with AN470AD-5 rivets.
t. Fabricate a new skin from 0.032-inch 7178-T6 bare sheet.

NOTE

LEAVE A MINIMUM OF ONE INCH ACCESS MATERIAL AROUND THE PERIMETER OF THE NEW SKIN FOR TRIM PURPOSES.

u. Cut access openings in the new skin as shown, except reduce the cutout dimensions by one inch.
v. Layout and drill holes through the contoured angle of the repair plate. See figure "Standard Minimum Rivet Spacing" in Section 1 for rivet spacing and edge distance.
w. Using a grease pencil, draw a center line on the exterior surface of the new skin.
x. Hold the new skin in place and align the center line on the new skin with the bottom center line of the airplane.
y. Using the existing holes in the fuselage frame work, drill holes through the skin. Use a No. 30 (0.128) drill.

NOTE

TO KEEP THE NEW SKIN FROM WRINKLING, COMMENCE THE DRILLING OPERATION APPROXIMATELY IN THE CENTER OF THE NEW SKIN AND PROCEED TO DRILL OUTBOARD, FORWARD AND AFT. INSERT CLECOS THROUGH THE HOLES AS THEY ARE DRILLED TO PREVENT THE SKIN FROM SLIPPING OUT OF ALIGNMENT.

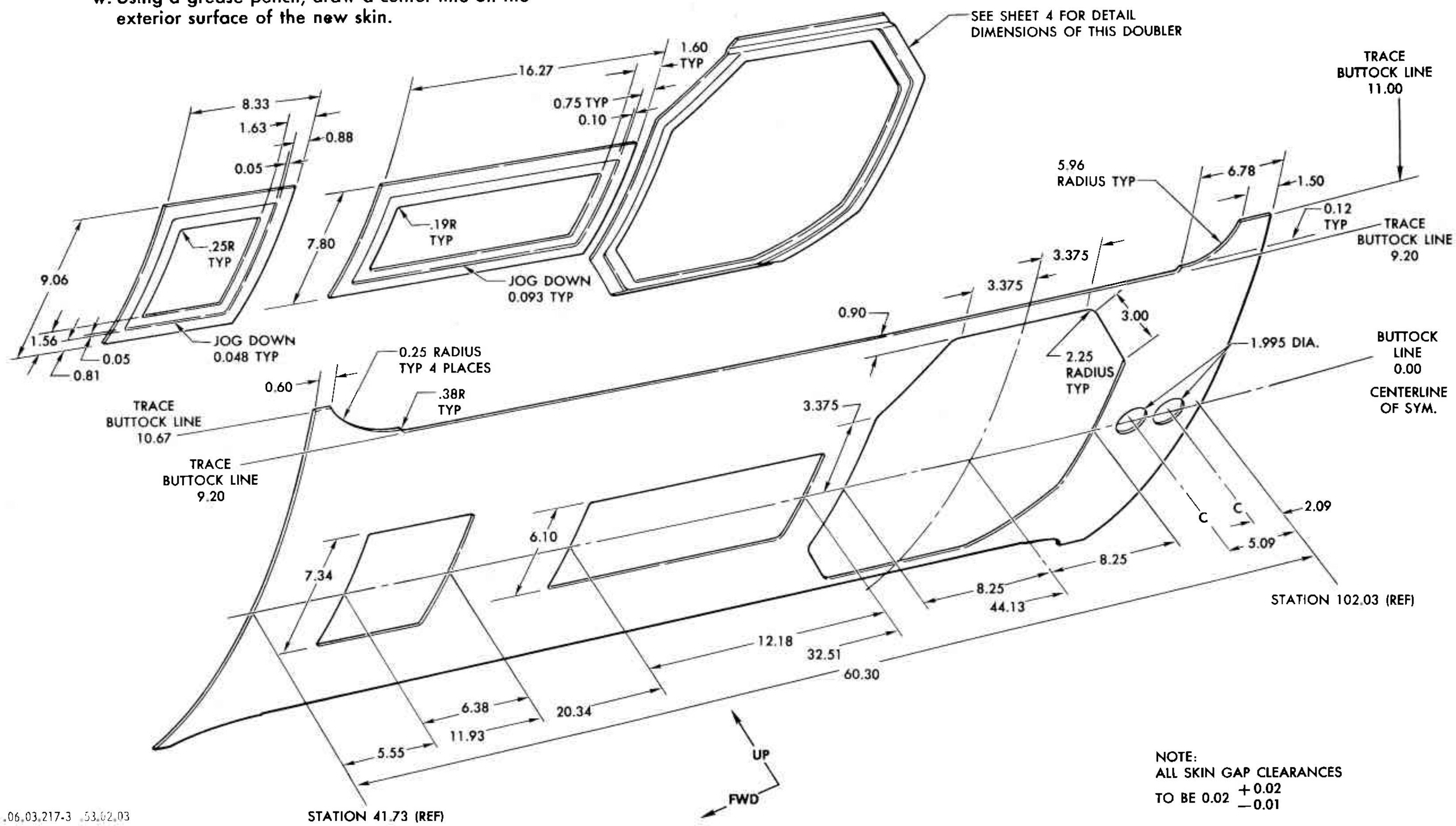
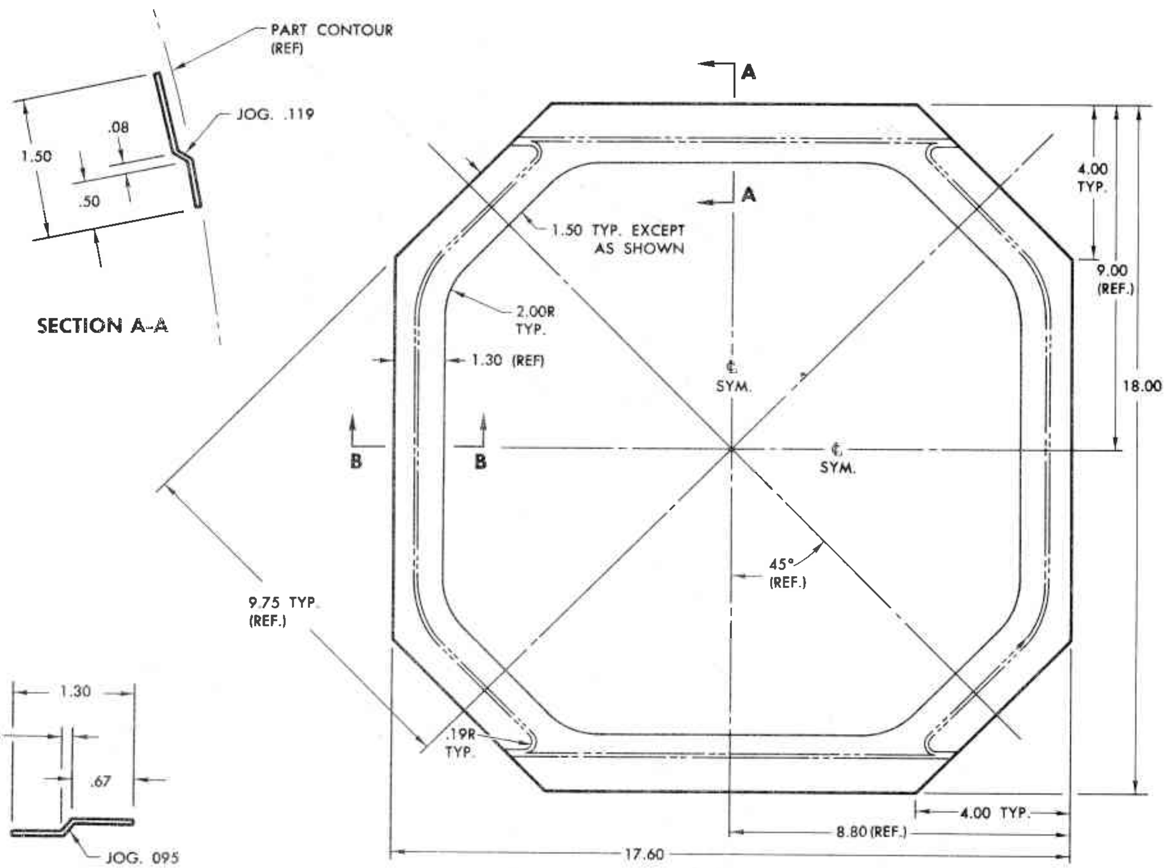


Figure 9-10. Lower Panel Damaged Frame Repair — Fuselage Stations 40.89 to 102.00 (Sheet 3 of 5)



SECTION B-B **DOUBLER**

z. Remove the skin and trim the perimeter according to the dimensions shown.

aa. All holes in the skin must be dimpled. See figure "Flush Rivet Installation" in Section 1 for dimpling information.

ab. Reinstall the skin and secure in place with AN426-AD-4 rivets.

NOTE
DO NOT INSTALL RIVETS THROUGH SKIN AND REPAIR PLATE UNTIL STEPS AC. AND AD. HAVE BEEN COMPLETED.

ac. Fabricate a splice from 0.040 7178-T6 bare sheet to the dimensions shown to splice the repair plate to the original frame.

ad. Place the splice on the inner surface of the repair plate flange and using the existing holes in the skin, repair plate and original frame as a guide, drill holes through the splice. Use a No. 20 (0.161) drill.

ae. Rivet the spliced area with AN426AD-5 rivets.

af. Trim the access openings to the dimensions shown.

ag. Fabricate doublers from 0.032 7178-T6 bare sheet to the dimensions shown in detail D.

ah. Layout and drill a double row of pilot holes around the access openings of the new skin. See figure "Standard Minimum Rivet Spacing" in Section 1. Use a No. 39 (0.099) drill.

ai. Clamp the doublers to the inner surface of the skin and using the predrilled holes in the skin as a guide, drill holes through the skin and doublers. Use a No. 30 (0.128) drill.

aj. Remove the doublers and dimple the skin and doublers for flush type rivets.

ak. Reinstall the doublers and secure in place with AN426AD-4 rivets.

al. Fabricate new doors for the access openings. The forward door is made from 0.080 7178-T6 bare sheet. The middle door is made from 0.125 7178-T6 bare sheet and the aft door is made from pre-impregnated fiberglas cloth.

NOTE
THE AFT FIBERGLASS DOOR CANNOT BE FABRICATED WITHOUT THE AID OF SPECIAL HEAT AND PRESSURE EQUIPMENT, THEREFORE, REPLACEMENT OF THIS COMPONENT SHOULD BE CONSIDERED.

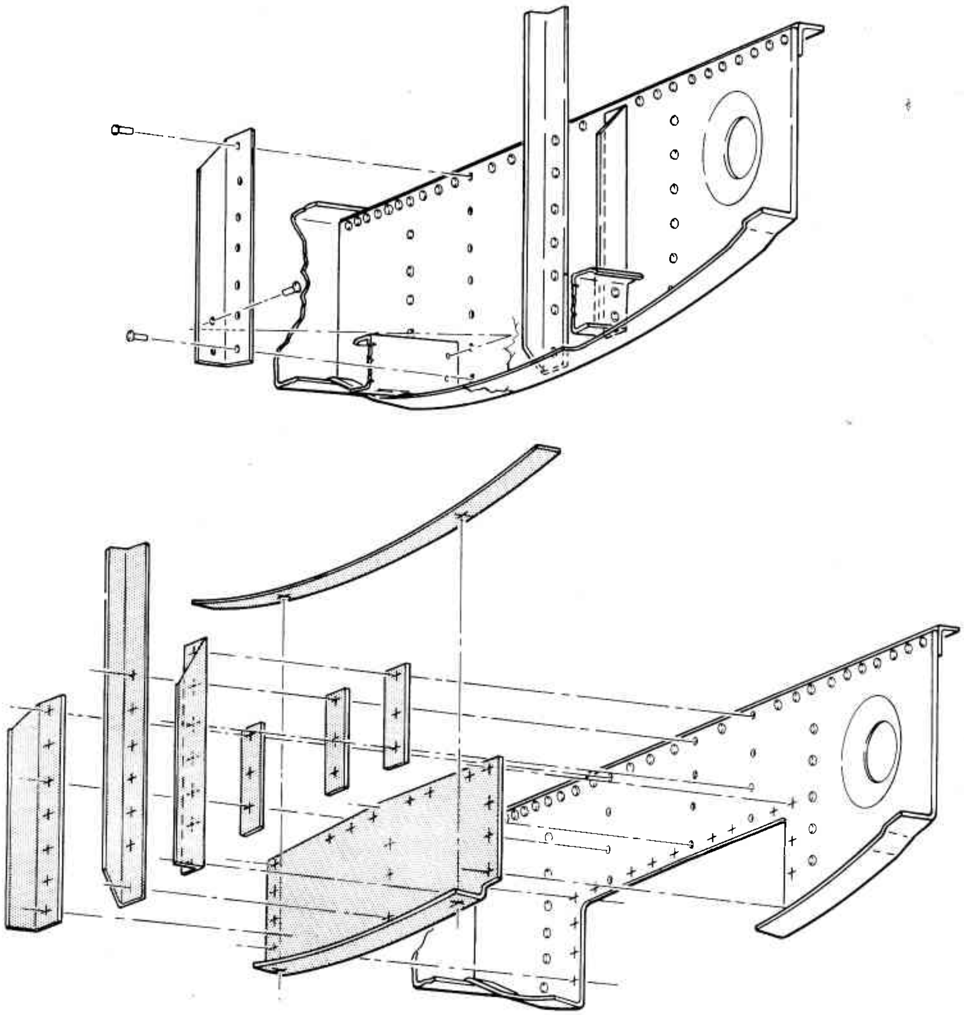
am. Layout and drill pilot holes in the new access doors. Determine the spacing of the pilot holes from the old doors.

an. Hold the access doors in place and using the pilot holes in the doors as a guide, drill holes through the doublers. Use a No. 39 (0.099) drill.

ao. Remove the doors and ream the holes in the doublers with a No. 10 (0.1935) drill.

.06.03.217-4 .53.02.03

Figure 9-10. Lower Panel Damaged Frame Repair — Fuselage Stations 40.89 to 102.00 (Sheet 4 of 5)



FINAL ASSEMBLY SEQUENCE VIEW

ap. Install gang channels and nut plates on the inner surface of the doublers as required. Refer to the old doublers for types of gang channels and nut plates to be used and their method of attachment.

aq. Ream the holes in the access doors with a No. 10 (0.1935) drill. Machine countersink 100° all holes in the access doors on the exterior surface.

ar. Install the forward and middle doors with AN509-10R10 screws. Install the aft door with AN509-10R18 screws.

as. Fill the gaps between the access doors and the skin with aerodynamic smoothing compound, EC1293.

at. Finish the exterior surface of the repair area according to Primer and Paint Coating Procedures in Section I.

CAUTION

REFER TO PAINTING PRECAUTIONS IN AREA OF CANOPY PLEXIGLASS WINDOW IN SECTION I PRIOR TO TOUCHUP PAINTING.

au. Reinstall the electronic equipment which was removed in step C. Refer to T.O. 1F-106A-2-27-2 for installation procedure.

av. Remove the 8-96013 (1730-522-2745) door lock assemblies, and close and lock the forward electronic equipment doors.

06.03.217-5A

Figure 9-10. Lower Panel Damaged Frame Repair — Fuselage Stations 40.89 to 102.00 (Sheet 5 of 5)

DAMAGE REMOVAL PROCEDURE

a. Support airplane in support fixtures to allow access to damaged area. See figure Airplane Handling Equipment in Section I.

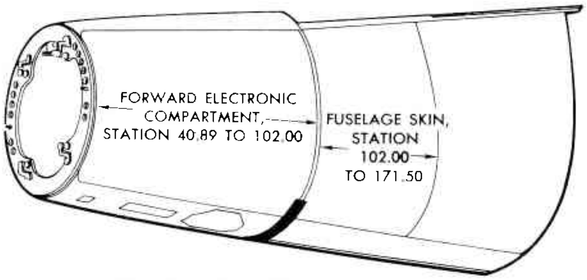
b. Open forward electronic compartment doors and install an 8-96013 (1730-522-2745) lock assembly in each door. Refer to T.O. 1F-106A-2-2-2-2 for lock assembly installation procedure.

c. Remove all electronic equipment from the opened compartment. Refer to T.O. 1F-106A-2-27-2 for electronic equipment removal procedure.

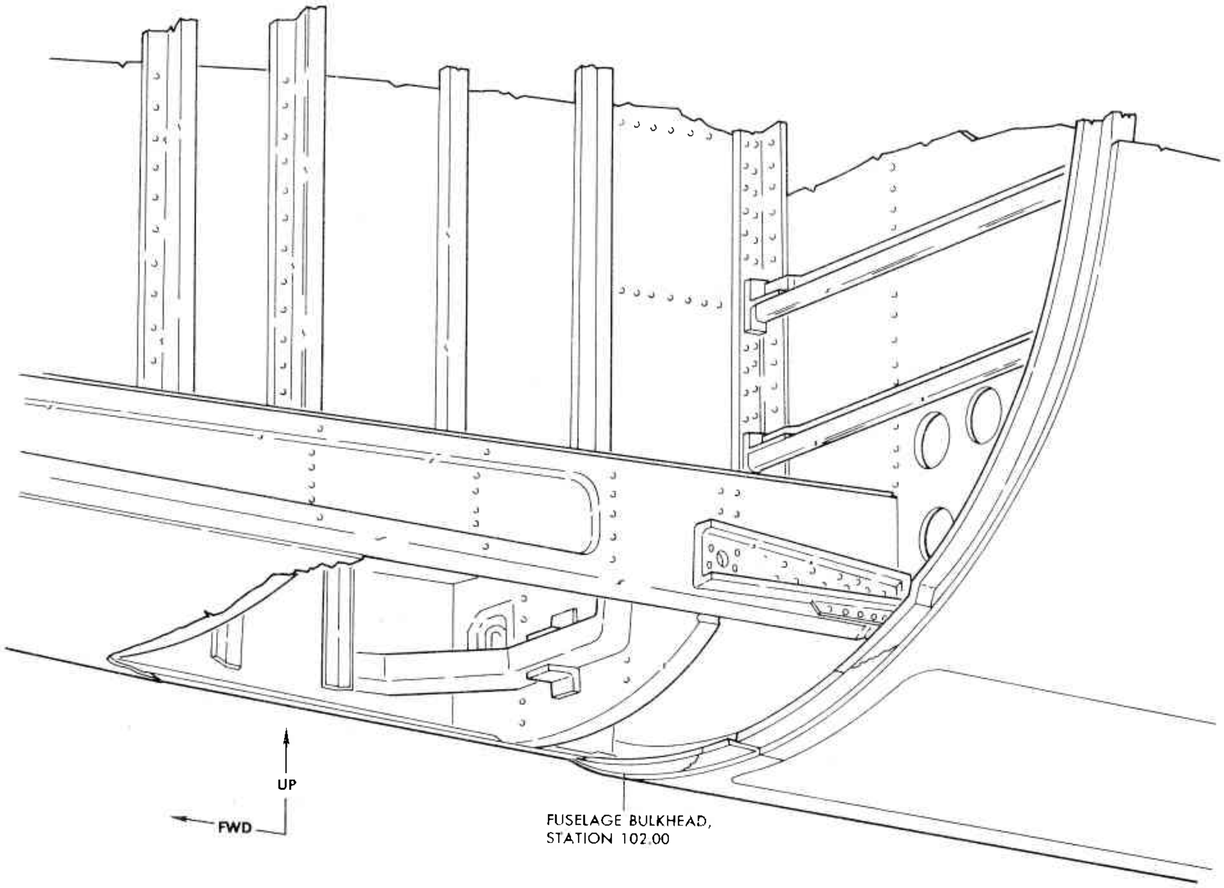
d. Remove damaged nose landing gear. Refer to T.O. 1F-106A-2-8-2-1 for nose landing gear removal procedure.

NOTE

SEE FIGURE 9-10 FOR REPAIR OF FUSELAGE FRAMES FORWARD OF STATION 102.00. SEE FIGURE 9-12 FOR REPAIR OF FUSELAGE FRAMES AFT OF STATION 102.00.

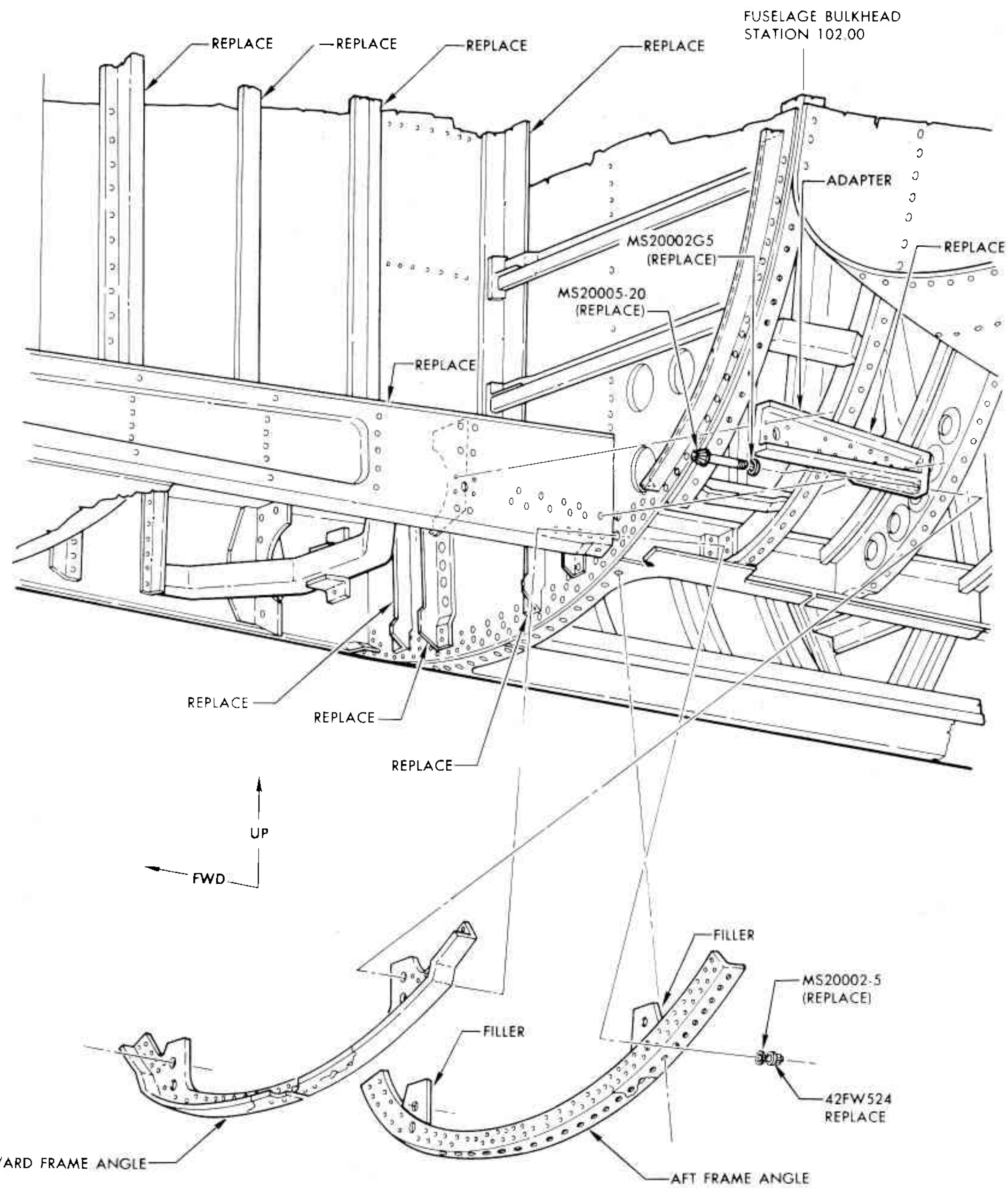


AREA OF ASSUMED DAMAGE
FUSELAGE STATION 102.00



06.03.218-1B

Figure 9-11. Bulkhead Repair — Fuselage Station 102.00 (Sheet 1 of 4)



REMOVAL OF DAMAGED FUSELAGE FRAMEWORK

e. Remove electrical equipment from nose wheel well. Refer to T.O. 1F-106A-2-10-2-1 for electrical equipment removal procedure.

f. Remove paint finish from exterior surface of airplane to completely expose damaged area. Refer to T. O. 1F-106A-2-2-2-2 for approved airframe cleaning materials and procedures.

g. Remove outer skin from lower panel between stations 40.89 and 102.00. Refer to Table 1-IXLIV for drill sizes and hole diameter limits.

NOTE

REPAIR PROCEDURES FOR LOWER PANEL BETWEEN STATION 40.89 AND 102.00 ARE SHOWN ON FIGURE 9-10.

h. Remove a sufficient number of rivets from left- and right-hand skin panels between stations 102.00 and 171.50 to allow removal of damaged portion of the skins and to allow access to the internal structure.

NOTE

THE COMPONENTS MARKED "REPLACE" SHOULD BE REMOVED AND REPLACED BY NEW PARTS IF THEY HAVE BEEN DAMAGED BEYOND THE ALLOWABLE NEGLIGIBLE DAMAGE LIMITS GIVEN IN TABLE 4-1.

i. Remove a sufficient number of rivets from fuselage bulkhead frame to allow removal of damaged portion of frame.

06.03.218-2B

Figure 9-11. Bulkhead Repair — Fuselage Station 102.00 (Sheet 2 of 4)

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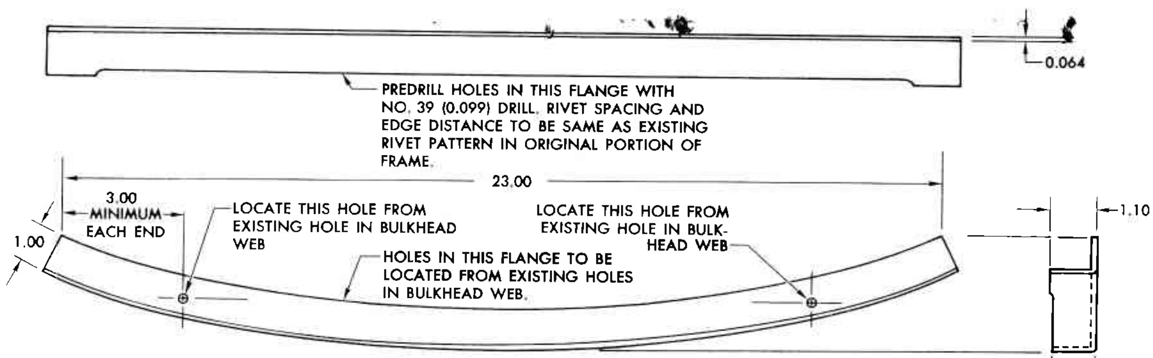
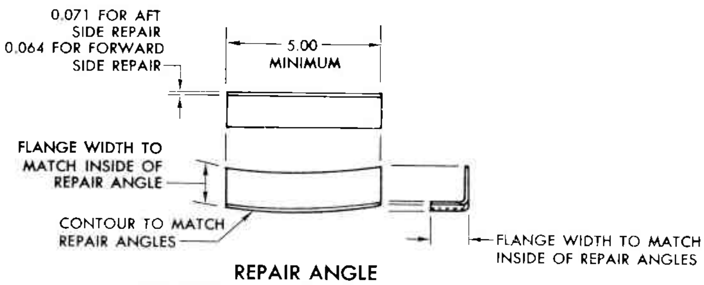
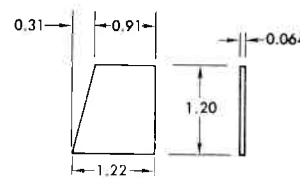


REPAIR ANGLE
FOR FORWARD SIDE OF BULKHEAD



REPAIR ANGLE
FOR FORWARD SIDE OF BULKHEAD



FILLER

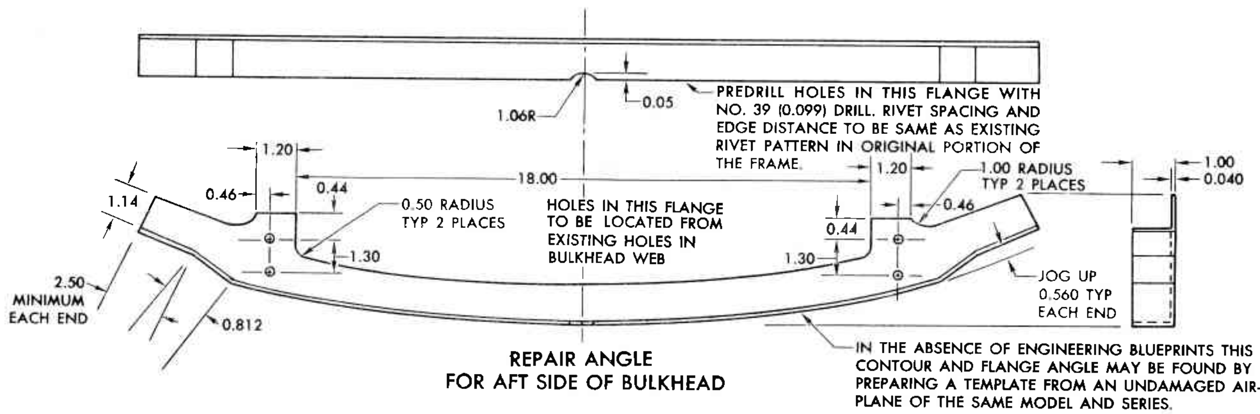
j. If necessary, remove fillers from between vertical angles and bulkhead web.
k. Thoroughly remove all sealant material from repair area. Use aliphatic naphtha, Federal Specification TT-N-95A and a clean, lint-free cloth to remove sealant material.

FABRICATION OF REPAIR PARTS

a. Fabricate a repair angle for the forward side of the bulkhead from 7178-O aluminum alloy. Heat treat to the -T6 condition per specification MIL-H-6088 after forming. Use dimensions shown on sheet 3 to form this part.
b. Fabricate a repair angle for the aft side of the bulkhead from either 7178-O or from extruded aluminum alloy No. E121209. If part is made from 7178-O, it must be heat treated to the -T6 condition per specification MIL-H-6088 after forming. Use dimensions shown on sheet 3 to form this part.
c. Fabricate skin patches as necessary from 0.040 gage, 7178-T6 aluminum alloy.

NOTE
ALL HOLES IN SKIN PATCHES MUST BE MACHINE COUNTERSUNK.

d. Apply a protective coating to all repair parts after forming. See figure "Primer and Paint Coatings" in Section 1.

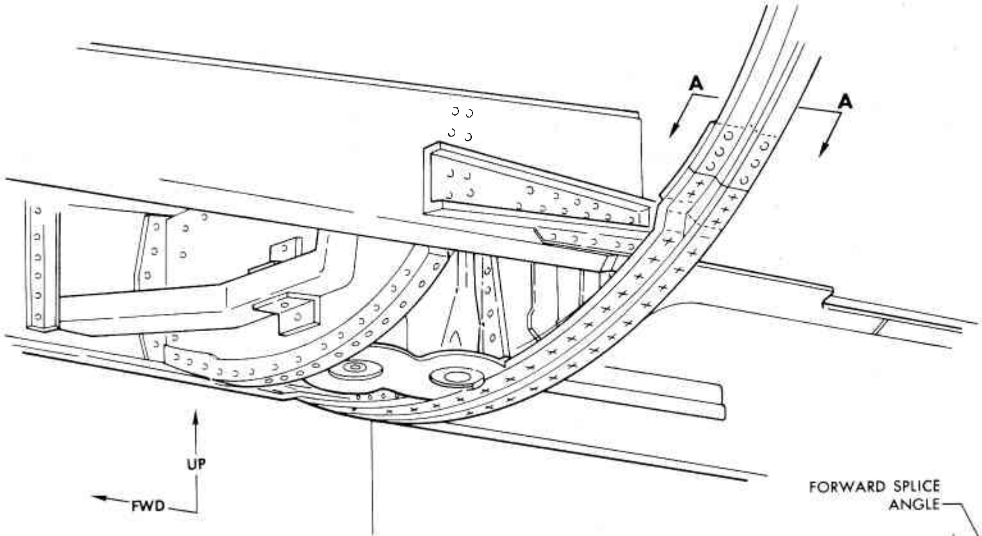


REPAIR ANGLE
FOR AFT SIDE OF BULKHEAD

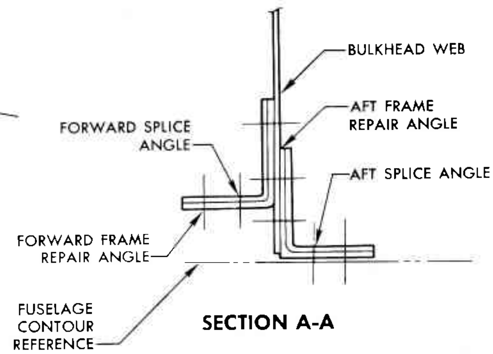
NOTE
SEE FIGURE "FLAT PATTERN LAYOUT" IN SECTION 1 FOR FLAT PATTERN LAYOUT INFORMATION, AND REFER TO TABLE "STANDARD BEND RADII FOR FLAT SHEET" IN SECTION 1 FOR BEND RADII INFORMATION.

.06.03.218-3 .53.04.03

Figure 9-11. Bulkhead Repair — Fuselage Station 102.00 (Sheet 3 of 4)



INSTALLATION VIEW



SECTION A-A

REPAIR INSTALLATION PROCEDURE

a. Install forward bulkhead repair angle and hold in place with clamps.
b. Using existing holes in bulkhead web as a guide, drill holes in repair angle.
c. Install aft bulkhead repair angle and hold in place with clamps.
d. Using holes in bulkhead web and forward repair angle as a guide, drill holes in aft repair angle.
e. Install splice angles on forward side of bulkhead and hold in place with clamps.
f. Using existing holes in bulkhead frame and repair angles as a guide, drill holes through splice angles.
g. Install splice angles on aft side of bulkhead in line with splice angles on forward side of bulkhead, and hold in place with clamps.
h. Using existing holes in bulkhead frame, repair angles and forward splice angles as a guide, drill holes through aft splice angles.
i. Remove all repair parts from bulkhead, then remove all burrs and drill chips.
j. Apply a film of MIL-S-8802 sealant to faying surfaces of repair parts.

k. Reinstall repair parts and hold in place with clecos.
l. Rivet repair angles and splice angles to bulkhead. Use Q4310 rivets.
m. Reinstall left- and right-hand adapters. Use same type and size of fasteners that were used in the original construction.
n. Install skin patches aft of station 102.00 and hold in place with clamps.
o. Using existing holes in fuselage framework as a guide, drill holes through skin patches. Countersink all holes in skin patch exterior surface 100°.

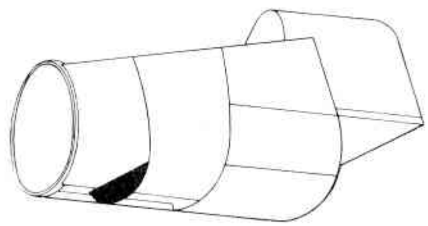
NOTE
HOLES THAT CANNOT BE LOCATED FROM THE INTERNAL STRUCTURE MAY BE DRILLED WITH THE AID OF A HOLE FINDER. SEE HOLE FINDER FIGURE IN SECTION 1.

p. Remove skin patches, then remove all burrs and drill chips.
q. Apply a film of MIL-S-8802 sealant to faying surface of repair angles and skin patches.

r. Reinstall skin patches and rivet in place. Use AN426AD-5 rivets.
s. Reinstall lower skin panel between stations 40.89 and 102.00.
t. Using predrilled holes in forward bulkhead repair angle as a guide, drill holes in lower panel. All holes in lower panel, exterior surface, must be countersunk 100°.
u. Rivet lower skin panel to bulkhead repair angle with AN426AD-4 rivets.
v. Refinish exterior surface of airplane in the damaged area according to procedures outlined on figure Primer and Paint Coatings in Section 1.
w. Reinstall electrical equipment in nose wheel well. Refer to T.O. 1F-106A-2-10-2-1 for electrical equipment installation and checkout procedure.
x. Reinstall nose landing gear. Refer to T.O. 1F-106A-2-8-2-1 for nose landing gear installation operational checkout procedures.
y. Reinstall electronic equipment in forward electronic compartment. Refer to T.O. 1F-106A-2-27-2 for electronic equipment installation and checkout procedure.
z. Remove the 8-96013 (1730-522-2745) door lock assemblies from left and right hand forward electronic compartment doors.
aa. Close and lock forward electronic compartment doors. Refer to T.O. 1F-106A-2-2-2-2 for door closing and locking procedure.

06.03.218-4A

Figure 9-11. Bulkhead Repair — Fuselage Station 102.00 (Sheet 4 of 4)



AREA OF ASSUMED DAMAGE

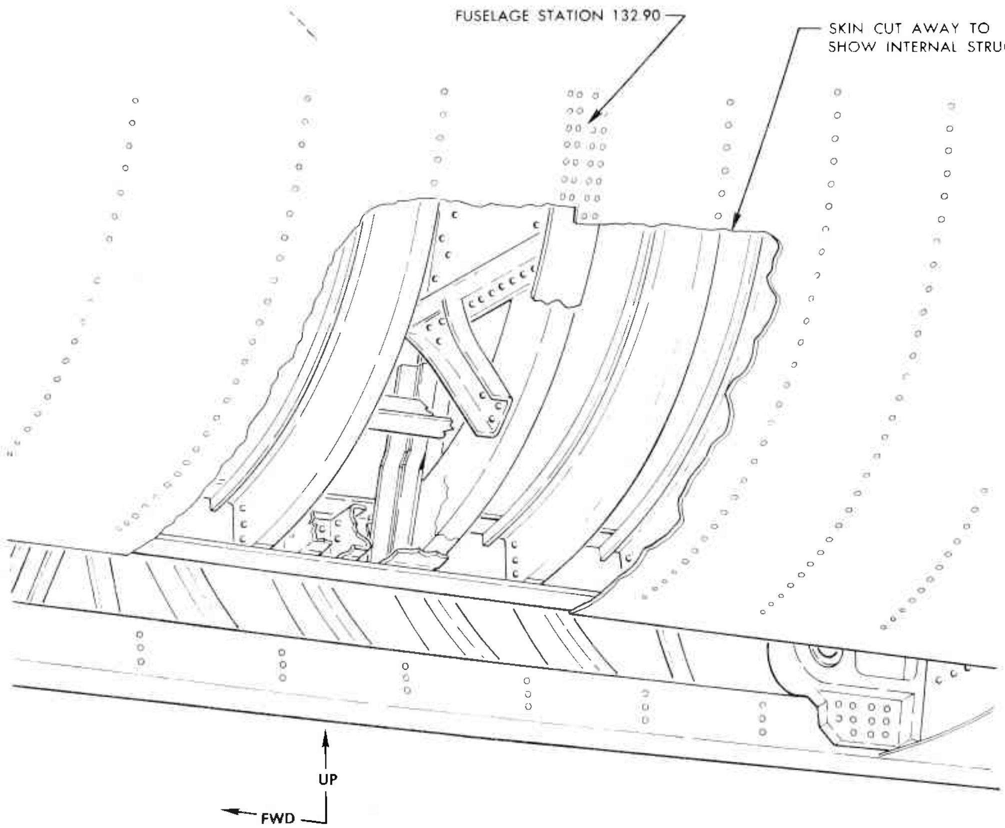
FRAME REPAIR PROCEDURE

a. Place airplane on support fixtures to allow access to damaged area. See figure "Airplane Handling Equipment" in Section 1.
b. Remove nose landing gear and nose landing gear door. Refer to T.O. 1F-106A-2-8-2-1 for removal procedure.
c. Remove electrical equipment from nose wheel well. Refer to T.O. 1F-106A-2-10-2-1 for removal procedure.
d. Using a No. 30 (0.128) drill, remove a sufficient number of rivets from the outer skin to allow removal of the damaged frame.

NOTE

A DRILL STOP SHOULD BE USED WHEN REMOVING RIVETS FROM THE OUTER SKIN TO PREVENT DAMAGE TO UNDERLYING STRUCTURE.

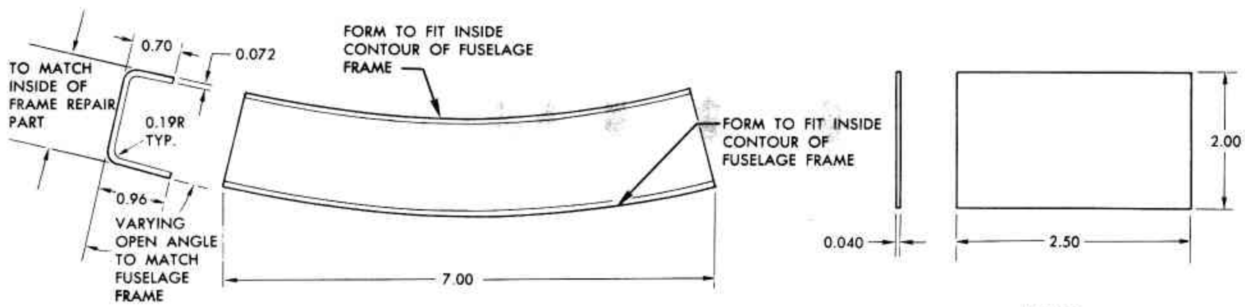
e. Using a No. 10 (0.1935) drill, remove rivets that attach clip to longeron.
f. Using a No. 10 (0.1935) drill, remove rivets that attach clip to frame and vertical stiffener channels.



VIEW LOOKING INBOARD
LEFT-HAND SIDE

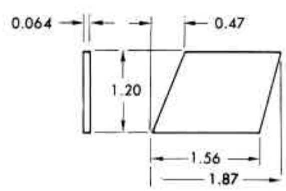
06.03.219-1 .53.04.03

Figure 9-12. Fuselage Frame Repair—Fuselage Station 132.90 (Sheet 1 of 4)

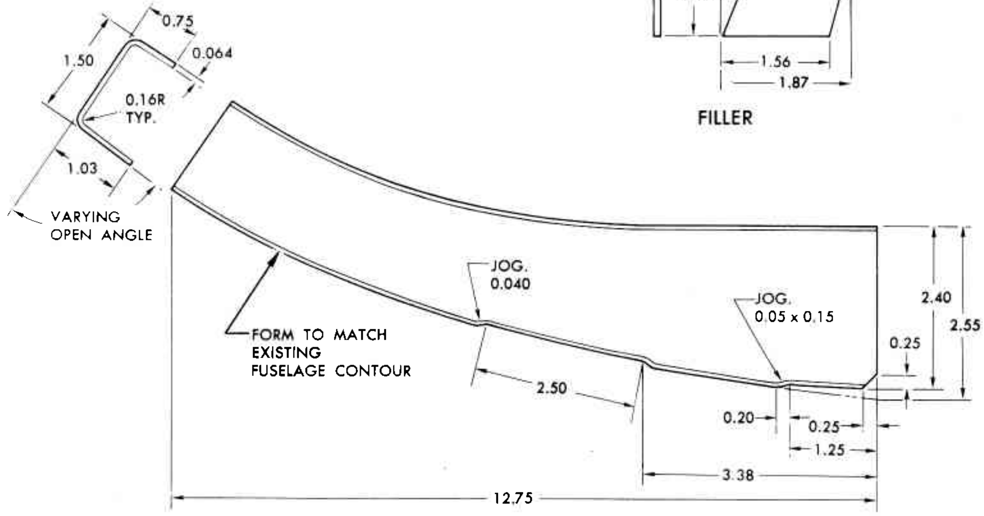


FUSELAGE FRAME CHANNEL SPLICE

FILLER



FILLER



FUSELAGE FRAME REPAIR PART

g. Using a No. 20 (0.161) drill, remove rivets that attach the diagonal stiffener channel to the frame. See figure "Modified Hi-Speed Drill for Titanium" in Section 1 for type of drill to be used when removing steel rivets.
h. Using a No. 30 (0.128) drill, remove rivets that attach the diagonal stiffener channel to the vertical stiffener channel. See figure "Modified Hi-Speed Drill for Titanium" in Section 1 for type of drill to be used when removing steel rivets.
i. Using a No. 10 (0.1935) drill, remove rivets which attach the vertical splice channel to the vertical stiffener channel.

NOTE

REMOVE ALL OLD SEALANT FROM AROUND DAMAGED AREA. USE ALIPHATIC NAPHTHA, FEDERAL SPECIFICATION TT-N-95A TO REMOVE SEALANTS.

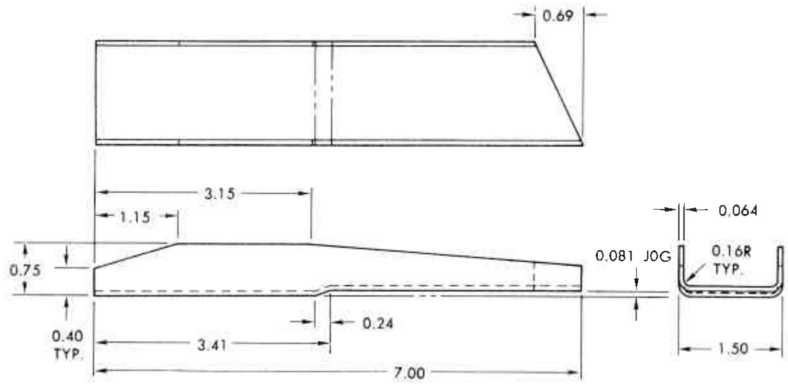
j. Remove diagonal stiffener and vertical splice channels.
k. Cutout damaged portion of fuselage frame.

CAUTION

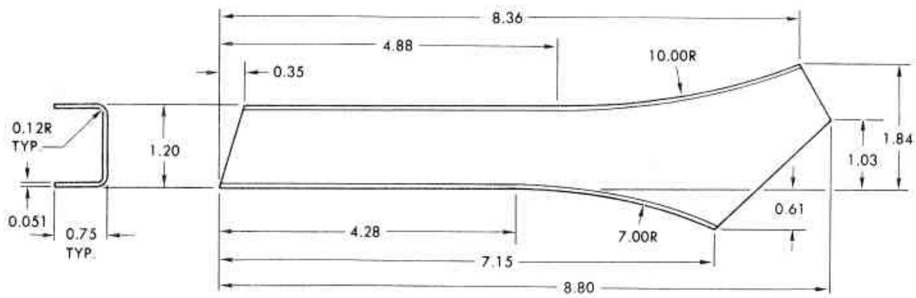
PLACE A PIECE OF SCRAP METAL BETWEEN FAYING SURFACES OF FRAME AND OUTER SKIN TO PREVENT DAMAGE TO THE SKIN WHILE CUTTING OUT DAMAGED PORTION OF FUSELAGE FRAME.

.06.03.219-2 .53.04.03

Figure 9-12. Fuselage Frame Repair — Fuselage Station 132.90 (Sheet 2 of 4)



VERTICAL SPLICE CHANNEL



DIAGONAL STIFFENER CHANNEL

l. Fabricate repair parts from 7178-0 bare sheet according to dimensions shown on sheets 2 and 3.

m. After forming, heat treat repair parts to -T6 condition per Specification MIL-H-6088.

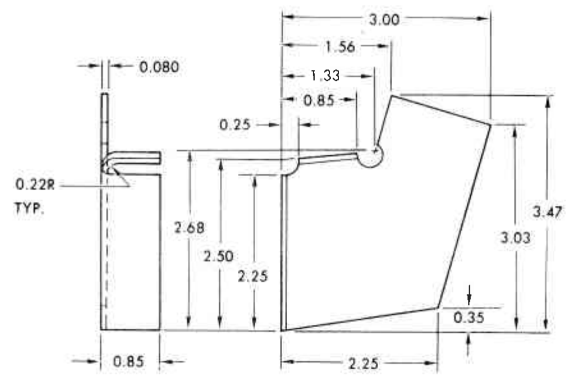
NOTE

APPLY A FILM OF MIL-S-8802 SEALANT TO FAYING SURFACES OF REPAIR PARTS PRIOR TO INSTALLATION.

n. Using existing holes in the outer skin, drill holes through new repair part. Use a No. 30 (0.128) drill.

NOTE

IF HOLES IN THE OUTER SKIN HAVE BEEN OVERSIZED DURING THE DRILLING OPERATIONS, IT IS PERMISSIBLE TO ENLARGE THE HOLE ONE RIVET SIZE.



CLIP

.06.03.219-3 .53.04.03

Figure 9-12. Fuselage Frame Repair — Fuselage Station 132.90 (Sheet 3 of 4)

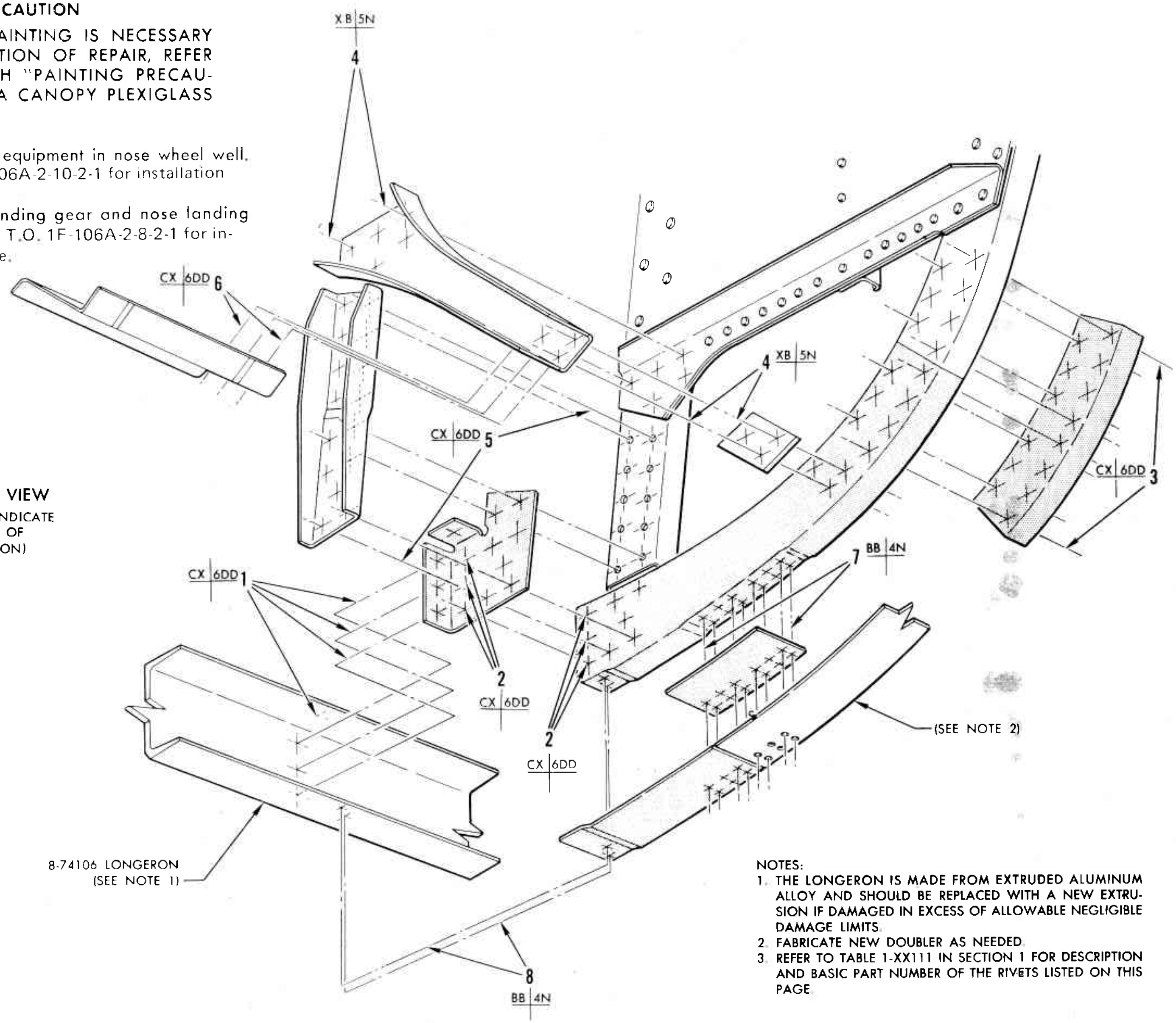
o. Rivet skin in place with AN426AD-4 rivets.

CAUTION

IF TOUCHUP PAINTING IS NECESSARY AFTER COMPLETION OF REPAIR, REFER TO PARAGRAPH "PAINTING PRECAUTIONS IN AREA CANOPY PLEXIGLASS WINDOWS."

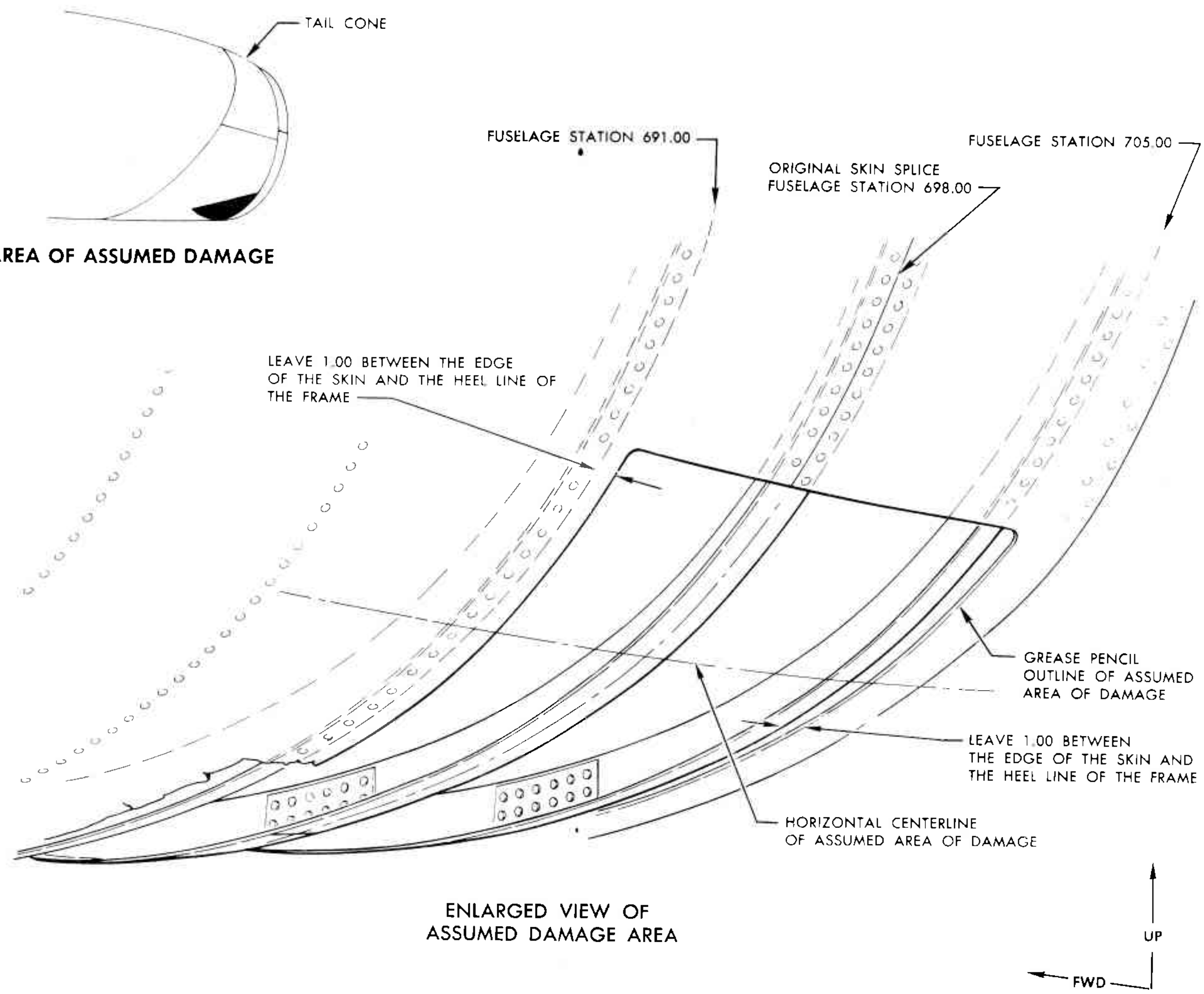
p. Re-install electrical equipment in nose wheel well. Refer to T.O. 1F-106A-2-10-2-1 for installation procedure.

q. Re-install nose landing gear and nose landing gear door. Refer to T.O. 1F-106A-2-8-2-1 for installation procedure.



NOTES:
1. THE LONGERON IS MADE FROM EXTRUDED ALUMINUM ALLOY AND SHOULD BE REPLACED WITH A NEW EXTRUSION IF DAMAGED IN EXCESS OF ALLOWABLE NEGLIGIBLE DAMAGE LIMITS.
2. FABRICATE NEW DOUBLER AS NEEDED.
3. REFER TO TABLE 1-XX111 IN SECTION 1 FOR DESCRIPTION AND BASIC PART NUMBER OF THE RIVETS LISTED ON THIS PAGE.

Figure 9-12. Fuselage Frame Repair — Fuselage Station 132.90 (Sheet 4 of 4)



TAIL CONE REMOVAL PROCEDURE

a. Place airplane on support fixtures to allow access to damaged area. See figure "Airplane Handling Equipment" in Section 1.

b. Remove tail cone and place on padded work bench or table. Refer to T.O. 1F-106A-2-4-2-1 for tail cone removal procedure.

DAMAGE REMOVAL PROCEDURE

a. Remove all paint and other foreign material from damaged area of tail cone, so that entire extent of damage may be detected.

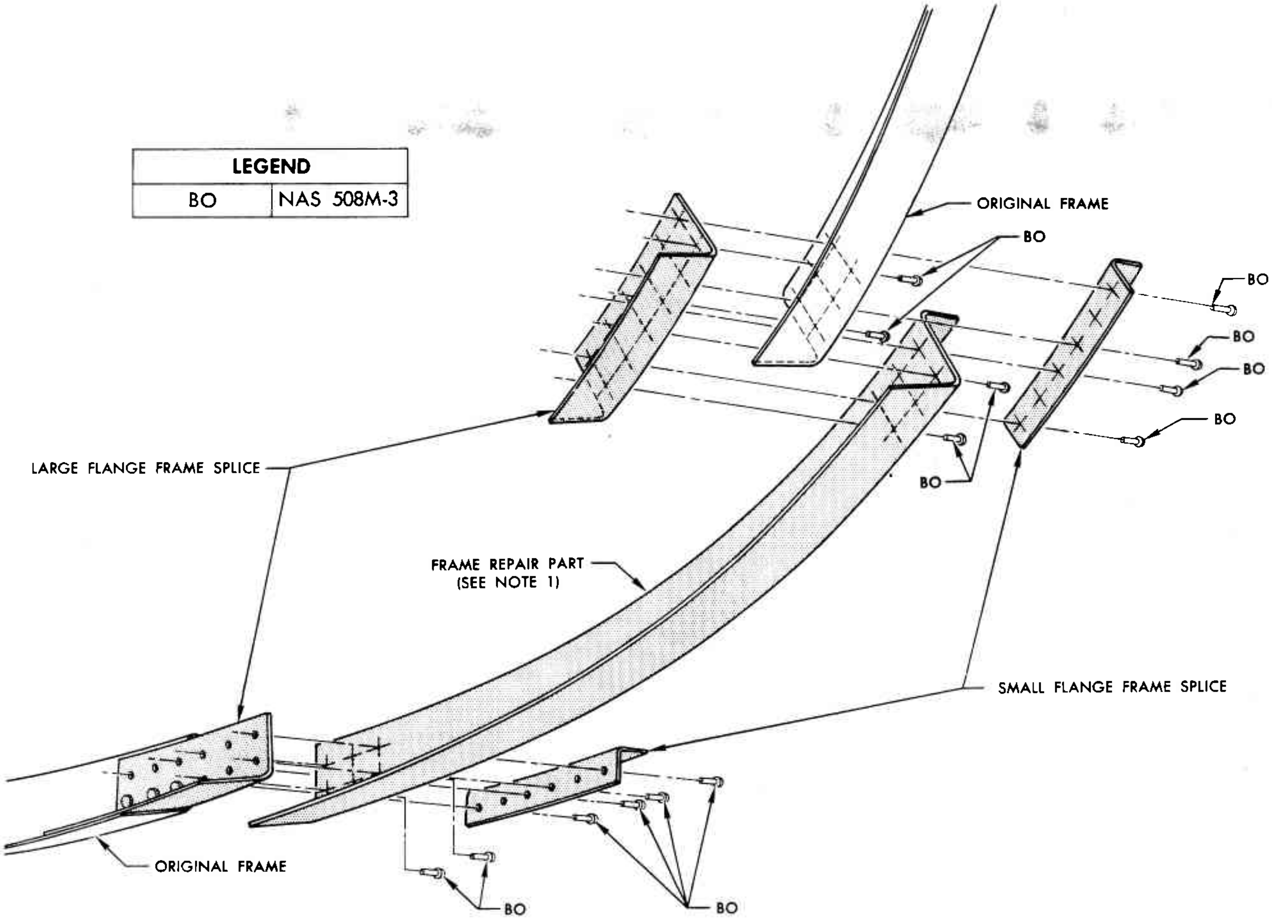
NOTE

EXTREME EDGES OF CRACKS MAY BE DETECTED WITH PENETRANTS. REFER TO PARAGRAPH "FLUORESCNET PENETRANT INSPECTION" IN SECTION 1 FOR THE DIFFERENT TYPES OF FLUORESCENT PENETRANT INSPECTION.

b. Using a straight edge and a grease pencil with "Listo Lead," outline damaged area of tail cone.

