EO 05-205A-1

CANADIAN FORCES

AIRCRAFT OPERATING INSTRUCTIONS

CF-5A & CF-5A/R

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NOTES TO USERS

1 This publication is divided into four Parts: Description, Handling, Emergency Handling, and Operating Data.

2 PART 1 - DESCRIPTION of the controls and equipment with which the pilot should be acquainted.

3 PART 2 - HANDLING describes the normal handling of the aircraft by the aircrew.

4 PART 3 - EMERGENCY HANDLING describes the emergency handling of the aircraft by the aircrew.

5 PART 4 - OPERATING DATA gives the flying and engine limitations and includes information on fuel consumption, range, and endurance under various conditions of flight.

6 These notes are complementary to EO 05-1-1 Aircraft Operating Instructions General and assume a thorough knowledge of its contents.

7 In the text, words written in capital letters indicate actual markings on the controls concerned.

8 The use of code Ⓐ in the text and illustrations signifies reference to the CF-5A; likewise, code A/R signifies reference to the CF-5A/R. Text and illustrations not coded are applicable to either aircraft.

9 Comments and suggestions should be forwarded through the usual channels to CFHQ.

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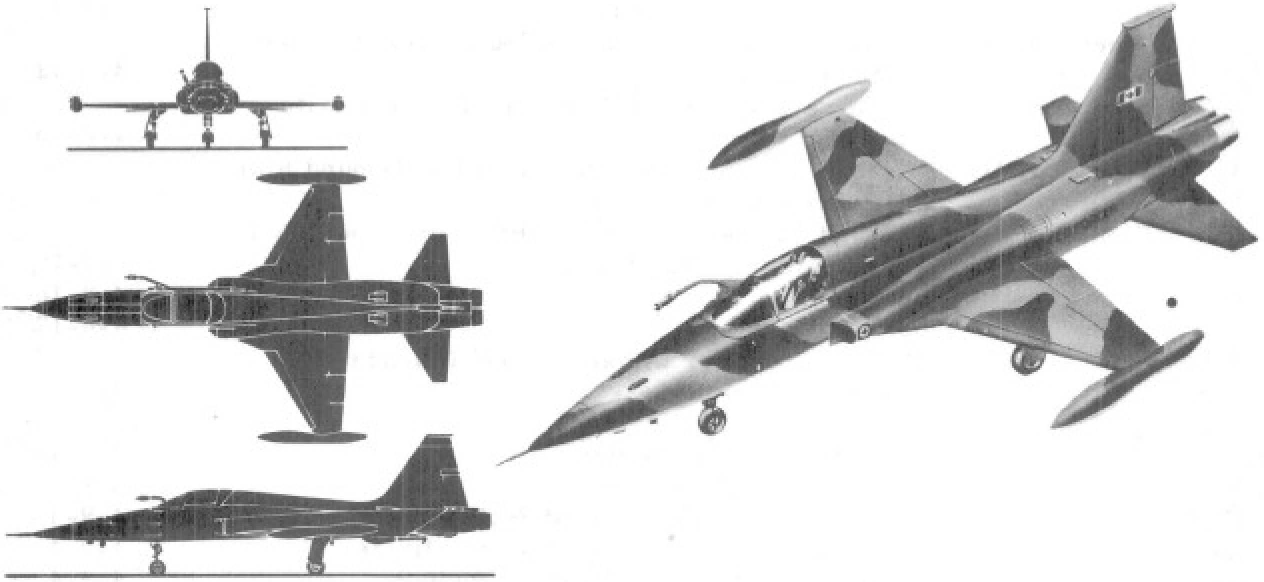
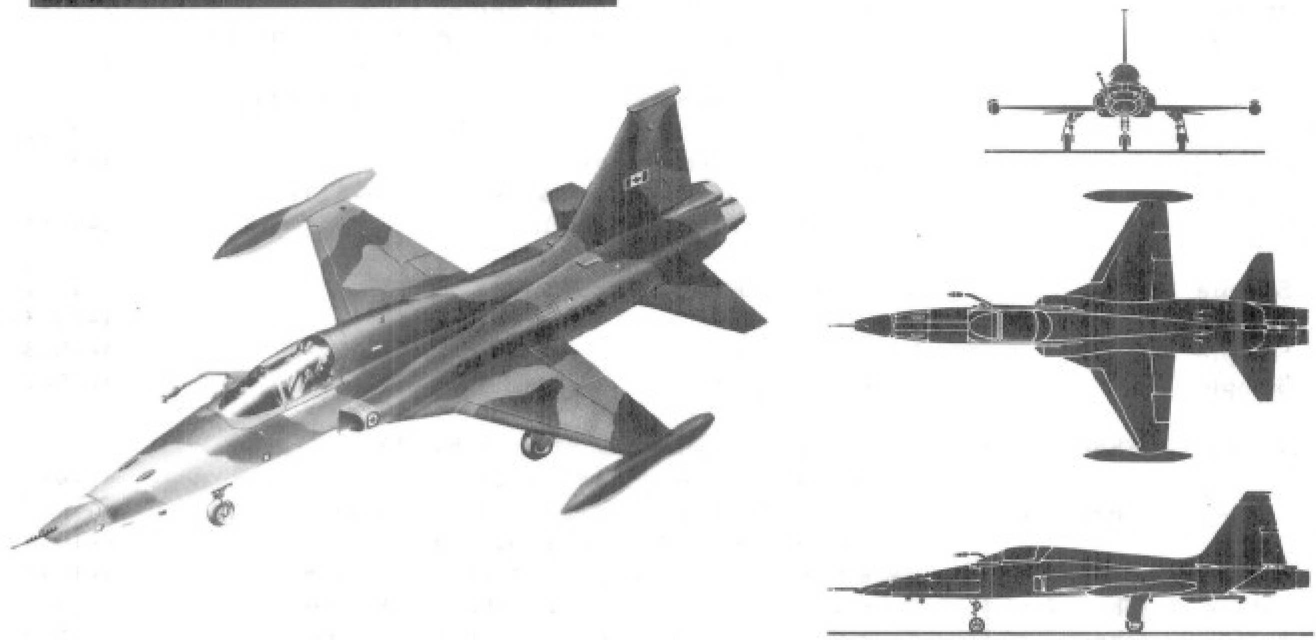
CF-5A TACTICAL FIGHTER



CF-5A/R RECONNAISSANCE TACTICAL FIGHTER



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PART 1

DESCRIPTION

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PART 1

DESCRIPTION

SECTION 1

GENERAL DESCRIPTION

GENERAL

1 The CF-5A (A) aircraft is a single-place tactical fighter. The aircraft becomes a multipurpose tactical fighter and photoreconnaissance fighter with the installation of a removable three-camera reconnaissance nose package and the necessary cockpit control panel. The (A) when configured for this multipurpose capability is identified as a CF-5A/R (A/R). The aircraft is designed to operate from unimproved airfields. In addition to twin-engine reliability, the aircraft are capable of supersonic flight. The fuselage is an area rule ("coke bottle") shape. The wing, horizontal stabilizer, and vertical stabilizer leading edges are moderately sweptback. Each wing incorporates both leading and trailing edge flaps for increased lift and improved low-speed handling characteristics. The speed brake is located at the lower midcentre of the fuselage for increased drag, and a drag chute is provided to decrease landing distances. The lower aft centreline fuselage section is provided with an arresting hook for barrier engagements. Thrust is supplied by two axial flow, turbojet engines equipped with afterburners. Each engine is provided with an inlet duct and a louvered auxiliary air inlet engine door on each side of the fuselage. The tricycle landing gear has a steerable nosewheel. A two-position nose gear strut, controlled from the cockpit, increases takeoff angle-of-attack position. Flight controls are hydraulically actuated by two independent hydraulic systems with artificial feel devices incorporated to simulate feel to the pilot. The cockpit is enclosed by a manually operated clamshell canopy. Emergency escape is accomplished by an upward trajectory ejection system capable of ground level ejection. External stores are carried on five jettisonable pylon stations. The aircraft is equipped with two 20mm guns in the nose, with or without the photoreconnaissance nose installed. The A/R aircraft provides a low-through-high altitude photoreconnaissance capability while still retaining the tactical fighter capability of the (A). A removable air refueling probe, when installed, permits in-flight refueling of the aircraft.

DIMENSIONS

2 The overall dimensions of the basic aircraft, with wingtip tanks, at normal tire and shock strut inflation pressures are:

(a) Length, maximum (with boom) . . . . 47.2 ft

(b) Wing Span . . . . 25.8 ft

(c) Height (to top of vertical stabilizer) . . . . 13.2 ft

(d) Track (main wheels) . . . . 11.0 ft

(e) Wheelbase . . . . 16.2 ft

AIRCRAFT WEIGHT

3 The average basic and gross weights of the aircraft are as follows:

		(A)	A/R
(a)	Basic Weight (empty wingtip and internal fuel tanks plus residual fuel)	9701 lb	9842 lb

(b) Maximum Permissible Takeoff Gross Weight . . . . No Limit

(c) Configuration Gross Weights (with full wingtip tanks):

CONFIGURATION	(A)	A/R
Clean (no pylons)	14,749 lb	14,890 lb
Pylons (all pylons w/o stores)	15,380 lb	15,521 lb
SUU-20/A Bomb-Rocket Dispenser, loaded (CL)	15,350 lb	15,491 lb
LAU-10/A Rocket Launcher, loaded (OUTBD)	16,063 lb	16,204 lb
CBU-2A/A Bomb Dispenser, loaded (OUTBD)	16,657 lb	16,798 lb
LAU-3/A Rocket Launcher, loaded (OUTBD & INBD)	16,943 lb	17,084 lb
MK-84 LDGP Bomb (CL)	16,864 lb	17,005 lb
MK-82 LDGP Bomb (OUTBD, INBD, & CL)	18,035 lb	18,176 lb
MK-83 LDGP Bomb (INBD & CL)	18,087 lb	18,228 lb
Tanks, full (INBD & CL)	18,506 lb	18,647 lb
BLU-1/B Fire Bomb (OUTBD, INBD, & CL)	18,865 lb	19,006 lb
M-117 GP Bomb (OUTBD, INBD, & CL)	19,480 lb	19,621 lb
Maximum Armament: MK-84 LDGP Bomb (CL) and M-117 GP Bomb (OUTBD & INBD)	20,630 lb	20,771 lb

NOTE

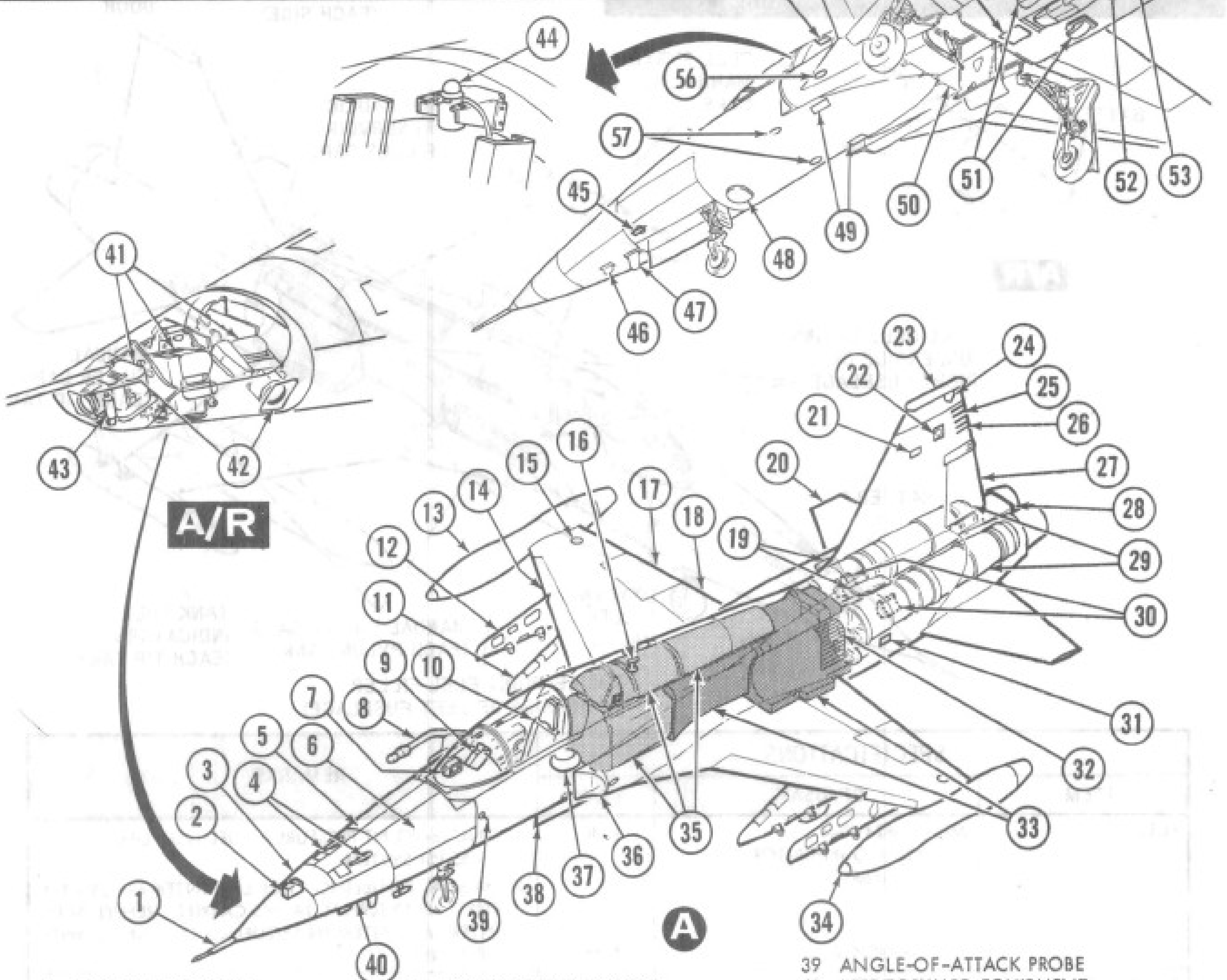
- No limit on maximum takeoff gross weight when aircraft is loaded with authorized configurations.
- Refer to Figure 4-1-4 for Authorized Configurations for Takeoff.
- Gross weights include aircrew, survival kit and parachute, full ammunition, full fuel, oil, and oxygen.
- Refer to Part 4, Section 1, and Part 4, Section 2; for authorized external store combinations and store weights for mission planning.

(d) See Figure 1-3-2 for fuel cell capacity and weights.

OIL TANK CAPACITIES

4 Oil Capacity - Each engine is provided with an individual oil tank. Each tank when fully serviced holds 0.83 Imperial (1.0 US) gallon with an expansion space of approximately 0.8 Imperial (1 US) quart.

GENERAL ARRANGEMENT



1 PITOT-STATIC BOOM
2 BATTERY
3 RIGHT FORWARD EQUIPMENT COMPARTMENT
4 GUNS
5 RIGHT GUN BAY
6 LEFT GUN BAY
7 GUN CAMERA
8 AIR REFUELING PROBE
9 COMPUTING SIGHT
10 EJECTION SEAT
11 INBOARD JETTISONABLE PYLON
12 OUTBOARD JETTISONABLE PYLON
13 WINGTIP TANK
14 LEADING EDGE FLAP
15 POSITION LIGHT (2 EACH SIDE)
16 T_2 HEATER SENSOR PROBE
17 AILERON
18 TRAILING EDGE FLAP
19 COOLING DUCTS
20 HORIZONTAL STABILIZER
21 ROTARY BEACON LIGHT
22 POSITION LIGHT (EACH SIDE)
23 VERTICAL STABILIZER
24 UHF ANTENNA
25 TACAN ANTENNA
26 IFF ANTENNA
27 RUDDER
28 DRAG CHUTE COMPARTMENT
29 ENGINES
30 ENGINE OIL RESERVOIRS
31 EXTERNAL ELECTRICAL RECEPTACLE
32 ENGINE AUXILIARY TAKE-OFF DOOR (EACH SIDE)
33 RIGHT FUEL SYSTEM CELLS
34 TIP TANK POSITION LIGHT (EACH SIDE)
35 LEFT FUEL SYSTEM CELLS
36 ENGINE AIR INLET DUCT (EACH SIDE)
37 LIQUID OXYGEN CONVERTER
38 CENTRELINE JETTISONABLE PYLON
39 ANGLE-OF-ATTACK PROBE
40 LEFT FORWARD EQUIPMENT COMPARTMENT
41 70MM RECON CAMERAS
42 CAMERA WINDOWS (7 TOTAL)
43 REFLECTED LIGHT LEVEL MONITOR
44 INCIDENT LIGHT LEVEL MONITOR
45 TOTAL TEMPERATURE PROBE
46 TACAN ANTENNA
47 UHF/IFF ANTENNA
48 UHF/ADF ANTENNA
49 LANDING-TAXI LIGHT (EACH SIDE)
50 SPEED BRAKE
51 CONSTANT SPEED DRIVE COOLING AIR SCOOPS
52 ARRESTING HOOK
53 ENGINE STARTER AIR INLET
54 LF/ADF ANTENNA
55 FORMATION LIGHT (EACH SIDE)
56 INTERPHONE RECEPTACLE
57 POSITION LIGHTS

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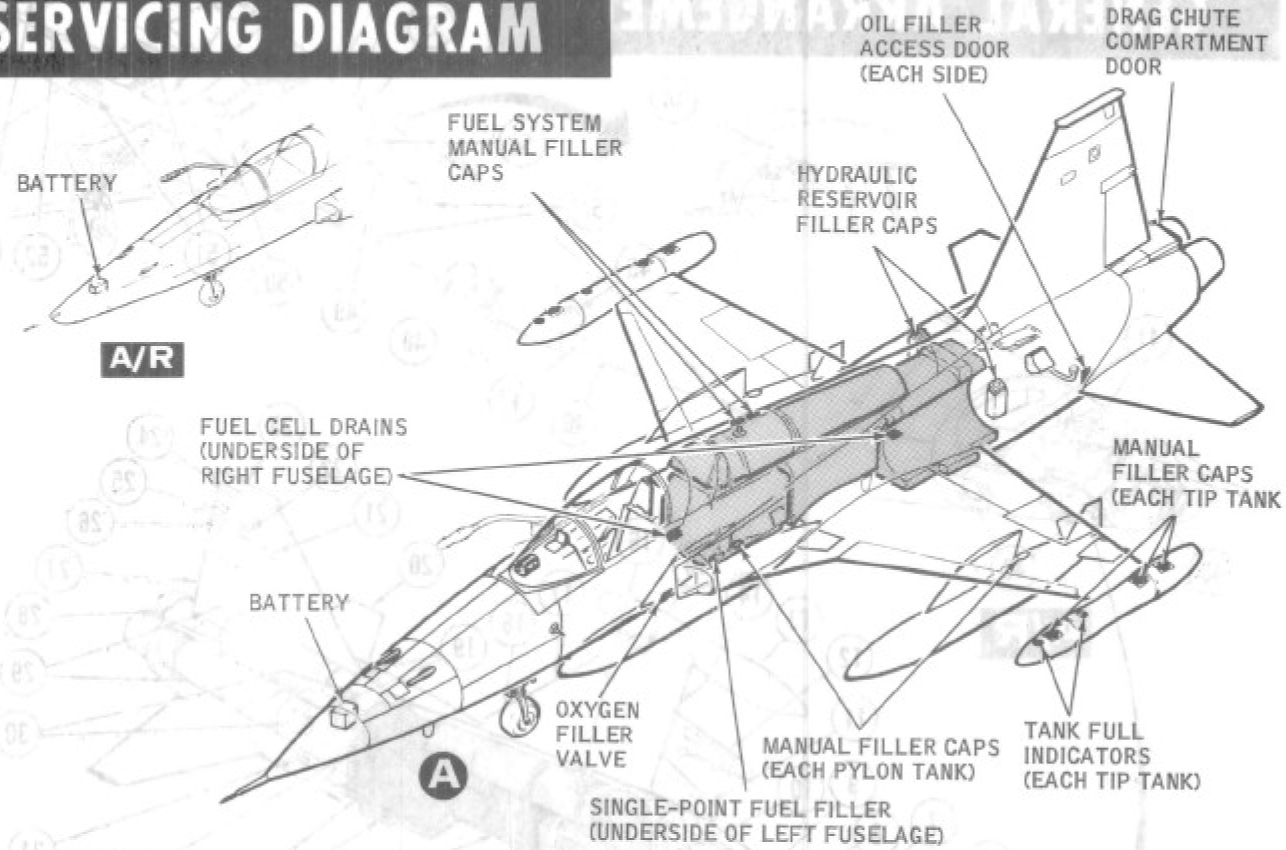
Figure 1-1-1

SERVICING DIAGRAM



SPECIFICATIONS			REMARKS
ITEM	CF/USAF	NATO	
FUEL	MIL-T-5624 JP-4 (WITH ICING INHIBITOR MIL-I-27686A) EMERGENCY JP-5	F-40 F-44	1 SINGLE-POINT PRESSURE REFUELING: USE 45-55 PSI SYSTEM. 2 MANUAL REFUELING: FILL LEFT INTERNAL SYSTEM FIRST. IF EXTERNAL TANKS CARRIED, REFUEL AFTER INTERNAL SYSTEM IN SEQUENCE CL, WING, AND TIP TANKS.
ENGINE OIL	MIL-L-7808 ALTERNATE — NONE	0-148	CHECK OIL LEVEL WITHIN 30 MINUTES AFTER ENGINE SHUTDOWN.
HYDRAULIC FLUID	MIL-H-5606	H-515	1 PRESS FILLER CAP DOWN TO VENT RESERVOIR PRESSURE. 2 CAP UNLOCKED WHEN RED DOT SHOWS; LOCKED — GREEN DOT.
LIQUID OXYGEN	MIL-O-27210, TYPE II	NONE	1 TO BE FILLED ONLY BY QUALIFIED PERSONNEL. 2 USE MA-1 OR TYPE TMU27M TANK FOR SERVICE.
TIRE PRESSURE	SEE DECAL INBOARD OF EACH MAIN GEAR STRUT, UNDERSIDE OF WING SKIN SURFACE.	NONE	**WARNING** DO NOT USE HIGH PRESSURE SERVICE SYSTEM.
EXTERNAL ELECTRICAL POWER	M32A-60 (USAF) NC-5 (USN) CAN-C (CF)	NONE	1 OR A POWER UNIT WHICH MUST SUPPLY 3-PHASE, 115/200-VOLT, 400-CPS AC. 2 BATTERY START (LEFT ENGINE)
EXTERNAL AIR (JASU)	MA-1A (USAF) OR EQUIVALENT GTC-85 OR MA-1E (USN) WELLS AIR START SYSTEM	NONE	JASU RECOMMENDED MINIMUM OUTPUT: 350°F 42 PSIA 100 LB/MIN

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Figure 1-1-2

COCKPIT ARRANGEMENT (TYPICAL)

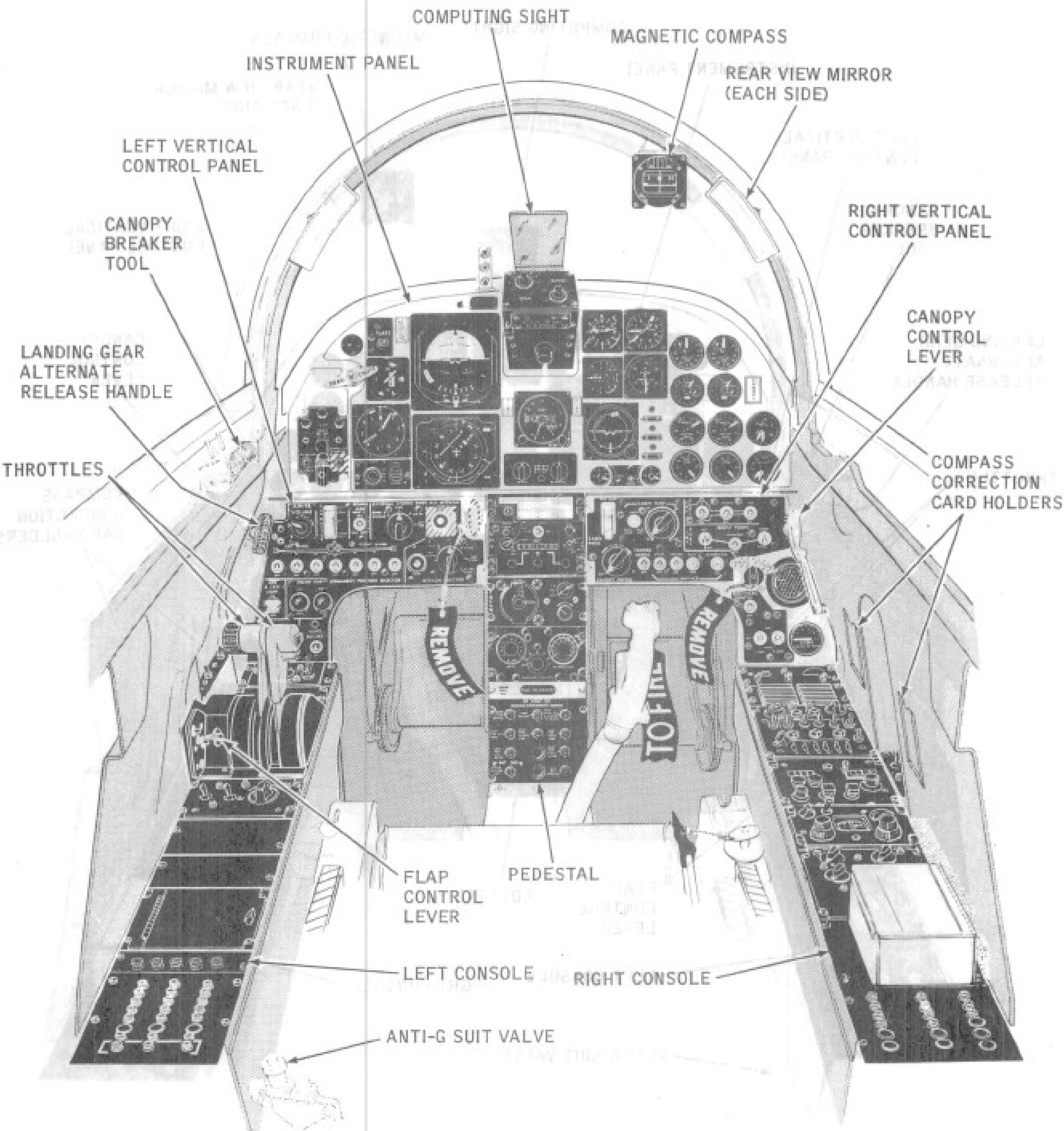


Figure 1-1-3

COCKPIT ARRANGEMENT (TYPICAL) A/R

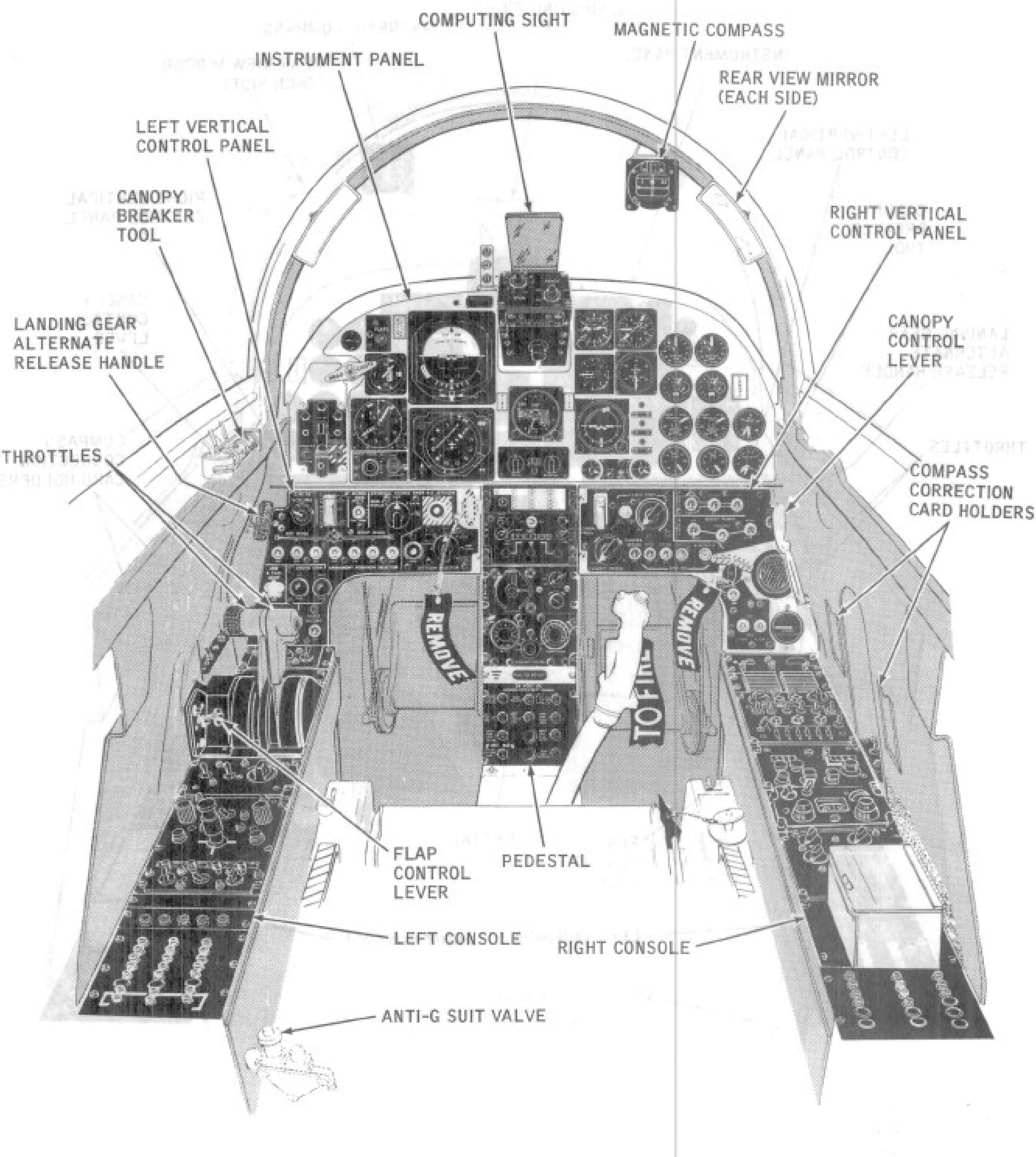
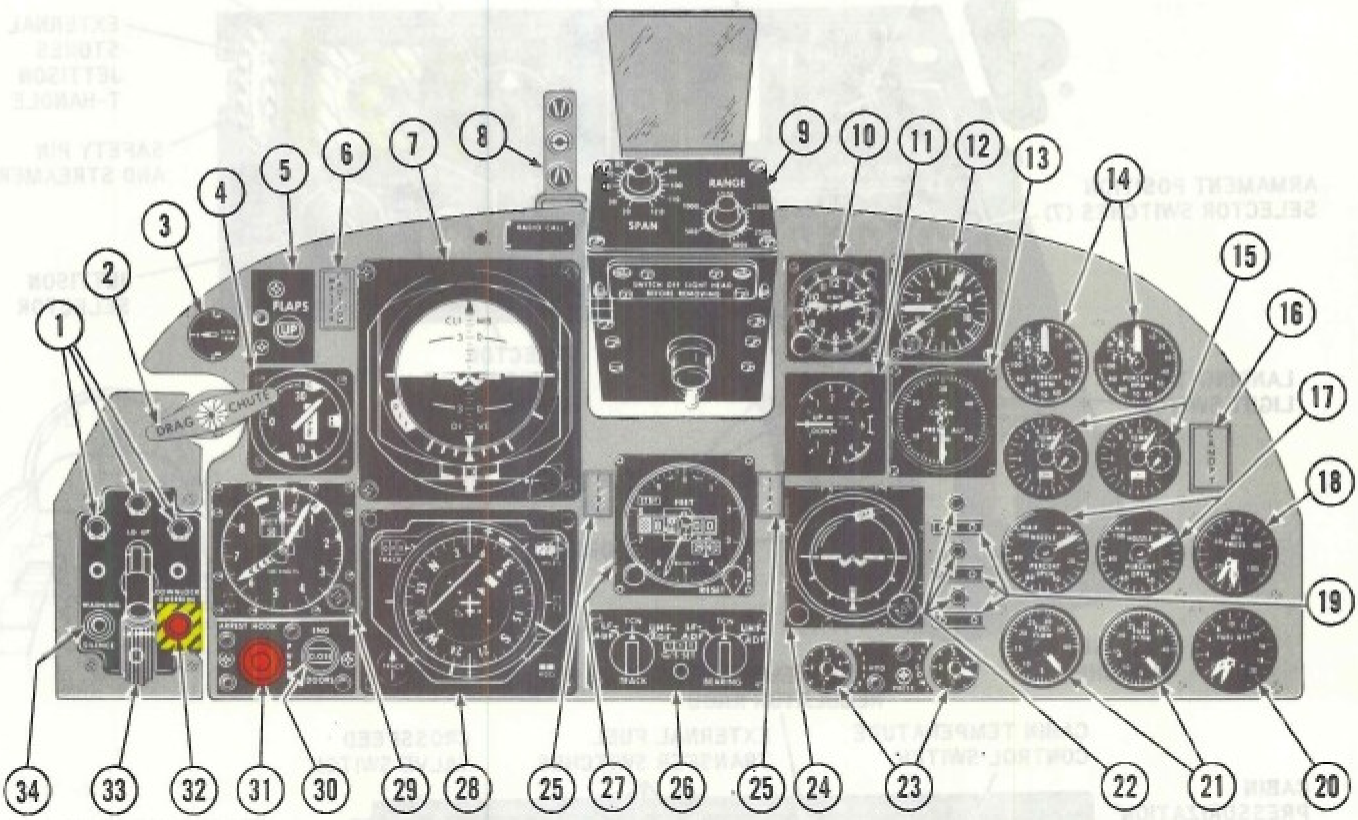