06.03.220-1 .53.15.03

Figure 9-13. Damaged Tail Cone Repair (Sheet 1 of 4)

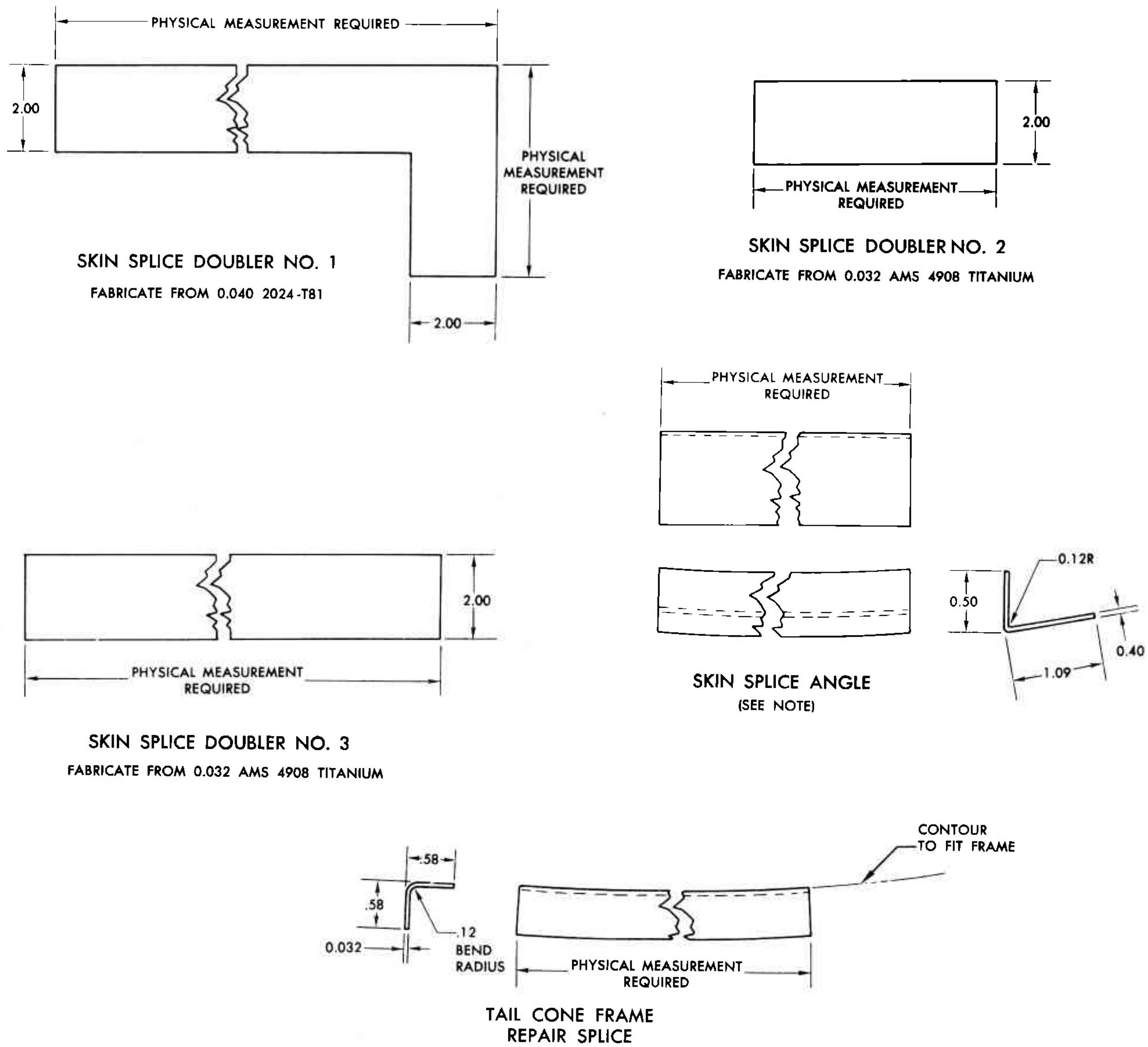


c. Using a No. 30 (0.128) drill, remove rivets from aluminum portion of tail cone skin that is within the outlined area.
d. Using a No. 39 (0.99) drill, remove rivets from titanium portion of tail cone skin that is within the outlined area.
e. Following the lines laid out in step B, cut away damaged portion of tail cone skin and frame. Metal cutting hand shears and hand files are most effective for cutting away damaged parts in the tail cone.
f. Using a No. 39 (0.099 drill, remove at least six rivets from tail cone frame and stiffener angles at each end of tail cone frame.
g. Cut away that portion of the tail cone frame stiffener angles from which rivets were removed in step F.

NOTE:
1. DIMENSIONS AND FLANGE ANGLE FOR FRAME REPAIR PART MAY BE OBTAINED FROM AN UNDAMAGED AIRPLANE OF THE SAME MODEL AND SERIES.
2. APPLY STA-BOND C-875 SEALANT TO FAYING SURFACES OF REPAIR PARTS PRIOR TO INSTALLATION.

.06.03.220-2 .53.15.03

Figure 9-13. Damaged Tail Cone Repair (Sheet 2 of 4)



FABRICATION OF DETAIL REPAIR PARTS

a. Fabricate a tail cone frame repair part from AMS4908 titanium alloy to dimensions shown on sheet 3. The linear dimension of the repair part must be determined by a physical measurement.

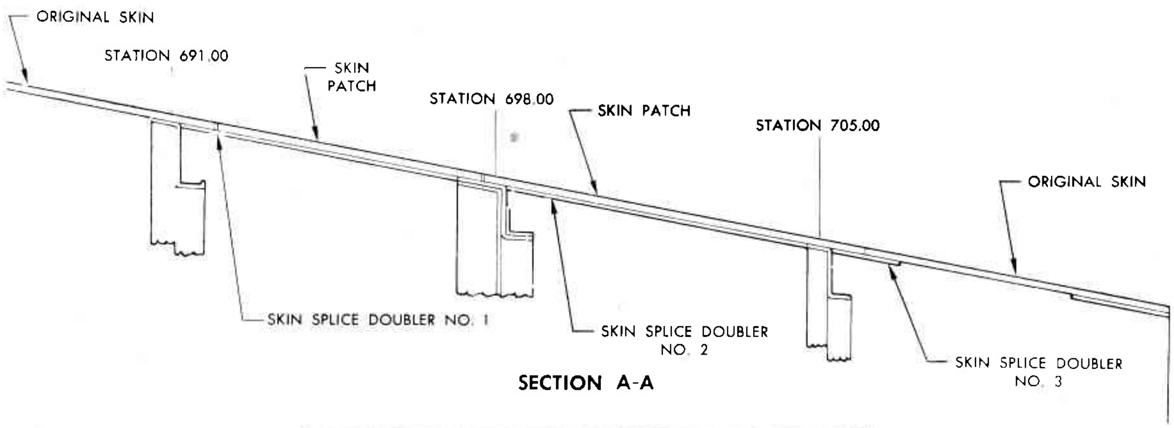
b. Fabricate tail cone frame stiffener angles from AMS4908 titanium alloy, according to the dimensions shown on sheet 3. The linear dimensions of the stiffener angles must be determined by a physical measurement.

c. Fabricate skin splice doublers according to dimensions shown on sheet 3. The linear dimensions must be determined by a physical measurement.

d. Fabricate skin patches. Skin patch forward of station 698.00 is to be made from 0.032-inch 2024-T81. Skin patch aft of station 698.00 is to be made from 0.025-inch AMS4908 titianium.

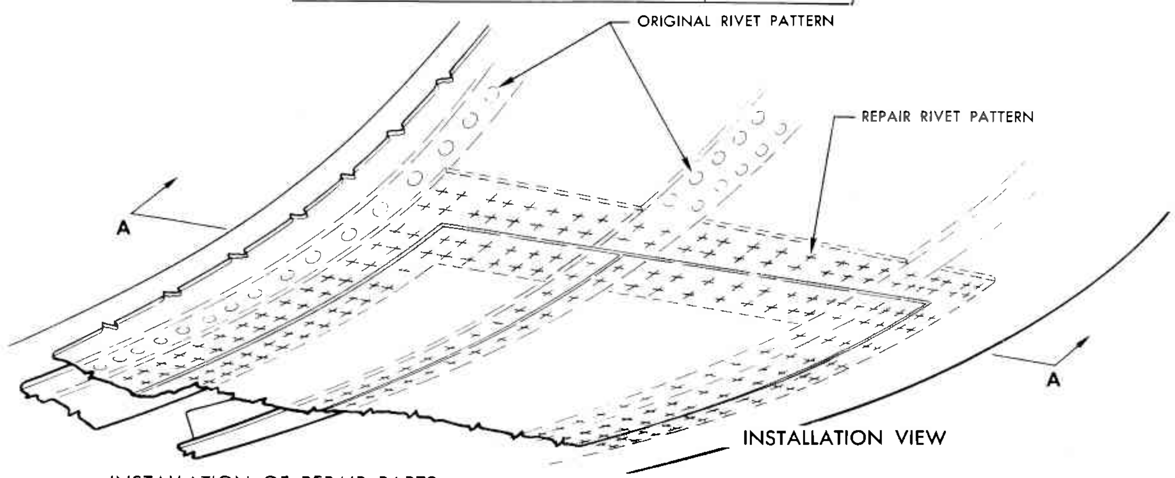
.06.03.220-3 .53.15.03

Figure 9-13. Damaged Tail Cone Repair (Sheet 3 of 4)



SECTION A-A

RIVET INSTALLATION CHART	
INTERNAL RIVETS FWD OF STA 698.00	AN470AD-4
INTERNAL RIVETS, STA 698.00 AND AFT	NAS508M-3
EXTERNAL RIVETS FWD OF STA 698.00	AN426AD-4
EXTERNAL RIVETS, STA 698.00 AND AFT	AN427M-3



INSTALLATION VIEW

INSTALLATION OF REPAIR PARTS

a. Attach wide-flanged stiffener angle to tail cone frame repair part as shown on sheet 2.

b. Insert frame repair part into tail cone and hold in place with clamps. Align repair part with original frame before drilling and riveting.

c. Install small-flanged stiffener angle and hold in place with clamps. See sheet 2 for hole size and rivet data.

d. Install skin splice doublers as shown on sheet 4.

NOTE

ALL HOLES FORWARD OF STATION 698.00 IN THE EXTERIOR SURFACE MUST BE MACHINE COUNTERSUNK 100°. ALL HOLES AFT OF STATION 698.00 IN THE EXTERIOR SURFACE MUST BE HEAT DIMPLED. REFER TO PARAGRAPH "FORMING CHARACTERISTICS OF TITANIUM" IN SECTION 1.

e. Install skin patches as shown on sheet 4.

NOTE

APPLY STA-BOND C-875 TO FAYING SURFACES OF TAIL CONE SKIN AND INTERNAL STRUCTURE PRIOR TO RIVETING.

f. Finish exterior surface of tail cone according to procedures outlined on figure "Primer and Paint Coatings" in Section 1.

TAIL CONE INSTALLATION PROCEDURE

a. Refer to T.O. 1F-106A-2-4-2-1 for installation procedure.

Figure 9-13. Damaged Tail Cone Repair (Sheet 4 of 4)

Section X

TYPICAL REPAIRS AND APPLICATION

10-1. TYPICAL REPAIRS.

10-2. This section covers repairs that are applicable to more than one specific component or section of the airplane structure. By application of the repair principles established in Section I, it is possible to design repairs for a large portion of the structure components of these airplanes. Damage requiring repairs exceeding those illustrated in this manual shall be designed by an authorized aeronautical structures engineer. Application of repairs that are designed by an authorized aeronautical structures engineer is authroized. In some instances, damages to structure will require a combination of two or more of the described repairs. A careful analysis of the damage will indicate which of the typical repairs may be applied.

10-3. APPLICATION OF TYPICAL REPAIRS.

10-4. Milled Skin Areas.

10-5. Conventional skin repairs are applicable to areas of milled skin where skin thickness is uniform and there are no integral ribs.

10-6. Engine Air Inlet Duct Areas.

10-7. Conventional skin repairs may be used in the repair of the engine air intake duct areas, provided damage is investigated carefully. A failure in the riveted areas of the duct could cause loosened rivets or pieces of metal to be drawn into the engine, resulting in more serious damage. Only flush-type repairs may be used on the inside of the duct. See figures 4-18A, 4-18B, 10-1 and 10-4 for typical flush repairs. Drag angle repairs may be accomplished by application of non-flush repairs in accordance with figure 10-4 when approved by an aeronautical structures engineer. In cases when it is impossible to use solid rivets, the following substitutions may be made:

SOLID	BLIND MECHANICAL LOCK
AD rivets	NAS 1399B SERIES
DD rivets	NAS 1739B SERIES
Steel screws and rivets	Blind bolt hi-shear part No. BB351

The above blind fasteners are authorized to fill the empty holes of missing rivets in the intake duct. Proper hole size and length of fasteners is of utmost importance in all blind fastener installations.

a. It is permissible to mechanically remove corrosion pits from air intake ducts. Material removed shall be ground out to form a saucer shaped depression within localized areas of one inch while maintaining three inches between pit area centers. Corrosion pits exceeding 0.025 depth limit shall be repaired as per Section IV, figure 4-18A and 4-18B. Treat and refinish all areas in accordance with T.O. 1-1-1 and T.O. 1-1-2, and this manual as applicable.

10-8. SHEET METAL SKIN AND WEB REPAIRS.

10-9. Tension-Shear Type Repair.

10-10. All skin repairs in this manual are designed to handle combination tension and shear type loads. These repairs are illustrated in principle only, and will require reference to Table 1-X. See figures 10-1 through 10-3 for applicable skin and web repair illustrations.

10-11. SHEET METAL STIFFENER REPAIRS.

10-12. Angle Repair.

10-13. Figure 10-4 shows a typical roll-formed flanged angle splice repair. Instructions for repair are given on the illustration. In all cases the splice angle must be of the same type material as the original flanged angle, but of the next heavier gage.

10-14. Channel Repair.

10-15. Figure 10-5 shows a typical roll-formed channel splice repair and figure 10-6 shows a typical channel rib repair. Refer to the instructions on the illustrations for repair procedures and the material gage to be used. In all cases the channel splice will be of the same material as the original roll-formed channel, but of the next heavier gage.

10-16. Zee Repair.

10-17. Figure 10-7 shows a typical roll-formed flanged zee splice repair. Refer to instructions on the illustration for repair procedures. An alternate repair is given when both flanges of the zee are attached to structure.

10-18. "U" Shape Repair.

10-19. Figure 10-8 shows a typical and an alternate roll-formed flanged channel splice repair. In all cases the splice angles will be fabricated from the same type of material as the original channels, but of the next higher gage.

10-20. EXTRUDED STIFFENER REPAIRS.

10-21. Extrusions and Sheet Metal Equivalents.

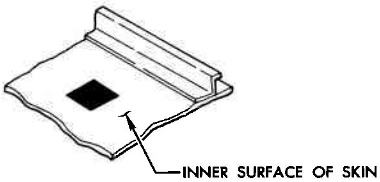
10-22. Extrusions are listed and illustrated in Section VIII. Material substitutes are listed in Table 1-I.

10-23. Extruded Tee Repair.

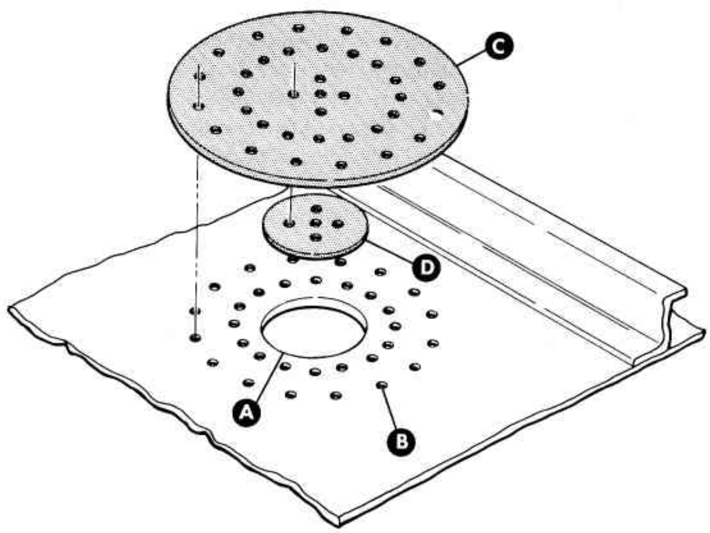
10-24. Figure 10-9 shows a typical extruded tee splice repair. Refer to the instructions on the illustration for repair procedures. In all cases, the splice angles will be of the same type material as the original extruded tee, but of the next heavier gage.

10-25. Extruded "I" Repair.

10-26. Figure 10-10 shows a typical extruded "I" splice repair. Refer to the instructions on the illustration for

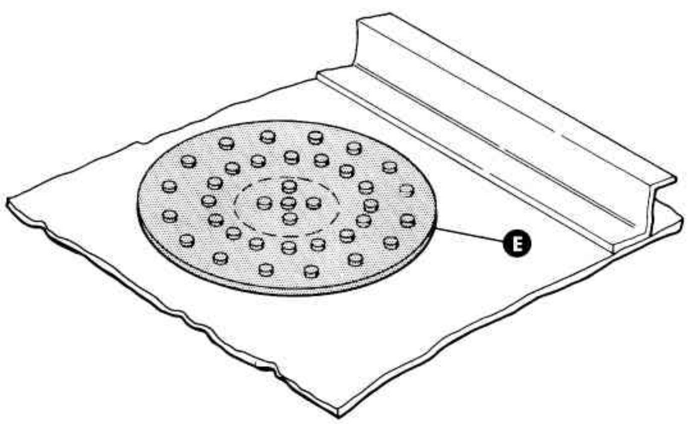


AREA OF
ASSUMED DAMAGE



REPAIR
(EXPLODED VIEW)

- **A** Clean up damaged area to regular shape. Round all corners and break all sharp edges.
- **B** Refer to illustration titled "Formula and Method for Patch Repair" in Section I for type, size, and number of rivets required. Layout and drill rivet pattern as indicated. Countersink holes 100° far side.
- **C** Fabricate doubler of next heavier gage and same type material as skin.
- **D** Fabricate flush plate of same gage and type material as skin.
- **E** If required, coat patch and doubler with sealer. Refer to applicable section for airplane component being repaired for fuel-tight or pressure sealing requirements. Install as shown.



ASSEMBLED REPAIR

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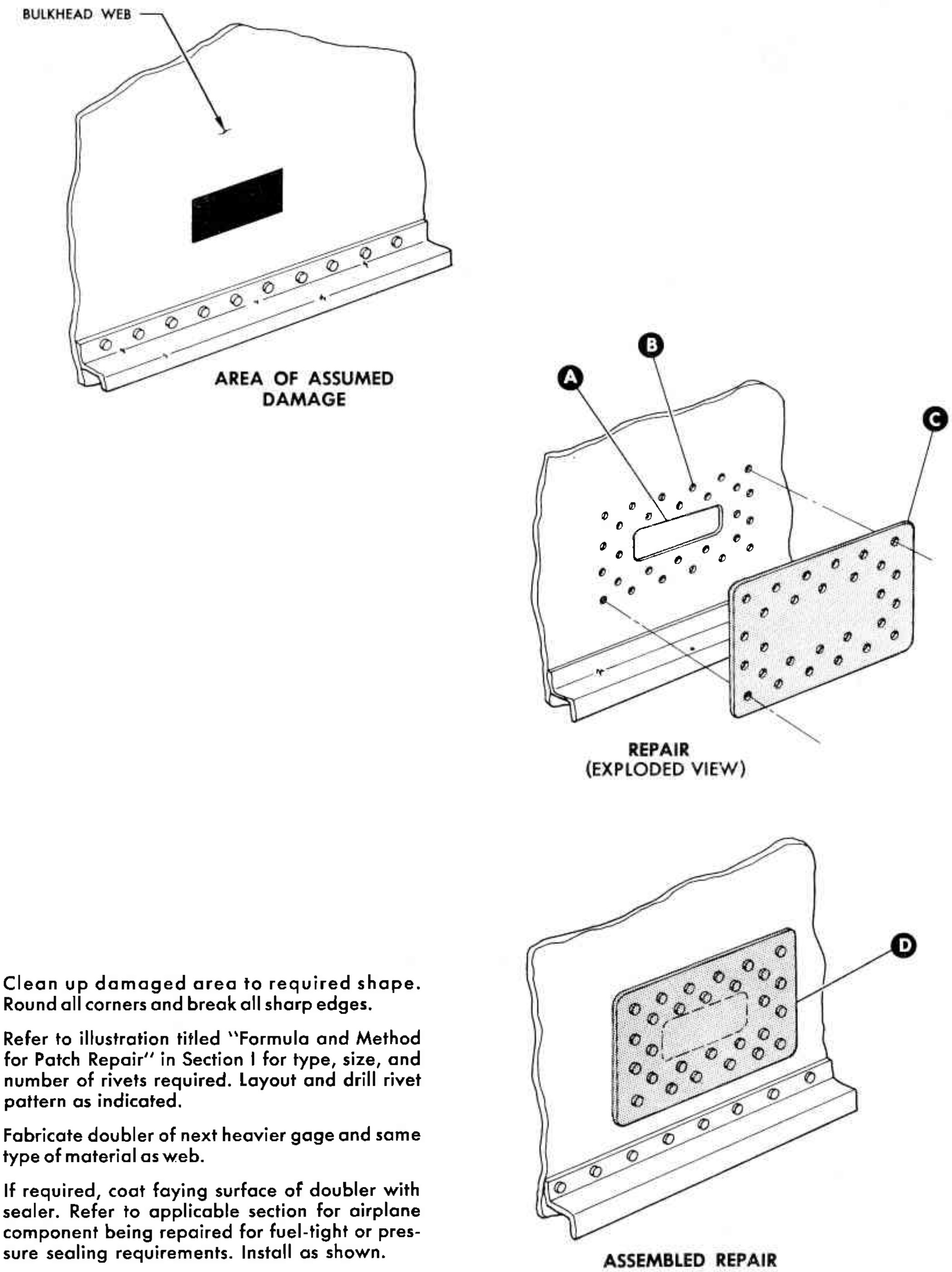
Figure 10-1. Flush Patch — Skin Repair



A Clean up damaged area to required shape. Round all corners and break all sharp edges.

B Refer to illustration titled "Formula and Method for Patch Repair" in Section I for type, size, and number of rivets required. Layout and drill rivet pattern as indicated.

C Fabricate doubler of next heavier gage and same type of material as web.

D If required, coat faying surface of doubler with sealer. Refer to applicable section for airplane component being repaired for fuel-tight or pressure sealing requirements. Install as shown.

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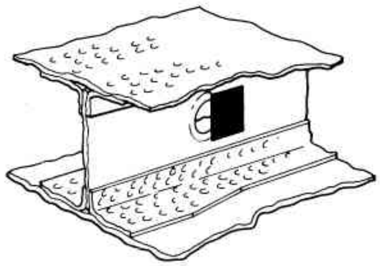
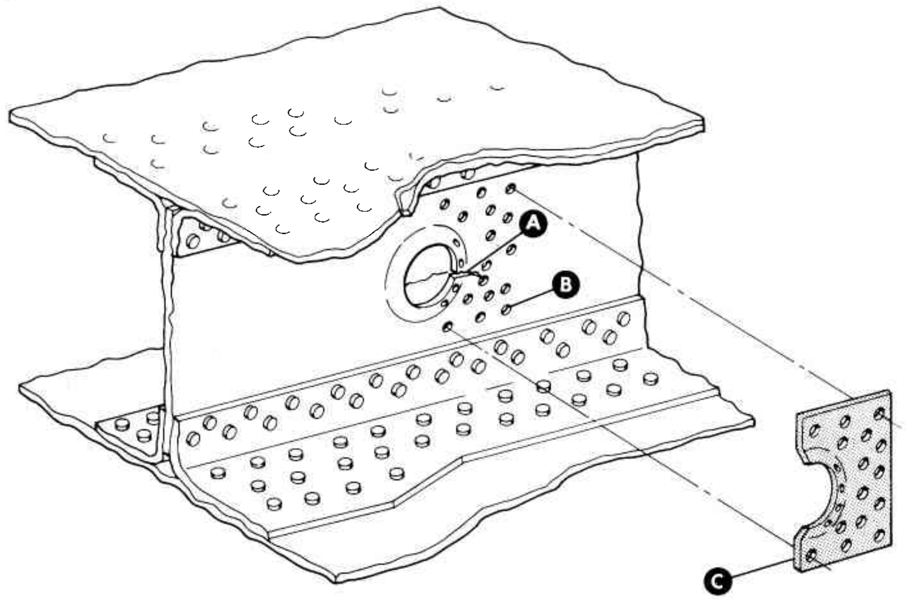
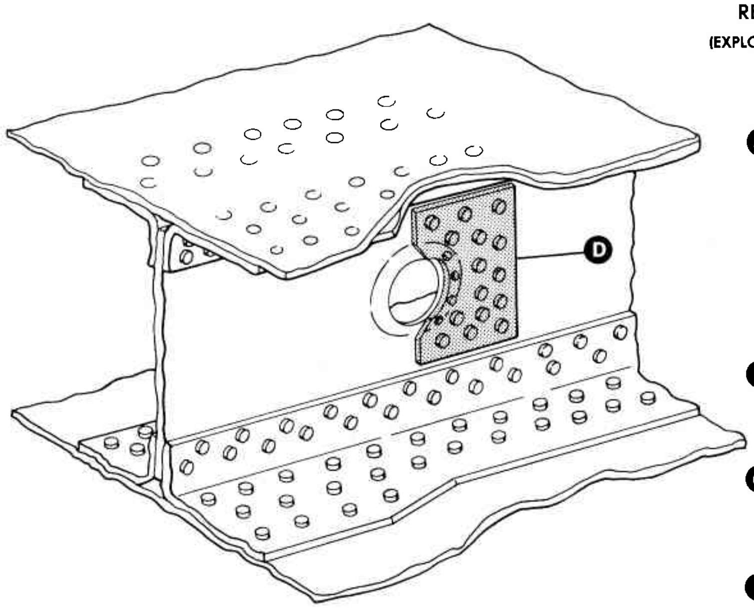
Figure 10-2. Doubler Patch — Bulkhead or Rib Web Repair



AREA OF
ASSUMED DAMAGE



REPAIR
(EXPLODED VIEW)



A Clean up damaged area to required shape. Round all corners and break all sharp edges. Hold 0.12 inch minimum radii.

NOTE

When damaged area exceeds more than 30 degrees of the hole, the repair shall be made with a circular doubler, attached around hole with 2 rows of rivets.

B Refer to illustration titled "Formula and Method for Patch Repair" in Section I for type, size, and number of rivets required. Layout and drill rivet pattern as indicated.

C Fabricate doubler of next heavier gage and same type material as web. Form to fit flange, provided 2 D rivet edge distance can be maintained, otherwise omit flange.

D Install doubler as shown.

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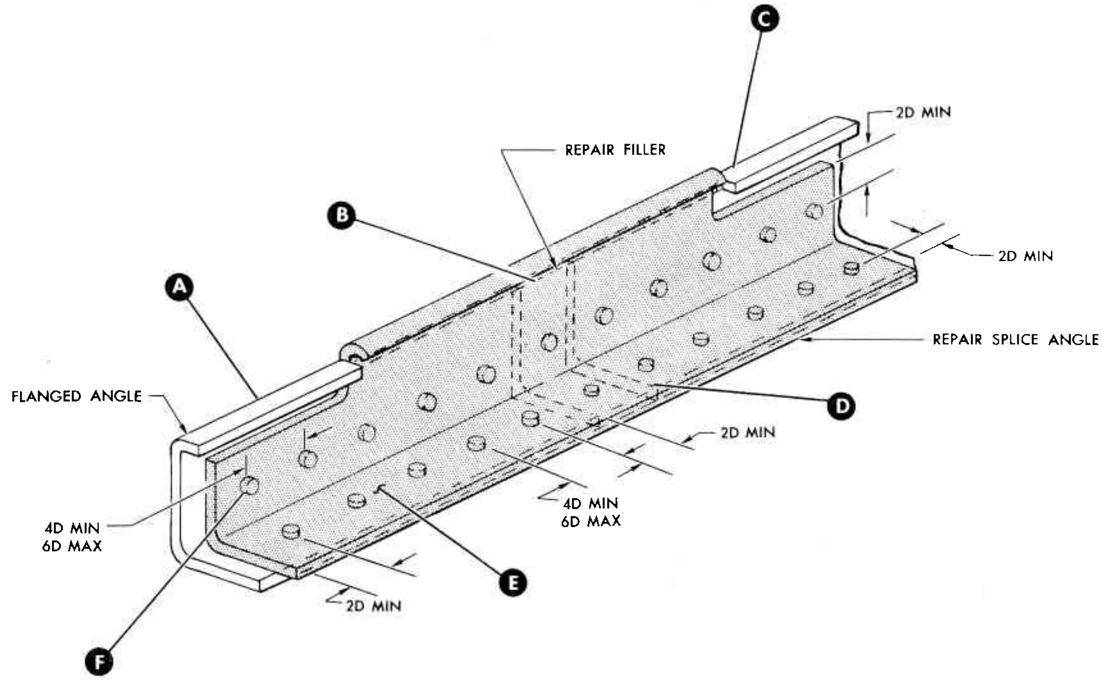
Figure 10-3. Cracked Lightening Hole Flange ***Repair***



A Identify flanged angle from standard roll-formed sections shown in Section VIII.

B Clean up damaged area to regular shape. Round all corners and break all sharp edges.

C Do not trim flange cut-out beyond absolute minimum required to avoid interference.

D Fabricate filler of the same type material and gage as the original angle.

E Fabricate splice angle of same type material and flange widths as original angle, but of the next heavier gage.

F Locate rivet pattern on each side of splice as shown to provide an equal number of rivets on each side of the flange cut-out. Continue rivet pattern across filler as shown. Refer to Section I for method of determining rivet pattern.

G Install filler and splice angles.

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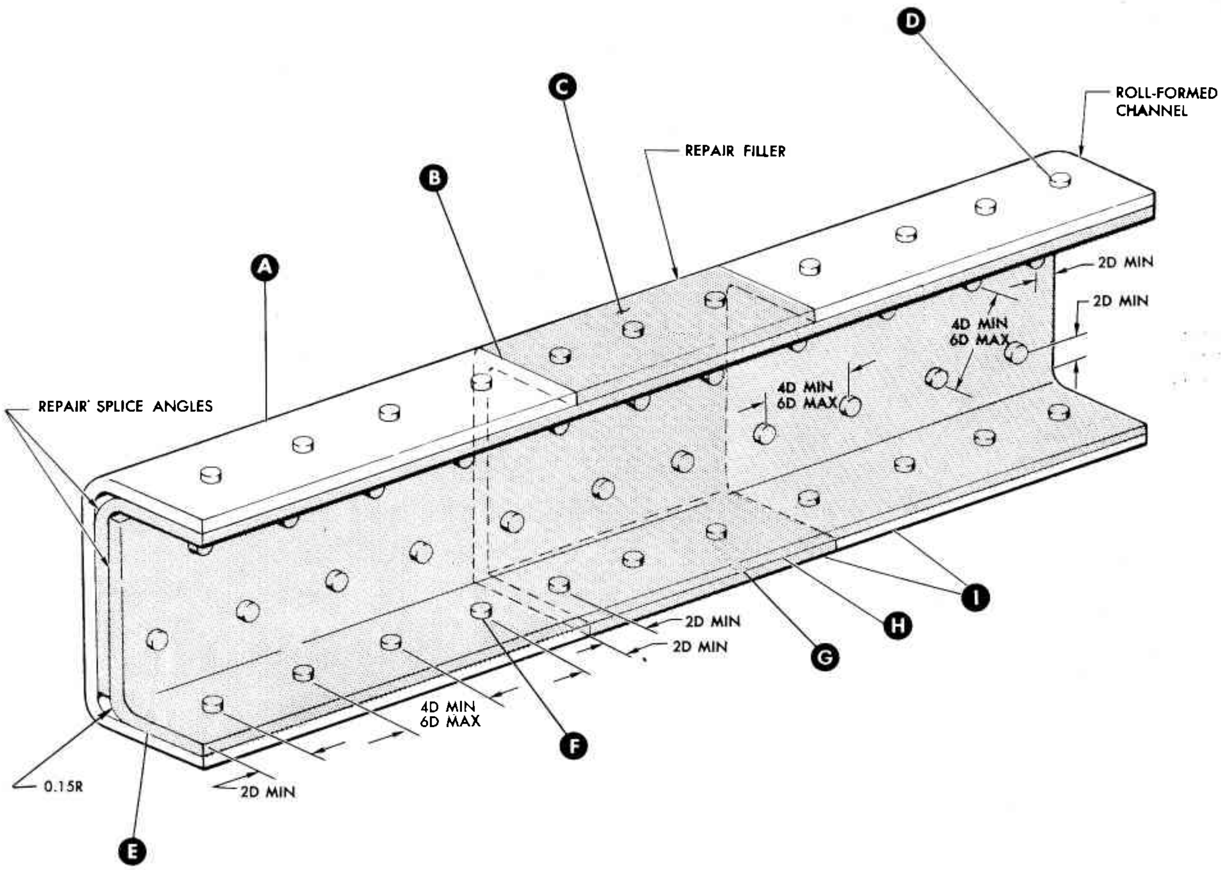
Figure 10-4. Roll-Formed Flanged Angle Splice Repair



A Identify channel from standard roll-formed sections shown in Section VIII.

B Cut out damaged area and smooth rough edges of channel.

C Fabricate filler of the same gage and material as original channel for use at former faying surface of channel and attaching structure. Filler may be omitted if channel is not attached to structure at damaged area.

D Refer to illustration "Formula and Method for Insertion Repair" in Section I for type, size, and number of rivets required. Layout rivet pattern with pencil marks on web and flanges of damaged channel.

E Fabricate splice angles of next heavier gage and same material as damaged channel. Determine length of angles by measuring 2D beyond each end of rivet pattern marked on channel.

F Clamp filler and angles into channel and drill rivet holes through channel, filler, and angles as indicated by pencil marks.

G Remove filler and angles. Clean out metal chips and remove burrs from edges of rivet holes. Refer to illustration "Primer and Paint Coatings" in Section I for applicable requirements and finish repair parts and bare edges of channel accordingly.

H If required, coat faying surfaces of repair parts with sealer. Refer to applicable airplane component section for fuel-tight or pressure sealing requirements. Install as shown:

I Install filler and angles as shown.

NOTES

1. WHEN WIDTH OF CHANNEL WEB EXCEEDS 4 RIVET DIAMETERS BETWEEN FLANGE RADII, THE RIVET PATTERN SHALL BE STAGGERED, BUT MINIMUM OF 2 RIVET DIAMETERS EDGE DISTANCE MUST BE MAINTAINED.
2. USE EQUAL NUMBER OF RIVETS ON EACH SIDE OF SPLICE AND CONTINUE RIVET PATTERN ACROSS FILLER AS SHOWN.

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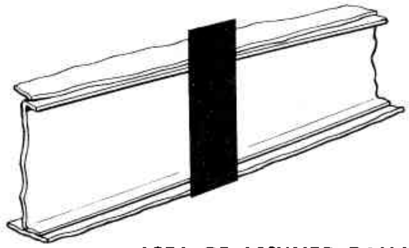
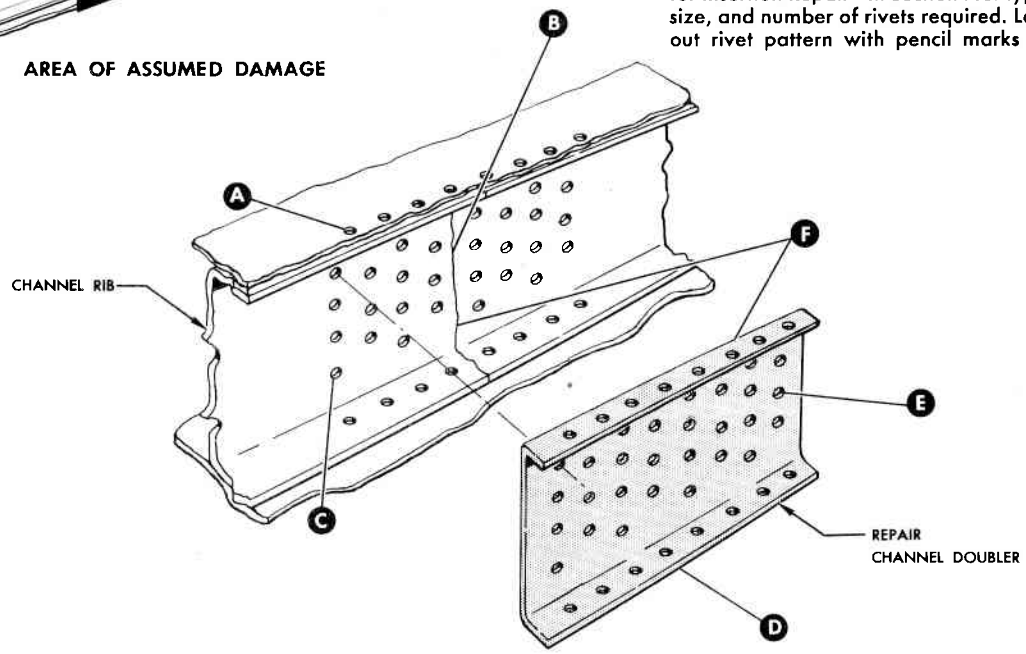
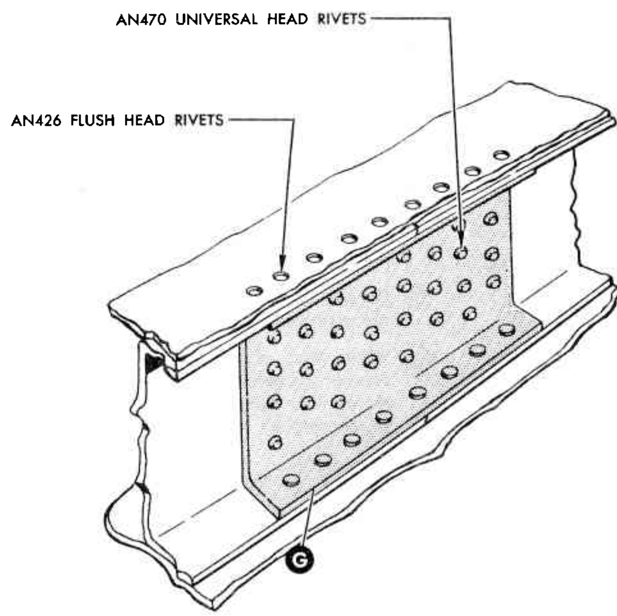
Figure 10-5. Roll-Formed Channel Splice Repair







A Remove existing skin rivets in both flanges.

B Smooth roughness and break sharp corners on edges of crack. Stop drill ends of partial cracks.

C Refer to illustration "Formula and Method for Insertion Repair" in Section I for type, size, and number of rivets required. Layout rivet pattern with pencil marks on web and flanges of the damaged channel.

D Fabricate channel doubler of next heavier gage and same type material as damaged channel. Determine length of doubler by measuring 2D beyond each end of rivet pattern marked on channel. Form channel doubler with material in the "O" or in the annealed condition to bend radii listed in Section I. Heat treat to final hardness after forming.

E Clamp doubler into channel and drill rivet holes through channel and doubler as indicated by pencil marks.

F Remove double, clean out metal chips and remove burrs from edges of rivet holes. Refer to illustration "Primer and Paint Coatings" in Section I for applicable requirements and finish doubler and bare edges of channel accordingly.

G If required, coat faying surface of doubler with sealer. Refer to applicable section for airplane component being repaired and for fuel-tight or pressure sealing requirements. Install as shown.

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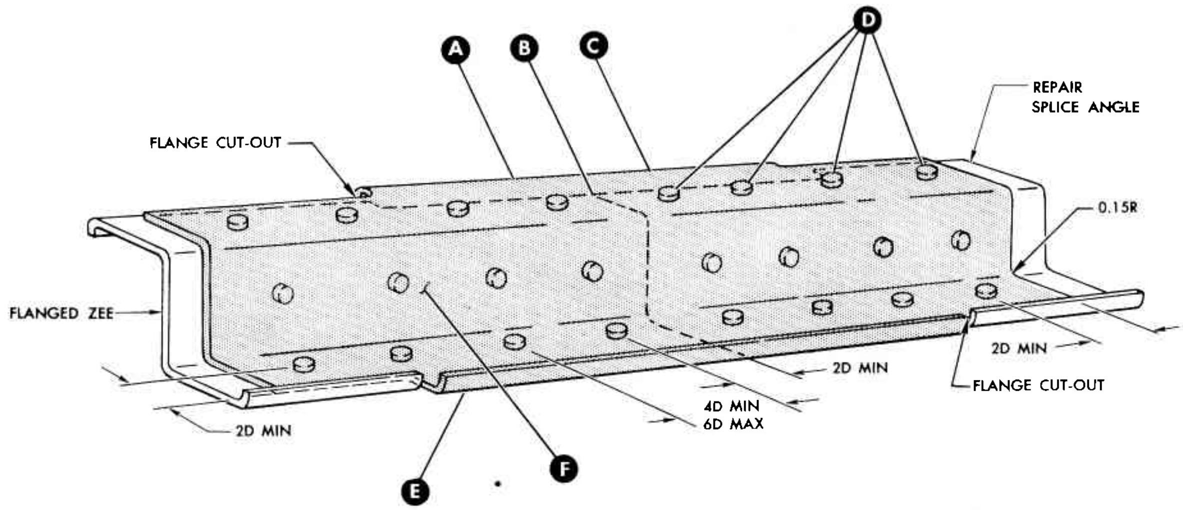
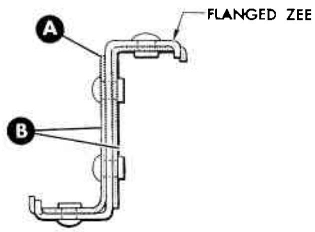
Figure 10-6. Channel Rib Repair





ALTERNATE REPAIR

A When both flanges of zee are attached cut off one flange of each splice angle and make splice repair as shown.

B Fabricate splice angles of same type material as original zee but next heavier gage. Layout rivet pattern as specified for single splice angle repair.

A Identify vertical flanged zee from standard roll-formed sections shown in Section VIII.

B Remove rough edges from butt ends.

C Fabricate splice angle of same type material and flange lengths as original zee, but next heavier gage.

D Locate rivet pattern on each side of the splice to provide an equal number of rivets on each side of the flange cut-out. Refer to Section I for standard minimum rivet spacing and method of determining rivet pattern.

E Do not trim flange cut-out beyond absolute minimum required to avoid interference.

F When web of zee exceeds 4 rivet diameters between flange radii, the rivet pattern shall be staggered, but minimum of 2 rivet diameters edge distance must be maintained.

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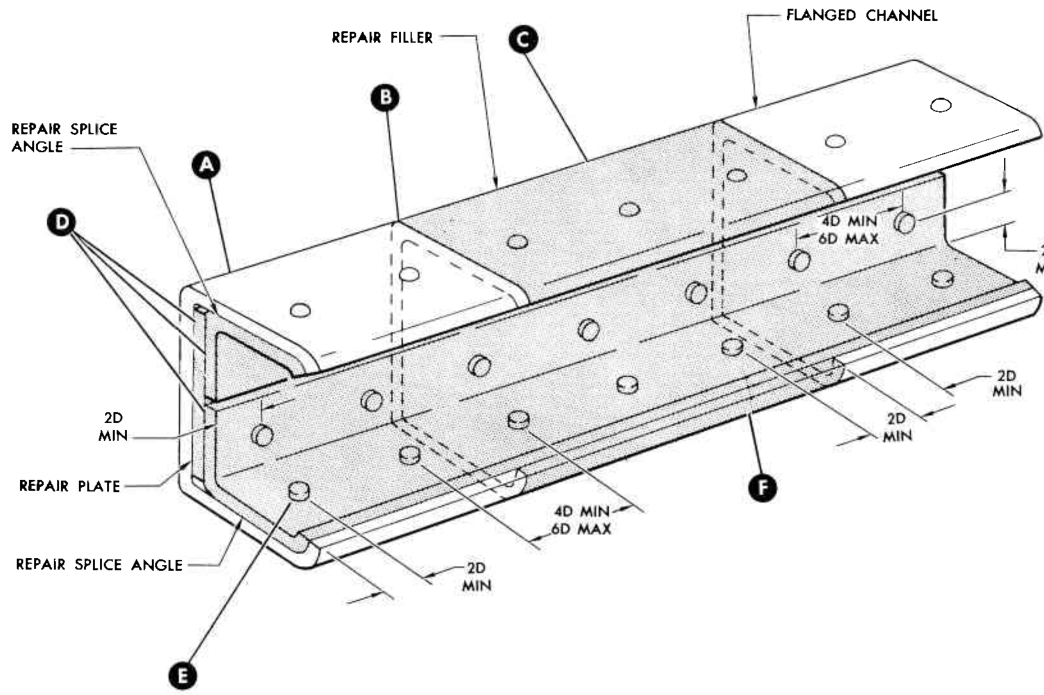
Figure 10-7. Roll-Formed Flanged Zee Splice Repair



A Identify flanged channel, from standard roll-formed sections shown in Section VIII.

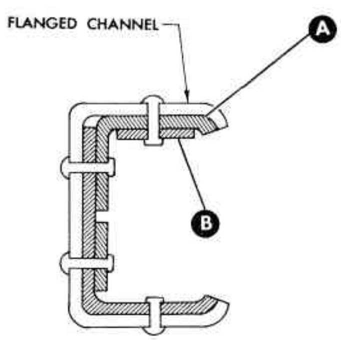
B Remove rough edges from butt ends of channel.

C Fabricate filler of same type material and gage as damaged channel.

D Fabricate splice angles and plate from same type material as original channel but next heavier gage.

E Locate and drill rivet pattern. Refer to Section I for method of determining rivet pattern.

F Install filler angles, and plate as shown.



ALTERNATE REPAIR METHOD.

A Fabricate splice angles from same type material but next heavier gage as original channel.

B Install radius block when necessary to prevent rivets from riding radius.

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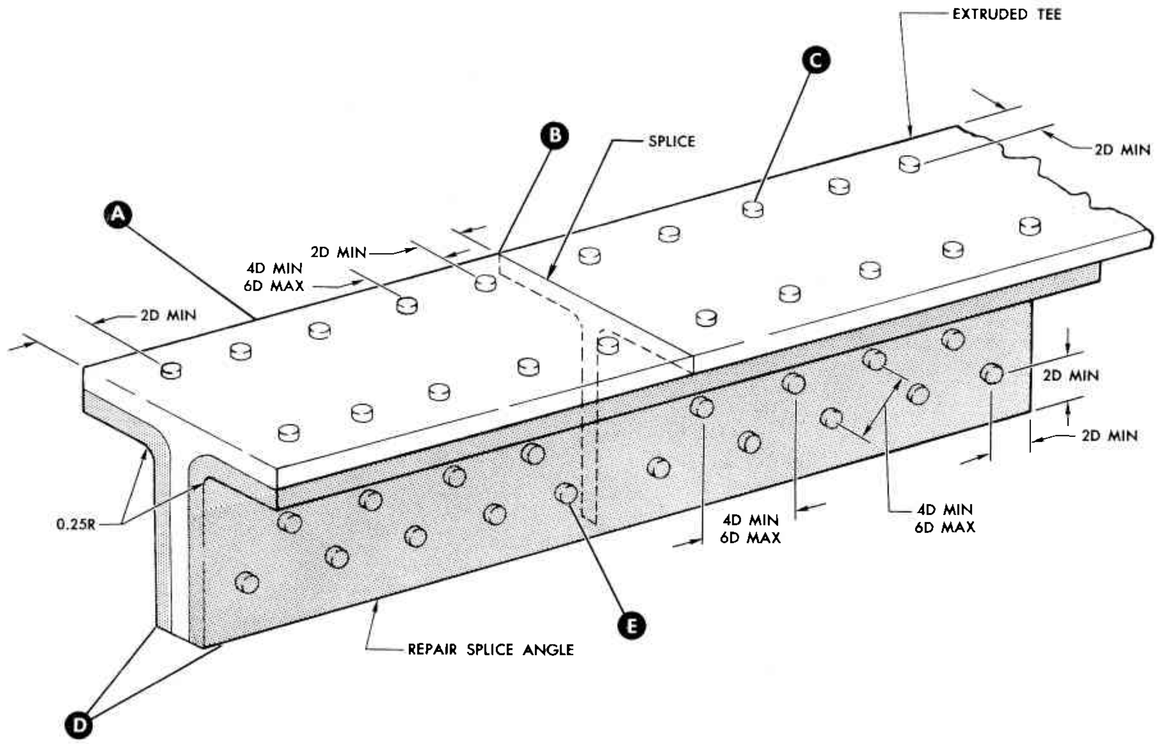
Figure 10-8. Roll-Formed Flanged Channel Splice Repair



A Identify extruded tee from extrusions shown in Section VIII.

B Remove rough edges from butt ends of tee.

C Refer to Section I for standard minimum rivet spacing and formula and method of determining rivet pattern.

D Fabricate and install roll-formed splice angles of the same type material as original tee but next heavier gage.

E When width of vertical leg of tee exceeds 4 rivet diameters, the rivet pattern shall be staggered, but minimum of 2 rivet diameters edge distance must be maintained.

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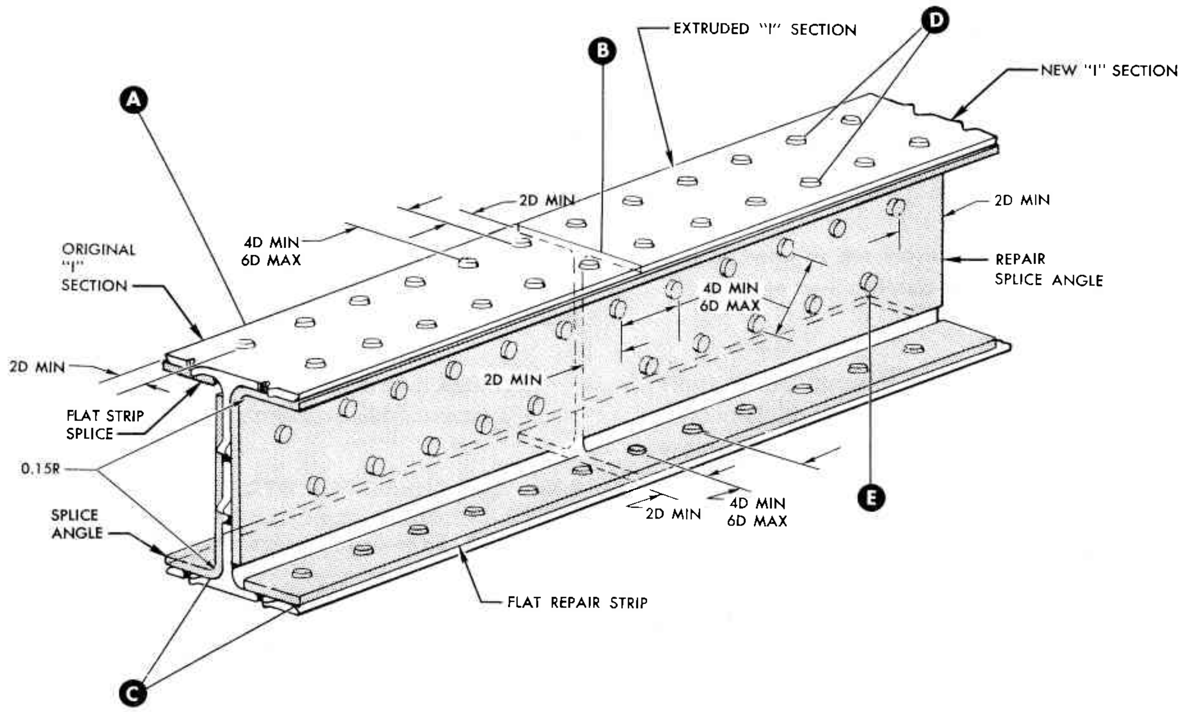
Figure 10-9. Extruded Tee Splice Repair



A Identify extruded "I" Section from extrusions shown in Section VIII.

B Remove rough edges from butt end of "I" Section.

C Fabricate and install roll-formed splice angles and flat strips of the same type material but next heavier gage.

D Use equal number of rivets on each side of splice. Refer to Section I for standard minimum rivet spacing and formula and method of repair.

E When web of "I" Section exceeds rivet diameters between flange radii, the rivet pattern shall be staggered, but minimum of 2 rivet diameters edge distance must be maintained.

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Figure 10-10. Extruded "I" Splice Repair

repair procedures. In all cases the splice angles will be of the same type material as the original "I" extrusions, but of the next heavier gage.

10-27. ALUMINUM HONEYCOMB REPAIRS.

10-28. Restoring structural strength in honeycomb panels with epoxy resins and other adhesives is an exacting process that involves specific cleaning, mixing, and handling problems. Resins and adhesives used are caustic and toxic. Personnel making repairs must be provided with the proper protective clothing and respirators. Caution must be exercised to observe the warnings displayed on the labels of adhesive containers.

NOTE

Most adhesives and accelerators are shipped in kit form under one stock number but in separate containers.

The instructions on figures 10-30 through 10-46 apply to honeycomb panels on all F-106 airplanes. However, in some instances a particular location is specified in the repair instructions. Where special tools are required, instructions for their manufacture are included. The repairs on the illustrations are arranged by number, category (dent, hole, etc.), and diameter. Before repair of any damage is attempted, the most applicable repair must be selected.

10-29. Repair Selection.

10-30. Figure 10-11 illustrates the fabrication and use of a damage locating template. This template is used to locate the center and area of the damage. However, any method, such as scribing lines across the damage area with a straight edge, may be used as long as the repair diameter can be determined.

10-31. Preparation of Damage Area.

10-32. To obtain a successful bond with adhesives, the damaged area must be properly cleaned. None of the various adhesives used in honeycomb repairs will adhere to oily or damp surfaces; even the small amount of moisture transferred from the hands when handling repair parts will seriously impair bonding strength. For this reason, clean white gloves must be worn when handling repair parts. The cleaning method illustrated in figure 10-13 is a complete step-by-step procedure. The steps required are listed in each individual repair procedure.

10-33. Repair Patch Plates and Rivet Patterns.

10-34. All patch plates used in F-106 honeycomb repairs are to be made of 0.025-inch aluminum alloy. The alloy and condition should conform to the original metal removed from the damaged area. Patch plate and rivet pattern data is shown in figure 10-12.

10-35. Repair Adhesives.

10-36. Table 10-I lists all the adhesives and accelerators required for repairs. Usually the first choice adhesive is shown. In some repairs, however, an alternate adhesive is listed. Table 10-I also gives the required cure time and temperatures. The use of heat, if so designated in the table, will decrease cure time and increase bonding strength. The manufacturer's instructions should be followed when mixing adhesives.

10-37. SPECIAL TOOLS.

10-38. Table 10-II lists special tools necessary to accomplish repairs on honeycomb structure. Also, see figures 10-11 through 10-29. Some of the tools may be fabricated locally as indicated on the table. Refer to the applicable figure number indicated in Table 10-II for information concerning fabrication and instructions for using the tools.

NOTE

It is anticipated that the required special tools will be available by March 1961.

10-39. BULKHEAD FUS. STA. 556.75 TEMPORARY REPAIR.

Applicable to F-106A-57-2465 and subsequent; F-106B-57-2522 and subsequent.

a. Locate ends of crack using fluorescent penetrant per TO 33B-1-1. Crack will be located at top center line of bulkhead on either fwd or aft face.

CAUTION

Care must be taken not to drill into the No. 2 vertical stabilizer spar attached in pocket of bulkhead. Thickness of cracked web is approximately 0.230 inch.

b. Stop drill both ends of crack using 1/4 inch drill bit.

c. Use eddy current probe to inspect stop drilled area. If probe indicates crack extends beyond stop drill, use 1/4 inch router to chase out cracks.

NOTE

If crack penetrates into either radius of the attachment lug hole bosses more than 0.1 inch, contact San Antonio ALC/MMEAS, Autovon 945, ext 7541.

d. Apply one coat of MIL-C-5541 chemical coating and MIL-P-8585 zinc chromate to inside of stop drilled area and crack.

e. Fabricate 90 degree repair angle patch in accordance with Figure 10-10G.

NOTE

Prior to final assembly, trim angle to fit and smooth all edges. Treat with one coat of MIL-C-5541 and MIL-P-8585. After final assembly, apply one coat of MIL-P-8585 to entire area.

f. Install repair angle patch in accordance with Figure 10-10G.

NOTE

Treat holes with MIL-C-5541 and, prior to installation of fasteners, coat holes and fasteners with MIL-P-8585.

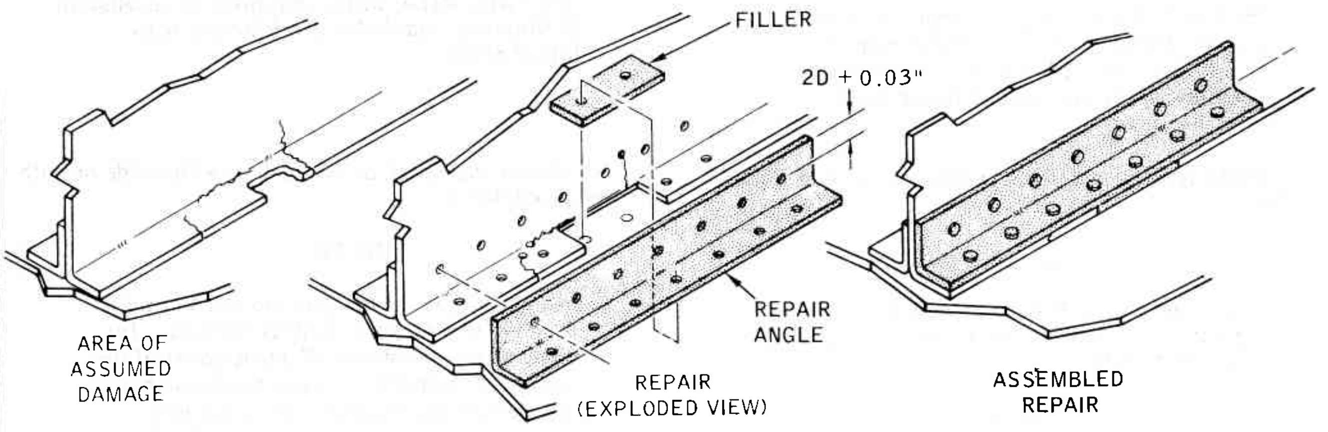
NOTE

Install bolts, Part No. AN4, with head down and with washer, Part No. AN960, under nut. Treat holes with MIL-C-5541, and, prior to installation of fasteners, coat holes and fasteners with MIL-P-8585.

g. Repair angle may be installed on either side or both sides of bulkhead.

NOTE

An entry must be made in aircraft Form 781 limiting this repair to 300 flying hours. Permanent repair consists of replacement of the bulkhead segment. Request serial number of all aircraft which receive this temporary repair be provided by message to San Antonio ALC/MMCTB.



REPAIR PROCEDURE

a. Smooth roughness and break sharp corners on edges of cracks; stop drill cracks. Trim out damaged area as required. Insert filler, in trimmed out area, of same gage material when required.

b. Fabricate repairs of same material and of next heavier gage as damaged area. Lower flange of repair to be same dimension as damaged flange. When damage extends into web leg, nested angle must extend past damage as shown in Section I, Formula and Method of Insertion Repair.

c. Locate and drill repair rivet pattern. Refer to Section I for method of determining rivet pattern.

NOTE

WHEN FLANGE IS LESS THAN ONE HALF INCH IN WIDTH, FLANGE RIVETS ARE NOT REQUIRED.

d. Refer to Section I for finish requirements of repair surfaces.

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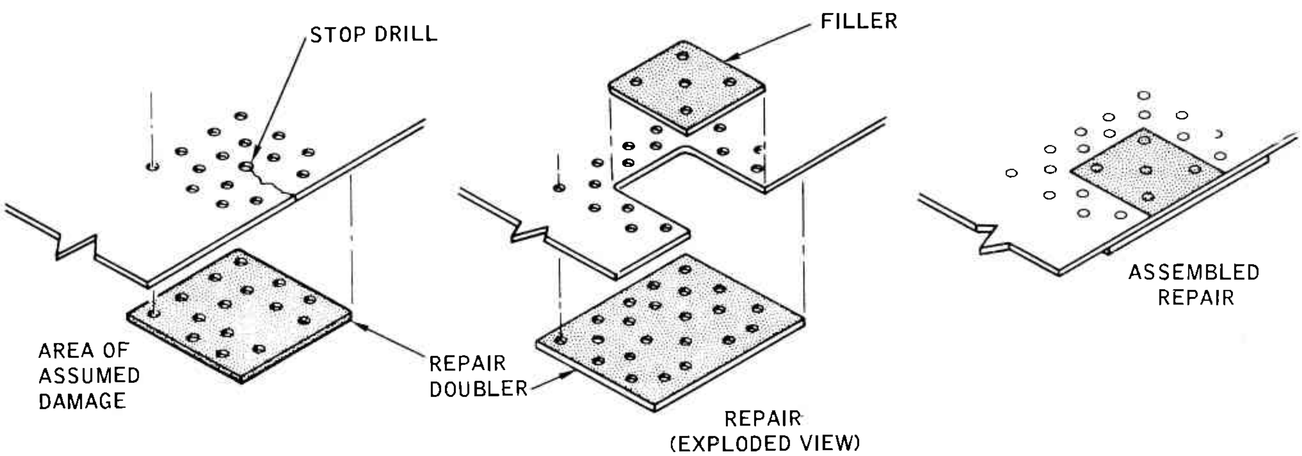
Figure 10-10A. Flange Repair



REPAIR PROCEDURE

a. Smooth roughness and break sharp corners on edges of cracks; stop drill cracks. Trim out area of damage if required. Insert filler in trimmed out area of same gage material as required.

b. Fabricate repair of same material and next heavier gage as damaged area. Repair must extend beyond damaged area same as Formula and Method of Insertion Repair in Section I.

c. Locate and drill rivet pattern. Refer to Section I for method of determining rivet pattern.

d. Install repair as shown.

e. If required, coat faying surfaces of doubler and filler with sealer. Refer to applicable section for airplane component being repaired for fuel-tight or pressure sealing requirements.

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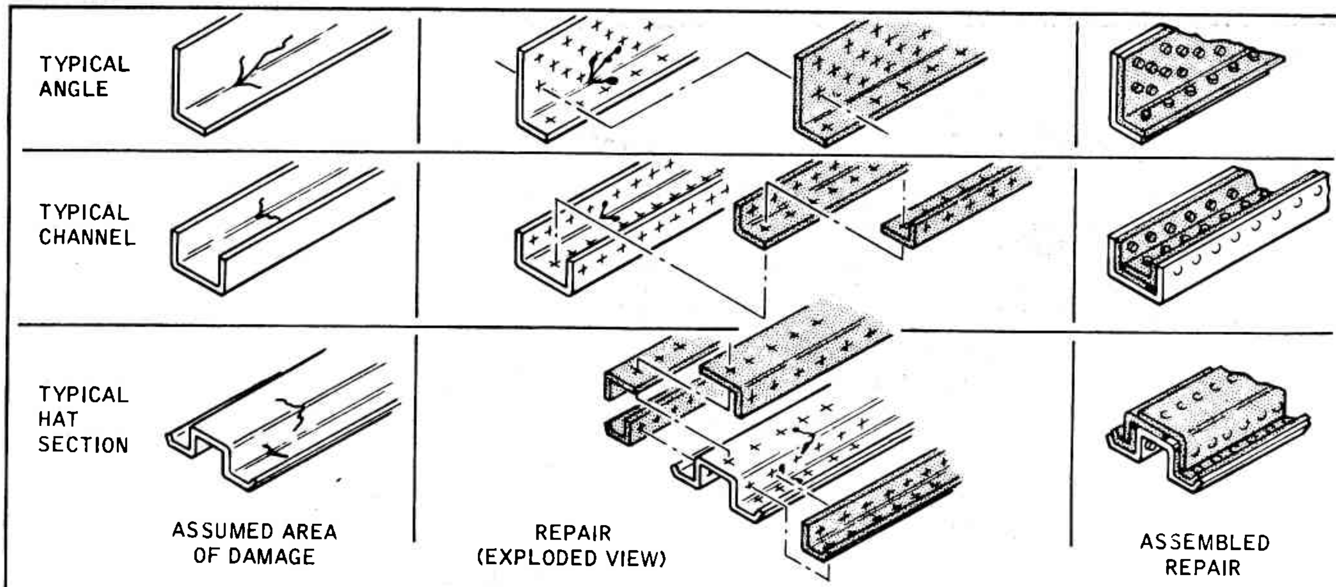
Figure 10-10B. Flat Plate - Edge Repair



REPAIR PROCEDURE

a. These repairs are for miscellaneous structural items.
b. Smooth roughness and break sharp corners on edges of cracks; stop drill cracks.
c. Repair parts shall be of same type material and one gage heavier than the original part.
d. Use all or part of channel or hat section repairs as required.
e. Locate and drill rivet pattern. Refer to Section I for method of rivet pattern determination.
f. Refer to Section I for finish requirements of repair surfaces.