Figure 1-1-4

INSTRUMENT PANEL (TYPICAL)



1 LANDING GEAR POSITION INDICATOR LIGHTS
2 DRAG CHUTE T-HANDLE
3 PITCH TRIM INDICATOR
4 ANGLE-OF-ATTACK INDICATOR
5 FLAP POSITION INDICATOR
6 MASTER CAUTION LIGHT
7 ATTITUDE INDICATOR
8 APPROACH INDEXER
9 COMPUTING OPTICAL SIGHT
10 CLOCK
11 VERTICAL VELOCITY INDICATOR
12 ACCELEROMETER
13 CABIN PRESSURE ALTIMETER
14 ENGINE TACHOMETER INDICATORS
15 EXHAUST GAS TEMPERATURE INDICATORS
16 CANOPY UNLOCKED WARNING LIGHT
17 NOZZLE POSITION INDICATORS
18 DUAL OIL PRESSURE INDICATOR
19 CAMERA POSITION DATA CARD HOLDERS
20 DUAL FUEL QUANTITY INDICATOR
21 FUEL FLOW INDICATORS
22 CAMERA OPERATE LIGHTS
23 HYDRAULIC PRESSURE INDICATORS
24 STANDBY ATTITUDE INDICATOR
25 ENGINE FIRE WARNING LIGHT
26 INTEGRATED DISPLAY-OF-SITUATION CONTROL PANEL
27 ALTIMETER
28 HORIZONTAL SITUATION INDICATOR
29 AIRSPEED-MACH INDICATOR
30 ENGINE DOORS POSITION INDICATOR
31 ARRESTING HOOK BUTTON
32 LANDING GEAR LEVER DOWNLOCK OVERRIDE BUTTON
33 LANDING GEAR LEVER
34 LANDING GEAR WARNING SILENCE BUTTON

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Figure 1-1-5

VERTICAL CONTROL PANELS (TYPICAL)

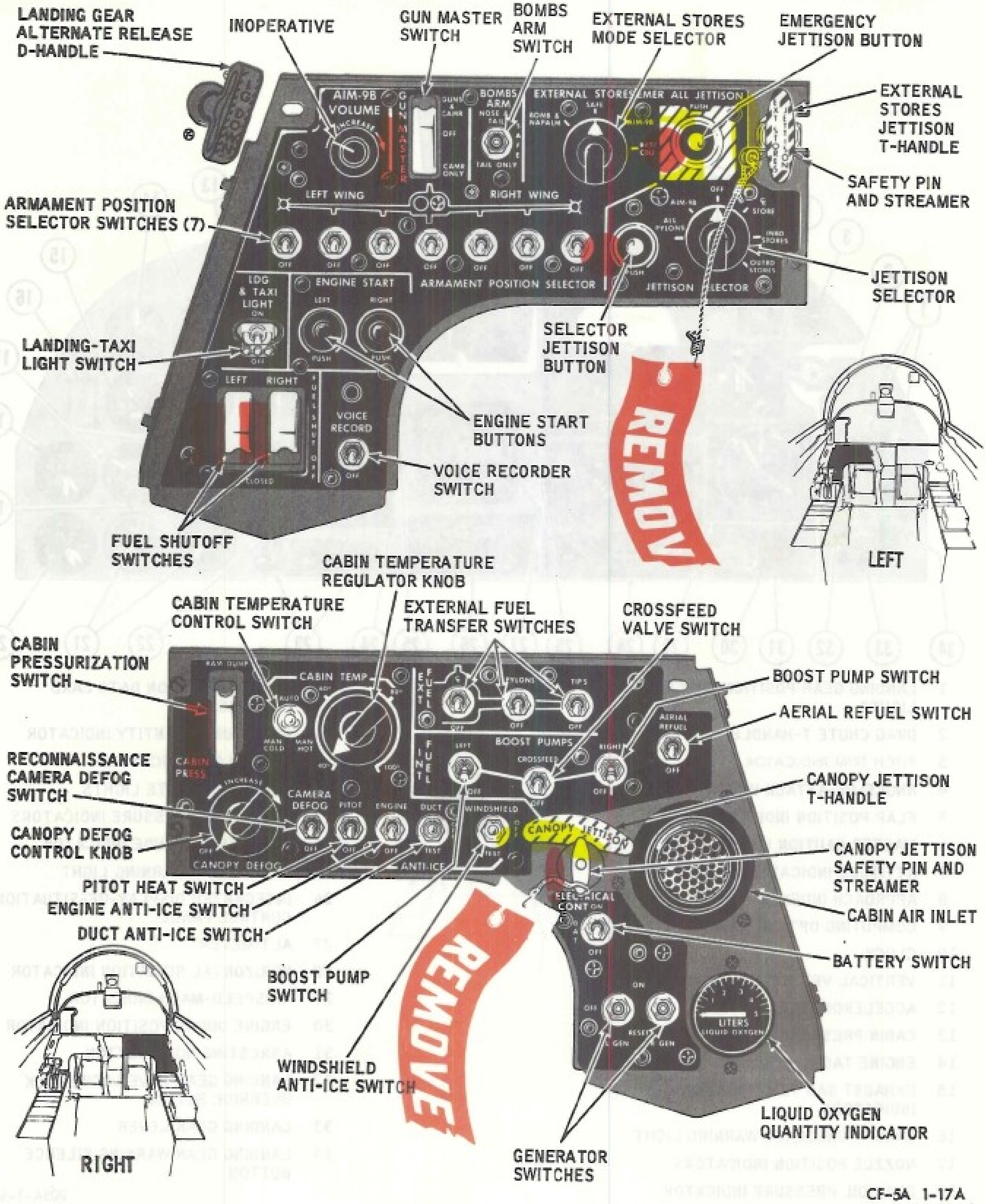

Figure 1-1-6

CONSOLE PANELS (TYPICAL)

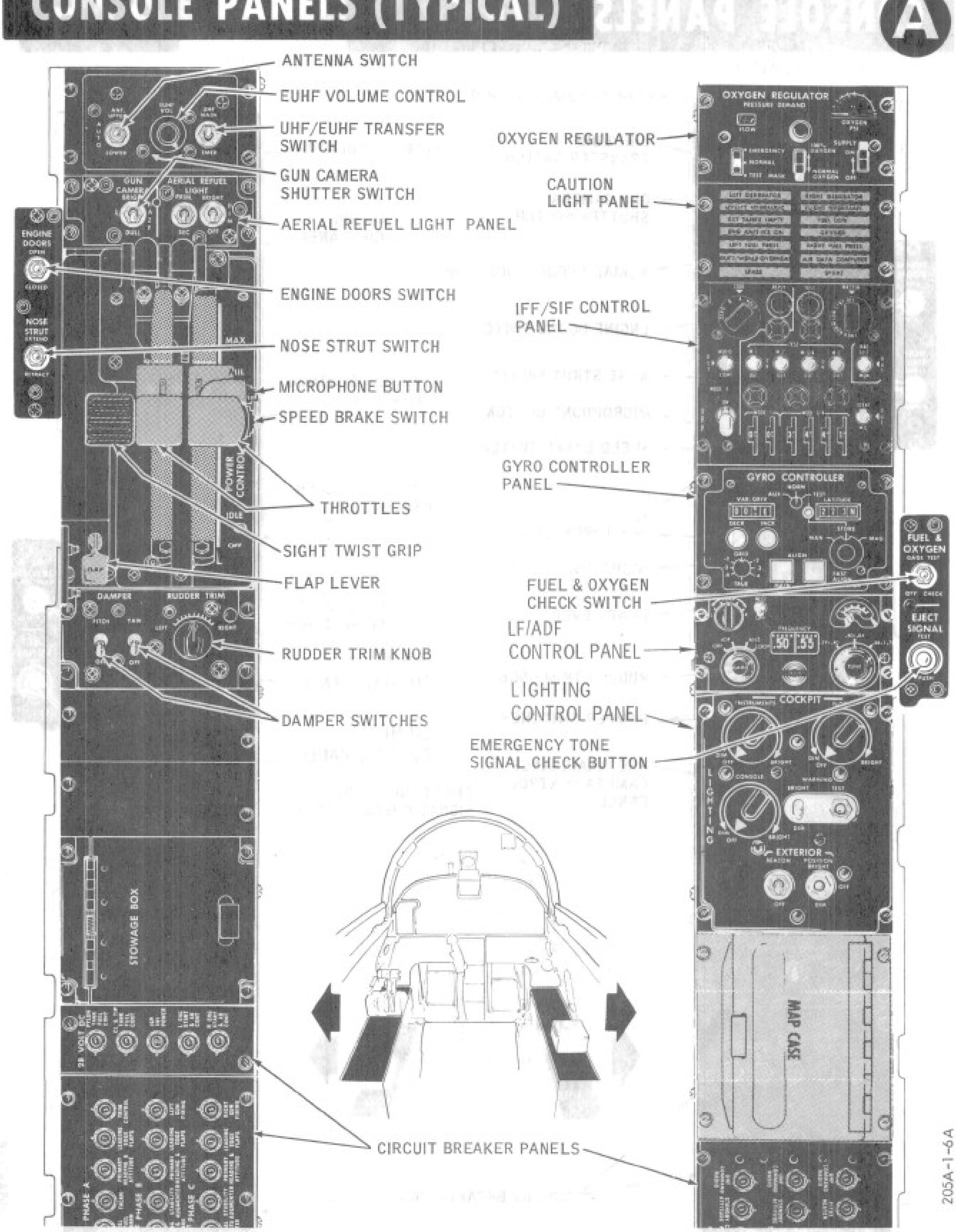

Figure 1-1-7

CONSOLE PANELS (TYPICAL) A/R

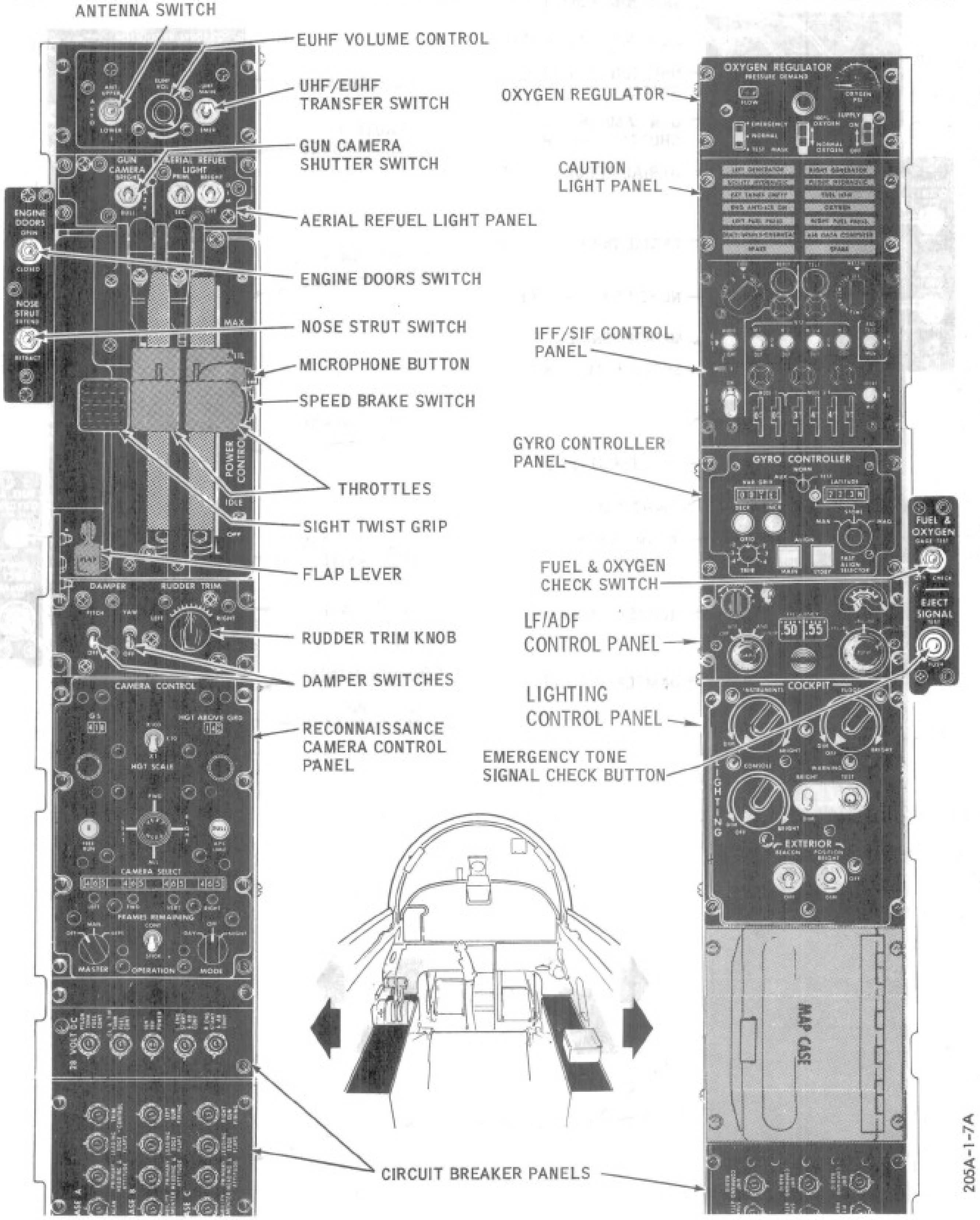

Figure 1-1-8

PEDESTAL (TYPICAL)