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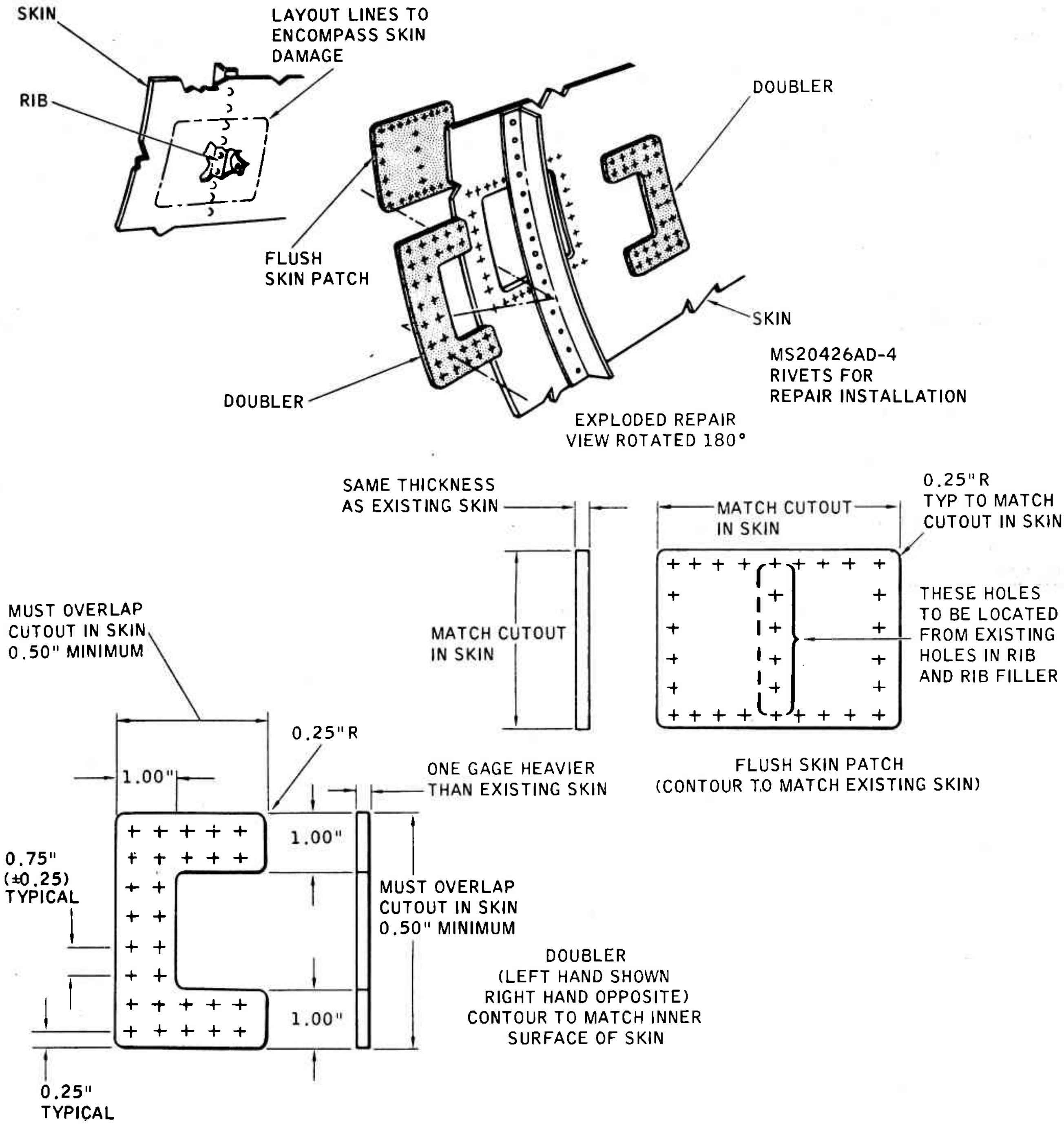
Figure 10-10C. Crack Repair of Miscellaneous Items



REPAIR PROCEDURE

a. Trim out damaged area to a regular shape.
b. Fabricate repair as shown.
c. Refer to Section X for finish requirements of repair surfaces.
d. Using predrilled holes in doublers as a guide, drill holes through existing skin. Use a No. 30 (0.128) drill forward of station 698.00. Use a No. 39 (0.099) drill aft of station 698.00.
e. Install flush skin patch.
f. Using predrilled holes in patch and existing holes in rib as a guide, perform drilling operation. Use a No. 30 (0.128) drill forward of station 698.00. Use a No. 39 (0.099) drill aft of station 698.00.
g. Remove doublers and patch and remove all burrs.
h. Reinstall doublers and patch and hold in place with clecos.
i. Rivet doublers and skin patch in place as shown.
j. Refer to applicable finish requirements in Section I if touch-up painting is required.

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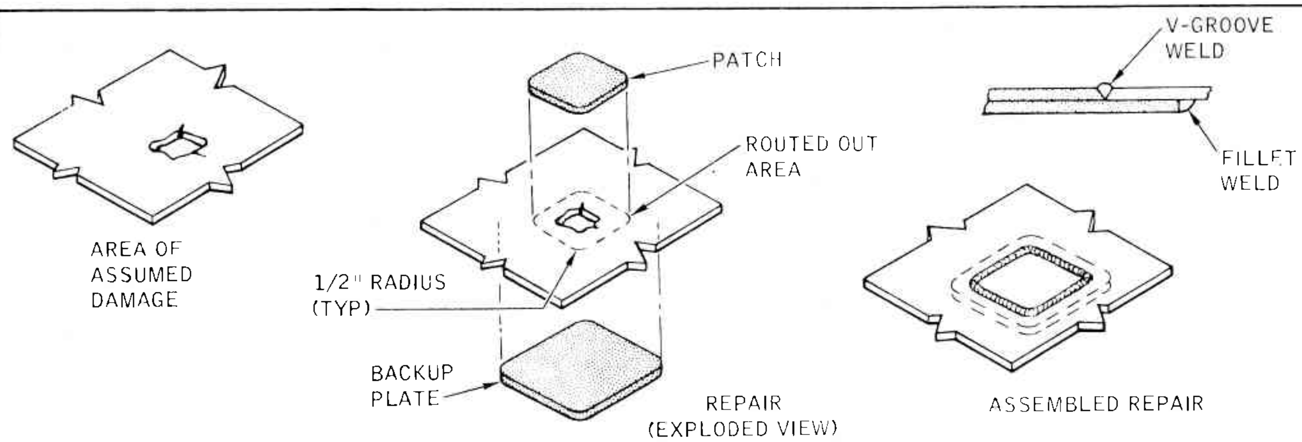
Figure 10-10D. Welded Repair for Welded Assemblies, Reinforced



REPAIR PROCEDURE

a. Reference should be made to the individual part drawing for welding and materials treatment specifications.

CAUTION

PARTS TO BE WELDED MUST BE REMOVED FROM AIRPLANE.

b. Route out cracked or damaged area.

c. Gap or V-groove crack or damaged area as required for full penetration weld to backup plate.

d. Fabricate backup plate of same material and gage or one gage lighter than damaged area. Plate to extend one half inch beyond edge of cutout or routed out area.

e. Weld backup plate over damaged area using fillet weld of same size as backup plate gage.

f. Fabricate flush plate of same gage and material as damaged area where routed out area cannot be filled with weld. Insert patch in routed out area.

g. Groove weld crack or gap closed with full penetration to backup plate.

h. Condition metal per part drawing.

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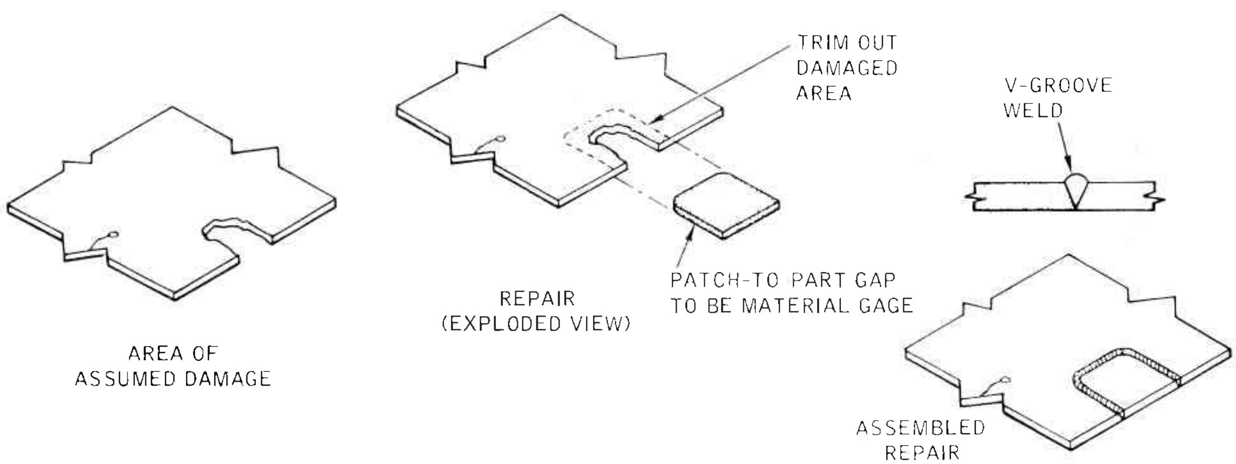
Figure 10-10E. Welded Crack Repair for Welded Assemblies



REPAIR PROCEDURE

a. Reference should be made to the individual part drawing for welding and material treatment specification.

CAUTION

PARTS TO BE WELDED MUST BE REMOVED FROM AIRPLANE.

b. Route out crack or damaged area.

c. Gap or V-groove crack or damaged area as required for full penetration weld.

d. Where required, fabricate filler patch of same material and gage as damaged area. Insert patch in damaged area.

e. Groove weld the routed out area with full penetration weld.

f. Condition metal per part drawing.

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Figure 10-10F. Dorsal Flairing Skin Repair

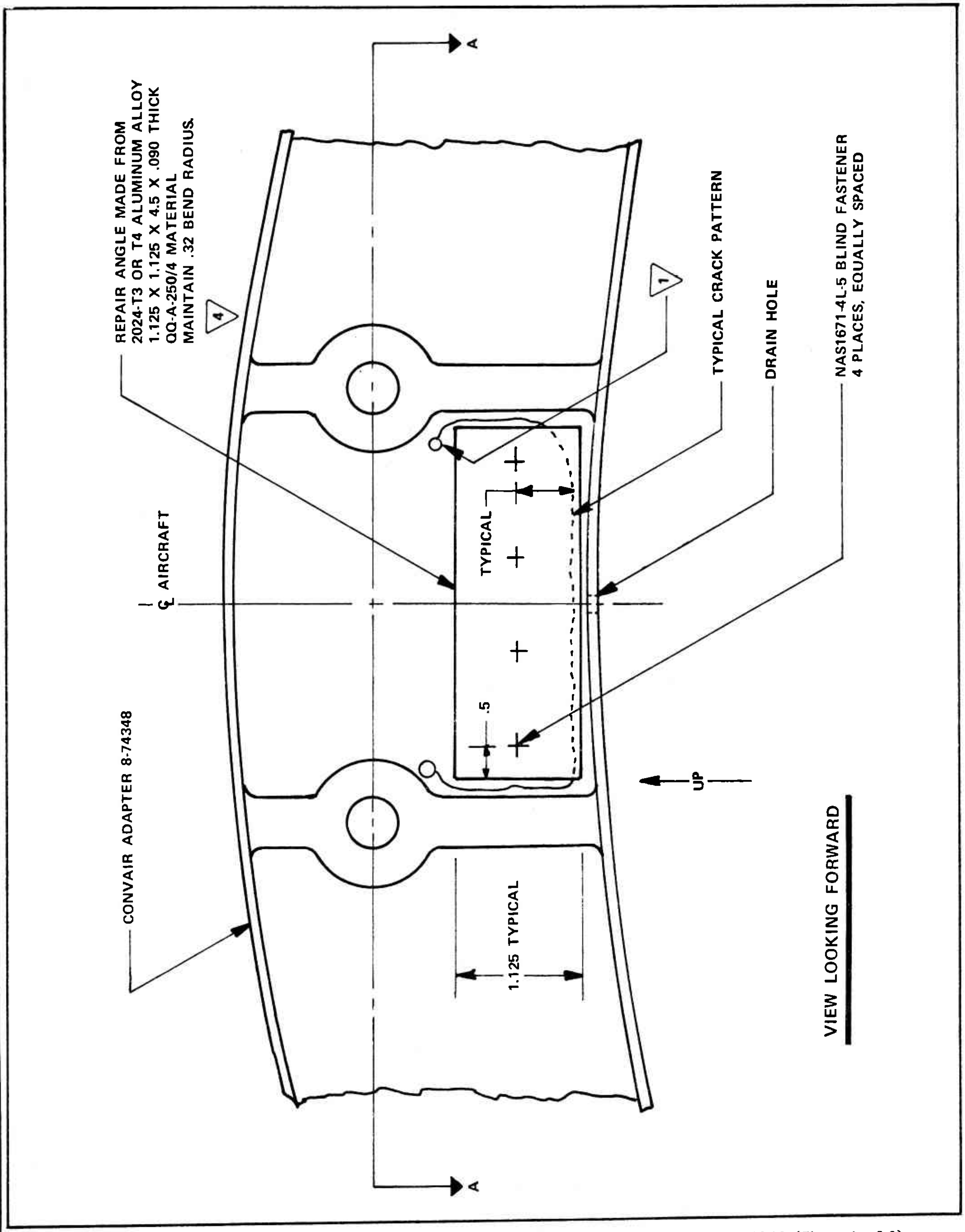


Figure 10-10G. Temporary Repair of F-106 Fuselage Sta. 556.75 Adapter, Part No. 8-74348 (Sheet 1 of 2)

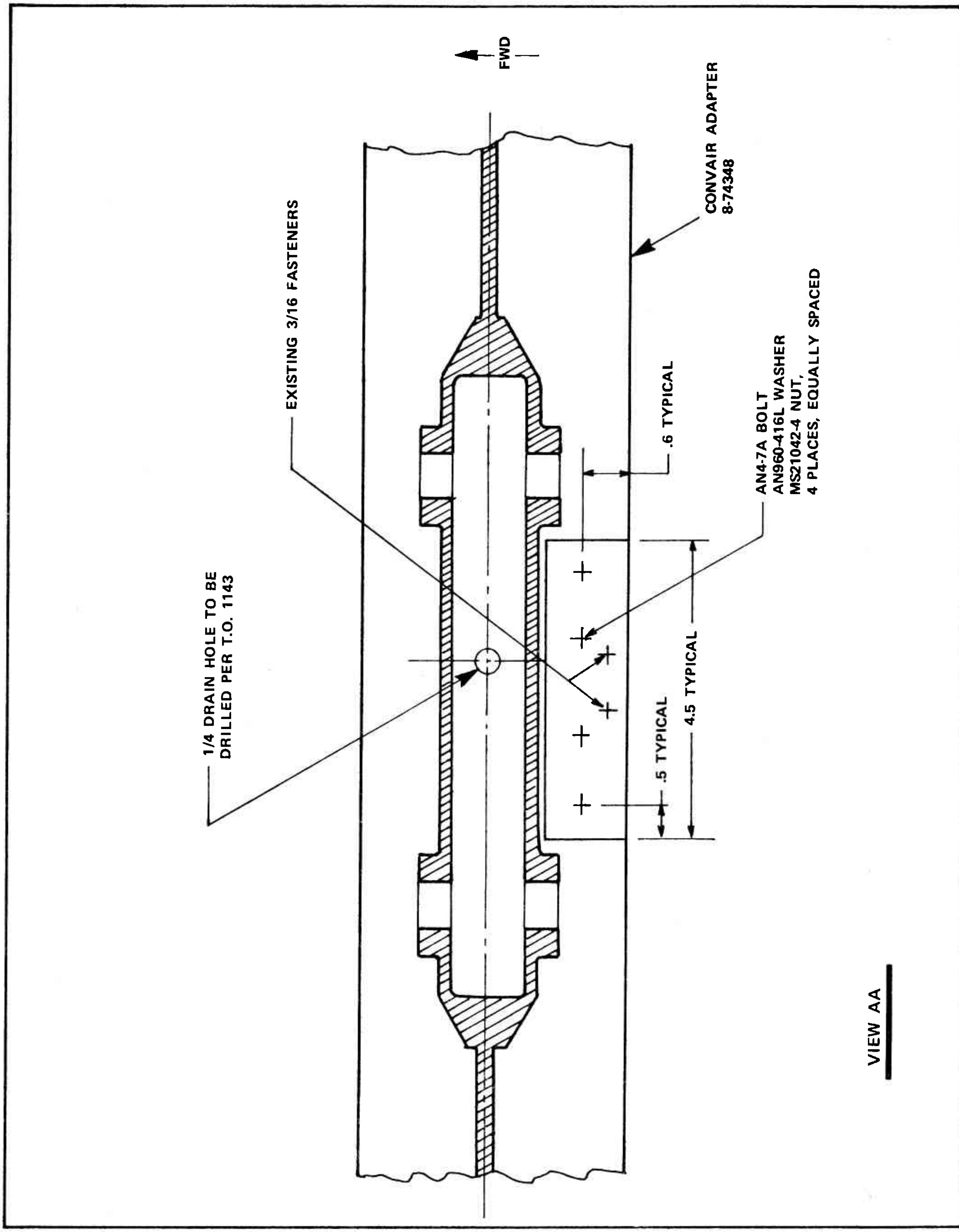
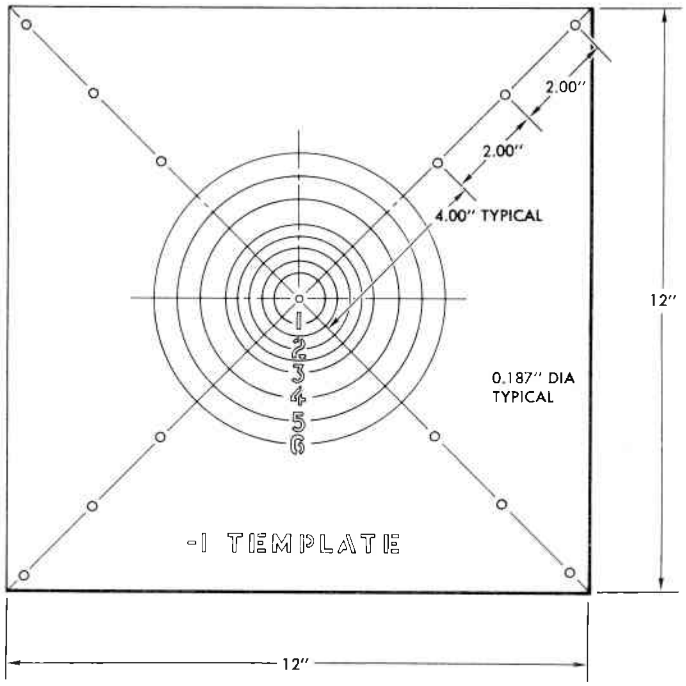
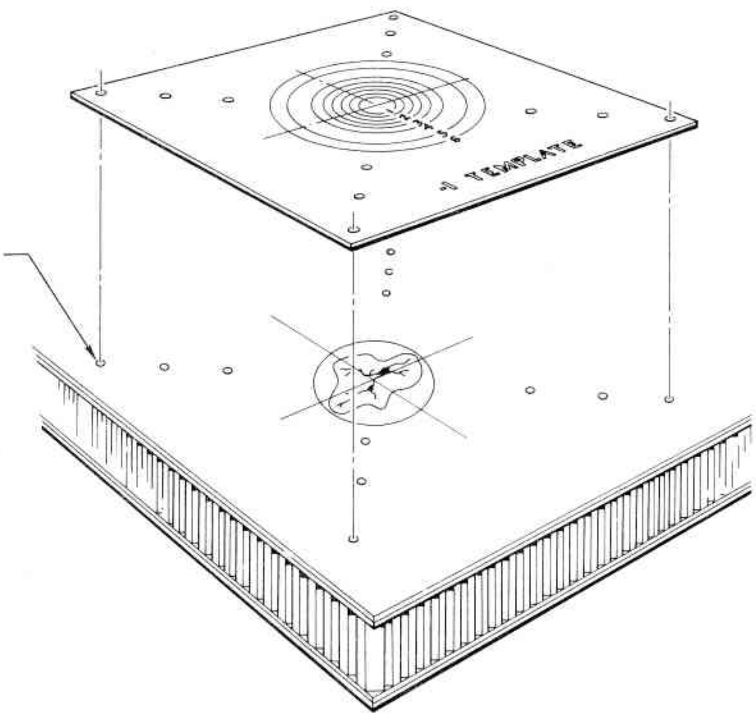


Figure 10-10G. Temporary Repair of F-106 Fuselage Sta. 556.75 Adapter, Part No. 8-74348 (Sheet 2 of 2)



NOTE:
1. MANUFACTURE TEMPLATE FROM PLEXIGLASS WITH CIRCLES AND DIAGONALS INSCRIBED ON UNDERSIDE.
2. NUMBERS INDICATE CIRCLE DIAMETER.



PROCEDURE

a. Center the template over outlined damage area.
b. Spot mark the skin through the hole in the template with a grease pencil.
c. Choose the diameter of the repair to be made (use the smallest diameter that encompasses the damage area.
d. Remove the template.
e. Mark the centerlines with a grease pencil.

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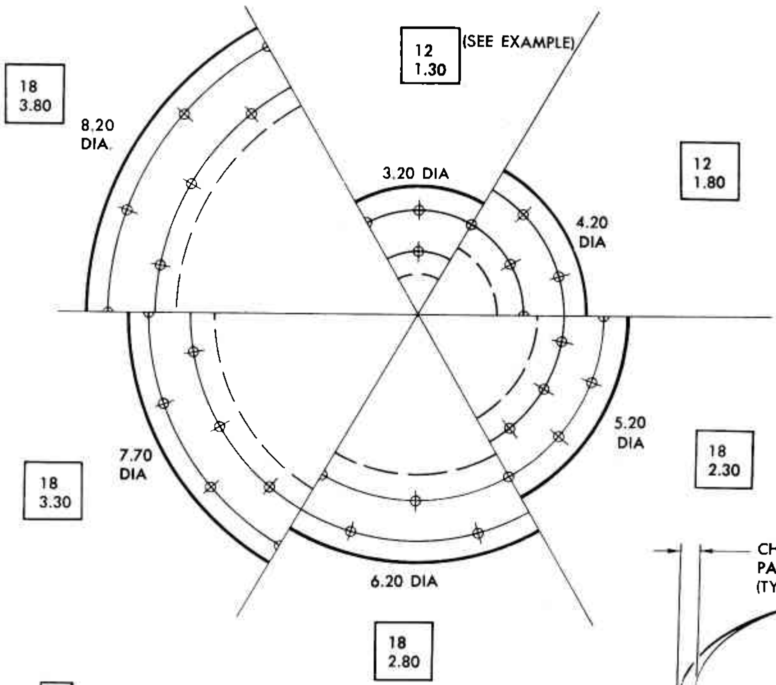
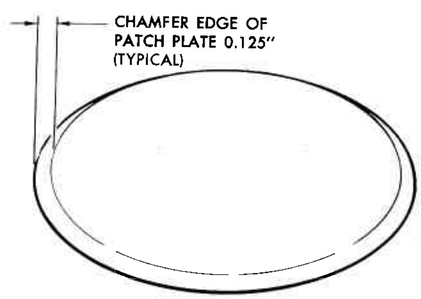
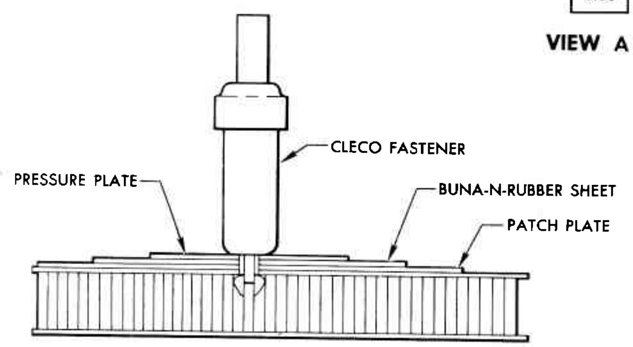
Figure 10-11. Honeycomb Repair — Damage Locating Template



VIEW A



VIEW B
(SEE NOTE 1)



VIEW C
(SEE NOTE 4)

DRILL SIZE FOR RIVETS	
RIVET	DRILL SIZE
⅛ EXPLOSIVE	NO. 29 DRILL
5/32 EXPLOSIVE	NO. 17 DRILL
⅛ CHERRY	NO. 30 DRILL
5/32 CHERRY	NO. 20 DRILL

EXAMPLE

12	← NUMBER OF RIVETS
1.30	← RADIUS OF HOLE PATTERN

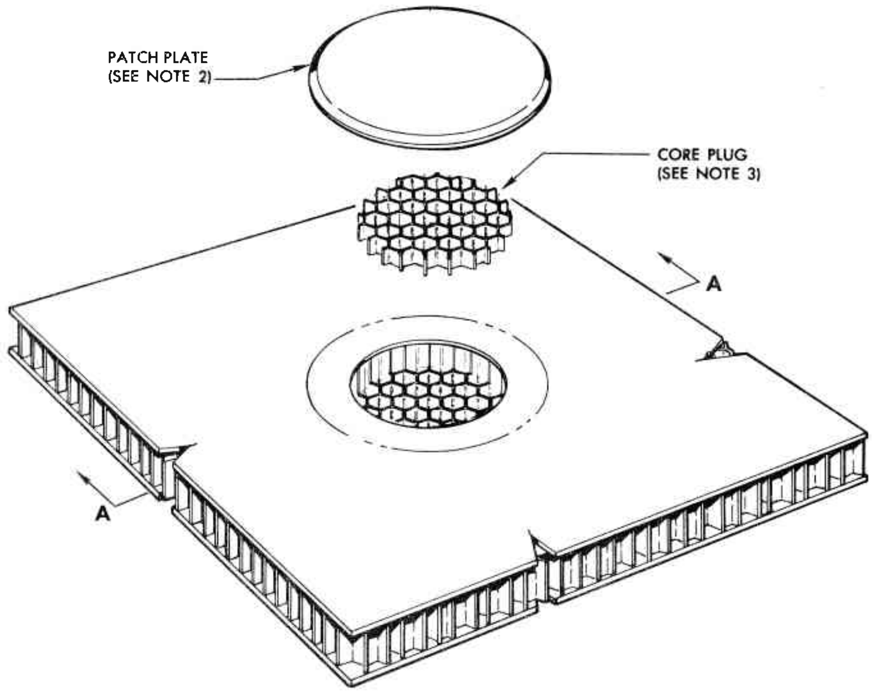
REPAIR PATCH TABLE	
DIAMETER OF DAMAGE	PATCH PLATE DIAMETER
1.000	3.200
1.500	3.700 (SEE NOTE 2)
2.000	4.200
2.500	4.700 (SEE NOTE 3)
3.000	5.200
4.000	6.200
5.000	7.200
6.000	8.200

NOTES:

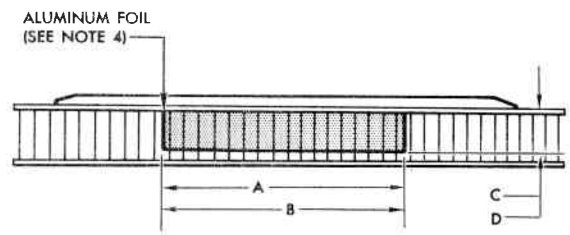
1. ALL PATCH PLATES USED IN HONEYCOMB REPAIRS ARE 0.025 INCH GAGE ALUMINUM ALLOY. THE TYPE AND CONDITION OF THE ALLOY SHOULD BE THE SAME AS THE METAL REMOVED FROM REPAIR AREA.
2. HOLE PATTERN SAME AS 3.200 INCH DIAMETER PATCH.
3. HOLE PATTERN SAME AS 4.200 INCH DIAMETER PATCH.
4. ALL PATCH PLATES ARE INSTALLED OVER FILM FM47 FILM. FM47 MUST HAVE 60 PSI TO 100 PSI PRESSURE WHILE HEAT IS BEING APPLIED FOR CURE. THE USE OF CLECO FASTENERS AS SHOWN IN VIEW C OR OTHER METHODS SHOWN IN THIS SECTION MAY BE USED TO APPLY THIS PRESSURE.

06.03.295-1B

Figure 10-12. Honeycomb Repairs — Patch Plates, Core Plugs, and Riveted Patterns (Sheet 1 of 2)



CORE PLUG DIMENSIONS			
REPAIR HOLE DIAMETER A	CORE PLUG DIAMETER B	REPAIR HOLE DEPTH C	CORE PLUG THICKNESS D
1.5	1.52	.250	.250
1.5	1.52	.375	.375
1.5	1.52	.500	.500
2.0	2.02	.250	.250
2.0	2.02	.375	.375
2.0	2.02	.500	.500
2.5	2.52	.250	.250
2.5	2.52	.375	.375
2.5	2.52	.500	.500
3.0	3.02	.250	.250
3.0	3.02	.375	.375
3.0	3.02	.500	.500
4.0	4.02	.250	.250
4.0	4.02	.375	.375
4.0	4.02	.500	.500
5.0	5.02	.250	.250
5.0	5.02	.375	.375
5.0	5.02	.500	.500
6.0	6.02	.250	.250
6.0	6.02	.375	.375
6.0	6.02	.500	.500
1.0	1.02	1.000	1.000
2.0	2.02	1.000	1.000
3.0	3.02	1.000	1.000
4.0	4.02	1.000	1.000
5.0	5.02	1.000	1.000
6.0	6.02	1.000	1.000



SECTION A-A

NOTE

1. THIS IS A CORE FILL OUTSIDE PATCH FOR DAMAGE ON ONE SIDE OF A HONEYCOMB STRUCTURE. APPLY PRESSURE AND HEAT PER APPLICABLE REPAIR INSTRUCTIONS.
2. SEE SHEET 1 FOR PATCH PLATE DIMENSIONS; AND RIVET PATTERN AND SIZES (IF PATCH PLATE IS RIVETED).
3. MAKE CORE PLUG FROM PERFORATED HONEYCOMB 1/8-INCH HEX CELL CORE, 0.002-INCH FOIL, SPECIFICATION MIL-C-7438B TYPE 1D. TOLERANCE FOR CORE PLUG THICKNESS (D) IS (±0.002-INCH).
4. CUT ALUMINUM FOIL FROM 0.004-INCH CLEAN STOCK MATERIAL.

06.03.295-2

Figure 10-12. Honeycomb Repairs — Patch Plates, Core Plugs, and Rivet Patterns (Sheet 2 of 2)

STEP 1

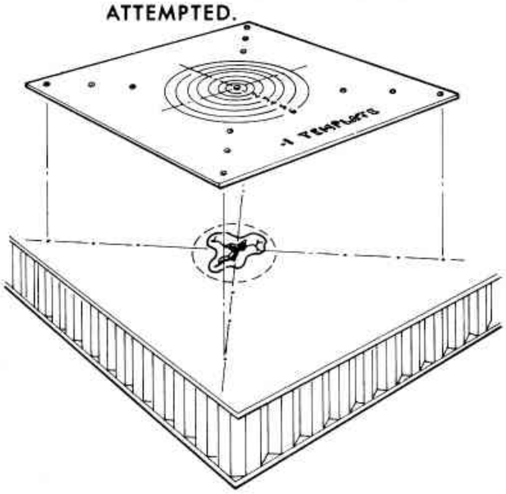
CLEAN THE DAMAGED AND SURROUNDING AREA WITH CHEESECLOTH SOAKED WITH METHYL ETHYL KETONE.

CAUTION

PAINT MUST BE REMOVED FROM REPAIR AREA BEFORE ANY CLEANING IS ATTEMPTED.

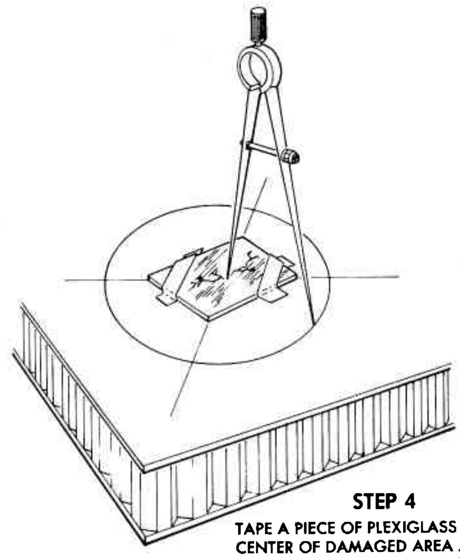
STEP 2

OUTLINE DAMAGED AREA WITH GREASE PENCIL.



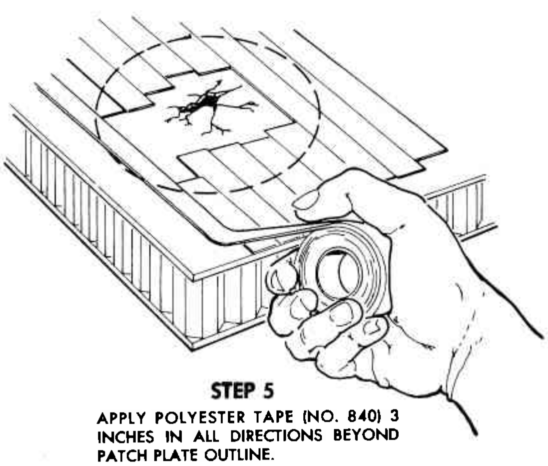
STEP 3

MARK CENTERLINES THROUGH CENTER OF DAMAGED AREA. USE TEMPLATE (SEE FIGURE 10-11).



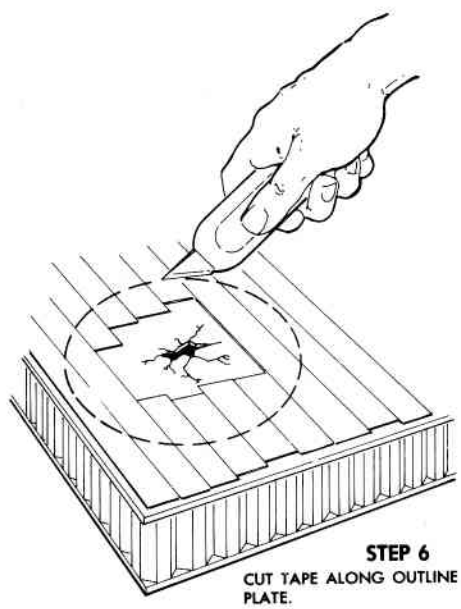
STEP 4

TAPE A PIECE OF PLEXIGLASS OVER THE CENTER OF DAMAGED AREA AND OUTLINE PATCH PLATE AREA AS SHOWN.



STEP 5

APPLY POLYESTER TAPE (NO. 840) 3 INCHES IN ALL DIRECTIONS BEYOND PATCH PLATE OUTLINE.



STEP 6

CUT TAPE ALONG OUTLINE OF PATCH PLATE.

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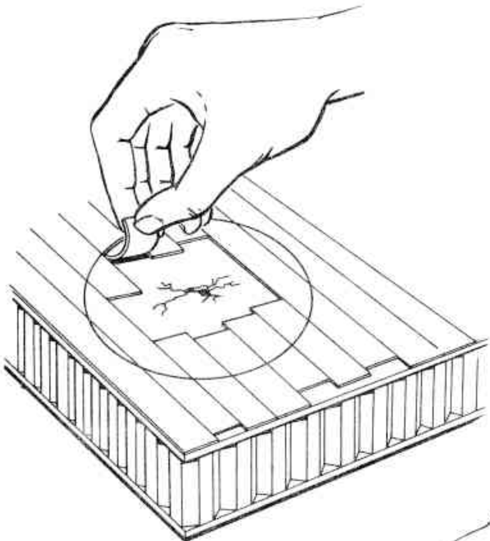
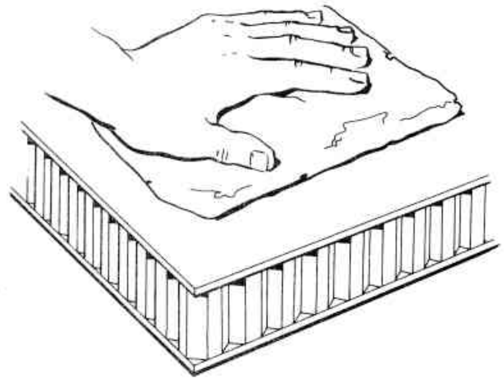
Figure 10-13. Honeycomb Repairs — Preparation and Cleaning (Sheet 1 of 2)



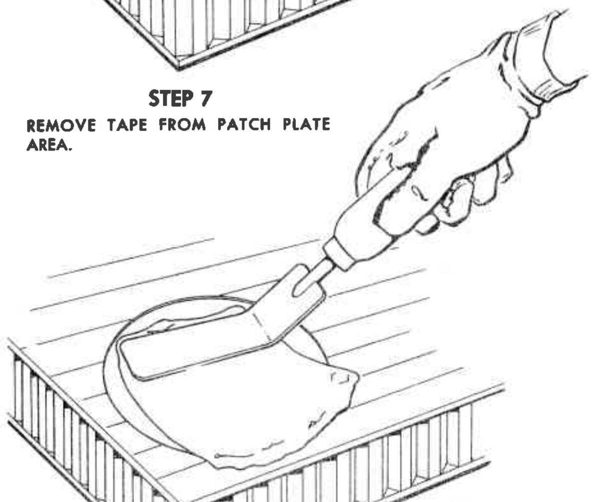
STEP 7

REMOVE TAPE FROM PATCH PLATE AREA.



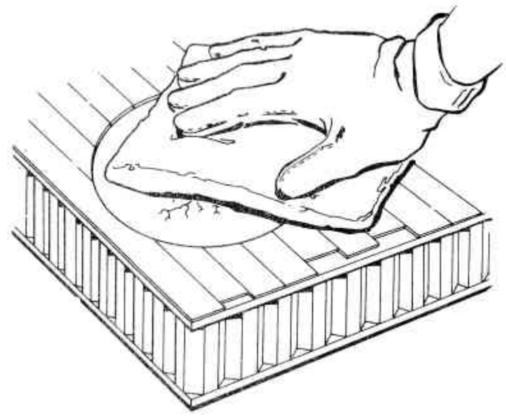
STEP 8

CLEAN BARED PATCH PLATE AREA WITH METHYL ETHYL KETONE.



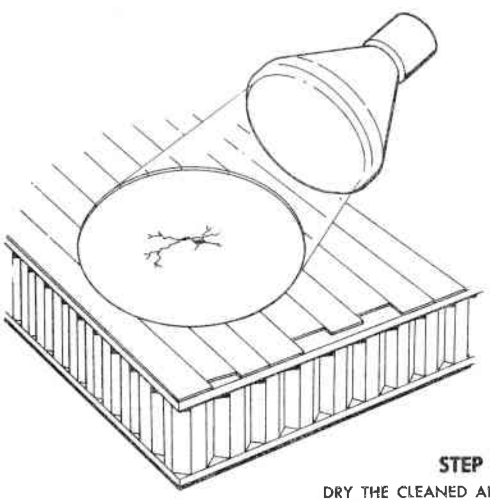
STEP 9

APPLY CLEANER 1/16 INCH TO 1/8 INCH THICK ON BARED SKIN AREA. ALLOW CLEANER TO REMAIN ON SKIN SURFACE AT LEAST 45 MINUTES. TEMPERATURE OF SKIN SHOULD BE 70°F OR ABOVE. (SEE NOTE).



STEP 10

REMOVE CLEANER. WASH BARED AREA THOROUGHLY WITH A CLEAN CHEESE-CLOTH SATURATED WITH CLEAR WATER.



STEP 11

DRY THE CLEANED AREA FOR 15 MINUTES WITH A HEAT LAMP.

NOTE

CLEANER IS MADE AS FOLLOWS:

1. COMBINE TEN PARTS OF SULFURIC ACID FEDERAL SPECIFICATION O-A-115(1), FOUR PARTS OF SODIUM DICHROMATE FEDERAL SPECIFICATION 0-5-595a(3), AND THIRTY PARTS OF WATER.
2. MIX ONE PART OF THE ABOVE SOLUTION WITH THREE PARTS OF BARIUM SULFATE (TECHNICAL GRADE).

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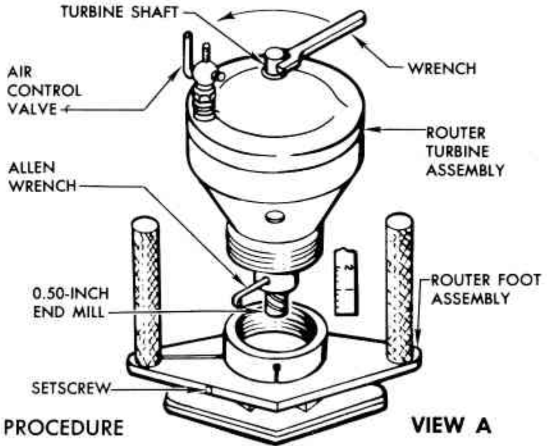
Figure 10-13. Honeycomb Repairs — Preparation and Cleaning (Sheet 2 of 2)



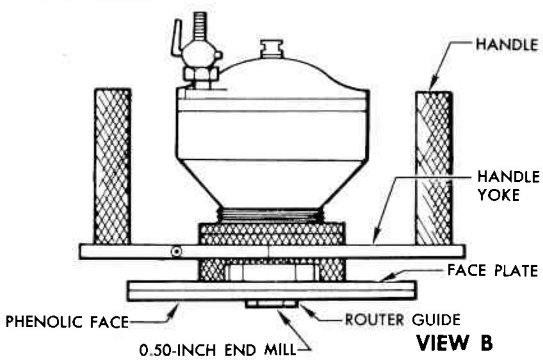
PROCEDURE

a. Remove router turbine assembly from router foot assembly as shown in View A.
b. Install 0.50-inch end mill in router turbine assembly chuck.

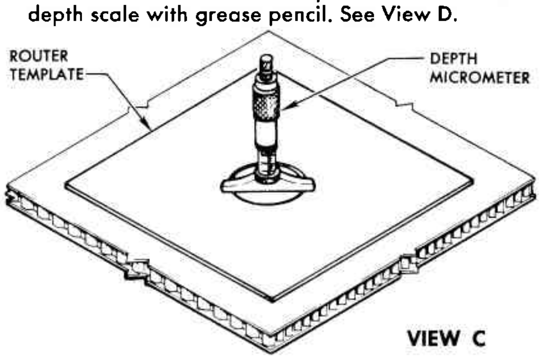
NOTE

THE END MILL MUST EXTEND 1.0 INCH BEYOND THE END OF THE CHUCK. WHEN USING THE ROUTER TURNTABLE ASSEMBLY (FIGURE 10-19) A LONGER END MILL MAY BE REQUIRED.

c. Tighten end mill by holding Allen wrench in chuck hole and turning turbine shaft counterclockwise with wrench.



d. Screw router turbine assembly into router foot assembly until end mill is flush with router guide. See View B.
e. Make reference mark on adjustment threads and depth scale with grease pencil. See View D.



f. Measure depth of tape and template in area of hole with depth micrometer as shown in View C.
g. Subtract 0.10-inch from tape and template depth measurement for depth of router guide.

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h. Increase depth of end mill one mark on depth scale for each 0.001-inch difference between depth of router template and router guide.
i. Place router on router template and rotate turbine shaft by hand for four or five revolutions.
j. Remove router from template and check depth setting.

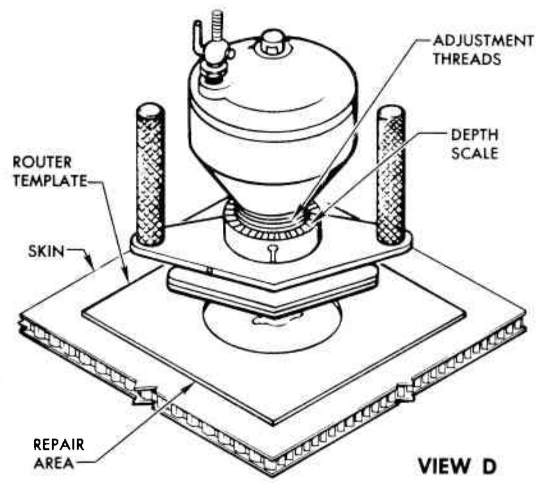
NOTE

THE END MILL MUST BE FLUSH WITH THE SKIN. IF THE END MILL IS TOO LOW AND WILL NOT TURN WHEN THE TURBINE SHAFT IS ROTATED, OR IS SCRATCHING THE SKIN SURFACE, DECREASE THE DEPTH SETTING UNTIL THE END MILL IS FLUSH WITH THE SKIN.

k. Increase depth setting 0.002-inch and tighten setscrew.
l. Attach router assembly hose to 90- to 100-pound air source.

WARNING

AIR CONTROL VALVE ON ROUTER ASSEMBLY MUST BE IN OFF POSITION PRIOR TO ATTACHING AIR SOURCE.



m. Place router assembly on template with end mill over template hole.
n. Start router assembly with air control valve and make initial cut.

WARNING

SAFETY GLASSES MUST BE WORN BY PERSONNEL IN VICINITY OF REPAIR DURING ROUTER OPERATION.

NOTE

OPERATE ROUTER ASSEMBLY IN FORWARD AND AFT MOTIONS PARALLEL TO CONTOUR OF THE SKIN.

o. Remove router assembly.

CAUTION

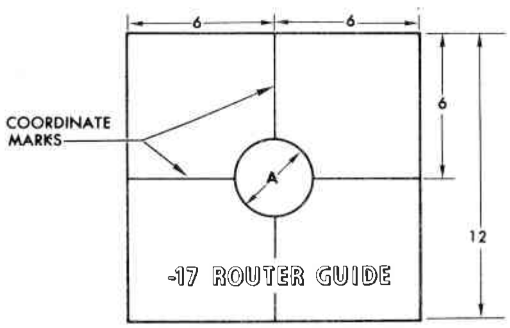
DO NOT LIFT ROUTER ASSEMBLY UNTIL TURBINE HAS STOPPED ROTATING.

p. Measure depth of cut with depth micrometer to determine accuracy of first setting.
q. Make series of router cuts, increasing end mill depth in increments no greater than 0.10-inch for core areas and 0.010-inch for step cuts in skin until desired routing depth is obtained.

NOTE

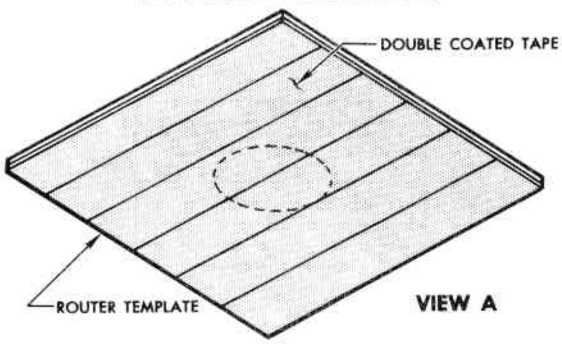
FOR A MORE ACCURATE STEP-CUT DEPTH, ROUTE LAST 0.004-INCH OF STEP-CUT DEPTH WITH TWO 0.002-INCH CUTS.

Figure 10-14. Honeycomb Repairs — Router Assembly



ROUTER GUIDE TEMPLATE
(SEE TABLE I)

TABLE I		
OVERALL DAMAGE SIZE	TEMPLATE HOLE SIZE A	TEMPLATE DASH NUMBER
1.440	1.0	-5
1.940	1.5	-1
2.440	2.0	-9
2.940	2.5	-11
3.440	3.0	-13
4.330	4.0	-15
5.440	5.0	-17
6.440	6.0	-19



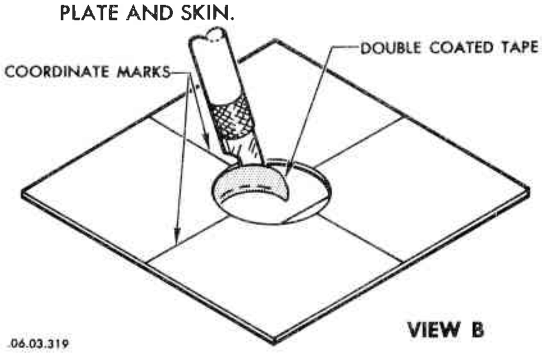
NOTE
1. FABRICATE GUIDE TEMPLATES FROM 0.125-INCH PHENOLIC, SPECIFICATION MIL-P-15035 FBM OR SIMILAR MATERIAL.
2. SCRIBE COORDINATE MARKS DEEP ENOUGH TO BE CLEARLY VISIBLE.
3. STENCIL APPLICABLE DASH NO. AND NAME ON EACH GUIDE TEMPLATE, AS SHOWN, USING 0.50-INCH BLACK LETTERS.

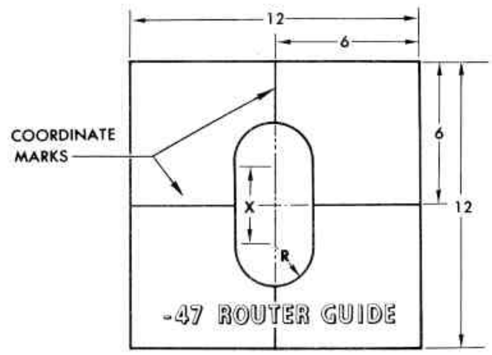
PROCEDURE

a. Tape one side of template using #400 double coated tape leaving protective backing on tape. See View A.

CAUTION

THE TAPE MUST COMPLETELY SURROUND THE TEMPLATE HOLE TO KEEP ALUMINUM CHIPS FROM GETTING BETWEEN THE TEMPLATE AND SKIN.





ROUTER GUIDE TEMPLATE
(SEE TABLE II)

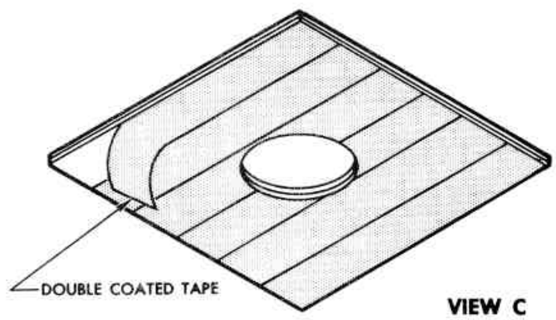
TABLE II			
OVERALL DAMAGE SIZE	TEMPLATE HOLE SIZE		TEMPLATE DASH NUMBER
	R	X	
3.0x6.25	1.720	3.250	-45
3.0x4.625	1.720	1.625	-47
2.0x5.25	1.220	3.250	-49
2.0x3.625	1.220	1.625	-51

b. Remove excess tape flush with edge of template hole using knife as shown in View B.

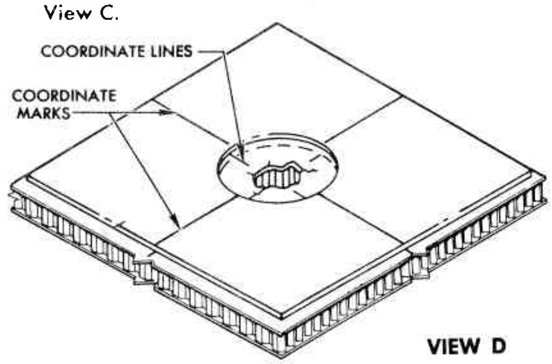
CAUTION

THE TEMPLATE MUST NOT BE PLACED ON THE SKIN WHILE CUTTING THE TAPE.

c. Clean area immediately surrounding the repair using Methyl-Ethyl-Ketone, Specification TT-M-261.



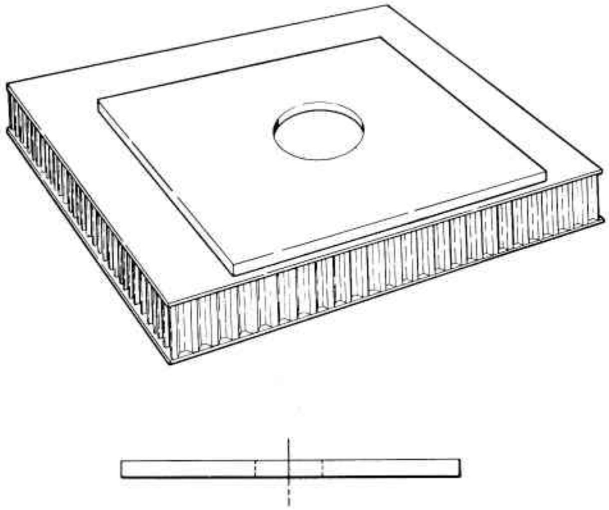
d. Remove protective backing from tape as shown in View C.



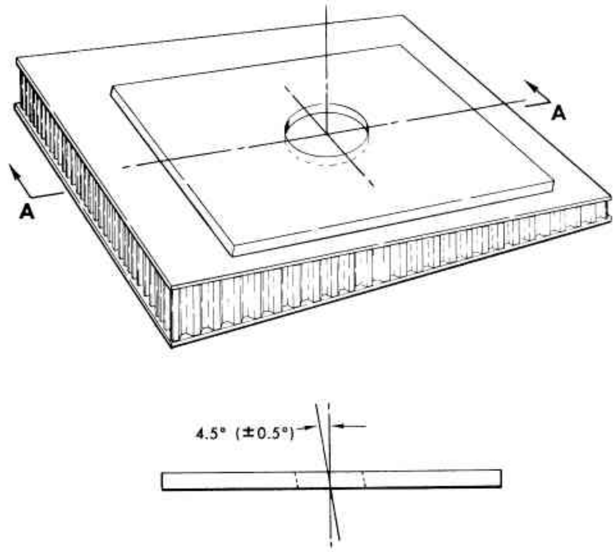
e. Using a grease pencil mark coordinate lines through center of the damaged area. See View D.

f. Align template coordinates with damaged area coordinates and secure template to skin.

Figure 10-15. Honeycomb Repairs — Router Guide Templates



VIEW A
HOLE SAW GUIDE TEMPLATE
(SEE TABLE I)



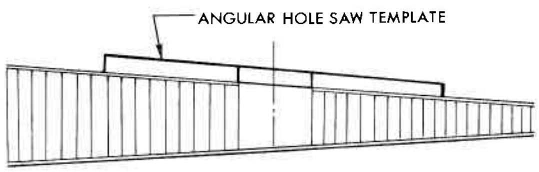
VIEW B
ANGULAR HOLE SAW GUIDE TEMPLATE
(SEE TABLE II)

TABLE I		
OVERALL DAMAGE SIZE	TEMPLATE HOLE SIZE	TEMPLATE DASH NUMBER
1.0	1.010	-21
2.0	2.010	-23
3.0	3.010	-25
4.0	4.010	-27
5.0	5.010	-29
6.0	6.010	-31

TABLE II		
OVERALL DAMAGE SIZE	TEMPLATE HOLE SIZE	TEMPLATE DASH NUMBER
1.0	1.010	-33
2.0	2.010	-35
3.0	3.010	-37
4.0	4.010	-39
5.0	5.010	-41
6.0	6.010	-43

NOTE

1. FABRICATE GUIDE TEMPLATES FROM 0.125-INCH PHENOLIC, SPECIFICATION MIL-P-15035 FBM OR SIMILAR MATERIAL. REFER TO TABLE I OR II FOR HOLE SIZES. SEE FIGURE 10-15, SHEET 1 FOR OUTSIDE DIMENSIONS.
2. DRILL HOLES IN ANGULAR GUIDE TEMPLATES AT A 4.5° (±0.5°) ANGLE AS SHOWN IN VIEW B.
3. STENCIL APPLICABLE DASH NO. (SEE TABLE I OR II) AND THE WORDS "HOLE SAW GUIDE" SIMILAR TO ILLUSTRATION SHOWN IN FIGURE 10-15 SHEET 1. USE 0.50-INCH BLACK LETTERS.



SECTION A-A

PROCEDURE

a. Accomplish steps "a" thru "d" of figure 10-15.
b. Align template so that hole encompasses entire damage area and secure template to skin. (Align angular guide template so that hole is parallel to the vertical direction ribbon as shown in Section A-A.)

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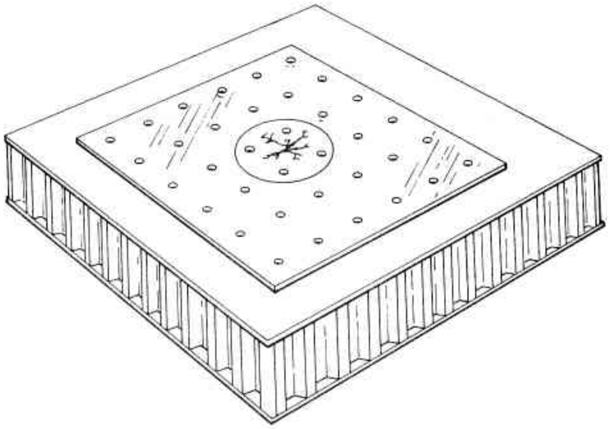
Figure 10-16. Honeycomb Repairs — Hole Saw Guide Templates



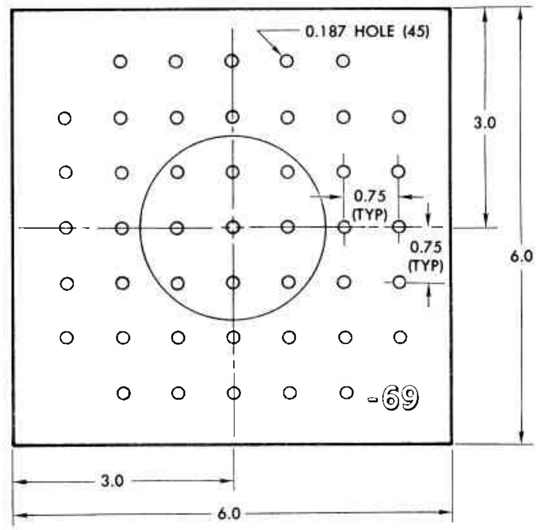
HOLE PATTERN TEMPLATE
VIEW A

NOTE

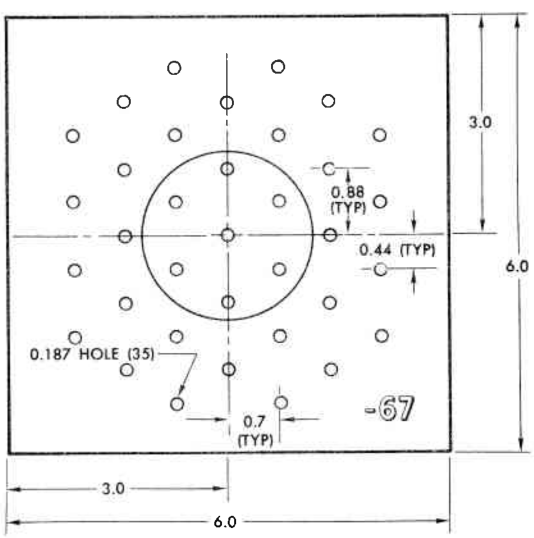
1. FABRICATE HOLE PATTERN TEMPLATES FROM 0.250-INCH CLEAR PLEXIGLASS, COMMERCIAL GRADE.
2. SCRIBE COORDINATES (CENTER-LINES) AND 2.50-INCH CIRCLE ON UNDERSIDE DEEP ENOUGH TO BE CLEARLY VISIBLE.
3. DRILL 0.187 HOLES LOCATED AS SHOWN.
4. SCRIBE OR SANDBLAST DASH NUMBER ON TOP SIDE AS SHOWN. MAKE NUMBERS APPROXIMATELY 3/8-INCH.

PROCEDURE

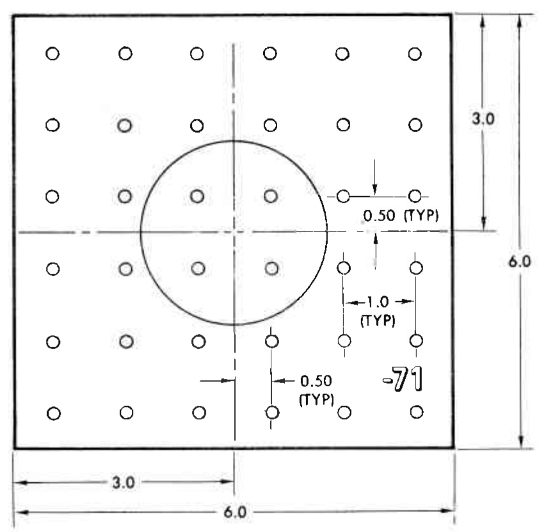
a. Select template that will permit the largest number of holes immediately surrounding the damaged area.
b. Center template so that circle encompasses entire damage area as shown in View A.
c. Secure template in place with #850 tape. See figure 10-15.
d. Centerpunch each hole lightly and remove template from part.



-69



-67



-71

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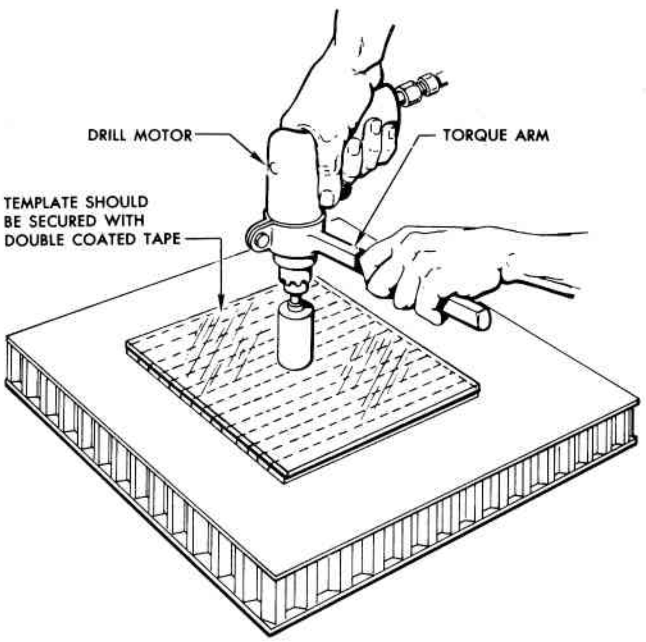
Figure 10-17. Honeycomb Repairs — Hole Pattern Templates



NOTE
THESE HOLE SAWS ARE USED IN CONJUNCTION WITH TEMPLATES -21 THRU -43. USE HOLE SAWS IN A SLOW SPEED DRILL MOTOR HAVING A ½-INCH CHUCK AND EQUIPPED WITH A TORQUE BAR.

PROCEDURE

a. Select template having hole large enough to encompass the damage area.
b. Prepare template and secure to part as described on figure 10-15.
c. Select hole saw of same diameter as template.
d. Install pilot holder in saw and install saw assembly in drill motor chuck.
e. Place saw in template at correct angle relative to part.
f. Turn on drill motor holding it firmly by torque arm.

CAUTION

BE ALERT FOR SUDDEN BINDING OR STALLING, AND DEPTH OF CUT AS SAW GOES THROUGH TOP SKIN.

g. Saw hole to the depth required by the repair instructions and remove saw and template.
h. Remove pilot holder and clean out inside of saw using punch through holder hole.

HOLE SAW DIMENSIONS					
SAW ASSY DASH NO.	SAW LENGTH	OVERALL DIAMETER	PLATE DIAMETER	WALL THICKNESS	NO. OF TEETH
-55	5.0	1.000	0.902	0.049	8
-57	5.0	2.000	1.902	0.049	12
-59	5.0	3.000	2.884	0.058	18
-61	5.0	4.000	3.870	0.065	18
-63	5.0	5.000	4.870	0.065	24
-64	5.0	6.000	5.870	0.065	24

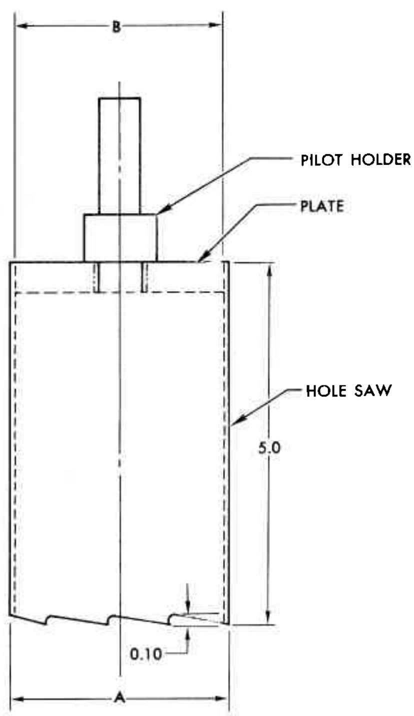


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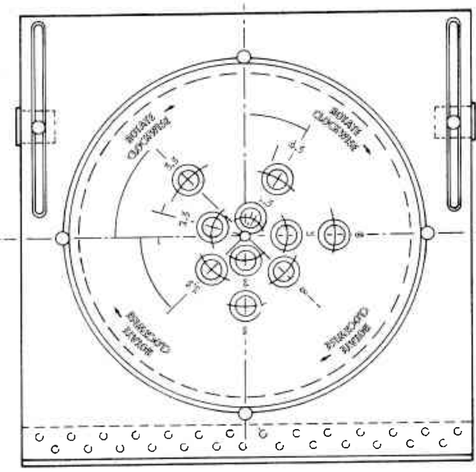
Figure 10-18. Honeycomb Repairs — Hole Saws



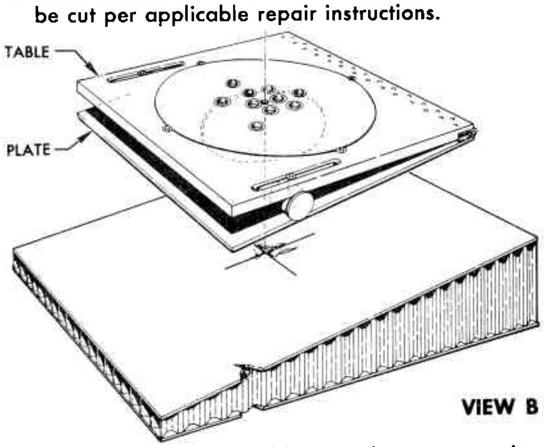
VIEW A

NOTE

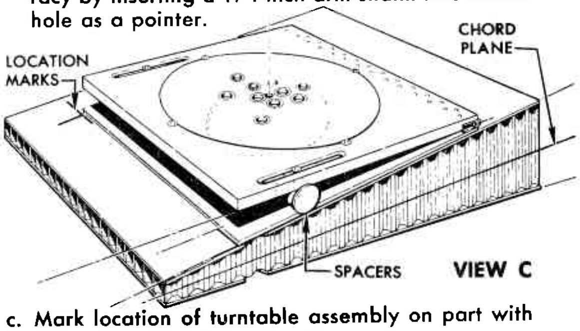
THIS TURNTABLE IS USED IN CONJUNCTION WITH THE -3 ROUTER ASSEMBLY. BY USING THE TURNTABLE ASSEMBLY IT IS POSSIBLE TO MACHINE HOLES FROM 1.5 TO 6.0 INCHES DIAMETER IN INCREMENTS OF ½ INCH. THE TABLE MAY BE SET SO THAT THE HOLE IS NORMAL TO THE SURFACE OR AT AN ANGLE OF FROM 2 TO 6 DEGREES MAKING IT POSSIBLE TO MACHINE HOLES IN TAPERED PARTS PERPENDICULAR TO THE CHORD PLANE OF THE PART. SEE VIEW D.

PROCEDURE

a. Establish coordinates and determine size of hole to be cut per applicable repair instructions.



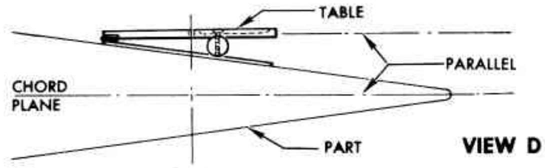
b. Center turntable assembly over damage area by sighting thru center (1/4-inch) hole to coordinate axis as shown in View B. Check location for accuracy by inserting a 1/4-inch drill shank thru center hole as a pointer.

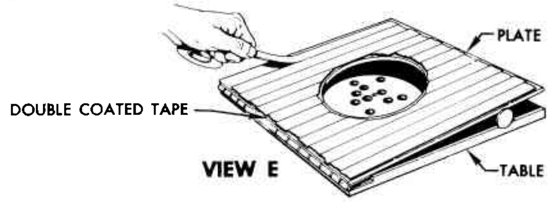


c. Mark location of turntable assembly on part with a grease pencil. See View C.

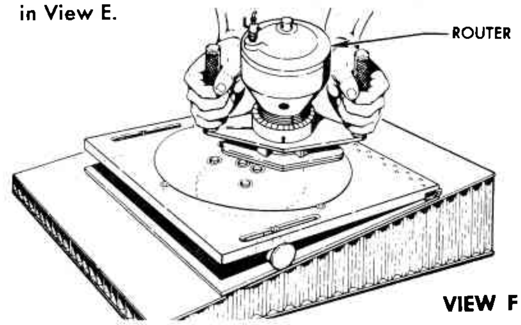
.06.03.323

d. Adjust spacers so that the table is parallel to the chord plane of the part. See View C and D.





e. Remove turntable assembly and apply #400 double coated tape to underside of plate as shown in View E.



f. Secure turntable assembly to part using location marks established in step "c." Alo see figure 10-15.

NOTE

CHECK TO SEE THAT TURNTABLE ASSEMBLY IS FIRMLY SECURED TO PART, THAT SPACER ADJUSTMENT IS SECURE, AND THAT TABLE IS PARALLEL TO CHORD PLANE OF PART.

g. Set up router assembly as outlined in figure 10-14.
h. Place router assembly on turntable with end mill in hole having number corresponding to diameter of hole to be routed out. See Views A and F.
i. Adjust router so that end mill makes contact at nearest point on skin of part.
j. Holding router assembly firmly by handles, start hole in skin with the aid of an assistant adjusting router depth downward.
k. Cut and adjust, moving router clockwise, in a complete circle, until the desired depth is reached.

NOTE

IF ADJUSTMENT LIMIT IS REACHED BEFORE DESIRED DEPTH, SHUT OFF ROUTER AND INSTALL NEXT LENGTH END MILL. CONTINUE AS DIRECTED IN THE PRECEDING STEP.

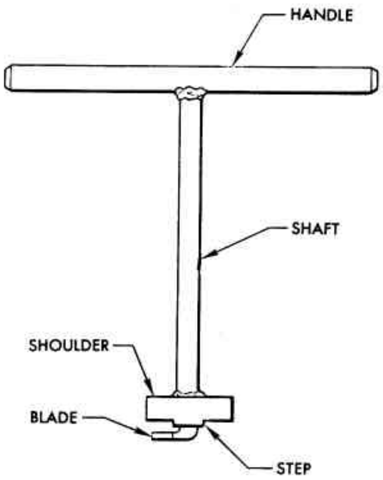
l. Remove router and turntable assembly.
m. Pry out skin and core left in center of hole being careful not to bend the skin at edge of hole.
n. If hole is to extend thru lower skin, proceed as in step "k" except stop router approximately 1/2-inch before the circle is completed.

NOTE

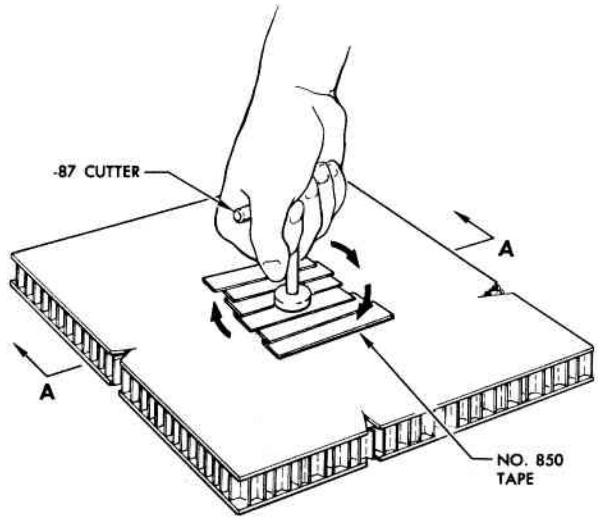
ROUTER END MILL SHOULD EXTEND APPROXIMATELY ¼ INCH THRU LOWER SKIN.

o. Turn off and remove router assembly.
p. Break off the skin disc left in the hole.
q. Repeat step "h" and finish routing out hole.
r. Remove router and turntable assemblies.

Figure 10-19. Honeycomb Repairs — Router Turntable Assembly



-87 CUTTER ASSEMBLY

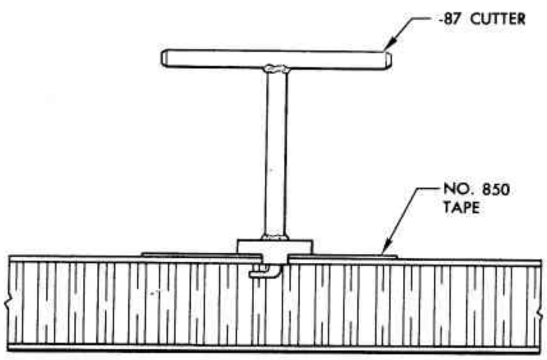


VIEW A

NOTE
THE -87 CUTTER ASSEMBLY IS A HAND OPERATED TOOL USED FOR UNDERCUTTING 0.375-INCH DIAMETER HOLES IN THE REPAIR OF HONEYCOMB STRUCTURE. SUCH UNDERCUTTING IS DONE PRIOR TO APPLICATION OF POTTING COMPOUND.

PROCEDURE

a. Apply #850 tape to area approximately one inch wide around repair hole as shown in view A.
b. Insert cutter assembly blade into 0.375-inch diameter hole and force tip of blade into core.
c. Place cutter assembly shaft in vertical position with shoulder parallel to skin.
d. Rotate handle slowly in clockwise direction and apply pressure until shoulder step moves down into 0.375-inch hole.
e. Rotate handle in clockwise direction until cutter turns freely.
f. Remove cutter assembly from hole and remove tape.



SECTION A-A

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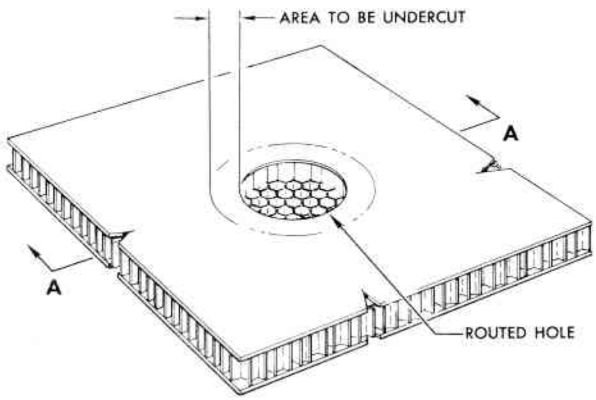
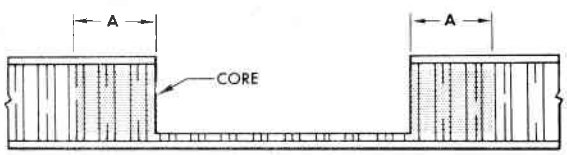
Figure 10-20. Honeycomb Repairs — Cutter Assembly

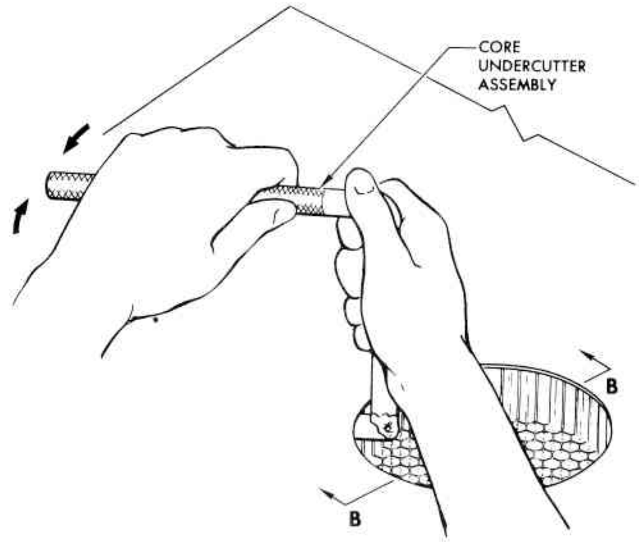


VIEW A

DIMENSION TO BE UNDERCUT



SECTION A-A



VIEW C

CORE UNDERCUTTING TABLE	
CORE UNDERCUTTER TOOL	CORE UNDERCUT DIMENSION A
8-01404-89	1.00-INCH
8-01404-91	1.50-INCHES
8-01404-93	.812-INCHES

NOTE

THESE TOOLS ARE USED FOR UNDERCUTTING CORE IN HONEYCOMB STRUCTURE 0.75-INCH TO 1.50-INCH. THE DEPTH AND SIZE OF THE UNDERCUT AREA WILL BE GIVEN IN THE APPLICABLE REPAIR INSTRUCTIONS.

PROCEDURE

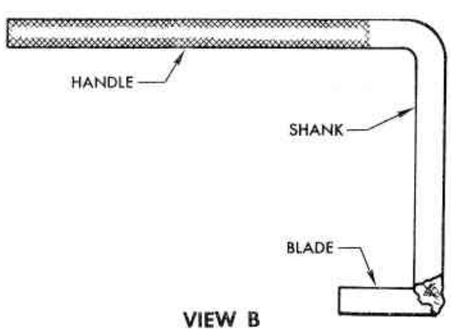
a. Determine the dimension of the area to be undercut from the repair instructions. See view A and Section A-A.
b. Select proper size tool. See core undercutting table.
c. Place core undercutting tool in routed hole.
d. Hold shank of tool in vertical position with one hand and move handle forward and aft with other hand forcing undercutter blade into core area. See View C.
e. Continue undercutting operation until all required core is cut away from around the routed hole. See Section B-B.

NOTE

REMOVE ALL CORE IN UNDERCUT AREA UP TO THE ORIGINAL ADHESIVE. SMOOTH OFF ALL ROUGH AND JAGGED EDGES.

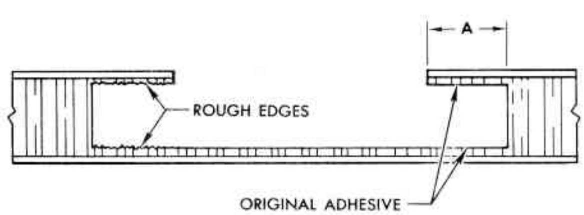
CAUTION

DO NOT REMOVE ORIGINAL ADHESIVE UNLESS REQUIRED BY REPAIR INSTRUCTIONS.



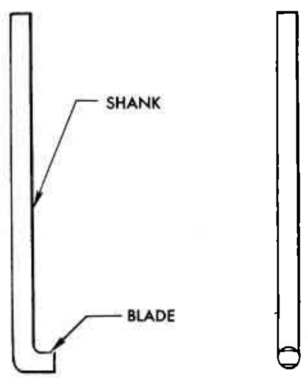
VIEW B

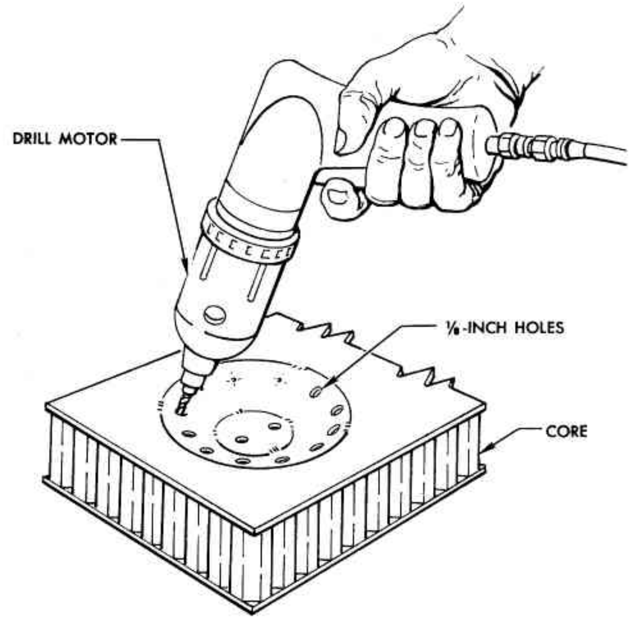
.06.03.325



SECTION B-B

Figure 10-21. Honeycomb Repairs — Hand-Operated Undercutter Assembly





NOTE

THESE TOOLS ARE USED IN CONJUNCTION WITH A ¼-INCH DRILL MOTOR. THEY ARE INSERTED IN ⅛-INCH HOLES TO UNDERCUT HONEYCOMB CORE PRIOR TO APPLICATION OF POTTING COMPOUNDS. THE -95 UNDERCUTS THE CORE TO ¼-INCH, AND THE -97 UNDERCUTS THE CORE TO ⅜-INCH.

PROCEDURE

a. Determine the size of the area to be undercut from repair instructions; select proper tool. Refer to note above.
b. Insert curved end of tool in ⅛-inch hole.
c. Raise drill motor to vertical position forcing cutter end into core.
d. Turn drill chuck by hand to free undercutter in core.
e. Turn on drill motor and undercut core by raising and lowering drill motor until tool contacts the skins.
f. Shut off drill motor and rotate chuck by hand tilting drill motor sufficiently to allow removal of the tool.

NOTE

DO NOT FORCE TOOL AGAINST SKIN.

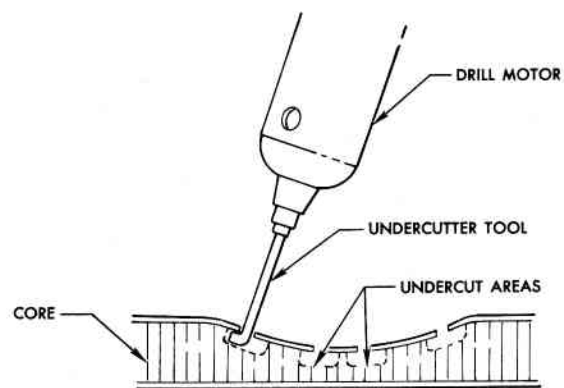


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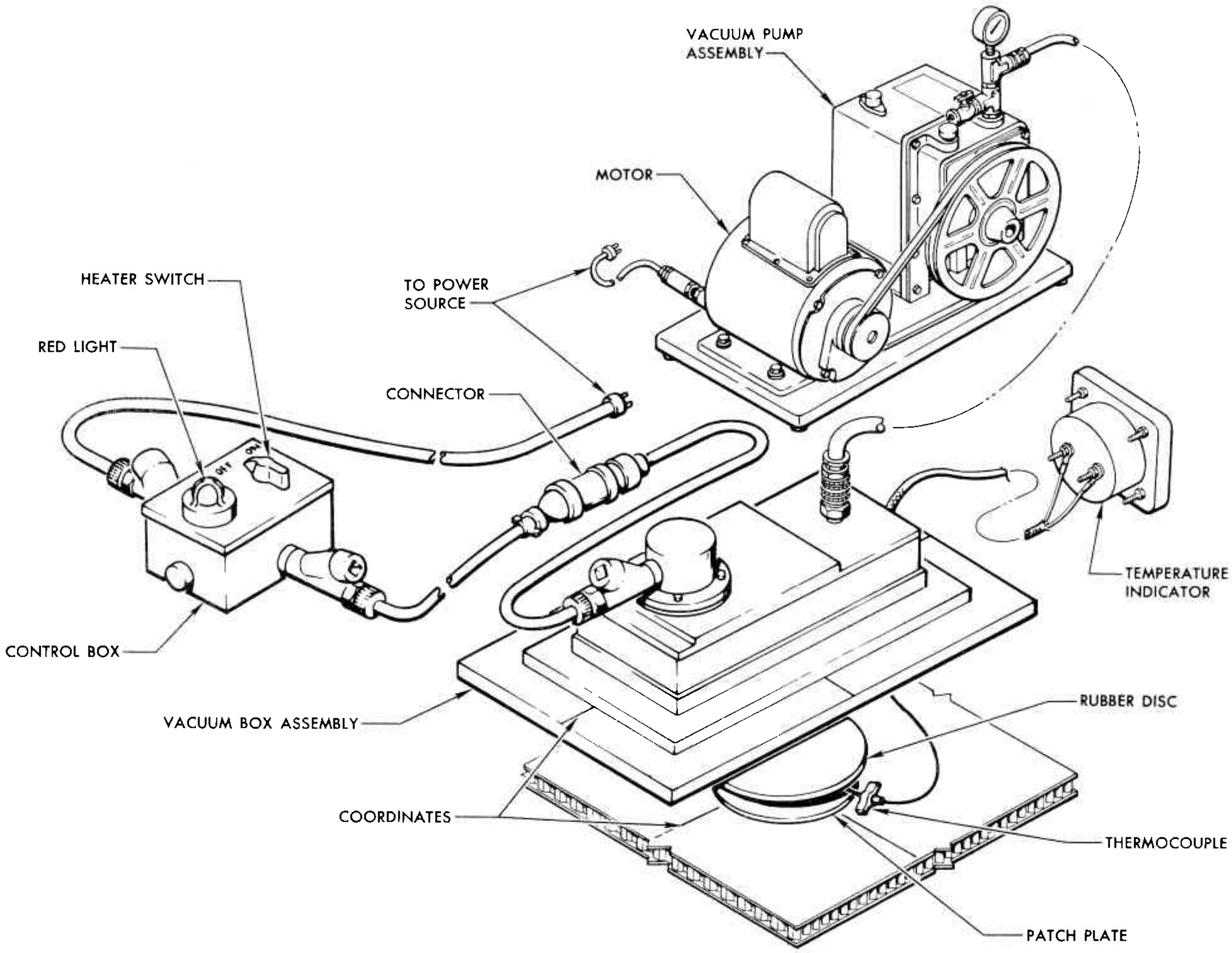
Figure 10-22. Honeycomb Repairs — Power-Operated Undercutter Assembly

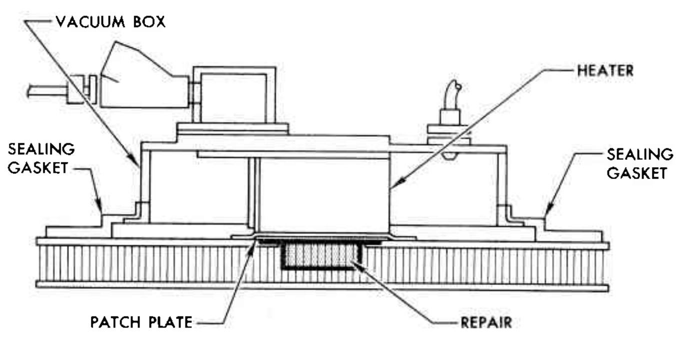


VIEW A
VACUUM BOX AND COMPONENTS

NOTE
THE -99 VACUUM BOX ASSEMBLY IS USED TO APPLY ATMOSPHERIC PRESSURE AND HEAT TO A REPAIR WHILE CURING THE ADHESIVE. IT IS USED IN CONJUNCTION WITH A VACUUM PUMP ASSEMBLY, A CONTROL BOX ASSEMBLY, AND A TEMPERATURE INDICATOR ASSEMBLY. THE CONTROL BOX ASSEMBLY CONTAINS A HEATER SWITCH, A RED LIGHT TO INDICATE WHEN THE HEATER IS IN OPERATION, A FUSE, AND AN ELECTRICAL LEAD. THE TEMPERATURE INDICATOR ASSEMBLY CONSISTS OF TWO THERMOCOUPLES AND TWO INDICATORS.

PROCEDURE

a. Place vacuum box heater switch in "OFF" position. See view A.
b. Plug vacuum box electrical lead into control box connector.
c. Plug control box electrical lead into 110-v ac power source.



VIEW B
VACUUM BOX CUTAWAY

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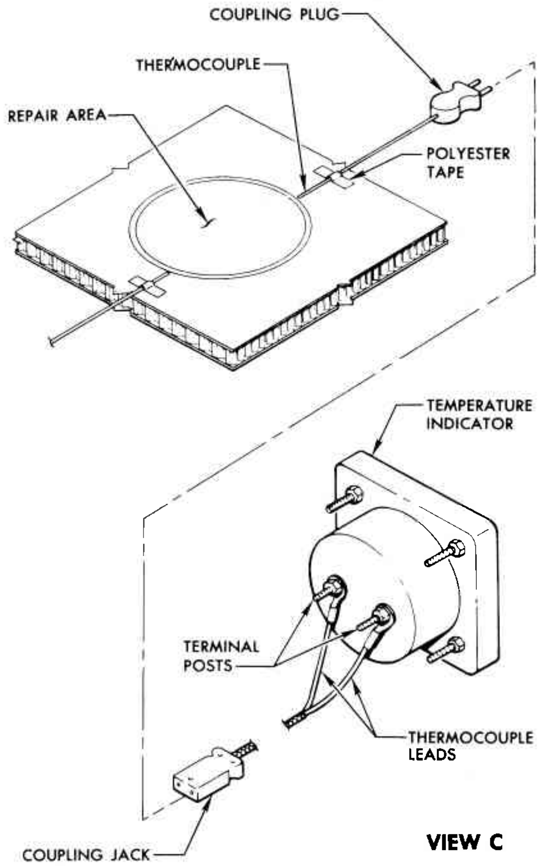
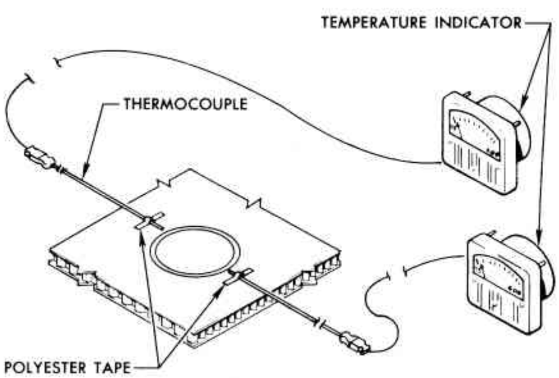
Figure 10-23. Honeycomb Repairs — Vacuum Box Assembly (Sheet 1 of 2)



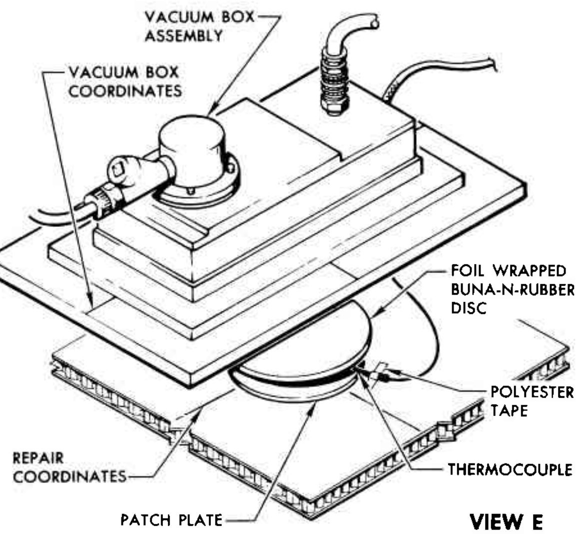
VIEW C

d. Attach thermocouple leads to terminal posts on back of temperature indicators as shown in view C.
e. Tape thermocouples down at edge of repair area using polyester tape #840. See view D.
f. Place temperature indicators in upright position for easy reading.
g. Connect coupling plug to coupling jack.



VIEW D

06.03.327-2



VIEW E

h. Center foil wrapped Buna-N-Rubber disc over repair patch plate as shown in view E.
i. Place vacuum box over rubber disc and align vacuum box coordinates with repair coordinates.
j. Attach vacuum box to vacuum pump with hose assembly.
k. Close vacuum pump valve by rotating valve handle clockwise.
l. Plug vacuum pump lead into 110-v ac power source.
m. Place vacuum pump switch in the "ON" position.
n. Allow vacuum pump gage to attain a reading of 20 to 30-inches of mercury vacuum.

NOTE

IF VACUUM GAGE DOES NOT ATTAIN A READING OF 20 INCHES OF MERCURY VACUUM, CHECK VACUUM BOX SEALING GASKET AND HOSE CONNECTIONS.

o. Place vacuum box heater switch in the "ON" position.

NOTE

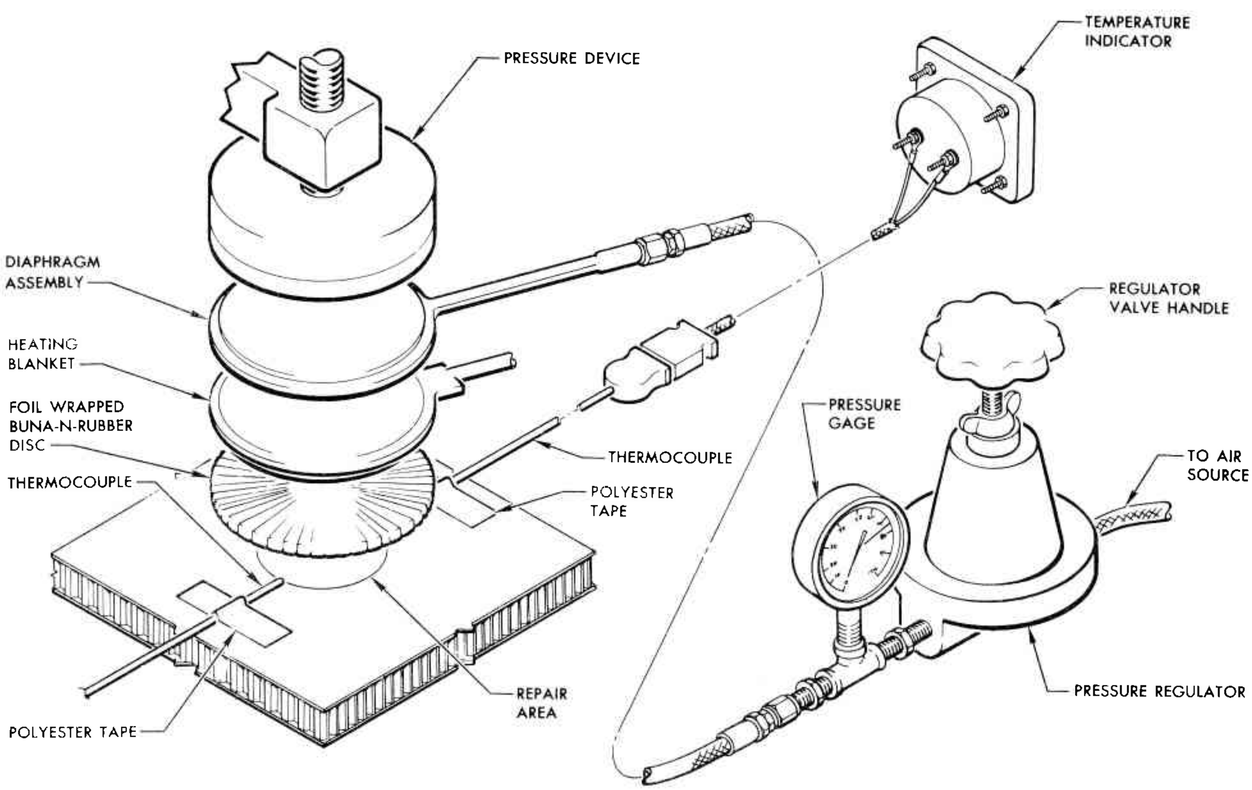
WHEN HEATER SWITCH IS IN "ON" POSITION, THE RED LIGHT ILLUMINATES. WHEN THE RED LIGHT GOES OFF, HEATER TEMPERATURE HAS REACHED 350°F. THE RED LIGHT WILL GO ON AND OFF THROUGHOUT THE CURE CYCLE.

p. Start timing the cure cycle when the red light goes off the first time.
q. Cure adhesives per repair instructions monitoring temperature indicators as specified.
r. Place heater and vacuum pump switches in the "OFF" position when cure cycle is completed.
s. Turn vacuum pump handle in counterclockwise direction to allow atmospheric pressure to enter vacuum box.
t. Remove vacuum box from the repair area.

WARNING

USE ASBESTOS GLOVES WHEN HANDLING HOT VACUUM BOX.

Figure 10-23. Honeycomb Repairs — Vacuum Box Assembly (Sheet 2 of 2)



NOTE
DIAPHRAGM ASSEMBLIES ARE USED IN CONJUNCTION WITH CLAMP ASSEMBLIES TO PROVIDE MEASURED PRESSURE TO REPAIR AREAS. THE -839 AND THE -841 AIR DIAPHRAGM ASSEMBLIES REQUIRE A 75 TO 100 PSI AIR SOURCE WITH PRESSURE BEING REGULATED BY AN AUTOMATIC REGULATOR.

PROCEDURE

a. Select diaphragm assembly large enough to amply cover repair.
b. Connect diaphragm assembly and regulator together as shown above.
c. With repair parts, rubber, and heating blanket in place, position diaphragm on the center of the repair area.
d. Install thermocouple assembly as described on figure 10-23.
e. Close regulator valve by turning knob counterclockwise.
f. Connect 75 to 100 psi air source to regulator.
g. Turn regulator knob slowly clockwise to obtain pressure required by repair instructions.

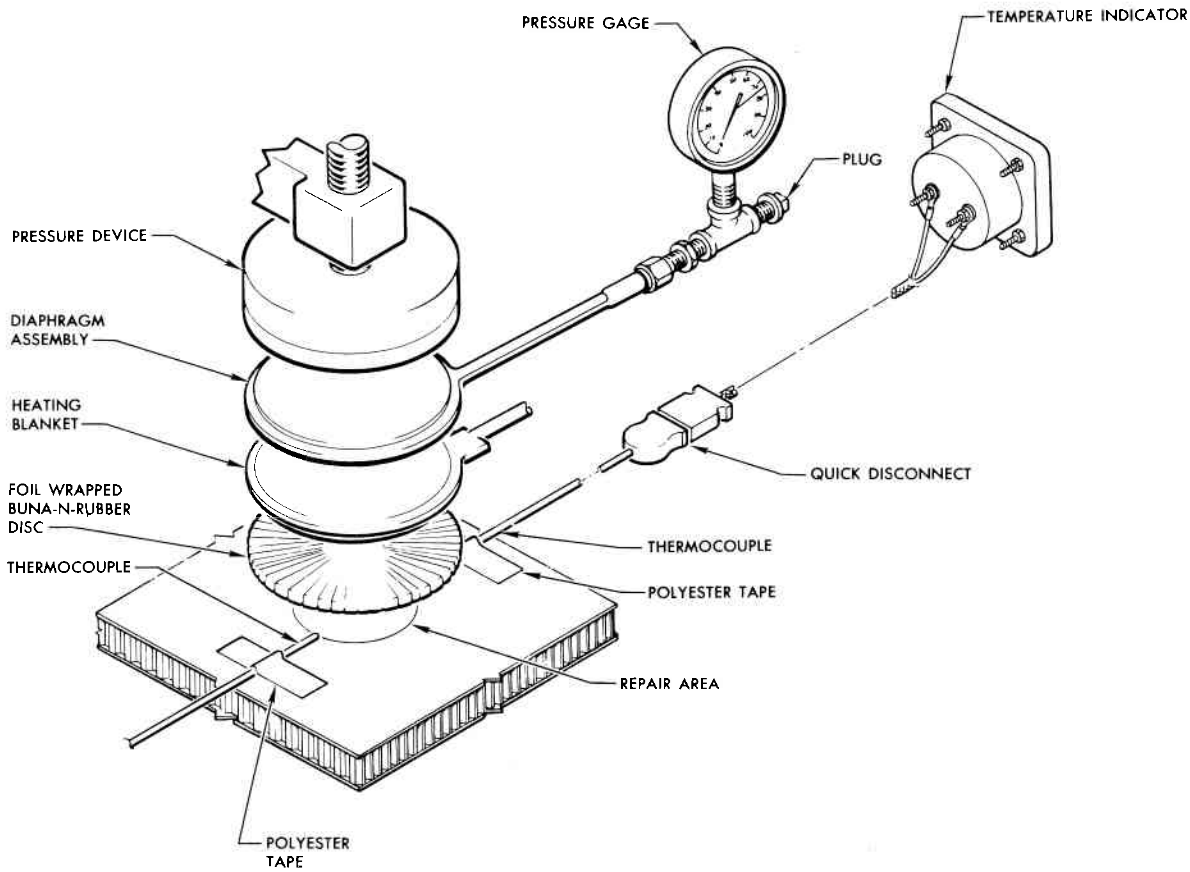
CAUTION

DO NOT EXCEED 75 PSI PRESSURE.

h. Cure adhesives per repair instructions regulating temperature and pressure indications as required.
i. Close regulator valve and disconnect air source from regulator upon completion of cure cycle.
j. Open regulator valve and bleed off diaphragm pressure.
k. Allow diaphragm assembly to cool and remove from repair area.

06.03.328

Figure 10-24. Honeycomb Repairs — Air Diaphragm Assembly



NOTES

1. DIAPHRAGM ASSEMBLIES ARE USED IN CONJUNCTION WITH CLAMP ASSEMBLIES TO PROVIDE MEASURED PRESSURE TO REPAIR AREAS. THE -843 AND THE -845 HYDRAULIC DIAPHRAGM ASSEMBLIES DO NOT FURNISH PRESSURE BUT DO REGISTER PRESSURE APPLIED BY CLAMP ADJUSTMENT.
2. PRIOR TO USE, IT IS NECESSARY TO PROPERLY FILL THE DIAPHRAGM WITH FLUID, SPECIFICATION MIL-H-8446B. TO ACCOMPLISH THIS, REMOVE PLUG AND FILL COMPLETELY FULL OF FLUID. REPLACE PLUG FINGER TIGHT. PLACE IN PLATEN PRESS WITH HEAT BLANKET AND HEAT TO 300°F. LOOSEN PLUG AND ALLOW EXCESS FLUID TO DRIP OFF UNTIL PRESSURE INDICATOR READS ZERO.

PROCEDURE

a. Select diaphragm assembly large enough to amply cover repair.
b. With repair parts, rubber, and heating blanket in place, position diaphragm on the center of the repair area.
c. Install thermocouple assembly as described on figure 10-23.
d. Position clamp on center of diaphragm and tighten finger tight.
e. Turn on the heat blanket.
f. Observe pressure gage and adjust clamp screw to maintain pressure required by repair instructions.

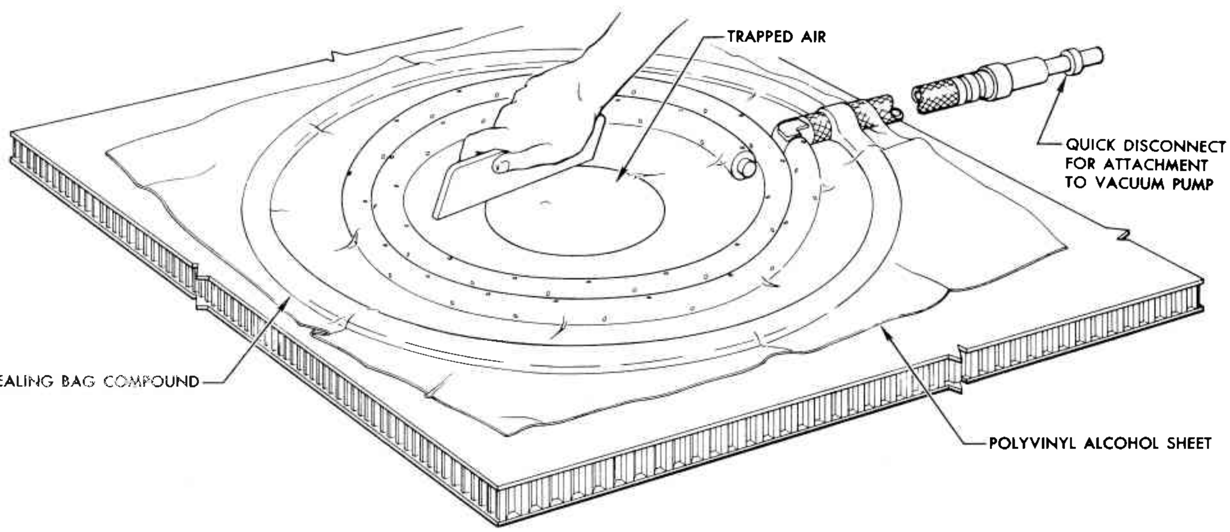
CAUTION

DO NOT EXCEED 75 PSI PRESSURE.

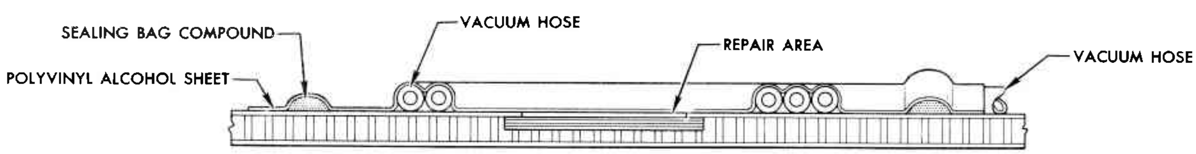
g. Cure adhesives per repair instructions regulating temperature and pressure as required.
h. Allow diaphragm assembly to cool and remove from repair area.

06.03.329

Figure 10-25. Honeycomb Repairs — Hydraulic Diaphragm Assembly



VIEW A



SECTION A-A

NOTE

THE -801 VACUUM BAG IS USED TO PROVIDE ATMOSPHERIC PRESSURE TO THE REPAIR AREA WHILE CURING ADHESIVES. THIS IS ACCOMPLISHED BY USE OF A VACUUM SOURCE TO REMOVE ATMOSPHERIC PRESSURE BETWEEN A POLYVINYL ALCOHOL SHEET AND THE REPAIR. VACUUM IS PROVIDED BY THE VACUUM PUMP USED WITH THE VACUUM BOX ASSEMBLY SHOWN IN FIGURE 10-23.

PROCEDURE

a. With repair ready for bonding per repair instructions, place a ¾-inch strip of 3985 sealing bag compound (W. P. Fuller Co., Los Angleles, Calif.) in a two-foot circle around the repair area. Pack in around the vacuum hose outlet.
b. Spread a polyvinyl alcohol sheet over the repair and press against the sealing bag compound.
c. Attach vacuum pump hose to sealing bag quick disconnect.
d. Close vacuum pump valve by rotating valve handle clockwise.
e. Plug vacuum pump lead into 110-v ac power source.
f. Place vacuum pump switch in the "ON" position.
g. Allow pump to attain a reading of 26-inches of mercury vacuum.

NOTE

IF VACUUM GAGE DOES NOT ATTAIN A READING OF 26-INCHES OF MERCURY VACUUM, CHECK SEALING AND CONNECTIONS.

h. Remove air trapped between repair and polyvinyl alcohol sheet by smoothing sheet and forcing air to the edge as shown in view A.
i. Place heating blanket over repair area.

NOTE

SILICONE RUBBER HEATING BLANKETS MAY BE PROCURED FROM SAFEWAY HEAT ELEMENTS CO., MIDDLETOWN, CONN. TEMPERATURE OF THE BLANKET MUST BE MONITORED BY THERMOCOUPLES WITH TEMPERATURE INDICATORS AND CONTROLLED BY A MODEL 2PE136 VARIABLE TRANSFORMER (SUPERIOR ELECTRIC CO., BRISTOL, CONN.). FOR INFORMATION CONCERNING THERMOCOUPLE INSTALLATION, SEE FIGURE 10-23.

j. Place asbestos pad on heating blanket and tape in place.
k. Cure adhesives per repair instructions monitoring temperature indicators as specified.
l. Turn off heating blanket upon completion of cure cycle.
m. Turn vacuum pump handle in counterclockwise direction to allow atmospheric pressure to enter the vacuum bag.
n. After the assembly has cooled, remove the vacuum bag assembly from the repair.

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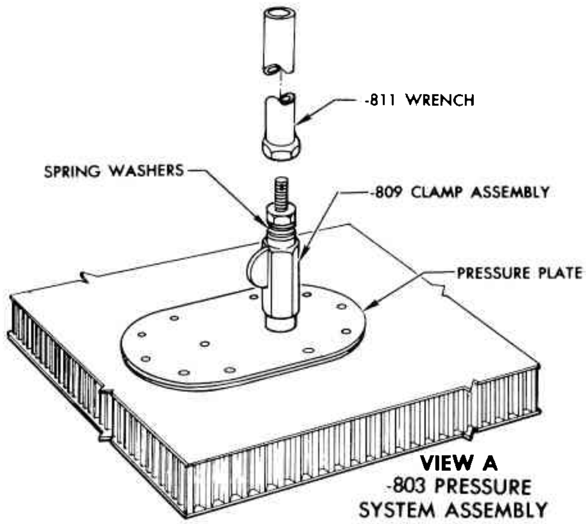
Figure 10-26. Honeycomb Repairs — Vacuum Bag Assembly



VIEW A
-803 PRESSURE SYSTEM ASSEMBLY

NOTE

THE SPRING WASHERS, STACKED AS SHOWN, APPLY A LOAD OF APPROXIMATELY 70 POUNDS WHEN FULLY COMPRESSED. A LOAD OF 50-60 POUNDS PER CLAMP IS SUFFICIENT IN REPAIR WORK. THEREFORE, WASHER DEFLECTION WITH AN OBSERVABLE GAP OF APPROXIMATELY 0.015-INCH IS REQUIRED.

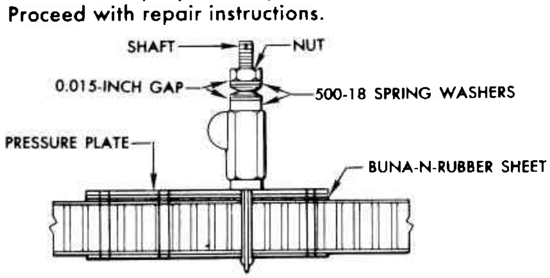
PROCEDURE

a. With repair patch, rubber sheet and pressure plate in place, drill out the 0.093 pilot holes to 0.1285-inch diameter with a No. 30 drill bit.
b. Screw the nut, on -809 clamp assembly, to top threads. Push down on shaft until nut is pushed completely down on the washers.
c. Insert the -809 clamp assembly through pressure plate hole, forcing the jaws through the rubber sheet. Push the -809 clamp assembly until it rests on the pressure plate.
d. Pull the nut until the jaws in -809 clamp engage.
e. Screw the nut down to the washers finger tight.
f. Install additional -809 clamps approximately one inch apart.
g. Tighten the nut on -809 clamps with -811 wrench until gap at edge of spring washers is approximately 0.015 inch.

CAUTION

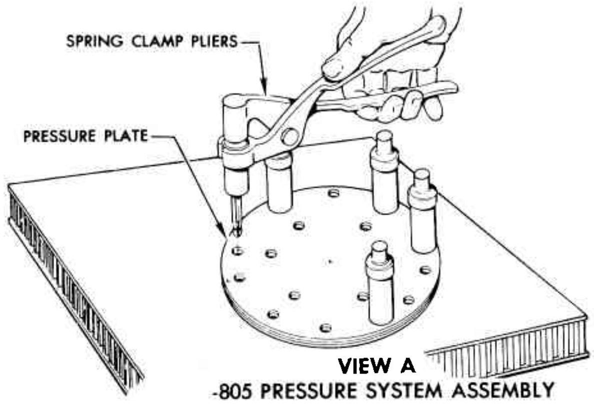
DO NOT OVERTIGHTEN NUT AS UNDERSKIN MAY BE DAMAGED.

h. After repair bonding is accomplished and parts have cooled, screw the nut to the top of threads. Push shaft down until the nut is down on the washers.
i. Twist the clamp to break any bond attachment.
j. Remove clamps, pressure plates and rubber sheet. Proceed with repair instructions.



VIEW B
-803 PRESSURE SYSTEM ASSEMBLY

.06.03.331



VIEW A
-805 PRESSURE SYSTEM ASSEMBLY

NOTE

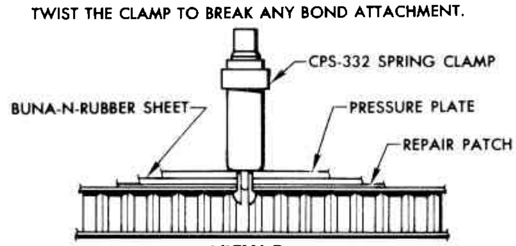
CPS-332 SPRING CLAMP HAS A SPRING LOAD OF 30 POUNDS AT 1/16-INCH EXTENSION TO 45 POUNDS AT 3/16-INCH EXTENSION. SPACE SPRING CLAMPS AT APPROXIMATELY 1-INCH APART FOR ADEQUATE PRESSURE.

PROCEDURE

a. Ascertain that repair patch, rubber sheet and pressure plate are in place.
b. Using the clamp pliers (Model 300), extend the clamp jaws and insert in pressure plate hole. Push the clamp jaws through the rubber sheet and into hole in the structure.
c. Release the pliers to engage the clamp.
d. Install spring clamps in all pressure plate holes and proceed with repair bonding instructions.
e. After repair bonding is accomplished and parts have cooled, remove all clamps, pressure plate, and rubber sheet.

NOTE

TWIST THE CLAMP TO BREAK ANY BOND ATTACHMENT.



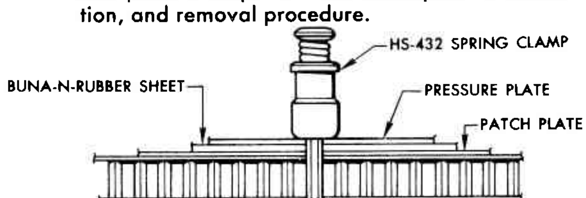
VIEW B
-805 PRESSURE SYSTEM ASSEMBLY

NOTE

HS-432 SPRING CLAMP HAS A SPRING LOAD OF 45 POUNDS AT 3/16-INCH EXTENSION TO 65 POUNDS AT 1/2-INCH EXTENSION. SPACE SPRING CLAMPS AT APPROXIMATELY 1-INCH APART.

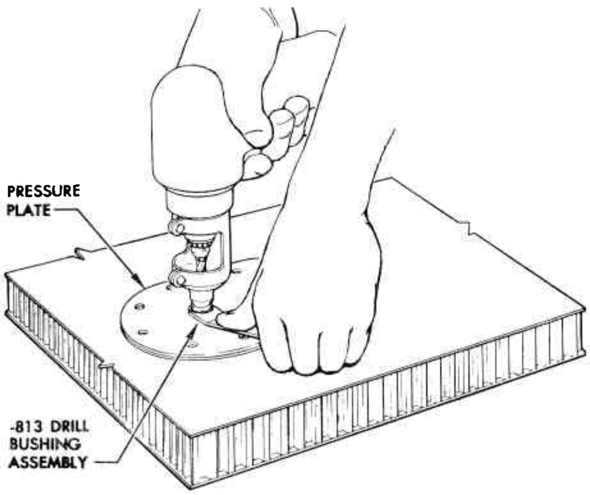
PROCEDURE

a. With repair patch, rubber sheet and pressure plate in place, drill out the 0.093 pilot holes to 0.1285-inch diameter with a No. 30 drill bit.
b. See Views A and B in -805 pressure system assembly, and perform steps "b" and subsequent in installation, and removal procedure.



VIEW A
-807 PRESSURE SYSTEM ASSEMBLY

Figure 10-27. Honeycomb Repairs — Pressure Systems



NOTES

1. THESE PLATES ARE USED WITH -813 DRILL BUSHING ASSEMBLY AND AN 0.093 DRILL TO PILOT DRILL REPAIR PARTS. THEY ARE ALSO USED WITH SPRING CLAMPS AS PRESSURE DISTRIBUTION PLATES FOR BONDING REPAIRS WHEN ASSEMBLED AS SHOWN ON FIGURE 10-27 FOR THE -803, -805, AND -807 PRESSURE SYSTEMS.
2. THE PRESSURE PLATES ARE MADE FROM 2024 ALUMINUM, SPECIFICATION QQ-A-362. AFTER FABRICATING THE PLATES, STENCIL IN 3/8 OR 1/2 INCH NUMERALS LOCATED APPROXIMATELY AS SHOWN TO IDENTIFY THE PLATE.
3. PRESSURE PLATES MUST HAVE ONE SIDE FLAT AS REQUIRED ON ORIGINAL SHEET MATERIAL. LIGHT SANDING TO OBTAIN A FLAT SURFACE IS ACCEPTABLE.
4. PRESSURE PLATES -315 THRU -321 ARE SIMILAR TO THOSE SHOWN. FOR DETAILED INFORMATION, SEE PRESSURE PLATE TABLE ON SHEET 2.

PROCEDURE

a. Coordinate #60 holes on the axes with repair. Coordinate axes to properly locate pattern in the structure.
b. Coordinate patch plate and pressure plate outside diameter to properly locate patch plate holes.
c. The -827 and -829 pressure plates are used with flush patches of smaller diameter. Locate the center of the patch, mark it with a grease pencil, and locate holes by sight through the center hole in the pressure plate.

NOTE

THE 0.125 INCH CENTER HOLE IN THE PRESSURE PLATE IS NOT FOR DRILLING PURPOSES.

d. If hole pattern in doubler and skin has been located and drilled, a more precise patch location is obtained by installing doubler and patch in repair location, placing template on top and securing with spring clamps. Drill the inside row of holes through patch and doubler. If holes through doubler and skin have not been drilled, template should be taped in place to prevent shifting during the drilling of holes.
e. Install patch and pressure plate assembly in conjunction with Buna-N-Rubber sheet and spring clamps. See figure 10-27 for the use of pressure plates.

NOTE

BUNA-N-RUBBER SHEET (UNCURED—1/16 INCH THICK) CAN BE OBTAINED FROM KIRKHILL RUBBER CO., 300 EAST CYPRUS STREET, BREA, CALIFORNIA. USE PRESSURE PLATE AS A TEMPLATE FOR CUTTING THE BUNA-N-RUBBER SHEET TO SIZE.

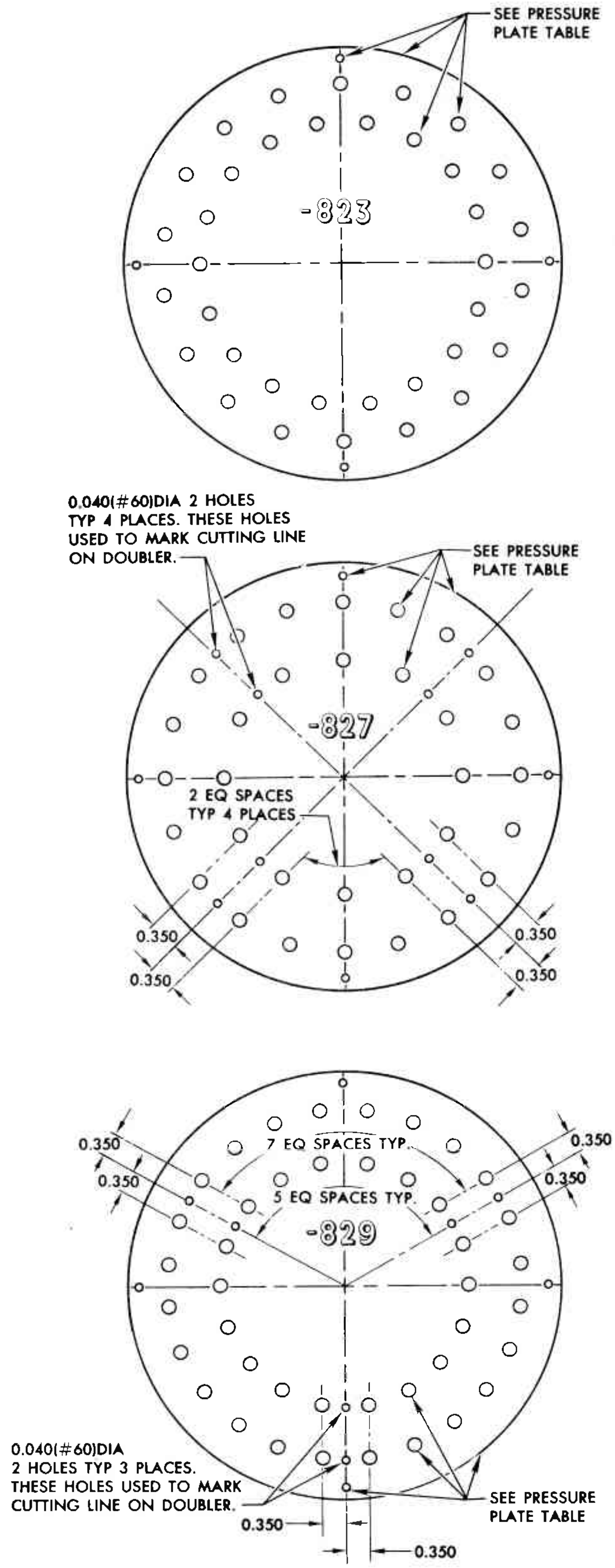


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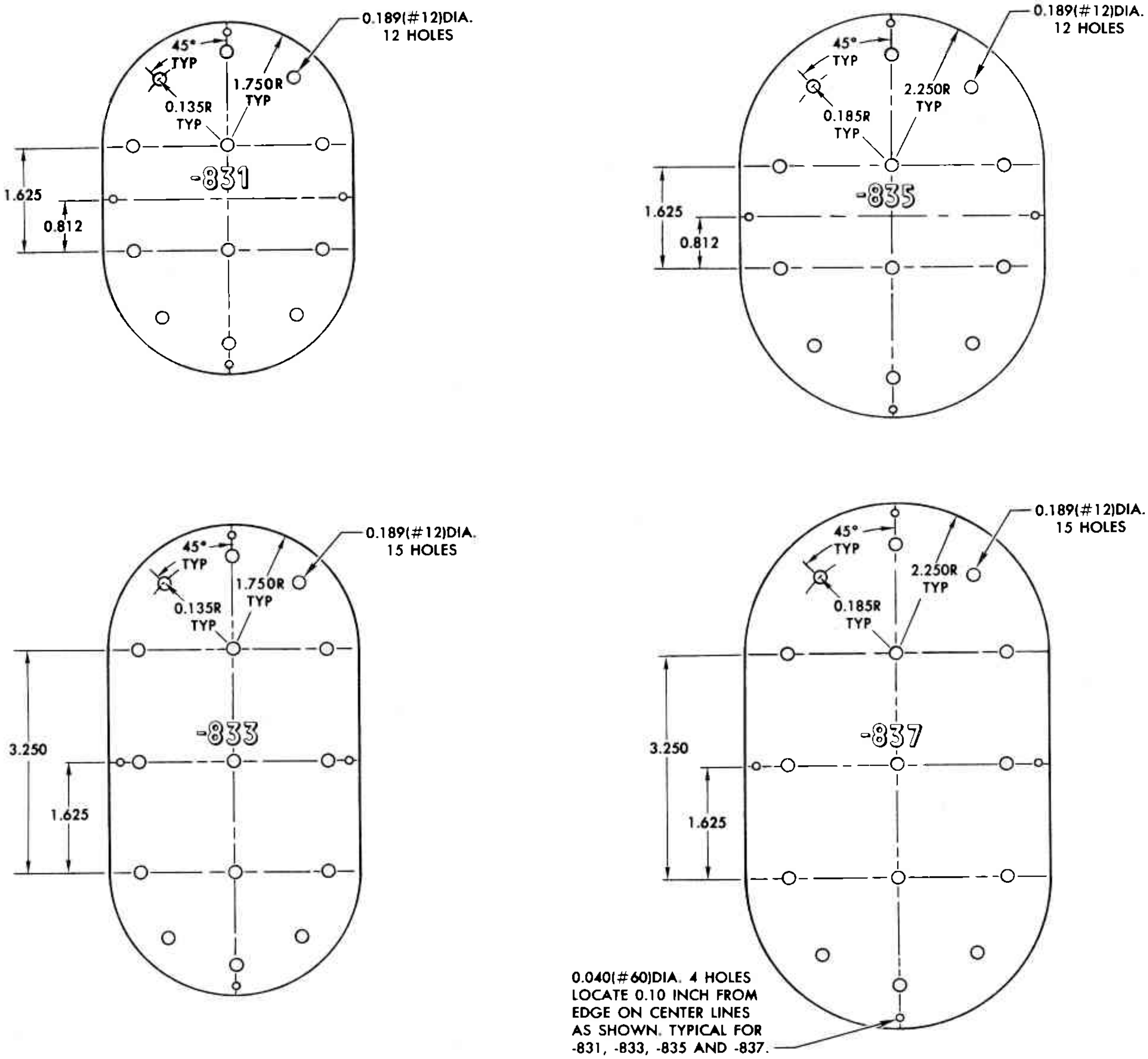
Figure 10-28. Honeycomb Repairs — Pressure Plates (Sheet 1 of 2)



PRESSURE PLATE TABLE

THIS TABLE GIVES DATA NOT OTHERWISE SHOWN. NUMBER OF SPACES EQUAL NUMBER OF NO. 12 HOLES REQUIRED. FOUR NO. 60 HOLES REQUIRED PER PART. LOCATE AND STAGGER AS SHOWN.