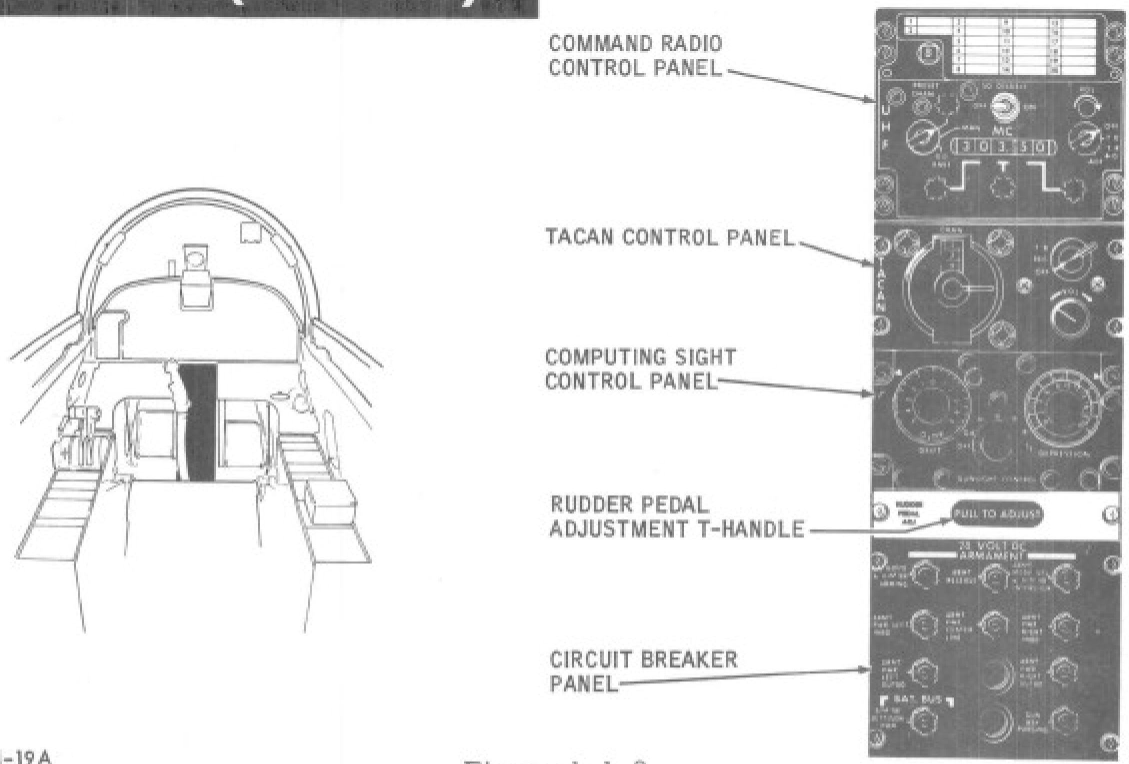

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Figure 1-1-9

SECTION 2

ENGINE

GENERAL

1 The aircraft is powered by two J85-CAN-15 eight stage, axial flow, turbojet engines equipped with afterburners, see Figure 1-2-1. Sea level static rated thrust for an uninstalled bare engine is 4300 pounds at maximum afterburner (MAX) power, 2925 pounds at military (MIL) power, and 2820 pounds at normal rated power. Air enters through two side fuselage structure inlet ducts and is directed into the compressor sections by variable inlet guide vanes. Additional air supply to each engine during aircraft takeoff operation is provided by engine auxiliary takeoff doors, which are structurally incorporated on the side of each fuselage air inlet duct. Automatically controlled inlet guide vanes and compressor air bleed valves reduce the possibility of a compressor stall. Compressor bleed air is also used to supply various pneumatic systems. The engine has a single-rotor eight-stage compressor coupled directly to a two-stage turbine. Exhaust gases from the combustor section pass through the two-stage turbine section and are discharged through a variable area nozzle. An exhaust gas temperature sensing system controls the variable exhaust nozzle area electrohydraulically to prevent excessive exhaust gas temperature at MIL and afterburner power settings while allowing optimum thrust efficiency.

ENGINE FUEL CONTROL SYSTEM

2 Each engine has a main fuel control and pump, an afterburner fuel control and pump, and an overspeed governor, see Figure 1-2-2. Fuel flow is controlled primarily by engine speed, engine inlet air temperature, compressor discharge pressure, and throttle position. The main fuel control automatically meters fuel according to engine requirements and provides automatic fuel flow control during ground or air starts.

J85-CAN-15 ENGINE

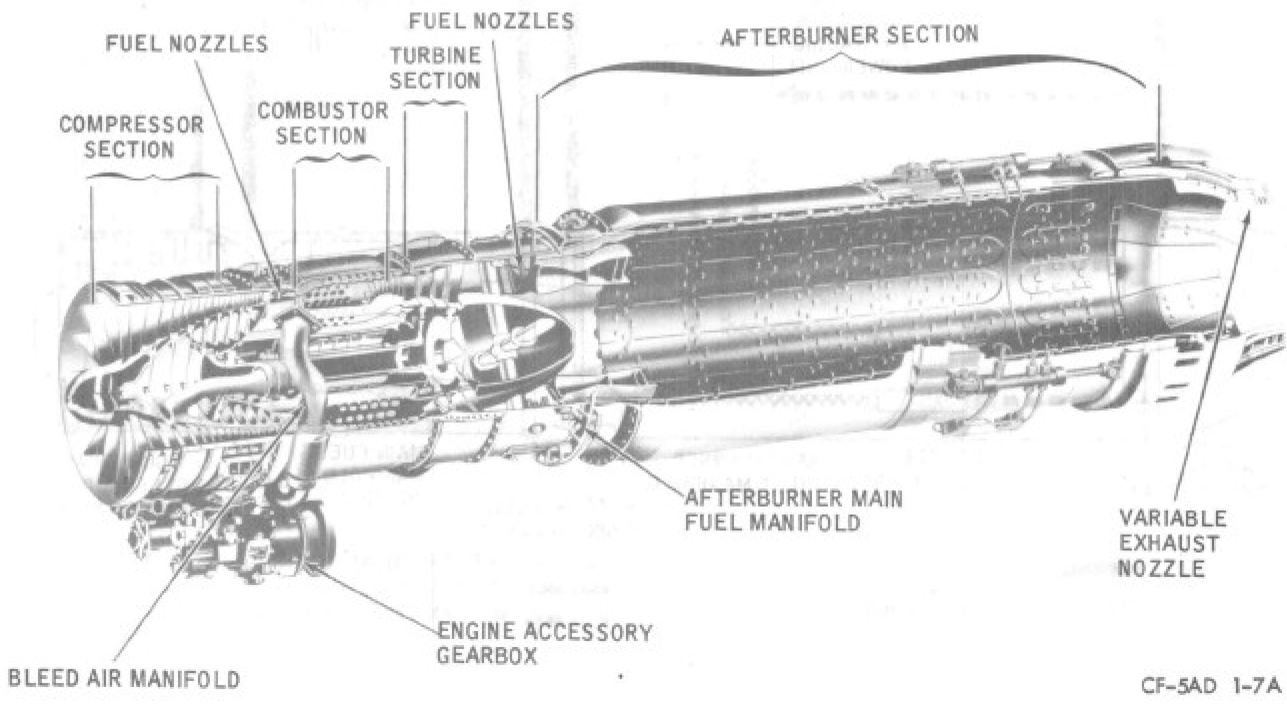


Figure 1-2-1

ENGINE FUEL CONTROL SYSTEM

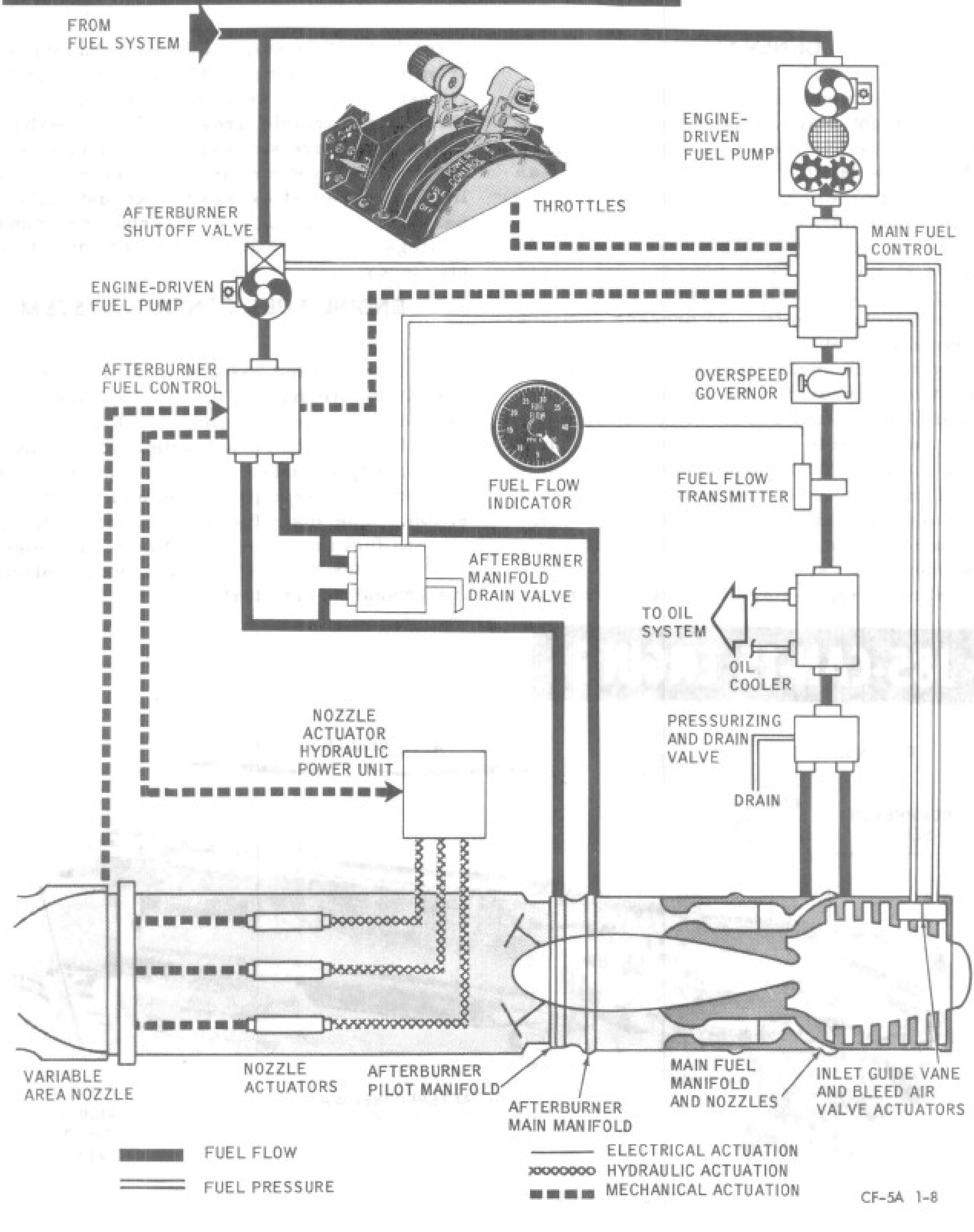


Figure 1-2-2

MAIN FUEL PUMP

3 The main engine-driven fuel pump is mounted on the forward right side of the engine accessory gearbox. The two-stage pump consists of a centrifugal impeller type boost element, which feeds through a filter screen to a positive displacement gear pump. A pressure relief valve allows excess pressure to bleed back to the pump inlet. The pump supplies fuel to the main fuel control.

MAIN FUEL CONTROL

4 The main fuel control mounted on the main engine-driven fuel pump has two sections, a metering section and a computing section. The fuel control computes and schedules fuel flow in conjunction with compressor discharge pressure, compressor inlet air temperature, engine speed, and throttle position. During normal engine operation, the fuel control regulates engine thrust by establishing an engine rpm and exhaust nozzle position according to the selected throttle position.

AFTERBURNER SYSTEM

5 The afterburner system consists of a shutoff valve, single-stage pump, main and pilot burner fuel manifold and spraybars, turbine discharge temperature sensing system, fuel control unit, and variable exhaust nozzle actuating controls. Afterburner operation is initiated by advance of the throttle from MIL. Ignition, simultaneous with fuel supply, occurs automatically when the throttle is placed in the afterburner range (MIL to MAX) and continues for approximately 30 seconds. At any time, afterburner operation may be discontinued when the throttle is retarded from the afterburner range to cut off ignition (if applicable) and fuel. Rate of fuel flow with the engine doors closed and with maximum afterburner is approximately 2700 pounds per hour per engine (pph/eng) static at sea level. On takeoff, with maximum afterburner and the engine doors open, the rate of fuel flow will increase approximately 200 to 300 pph/eng.

AFTERBURNER FUEL CONTROL

6 The afterburner fuel control is mounted directly on the afterburner fuel pump, which is mounted on the engine accessory gearbox left forward drive pad. The fuel control contains a fuel metering section, a computing section, and an exhaust nozzle control section. The control initiates and schedules fuel flow to the afterburner main and pilot burner spraybars under all conditions of afterburning. If a failure in the variable exhaust nozzle actuating system causes the nozzle to remain in a fixed position, afterburner fuel flow is automatically limited by the actual nozzle position to prevent excessive exhaust gas temperatures.

THROTTLES

7 The engines are controlled by individual throttles, see Figure 1-2-3, located on the left console. The throttles have fingerlifts on each lever which must be raised to move the throttle levers from IDLE to the OFF position. An increase in throttle friction (ramp effect) is provided at the MIL position (detent). This

THROTTLE QUADRANT

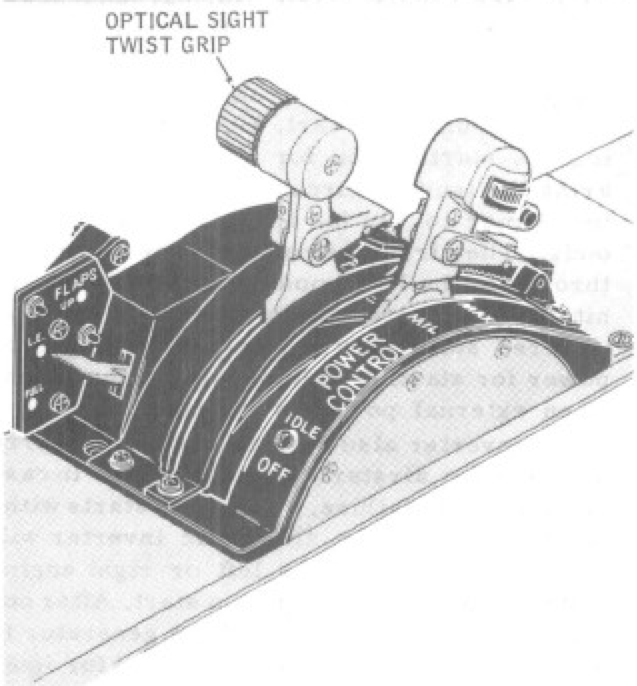


Figure 1-2-3 CF-5A 1-67

ramp effect must be overcome to ignite the afterburner. Afterburner thrust can be varied within the afterburner range and throttle friction is slightly more than that provided within the IDLE to MIL positions. The afterburner range of the throttle quadrant is from the high cam of the ramp at the MIL position (minimum afterburner) to the MAX position (maximum afterburner), providing a range of thrust modulation in afterburner operation. Each throttle controls main fuel shutoff valves, ignition circuitry, engine speed, and afterburner operation, and is ground adjustable for friction drag and load gradient feel for the MIL and MAX positions.

ENGINE START AND IGNITION SYSTEM

GENERAL

8 The aircraft uses a single-point, external, low-pressure air source for initial motoring of the engines during ground start. The external air inlet is in the underside of the fuselage aft section. A manually operated diverter valve, mounted on the left engine, is externally positioned by the ground crew to direct the flow of air to the selected engine during the start cycle. Two engine start buttons are provided on the left vertical control panel, see Figure 1-1-6. Normal sequence of starting is left engine first. External ac power for starting either engine, allowing monitoring of either engine instruments, is furnished by the ground start cart. Momentarily pressing the start button for the selected engine arms the ignition circuit and allows the ignition timer to run for approximately 30 seconds. The circuit is completed by moving the throttle to the IDLE position, causing the igniters to initiate engine start. A battery-powered static inverter provides ignition power for starting either engine on the ground when external power is not connected. The static inverter also acts as a standby source of power for airstart of either engine in case of dual engine failure. For engine starts without external power, the static inverter will supply ac power to the left or right engine instruments for monitoring a start. After one engine has been started and the generator is on the line, normal ac power is used for ignition when starting the other engine.

CROSSBLEED START

9 A crossbleed start cabability without external air is provided for starting the right engine after the left engine has been started. Compressed air from the eighth stage of the left engine compressor section is used for initial motoring of the right engine. An electrically operated crossbleed control valve, installed as part of the left engine compressor ducting, is armed when the left throttle is advanced above 70% rpm. When the right engine start button is pushed, the circuit is completed and the valve opens to permit a flow of compressor air from the left to the right engine. The right engine ignition circuit is then completed by moving the right throttle from OFF to IDLE position. In order to ensure adequate flow of air for starting, the left engine should be operating between 85% and 90% rpm. The crossbleed valve closes and power is removed from the valve-open circuit any time the left throttle is below approximately 70% rpm, the aircraft is airborne (the circuit is routed through the weight switch of the nose landing gear), or approximately 30 seconds after the right engine start button has been actuated.

EXHAUST GAS TEMPERATURE INDICATORS

10 The exhaust gas temperature (EGT) indicators on the instrument panel, see Figure 1-1-5, indicate biased engine EGT in degrees centigrade. Temperature signals are received from a system of thermocouples evenly spaced around the diffuser section. The instruments require ac power for operation. An OFF flag on the face of the instrument will appear when ac power is not available.

ENGINE TACHOMETER INDICATORS

11 Engine rotor speed in percent of maximum rated rpm of each engine is provided by engine tachometer indicators on the instrument panel, see Figure 1-1-7. Each indicator is self-powered by a tachometer generator mounted on each engine. The indicator is calibrated to indicate from 0 to 110%, with the main dial face marked in 10% increments and a second inner dial to indicate 1% increments up to 10%.

NOZZLE POSITION INDICATORS

12 A nozzle position indicator for each engine, located in the engine instrument group of the instrument panel, see Figure 1-1-5, indicates nozzle position in percent of full open. The nozzle is mechanically restricted to 98% of full open position.

ENGINE DOORS

GENERAL

13 Auxiliary air inlet duct engine doors on each side of the aft fuselage, see Figure 1-1-1, are provided to furnish additional air intake to engines. The doors bypass the "duct loss" effect during takeoff and low airspeed flight, thereby providing additional thrust. Each door consists of individually interconnected louvres which are opened and closed by electrical rotary actuators. The doors can be opened or closed manually and will close automatically, if open, at 255 (±10) KIAS. The doors will not reopen automatically when airspeed is reduced below 255 (±10) KIAS. The automatic closing function is provided by air pressure sensing within the central air data computer (CADC). An engine doors position indicator is provided to allow monitoring of the doors' position.

ENGINE DOORS SWITCH

14 The engine doors switch is located on the ENGINE DOORS control panel, which is located outboard of the throttle quadrant on the left console, see Figures 1-1-7 and 1-1-8. The switch has two placarded positions, OPEN and CLOSED, and is spring-loaded to a neutral off position. To open or close the doors manually, momentary action of the switch is sufficient.

CAUTION

Engine doors should not be opened at or above 0.45 mach to preclude loss of engine performance.

ENGINE DOORS POSITION INDICATOR

15 The engine doors position indicator on the instrument panel, see Figure 1-1-5, indicates the position of both doors. When both doors are fully open, the word OPEN will appear in the indicator window; when closed, the word CLOSE will appear. When one or both doors are in an intermediate position, or when one door is open and one door closed, a "barber-pole" indication will appear in the window.

OIL SYSTEM

16 Each engine has an independent, self-contained oil supply and lubrication system with a reservoir capacity of 0.83 Imperial (1.0 US) gallon. The system consists of an oil reservoir, a lube and scavenge six-element pump, check valves, relief valve, and oil filter and bypass. During engine operation, oil is pumped from the reservoir and delivered under pressure through the oil cooler and the oil filter to the engine accessory drive gearbox, main bearings, and other internal moving parts. Oil is returned to the reservoir through the scavenging system. A sump vent system maintains a positive pressure, making the lubrication system insensitive to altitude.

OIL PRESSURE INDICATOR

17 A dual-pointer oil pressure indicator, with the letters "L" and "R" for respective engines, on the instrument panel, see Figure 1-1-5, indicates oil pressure in pounds per square inch (psi).

ACCESSORY DRIVE SYSTEMS

18 An accessory drive system (ADS) is connected to each engine by a power shaft and operates the respective left and right hydraulic pump and ac generator. Scoops are provided in the underside of the fuselage, see Figure 1-1-1, to provide ram air cooling to the ADS's.

SECTION 3

FUEL SUPPLY SYSTEM

GENERAL

1 The aircraft fuel supply system, see Figure 1-3-1, consists of five bladder type cells, semipermanently mounted in the fuselage and divided into two independent systems, one for each engine. The left engine is supplied by the forward fuselage cell and the forward and aft dorsal cells; the right engine, by the centre and aft fuselage cells. Either system can supply both engines, or both systems can supply either engine through an electrically operated crossfeed valve. To increase the fuel supply, auxiliary tanks may be carried. The auxiliary tanks consist of removable nonjettisonable wingtip tanks, jettisonable inboard pylon tanks carried under each wing, and a jettisonable centreline pylon tank carried under the centre section of the fuselage. A quick-disconnect fitting, incorporating a check valve, closes when the pylon tanks are jettisoned to prevent fuel from flowing overboard. External fuel is transferred from the selected tanks to both internal systems through the single-point manifold by air pressure supplied by the cockpit pressurization system. Fuel is then supplied to the engines from the internal systems. An electrically driven boost pump within each internal system provides fuel under pressure to the engine-driven fuel pump. Each engine can be supplied with fuel by gravity flow from its respective system if a fuel boost pump fails.

NOTE

Air pressure will not be available for fuel transfer with the cockpit pressurization switch in the RAM DUMP position, nor with the aerial refuel switch at AERIAL REFUEL.

FUEL TANK CAPACITIES

2 For fuel quantity data, see Figure 1-3-2.

FUEL CONTROL PANEL

3 The fuel control panel is part of the right vertical control panel, see Figure 1-1-6. The panel contains left and right fuel boost pump switches, crossfeed valve switch, and fuel transfer switches for the externally carried auxiliary fuel tanks. All switches are two-position function switches (on/off). The boost pump switches are up-and-over lock-on type switches. This type switch is turned on in the normal manner, but the toggle must be pulled out and pushed down to turn the respective boost pump off. These switches prevent the boost pumps from being inadvertently turned off when the external fuel transfer switches are activated. For all normal operation, the fuel boost pump switches will be in the on (placarded LEFT and RIGHT) position and crossfeed valve switch in the OFF position. The external fuel transfer switches are positioned on (placarded CL, PYLONS, TIPS) as needed and returned to OFF when fuel transfer from the respective external tank group is completed.

CROSSFEED VALVE SWITCH

4 The crossfeed valve switch on the fuel control panel is used to electrically open or close the crossfeed valve in the bottom of the aft fuel cell. When the crossfeed switch is placed in the CROSSFEED position, the valve is opened, allowing fuel to be supplied to both operating engines with one fuel boost pump operative, supplying a single operative engine with both fuel boost pumps on, or permitting both fuel systems to supply fuel by gravity flow to a single operative engine if both fuel boost pumps fail. Refer to Part 2, Section 3, for use of crossfeed valve to balance fuel systems and while operating engine(s) with low fuel quantity.

FUEL TRANSFER

5 Sequencing of external fuel when all external tanks are installed is wing pylons, centreline, and tips. When transferring external

FUEL SYSTEM (TYPICAL)

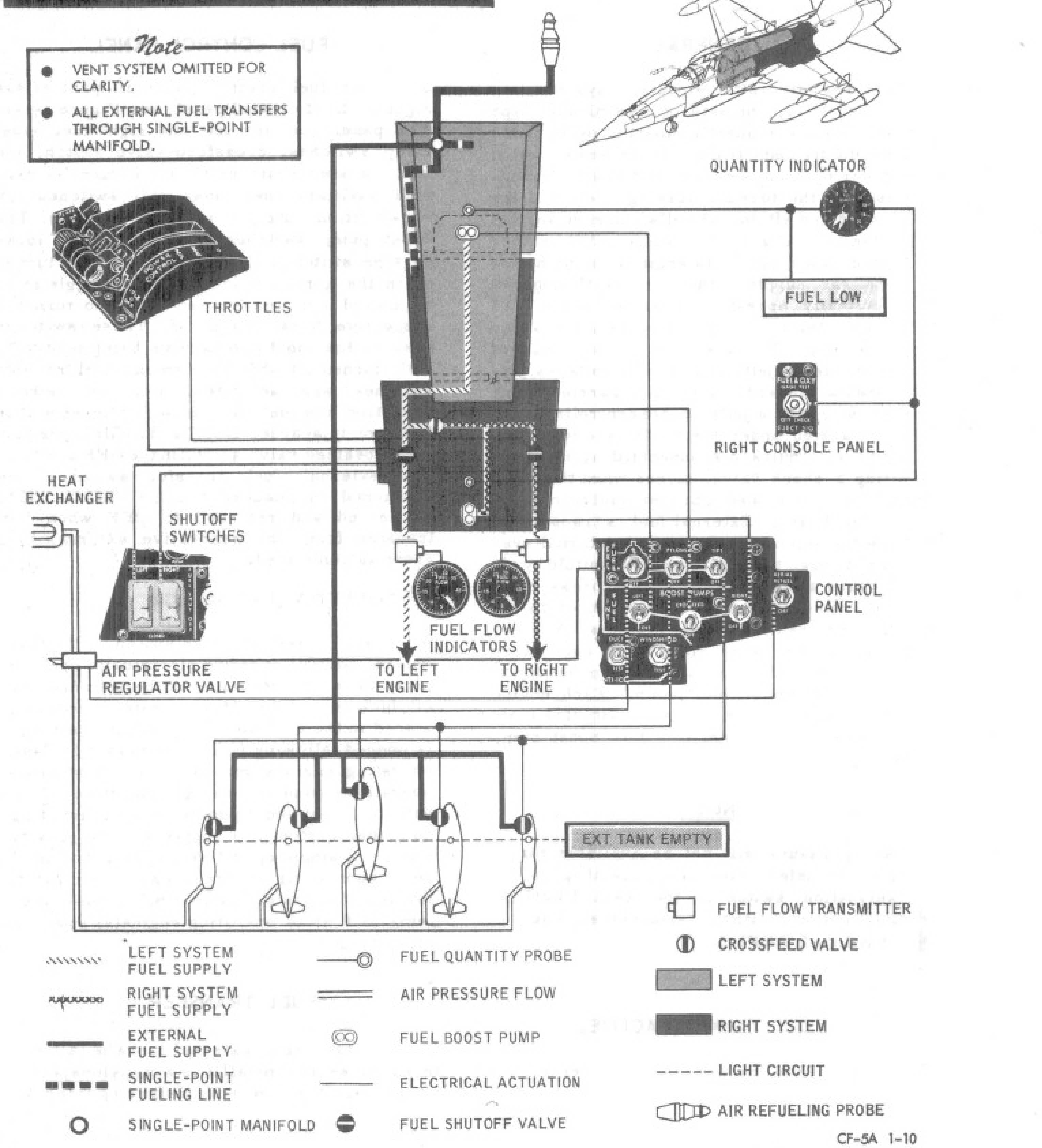
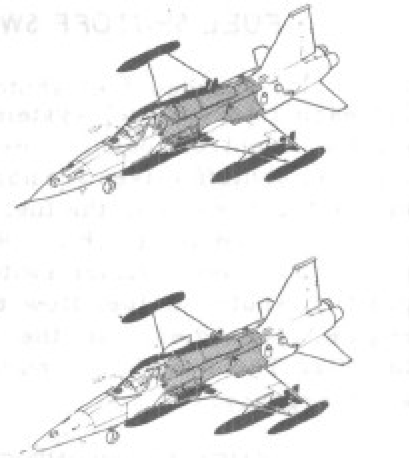


Figure 1-3-1

FUEL QUANTITY DATA

SINGLE-POINT REFUELING

FULLY SERVICED		SYSTEMS/TANKS FUELED	USABLE FUEL	
GALLONS IMPERIAL (US)	POUNDS		GALLONS IMPERIAL (US)	POUNDS
498 (598)	3888	TOTAL FUEL — INTERNAL	485 (583)	3790
244 (293)	1905	LEFT SYSTEM	238 (286)	1859
254 (305)	1983	RIGHT SYSTEM	247 (297)	1931
461 (554)	3601	TOTAL FUEL — EXTERNAL	455 (546)	3549
126 (152)	988	CENTRELINE PYLON TANK 125 IMP. GAL (150 US GAL)	125 (150)	975
253 (304)	1976	WING PYLON TANKS (2) 125 IMP. GAL (150 US GAL)	250 (300)	1950
82 (98)	637	TIP TANKS (2) 42 IMP. GAL (50 US GAL)	80 (96)	624
580 (696)	4525	MAXIMUM FUEL — INTERNAL PLUS TIPS	565 (679)	4414
959 (1152)	7489	MAXIMUM FUEL — INTERNAL, TIPS, CENTRELINE, AND WING TANKS	940 (1129)	7339



DATA BASIS
- CALIBRATED
- STANDARD DAY
- FUEL: JP-4
- 7.8059 LB/IMP. GAL (6.5 LB/US GAL)

Note

MANUAL REFUELING WILL REDUCE TOTAL USABLE INTERNAL FUEL 5 IMPERIAL (6 US) GALLONS AND REDUCE WEIGHT 39 POUNDS.

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Figure 1-3-2

fuel, each tank group must be selected individually and the respective switch returned to OFF when the transfer is completed. If the switch is not returned to OFF, the external tanks empty light, refer to Paragraph 13, following, will remain on and no longer function as an indicator for the completion of later external fuel transfer selections. Placing the fuel transfer switch to the placarded position not only opens the fuel valve of the selected tank or tanks but also allows the compressed air from the air-conditioning system to pressurize all external tanks. If an improper tank group is selected, returning the transfer switch to OFF will close the valve or valves, stopping fuel transfer. Tanks will then depressurize unless another external tank group is selected. Refer to Part 2, Section 3, for system operation and Part 4, Section 1, for limitations.

FUEL BOOST PUMPS

6 Two electrically driven, double-ended fuel boost pumps are provided, one for each fuel system. The left system boost pump is in the inverted flight compartment of the forward fuel cell; the right system boost pump is in the inverted flight compartment of the aft fuel cell. Each pump is capable of supplying sufficient fuel to both engines at MAX power, if necessary. If both pumps fail, sufficient fuel will flow by gravity to maintain MAX power from sea level to 6000 feet. Operating with boost pumps off at altitudes above 6000 feet may result in engine fuel starvation. During inverted flight, the boost pumps will provide a continuous fuel flow to the engines for a limited time at maximum AB (approximately 10 to 30 seconds) dependent upon altitude. Refer to Part 4, Section 1, for limitations.

FUEL SHUTOFF SWITCHES

7 Two guarded fuel shutoff switches, one for each internal fuel system, are on the left vertical control panel, see Figure 1-1-6. The fuel shutoff valves are normally controlled by the throttles, with the fuel shutoff switches in the normal (LEFT, RIGHT) guarded positions. Placing either switch in the CLOSED position shuts off fuel flow to the respective engine, regardless of the position of the throttle. The switches should be used only in an emergency.