PLATE DASH NO.	PLATE OUTSIDE DIA.	0.189 (NO. 12) DIAMETER HOLES				0.040 (NO. 60) 2 HOLES DIA.	CENTER HOLE SIZE
		OUTSIDE ROW		INSIDE ROW			
		DIA.	SPACES	DIA.	SPACES		
815	3.200	2.600	12	1.600	6	3.00	.125
817	4.200	3.600	12	2.600	12	4.00	.125
819	5.200	4.600	18	3.600	18	5.00	.125
821	6.200	5.600	18	4.600	18	6.00	.125
823	7.200	6.600	18	5.600	18	7.00	.125
825	8.200	7.600	18	6.600	18	8.00	.125
827	4.500	3.750	20 HOLES	2.250	12 HOLES	4.30	.125
829	6.000	5.250	24 HOLES	3.750	18 HOLES	5.80	.125

NOTE:
PRESSURE PLATE (TEMPLATES) REFER TO SHEET 1 FOR METHOD OF FABRICATION.

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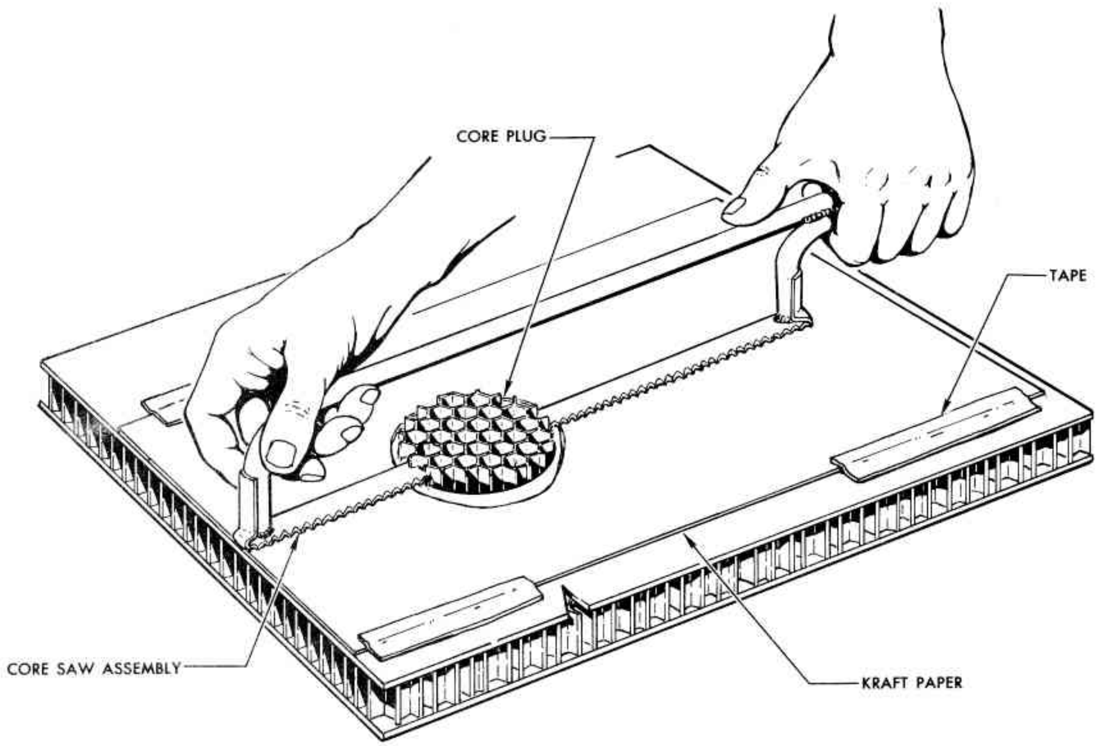
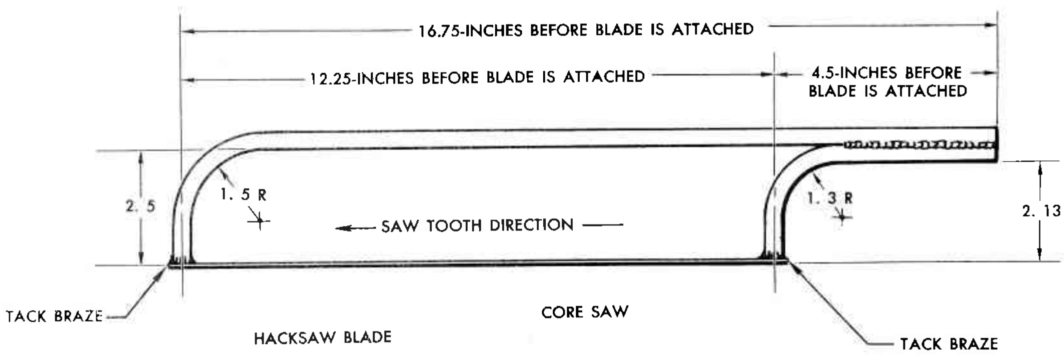
Figure 10-28. Honeycomb Repairs — Pressure Plates (Sheet 2 of 2)





NOTE
THE CORE SAW IS USED FOR ROUGH CUTTING THE CORE PLUGS AND ALUMINUM TUBES NEARLY FLUSH WITH SURFACE OF SKIN WHEN PERFORMING HONEYCOMB REPAIRS WHERE HOLES HAVE BEEN ROUTED OUT. FABRICATE CORE SAW AS SHOWN USING 3/8-INCH O.D. STEEL ROD FOR CORE SAW FRAME AND HACKSAW BLADE WITH VERY FINE TEETH. BRAZE THE FRAME TOGETHER TO DIMENSION SHOWN. SPRING THE LEGS TOGETHER TO CENTER THEM OVER HOLES IN BLADE. TACK BRAZE BLADE TO FRAME AS SHOWN.

PROCEDURE

a. Apply 2 layers of #850 tape around ends of blade. and up into the frame legs. Masking tape can be used in place of #850 tape.
b. Cut a hole, the size of core plug, on the center of a piece of Kraft paper approximately 18 inches.
c. Place the Kraft paper over the core plug, and attach the paper with masking tape to the surface of skin to be repaired.

NOTE
THE #850 TAPE AND MASKING TAPE ARE USED TO PROTECT THE SURFACE SKIN FROM BEING SCRATCHED.

d. Cut off the core plug and tube as close to the skin surface as possible being careful not to scratch the skin.
e. Remove Kraft paper. Finish machining core plug with router assembly. See figure 10-14 for instructions on use of router assembly.

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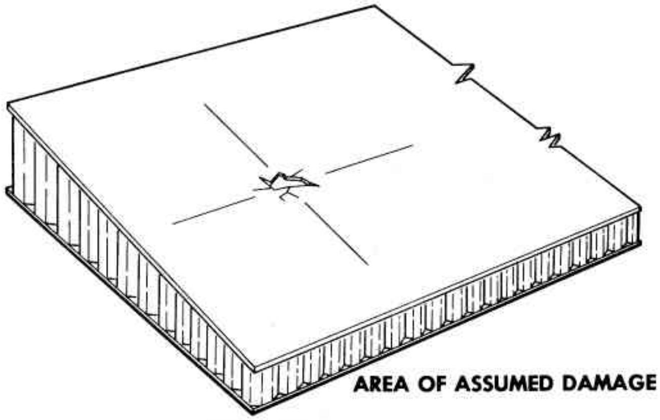
Figure 10-29. Honeycomb Repairs — Core Saw



REPAIR NO. 1

NOTE

REPAIR NO. 1 IS FOR REPAIR OF HOLES THROUGH BOTH SIDES OF PANEL USING E3045 CORE FILLER TO FILL ROUTED HOLE. MAXIMUM ALLOWABLE HOLE DIAMETER IS 6 INCHES.

PROCEDURE

a. Determine diameter of damage area with damage locating template; see figure 10-11.
b. Select proper patch plates. See figure 10-12.
c. Prepare and clean skin surfaces. Use steps 4 through 7, figure 10-13.
d. Route out damaged area to a round, preselected size; see View A. Use -3 router assembly and -53 router turntable assembly; or hole saws and hole saw guide templates. Use these tools in accordance with instructions contained in this section.
e. Clean upper and lower patch plate bonding area on skin. Use steps 8, 9, 10, and 11, figure 10-12.

CAUTION

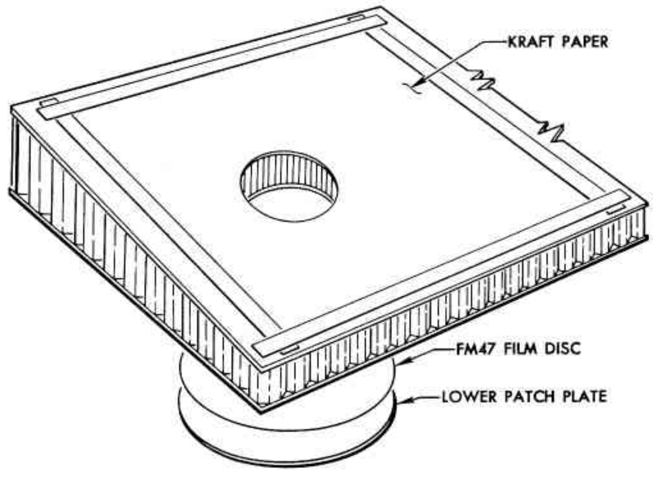
HONEYCOMB CORE MUST BE KEPT DRY AND CLEAN. ADHESIVES WILL NOT BOND TO WET SURFACES.

f. Brush thin coat of liquid primer FM47 on patch plate area. (Both upper and lower skin.)
g. Tape Kraft paper over repair area. Cut hole through paper around routed hole; see View A.
h. Cut a disc of film FM47 1/16-inch larger than lower patch plate.

CAUTION

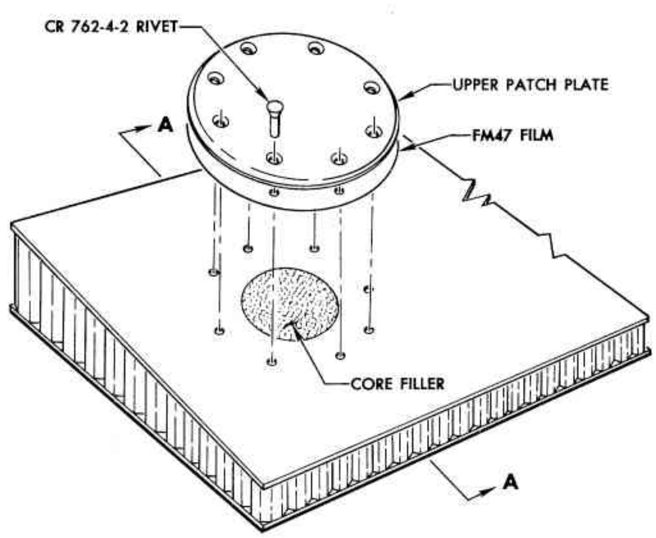
USE CLEAN WHITE GLOVES TO HANDLE FILM TO PREVENT TRANSFER OF MOISTURE OR GREASE FROM HANDS.

i. Install film disc FM47 and patch plate to lower repair area. Hold in place with tape #850; see View A.
j. Mix core filler E3045 in accordance with Table 10-1 and fill repair 1/32-inch above skin. Work in with a spatula or tongue blade until all air bubbles have been removed and core filler is solid and smooth.
k. Cure core filler. See Table 10-1. Sand down to contour of existing skin; see View B.
l. Remove Kraft paper and install upper patch plate over film disc FM47; see View B.
m. Drill rivet pattern for patch plates selected from figure 10-12.
n. Bond and cure repair parts to skin; see Section A-A. Use heat lamp, pressure plate, and -805 pressure system. See figure 10-27 and 10-28 for pressure plate and pressure system instructions. Refer to Table 10-1 for FM47 film curing instructions.
o. Install rivets in patch plates and mill rivets flush in accordance with figure 10-12. See Section A-A for cross-section of completed repair.

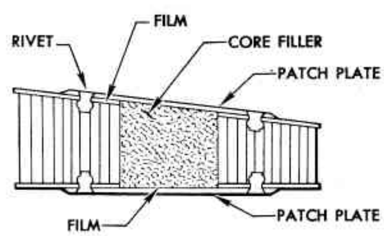


VIEW A

ROUT OUT DAMAGE.
TAPE KRAFT PAPER,
ADD LOWER DISC
AND PATCH PLATE.



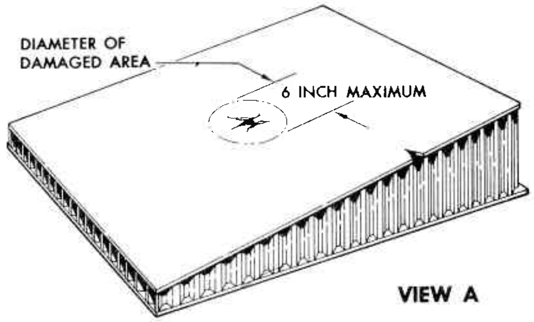
VIEW B



SECTION A-A
COMPLETED REPAIR

.06.03.296B

Figure 10-30. Honeycomb Repair No. 1 — Hole



VIEW A

REPAIR NO. 2

NOTE

REPAIR NO. 2 IS FOR REPAIR OF HOLES THROUGH BOTH SIDES OF PANEL. USE A ⅛-INCH CELL HEX CELL PERFORATED ALUMINUM HONEYCOMB CORE PLUG AND AN ALUMINUM TUBE SHEET TO FILL ROUTED HOLE. MAXIMUM ALLOWABLE HOLE DIAMETER IS 6 INCHES.

PROCEDURE

a. Determine diameter of damaged area with damage locating template; see figure 10-11.
b. Select proper patch plates. See figure 10-12.
c. Prepare and clean skin surfaces. Use steps 4 through 7 of figure 10-13.
d. Route out damaged area to a round, preselected size. Use -3 router assembly and -53 router turntable assembly; see figure 10-14 and 10-19 for operating instructions.
e. Clean upper and lower patch plate bonding area on skin. Use steps 8, 9, 10, and 11 of figure 13.

CAUTION

HONEYCOMB CORE MUST BE KEPT DRY AND CLEAN. ADHESIVES WILL NOT BOND TO WET SURFACES.

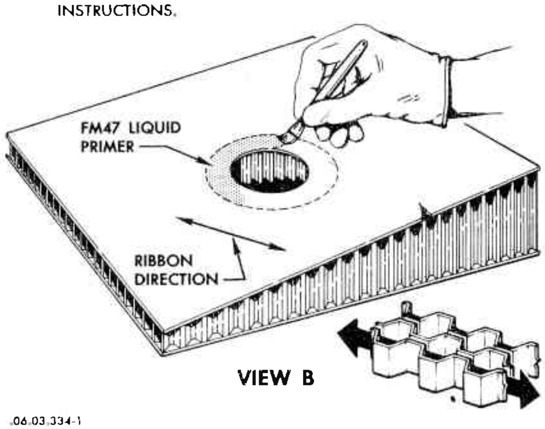
f. Brush a thin coat of FM47 liquid primer on patch plate area. (Both upper and lower skins.)
g. On skin near routed hole, mark ribbon direction.

CAUTION

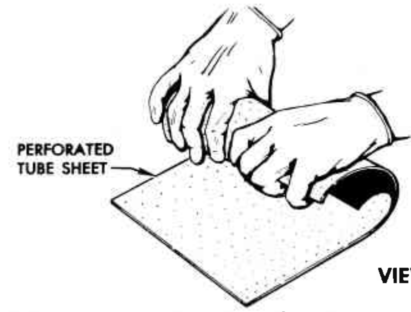
USE CLEAN WHITE GLOVES TO HANDLE ALL REPAIR PARTS TO PREVENT TRANSFER OF MOISTURE OR GREASE FROM HANDS.

NOTE

FOR 1 INCH DIAMETER REPAIR, OMIT CORE PLUG INSTALLATION. PERFORM THE FOLLOWING REPAIR STEPS USING ONLY A TUBE SHEET OF THE CORRECT SIZE. USE E3045 CORE FILL IN PLACE OF THE CORE PLUG. REFER TO TABLE 10-1 FOR CORE FILL MIXING, APPLICATION, AND CURING INSTRUCTIONS.

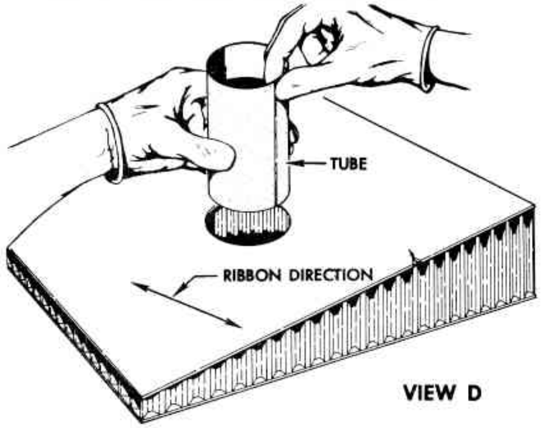


VIEW B



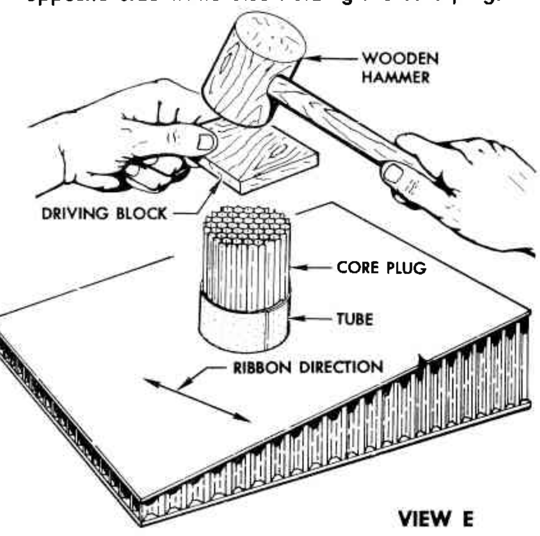
VIEW C

h. Select correct tube sheet for diameter core plug being used from the core plug and tube sheet table. Hand roll tube sheet into a cylindrical form.



VIEW D

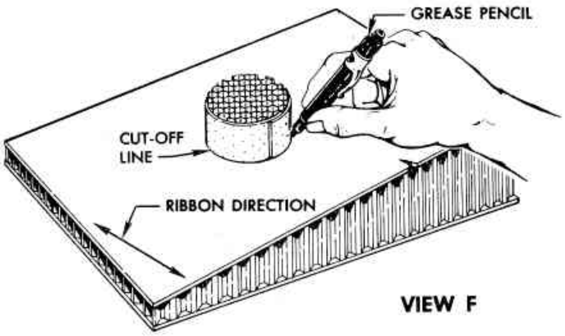
i. Cut tube sheet to length to obtain approximately ½-inch overlap when placed in routed hole. Correct length will be 3.14 x hole diameter +0.50-inch.
j. Push tube sheet into routed hole and slightly through the far side of skin.
k. Place proper size core plug inside tube sheet. Core plug ribbon shall run parallel to mark placed on skin in step "g."
l. With a wooden driving block and hammer, drive core plug into the tube sheet. Have an assistant hold a block or plate against tube sheet on the opposite side while also holding the core plug.



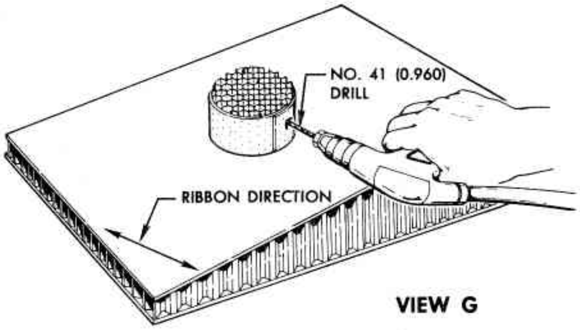
VIEW E

.06.03.334-1

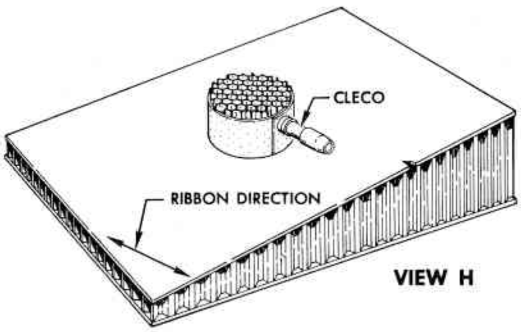
Figure 10-31. Honeycomb Repair No. 2—Hole (Sheet 1 of 3)

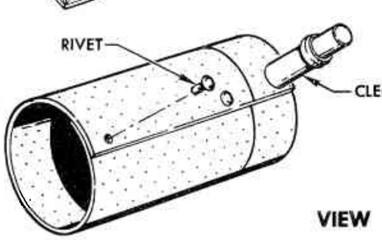


m. Scribe mark a cut-off line on tube sheet at skin levels. With a grease pencil, mark core and tube sheet positions.

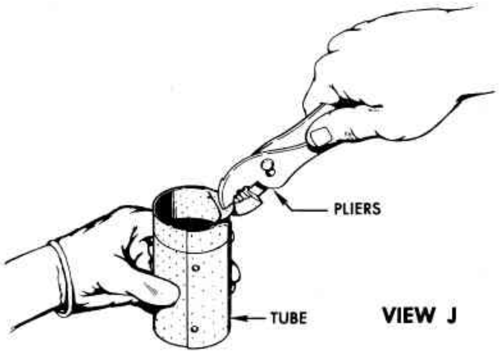


n. Drill a cleco hole and install cleco as shown to hold tube sheet to proper size. Use a No. 41 (0.960) drill.
o. Remove tube sheet and core plug from panel and fasten tube sheet in a cylindrical form with spring clamps.

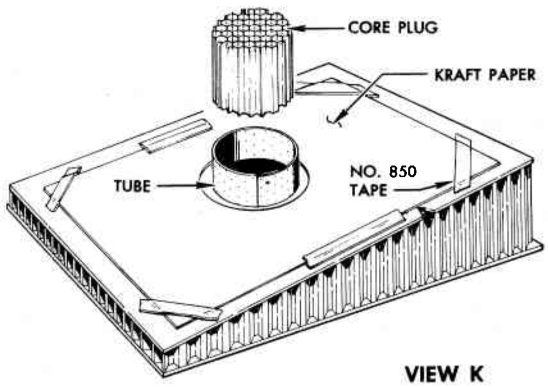




p. Drill at least two holes with a No. 50 (0.70) drill and install 1/16-inch diameter aluminum brazier head rivets to hold tube sheet in shape.



q. Bend tube sheet with pliers as shown in View J for tube sheet and core plug final installations.



r. Tape Kraft paper over repair area. Cut hole through paper around routed hole; see View A.
s. Apply 338 adhesive to inside of routed hole. Refer to Table 10-1 for adhesive mixing instructions.
t. Place tube sheet in proper position in routed hole. With a tongue blade, work adhesive out through tube sheet perforations to fill space outside tube.
u. Apply 338 adhesive to core plug and place core plug in proper position in panel. Repeat steps "k" and "l."
v. Cure 388 adhesive with heat lamps in accordance with Table 10-1.
w. Cut away excess core and tube sheet 1/32-inch to 1/16-inch above skin levels with -861 core saw. See figure 10-29 for core saw instructions.
x. Machine core and tube sheet 0.004-inch to 0.005-inch above skins, see View L. Use -3 router assembly and a router guide template for oversized hole. See figure 10-14 for -3 router assembly instructions, and figure 10-15 for router guide template instructions.

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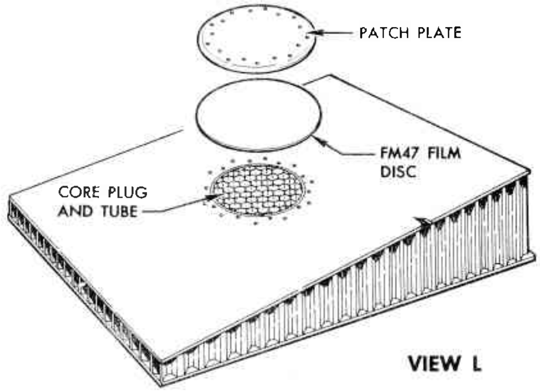
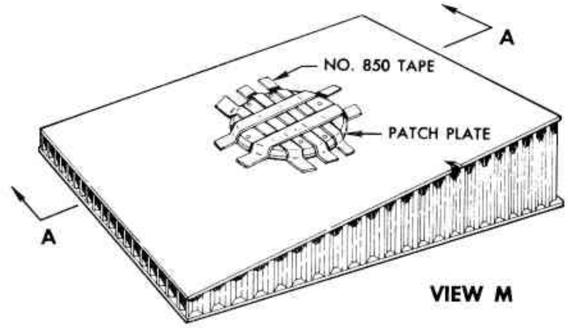
Figure 10-31. Honeycomb Repair No. 2 — Hole (Sheet 2 of 3)





y. Remove Kraft paper. If repair area is contaminated, clean with aliphatic naphtha, Federal Specification TT-N-95, and touch-up any bare spots with FM47 liquid primer.

z. Center patch plates selected from figure 10-12 and pilot-drill rivet patterns. Drill pilot holes through patch plate and skin only.

aa. Cut disc of FM47 film 1/16-inch larger in diameter than patch plates.

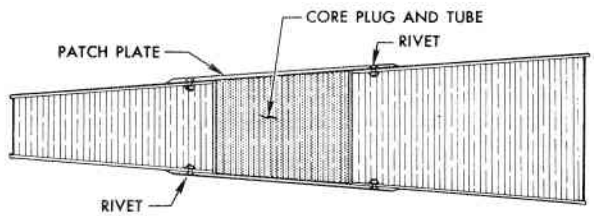
ab. Center FM47 film discs on repair. Discs may be held in place with CPS-332 spring clamps.

ac. Apply 1/16-inch coat of EC1469 adhesive with a brush over areas of FM47 film discs which cover core only.

ad. Turn discs over and center EC1469 adhesive area over core. Install patch plates and hold in place with #850 tape.

ae. Bond and cure repair parts to skin. Use heat lamp, pressure plate, and -805 pressure system. See figures 10-27 and 10-28 for pressure system instructions. Refer to Table 10-1 for FM47 film and EC1469 adhesive curing instructions.

af. Drill rivet holes of correct diameter and install rivets to patch plates. Install and mill rivets flush in accordance with figure 10-12. See Section A-A for cross-section of completed repair.



SECTION A-A

CORE PLUG AND TUBE SHEET TABLE		
DAMAGE DIAMETER	CORE PLUG USED (SEE NOTE 1)	TUBE SHEET USED (SEE NOTE 2)
1.000	NONE (USE E 3045 CORE FILLER	0.035 WALL X 1.0 DIAMETER X 5.0 WIDTH (SEE NOTE 3)
2.000	2.0 DIAMETER X 4.0 LENGTH	.012 GAGE X 5.0 WIDTH X 6.7 LENGTH
3.000	3.0 DIAMETER X 4.0 LENGTH	.012 GAGE X 5.0 WIDTH X 9.8 LENGTH
4.000	4.0 DIAMETER X 4.0 LENGTH	.012 GAGE X 5.0 WIDTH X 13.0 LENGTH
5.000	5.0 DIAMETER X 4.0 LENGTH	.012 GAGE X 5.0 WIDTH X 16.2 LENGTH
6.000	6.0 DIAMETER X 4.0 LENGTH	.012 GAGE X 5.0 WIDTH X 19.4 LENGTH

NOTES:
1. CORE PLUG MATERIAL IS PERFORATED 1/8-INCH HEX CELL, 0.002 FOIL HONEYCOMB, SPECIFICATION MIL-C-7438C.
2. TUBE SHEET MATERIAL IS 0.12 GAGE 2024-T3 PERFORATED ALUMINUM ALLOY.
3. TUBE SHEET MATERIAL FOR 1.0-INCH DIAMETER DAMAGE IS 0.035 WALL 5052 ALUMINUM ALLOY.

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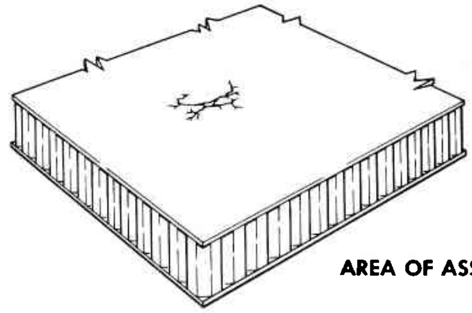
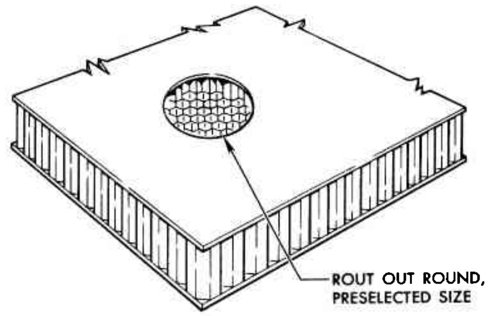
Figure 10-31. Honeycomb Repair No. 2 — Hole (Sheet 3 of 3)





VIEW A

REPAIR NO. 3

NOTE

REPAIR NO. 3 IS FOR REPAIR OF HOLES THROUGH ONE SIDE OF PANEL USING E3045 CORE FILLER TO FILL ROUTED HOLE. MAXIMUM ALLOWABLE CORE HOLE DIAMETER IS 6 INCHES.

PROCEDURE

a. Determine center and diameter of damage; see figure 10-11.
b. Fabricate patch plate; see figure 10-12 for details.
c. Prepare and clean damage area. Perform all steps in figure 10-13.
d. Brush liquid primer FM47 over patch plate attachment area of skin.
e. Rout out damaged area to a round; preselected size; see View A. Use -3 router assembly and proper router guide template. See figure 10-14 for -3 router assembly instructions, and figure 10-15 for router guide template instructions.
f. Blow out burrs and cutter dust with dry, clean, filtered air.
g. Center patch plate selected from figure 10-12 and drill rivet pattern.

CAUTION

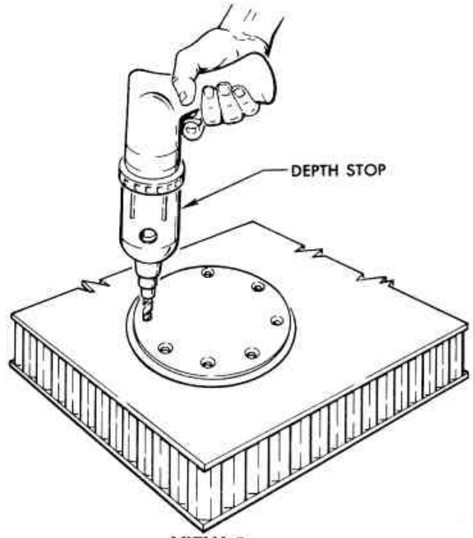
TO PREVENT EXCEEDING DESIRED RIVET HOLE DEPTH, USE A DRILL DEPTH STOP. (SEE VIEW B.) DESIRED RIVET PATTERN MAY ALSO BE DRILLED PROPERLY USING CORRECT PRESSURE PLATE AND -813 DRILL BUSHING ASSEMBLY. SEE FIGURE 10-28 FOR DRILL BUSHING AND PRESSURE PLATE INSTRUCTIONS.

h. Tape Kraft paper over repair area and cut a hole from paper to expose routed hole.
i. Mix core filler E3045 (see Table 10-1 for instructions) and fill core cavity flush at edge of skin and 1/32-inch above in center. See View C.

CAUTION

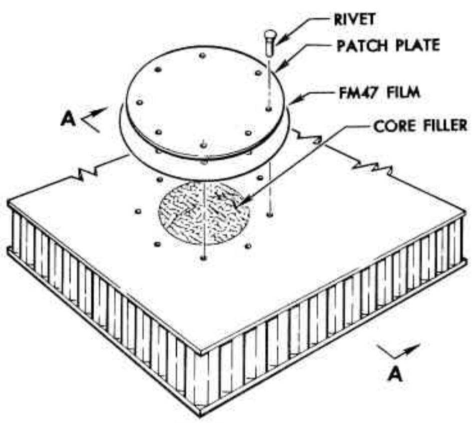
WEAR CLEAN WHITE GLOVES TO HANDLE ALL REPAIR PARTS TO PREVENT TRANSFER OF GREASE OR MOISTURE FROM HANDS.

j. Remove Kraft paper and clean primed area with Aliphatic Naphtha, Specification TT-N-95.
k. Cut a disc from FM47 film 1/16-inch larger in diameter than patch plate and center over repair area. Place patch plate over FM47 film disc. See View C. Hold repair parts in place with #850 tape.
l. Bond and cure repair parts to skin; see Section A-A. Use heat lamp, pressure plate, and -805 pressure system. See figure 10-27 and 10-28 for pressure plate and pressure system instructions. Refer to Table 10-1 for FM47 film curing instructions.
m. Install rivets in patch plate and mill rivets flush in accordance with figure 10-12. See Section A-A for cross-section of completed repair.

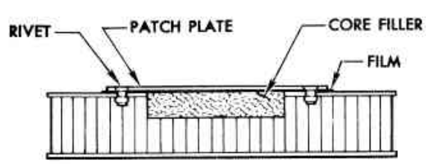


VIEW B

USE A DRILL STOP WHEN DRILLING RIVET HOLES



VIEW C



SECTION A-A

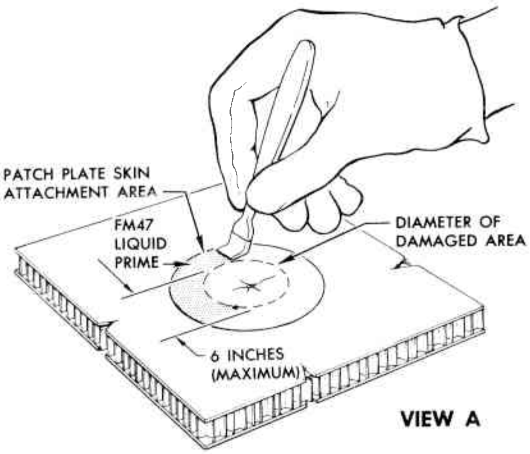
06.03.297

Figure 10-32. Honeycomb Repair No. 3 — Hole

REPAIR NO. 4 AND 5

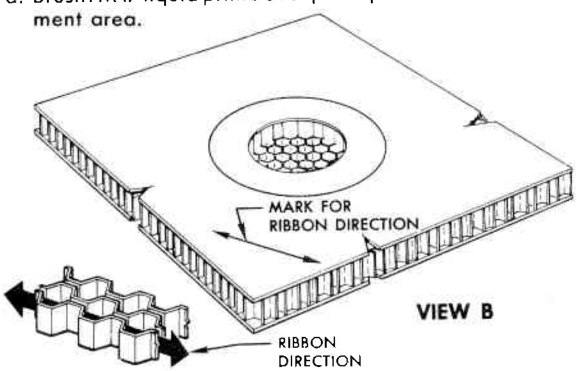
NOTE

REPAIRS 4 AND 5 ARE FOR HOLES THROUGH ONE SIDE OF PANEL USING A 1/8-INCH CELL HEX-CELL CORE PLUG AND AN ALUMINUM FOIL DISC TO FILL ROUTED HOLE. MAXIMUM ALLOWABLE HOLE DIAMETER IS 6 INCHES. MAXIMUM ALLOWABLE HOLE DEPTH IS 1 INCH. FASTENERS ARE INSTALLED AFTER BONDING IN REPAIR NO. 5 TO SAFEGUARD AGAINST PEEL PROPAGATION.



PROCEDURE, REPAIR NO. 4

a. Determine diameter of damaged area with damage locating template; see figure 10-11.
b. Select proper patch plate and core plug. See figure 10-12.
c. Prepare and clean repair area in accordance with figure 10-13.
d. Brush FM47 liquid prime over patch plate skin attachment area.

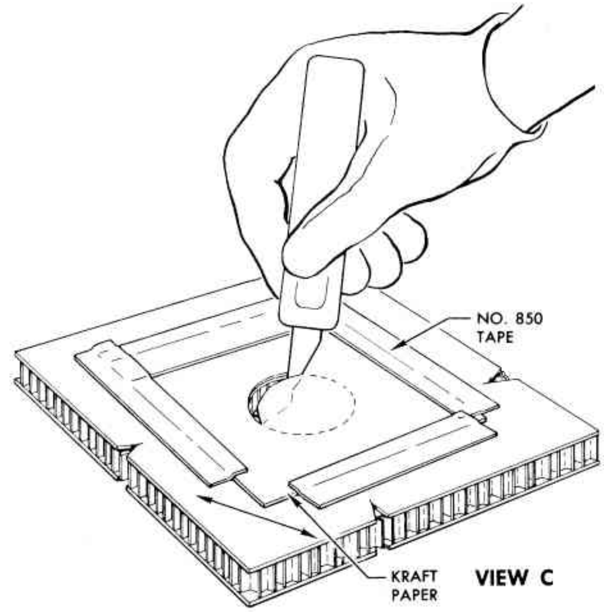


e. Rout out damaged area to a round, preselected size; see view A. Use -3 router assembly and proper router guide template. See figure 10-14 for -3 router assembly instructions, and figure 10-15 for router guide template instructions. Remove damaged core to match applicable dimensions in core plug table, figure 10-12 (hole depth, dimension "C," and core plug thickness, dimension "D"). Blow out burrs and cutter dust with dry, clean filtered air.
f. On skin near routed hole, mark ribbon direction.

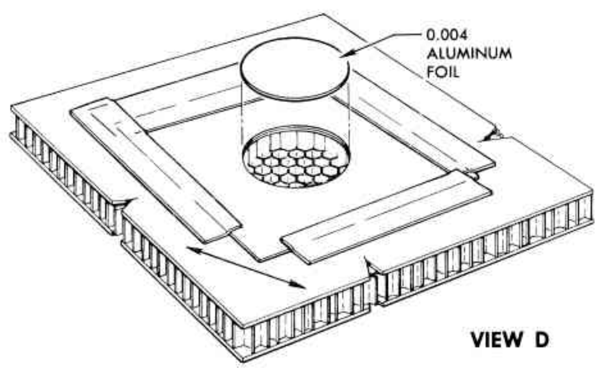
CAUTION

USE CLEAN WHITE GLOVES TO HANDLE ALL REPAIR PARTS TO PREVENT TRANSFER OF MOISTURE OR GREASE FROM HANDS

06.03.335-1



g. Tape Kraft paper over repair area. Cut hole through paper around routed hole;



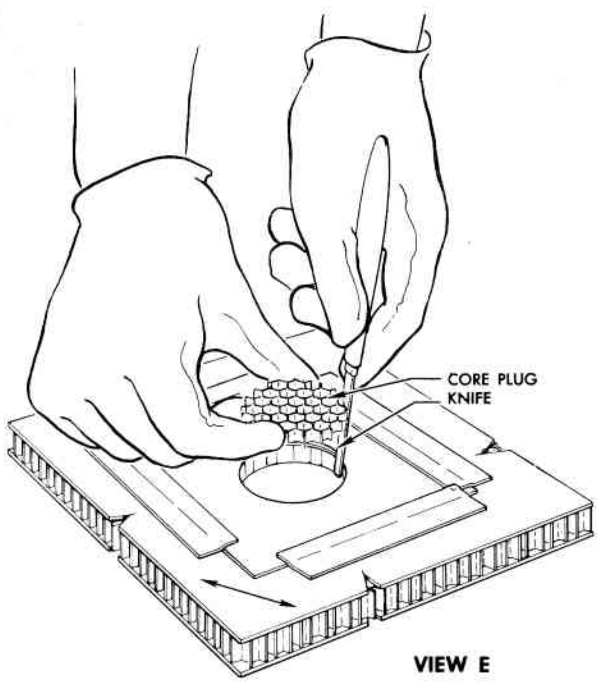
h. Cut a disc of 0.004 aluminum foil, Specification MIL-A-148C, the diameter of the routed hole.

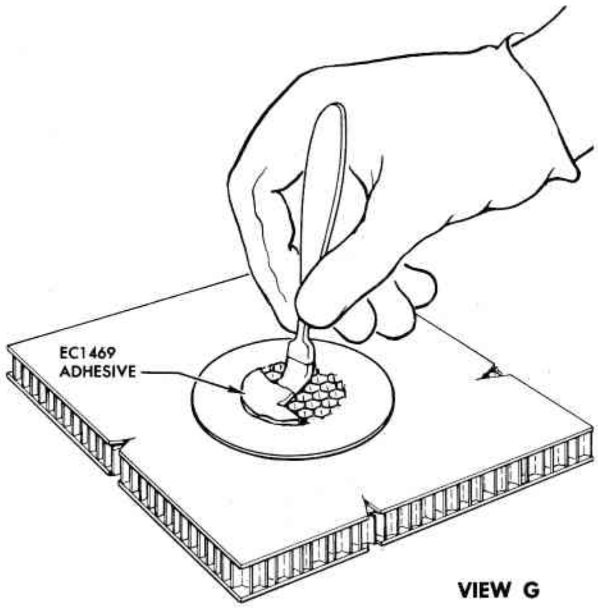
CAUTION

IF FOIL IS SOILED, CLEAN WITH ALIPHATIC NAPHTHA, FEDERAL SPECIFICATION TT-N-95.

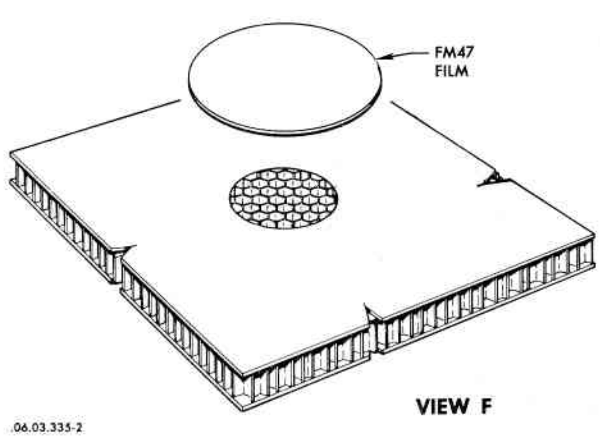
i. Hold foil disc with aliphatic naphtha cleaned needle nose pliers. Brush 338 adhesive on one side of foil disc 1/16 -inch to 1/8-inch thick.
j. Install foil disc in routed hole with adhesive coated side down. Push foil down smoothly and evenly on core.
k. Apply 338 adhesive 1/16-inch to 1/8-inch thick on top side of foil disc and to routed hole's exposed core surfaces. Work adhesive into core surfaces to fill all cavities.

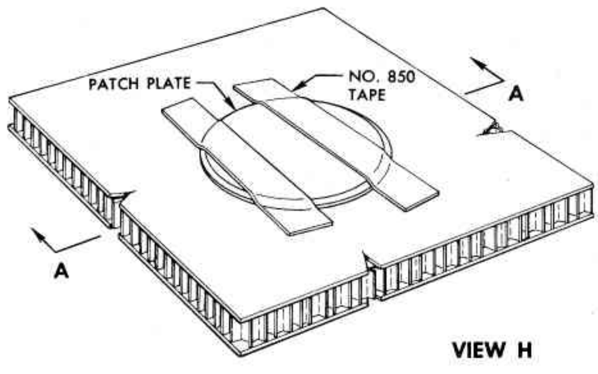
Figure 10-33. Honeycomb Repair No. 4 and 5 — Hole (Sheet 1 of 3)

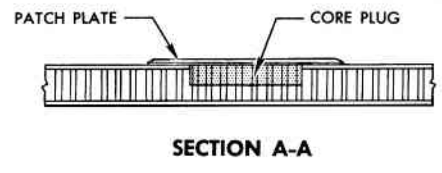




l. Apply 338 adhesive to outside edge of core plug selected from figure 10-12 and place in routed hole. Core plug ribbon shall run parallel to mark placed on skin in step "f." Use knife to guide core plug in position. Exercise care to prevent crushing core cells.
m. Press core plug firmly in place.
n. Cure 338 adhesive with heat lamps in accordance with Table 10-1.
o. Remove Kraft paper. If repair area is contaminated, scrape and then clean with aliphatic naphtha. Touch-up any bare spots with FM47 liquid primer.
p. Cut a disc of FM47 film 1/16-inch larger in diameter than patch plate.
q. Center FM47 disc on repair.
r. Apply 1/16-inch coat of EC1469 adhesive with a brush over the area of film disc which covers core only.
s. Turn film disc over and center EC1469 adhesive area over core. Brush patch plate bonding surface with FM47 liquid prime. Install patch plate and hold in place with #850 tape; see view H.
t. Bond and cure repair parts to skin. Use heat lamp and vacuum box assembly or air diaphragm assembly. See figure 10-23 for vacuum box assembly instructions. Refer Table 10-1 for FM47 film and EC1469 adhesive curing instructions.
u. See Section A-A for cross-section of completed repair.
v. After repair equipment is removed, inspect repaired area for voids by tapping. Voids up to 0.25 of an inch are permissible.

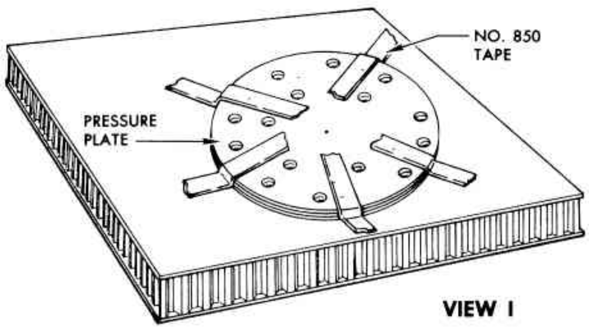






06.03.335-2

Figure 10-33. Honeycomb Repair No. 4 and 5 — Hole (Sheet 2 of 3)



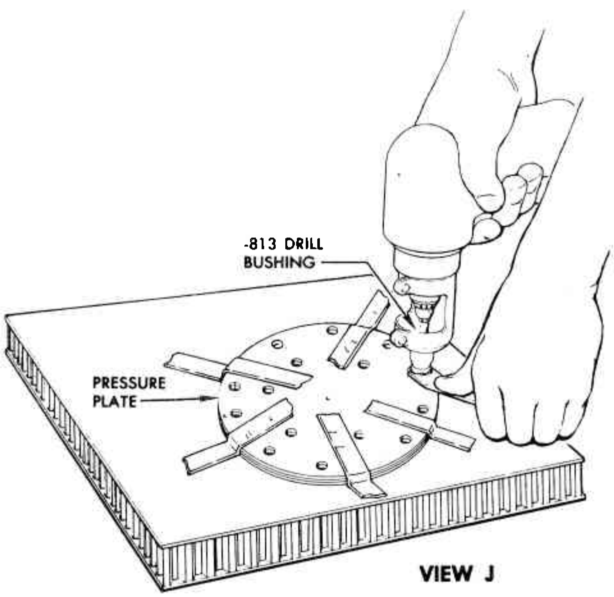
VIEW I

PROCEDURE, REPAIR NO. 5

NOTE

REPAIR NO. 5 IS THE SAME AS REPAIR NO. 4 EXCEPT THAT FASTENERS ARE INSTALLED AFTER THE REPAIR IS BONDED.

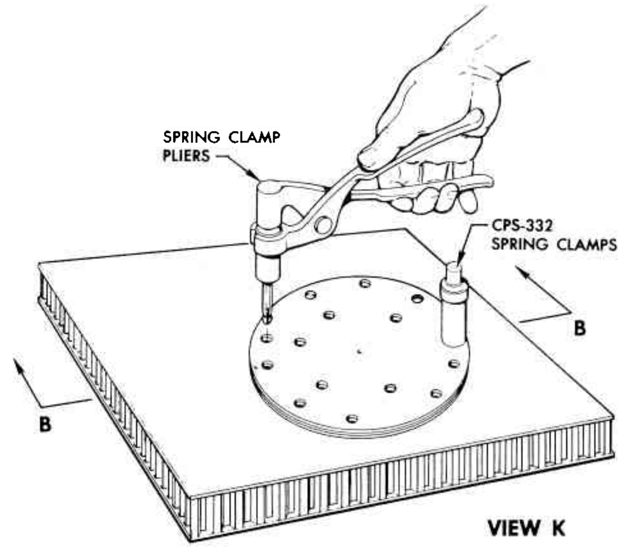
a. Perform steps "a" through "t" of Repair No. 4.
b. Select a pressure plate the same size as repair patch plate; see figure 10-28.
c. Using selected pressure plate as a hole pattern template, locate and center pressure plate on repair patch plate. Tape in place with #850 tape.



VIEW J

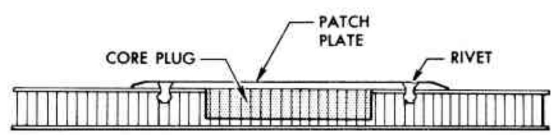
d. With -813 drill bushing, drill two pilot holes in opposite edges of plates as shown. Use a drill stop and drill through patch plate and skin only. See figure 10-28 for detailed instructions on the use of the -813 drill bushing with the pressure plates.

.06.03.335-3



VIEW K

e. Install CPS-332 spring clamps in holes drilled in step "d."
f. Drill remaining pilot holes.
g. Drill holes of correct diameter for rivets to be installed. See figure 10-12 for patch plate and rivet patterns.
h. Remove pressure plate and clamps.
i. Countersink holes in accordance with figure 10-12.
j. Install and mill rivets flush in accordance with figure 10-12.
k. See Section B-B for cross-section of completed repair.



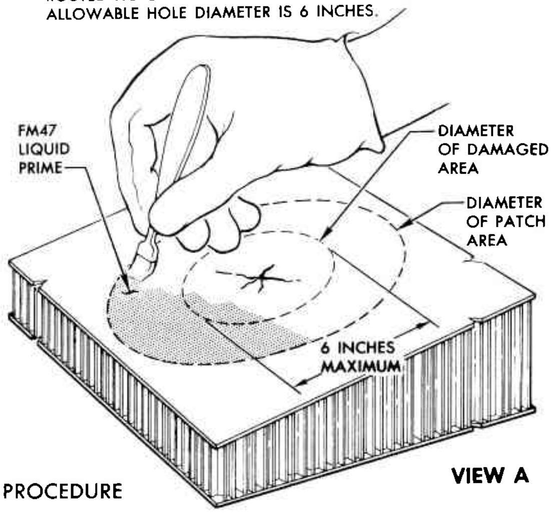
SECTION B-B

Figure 10-33. Honeycomb Repair No. 4 and 5 — Hole (Sheet 3 of 3)

REPAIR NO. 6

NOTE

THIS PROCEDURE IS FOR REPAIR OF HOLES THROUGH ONE SIDE OF A PANEL, USE 1/8-INCH CELL HEX-CELL PERFORATED ALUMINUM HONEYCOMB CORE PLUG AND AN ALUMINUM FOIL DISC TO FILL ROUTED HOLE. THIS REPAIR IS SIMILAR TO REPAIRS 4 AND 5 EXCEPT THAT ROUTED HOLE DEPTH MAY EXCEED 1 INCH. MAXIMUM ALLOWABLE HOLE DIAMETER IS 6 INCHES.



VIEW A

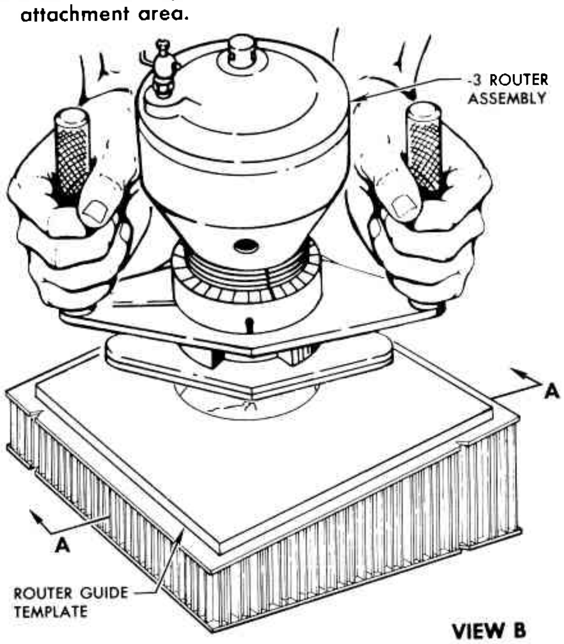
PROCEDURE

a. Determine diameter of damaged area with damage locating template; see figure 10-11.
b. Select proper patch plate and core plug; see figure 10-12.

NOTE

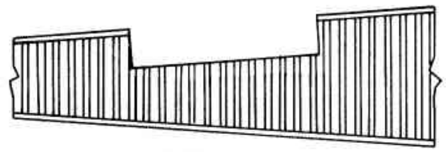
IF ROUTED HOLE DEPTH EXCEEDS 1 INCH, SELECT A CORE PLUG OF APPLICABLE DIAMETER FROM CORE PLUG AND TUBE SHEET TABLE ON FIGURE 10-31.

c. Cut core plug selected to length required to fit routed hole.
d. Prepare and clean repair area in accordance with figure 10-13.
e. Brush FM47 liquid prime over patch plate skin attachment area.



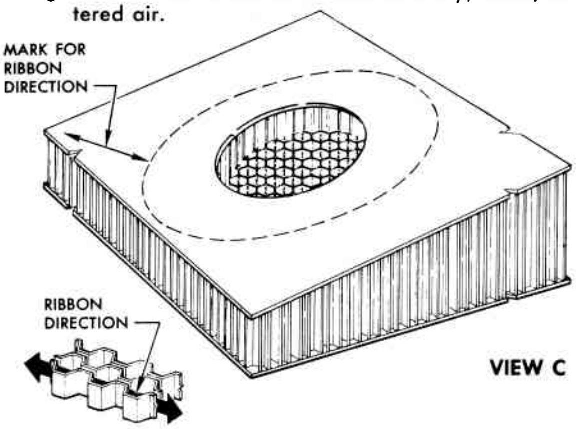
VIEW B

06.03.336-1



SECTION A-A

f. Rout out damage to a round preselected size; see view B. Use -3 router assembly and proper router guide template. See figure 10-14 for -3 router assembly instructions, and figure 10-15 for router guide template instructions. Remove damaged core to match applicable dimensions in core plug table, figure 10-12 (hole depth, dimension "C," and core plug thickness, dimension "D").
g. Blow out burrs and cutter dust with dry, clean, filtered air.

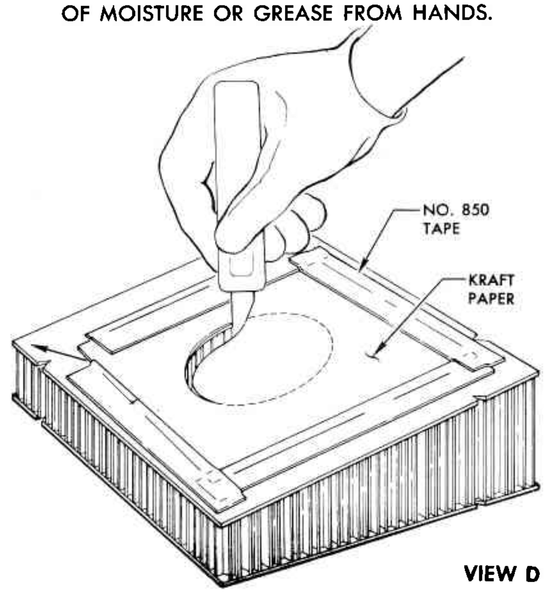


VIEW C

h. On skin near routed hole, mark ribbon direction.

CAUTION

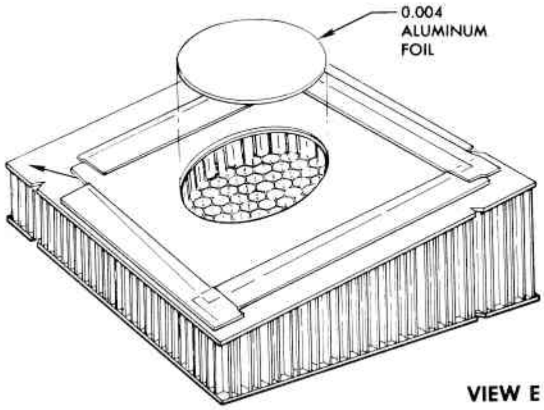
USE CLEAN WHITE GLOVES TO HANDLE ALL REPAIR PARTS TO PREVENT TRANSFER OF MOISTURE OR GREASE FROM HANDS.



VIEW D

i. Tape Kraft paper over repair area. Cut hole through paper around routed hole.

Figure 10-34. Honeycomb Repair No. 6 — Hole (Sheet 1 of 3)



j. Cut a disc of 0.004 aluminum foil, Specification MIL-A-148C, the diameter of routed hole.

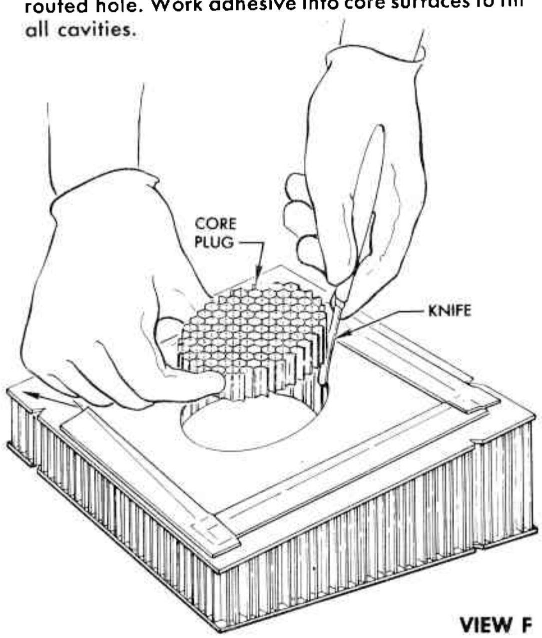
CAUTION

IF FOIL IS SOILED, CLEAN WITH ALIPHATIC NAPHTHA, FEDERAL SPECIFICATION TT-N-95.

k. Brush 338 adhesive on one side of foil disc 1/16-inch to 1/8-inch thick. Hold foil disc with aliphatic naphtha cleaned needle nose pliers.

l. Install foil disc in routed hole with adhesive coated side down. Push foil down smoothly and evenly on core.

m. Apply 338 adhesive 1/16-inch to 1/8-inch thick on top side of foil disc and to exposed core surfaces of routed hole. Work adhesive into core surfaces to fill all cavities.



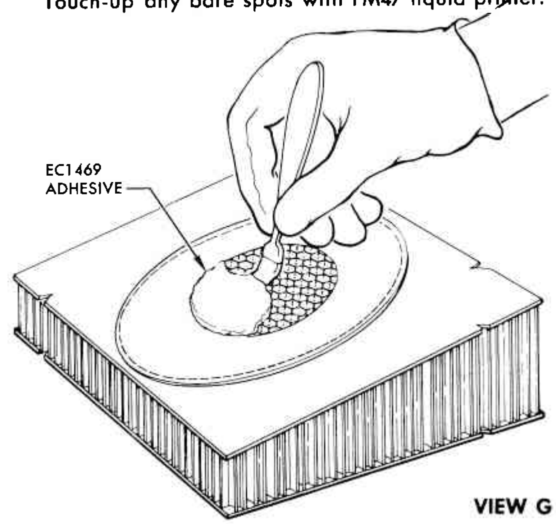
n. Using core plug selected from figure 10-12, apply 338 adhesive to outside edge of core and place in routed hole. Core plug ribbon shall run parallel to mark placed on skin in step "h." Use knife to guide core plug in position. Be careful not to crush core cells.

o. Press core plug firmly in place.

06.03.336-2

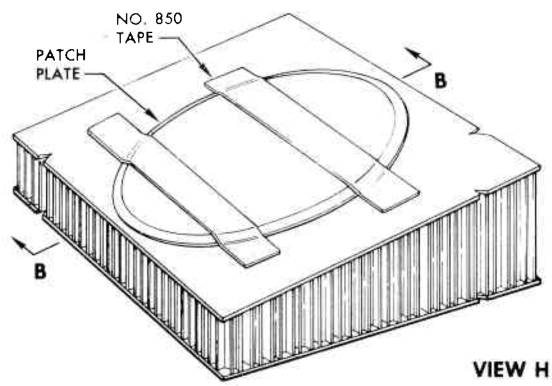
p. Cure 338 adhesive with heat lamps in accordance with Table 10-1. Cut away excess core. See figure 10-29 for core saw instructions.

q. Remove Kraft paper. If repair area is contaminated, scrape and then clean with aliphatic naphtha. Touch-up any bare spots with FM47 liquid primer.



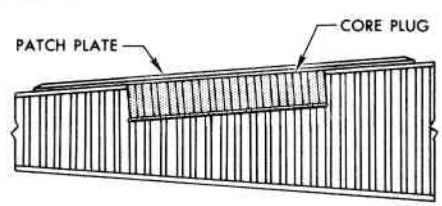
r. Cut disc of FM47 film 1/16-inch larger in diameter than patch plate.

s. Center FM47 film disc on repair. Apply 1/16-inch coat of EC1469 adhesive with a brush over area of film disc which covers core only.



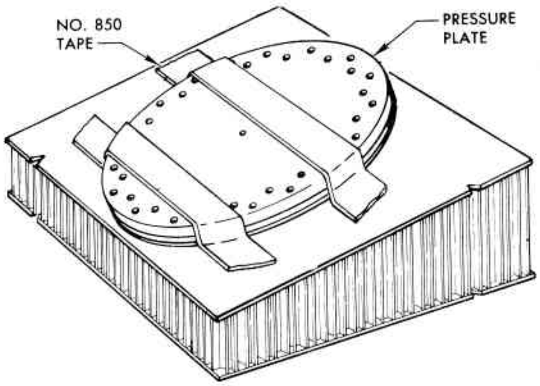
t. Turn film disc over and center EC1469 adhesive area over core. Brush patch plate bonding surface with FM47 liquid prime. Install patch plate and hold in place with #850 tape.

u. Bond and cure repair parts to skin. Refer to Table 10-1 for FM47 film and EC1469 adhesive curing instructions. Remove #850 tape after curing adhesive.