FUEL FLOW INDICATORS

8 A fuel flow indicator on the instrument panel, see Figure 1-1-5, indicates fuel flow for each engine. Rate of fuel flow is measured in pounds per hour by an engine-mounted, electrically-powered fuel flowmeter transmitter which furnishes fuel flow rate to the indicator. Only fuel consumption by the main engine is measured and indicated by the system; afterburner fuel consumption is not measured or indicated. The indicator dial is calibrated in pounds per hour (pph) from 0 through 40 (X 100) with numerals at the 500 pph graduations and incremental graduation markers at every 100 pph.

FUEL QUANTITY INDICATOR

9 A dual-pointer fuel quantity indicator, with the letters "L" and "R" for respective left and right engine internal fuel systems, is provided on the lower right side of the instrument panel, see Figure 1-1-5. The indicator is a capacitance type, operating on alternating current, and indicates pounds (LBS x 100) of usable fuel in each internal fuel system. No provisions are made for indicating external fuel quantity; therefore, fuel in external fuel tanks should be checked during walk-around inspection. Visual indicators are provided on the wingtip tanks. These are two-position type and indicate the word FULL when tank compartment is full and blank when other than full. If tip tanks have been serviced, both indicators on each tank must indicate FULL before flight. No visual method of checking servicing of the pylon and centreline tanks is possible without removing the filler caps.

FUEL AND OXYGEN CHECK SWITCH

10 A fuel and oxygen check switch outboard of the gyro controller panel on the right console panel, see Figures 1-1-7 and 1-1-8, has three positions: GAGE TEST, QTY CHECK, and a spring-loaded unmarked off position. Before starting the engines, the switch should be used to check the operation of the static inverter as well as check the fuel and oxygen quantity. With the battery switch in the ON position, and no external power or generators operating, placing the fuel and oxygen check switch in either the QTY CHECK or GAGE TEST position actuates the static inverter, which then supplies ac power to the fuel and oxygen quantity indicators. With external power or the generators operating, ac power is supplied to the fuel and oxygen quantity indicators to indicate fuel and oxygen quantity. When the switch is held in the GAGE TEST position, the fuel and oxygen quantity indicator pointers will move toward zero if the indicators are operative.

FUEL LOW CAUTION LIGHT

11 The fuel low caution light on the caution light panel, see Figures 1-1-7 and 1-1-8, and the master caution light on the instrument panel, see Figure 1-1-5, will come on whenever the fuel quantity indicator pointer for either system indicates approximately 250 pounds or less for more than 7.5 seconds.

FUEL PRESSURE CAUTION LIGHTS

12 Individual fuel pressure caution lights for left and right fuel systems are located on the caution light panel on the right console, see Figures 1-1-7 and 1-1-8. When illuminated, the lights display the legends LEFT FUEL PRESS and RIGHT FUEL PRESS respectively. When fuel pressure falls to 6 psi or less in a system, the related caution light and the master caution light on the instrument panel, see Figure 1-1-5, will illuminate.

EXTERNAL TANKS EMPTY LIGHT

13 An external tanks empty light placarded **EXT TANKS EMPTY** on the caution light panel, see Figures 1-1-7 and 1-1-8, and the master caution light on the instrument panel, see Figure 1-1-5, will illuminate when the fuel transfer of a selected tank group is completed. When the applicable fuel transfer switch is turned OFF, the master caution light and the caution panel light will extinguish. The EXT TANKS EMPTY caution light system is rearmed after a 30-second time delay when another external tank group is selected.

SECTION 4

AIR REFUELING SYSTEM

GENERAL

1 Air refueling permits all internal fuel cells and external fuel tanks to be filled from tanker aircraft by means of probe-and-drogue type refueling equipment. The probe, see Figure 1-4-1, consisting of a boom and nozzle, is a detachable unit which is mounted on the right side of the fuselage and connected through an adapter elbow to the single-point refueling system. The system is capable of air refueling total fuel in approximately 4 minutes. A light at the forward probe fairing shines forward to light the probe-and-tanker drogue for night refueling operations. Refer to Part 2, Section 4, for air refueling procedures.

AIR REFUELING CONTROLS

AERIAL REFUELING CONTROL SWITCH

2 The aerial refueling control switch on the right vertical control panel, see Figure 1-4-1, has two placarded positions: AERIAL REFUEL and OFF. The OFF position allows the external fuel tanks to pressurize and feed normally as selected. The AERIAL REFUEL position will allow selected external tanks to depressurize when tank positions are selected for refueling. If external tanks are to be air refueled, the air refueling switch must be placed in the AERIAL REFUEL position just prior to hookup with the tanker drogue.

NOTE

Return switch to OFF position after completion of air refueling to restore external tank pressure for fuel system operation when external tanks are selected.

EXTERNAL FUEL TRANSFER SWITCHES

3 The external fuel transfer switches (placarded CL, PYLONS, and TIPS) on the right vertical control panel, see Figure 1-4-1, are selected, if required, prior to engagement with the tanker drogue. Placing each switch in the position selected will open the respective fuel valve of selected external tanks. With the aerial refueling switch positioned at AERIAL REFUEL, the external tanks are depressurized and are filled by reverse flow of fuel from the internal fuel system.

CAUTION

Fuel transfer switches for each tank group must be turned off when tanks are full; otherwise, improper fuel management will result after aerial refueling is completed.

AIR REFUELING PROBE LIGHT SWITCH

4 The light on the refueling probe fairing is controlled by two switches on the panel placarded AERIAL REFUEL LIGHT on the left console, see Figure 1-4-1. The switch on the left side of the panel controls a primary (PRIM.) and secondary (SEC) lamp selection. The switch on the right has three positions, BRIGHT, DIM, and OFF, and is used to control operation of either the PRIM. or SEC lamp in the probe light.

AIR REFUELING SYSTEM

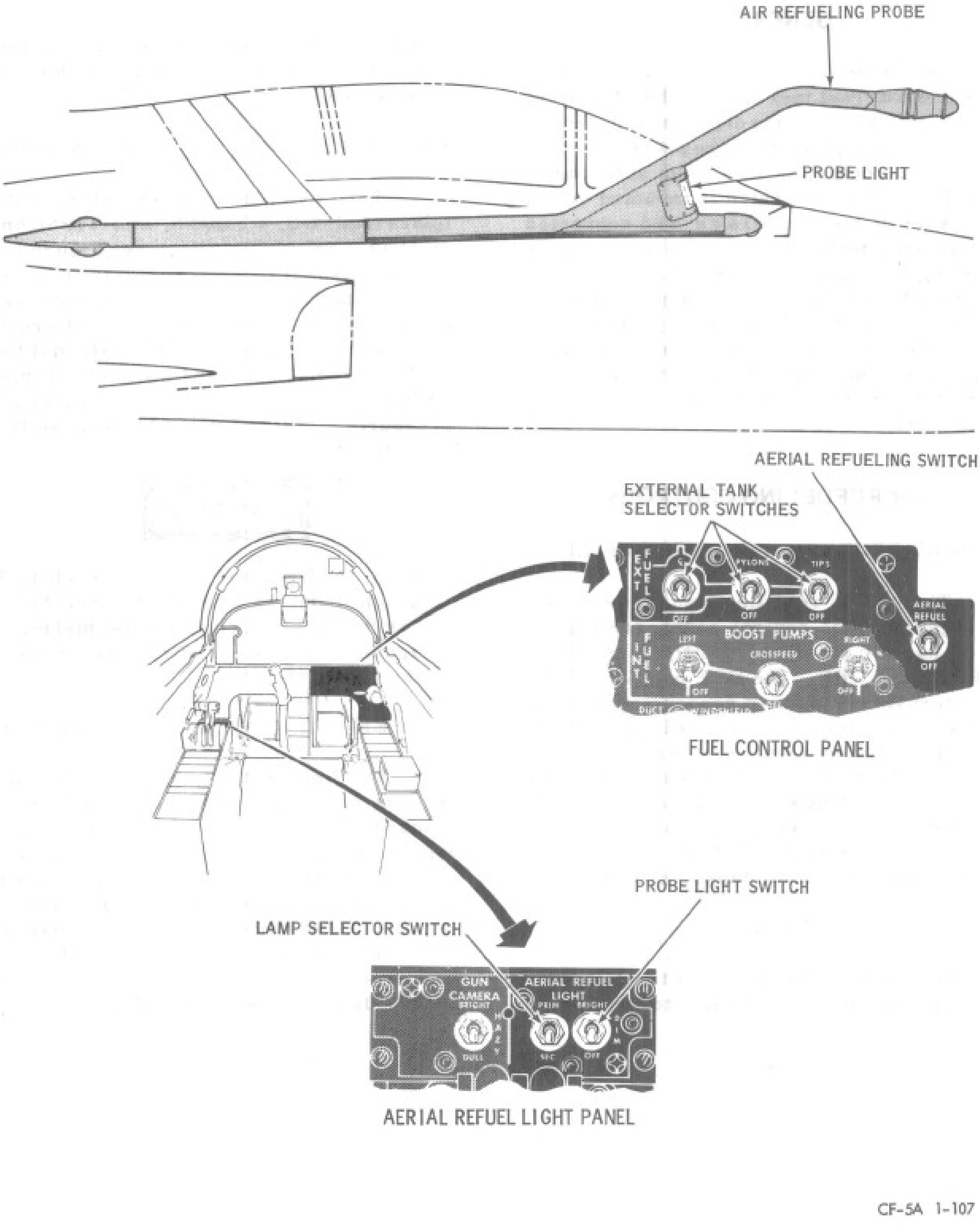


Figure 1-4-1

SECTION 5

EXTERNAL STORES JETTISON SYSTEM

GENERAL

1 Three methods of jettisoning external stores are available to the pilot; selective jettison and two methods of emergency jettison, see Figure 1-5-1. When all external stores are salvo jettisoned, wing shock loads are reduced by the following electrical sequencing of the release mechanism: outboard and centreline pylon stores released simultaneously followed in 150 milliseconds by the inboard pylon stores. Jettison of either the external stores and/or pylons can be accomplished on the ground or in-flight with the landing gear up or down. Refer to Part 4, Section 1, for external store and/or pylon release or jettison limits. To prevent inadvertent release of either the external stores and/or the pylons from the aircraft, prior to flight, ground safety pins are installed, see Figure 1-11-1. Refer to Part 2, Section 7, for complete weapons system description and operation.

WARNING

Following an attempted release or jettison, any munition that does not separate from the aircraft should be considered armed and susceptible to inadvertent release during landing impact.

EXTERNAL STORES JETTISON

2 External stores jettison controls are located on the left vertical control panel, see Figure 1-1-6.

SELECTIVE JETTISON

3 Selective jettison is accomplished by positioning the switch placarded JETTISON SELECTOR to the stores station desired and pushing the button immediately to the left of the selector switch. Pushing the button jettisons the particular external stores carried on the selected pylon station, using battery bus power.

EMERGENCY ALL JETTISON BUTTON

4 Emergency jettison of all stores except tip tanks, using battery bus power, can be accomplished by pushing the button placarded EMER ALL JETTISON.

EXTERNAL STORES JETTISON T-HANDLE

5 Pulling the T-handle placarded EXT STORES JETTISON, excites a one-time emergency jettison battery, which will supply electrical power to the stores jettison circuit. The T-handle is primarily designed for use

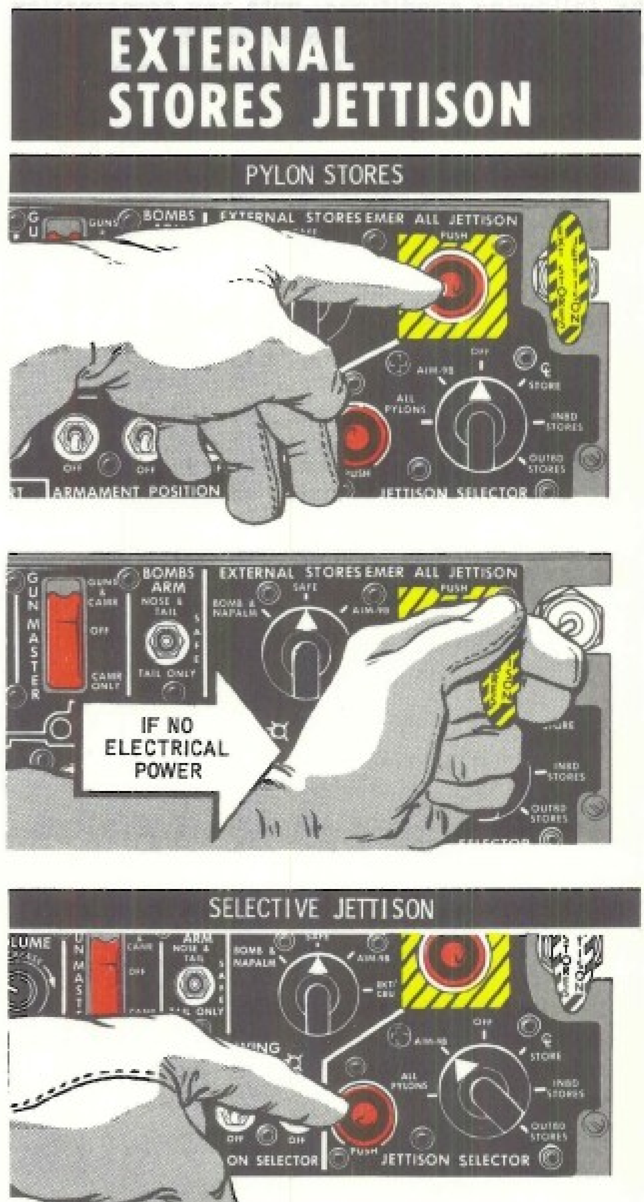


Figure 1-5-1

when the aircraft battery bus power fails; however, the handle may be used at any time emergency jettison is desired. The T-handle, when pulled, will not return to the fully stowed position. The T-handle is repositioned when the thermal battery is replaced by ground service personnel.

PYLON JETTISON

6 The five pylons are jettisonable units when the explosive bolt kits are used for pylon installation. Pylon jettison may be accomplished in salvo only, but under any of the following conditions: with any combination of empty or loaded pylons; on the ground or in the air; and with the landing gear up or down.

7 With the JETTISON SELECTOR switch positioned at ALL PYLONS, depression of the normal jettison button to the left of the selector will electrically initiate the explosive charge in each bolt to release the pylons. If ALL PYLONS is selected before the stores have been released, pylon jettison will occur 1 to 1.5 seconds after release of stores, refer to Paragraph 1, preceding.

WARNING

If a "hung" store has resulted from an earlier selection, attempt to release the store by G application and additional store selection before releasing the pylon. The release path of a store/pylon combination is unpredictable.

SECTION 6

ELECTRICAL POWER SUPPLY SYSTEM

GENERAL

1 Two alternating current (ac) systems and one direct current (dc) system, see Figure 1-6-1, supply electrical power to the aircraft. The ac power supply systems consist of two identical ac generator systems and an ac external power receptacle. The dc power supply system consists of a battery and two 25-ampere, 28-volt dc, transformer-rectifiers. An additional transformer-rectifier is installed when the aircraft is fitted with the photoreconnaissance camera nose.

AC POWER SYSTEM

2 AC power is supplied by two 15-kva 380 to 420-cps generators, one operating from the accessory drive system of each engine. Each generator functions independently and supplies 115/200-volt 3-phase power to the primary buses of the aircraft. Normally, power distribution is divided between the right and left systems. However, should one generator fail or be turned off, the functioning generator will assume the full load without disruption. Generators are cut in individually when engine rpm reaches approximately 35%-40% and should be "on-line" at engine idle.

GENERATOR SWITCHES AND CAUTION LIGHTS

3 Two generator switches, see Figure 1-1-6, one for each generator, are located on the right vertical control panel. Generator caution lights are located on the caution light panel, see Figures 1-1-7 and 1-1-8. The generator caution light will come on any time the respective generator fails or is turned off. The generator switch has a RESET position, allowing the pilot to reset the generators if necessary.

DC POWER SYSTEM

4 Primary dc power is obtained from each ac system through a transformer-rectifier which converts ac power to dc power. The 24-volt, 11-ampere-hour battery serves as a standby source of power for all dc circuits and is charged by the transformer-rectifiers. If one transformer-rectifier fails, the other(s) continues to supply all dc power requirements.

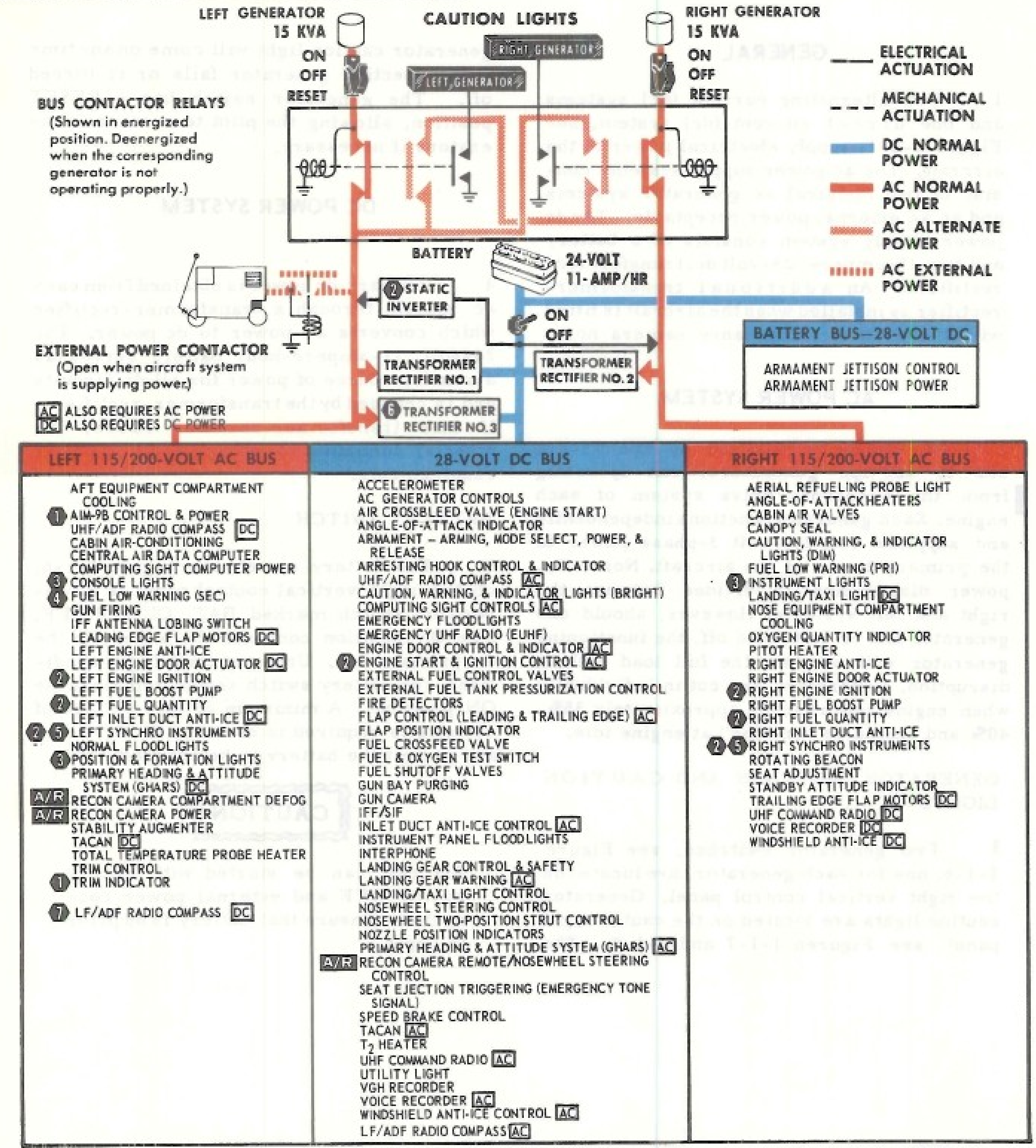
BATTERY SWITCH

5 The battery switch, see Figure 1-1-6, on the right vertical control panel is a two-position switch marked BAT. ON and OFF. The ON position connects the battery to the dc primary bus. Under normal flight conditions, the battery switch should remain in the ON position. A minimum battery voltage of 18 volts is required to close the battery relay and permit the battery to be charged.

CAUTION

Aircraft can be started with battery switch OFF and external power connected. Ensure that battery is on prior to flight.

ELECTRICAL SYSTEM



1. POWERED BY 28-VOLT AC THROUGH TRANSFORMER.
2. STATIC INVERTER SUPPLIES 115-VOLT AC POWER FOR:
   (a) NORMAL ENGINE STARTS (ONE ENGINE)
   (b) EMERGENCY IN-FLIGHT ENGINE STARTS
   (c) SYNCHRO INSTRUMENTS DURING NORMAL ENGINE STARTING (FIRST ENGINE)
   (d) FUEL & OXYGEN QUANTITY INDICATORS WHEN TEST SWITCH IS OPERATED
3. POWERED BY 6-VOLT AC THROUGH TRANSFORMER.
4. POWERED BY LEFT AC BUS ONLY WHEN RIGHT AC BUS NOT ENERGIZED.
5. SYNCHRO INSTRUMENTS CONSIST OF:
   (a) EXHAUST GAS TEMPERATURE INDICATORS
   (b) FUEL FLOW INDICATORS
   (c) OIL PRESSURE INDICATORS
   (d) HYDRAULIC PRESSURE INDICATORS
6. INSTALLED WHEN A/R CAMERA NOSE CONFIGURED, TO POWER RECON CAMERAS.
7. POWERED BY 26-VOLT AC THROUGH TRANSFORMER

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Figure 1-6-1

CIRCUIT BREAKER PANELS

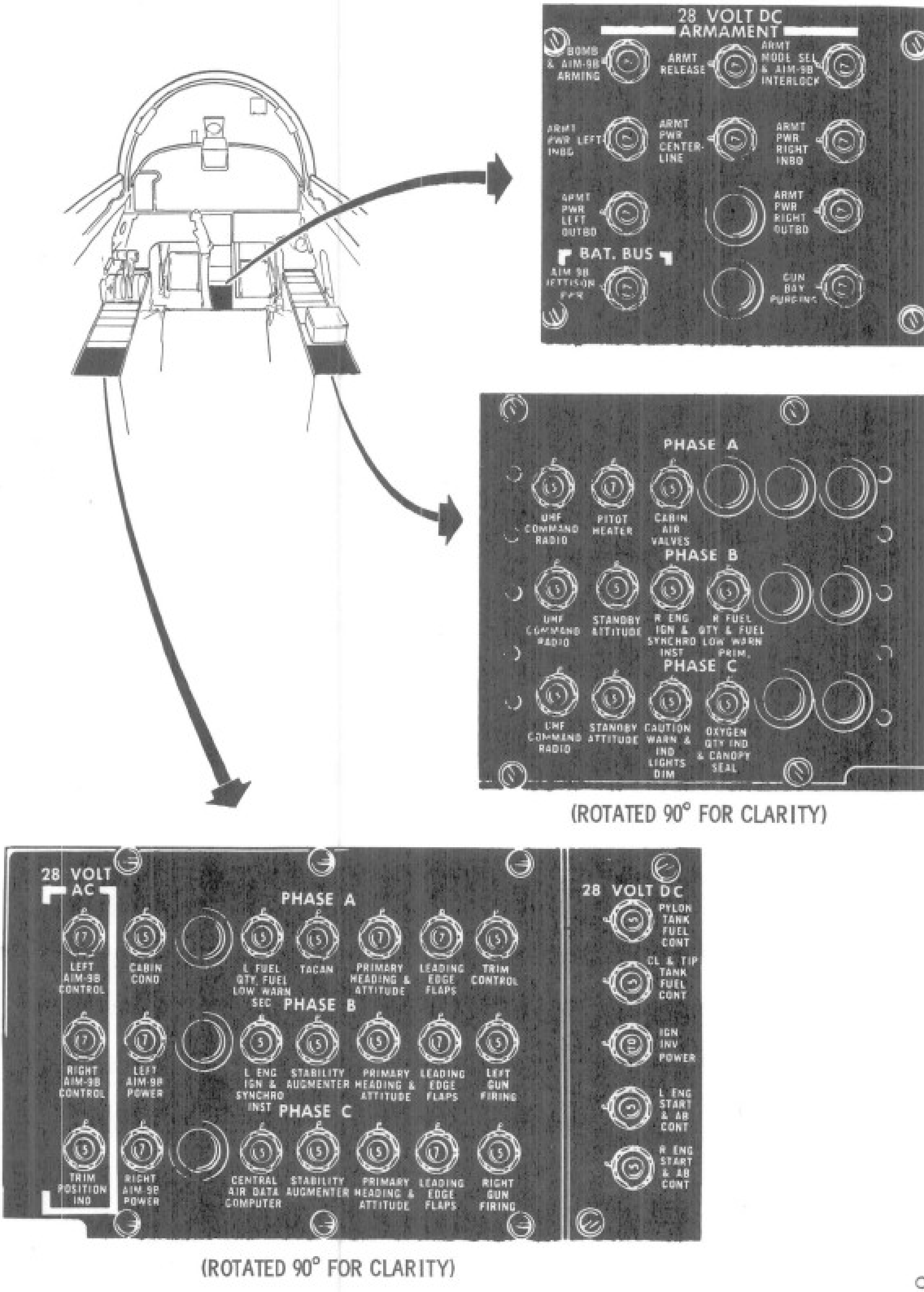

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Figure 1-6-2

SECTION 7

HYDRAULIC POWER SUPPLY SYSTEMS

GENERAL

1 Hydraulic power, see Figure 1-7-1, is supplied by two systems, the flight control system and the utility system, each one independent of the other. Each system is powered by a positive displacement piston type pump. The right accessory drive system (ADS) drives the flight control system pump, and the left ADS drives the utility system pump. Both systems operate at 3000 psi. Each system provides one-half the hydraulic power for the flight controls. In addition, the left system provides the hydraulic power to operate the landing gear, two-position nose strut, gear doors, speed brake, wheel brakes, stability augmenter, nosewheel steering, and the gun gas purge doors and gun gas deflector doors. Hydraulic pressure indicators and low-pressure warning lights are provided for each system. Individual system hydraulic reservoirs supply each system with hydraulic fluid. See Figure 1-1-2 for servicing data.

HYDRAULIC PRESSURE INDICATORS

2 The hydraulic systems pressure indicators are located on the instrument panel, see Figure 1-1-5. The indicators provide a visual indication of the pressure available in each hydraulic system. The indicator dials are calibrated in 500-psi graduations from 0 to 4000 psi. See Figure 4-1-1 for instrument marking and pressure limits.

HYDRAULIC PRESSURE CAUTION LIGHTS

3 A hydraulic pressure light for each system is provided on the caution light panel, see Figures 1-1-7 and 1-1-8. The caution lights are placarded UTILITY HYDRAULIC and FLIGHT HYDRAULIC. The master caution light on the instrument panel, see Figure 1-1-5, and the hydraulic pressure caution lights will come on when the respective system pressure drops to 1500 psi or less to indicate a low-pressure condition. The lights will automatically go out when a pressure of approximately 1800 psi is restored.

HYDRAULIC SYSTEMS

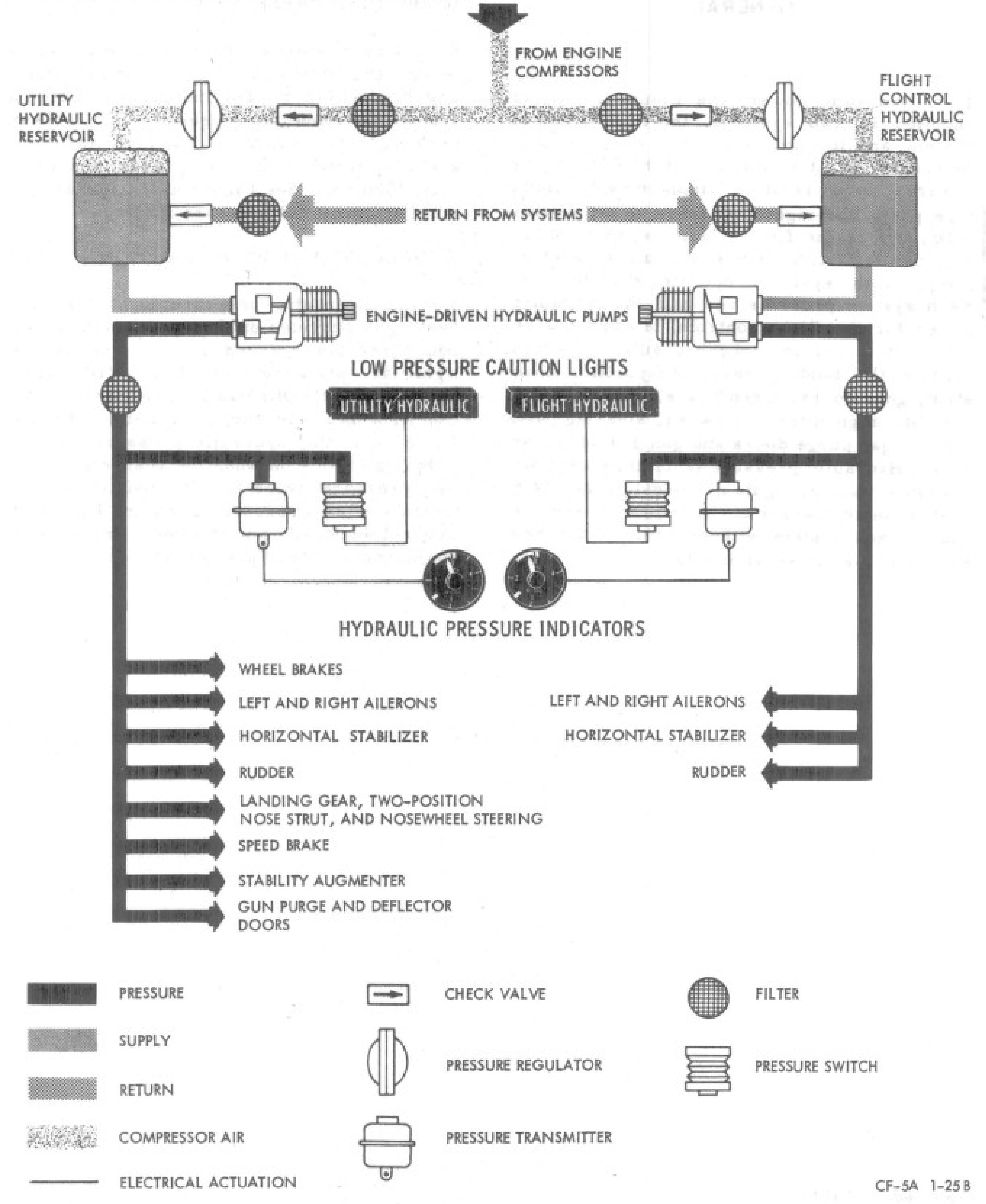


Figure 1-7-1

SECTION 8
FLIGHT CONTROL SYSTEM

GENERAL

1 The flight control system consists of an all-movable horizontal stabilizer for pitch control, ailerons for roll control, and a rudder for yaw control. All control surfaces are actuated by dual hydraulic actuators, one powered by the utility hydraulic system (left engine) and the other by the flight control hydraulic system (right engine). If either system malfunctions, hydraulic power to the flight control system will continue to be available. With one hydraulic system inoperative, there is no noticeable difference in flight control response. Artificial "feel" is built into the system and electrical trim actuators change the relationship of the "feel" springs to the control stick. Flight control trim is available to the pilot around the three axes of the aircraft; however, the yaw damper switch must be on before rudder trim can be accomplished.