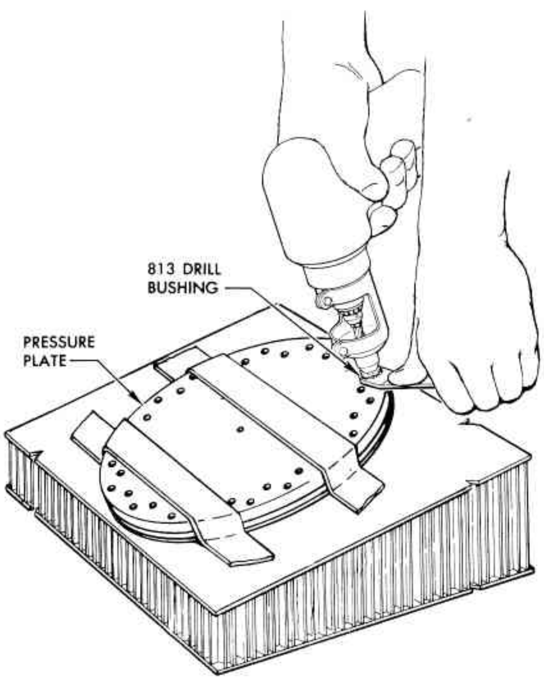
SECTION B-B

Figure 10-34. Honeycomb Repair No. 6 — Hole (Sheet 2 of 3)



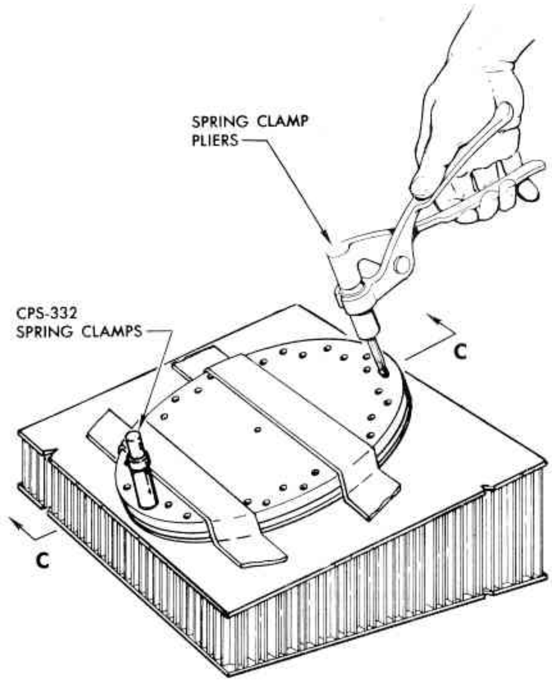
VIEW I

v. Select a pressure plate the same size as repair patch plate; see figure 10-28.
w. Using selected pressure plate as a hole pattern template, locate and center pressure plate on repair patch plate. Tape in place with #850 tape.



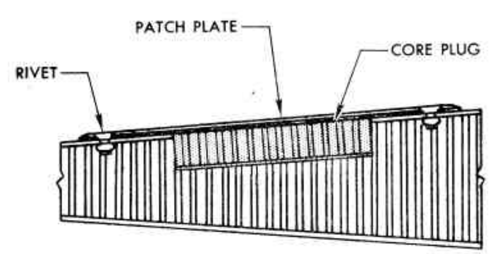
VIEW J

x. With -813 drill bushing, drill two pilot holes in opposite edges of plate as shown. Using a drill stop, drill through patch plate and skin only. See figure 10-28 for detailed instructions on the use of the -813 drill bushing with pressure plates.



VIEW K

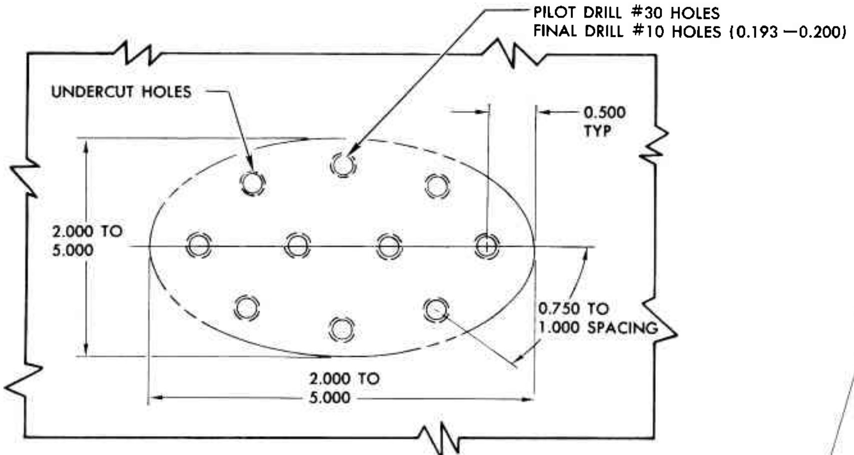
y. Install CPS-332 spring clamps in holes drilled in step "x"; drill remaining pilot holes.
z. Drill holes of correct diameter for rivets to be installed. See figure 10-12 for patch plate and rivet patterns.
aa. Remove pressure plate and clamps; countersink holes in patch plate.
ab. Install and mill rivets flush in accordance with figure 10-12.
ac. See section C-C for a cross-section of completed repair.



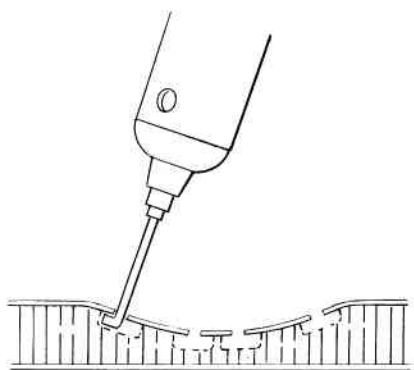
SECTION C-C

06.03.336-3

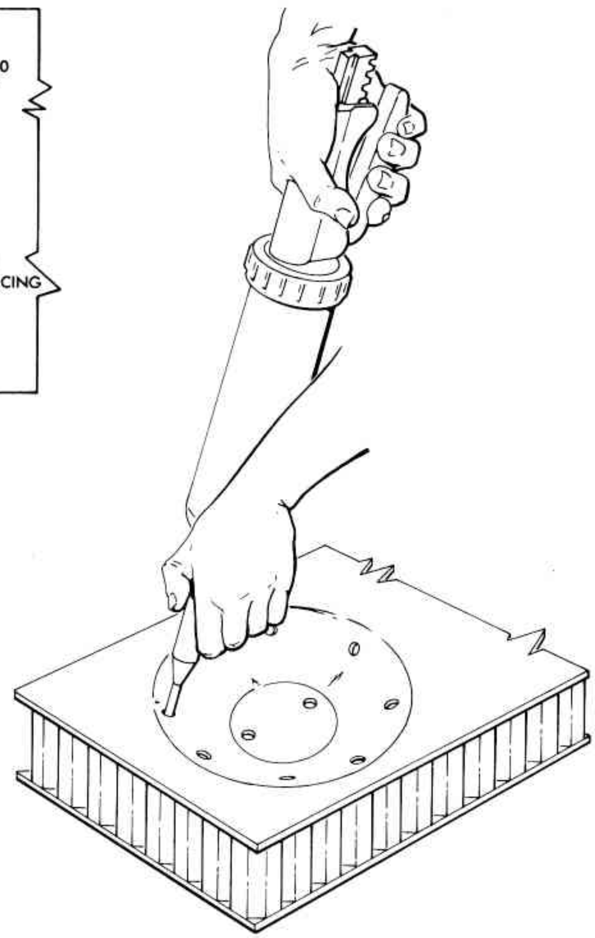
Figure 10-34. Honeycomb Repair No. 6 — Hole (Sheet 3 of 3)



VIEW A



VIEW B



VIEW C

INJECT ADHESIVE OR CORE FILL WITH SEALANT GUN

REPAIR NO. 7, 8, 9, 10 AND 11

NOTE

THESE ARE DENT OR DEPRESSION REPAIRS USED WHEN ONE SIDE OF THE PANEL IS DAMAGED. SKIN IS NOT REMOVED, DAMAGE DIAMETER SHALL NOT EXCEED 5 INCHES.

EVALUATE STRESS LOADS ON DAMAGED AREA AND TIME REQUIRED TO COMPLETE THE REPAIR BEFORE SELECTING REPAIR NUMBER. REPAIRS 7 AND 11 ARE THE STRONGEST, BUT TAKE LONGER TO PERFORM. REPAIR NO. 8 IS NOT QUITE AS STRONG AS REPAIR 7 AND 11, BUT TAKES LESS TIME TO PERFORM. REPAIRS 9 AND 10 ARE THE WEAKEST, BUT REQUIRE THE LEAST TIME TO PERFORM.

PROCEDURE, REPAIR NO. 7

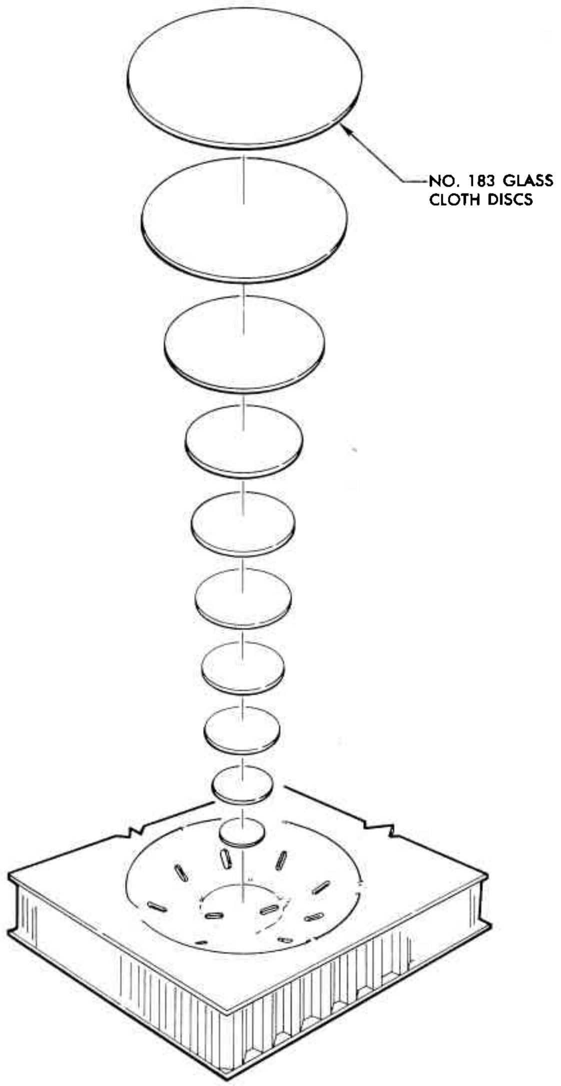
a. Prepare and clean damaged area. Use steps 1, 2 and 5 through 10 of figure 10-13.
b. Select a hole pattern template to fit damaged area from figure 10-17. Lightly centerpunch a hole pattern; see View A.
c. Drill No. 30 (0.128) pilot holes; use a drill stop to control depth.
d. Undercut holes drilled in step "c" to 0.25 inch with -95 undercutter; see View B. See figure 10-22 for instructions on the use of the -95 power-operated undercutter.
e. With a No. 10 (0.193) drill, drill out pilot holes drilled in step "c" to a 0.193 —0.200 inch diameter.
f. Smooth-up rough edges and blow out burrs and cutter dust with dry, clean, filtered air.
g. Mix Narmco 3135 adhesive in accordance with Table 10-1.
h. Use a sealant gun with a small tip and fill drilled holes with adhesive; see View C. Allow adhesive to cure before continuing repair. A heat lamp may be used to decrease curing time. See Table 10-1 for curing instructions.

NOTE

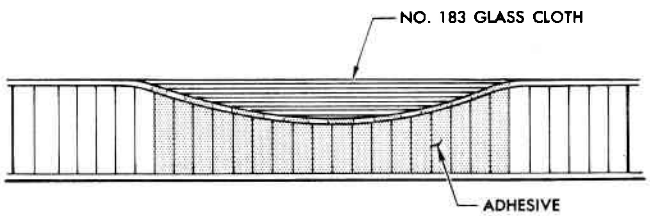
IF ANY DEPRESSIONS EXIST AT HOLE LOCATIONS AFTER COMPLETING STEP "H," REFILL AND CURE.

06.03.298-1A

Figure 10-35. Honeycomb Repair No. 7, 8, 9, 10, and 11 — Dent (Sheet 1 of 3)



VIEW D



VIEW E
(COMPLETED REPAIR NO. 7)

i. Cut discs of No. 183 glass cloth of graduated sizes so when laminated they will more than fill the depression. See View D.
j. Mix Narmco 3135 adhesive in accordance with Table 10-1. Brush a coat of the adhesive over the depression.
k. Soak each glass disc in the 3135 adhesive mixture. Install soaked discs in depression one at a time until depression is built up to 1/32-inch above existing skin.

CAUTION

AS EACH DISC IS INSTALLED, WORK OUT ANY BUBBLES WITH A TONGUE BLADE.

l. Allow repair to cure as indicated in Table 10-1 and sand down to skin contour. See View E for a cross-section of completed repair.

PROCEDURE, REPAIR NO. 8

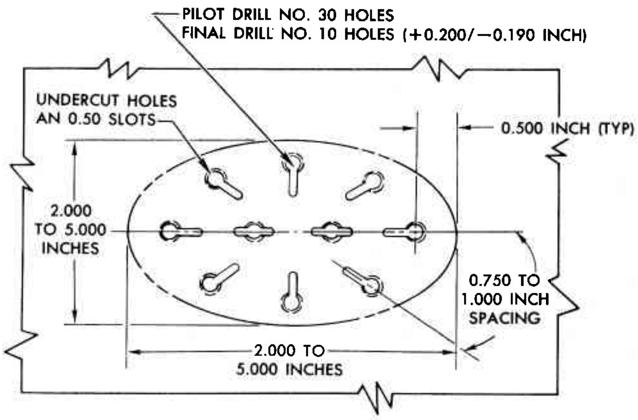
a. Perform steps "a" through "h" of Repair No. 7.
b. Mix Narmco 3135 adhesive in accordance with Table 10-1.
c. Stir Owens-Corning glass-fiber mill ends with 3135 adhesive until a paste consistency is obtained.
d. Fill depression 1/32-inch above skin levelwith paste and allow to cure.
e. Sand down to skin contour.

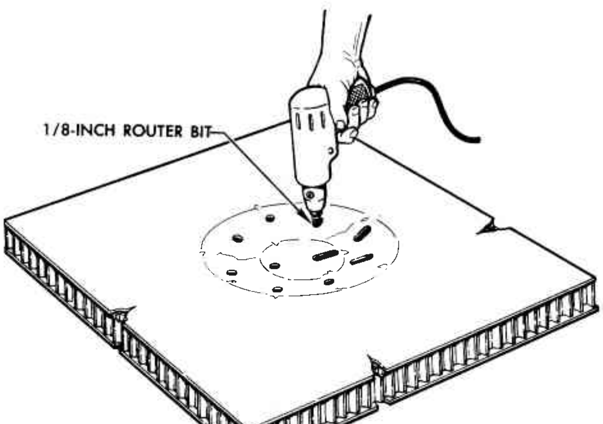
PROCEDURE, REPAIR NO. 9

a. Perform steps "a" through "h" of Repair No. 7.
b. Mix Narmco 3135 adhesive in accordance with Table 10-1.
c. Fill depression with the 3135 adhesive 1/32-inch above skin level; cure and sand down to skin contour.

06.03.298-2A

Figure 10-35. Honeycomb Repair No. 7, 8, 9, 10, and 11 — Dent (Sheet 2 of 3)

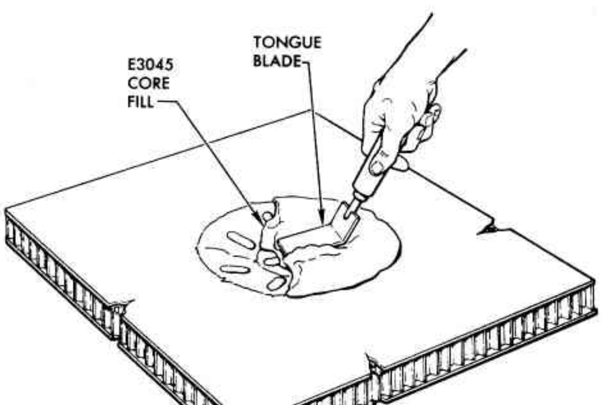




VIEW G

PROCEDURE, REPAIR NO. 10

a. Perform steps "a" through "e" of Repair No. 7; see view F.
b. With an 1/8-inch router bit, cut slots in skin 0.50 of an inch in length at drilled holes as shown in view G.
c. Smooth-up rough edges and blow out burrs and cutter dust with dry, clean, filtered air.



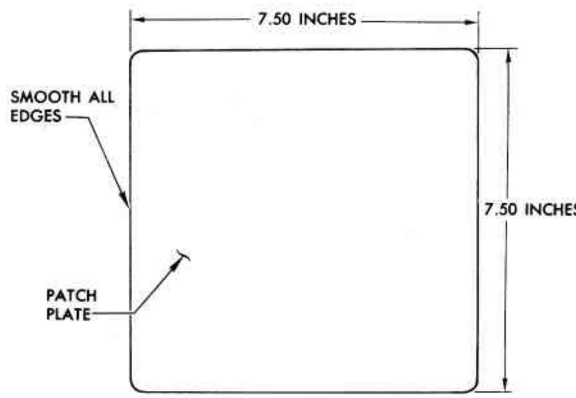
VIEW H

d. Mix E3045 core fill in accordance with Table 10-1.

06.03.298-3

e. Use a sealant gun with a small tip and fill drilled and slotted holes with core fill; see view C. With a tongue blade push and work fill until slots and holes are completely and evenly filled as shown in view H. Allow core fill to cure before continuing repair. A heat lamp may be used to decrease curing time. See Table 10-1 for curing instructions.

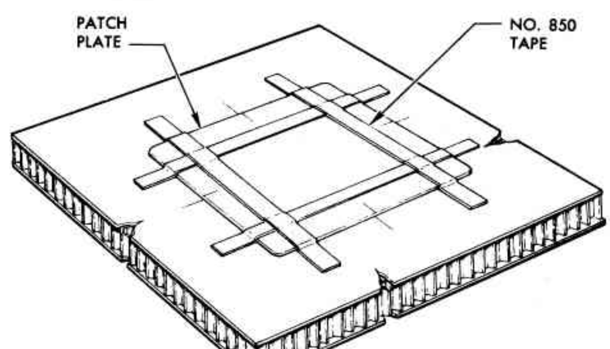
NOTE
IF ANY DEPRESSIONS EXIST AT HOLE LOCATIONS AFTER COMPLETING STEP "e," REFILL AND CURE.



VIEW I

PROCEDURE, REPAIR NO. 11

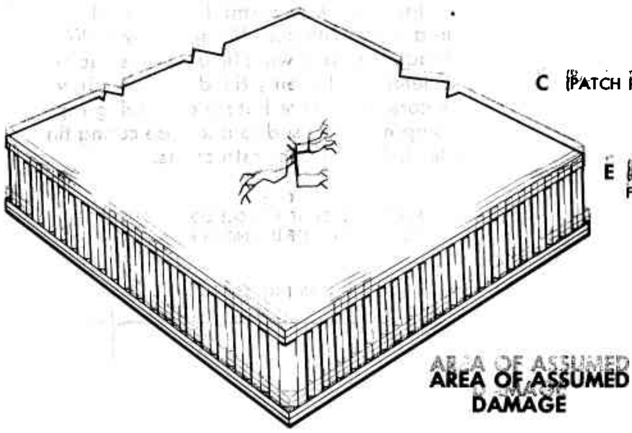
a. Prepare and clean damaged area. Use steps 1, 2, and 4 through 10 of figure 10-13. Perform steps "b" through "f" of Repair No. 10.
b. Fabricate a patch plate from 0.012 gage 2024-T3 aluminum alloy as shown in view I.

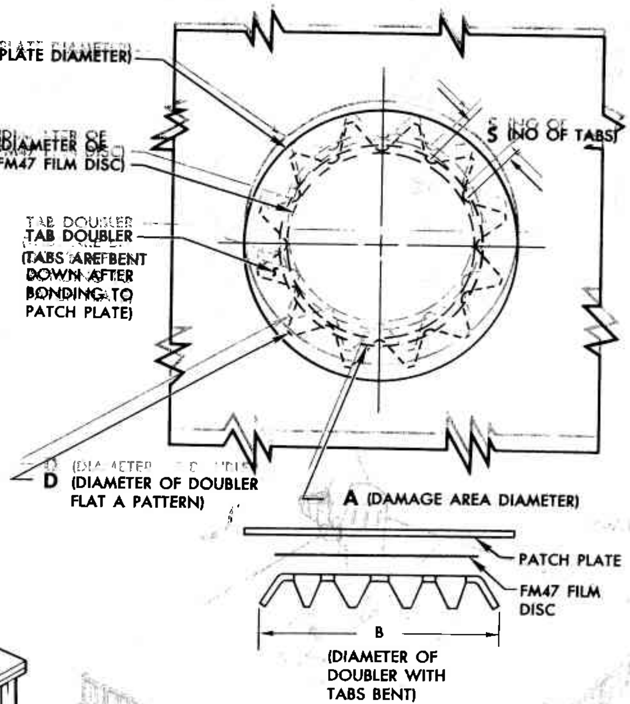


VIEW J

c. Apply EC1660 adhesive to one side of patch plate for bonding. Refer to Table 10-1 for adhesive application instructions.
d. Center and install patch plate on repair area with adhesive side down; see view J. Hold patch plate in place with #850 tape.
e. Bond and cure patch plate to skin. Use -801 vacuum bag assembly. See figure 10-26 for vacuum bag assembly installation and operation instructions. Refer to Table 10-1 for EC1660 adhesive curing instructions.

Figure 10-35. Honeycomb Repair No. 7, 8, 9, 10, and 11 — Dent (Sheet 3 of 3)





NOTE:
MAKE A PATCH PLATE AND TAB DOUBLER FROM 0.025 GAGE 2024-T3 ALUMINUM ALLOY.

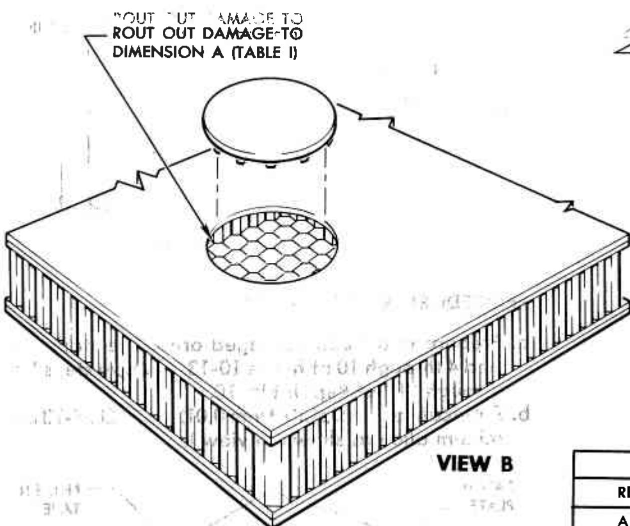


TABLE I					
REPAIR NO. 12 AND 13, PATCH PLATE ASSEMBLY DIMENSIONS					
A	B	C	D	E	S
1.000	0.630	1.062	1.000	0.50	8
1.500	1.060	1.562	1.500	0.94	10
2.000	1.50	2.062	2.000	1.380	12

REPAIR NO. 12

NOTE

REPAIR NO. 12 IS A DENT OR CRACK REPAIR 2 INCHES IN DIAMETER MAXIMUM.
REPAIR NO. 13 IS A DENT OR CRACK REPAIR 1 INCH IN DIAMETER MAXIMUM.

PROCEDURE

a. Determine diameter of damage with damage locating template and fabricate applicable patch plate assembly. See figure 10-11 for damage locating template instructions. See View A and Table I for patch plate assembly fabrication.

b. Rout out damage area to expose honeycomb core. Use -3 router assembly and proper router guide template in accordance with figures 10-14 and 10-15. Remove adhesive scraps and cuttings from core; see View B.

c. Center patch plate assembly over hole and push prongs of doubler down into hole.

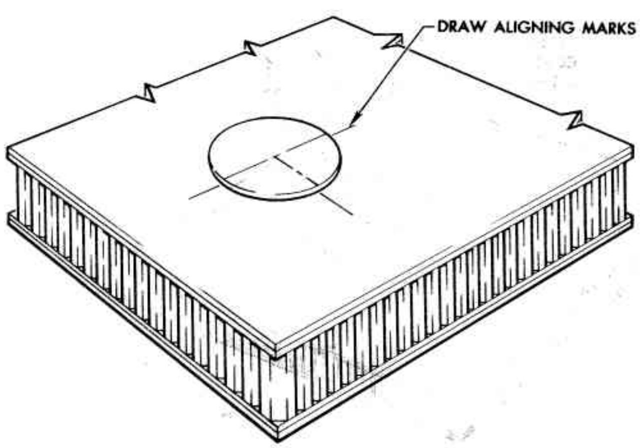
NOTE

PATCH PLATE IS LARGER THAN HOLE; PUSH DOWN UNTIL PATCH PLATE CONTACTS SKIN.

d. Mark patch plate so it can be reinstalled in the same position; see View C.

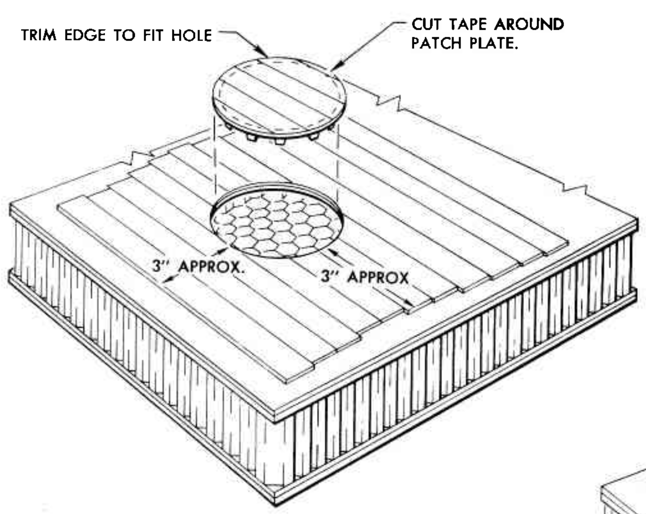
CAUTION

USE CLEAN WHITE GLOVES TO HANDLE ALL REPAIR PARTS TO PREVENT TRANSFER OF MOISTURE OR GREASE FROM HANDS.

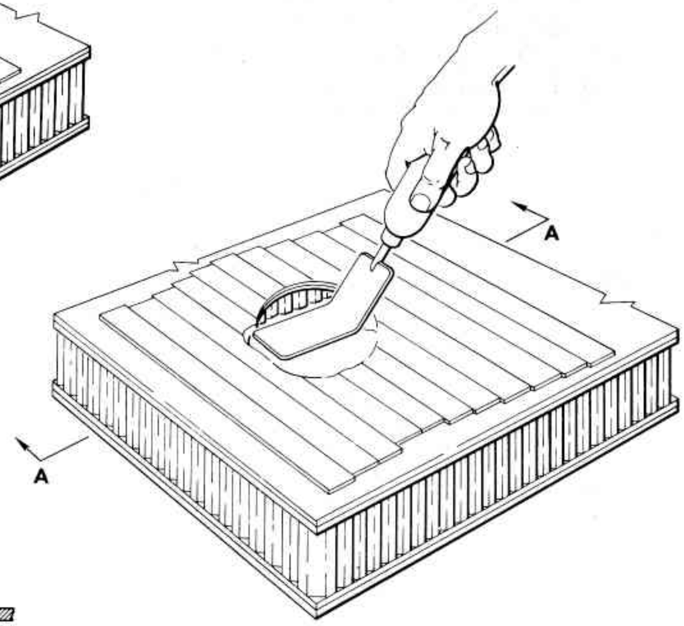


06.03.299-1B

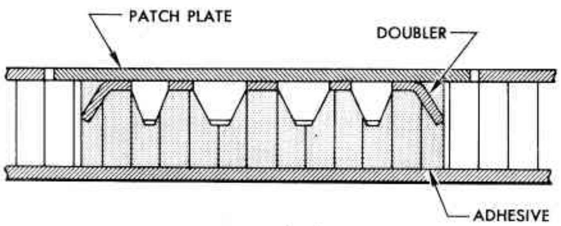
Figure 10-36. Honeycomb Repair No. 12 and 13 — Dent or Crack (Sheet 1 of 2)



VIEW D



VIEW E



SECTION A-A

e. Remove patch plate and trim until it fits hole.

NOTE
MAXIMUM ALLOWABLE GAP BETWEEN PATCH PLATE AND SKIN IS 0.010 INCH.

f. Install patch plate to align with marks made in step "d" and cover patch and surrounding skin with #850 tape to approximately 3 inches in all directions; see View D.
g. Cut tape flush with patch plate and hole edge and remove patch plate.
h. Select adhesive from Table 10-I. Mix per manufacturer's instructions.
i. Apply adhesive to hole with tongue blade or spatula; force down into core cells until holes are full; see View E. Apply adhesive to bonding surface of patch and prongs.
j. Install patch and push down until adhesive squeezes from around patch plate.
k. Allow adhesive to cure; remove tape and sand away any excess adhesive. See Section A-A for completed repair.

06.03.299-2A

REPAIR NO. 13

NOTE
REPAIR 13 IS SIMILAR TO REPAIR 12 EXCEPT THAT A PATCH PLATE IS NOT USED.

PROCEDURE

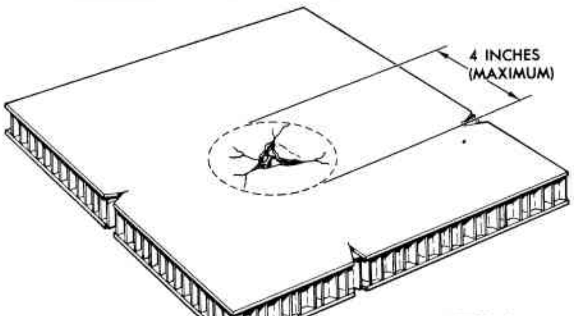
a. Rout out damaged area with a spot facer until honeycomb is exposed.
b. Straighten any damaged cores and blow out cutting dust.
c. Cover skin 3 inches in all directions around hole.
d. Mix adhesive. See Table 10-I for adhesive and mixing instruction.
e. Fill cores and hole 1/32 inch above skin and allow adhesive to cure.
f. Sand down adhesive to contour and remove tape.

Figure 10-36. Honeycomb Repair No. 12 and 13 — Dent or Crack (Sheet 2 of 2)

REPAIR NO. 14

NOTES

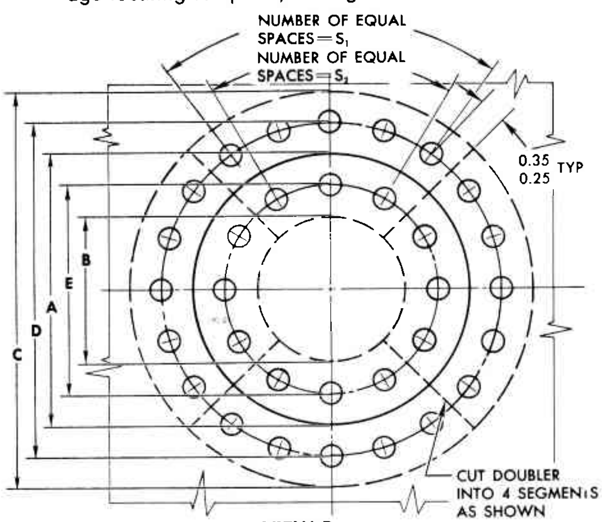
1. REPAIR NO. 14 IS FOR REPAIR OF HOLES THROUGH ONE SIDE OF THE PANEL USING A FLUSH PATCH PLATE, A DOUBLER, AND CORE FILL. ACCESS IS TO ONE SIDE OF THE PANEL ONLY. USE THIS REPAIR FOR 2-, 3-, AND 4-INCH DIAMETER HOLES.
2. MAKE PATCH PLATES AND DOUBLERS FROM 0.025 GAGE 2024-T3 ALUMINUM ALLOY.
3. MAKE FLUSH PATCH PLATES 0.030 INCH LARGER THAN DIMENSION A TO ALLOW FOR TRIM FITTING MATERIAL.



VIEW A

PROCEDURE

a. Determine diameter of damaged area with damage locating template; see figure 10-11.



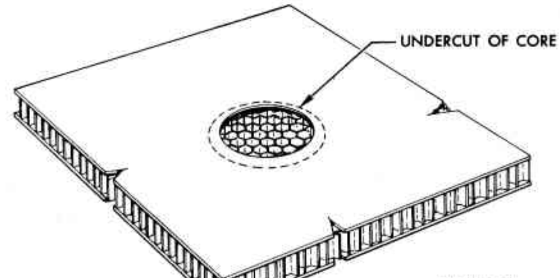
VIEW B

TABLE I							
FLUSH PATCH PLATE AND DOUBLER DIMENSIONS							
DIAMETETER OF DAMAGE	A	B	C	D	E	S_1	S_2
2.000	2.000	1.000	3.50	2.75	1.50	3	1
3.000	3.000	1.500	4.50	3.75	2.25	4	2
4.000	4.000	2.500	5.50	4.75	3.25	5	3

b. After a diameter has been determined, select applicable flush patch plate and doubler from Table I (see view B).

c. Rout out hole to a round, preselected size of applicable A diameter; see View B. Rout hole to a depth of 9/32 (0.281) inch below underside of skin. Use -3 router assembly and applicable router guide template. See figure 10-14 for -3 router assembly instructions, and figure 10-15 for router guide template instructions.

06.03.337-1



VIEW C

d. Undercut honeycomb core to doubler diameter as shown in View B. Use the -93 undercutter. See figure 10-21 for instructions on the use of the undercutter.

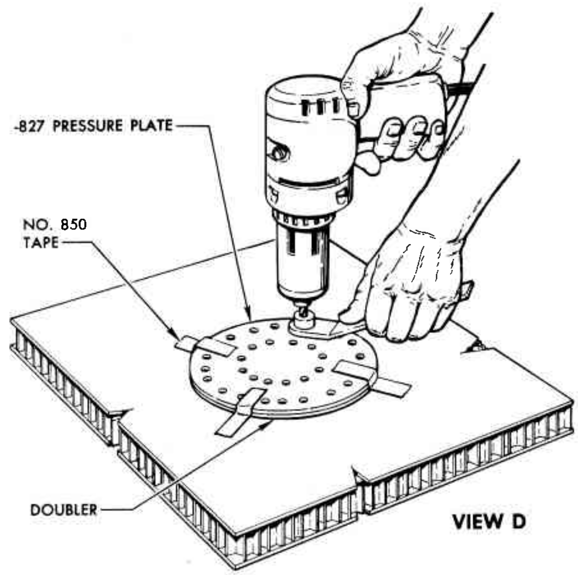
CAUTION

DO NOT SCRATCH UNDERSIDE OF SKIN WITH UNDERCUTTER.

e. Remove dust and cuttings from hole. Use an inspection mirror to check removal of cuttings and smoothness of undercut surfaces.

CAUTION

USE CLEAN WHITE GLOVES TO HANDLE ALL REPAIR PARTS TO PREVENT TRANSFER OF MOISTURE OR GREASE FROM HANDS.



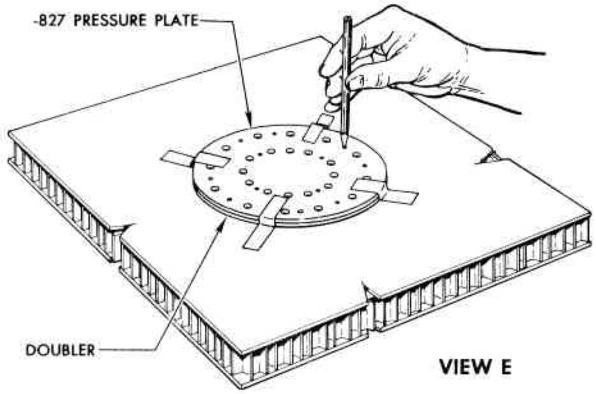
VIEW D

f. Center doubler selected in step "b" on repair.

g. With doubler centered on repair, place an -827 pressure plate in position as a template on top of doubler and secure in place with No. 850 tape; see View D. See figure 10-28 for pressure plate information.

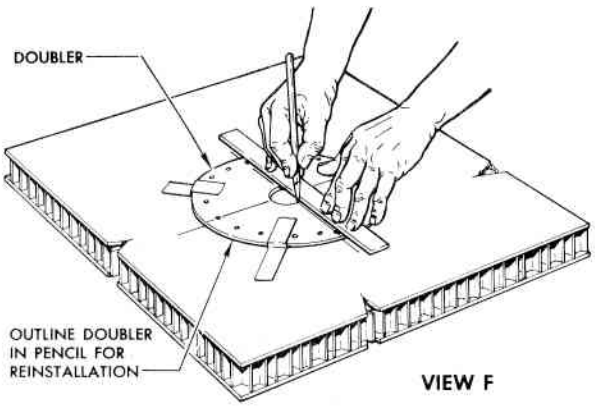
h. With an -813 drill bushing and the -827 plate, drill pilot holes for the outside row of rivet holes through doubler and skin only. See View B for applicable rivet pattern and figure 10-12 for drilling and fastener installation procedures. See figure 10-28 for detailed instructions on the use of the -813 drill bushing with pressure plates.

Figure 10-37. Honeycomb Repair No.14 — Hole (Sheet 1 of 3)



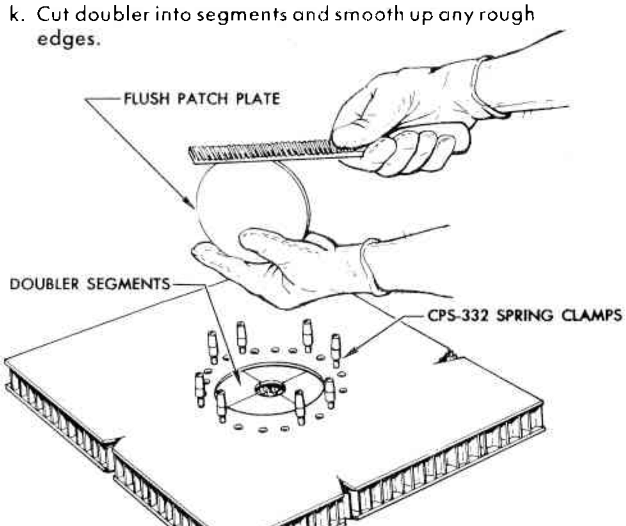
VIEW E

i. Locate and mark doubler segment cutting lines using -827 plate. Use No. 60 holes in -827 plate to locate segment cutting lines.



VIEW F

j. Remove -827 plate and mark position of doubler for reinstallation.
k. Cut doubler into segments and smooth up any rough edges.



VIEW G

l. Place doubler segments in position on repair and hold in place with CPS-332 spring clamps.

.06-03.337-2

m. Install flush patch plate selected in step "b" on doubler segments and file plate to fit hole.

NOTE

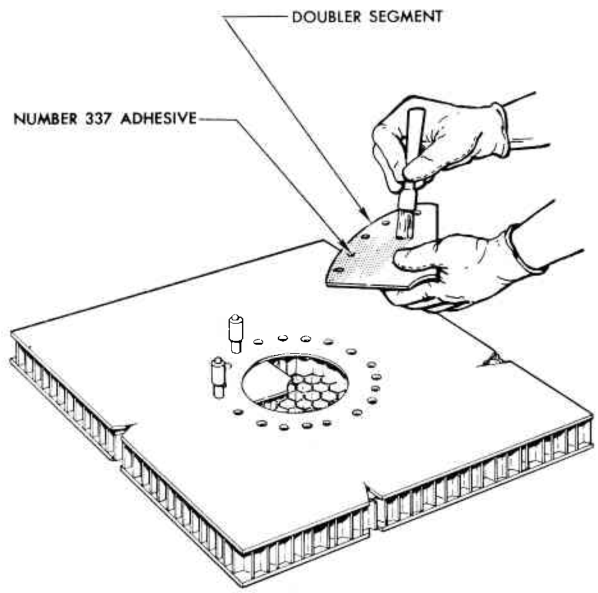
MAXIMUM ALLOWABLE GAP BETWEEN PATCH PLATE AND SKIN IS 0.010 INCH.

n. Place -827 plate on repair in the same position placed in step "g" and "h." See View D.

NOTE

TO PERFORM STEP "n," REMOVE FLUSH PATCH PLATE, HOLD DOUBLER SEGMENTS IN PLACE BY HAND OR WITH NO. 850 TAPE, AND REMOVE CPS-332 SPRING CLAMPS. THEN REINSTALL FLUSH PATCH PLATE, PLACE -827 PLATE IN POSITION AND REINSTALL CPS-332 SPRING CLAMPS THROUGH -827 PLATE, SKIN, AND DOUBLER SEGMENTS.

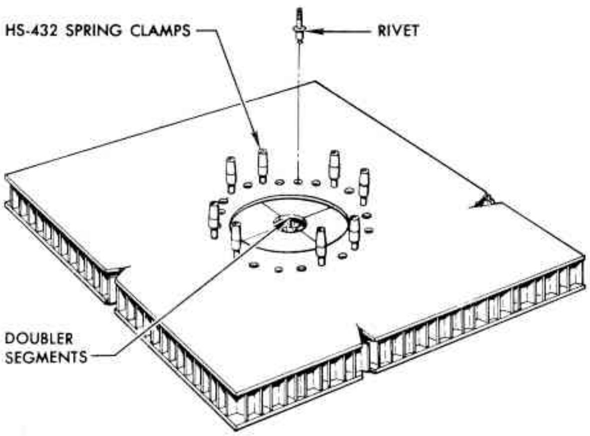
o. Pilot drill holes for inside row of rivets through flush patch plate and doubler segments; see View D. See View B for applicable rivet pattern and figure 10-12 for drilling and fastener installation procedures.
p. Drill out all pilot holes to rivet hole size selected from figure 10-12.
q. Remove -827 plate and countersink rivet holes in accordance with figure 10-12 in flush patch plate and skin only.



VIEW H

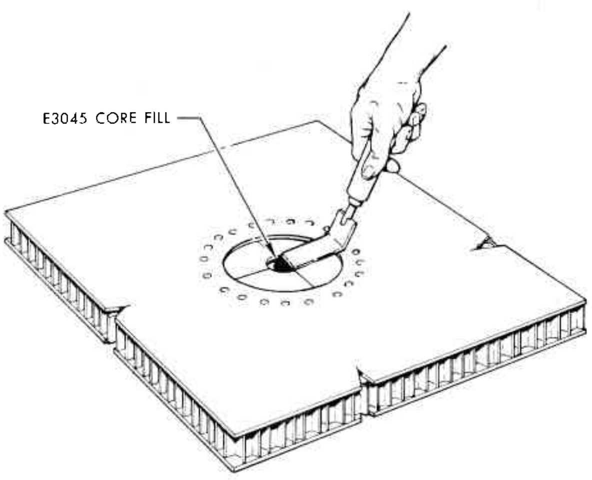
r. Remove all repair parts. Remove burrs and rough edges from repair area and parts. Clean repair parts and skin surface with Aliphatic Naphtha, Federal Specification TT-N-95. Apply a 1/32-inch layer of No. 337 adhesive to doubler segment and panel skin bonding surfaces. Refer to Table 10-1 for No. 337 adhesive mixing and application instructions.

Figure 10-37. Honeycomb Repair No. 14 — Hole (Sheet 2 of 3)



VIEW I

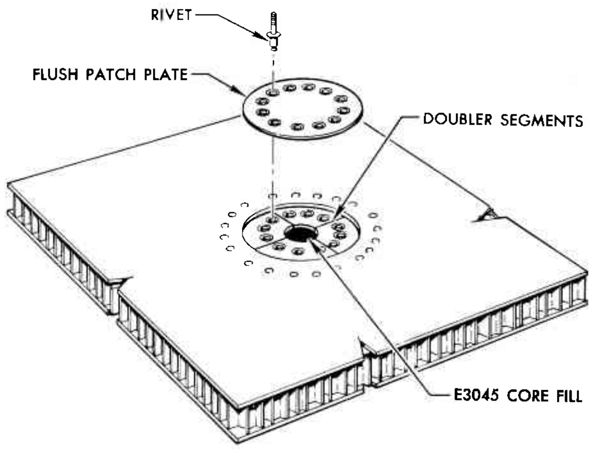
s. Reinstall doubler segments in their proper position and hold in place with HS-432 spring clamps. Rivet segments to panel skin in accordance with figure 10-12 using CR762-4-2 cherry rivets.



VIEW J

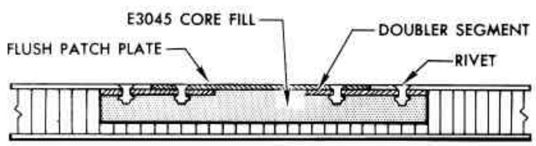
t. Mix E3045 core fill in accordance with Table 10-1.
u. Fill hole level with doubler as shown in View G with E3045 core fill. Force core fill under doubler segments; be sure cavity is filled evenly and completely.
v. Apply No. 337 adhesive to patch plate and doubler bonding surfaces.

.06.03.337-3

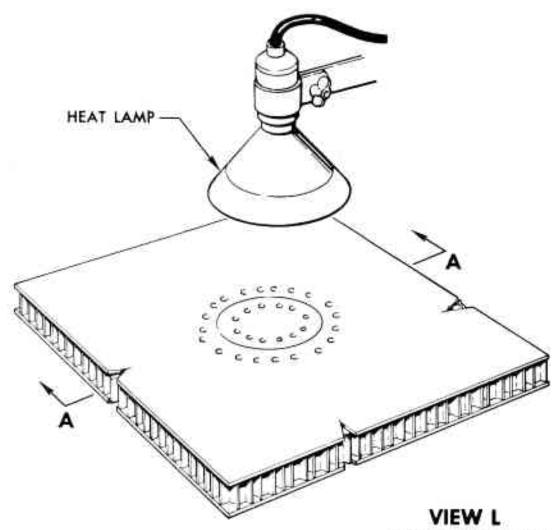


VIEW K

w. Reinstall patch plate and rivet in place. Install CR762-4-2 cherry rivets. Mill rivets flush in accordance with figure 10-12.
x. Cure No. 337 adhesive and E3045 core fill in accordance with Table 10-1.
y. See Section A-A for a cross section of completed repair.



SECTION A-A



VIEW L
(COMPLETED REPAIR)

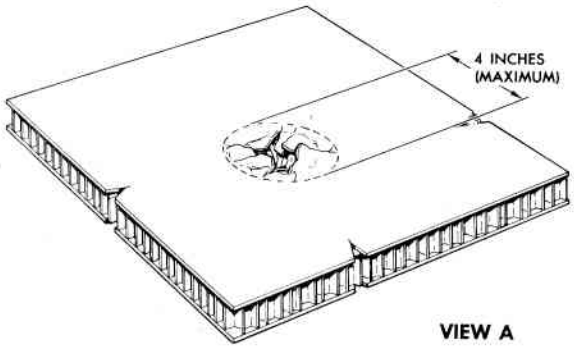
NOTE:
CURE ADHESIVE 24 TO 48 HOURS AT AMBIENT TEMPERATURE; OR 2 HOURS AT 200° USING HEAT LAMPS. DO NOT HEAT RAPIDLY.

Figure 10-37. Honeycomb Repair No. 14 — Hole (Sheet 3 of 3)

REPAIR NO. 15

NOTES

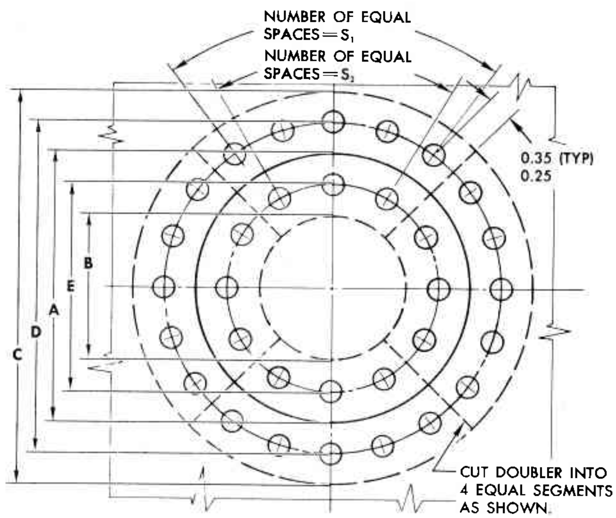
1. REPAIR NO. 15 IS FOR REPAIR OF 2-, 3-, AND 4-INCH DIAMETER HOLES THROUGH BOTH SIDES OF THE PANEL. THIS REPAIR IS SIMILAR TO REPAIR NO. 14 ON ONE SIDE USING A FLUSH PATCH PLATE, A DOUBLER, AND CORE FILL. A NON-FLUSH INSIDE PATCH PLATE IS USED TO COVER THE PANEL'S INSIDE SURFACE. ACCESS IS TO BOTH SIDES OF THE PANEL.
2. MAKE FLUSH PLATES, INSIDE SURFACE PATCH PLATES, AND DOUBLERS FROM 0.025 GAGE 2024-T3 ALUMINUM ALLOY.
3. MAKE FLUSH PATCH PLATES 0.030 INCH LARGER THAN DIMENSION A TO ALLOW FOR TRIM FITTING MATERIAL.



VIEW A

PROCEDURE

a. Determine diameter of damaged area with damage locating template; see figure 10-11.



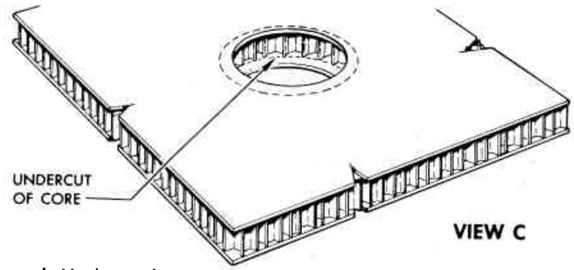
VIEW B

TABLE I							
FLUSH PATCH PLATE, INSIDE SURFACE PATCH PLATE, AND DOUBLER DIMENSIONS							
DIAMETER OF DAMAGE	A	B	C	D	E	S_1	S_2
2.000	2.000	1.000	3.50	2.75	1.50	3	1
3.000	3.000	1.500	4.50	3.75	2.25	4	2
4.000	4.000	2.500	5.50	4.75	3.25	5	3

b. After a diameter has been determined, select applicable flush patch plate, inside surface patch plate, and doubler from Table I (see View B).

c. Rout out hole to a round, preselected size of applicable A diameter; See View B. Use -3 router assembly and -53 router turntable assembly; see figure 10-14 and 10-19 for operating instructions.

06.03.338-1



VIEW C

d. Undercut honeycomb core to doubler diameter. Use the -93 undercutter. See figure 10-21 for instructions on the use of the undercutter.

CAUTION

DO NOT SCRATCH UNDERSIDE OF SKIN WITH UNDERCUTTER.

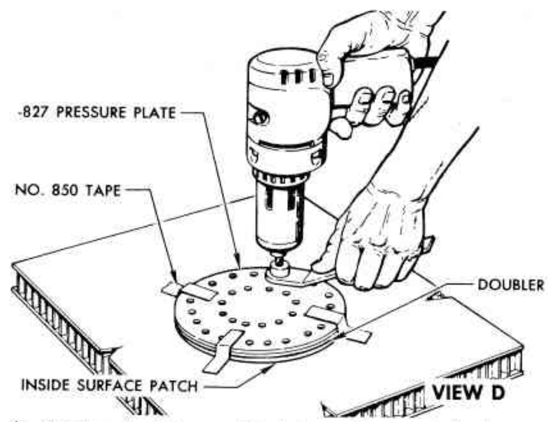
e. Remove dust and cuttings from hole. Use an inspection mirror to check removal of cuttings and smoothness of undercut surfaces.

f. Mark damaged area center lines on panel's inside skin surface with damage locating template in accordance with figure 10-11.

g. Clean and prepare panel's inside skin surface for patch plate in accordance with instructions in figure 10-13.

CAUTION

USE CLEAN WHITE GLOVES TO HANDLE ALL REPAIR PARTS TO PREVENT TRANSFER OF MOISTURE OR GREASE FROM HANDS.



VIEW D

h. Center doubler and inside surface patch plate selected in step "b" on repair and tape in place with NO. 850 tape; see View D and Section A-A.

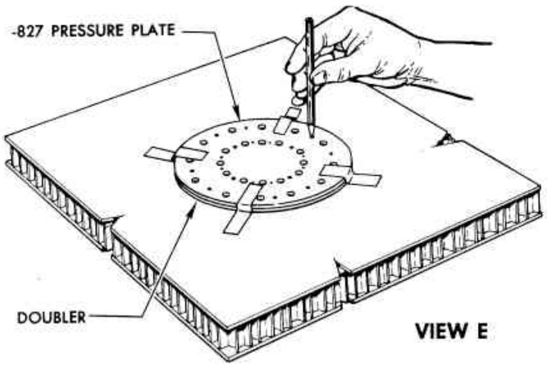
NOTE

CLEAN ALL REPAIR PARTS WITH ALIPHATIC NAPHTHA, FEDERAL SPECIFICATION TT-N-95.

i. With doubler and inside surface patch plate centered on repair, place an -827 pressure plate in position as a template on top of doubler. See figure 10-28 for pressure plate information.

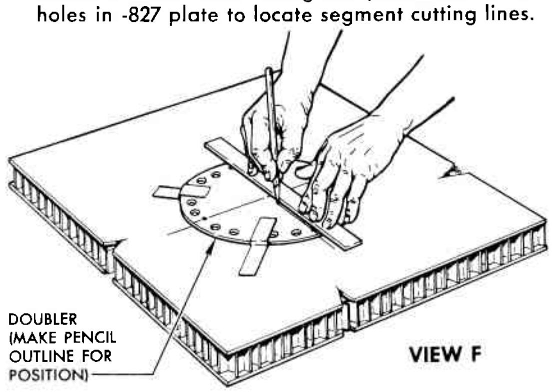
j. With an -813 drill bushing and the -827 plate, drill holes through doubler, skins, and inside surface patch plate for the outside row of rivets as shown in View D. See View B for applicable rivet pattern and figure 10-12 for drilling and fastener installation procedures. See figure 10-28 for detailed instructions on the use of the -813 drill bushing with pressure plates.

Figure 10-38. Honeycomb Repair No. 15 — Hole (Sheet 1 of 3)



VIEW E

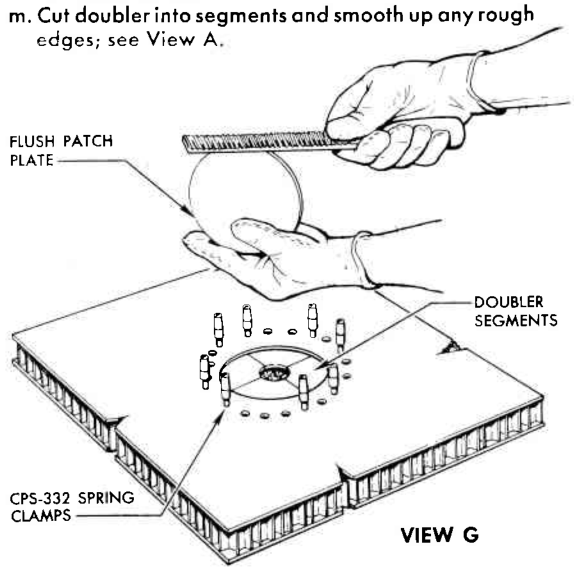
k. Locate and mark doubler segment cutting lines shown in View E and F using -827 plate. Use No. 60 holes in -827 plate to locate segment cutting lines.



VIEW F

l. Remove -827 plate and mark position of doubler for reinstallation.

m. Cut doubler into segments and smooth up any rough edges; see View A.



VIEW G

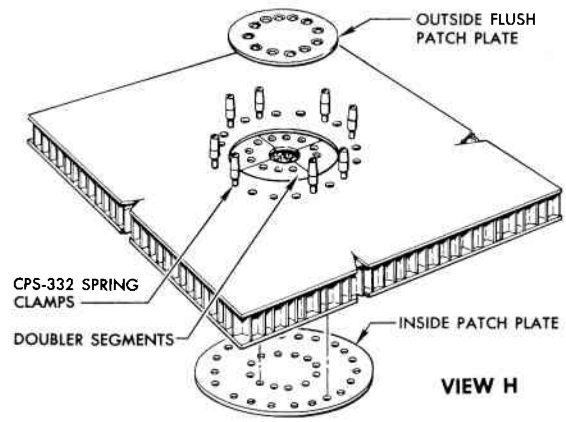
n. Place doubler segments in position on repair and hold in place with CPS-332 spring clamps; see View G.

o. Install flush patch plate selected in step "b" on doubler segments (file plate to fit hole).

NOTE

MAXIMUM ALLOWABLE GAP BETWEEN PATCH PLATE AND SKIN IS 0.010 INCH.

06.03.338-2



VIEW H

p. Place -827 plate on repair in the same position placed in steps "i" and "j."

NOTE

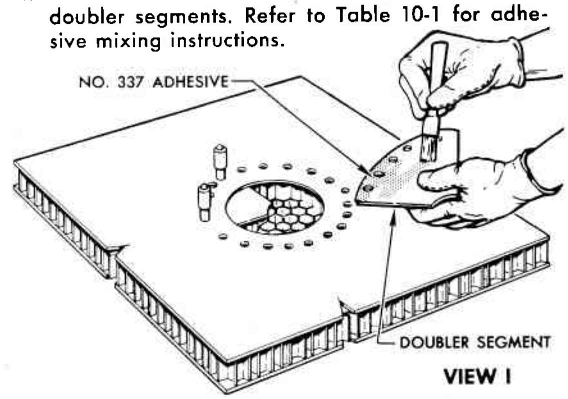
TO PERFORM STEP "p," REMOVE FLUSH PATCH PLATE, HOLD DOUBLER SEGMENTS IN PLACE BY HAND OR WITH NO. 850 TAPE, AND REMOVE CPS-332 SPRING CLAMPS. THEN REINSTALL FLUSH PATCH PLATE, PLACE -827 PLATE IN POSITION AND REINSTALL CPS-332 SPRING CLAMPS THROUGH -827 PLATE, SKIN, AND DOUBLER SEGMENTS.

q. Pilot drill holes for inside row of rivets through flush patch plates, doubler segments, and inside surface patch plate; see View D. Drill out all pilot holes in inside surface patch plate and skin to rivet hole size selected from figure 10-12. See View B for applicable rivet pattern and figure 10-12 for drilling and fastener installation procedures.

r. Remove all repair parts. Remove burrs and rough edges from repair area and repair parts. Clean repair parts and skin surfaces with Aliphatic Naphtha, Federal Specification TT-N-95.

s. Assemble flush patch plate and -805 pressure system. See figure 10-27 for detailed instructions on the -805 pressure system.

t. Mix a sufficient amount of No. 337 adhesive to cover doubler segments. Refer to Table 10-1 for adhesive mixing instructions.



VIEW I

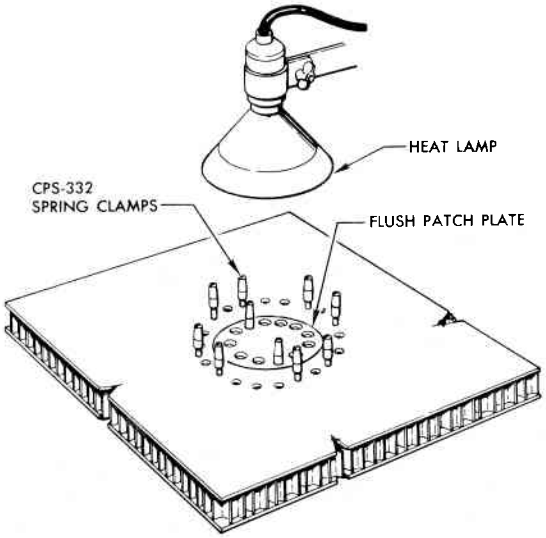
u. Apply a 1/32-inch coat of No. 337 adhesive to doubler segment and panel skin bonding surfaces.

v. Place doubler segments in position on repair one at a time and install with CPS-332 spring clamps. See View I.

w. Apply No. 337 adhesive to patch plate and doubler bonding surface.

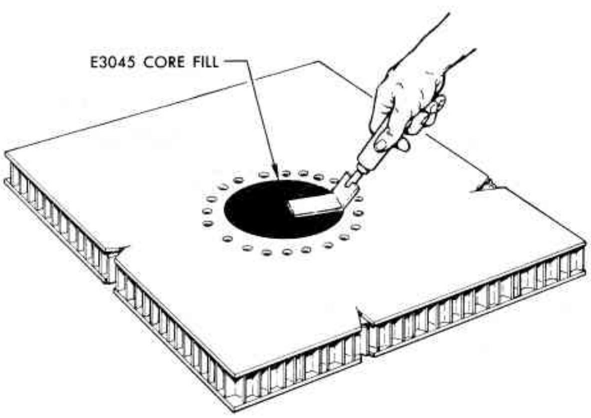
x. Install flush patch plate and -805 pressure system assembled in step "s" in position on repair and secure with CPS-332 spring clamps.

Figure 10-38. Honeycomb Repair No. 15 — Hole (Sheet 2 of 3)



VIEW J

y. With a heat lamp placed approximately 8 inches above repair, partially cure adhesive for 45 minutes.
z. Remove heat lamp and CPS-332 spring clamps from outside surface repair. Drill out pilot holes to rivet hole size and countersink holes in flush patch plate and outside surface panel skin in accordance with figure 10-12.

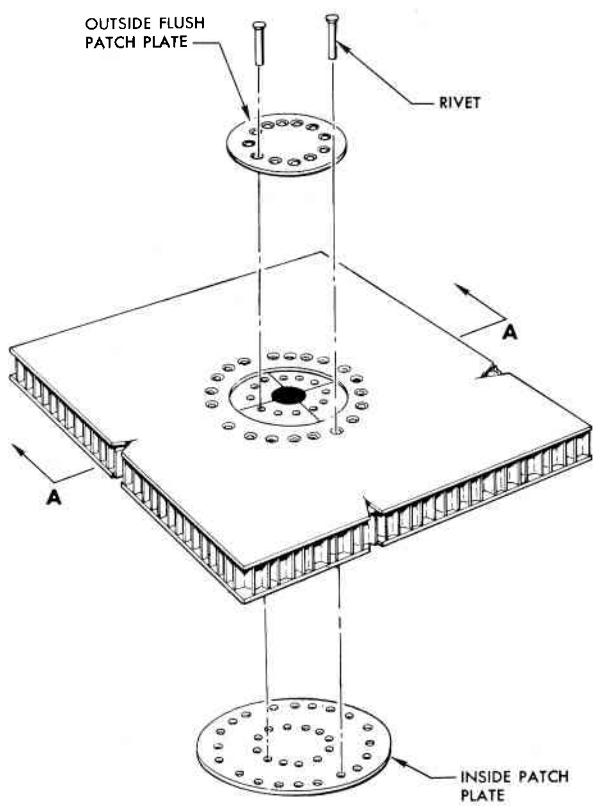


VIEW K

aa. Mix a sufficient amount of E3045 core fill to fill repair hole and a sufficient amount of No. 337 adhesive to bond inside surface patch plate to skin. Refer to Table 10-1 for mixing instructions.
ab. Fill hole flush with inside surface skin with E3045 core fill as shown in View K. Force core fill under inside surface skin, being sure cavity is filled evenly and completely.

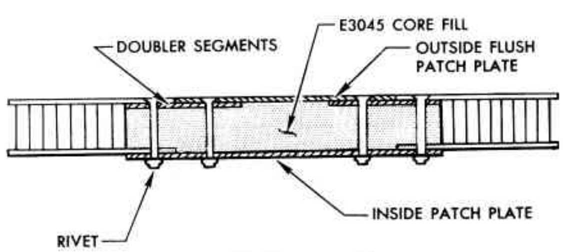
06.03.338-3

ac. Apply No. 337 adhesive to inside surface patch plate bonding surfaces and rivet patch plate in place using CR762 cherry rivets. See Section A-A.
ad. Install CR726 cherry rivets in flush patch plate and panel's outside skin surface. See Section A-A.



VIEW L

ae. Cure core fill and adhesive in accordance with Table 10-1.
af. Mill rivets installed in panel's outside skin surface flush in accordance with figure 10-12. See Section A-A for a cross section of completed repair.



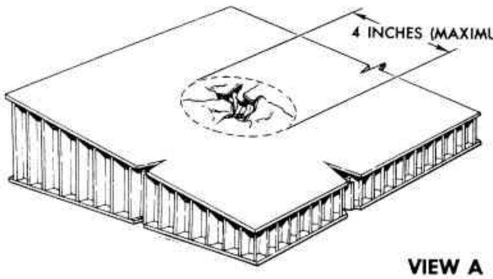
SECTION A-A

Figure 10-38. Honeycomb Repair No. 15 — Hole (Sheet 3 of 3)

REPAIR NO. 16

NOTE

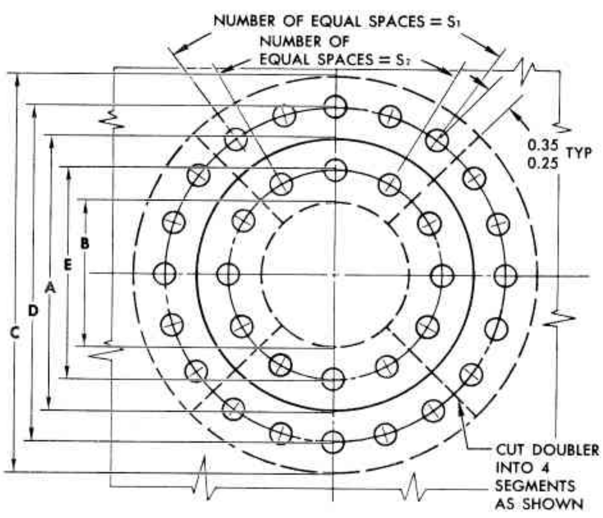
1. REPAIR NO. 16 IS SIMILAR TO REPAIR NO. 14 AND IS FOR REPAIR OF 2-, 3-, AND 4-INCH DIAMETER HOLES THROUGH BOTH SIDES OF A TAPERED PANEL. CORE FILL IS USED AS A FILLER AND FLUSH PATCH PLATES AND DOUBLERS ARE USED ON BOTH SKIN SURFACES.
2. MAKE FLUSH PATCH PLATES AND DOUBLERS FROM 0.025 GAGE 2024-T3 ALUMINUM ALLOY.
3. MAKE FLUSH PATCH PLATES 0.030 INCH LARGER THAN DIMENSION A TO ALLOW FOR TRIM FITTING MATERIAL.



VIEW A

PROCEDURE

a. Determine diameter of damaged area with damage locating templates; see figure 10-11.

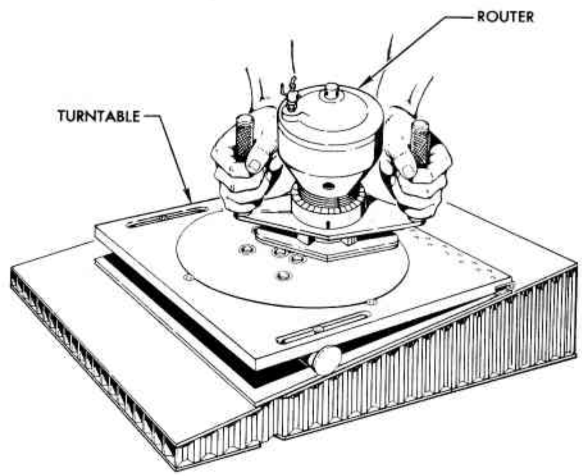


VIEW B

TABLE I							
FLUSH PATCH PLATE AND DOUBLER DIMENSIONS							
DIAMETETER OF DAMAGE	A	B	C	D	E	S_1	S_2
2.000	2.000	1.000	3.50	2.75	1.50	3	1
3.000	3.000	1.500	4.50	3.75	2.25	4	2
4.000	4.000	2.500	5.50	4.75	3.25	5	3

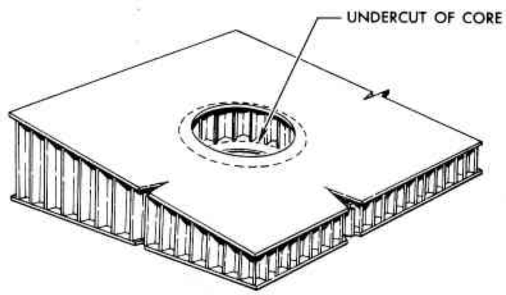
b. After a diameter has been determined, select applicable flush patch plates and doublers from Table I.

06.03.340-1



VIEW C

c. Rout out hole to a round, preselected size of applicable A diameter; see View B. Use -3 router assembly and -53 router turntable assembly; see figure 10-14 and 10-19 for operating instructions.



VIEW D

d. Undercut honeycomb core to doubler diameter as shown in View D. Use the -93 undercutter. See figure 10-21 for instructions on the use of the undercutter.

CAUTION

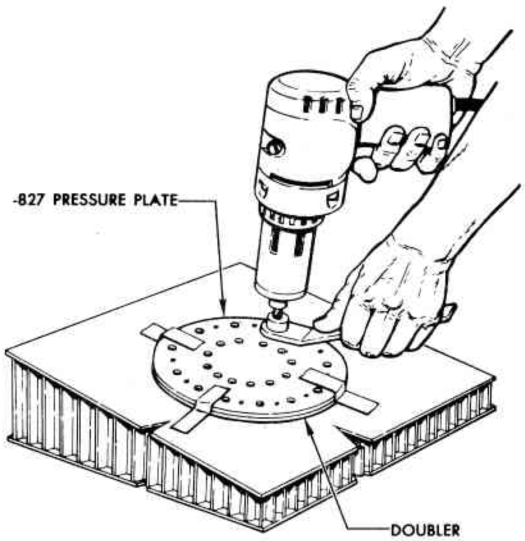
DO NOT SCRATCH UNDERSIDE OF SKIN WITH UNDERCUTTER.

e. Remove dust and cuttings from hole. Use an inspection mirror to check removal of cuttings and smoothness of undercut surfaces.

CAUTION

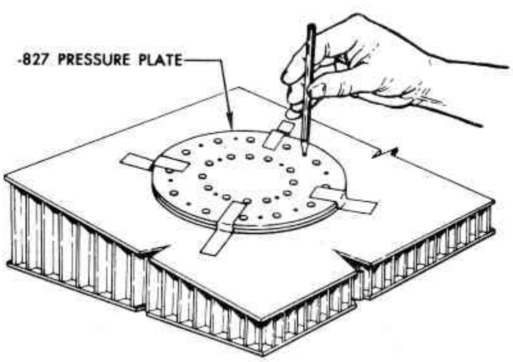
USE CLEAN WHITE GLOVES TO HANDLE ALL REPAIR PARTS TO PREVENT TRANSFER OF MOISTURE OR GREASE FROM HANDS.

Figure 10-39. Honeycomb Repair No. 16 — Hole (Sheet 1 of 3)



VIEW E

f. Center one of the doublers selected in step "b" on one skin surface of repair.
g. With doubler centered on repair, place an -827 pressure plate in position as a template on top of doubler and tape in place with No. 850 tape; see View E. See figure 10-28 for pressure plate information.
h. With an -813 drill bushing and the -827 plate, drill pilot holes for the outside row of rivet holes through doubler and the one skin surface only as shown in View E. See View B for applicable rivet pattern and figure 10-12 for drilling and fastener installation procedures. See figure 10-28 for detailed instructions on the use of the -813 drill bushing with pressure plates.



VIEW F

i. Locate and mark doubler segment cutting lines shown in View F using -827 plate. Use No. 60 holes in -827 plate to locate segment cutting lines.

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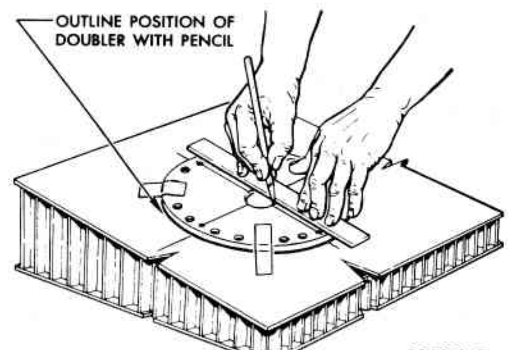
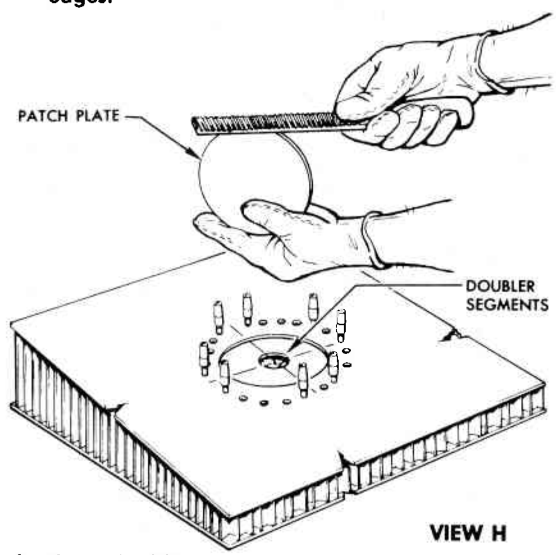


VIEW G

j. Remove -827 plate and mark position of doubler for installation.
k. Cut doubler into segments and smooth up any rough edges.



VIEW H

l. Place doubler segments on repair in positions marked in step "i" and hold in place with CPS-332 spring clamps; see View H.
m. Install one of the flush patch plates selected in step "b" on doubler segments and file plate to fit hole. See View H.

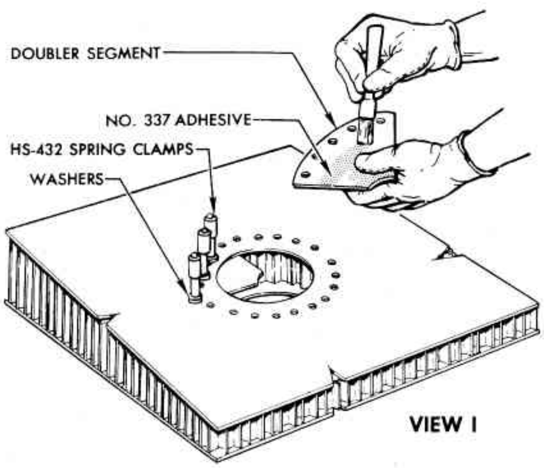
NOTE
MAXIMUM ALLOWABLE GAP BETWEEN PATCH PLATE AND SKIN IS 0.010 INCH.

n. Place -827 plate on repair in the same position placed in step "g" and "h." See View E.

NOTE
TO PERFORM STEP "n," REMOVE FLUSH PATCH PLATE, HOLD DOUBLER SEGMENTS IN PLACE BY HAND OR WITH NO. 850 TAPE, AND REMOVE CPS-332 SPRING CLAMPS. THEN REINSTALL FLUSH PATCH PLATE, PLACE -827 PLATE IN POSITION, AND REINSTALL CPS-332 SPRING CLAMPS THROUGH -827 PLATE, SKIN, AND DOUBLER SEGMENTS.

o. Pilot drill holes for inside row of rivets through flush patch plate and doubler segments; see View E.
p. Drill all pilot holes to rivet hole size selected from figure 10-12.
q. Remove -827 plate and countersink rivet holes in accordance with figure 10-12 in flush patch plate and skin only.

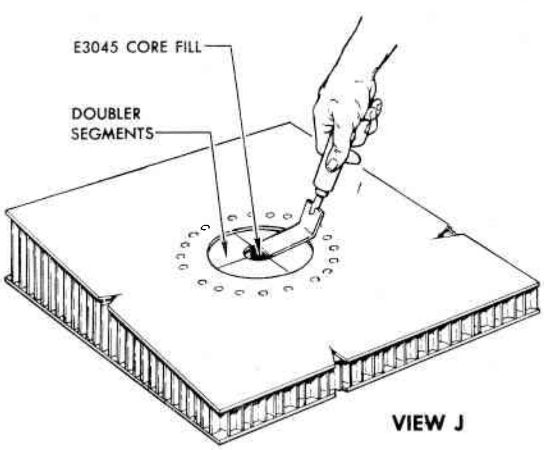
Figure 10-39. Honeycomb Repair No. 16 — Hole (Sheet 2 of 3)



r. Remove all repair parts. Remove burrs and rough edges from repair area and parts. Clean repair parts and skin surface with Aliphatic Naphtha, Federal Specification TT-N-95.
s. Repeat steps "f" through "r" on the other side of panel.
t. Apply a 1/32-inch coat of No. 337 adhesive to the bonding surfaces of one set of doubler segments. Refer to Table 10-1 for adhesive mixing instructions.
u. Install adhesive coated doubler segments and a flush patch plate in position on one side of repair only. Hold parts in place with HS-432 spring clamps and rivet flush patch plate and doubler segments in place using CR762-4-2 cherry rivets.
v. Apply adhesive to the other set of doubler segment bonding surfaces and rivet in place to the other side of panel.

NOTE

PLACE THREE AN960-10 WASHERS ON EACH HS-432 SPRING CLAMP INSTALLED IN STEP "u." INSTALL SPRING CLAMPS IN EVERY OTHER RIVET HOLE AND PULL REPAIR PARTS TOGETHER WITH SUFFICIENT PRESSURE TO SQUEEZE OUT ANY EXCESS ADHESIVE. BE SURE TO PUSH RIVETS DOWN TIGHTLY WHEN DRIVING.



w. Mix E3045 core fill in accordance with Table 10-1.
x. Fill hole as shown in View K with E3045 core fill. Force core fill under doubler segments, be sure cavity is filled evenly and completely.

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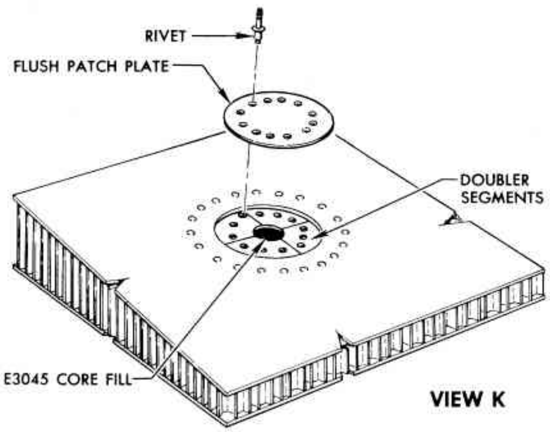
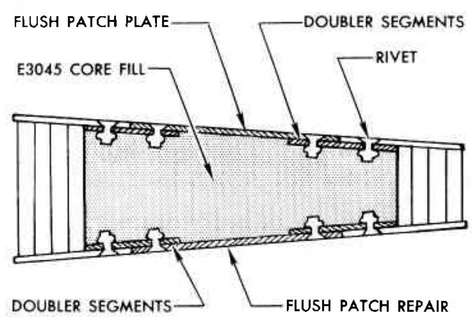
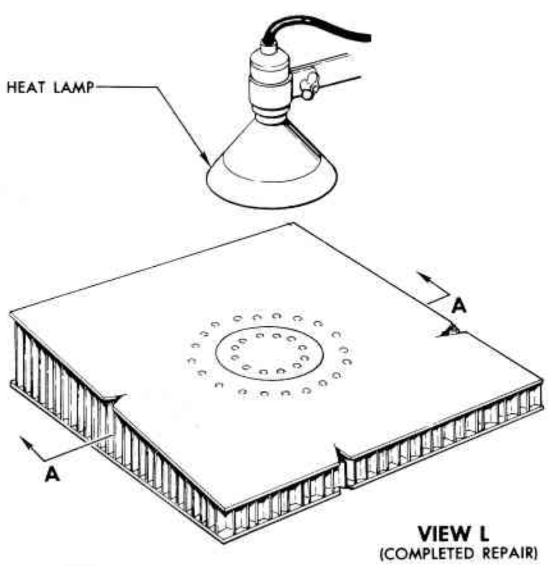


y. Install and rivet remaining flush patch plate to repair using CR762-4-2 cherry rivets. Mill rivets flush on both panel skin surfaces in accordance with figure 10-12.
z. Cure No. 337 adhesive and E3045 core fill in accordance with Table 10-1. See Section A-A for a cross section of a completed repair.



SECTION A-A



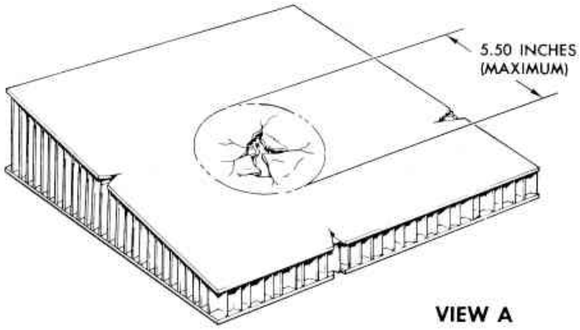
NOTE:
CURE ADHESIVE 24 TO 48 HOURS AT AMBIENT TEMPERATURE; OR 2 HOURS AT 200°F USING HEAT LAMPS. DO NOT HEAT RAPIDLY.

Figure 10-39. Honeycomb Repair No. 16 — Hole (Sheet 3 of 3)

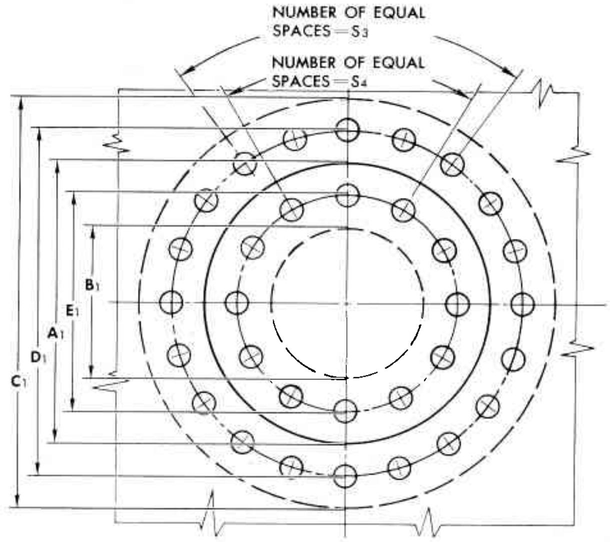
REPAIR NO. 17

NOTES

1. REPAIR NO. 17 IS ESSENTIALLY THE SAME AS REPAIR NO. 16, EXCEPT THAT IT ALLOWS REPAIR OF HOLES ON ONE SIDE OF THE HONEYCOMB PANEL UP TO 5.50 INCHES IN DIAMETER.
2. MAKE LARGE AND SMALL FLUSH PATCH PLATES AND DOUBLERS FROM 0.025 GAGE 2024-T3 ALUMINUM ALLOY.
3. MAKE LARGE AND SMALL FLUSH PATCH PLATES 0.030 INCH LARGER THAN DIMENSION A OR A1 TO ALLOW FOR TRIM FITTING MATERIAL.



VIEW A



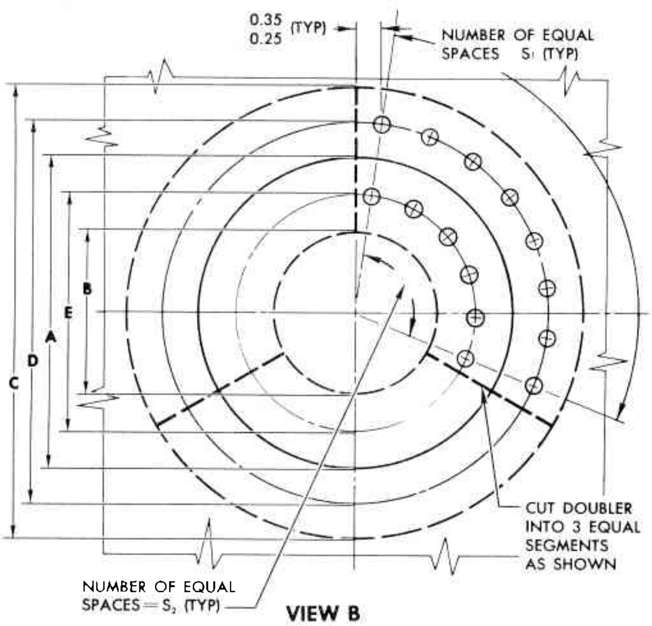
VIEW C

PROCEDURE

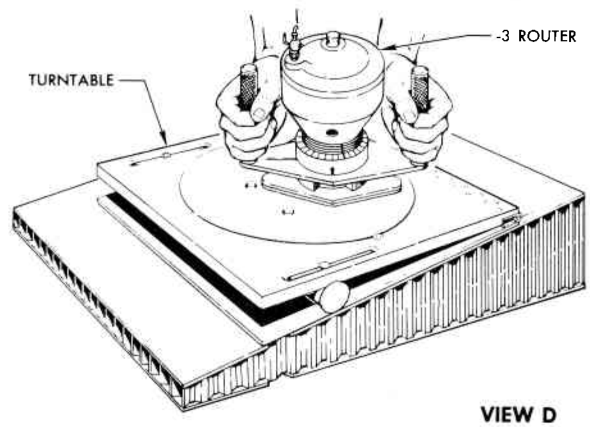
a. Determine exact diameter of damaged area on panel skin surface which exhibits the largest damaged area. Use damage locating template in accordance with Figure 10-11.

TABLE II						
SMALL FLUSH PATCH PLATE AND DOUBLER DIMENSIONS						
A_1	B_1	C_1	D_1	E_1	S_3	S_4
2.000	1.000	3.50	2.75	1.50	3	1
3.000	1.500	4.50	3.75	2.25	4	2
4.000	2.500	5.50	4.75	3.25	5	3

b. After a diameter has been determined, select applicable large and small flush patch plates and doublers from Table I and Table II (see views B and C).



VIEW B



VIEW D

TABLE I								
LARGE FLUSH PATCH PLATE AND DOUBLER DIMENSIONS								
DIAMETER OF DAMAGE	A	A_1	B	C	D	E	S_1	S_2
3.50	3.50	2.000	2.000	5.000	4.25	2.75	6	4
4.50	4.50	3.000	3.000	6.000	5.25	3.75	7	5
5.50	5.50	4.000	4.000	7.000	6.25	4.75	9	7

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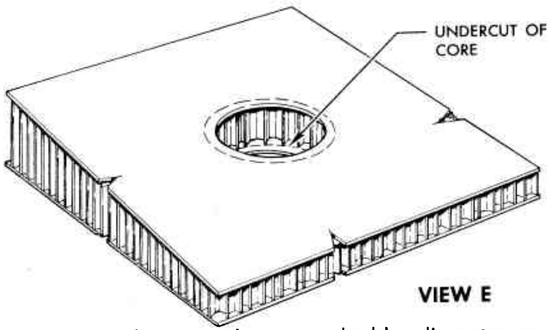
c. Install -53 router turntable assembly and -3 router assembly on panel surface with the largest diameter of damage.

d. Rout a hole through both sides of the panel to a round, preselected size of applicable B diameter; see View B and D. See figure 10-14 and 10-19 for router and turntable installation and operating instructions.

e. With -3 router assembly and -53 router turntable assembly, rout out honeycomb core to adhesive on panel's opposite skin to an applicable A diameter; see View B.

Figure 10-40. Honeycomb Repair No. 17—Hole (Sheet 1 of 3)



VIEW E

f. Undercut honeycomb core to doubler diameter as shown in View E. Use the -93 undercutter. See figure 10-21 for instructions on the use of the undercutter.

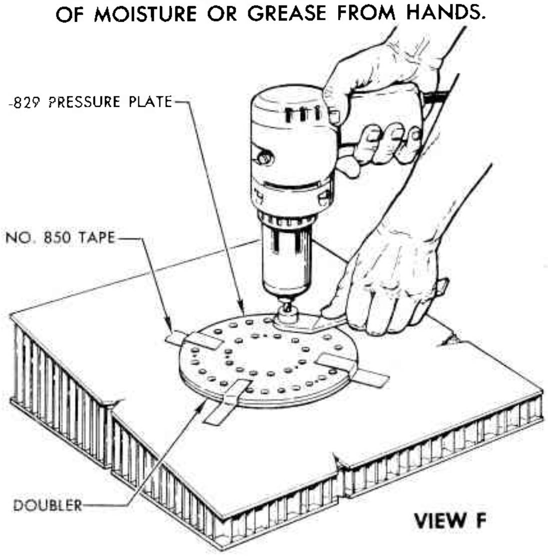
CAUTION

DO NOT SCRATCH UNDERSIDE OF SKIN WITH UNDERCUTTER.

g. Remove dust and cuttings from hole. Use an inspection mirror to check removal of cuttings and smoothness of undercut surfaces. Mark damaged area center lines on the panel's other skin surface with damage locating template in accordance with figure 10-11.

CAUTION

USE CLEAN WHITE GLOVES TO HANDLE ALL REPAIR PARTS TO PREVENT TRANSFER OF MOISTURE OR GREASE FROM HANDS.



VIEW F

h. Center large doubler, selected in step "b," in place on skin surface of repair.

i. With large doubler centered on repair, place an -829 pressure plate in position as a template on top of doubler. Tape doubler and pressure plate in place with No. 850 tape; see view F. See figure 10-28 for pressure plate information.

j. With an -813 drill bushing and the -829 plate, drill pilot holes for the outside row of rivet holes through doubler and skin only as shown in View F. See View B for applicable rivet pattern and figure 10-12 for drilling and fastener installation procedures. See figure 10-28 for detailed instructions on the use of the -813 drill bushing with pressure plates.

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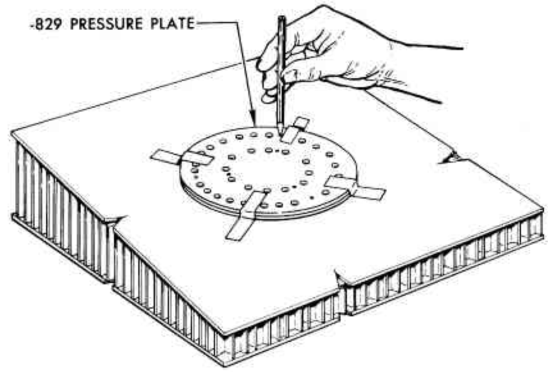
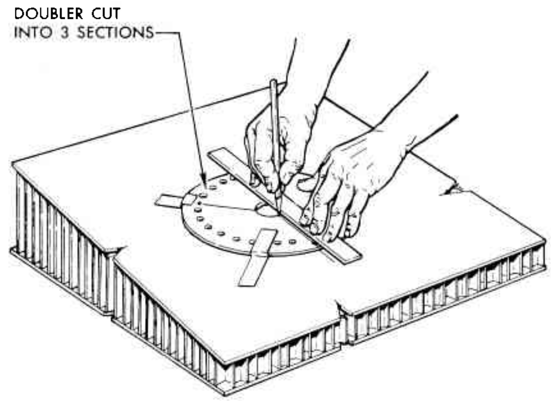


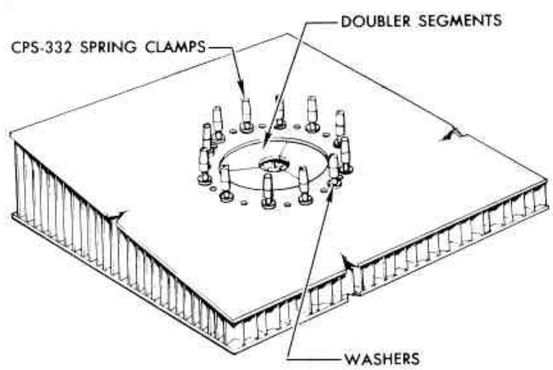
VIEW G

k. Locate and mark large doubler segment cutting lines shown in View G using -829 plate. Use No. 60 holes in -829 plate to locate segment cutting lines.



VIEW H

l. Remove -829 plate and mark position of doubler for reinstallation.

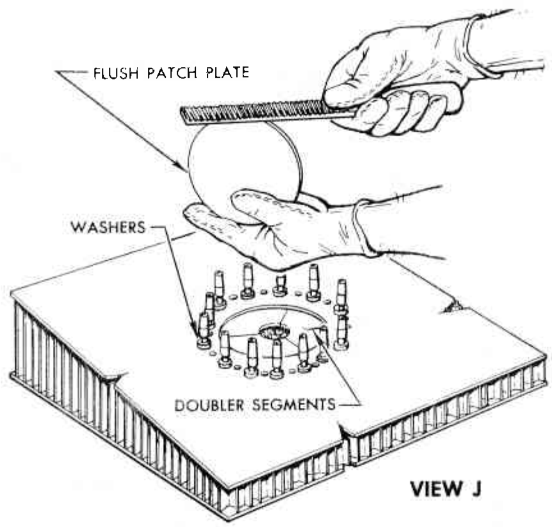


VIEW I

m. Cut large doubler into segments and smooth up any rough edges; see View B.

n. Place large doubler segments on repair in positions marked in step "l" and hold in place with CPS-332 spring clamps; see View I.

Figure 10-40. Honeycomb Repair No. 17—Hole (Sheet 2 of 3)



o. Install large flush patch, selected in step "b," on doubler segments and file plate to fit hole. See View J.

NOTE

MAXIMUM ALLOWABLE GAP BETWEEN PATCH PLATE AND SKIN IS 0.010 INCH.

p. Place -829 plate on repair in the same position placed in steps "i" and "j." See View F.

NOTE

TO PERFORM STEP "p," REMOVE FLUSH PATCH PLATE, HOLD DOUBLER SEGMENTS IN PLACE BY HAND OR WITH NO. 850 TAPE, AND REMOVE CPS-332 SPRING CLAMPS. THEN REINSTALL FLUSH PATCH PLATE, PLACE -829 PLATE IN POSITION, AND REINSTALL CPS-332 SPRING CLAMPS THROUGH -829 PLATE, SKIN, AND DOUBLER SEGMENTS.

q. Pilot drill holes for inside row of rivets through large flush patch plate and doubler segments.

r. Drill out all pilot holes to rivet hole size selected from figure 10-12.

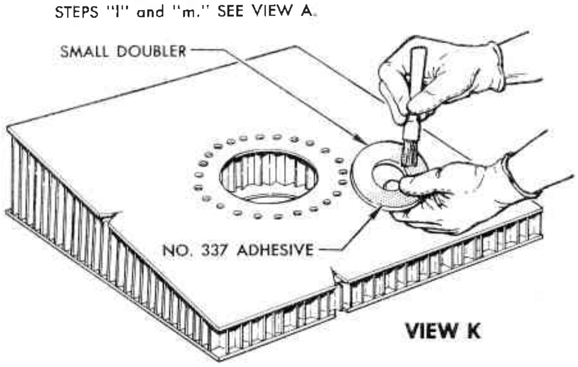
s. Remove -829 plate and countersink rivet holes in accordance with figure 10-12 in large flush patch plate and skin only.

t. Remove repair parts. Remove burrs and rough edges from repair area and large repair parts. Clean repair parts and skin surface with Aliphatic Naphtha, Federal Specification TT-N-95.

u. With small patch plate and small doubler selected in step "b," and an -827 pressure plate, repeat steps "h" through "k" and "n" through "u" on the panel's other skin surface.

NOTE

DO NOT CUT SMALL DOUBLER INTO SEGMENTS; OMIT STEPS "l" and "m." SEE VIEW A.



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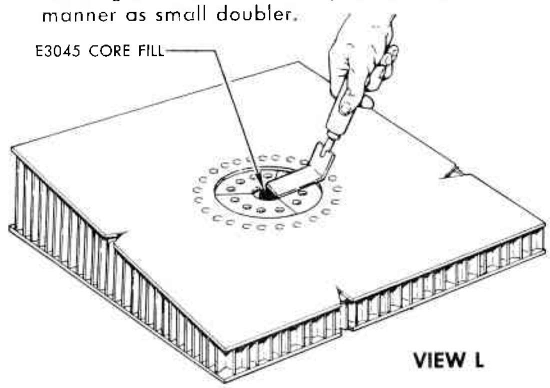
v. Apply a 1/32-inch coat of No. 337 adhesive to the bonding surface of the small doubler. Refer to Table 10-1 for adhesive mixing instructions.

w. Install small adhesive coated doubler and flush patch plate in position. Hold parts in place with HS-432 spring clamps and rivet small flush patch plate and doubler to skin using CR762-4-2 cherry rivets.

NOTE

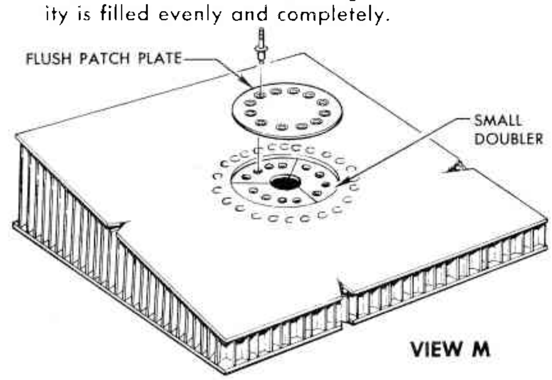
PLACE THREE AN960-10 WASHERS ON EACH HS-432 SPRING CLAMP INSTALLED IN STEP "w." INSTALL SPRING CLAMPS IN EVERY OTHER RIVET HOLE AND PULL REPAIR PARTS TOGETHER WITH SUFFICIENT PRESSURE TO SQUEEZE OUT ANY EXCESS ADHESIVE. BE SURE TO PUSH RIVETS DOWN TIGHTLY WHEN DRIVING.

x. Apply No. 337 adhesive to large doubler segment bonding surfaces and rivet to panel skin in the same manner as small doubler.



y. Mix E3045 core fill in accordance with Table 10-1.

z. Fill hole as shown in View L with E3045 core fill. Force core fill under doubler segments. Be sure cavity is filled evenly and completely.



aa. Install and rivet large flush patch plate to repair using CR762-4-2 cherry rivets.

ab. Mill rivets flush on both panel skin surfaces in accordance with figure 10-12.

ac. Cure No. 337 adhesive and E3045 core fill in accordance with Table 10-1.

ad. See Section A-A for a cross section of completed repair.

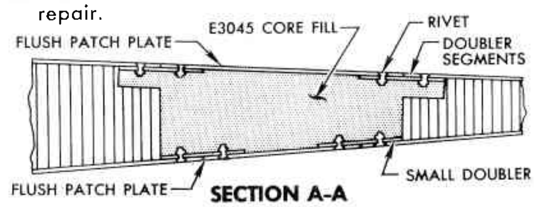
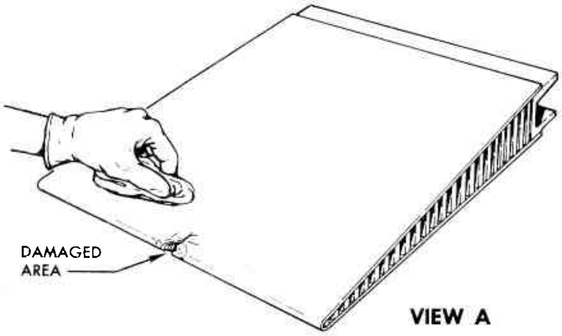


Figure 10-40. Honeycomb Repair No. 17—Hole (Sheet 3 of 3)

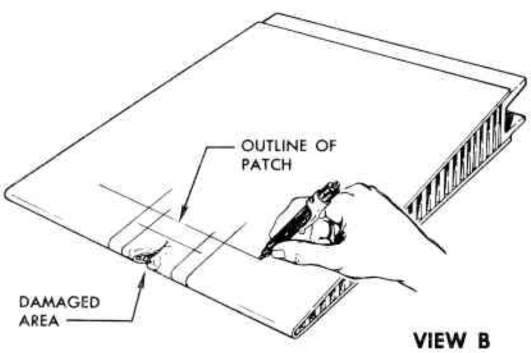
REPAIR NO. 18

NOTE
THE MAXIMUM ALLOWABLE DIAMETER OF THIS REPAIR IS 2.00 INCHES.

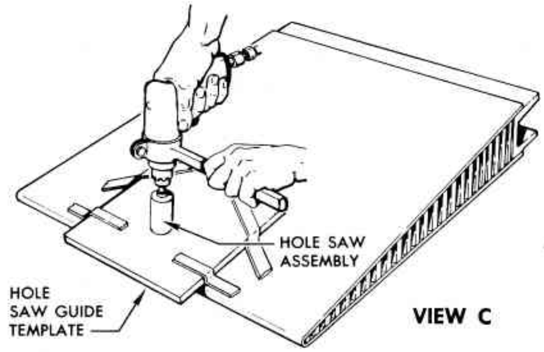


PROCEDURE

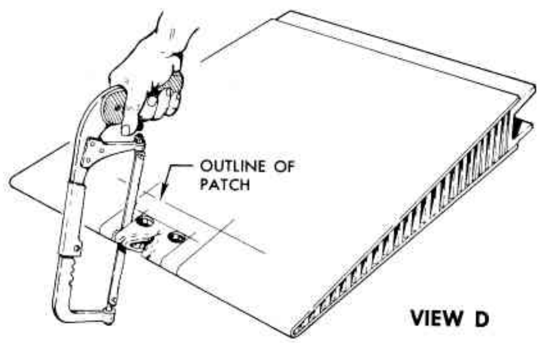
a. Clean damaged area with Methyl-Ethyl-Ketone, Specification, TT-M-261.



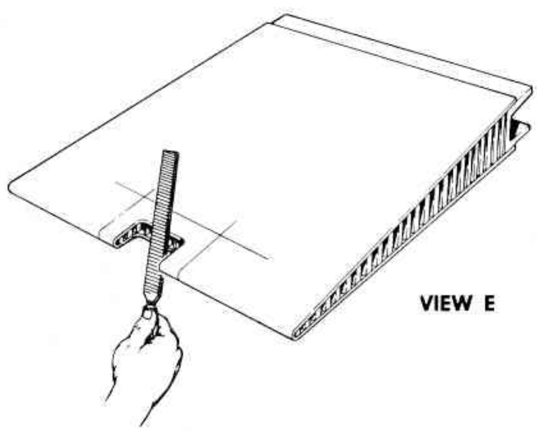
b. Using grease pencil, draw smallest rectangle that will enclose the damaged area as shown.
c. Draw outline of patch. Continue patch outline over trailing edge and complete on bottom side of part.



d. Locate -33 hole saw guide template over one corner of the damaged area outline so that hole saw will cut 0.50 corner radius. See figure 10-16.
e. With -55 hole saw assembly and slow speed drill motor, saw hole thru as shown on figure 10-18.
f. Repeat steps "d" and "e" to cut hole in other corner of damage area.



g. Using common hack saw, cut along lines established in step "b" from trailing edge to both corner radii and along forward line parallel to trailing edge.



h. File edges of saw cuts smooth.

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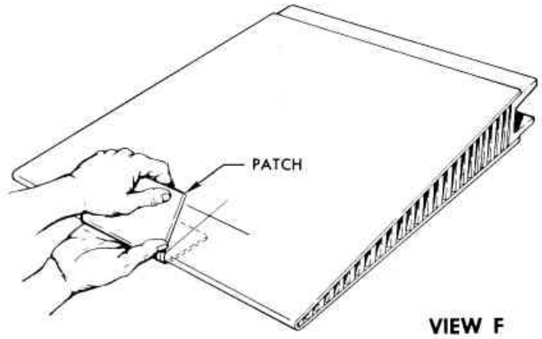
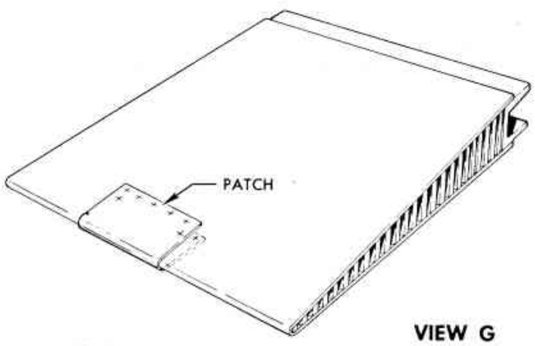
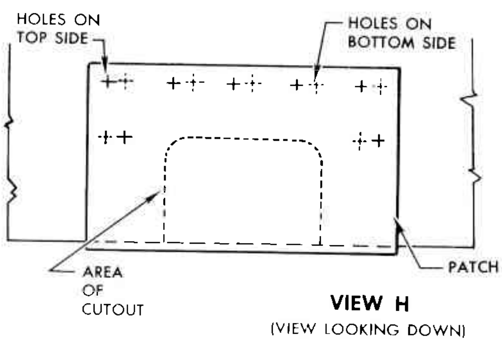
Figure 10-41. Honeycomb Repair No. 18 (Trailing Edge Damage) (Sheet 1 of 3)

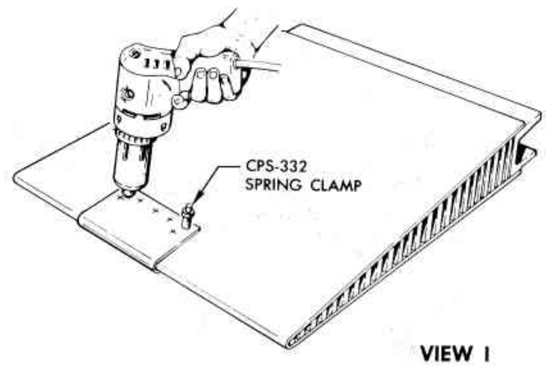


i. Bend 2024-T3 aluminum alloy patch (0.020 thick) to conform to contour of trailing edge.
j. Trim patch to correct dimensions.

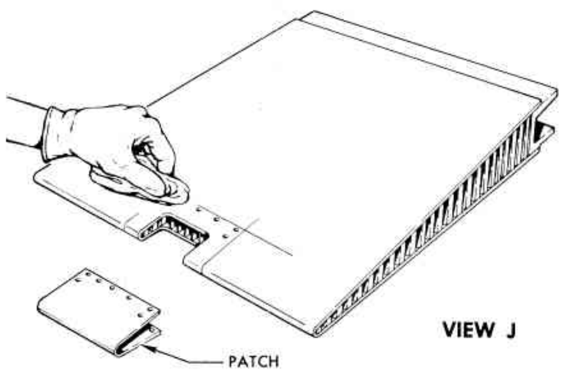




k. Lay out hole pattern, staggering rivets so that those on bottom side (shown in dotted lines) do not interfere with those on top side.



l. Holding patch firmly in place, drill two pilot holes in top side of patch in opposite corners and install CPS-332 spring clamps. Drill thru patch and skin only, using depth stop.
m. On bottom side of patch, drill two additional pilot holes in opposite corners and install CPS-332 spring clamps.
n. Drill remaining pilot holes.
o. Finish drill pilot holes with No. 30 drill for rivets.
p. Remove spring clamps and patch.
q. Blow out shavings and filings with clean, dry, filtered air.



r. Clean surfaces to be covered by patch as outlined in figures 10-13, steps 5 thru 10.
s. Clean patch with Aliphatic Naphtha, Specification TT-N-95.

NOTE
USE CLEAN WHITE GLOVES TO HANDLE ALL REPAIR PARTS TO PREVENT TRANSFER OF MOISTURE OR GREASE FROM HANDS.

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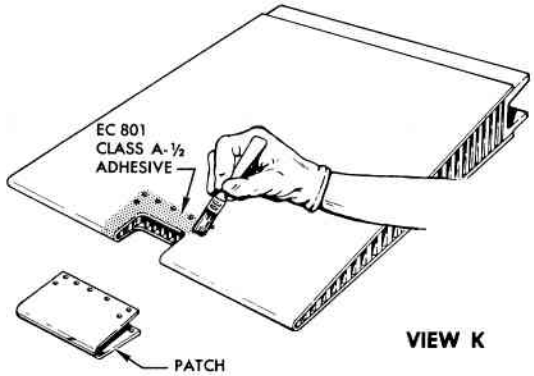
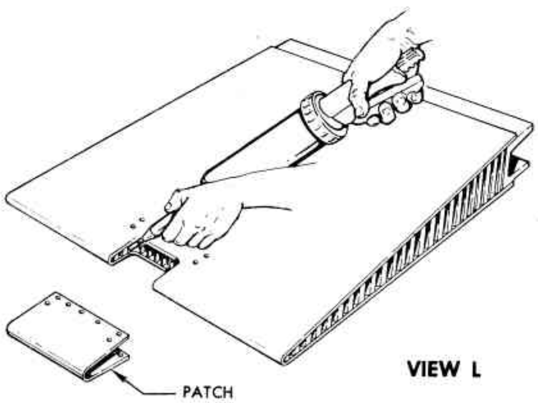
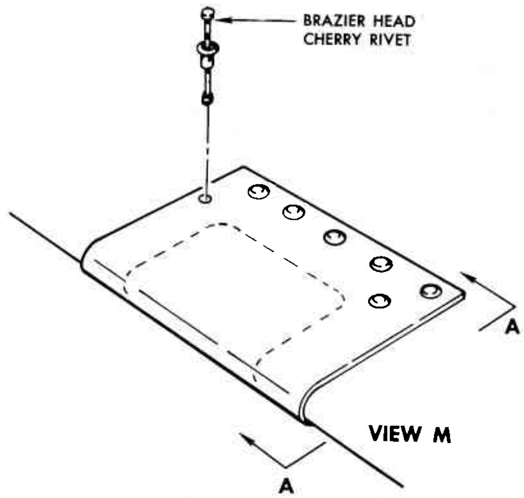
Figure 10-41. Honeycomb Repair No. 18 (Trailing Edge Damage) (Sheet 2 of 3)



t. Brush coat the bond areas with EC801 Class A-1/2 adhesive. Refer to Table 10-1 for mixing instructions. Use enough adhesive to squeeze out when the parts are pressed together.
u. Fill exposed core cells with adhesive used in preceding step.



v. Install patch. Press firmly by hand to work out trapped air and ensure good adhesive contact.



w. Install 1/8-inch brazier head cherry rivets. Exert pressure on rivet heads during pulling operation.
x. Clean surplus adhesive from repair area.

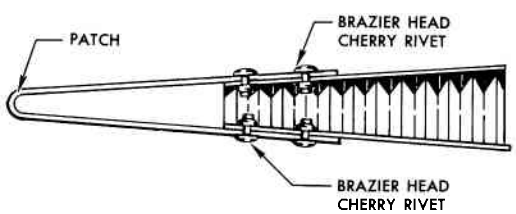


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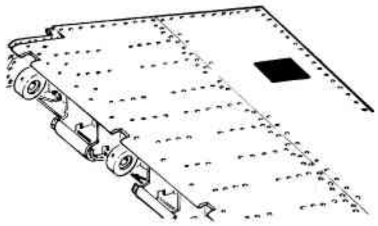
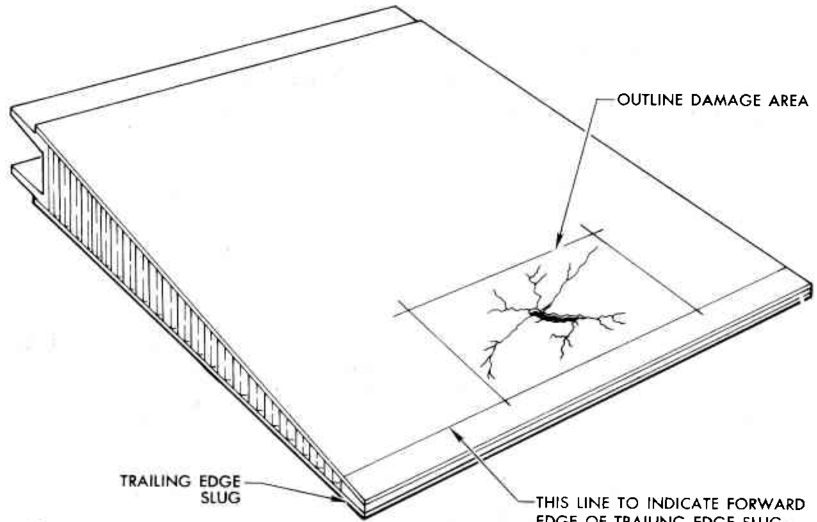
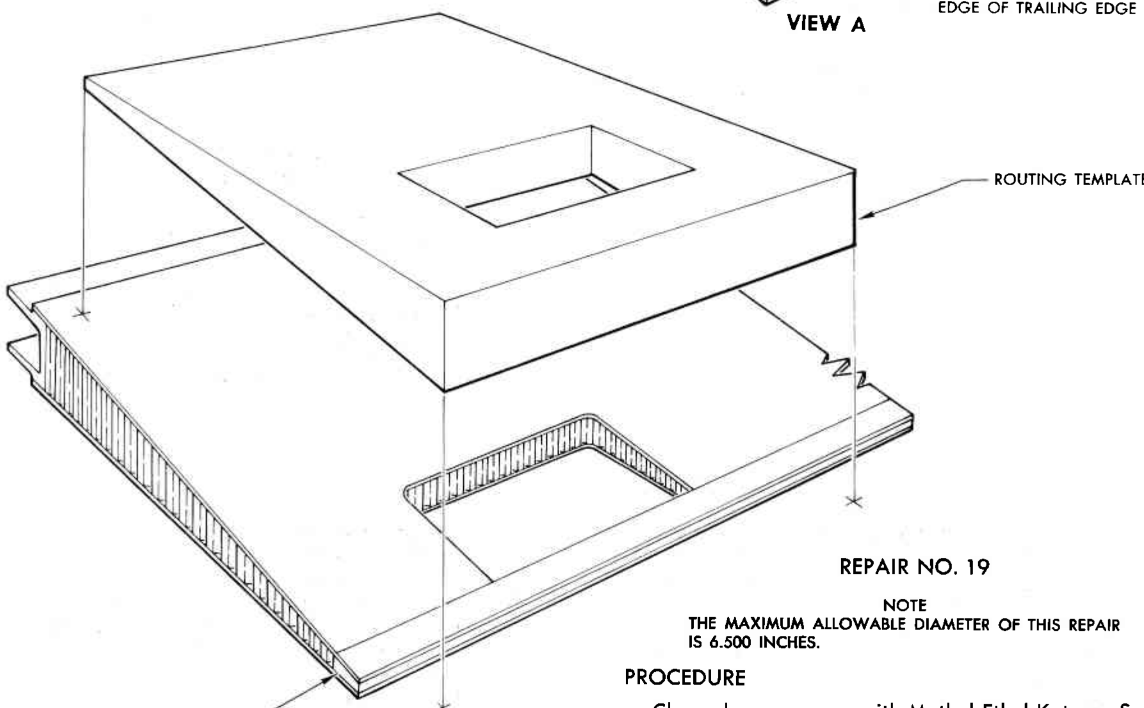
Figure 10-41. Honeycomb Repair No. 18 (Trailing Edge Damage) (Sheet 3 of 3)



AREA OF ASSUMED DAMAGE



VIEW A



VIEW B

REPAIR NO. 19

NOTE
THE MAXIMUM ALLOWABLE DIAMETER OF THIS REPAIR IS 6.500 INCHES.

PROCEDURE

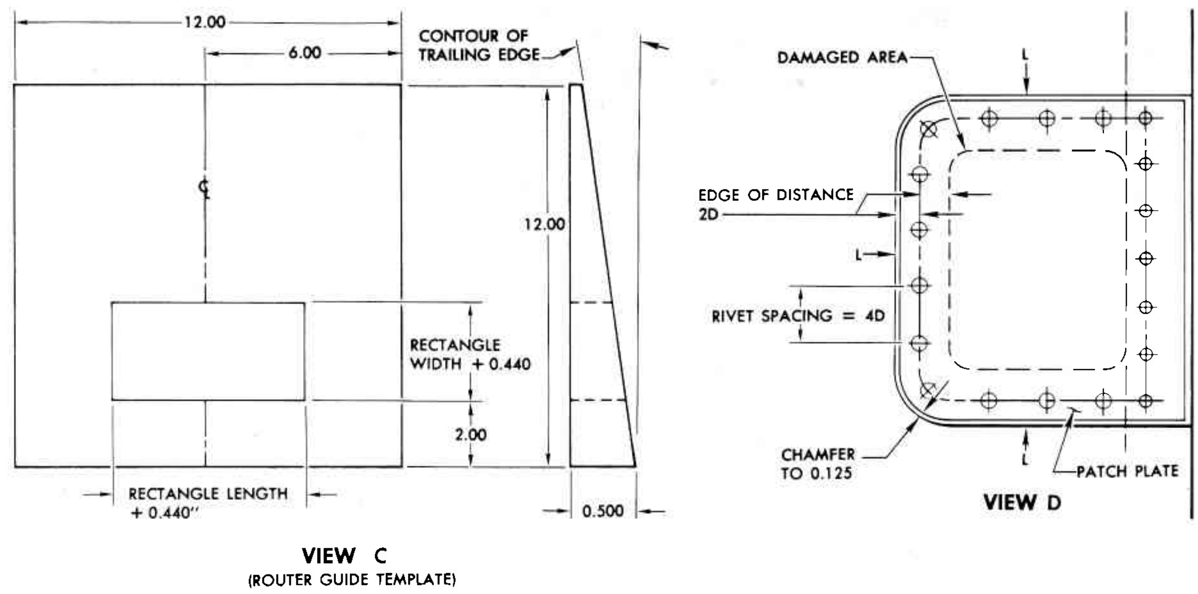
a. Clean damage area with Methyl Ethyl Ketone, Specification TT-M-261.
b. Outline damage area as shown in view A.
c. Determine existing skin gage of damage area, then find dimension L from view D and Table I.
d. Fabricate a router guide template from a piece of plywood 1/2 inch thick and 12 inches square; see views B and C.
e. Place strips of double coated scotch tape on lower surface of template to hold it in place on skin. Place cut out in template over damage area and mark outline of template.
f. Install template on reverse side of damage area and mark template outline exactly opposite outline made in step "e."

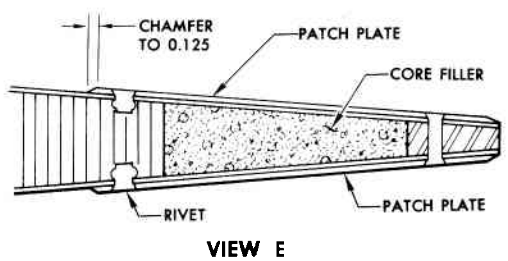
TABLE I

EXSTING SKIN GAGE	L DIM.
0.012 0.016 0.020	0.625
0.025	0.750
0.032	0.960
0.040	1.200
0.050	1.500

06.03.300-1 A

Figure 10-42. Honeycomb Repair No. 19 (Trailing Edge Damage) (Sheet 1 of 2)





g. Center template over upper damage area marked in step "e" and rout out damage area through existing skin. Use -3 router assembly; see figure 10-14 for operating instructions.
h. Install template on opposite side. Use coordinating marks in step "f" so template is exactly opposite area routed in step "g" and rout out area through existing skin.
i. Use a sharp knife or scroll saw and trim out damaged core. Use existing skin edges and trailing edge slug as the trim boundary; see view B.
j. Fabricate patch plates. Form, trim and chamfer as shown in view D.
k. Clean faying surfaces of patch plates and existing skin. Use steps 5 through 10, figure 10-13.
l. Mix core filler E3045 as directed by manufacturer.
m. Apply adhesive No. 337 to lower skin bond area.
n. Install lower patch plate in position and secure with tape No. 850; remove excessive adhesive.
o. Fill core cavity with core filler to 1/32 inch above skin level in center of cavity.
p. Apply adhesive No. 337 to upper patch plate and tape in place.
q. Drill rivet pattern and countersink holes. Refer to view D for rivet pattern.
r. Install P56S-134-100-8 rivets. Mill down protruding rivet heads.
s. Use a heat lamp and allow repair to cure. (See Table 10-I.) See view E for cross section of completed repair.

06.03.300-2A

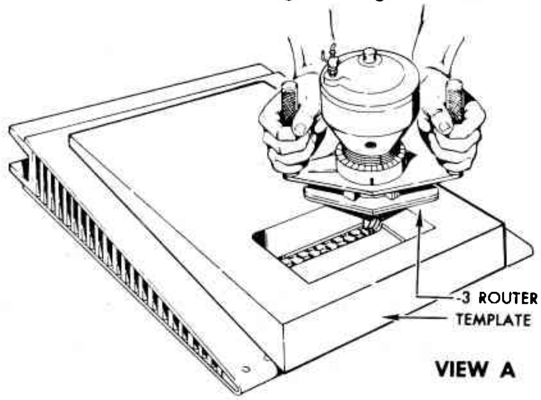
Figure 10-42. Honeycomb Repair No. 19 (Trailing Edge Damage) (Sheet 2 of 2)

REPAIR NO. 20

NOTE
THE MAXIMUM ALLOWABLE DIAMETER OF THIS REPAIR IS 6.500 INCHES.

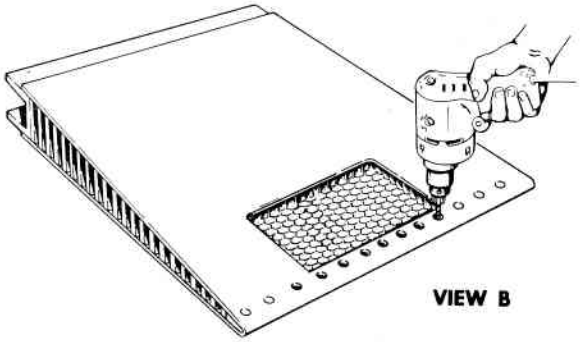
PROCEDURE

a. Accomplish steps a through e of figure 10-42.



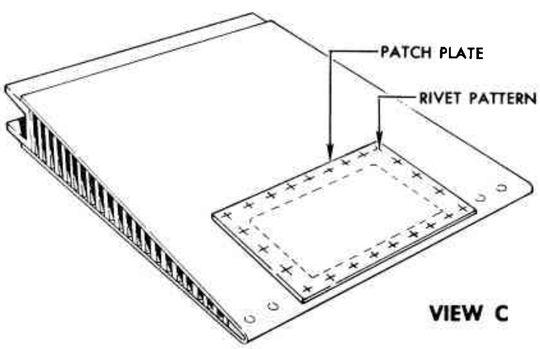
VIEW A

b. Center template over damage area and install -3 router assembly. See figure 10-14 for operating instructions.
c. Remove skin from damage area using router cuts of 0.010 inch or less. Do not remove skin over trailing edge slug.
d. Route out the damaged core to the required depth.



VIEW B

e. Drill and punch out original trailing edge rivets (if installed) in patch plate area.



VIEW C

f. Fabricate 2024-T3 aluminum alloy patch plate from 0.025 material. Trim, chamfer, form, and lay out rivet pattern as shown in view C. Pick up existing rivet holes in trailing edge (if holes exist).

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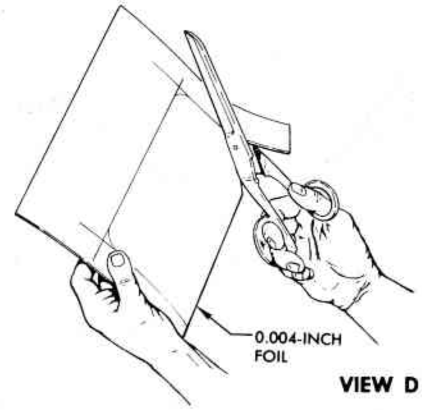
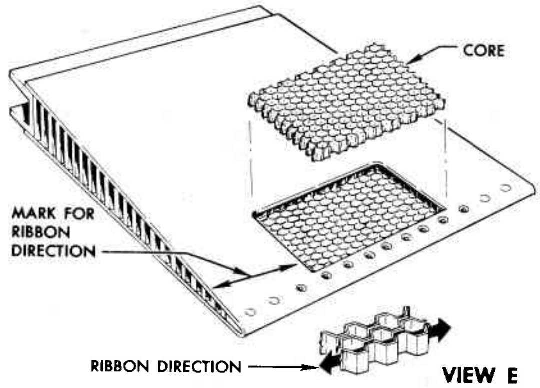


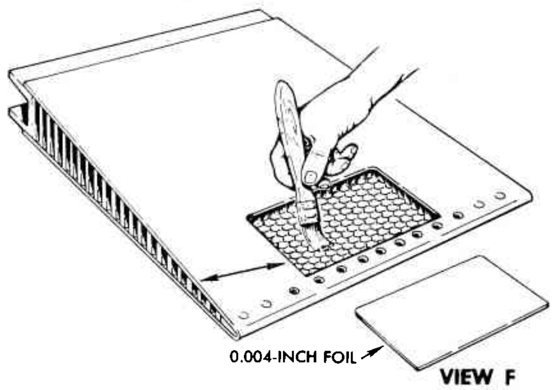
VIEW D

g. Cut out sheet of 0.004-inch foil to fit bottom of core cavity.



VIEW E

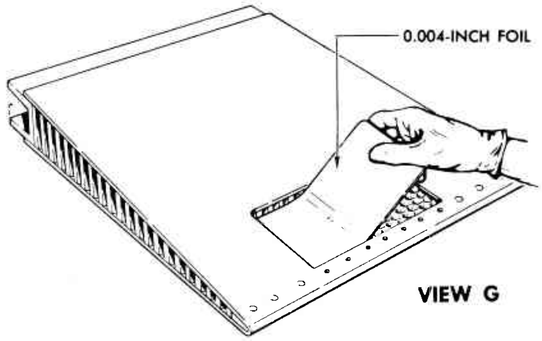
h. Cut core plug 1/32-inch oversize with ribbon direction same as core in the part.



VIEW F

i. Blow shavings out of core cavity with clean, dry, filtered air.
j. Clean faying surfaces of patch plate and existing skin as outlined in figure 10-13, steps 5 thru 10.
k. Clean core plug and foil sheet with Aliphatic Naphtha, Specification TT-N-95.
l. Brush coat one side of 0.004-inch foil 1/16 to ⅛-inch thick with No. 338 adhesive.

Figure 10-43. Honeycomb Repair No. 20 (Trailing Edge Damage) (Sheet 1 of 2)

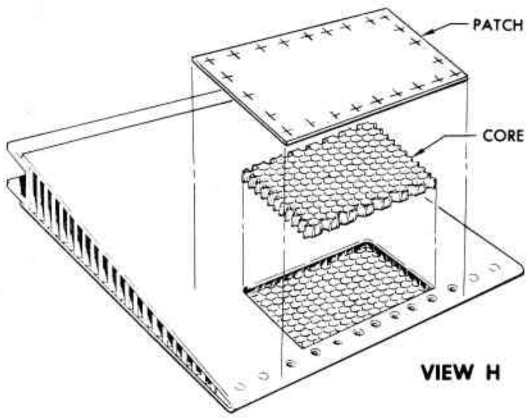


VIEW G

m. Push foil (adhesive side down) into core cavity.

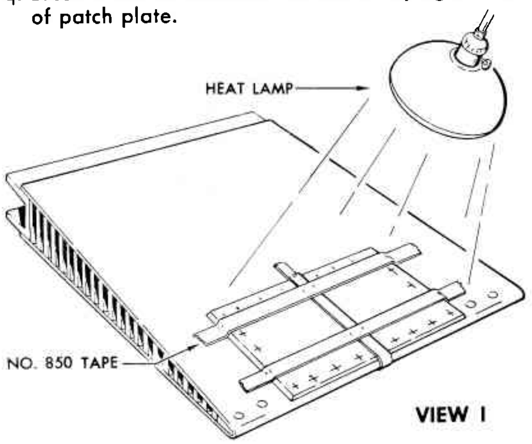
NOTE

USE CLEAN WHITE GLOVES TO HANDLE ALL REPAIR PARTS TO PREVENT TRANSFER OF MOISTURE OR GREASE FROM HANDS.



VIEW H

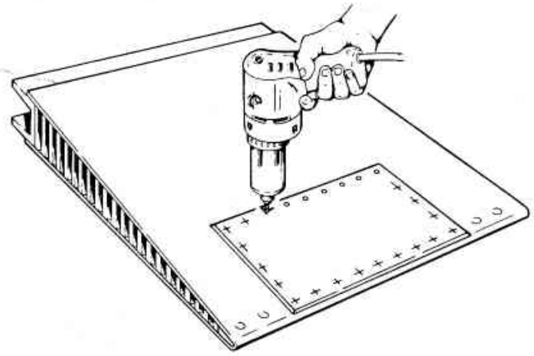
n. Coat foil and core at sides of cavity heavily with No. 338 adhesive.
o. Install core plug; push down firmly.
p. Brush a light coat of adhesive No. 337 on faying surfaces of part.
q. Brush a coat of adhesive No. 338 on faying surfaces of patch plate.



VIEW I

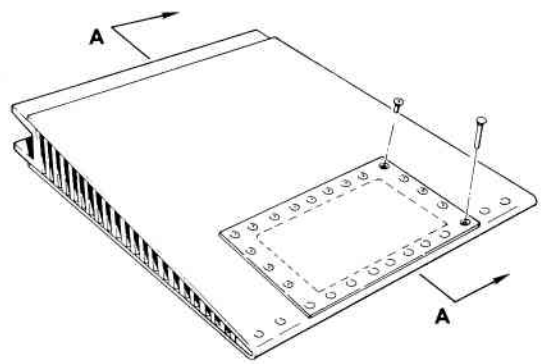
r. Install patch plate in position and secure with No. 850 tape.
s. Allow repair to cure using heat lamp. See Table 10-1.

06.03.341-2



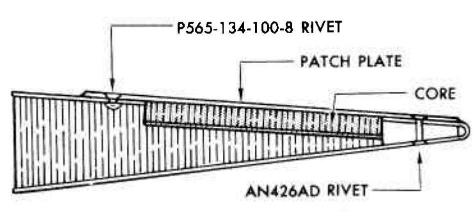
VIEW J

t. Drill holes in patch plate along trailing edge the same size as the original holes. Dimplet patch plate into old countersink. If no holes exist in trailing edge, drill No. 39 holes as shown in view B. Countersink patch plate only.
u. Drill No. 29 holes in skin area of patch plate and countersink. Drill thru patch plate and skin only using depth stop.



VIEW K

v. Install 426AD rivets in trailing edge and double flush.
w. Install P565-134-100-8 explosive rivets in skin area of patch.
x. Mill down all protruding rivet heads.
y. Clean surplus adhesive from repair area.



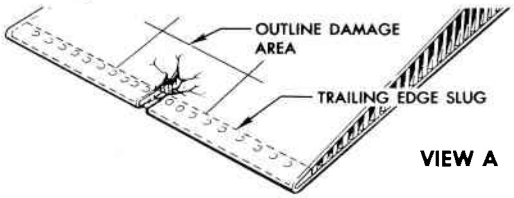
SECTION A-A

Figure 10-43. Honeycomb Repair No. 20 (Trailing Edge Damage) (Sheet 2 of 2)

REPAIRS NO. 21 AND 22

NOTE

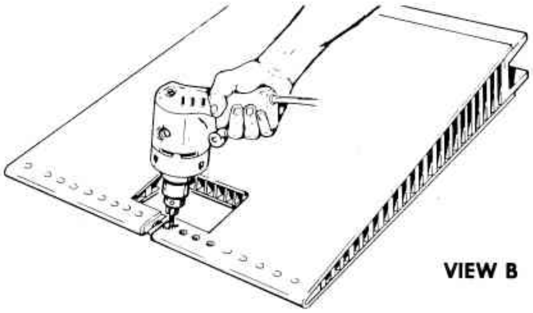
THESE REPAIRS ARE USED WHEN IT IS NECESSARY TO REPLACE A SECTION OF THE TRAILING EDGE SLUG. THEY ARE USED IN CONJUNCTION WITH HONEYCOMB TRAILING EDGE REPAIR NO. 19, DESCRIBED IN FIGURE 10-42.



VIEW A

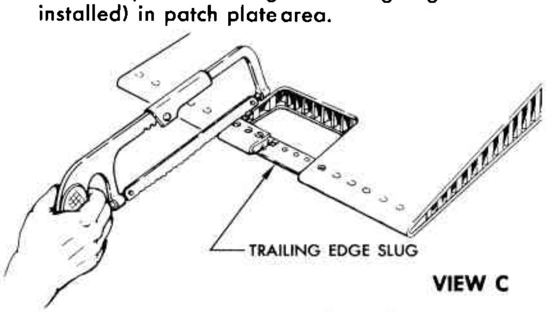
PROCEDURE

a. Accomplish steps "a" thru "i" of figure 10-42.



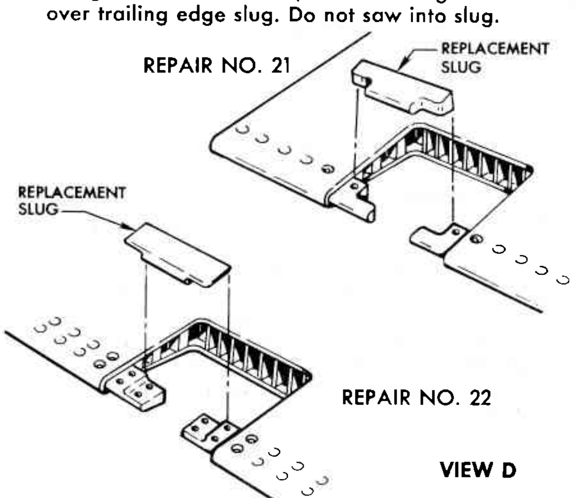
VIEW B

b. Drill and punch out original trailing edge rivets (if installed) in patch plate area.



VIEW C

c. Using common hack saw, cut remaining skin from over trailing edge slug. Do not saw into slug.



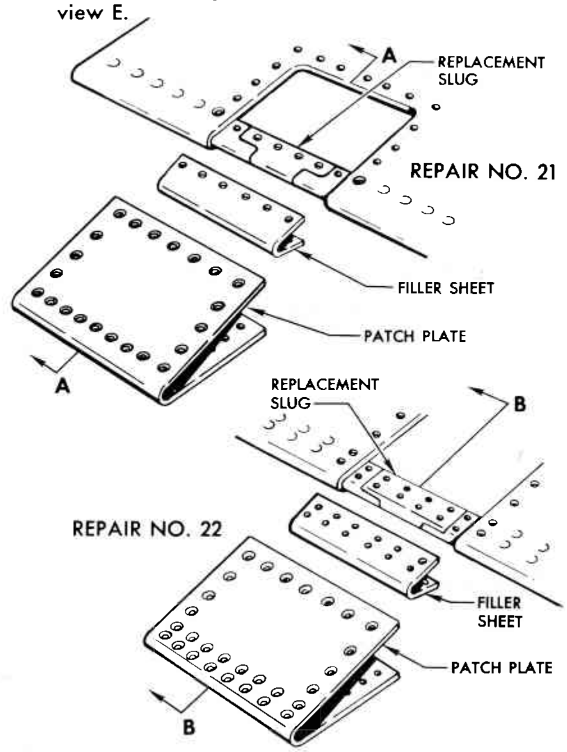
VIEW D

.06.03.342

d. Outline applicable slug repair to be made as shown in view D.

e. Using a hack saw and mill file, make appropriate cuts in trailing edge slug as shown.

f. Fabricate replacement slug from a section of a production trailing edge slug or from 2024-T3 stock. Pick up trailing edge rivet pattern as shown in view E.

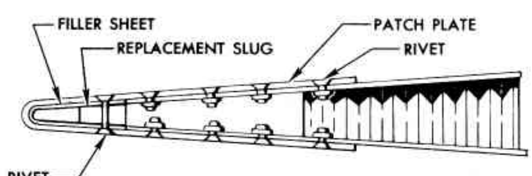


VIEW E

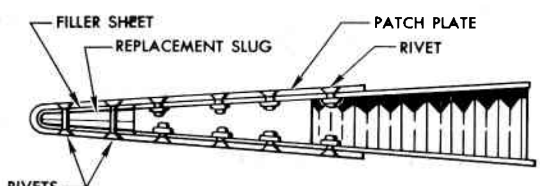
g. Fabricate filler sheet from 2024-T3 aluminum alloy the same thickness as the adjacent skin.

h. Fabricate patch plate from one piece of 2024-T3 aluminum alloy 0.025-inch thick to wrap around trailing edge as shown in view E.

i. Accomplish steps "j" thru "s" of figure 10-42.

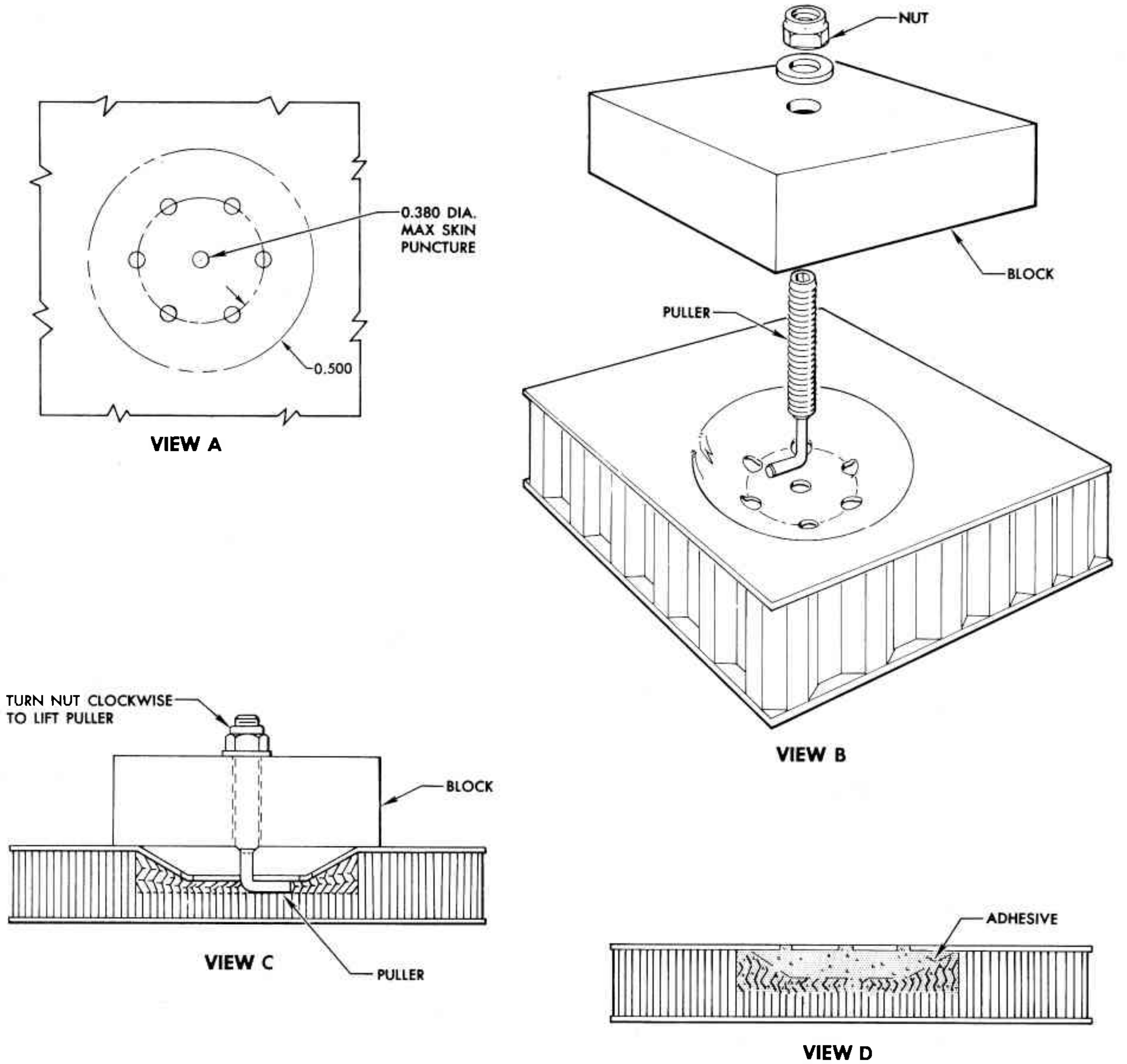


SECTION A-A



SECTION B-B

Figure 10-44. Honeycomb Repair No's. 21 and 22 (Trailing Edge Damage)



REPAIR NO. 23

NOTE
THIS REPAIR IS USED FOR DENTS WHICH DO NOT EXCEED 2.500 INCHES IN DIAMETER.

PROCEDURE

a. Prepare and clean damage area. Use steps 1, 2, 5 thru 9 and 10 of figure 10-13.
b. Lay out and centerpunch holes with proper hole pattern template as shown in view A. See figure 10-17 for illustration of -67, -69, and -71 hole pattern templates.
c. Drill holes (0.161 inch diameter) through skin only. Use a No. 20 drill and a depth stop.
d. Fabricate dent puller as shown in Views E and F.
e. Install dent puller by inserting in drilled holes; see View B.
f. Remove dent by tightening nut; see View C.
g. Select and mix proper adhesive from Table 10-1.
h. Inject adhesive in drill holes and fill depression under skin to 1/32 inch above existing skin level.
i. Allow adhesive to cure, then sand down to contour of existing skin. See View D for cross-section of completed repair.

06.03.301-1B

Figure 10-45. Honeycomb Repair No. 23 (Crack or Dent) (Sheet 1 of 2)

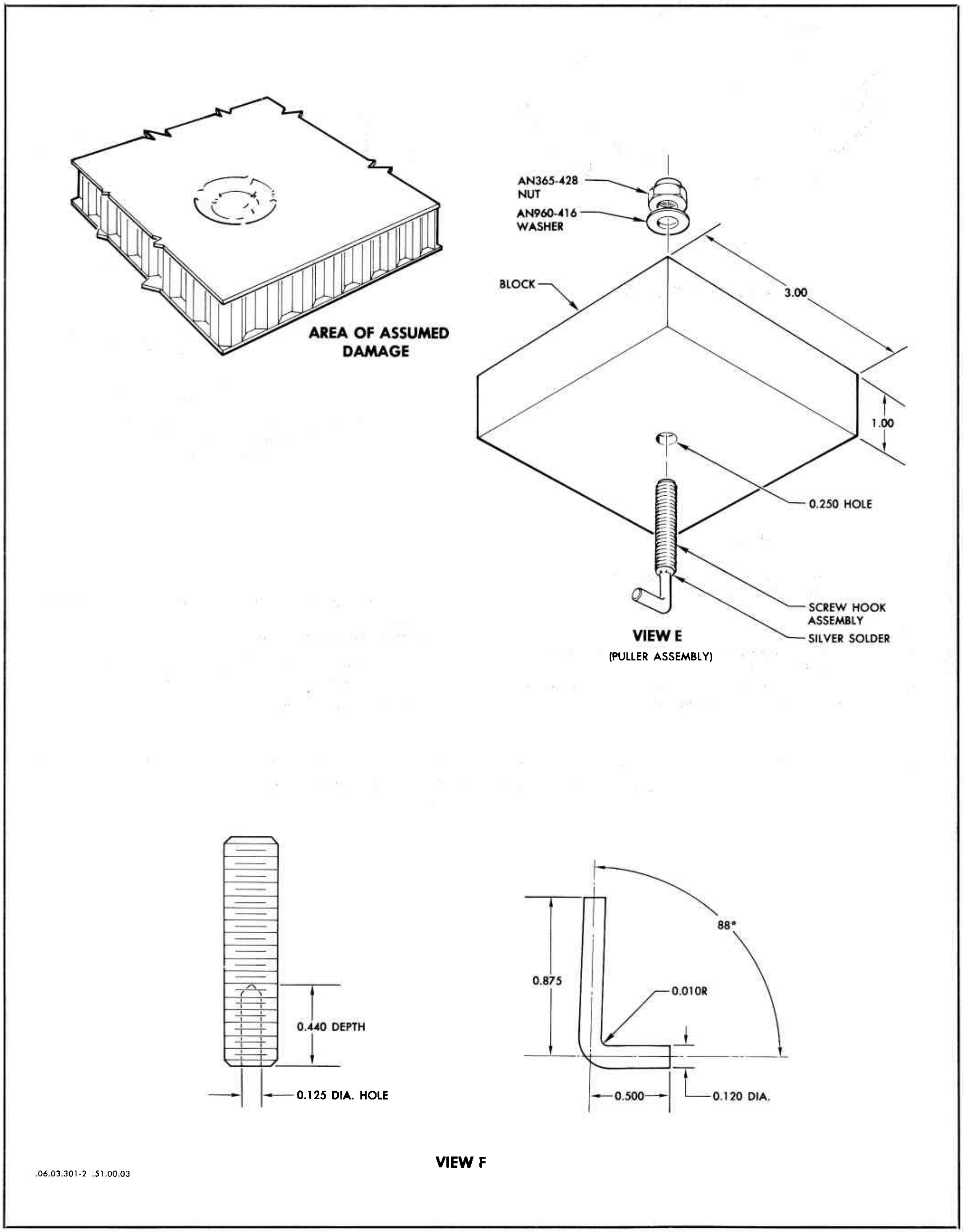
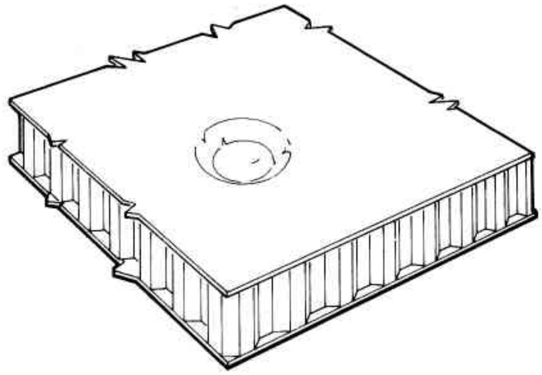
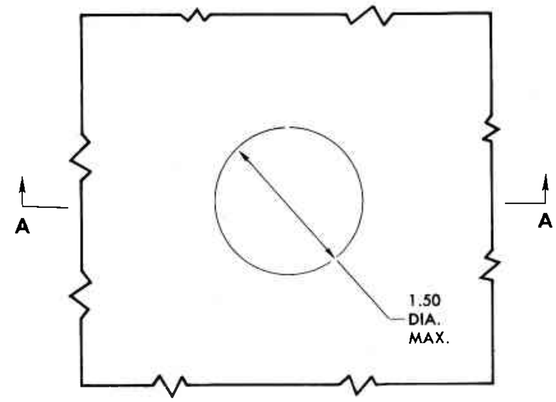
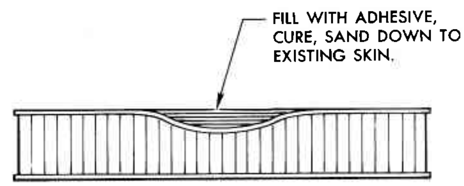


Figure 10-45. Honeycomb Repair No. 23 (Crack or Dent) (Sheet 2 of 2)



AREA OF ASSUMED DAMAGE





SECTION A-A

REPAIR NO. 24

NOTE
THIS IS A DENT REPAIR USED WHEN DAMAGE DOES NOT EXCEED 1.500 INCHES IN DIAMETER.

PROCEDURE

a. Prepare and clean damaged area. Use steps 1, 2, 5 thru 9 and 10 of figure 10-13.
b. Select adhesive and mix adhesive; mix adhesive according to manufacturers instructions.
c. Fill damage area 1/32 inch above level of existing skin.
d. Allow adhesive to cure.

NOTE
IF ADHESIVE EC1184 IS USED, ALLOW 3 HOURS CURING TIME FOR EACH 1/16 INCH OF THICKNESS.

e. Sand down to contour of existing skin.

06.03.302 A

Figure 10-46. Honeycomb Repair No. 24 (Dent)

REPAIR NO.	ADHESIVES A	NUMBER	CURING AGENT B	NUMBER	MANUFACTURER
1, 2, 3, 4, 5, and 6	FM 47 Liquid Primer	FM 47 Liquid	——	——	Bloomingdale Rubber Co. Aberdeen, Md.
1, 2, 3, 4, 5, and 6	FM 47 Film	FM 47 Film (Supported)	——	——	Bloomingdale Rubber Co. Aberdeen, Md.
3, 4, 5, 7, 8, 9, and 10	Narmco 3135	A	Narmco 3135	B	Narmco Resins and Coating Co. Costa Mesa, Calif.
2, 3, 4, 5, 6, 12, and 13	EC 1469 Compound	EC 1469	EC 1470	——	Minnesota Mining and Mfg. Co. St. Paul, Minn.
1, 2, 7, 8, 10, 14, 15, 16, 17, 19, 21, 22, 23, and 24	Core Filler	E 3045	Catalyst	E 3045	Boray Mfg. Co. 11434 S. Downey Ave. Downey, Calif.
2, 4, 5, 6, 15, and 20	338 Compound	338	Hardener 19	——	Boray Mfg. Co. 11434 S. Downey Ave. Downey, Calif.
3, 4 and 5	EC 1653 Compound	EC 1653	EC 1653A	——	Minnesota Mining and Mfg. Co. St. Paul, Minn.
3, 4, 5, 6, 7, 9, 10, 12, and 13	Narmco 3143 Compound (F.S.N. 5305-281-4648)	Narmco 3143A	Narmco 3143	Narmco 3143C	Narmco Resins and Coating Co. Costa Mesa, Calif.
6, 7, 12, 13, 23, and 24	Metalset A2	Metalset A2	01 / 08	——	Smooth-on Mfg. Co. 572 Communipaw Ave. Jersey City, New Jersey
3, 4 and 5	Epon 828 (F.S.N. 8040-291-0162)		Z	——	Shell Chemical Co. 10642 Downey Ave. Downey, Calif.
10, 23, and 24	EC 1184	EC 1184 A	EC 1184	B	Minnesota Mining and Mfg. Co. St. Paul, Minn.
10 and 11	EC 1660		——		Minnesota Mining and Mfg. Co. St. Paul, Minn.
18	EC 801 Compound Class A-½	EC 801	EC 1240		Minnesota Mining and Mfg. Co. St. Paul, Minn.
14, 15, 16, 17, 19, 20, 21, 22, 23, and 24	337 Compound	337	Hardener	37	Boray Mfg. Co. 11434 S. Downey Ave. Downey, Calif.

NOTES: 1. Cure cycles are optional unless otherwise directed by repair instructions. 2. Adhesive and accel

TABLE 10-I

Honeycomb Repair Adhesives and Processing

COMPONENT PARTS MIXING INSTRUCTIONS	APPLICATION INSTRUCTIONS	CURE CYCLE TIME	TEMP.	PRESS.	POT LIFE
Stir Thoroughly ee Paragraph 11-68	1. Brush or spray thin coat on surface to be primed. 2. Air dry 1 to 2 hours at ambient temperature (above 65°F).	45 Minutes	220°F to 235°F	None	——
o Mixing Required	1. Cut FM 47 film to size of area to be bonded. 2. Install in place between parts to be bonded.	30 Min. 1 Hour	300°F 275°F See Figure 10-12	60-100 Psi	——
Refer to Note 2 and Paragraph 11-65	1. Apply thin coat of mixture on one faying surface with brush. 2. Press part together firmly, clamp and cure.	24 Hours	Ambient (above 65°F) Refer to Note 1	Clamp	1 Hour
Refer to Note 2 and Paragraph 11-57	Fill area full plus 1/32" above surface with tongue blade. Work out any air bubbles. Cure. Sand down to contour.	Cure by raising gradually to 200°F and holding for 60 minutes.			4 Hours
Refer to Note 2 and Paragraph 11-70	Fill area full plus 132" above surface with tongue blade. Work in solid and smooth. Cure. Sand to contour.	1 Hour 20 Min.	250°F 325°F Refer to Note 1	None None	30 to 60 Min.
Refer to Note 2 and Paragraph 11-63	Fill area full plus 1/32" above surface with brush or tongue blade. Work in solid. Cure. Sand to contour.	2 Hours	Ambient (above 65°F) Refer to Note 1	None	30 Min.
Refer to Note 2 and Paragraph 11-54	Same as EC 1469.	2 Hours	150°F to 200°F	None	4 to 6 Hours
Refer to Note 2	For repairs 3, 4, 5, 9 and 10 apply same as EC 1469.	4 Hours	Ambient (above 65°F)	None	25 Min.
	For repairs 6 and 7 apply as per repair instructions.	1 Hour	200°F	None	
Refer to Note 2	Apply per repair instructions.	65 Min. 35 Min.	Ambient (above 65°F) Ambient (above 65°F)	—— ——	35 Min. 20 Min.
Refer to Note 2 and Paragraph 11-68	Same as EC 1469.	1 Hour	250°F to 300°F	——	1 Hour
Refer to Note 2 and Paragraph 11-42	Apply per repair instructions.	2 Hour	Ambient (above 65°F)	——	45 Min.
None	1. Apply thin coat to one side of repair patch. 2. Air dry approximately 20 minutes.	30 Min.	250°F to 300°F	3 to 15 Psi	——
A plus 12% of B by weight Mix thoroughly	Apply per repair instructions.	24 Hours 2 Hours	Ambient (above 65°F) 200°F	None None	2 Hours
2 parts A plus 1 part B by weight Mix thoroughly	Apply per repair instructions.	2 Hours 45 Min.	Ambient (above 65°F)	None None	30 Min.

erator are issued as a kit and should be mixed per manufacturer's instructions.

TABLE 10-II

Special Tools

FIGURE	NAME	PART NO.	USE AND APPLICATION
10-11	Damage Locating Template	8-01404-1 (Fabricate Locally)	To determine overall diameter of damage area and establish coordinate centerlines.
10-15	Router Guide Templates	8-01404-5 thru -19 and 8-01404-45 thru -51 (Fabricate Locally)	To guide router in order to route out damage area to a uniform shape and depth.
10-16	Hole Saw Guide Templates	8-01404-21 thru -43 (Fabricate Locally)	To locate and establish hole saw at correct angle while cutting out damage area.
10-17	Hole Pattern Templates	8-01404-67 thru -71 (Fabricate Locally)	To establish rivet hole pattern in damage area.
10-14	Router Assembly	8-01404-3	To rout out damage area to required shape and depth.
10-18	Hole Saws	8-01404-55 thru -65	To cut out damage area in a circular shape.
10-19	Router Turntable Assembly (used in conjunction with router assembly)	8-01404-53	May be used instead of templates to route out holes, in tapered parts, normal to the chord plane of the part.
10-20	Cutter Assembly	8-01404-87	Used to undercut core in the repair of 0.375-inch holes in honeycomb structure.
10-21	Hand Operated Undercutter Assembly	8-01404-89 thru -93	Used to undercut core in the repair of 0.75-inch to 1.50-inch holes in honeycomb structure.
10-22	Power Operated Undercutter Assembly	8-01404-95 and -97	Inserted in 1/8-inch holes to undercut core to 3/16-inch or 1/4-inch diameter prior to application of potting compounds.
10-23	Vacuum Box Assembly	8-01404-99	To apply pressure and heat to a repair while curing the adhesives.
10-24	Diaphragm Assembly	8-01404-839 and -841	Used in conjunction with clamp assemblies to provide measured pressure to repair while curing adhesives.
10-25	Hydraulic Diaphragm Assembly	8-01404-843 and -845	Same as diaphragm assembly.
10-26	Vacuum Bag Assembly	8-01404-801	To apply pressure to areas where other methods are impractical.
10-27	Pressure Systems	8-01404-803 thru -807	To hold patch in place and apply pressure while bonding.
10-28	Pressure Plates (used in conjunction with spring clamps and -813 drill bushing assembly)	8-01404-815 thru -837 (Fabricate Locally)	To locate rivet holes and doubler cutting lines, and to apply pressure to repair while bonding.
10-29	Core Saw	8-01404-861 (Fabricate Locally)	Used for rough cutting core plugs and tubes nearly flush with skin.

to 6
Hours

Hours

Section XI

REPAIR MATERIALS

11-1. REPAIR MATERIALS.

11-2. The following lists contain materials and attaching parts required to make the repairs discussed in this manual.

11-3. BOLTS.

11-4. The following is general practice information. Always use bolts of the original type and material for replacement. Use a nut made of the same material as the bolt. Use close tolerance bolts in shear installations. Use drilled shank bolts with castle nuts. Use undrilled shank bolts with self-locking nuts. Refer to Table 11-I for list of bolts.

TABLE 11-I

Bolts

DESIGNATION	MATERIAL	IDENTIFICATION	USE
AN3 to AN20	Steel, cadmium plated.	Raised X on head.	Ordinary Installation.
AN3C to AN20C	Corrosion resistant steel (CRES).	Raised dash mark (—) on head.	High-temperature installations.
AN3DD to AN20DD	Aluminum alloy, anodized.	Two raised radial dash marks (- -).	Use only as replacement for aluminum.
AN173 to AN186	Steel, cadmium plated close tolerance.	Raised triangle and X.	Shear installations.
MS 20004 thru MS 20024 series	High tensile strength steel.	Internal wrenching head.	Tension installations with high-strength nuts only.
NAS333 to NAS340	100° countersunk head. High tensile strength steel, cadmium plated. Close tolerance.	Number and triangle on head.	Shear installations with standard nuts. May be substituted for Huck lockbolts.
NAS 464	High tensile strength steel. Close tolerance.	Number and triangle on head.	High shear installations with standard nuts.
NAS 501	Nonmagnetic corrosion resistant steel (CRES).	Number and dash mark on head.	In area of equipment sensitive to magnetic materials or subject to high temperatures.
NAS 563 thru NAS 572	Full threaded steel, cadmium plated.	Full part number on head.	Use only as replacement for bolt of same number.
NAS 625	High tensile strength steel, higher than the MS 20004 thru MS 20024 series.	"NAS 625" on head.	Tension installations with high-strength nuts only.

11-5. NUTS.

11-6. The following is general practice information. Use a nut of the same material as the bolt. Use castle nuts with drilled shank bolts only. Use self-locking nuts with undrilled shank bolts only. Use shear nuts where bolts are in shear load only. Use high-strength nuts with all internal wrenching bolts. Aluminum nuts may be identified by weight. Corrosion resistant steel nuts may be identified with a magnet — they are nonmagnetic. Refer to Table 11-II for list of nuts.

TABLE 11-II

Nuts

DESIGNATION	TYPE	MATERIAL
AN310	Castle	Steel, aluminum, or corrosion resistant steel.
AN316	Plain	Steel or corrosion resistant steel.
AN320	Shear castle	Steel, aluminum, or corrosion resistant steel.
MS21049 OR MS21050	Countersunk plate nut, self-locking	Steel or corrosion resistant steel, countersunk.
MS21047 OR MS 21048	Plate nut, self-locking	Steel or corrosion resistant steel.
MS21044 - MS21045 OR MS21046	Self-locking	Steel or corrosion resistant steel.
MS20364	Shear self-locking	Steel, aluminum, or corrosion resistant steel.
MS20365	Self-locking	Steel, aluminum, or brass.
MS21047	Plate nut, self-locking	Steel or aluminum.
22NG41 or G-1000	Gang channel	Steel nuts, aluminum alloy channel.
EB-048 to -164	12-point wrench, self-locking	High-strength steel.

11-7. SCREWS.

11-8. Do not use low-strength steel screws in structural installations. The following screws are identified by an X on the head, indicating 125,000-145,000 psi tensile strength steel. Refer to Table 11-III for list of screws.

TABLE 11-III

Screws

DESIGNATION	HEAD TYPE
AN502	Fillister head, drilled.
MS24694	100° countersunk head.
MS27039	Brazier head.
NAS 514	100° countersunk head. Same as AN509 except for full threaded shank.

11-9. WASHERS.

11-10. Use regular or thin washers as needed. The letter "L" added to AN960 callout indicates a thin washer. Use corrosion resistant steel washers with corrosion resistant steel bolts and nuts only. Identify by being nonmagnetic. Use hard aluminum washers only for dissimilar metal insulation. Do not use aluminum washers with internal wrenching bolts except when called out on the print or as a replacement for an aluminum washer. Refer to Table 11-IV for list of washers.

TABLE 11-IV

Washers

DESIGNATION	MATERIAL	TYPE AND USE
AN960	Steel	Plain — ordinary installations.
AN960C	Corrosion resistant steel (CRES)	Plain — high-temperature installations.
AN 960D	Aluminum	Plain — insulation purposes.
MS 20002	Steel	Plain—under high-strength nuts.
MS 20002C	Steel	Countersunk. Use with countersunk face against head of internal wrenching bolts.

11-11. RIVETS.

11-12. Refer to paragraph 1-119 for usage, substitution and description of special fasteners. Refer to Table 11-V for list of standard rivets.

TABLE 11-V

Standard Rivets

DESIGNATION	MATERIAL	SHANK DIAMETERS IN $^1/_{32}$ INCH
MS20426B AND MS20470B	5056-F	—3 to —6
MS20426AD AND MS20470AD	2117-T3	—3 to —6
MS20426DD AND MS20470DD	2024-T31	—3 to —10

11-13. STRUCTURAL MATERIALS.

11-14. Refer to Section I for general characteristics, tempers and Table 1-I for substitutions of structural materials. Refer to Table 11-IV for list of structural materials.

TABLE 11-VI

Structural Materials

DESIGNATION	SPECIFICATION	REMARKS
	ALUMINUM ALLOY BAR STOCK	
2024-T4	QQ-A-200/3 or QQ-A-225/6	Used to machine replacements for extrusions.
6061-T6	QQ-A-200/8 or QQ-A-225/8	Used to machine replacements for extrusions.
7075-T6	QQ-A-200/11 or QQ-A-225/9	Used to machine replacements for extrusions.
7178-T6	QQ-A-200/13	Used to machine replacements for extrusions.

TABLE 11-VI

Structural Materials (Cont)

DESIGNATION	SPECIFICATION	REMARKS
ALUMINUM ALLOY SHEET STOCK		
2024-T3	Bare QQ-A-250/4 Clad QQ-A-250/5	Material has been heat treated and then given a controlled straightening operation to increase the tensile strength over 2024-T4.
2024-T4	Bare QQ-A-250/4 Clad QQ-A-250/5	Material has been heat treated without the subsequent controlled increase in tensile strength of 2024-T3.
2024-T6	Bare QQ-A-250/4 Clad QQ-A-250/5	Material has been artificially age-hardened from the 2024-T4 condition to increase its mechanical properties.
2024-T36	Clad QQ-A-250/5	Material has received controlled cold work (1%) after solution heat treatment and prior to age hardening. Exceeds normal 2024-T3 condition.
2024-T81	Bare QQ-A-250/4 Clad QQ-A-250/5	Material has been heat treated and rolled; very limited forming is possible. Should be used only for flat parts or parts having extremely simple bends.
2024-T86	Bare QQ-A-250/4 Clad QQ-A-250/5	Material has received controlled cold work (6%), after solution heat treatment and prior to age hardening. Should be used where the part is not formed or contoured.
6061-T4	Bare QQ-A-250/11	Material is used for general mechanical and lightly loaded parts. It is heat treated and has good forming characteristics. Can be aged without appreciable distortion.
6061-T6	Bare QQ-A-250/11	Material is heat treated and aged. Same characteristics as 6061-T4.
7075-T6	Bare QQ-A-250/12 Clad QQ-A-250/13	Material is a high-strength alloy suitable for all general mechanical and structural application. It is heat treated and aged.
7178-T6	Bare QQ-A-250/14 Clad QQ-A-250/15	Material is similar to 7075-T6 with increased strength properties.

TABLE 11-VI

Structural Materials (Cont)

DESIGNATION	SPECIFICATION	REMARKS
	TITANIUM	
Commercially Pure Sheet	AMS 4901	70,000 psi, annealed Min. Yield.
Commercially Pure Bar	AMS 4921	70,000 psi, annealed Min. Yield.
Alloy Sheet—8% Mn	AMS 4908	110,000 psi, annealed Min. Yield.
Alloy Sheet—5% Al and 2.5% Sn	Convair 0-01014	110,000 psi, annealed Min. Yield.
Alloy Sheet—6% Al and 4% V	Convair 0-01038	120,000 psi, annealed Min. Yield.
Alloy Bar—5% Al and 2.5% Sn	Convair 0-01015	110,000 psi, annealed Min. Yield.
Alloy Bar—6% Al and 4% V	Convair 0-01022	120,000 psi, annealed Min. Yield.
Alloy Bar—4% Al and 4% Mn	AMS 4925	130,000 psi, annealed Min. Yield.
	CORROSION RESISTANT STEEL SHEET	
Type 301, ¼ H	MIL-S-5059 Comp 301, ¼ H	Used when excellent corrosion resistance properties plus toughness and ductility are required. Not to be used where temperatures exceed 800°F.
Type 301, ½ H	MIL-S-5059 Comp 301, ½ H	Used when excellent corrosion resistance properties plus toughness and ductility are required. Not to be used where temperatures exceed 800°F.
Type 302, Annealed	MIL-S-5059 Comp 302	Used when excellent corrosion resistance properties plus toughness and ductility are required. Not to be used where temperatures exceed 800°F.
Type 321 and 347	MIL-S-6721	Used for applications requiring high heat resistance, good weldability and corrosion resistance. Used where temperatures are 800°F to 1650°F.
17-7 PH	MIL-S-25043, cond. TH1050	Used for general mechanical and structural application requiring corrosion resistance and strength up to 600°F.
Type 410	Convair 0-01047	Used for general mechanical and structural applications requiring moderate resistance to corrosion plus good physical properties. Not to be used where temperatures exceed 700°F.

TABLE 11-VI

Structural Materials (Cont)

DESIGNATION	SPECIFICATION	REMARKS
	ALLOY STEEL SHEET	
4130	AN-QQ-S-685	For general structural and mechanical parts.
	MAGNESIUM	
AZ31A	Convair, QQ-M-44, Condition H24	Material possesses a combination of good physical properties and good formability.
HK31	Convair, 0-01024 Condition H24	Primarily for components requiring weldability and good strength-to-weight ratio between 400°F and 600°F.

11-15. SODIUM HYDROXIDE.

11-16. Sodium hydroxide solution is used to determine if scratches on clad aluminum alloy sheets have penetrated through the clad layer. Mix sodium hydroxide, Specification O-S-60S, to a 10 percent solution in water.

11-17. TEMPILAQ.

11-18. Tempilaq is a temperature indicating lacquer manufactured by Tempil Corporation, 132 West 22nd Street, New York 11, N. Y. When hot dimpling is performed on aluminum and magnesium, the temperature of the area to be dimpled shall be heated to 162.7° to 176.7°C (325° to 350°F). To determine the temperature, a thin smear or daub of Tempilaq is applied adjacent to the area where the dimple is formed. When the temperature of 325°F is reached on aluminum and magnesium, the Tempilaq (orange in color) liquifies quickly, turns clear and upon cooling, solidifies, leaving a glossy or vitreous surface. Refer to paragraphs 1-174 through 1-178 for additional information on the use of Tempilaq.

11-19. SURFACE TREATMENTS.

11-20. The materials listed are for use as surface treatments on aircraft metal surfaces either alone or top-coated with enamel. Primers are intended for spray application over bare metal or chemical pretreatment coating. If used for dip application, the suitability for dipping shall be determined by the user. For further information concerning the use of surface treatments, refer to paragraph 1-181. Refer to Table 11-VII for list of surface treatment materials.

TABLE 11-VII

Surface Treatments

DESIGNATION	SPECIFICATION	APPLICATION
Paint Remover	MIL-R-25134	To remove damaged paint coating.
Chrome Pickle	MIL-M-3171 (Type 1)	Pre-paint surface treatment of magnesium alloy.
Alodine 1200	MIL-C-5541	Pre-paint surface treatment of aluminum alloy.
Wash Primer	MIL-C-8514	Primer pretreatment coating for use on clean metal surfaces of all types prior to application of paints.
Zinc Chromate Primer 32 (Yellow) 32-1 (Green)	MIL-P-8585 (Aer) or MIL-P-6808A	Primer coating for use over wash primer or on clean metal surfaces of all types and fin tip antenna.

TABLE 11-VII

Surface Treatments (Cont)

DESIGNATION	SPECIFICATION	APPLICATION
Aircraft Gray Enamel	TT-E-489 Color No. 36231	Cockpit interior protective coating over zinc chromate primer
Aliphatic Polyurethane	MIL-C-83286 Color No. 16473	Exterior protective coating over epoxy primer
Epoxy Primer	MIL-P-23377	
Aircraft Black Acrylic Lacquer Aliphatic Polyurethane	MIL-L-19538 Color No. 37038 MIL-C-83286 Lusterless	Exterior antiglare surface used in area forward of the cockpit.
Aerodynamic Smoothing Compound (Federal Stock Number 8030-684-8743	Sherwin-Williams Aluminized Putty No. DX4045.	To fill gaps less than 0.09-inch where temperature does not exceed 135°C (275°F).
Aerodynamic Smoothing Compound	MIL-S-38228 FSCM 81349 NSN 8030-00-782-1420	Refer to paragraph 11-54.
Aerodynamic Smoothing Compound (Federal Stock Number 8030-579-2527)	Stabond C-875 American Latex Product.	Refer to paragraph 11-52.
Epon Putty	EC-1751, Manufactured by Minnesota Mining and Mfg. Co., 6411 Randolph Street, Los Angeles, California.	To build up voids between two mated surfaces of different contour (IR receiver fairing installation).
Epon Putty Accelerator	EC-1752	Accelerator for EC-1751
Epon Putty	EC-1969, Manufactured by Minnesota Mining and Mfg. Co., 6411 Randolph Street, Los Angeles, California.	Alternate for EC-1751
Lacquer Primer	MIL-P-7962	Primer coating for use over wash primer, under acrylic lacquer
Lacquer, acrylic Nitrocellulose,	MIL-L-19537	Exterior protective coating over Lacquer Primer – Standard A/C Finish.

11-21. ADHESIVES.

11-22. Adhesives are used for bonding metal to metal or to various other materials such as phenolic or polyester sheet. Epoxy adhesives, listed under Specification MIL-A-8623, are permanent adhesives. Refer to Table 11-VIII for list of adhesives.

TABLE 11-VIII

Adhesives

MATERIAL	FEDERAL SPECIFICATION	FEDERAL STOCK NUMBER	REMARKS
Epon 3119	MIL-A-8623 (Aer) Type II	8030-687-3540	Refer to paragraph 11-59.
Epon VIII	MIL-A-8623 (Aer) Type II (Green)	8040-270-8136 (QT) 8040-691-1322 (PT)	Refer to paragraph 11-61.
EC-1469	MIL-A-8623 (Aer) Type II (White)	Manufactured by: Minnesota Mining and Mfg. Co., 6411 Randolph St., Los Angeles, Calif.	Refer to paragraph 11-57.
	MIL-A-25457	8040-721-901	For bonding silicon rubber to itself or to aluminum without use of heat and pressure.

11-23. Sealing Materials.

11-24. The manufacture of the F-106 airplane requires the use of many types of rubber and synthetic rubber. The type of rubber used depends upon the job it must do. For example, the electronic doors are sealed with silicone rubber. This particular material is used because of its excellent resistance to heat and cold. If an unauthorized substitute is made for this seal, failure of the electronics system could result. Most compounded natural rubber, synthetic, and especially silicone rubber, are difficult to cement. This is one of the reasons rubber parts are usually attached with rivets or screws. However, since this method of attachment is not always practical, various types of sealers and cements are provided for attaching the different types of rubber. These cements and sealers, like rubber, are provided for a specific use. Use of the wrong

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type of cement or sealer, or improper substitution of rubber seals, can cause the loss of a system or possible damage. Tables 11-IX and 11-X contain the authorized repair materials and sealing agents required to make the repairs discussed in this manual.

11-25. CABIN PRESSURIZATION SEALING MATERIALS.

11-26. For materials used in cabin pressurization sealing and their purpose and handling information, refer to Table 11-IX. Refer to paragraph 1-38 and 1-181 for sealant application procedures.

TABLE 11-IX

Cabin Pressurization Sealing Materials

MATERIAL	FEDERAL SPECIFICATION	FEDERAL STOCK NO.	REMARKS
Sealing Compound	MIL-S-8802 A 1/2 A 2 B 1/2 B 2	8030-753-5008 8030-753-5003 8030-753-5004 8030-735-5005	Faying surface sealant. For sealing integral fuel tanks, fuel cell cavities and other areas subjected to temperature range of 65° F to 275° F.
Sealing and Coating Compound Corrosion Inhibitive Faying Surface Sealant	MIL-S-81733 TYPE II - 2 I - 2 II - 1/2 IV - 12	8030-491-2933 8030-433-9028 8030-470-9154 8030-419-2954	In all corrosion prone areas, all faying surfaces, seams and lap joints shall be protected with sealant, Specification MIL-S-81733. Except integral fuel tank areas.
Deleted	MIL-S-8802		Suitable substitute for MIL-S-81733.
Canopy Sealer	90-006-2	8030-145-0372	Refer to Paragraphs 11-44 thru 11-45B
Canopy Sealer	RTV-106	8040-941-9984	
Canopy Sealer			
Surface Cleaner	Aliphatic Naphtha TT-N-95	6810-238-8118	Use to clean painted and unpainted surfaces.
Sealer Primer	A-4004 DC 1200	8040-653-3104 8030-870-0877	Refer to Paragraph 11-48
Parting Agent	Petrolatum VV-P-236	9150-250-0926	
Synthetic Rubber	M5570. Fairprene Sheet. 0.015" Thick. E. I. Dupont Co., Wilmington, Del.	9320-624-1700 0.031 36 X 36	Use for cabin pressure access door gasket.
Rubber Plug	Convair — 370994		To hold sealants in the voids.
Windshield Sealer	EC 1667-MMM Co. DC20-103 Dow Corning Corp. RTV-106 GE Co.	8030-850-2495 8030-951-0439 8040-941-9984	Refer to Paragraph 11-46

11-27. FUEL TANK SEALING MATERIALS.

11-28. Refer to Table 11-X for materials used in fuel tank sealing, their purpose and handling information. Repairs made in the fuel tank area, which create a new exposed metal surface, must first receive the Alodine 1200 treatment before sealants or finishes are applied. Refer to

Table 11-VII for Alodine 1200 information. Refer to paragraph 2-45 for sealant application procedure.

11-29. The materials listed in Table 11-X have been approved for fuel-tight sealing for the F-106A and F-106B airplanes.

CAUTION

Fuel tank sealants listed in Specification MIL-S-7502 are not to be used under any conditions in the fuel tank areas of the F-106A and F-106B airplanes.

TABLE 11-X

Fuel Tank Sealing Materials

MATERIAL	FEDERAL SPECIFICATION	FEDERAL STOCK NO.	REMARKS
Sealing Compound	MIL-S-8802 MIL-S-83430 (SUB) PRO SEAL 899		Refer to T.O. 1-1-3 for Mixing and Application Procedures.
Coating Corrosion Protection	MIL-S-4383 EC776		Refer to T.O. 1-1-3 for Mixing and Application Procedures.
Primer	EC 1662 (Red)	8010-973-1565	See Paragraph 11-38.
Sealer	EC 1663B (Red)		See Paragraph 11-40.
Sealer Accelerator	EC 1663A (Black)		
Cement	C319 Manufactured by Connecticut Hard Rubber Co., New Haven, Conn.		Second choice for bonding silicone rubber.
Solvent-Thinner	Methyl Isobutyl Ketone TT-M-268		Solvent or thinner for EC 776.
Dome Nut	Nutt-Shell 14630H048 Convair NU 453	4806-1650-605-8590	Replacement for access door nuts.
Solvent	Methyl Ethyl Ketone TT-M-261	6810-290-0055	Solvent for cleaning surfaces.
Solvent	Toluene TT-T-548		Solvent for cleaning surfaces.
Sealing Plug	1100 Aluminum Bar (Annealed) QQ-A-411		To plug ends of voids.

11-30. Deleted.

11-31. Deleted

11-32. Deleted.

11-33. Deleted.

11-34. Deleted.

11-35. Deleted

11-36. Structural Adhesive Prime — EC 1290.

11-37. EC 1290 is a high-temperature spray prime provided for use with "Scotch-Weld" and epoxy adhesives. It is used on fuel tank inner surfaces where the original prime has been scratched. Clean surfaces with methyl ethyl ketone (6810-290-0055). Apply prime with a siphon cup gun at 30 psi to 50 psi pressure. Cure at not less than 93.3°C (200°F) for 45 minutes.

WARNING

This prime contains a volatile and flammable solvent and must be kept away from sparks or flame. It also contains a toxic ingredient which can cause damage to the eyes, lungs, kidneys, and liver. Personnel who are to be exposed to this material for prolonged periods should wear chemical safety goggles, respirator, and protective clothing.

11-38. Primer — EC 1662.

11-39. Primer EC 1662 is a sprayable or brush type compound used to prepare metal surfaces for bonding with silicone rubber. It is used as a primer for sealer EC 1663. Before applying primer, make certain that surface to which primer is to be applied is dry and free of oil and grease. Clean surfaces with a lint-free cloth soaked with methyl ethyl ketone (6810-290-0055).

WARNING

EC 1662 is flammable and toxic material. Keep away from heat and open flame. Using personnel must wear a respirator and chemical safety goggles. Wash hands before eating or smoking.

After spray or brush application, allow primer to dry for at least 30 minutes.

11-40. Sealer — EC 1663 B/A.

11-41. Sealer EC 1663 B/A is a two-part heat-resistant sealer applied with a brush. It has a silicone base and is the first choice sealer for bonding silicone seals to metal. Sealing surfaces must be primed with EC 1662 and allowed to dry for at least 30 minutes. Sealer should be mixed in its own container, if supplied in kit form, until it becomes one color. If not supplied in kit form, it must be mixed with 10 parts of accelerator EC 1663A (black) to 100 parts of sealer EC 1663B (red) by weight. Apply to seal with brush and allow to dry for one hour before installing seal. After seal is installed against metal surface, allow an air cure period of 24 hours.

WARNING

Accelerator EC 1663A contains ingredients that can cause skin irritation. If material comes in contact with skin, wash affected area with soap and warm water. Contaminated clothing should be laundered before reuse. If eyes are contacted by material, irrigate eyes with water and get medical attention immediately.

11-42. Sealer — EC 1184.

11-43. Sealer EC 1184 is an aluminum colored compound used to aid in aerodynamic smoothing of airplane surfaces. Sealer EC 1184 can be hand-sanded and smoothly feathered into the surrounding surface, and is resistant to water, oil, gasoline, cold, heat, shock, and vibration. This material has a pot life of 45 minutes and is to be mixed per manufacturer's instructions. The cure time is two hours at ambient air temperature (above 65°F).

WARNING

This material is a flammable mixture; keep away from heat and flame. Avoid prolonged breathing of vapor; avoid prolonged or repeated contact with skin.

11-44. Canopy Sealer- Dow Corning 90-006-2.

11-45. 90-006-2. Silicone Sealant is a two-part sealer. Each of the parts comes from the manufacturer in a kit form. The sealer is used for pressure sealing around the windshield and canopy and is mixed by combining 10 parts of catalyst to 100 parts of base. The mixture is stirred with a spatula until it becomes a uniform blend. This sealer should be mixed just prior to use and only in a sufficient quantity to do the job, since its work life is only two hours. The method of application will depend upon its use. Ordinarily, it is applied over a very thin coating of petrolatum to prevent adhesion to a faying surface, or over DC 1200 primer to increase adhesion to a faying surface. RTV-106 used with primer A4004 is a suitable substitute.

11-45A. Canopy Sealer —RTV-106. *Applicable to F-106A airplanes after incorporation of TCTO 1F-106A-556.*

11-45B. RTV-106 sealer (01139) is a one part, high temperature silicone rubber sealer, red in color, and is dispensed in tube form using an attached applicator. This sealant, which has a shelf life of 12 months when stored at temperatures below 80°F (26.67°C), is used when installing an F-106A clear top canopy panel as instructed in T.O. 1F-106A-2-2-2-2. The sealer is applied to the canopy panel faying surface over a coating of A4004 primer which provides increased adhesion. A light film of petrolatum, Federal Specification VV-P-236, is applied to the canopy frame faying surface to prevent adhesion. Recommended sealant curing time before conducting a cockpit pressure leak test is a minimum of four hours at 70°F (21.1°C) temperature.

11-45C. F-106A Clear Top Canopy Bonding Cement — PS-18. *Applicable after incorporation of TCTO 1F-106A-556.*

11-45D. Cement PS-18 (77902) is a three part room temperature cement used for bonding attachments to the F-106A clear top canopy panel. The three parts of the cement consist of component A, component B, and component C.

WARNING

Do not mix components B and C directly together. Always mix components A and B together first, then add component C as directed to prevent possible danger to personnel.

Component A, when stored in the original container or a glass container, has a shelf life of six months at 45°F(7.22° C) temperature, or a shelf life of four months at 73°F (22.78°C). Component C becomes discolored when exposed to sunlight. Discoloration does not reduce the effectiveness of component C, but it should be stored in original containers or dark glass bottles to prevent discoloration. Do not mix more cement than can be used in 30 minutes, which is the average useful life under normal room temperature conditions. Since the cement will harden during polymerization, it should be mixed in a disposable container. Paper cups may be used but care should be taken to assure that the cups do not have a surface coating that might be disolved by the cement. Mix four ounces of component A and one capsule of component B together. Stir slowly, avoiding entrapment of air bubbles in the mixture. After thoroughly mixed, the mixture of components A and B has a storage life of one week at a temperature of 40°F (4.44° C) or lower. When cement is to be used, warm mixture of components A and B to room temperature, assuring that no moisture condensation enters the mixture. Add five cubic centimeters of component C to the components A and B mixture. Do not mix more cement than can be used in 30 minutes.

WARNING

Do not dispose of the cement mixing container until all remaining cement has set and the container is cool to prevent possible harmful effects to personnel.

The three part mixed cement has an average useful life of 30 minutes at 70°F (21.11°C). The use life may be extended to 90 minutes providing the mixture is kept at a temperature of 50°F (10.00°C) or lower.

11-46. Windshield Sealer — EC 1667.

11-47. Windshield sealer EC 1667 is a two-part, high-temperature, silicone-base sealer used to pressure seal the windshield faying surfaces. This sealer is supplied in kit form and should be mixed according to manufacturer's instructions. To insure proper adhesion, all surfaces must be sprayed or painted with primer EC 1662 before the sealer is applied. Cure time is approximately 24 hours at ambient temperatures. Curing time for EC 1667 sealer

may be reduced to approximately four hours when installing windshield glass panels if a locally manufactured sheet metal cover is used with the SE 0973 heater. Refer to Section V of T.O. 1F-106A-2-2-2-2 for windshield cover manufacturing details and for windshield glass panel installation procedure.

11-48. Primer — DC1200—A4004

11-49. Primer DC1200 is a surface treatment which improves the bond of 90-006-2 sealer to faying surfaces. Before applying primer, clean surfaces with a clean cloth dampened with aliphatic naphtha. Spray or brush primer directly from its container and allow it to dry for five minutes. A4004 primer is used with RTV-106 sealant.

CAUTION

Do not apply primer to 90-006-2 sealant but only to the metal surface to be bonded.

11-50. AERODYNAMIC SMOOTHING COMPOUNDS.

11-51. Aerodynamic smoothing compounds are used extensively in production and repair of F-106 airplanes to smooth-down protruding laps and seams which otherwise would cause aerodynamic drag. Several types of these compounds are used in repair procedures outlined in this manual. Because of the different size gaps they must fill and the wide heat range in which they must operate, substitutes must not be used unless approved by competent authority.

11-52. Aerodynamic Smoothing Compound — Stabond 875.

11-53. Stabond 875 is a two-part heat-resistant aerodynamic compound used to fill gaps of up to ¼ inch in laps and seams. It has the consistency of putty when mixed with Stabond "C" thinner. Since this compound requires varying amounts of thinner for different thicknesses and cure times, mixing and curing should be accomplished in accordance with the manufacturer's instructions on the container.

11-54. Aerodynamic Smoothing Compound — MIL-S-38228.

11-55. MIL-S-38228 is a two-part synthetic rubber compound in two types for filling and sealing exterior skin seams to smooth contours. Type I is for use in the temperature range of -65°F to 250°F and Type II is for use in the temperature range of -65°F to 500°F. Type I is more resistant to jet fuel than Type II. Both are available in 1/2, 2 and 4 hours application times, with cure times of 24, 48 and 72 hours respectively at room temperature. This sealant may be applied with extrusion gun or spatula. See figure 1-27 for method of application. The cured compound color is aluminum.

NOTE

Refer to T.O. 1F-106A-23 for surface preparation instructions.

11-56. ADHESIVES.

11-57. Adhesive EC 1469.

11-58. Adhesive EC 1469 is a one- or two-part epoxy resin compound with the consistency of flowable paste. It is used to re-bond separations between honeycomb and skin in the elevon and rudder trailing edges. When used as a one-part adhesive, it requires temperatures of 176.7°C (350°F) and pressure of 25 to 50 psi to effect a good cure. When used as a two-part adhesive, it will cure at room temperature in about 60 hours. For use as a two-part adhesive, blend 4 percent EC 1470 accelerator to the desired amount of EC 1469 adhesive. Mix only the amount of material that will be used within four hours.

WARNING

Both EC 1469 and EC 1470 contain caustic and toxic components. Wear a face shield or goggles while handling or mixing and provide for adequate ventilation. Avoid contact with skin and eyes. In case of contact with eyes, flush with plenty of water for 15 minutes and get prompt medical treatment. Clean areas of skin contact with soap and water. Remove contaminated clothing at once.

Inject the mixed adhesive with a sealant gun or a large hypodermic needle.

NOTE

To facilitate ease of flow, EC 1469 may be warmed before mixing.

11-59. Epon 3119 Adhesive.

11-60. Epon 3119 is a two-part epoxy adhesive used as an alternate for Epon VIII when re-bonding separations between the honeycomb and skin in the elevon and rudder trailing edges. It is supplied in kit form; the accelerator is shipped with the adhesive but in a different container. Mixing ratios, cure times, and temperatures are shown on the containers. When small batches are mixed, as is usually the case when repairing separations, follow the recommended mixing instructions.

WARNING

Both Epon 3119 and its accelerator contain toxic and caustic compounds. Provide for adequate ventilation and protect hands and eyes while mixing.

This adhesive should be applied with either a sealant gun with a small tip or a large hypodermic needle.

11-61. Epon VIII Adhesive.

11-62. Epon VIII is a two-part epoxy adhesive used in re-bonding separations of honeycomb from skin in the elevon and rudder trailing edges. It is supplied in kit form; the accelerator is shipped with the adhesive but in a separate container. Mixing ratios, cure times, and temperatures are indicated on the container. For full bonding strength, Epon VIII requires heat cures. Mix only the amount that will be used immediately, since the pot life of this adhesive after mixing with its accelerator is one to four hours.

WARNING

Both Epon VIII and its accelerator contain toxic and caustic components. Provide for adequate ventilation; wear rubber gloves and goggles while mixing.

Because of the varying amounts of material needed to complete repairs, the manufacturer's instructions on the container should be followed. Inject adhesive in area of separation with a sealant gun with a small tip or a large hypodermic type needle.

11-63. Adhesive — 338.

11-64. Adhesive 338 is a two-part compound used as an airplane honeycomb edge sealer, or for embedding and attaching inserts and fittings in the honeycomb. This adhesive is mixed by adding hardener 19 (7 percent of adhesive by weight) and stirring well. Pot life is 30 minutes. Cure time is two to three hours at ambient air temperature (above 65°F), or 45 minutes at 200°F (93°C).

11-65. Adhesive — Narmco 3135.

11-66. Narmco 3135 is a chemical and solvent resistant two-part adhesive used to fill cracks or dents. This adhesive is mixed by adding equal parts of 3135A to 3135B, and stirring in glass fiber mill ends to the solution to obtain paste consistency. Cure time is 24 hours at ambient temperature (above 65°F), or one hour at 180°F to 200°F (82°C to 93°C).

11-67. Adhesive — Epon 828.

11-68. Epon 828 is a chemical resistant epoxy-type adhesive having a high heat distortion point. This adhesive is used to fill surface dents on the airplane. Curing agent Z is used in a 1-to-5 ratio. Cure time is one hour at 250°F to 300°F (121°C to 149°C). Pot life is one hour.

WARNING

Skin contact must be prevented. Any accidental contamination of skin areas should be thoroughly cleansed immediately.

11-69. FILLERS.

11-70. Core Filler XE 3045 is a solvent, water, and chemical resistant thermosetting polymer that is used to replace damaged honeycomb areas. This compound is mixed by adding the catalyst (7.5 percent by weight of the core filler) and stirring well. Pot life is 30 to 60 minutes when mixed with catalyst. Air cure for 24 to 48 hours or heat cure for 20 minutes at 325°F (162.8°C).

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11-71. BONDING FILM.

11-72. Bonding film FM-47 is used in conjunction with FM-47 liquid primer, to adhere patches to areas being repaired. Primer FM-47 is to be applied to a thoroughly clean surface only. Air dry for one to two hours, then apply heat at 220°F to 235°F (105°C to 113°C) for 45 minutes. Film FM-47 is cut to size and placed on the repair surface. Cure time is 30 minutes at 300°F (149°C) at 60-100 psi, or one hour at 275°F (135°C) at 60-100 psi.

11-73. AERIAL REFUELING SEALING MATERIALS.

11-74. Refer to Table 11-XI for aerial refueling sealing materials. When repairs are to be made in the aerial refueling area, refer to Section I for surface treatment of repair parts.

TABLE 11-XI

Aerial Refueling Sealing Materials

MATERIAL	FEDERAL SPECIFICATION	FEDERAL STOCK NO.	REMARKS
Sealing Compound	MIL-S-8802, Class B2		Refer to paragraph 4-17A, and to T.O 1-1-3 for mixing and application procedures. Refer to paragraph 4-52C and paragraph 4-52E.

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