CONTROL STICK

2 The control stick is equipped with a standard stick grip, see Figure 1-8-1, incorporating a trim button, bomb-rocket button, trigger, nosewheel steering button, and pitch damper cutoff switch. During flight the nosewheel steering button is used to operate the reconnaissance cameras of A/R aircraft. The trim button provides aileron and horizontal stabilizer trim, which allows the pilot to reduce control stick forces to a minimum. Control stick movement beyond trim settings is possible in flight by exerting pressure in the desired direction.

PITCH TRIM INDICATOR

3 The pitch trim indicator on the instrument panel, see Figure 1-1-5, indicates the trimmed position of the horizontal stabilizer. Full trim indication ranges from 5 increments nose-up trim to 5 increments nose-down trim.

CONTROL STICK

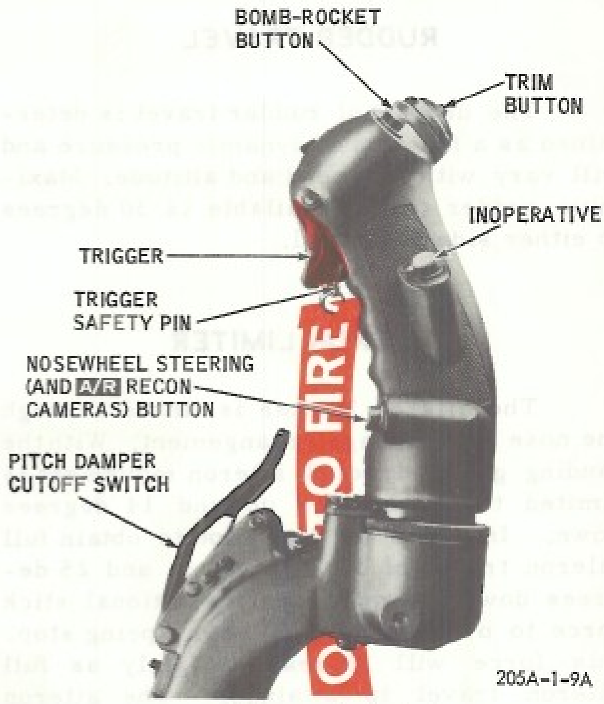


Figure 1-8-1

RUDDER TRIM KNOB

4 A rudder trim knob on the left console, see Figures 1-1-7 and 1-1-8, provides rudder trim only when the yaw damper switch of the stability augmenter system is at YAW. The trim knob should be manually set to neutral position before takeoff.

RUDDER PEDAL ADJUSTMENT T-HANDLE

5 A rudder pedal adjustment T-handle on the pedestal, see Figure 1-1-9, provides for adjustment of the rudder pedals. To adjust pedals, pull the T-handle out, move rudder pedals to the position desired, and manually return handle to the stowed position to lock the pedals in position.

CAUTION

Allowing the handle to snap back may trip circuit breakers on the pedestal and cause the cable to kink and wear excessively.

RUDDER TRAVEL

6 The degree of rudder travel is determined as a function of dynamic pressure and will vary with airspeed and altitude. Maximum rudder travel available is 30 degrees to either side of neutral.

AILERON LIMITER

7 The aileron limiter is linked through the nose gear linkage arrangement. With the landing gear retracted, aileron movement is limited to 18 degrees up and 14 degrees down. In order for the pilot to obtain full aileron travel of 35 degrees up and 25 degrees down, he must apply additional stick force to overcome the aileron spring stop; this force will increase slightly as full aileron travel is attained. The aileron limiter is disconnected when the landing gear is in the extended position, allowing full aileron travel.

STABILITY AUGMENTER SYSTEM

8 The stability augmenter system positions the vertical and horizontal control surfaces to automatically damp out pitch and yaw oscillations and also provides rudder trim. The central air data computer senses the aircraft's altitude and airspeed and determines the amount of control surface movement required. The aircraft can be safely flown without augmentation; however, augmentation improves handling characteristics and may be desirable for particular missions. The system is controlled by two damper switches on the left console, see Figures 1-1-7 and 1-1-8. The switches are spring-loaded to the OFF position, are held electromagnetically in the PITCH and YAW positions, and will disengage automatically in case of an electronic or power failure. The system may be disengaged by placing the applicable damper switch to the OFF position. Actuating a cutoff switch on the control stick will disengage the pitch damper only. The system may be reengaged in flight provided the flight limitations in Part 4, Section 1, are observed.

SECTION 9
WING FLAP SYSTEM

GENERAL

1 The wing flap system consists of trailing edge and leading edge flaps, electrically powered and operated by a three-position flap lever. Full flap extension or retraction (leading and trailing edge flaps) takes 10 to 17 seconds. Two ac motors operate each set of flaps through gear reduction units and actuators. The two trailing edge flaps are mechanically interconnected by a flexible rotary shaft to allow flap operation should one motor or actuator fail. The two leading edge flaps are mechanically interconnected in the same manner. The trailing edge flaps are mechanically interconnected to the horizontal stabilizer operating mechanism to minimize trim change automatically when the flaps are operated.

WING FLAP LEVER AND FLAP POSITION INDICATOR

2 The wing flap lever, see Figures 1-1-7 and 1-1-8, is a three-position lever controlling the electrical operation of the leading and trailing edge flaps. The placarded positions are UP, LE, and FULL. These positions are visually displayed to the pilot by the flap position indicator on the instrument panel, see Figure 1-1-5. With the lever in the LE position, the leading edge flaps are extended and the trailing edge flaps remain retracted. Placing the lever in the FULL position extends both leading and trailing edge flaps to full extension. A "barber pole" will appear on the indicator when aircraft electrical power is off, while flaps are being positioned, or if flap position does not agree with position of flap lever. Due to limit switches on the right main landing gear strut, leading edge flaps will not fully retract with weight on the gear; therefore, a "barber pole" indication on the flap position indicator is normal with flap lever at UP when the aircraft is on the ground.

SECTION 10
SPEED BRAKE SYSTEM

GENERAL

1 An electrically-controlled, hydraulically-actuated speed brake is located in the lower centre fuselage. Hydraulic power for the system is supplied from the utility system. The speed brake is the variable position type, requiring approximately 4 seconds to open and 3 seconds to close. At high speeds, the speed brake may not fully extend but as speed decreases will move out to fully extended position. The speed brake and horizontal stabilizer are mechanically interconnected to minimize trim change when the speed brake is operated.

SPEED BRAKE SWITCH

2 A three-position speed brake switch on the right throttle controls the operation of the speed brake, see Figure 1-2-3. Positioning the switch aft opens the speed brake (out), and moving the switch forward closes the speed brake (in). The centre (off) position neutralizes hydraulic pressure. Intermediate positions of the speed brake can be obtained by short intermittent actuation of the speed brake switch. The switch should be left in the forward position when the speed brake is closed to prevent speed brake from creeping open during flight. For the open and intermediate speed brake positions, the switch should be returned to the centre position after the speed brake has been positioned.

SECTION 11

LANDING GEAR SYSTEM

GENERAL

1 The tricycle landing gear incorporates a single wheel on each air-oil shock strut and fairing doors to enclose each gear well. The system is electrically controlled, and hydraulically powered by the utility hydraulic system. Electrical sequencing of gear and doors provides optimum low drag from takeoff to landing and is automatic during gear operation; doors unlock and open to downlock, gear unlocks and extends or retracts to downlock or uplock, doors unlock and return to uplock. The two-position nose shock strut may be extended 10 inches by the pilot to improve takeoff performance. Gear retraction time with the nose shock strut at normal extension (de-hiked) is 6 seconds; with the strut extended 10 inches (hiked), 9 seconds. Gear extension time is 6 seconds. Associated systems of the landing gear are nosewheel steering, wheel brakes, drag chute, and arresting hook. A mechanical landing gear alternate release system is provided for gear extension in the event of a malfunction in the utility hydraulic or electrical systems. With the exception of the nosewheel strut control switch, cockpit landing gear controls, indicator lights, warning lights and audible warning are conventional for high-performance aircraft.

LANDING GEAR CONTROLS

NOSEWHEEL STRUT CONTROL SWITCH

2 The nosewheel strut control switch placarded NOSE STRUT is located outboard of the throttle quadrant on the left console, see Figures 1-1-7 and 1-1-8. The switch has two placarded positions, EXTEND and RETRACT. When the switch is placed in the "solenoid-held" EXTEND position, hydraulic fluid is routed to the outside chamber of the nose gear strut to increase strut extension 10 inches. Takeoff angle-of-attack is increased approximately 3 degrees to provide a shorter takeoff run, and a more desirable attitude with forward centre of gravity configurations particularly. After takeoff, when the landing gear lever is selected LG UP, the nosewheel strut control switch will move automatically to RETRACT, fluid from the outer chamber of the strut will bleed back to the hydraulic system, and the strut will collapse (de-hike) to enter the wheel well. The strut will not extend (hike) when the weight of the aircraft is off the gear.

LANDING GEAR LEVER

3 The landing gear lever is on the lower left side of the instrument panel, see Figure 1-1-5. The lever has two positions, LG UP and LG DOWN, and can be moved from LG DOWN to LG UP when dc bus power is available and the landing gear struts are fully extended or when the landing gear lever downlock override button is pressed and held. When the weight of the aircraft is on the struts and the lever is in the down position, a locking solenoid is deenergized to permit engagement of a mechanical latch and prevent lever movement from LG DOWN. The override button energizes the solenoid to withdraw the latch. The same function is performed by a circuit through the weight switches when the shock struts extend at takeoff. The normal cycle may be interrupted and reversed at any time by placing the gear lever to the other position.

LANDING GEAR WARNING LIGHT AND AUDIBLE SIGNAL

4 Following gear selection, a red warning light in the wheel-shaped knob of the landing gear lever illuminates until sequencing is complete, refer to Paragraph 1, preceding. The audible warning signal (beeper), and the red light indicate an unsafe gear condition in flight. During flight, the altitude and airspeed sensing switches in the central air data computer will cause the red warning light to come on and the audible signal to sound at any time the landing gear is not extended and locked, aircraft below 10,000 ± 250 feet, airspeed less than 210 ± 10 KIAS, and either left or right throttle is reduced below 96% rpm. The warning signal may be silenced by pressing the landing gear warning signal silence button, see Figure 1-1-5. If the warning signal sounds

because of an unsafe gear (gear down and any downlock not fully engaged or gear up and not properly locked) and is silenced, the signal will not sound again until the gear is recycled. If the warning signal is due to wheels up configuration/flight profile and is silenced, the signal will sound again if throttles are advanced and again retarded, provided the same conditions exist.

4A A gear doors switch marked OPEN and NORMAL, located in the right main gear well is for ground maintenance use only. When the switch is at OPEN and electrical power is applied to the aircraft, the warning light will illuminate and the audible warning will sound.

LANDING GEAR POSITION INDICATOR LIGHTS

5 The landing gear position indicator lights are above the landing gear lever, see Figure 1-1-5. Illumination of all three lights indicates safe extension and locking of the landing gear. When the landing gear is not in the down and locked position, the green lights are out. For gear extension, each green light will come on as the respective gear reaches its down and locked position. If the doors do not lock up following a "three green" indication, the warning light in the landing gear lever will remain on but the audible warning will not sound. A safe landing may be carried out with this condition.

LANDING GEAR LEVER DOWNLOCK OVERRIDE BUTTON

6 A landing gear lever downlock override button to the right of the lever, see Figure 1-1-5, enables the pilot to raise the landing gear lever to LG UP position while the aircraft is on the ground and the struts are compressed. If the locking solenoid fails to release the landing gear lever from the LG DOWN position when the struts are extended, as after takeoff, the button can be pressed and held to allow the lever to be placed in the LG UP position.

LANDING GEAR ALTERNATE RELEASE

7 A landing gear alternate release handle to the left of the left vertical control panel, see Figure 1-1-6, permits gear extension with landing gear lever up or down should the normal extension system fail. To release the landing gear, slowly and steadily pull the handle (approximately 10 inches) and hold in the fully extended position until all three gears are unlocked. Pulling the handle deenergizes the normal landing gear hydraulic and electrical systems and releases the main gear uplocks, main gear inboard door locks, nose gear, and nose gear forward door. Gear doors open and landing gears extend, assisted by gravity and airloads. The nose gear is forced over centre by spring tension. Spring bungees on each landing gear strut assist landing gears into a down and locked over-centre position. As each landing gear is extended into the down and locked position, the respective green indicator on the control panel will come on. Safe gear down indication may take approximately 35 seconds. With the landing gear lever in the LG DOWN position, the red warning light will go out when all gears are indicated as down and locked. If alternate release of the gear is made with the landing gear lever at LG UP, the red warning light will not extinguish. The landing gear doors will remain open in either case.

NOSEWHEEL STEERING SYSTEM

GENERAL

8 The nosewheel steering system provides directional control and shimmy damping during taxiing, takeoff, and landing. The system uses utility hydraulic system pressure electrically controlled by the nosewheel steering button on the stick grip, see Figure 1-8-1. Nosewheel steering is accomplished by pressing and holding the nosewheel steering button and moving the rudder pedals in the direction desired. If the nosewheel position does not correspond to the position of the rudder pedals when the steering button is pressed, the nosewheel will turn to correspond to rudder pedal position. The nosewheel steering function is deactivated when the weight of the aircraft is removed from the nosewheel strut.

CAUTION

Nosewheel steering is not available while the two-position nose strut is in transit from the extended to the retracted position.

NOSEWHEEL STEERING BUTTON

9 The nosewheel steering button electrically engages and disengages the nosewheel steering system. Pressing and holding the button energizes and opens a normally closed hydraulic shutoff valve, allowing utility hydraulic system pressure into the steering actuator.

WHEEL BRAKE SYSTEM

10 Each main gear wheel is provided with a hydraulically operated multiple-disk power brake assembly. Brakes are operated by conventional toe-type brake pedals (rudder pedals). When the pedals are depressed, the brake control valve meters hydraulic pressure, proportional to pedal depression, to the brakes. Proper brake disk operating clearances are automatically provided when the brake pedals are momentarily pressed hard while engines are running and the aircraft not rolling. Should the utility system fail, the brake control valve acts as a brake master cylinder, and brake pressure is dependent on the amount of foot pressure applied to the brake pedal.

DRAG CHUTE SYSTEM

GENERAL

11 To aid in reducing landing roll distance after touchdown, a 15-foot ring-slot drag chute is packed in a deployment bag and stowed in an air-cooled compartment at the base of the rudder. The drag chute, while in the compartment, is not connected to the aircraft until the drag chute T-handle is pulled. A riser connects the drag chute to a release pin in the compartment. The compartment is equipped with a door, a door latch, and a drag chute release mechanism controlled by a T-handle in the cockpit.

DRAG CHUTE T-HANDLE

12 The drag chute T-handle on the instrument panel, see Figure 1-1-5, is mechanically connected to the drag chute release mechanism of the drag chute compartment. To deploy the chute, the handle is pulled straight out (without turning) to the first stop (approximately 3-1/4 inches). Initial movement of the handle latches the drag chute release mechanism, locking the drag chute to the aircraft. Further movement of the handle unlocks the compartment door latch, allowing the spring-loaded pilot chute to deploy and withdraw the drag chute into the airstream. The handle will lock in the deployed position. The drag chute can be jettisoned by turning the T-handle 90 degrees in a clockwise direction and pulling it out to the next stop (approximately an additional 3-1/4 inches). The handle is under spring tension during the final pull to jettison chute. When released, the handle will retract to the first stop.

CAUTION

To avoid inadvertent jettisoning of the drag chute, do not rotate the handle when deploying the drag chute.

ARRESTING HOOK SYSTEM

GENERAL

13 The arresting hook attached to the aircraft structure on the aft fuselage centreline is provided for runway barrier engagement. The hook is held in the up position by a lock assembly. A ground safety pin is provided to prevent inadvertent actuation on the ground and must be removed before flight, see Figure 1-11-1. For hook extension, the uplock is released electrically by pushing the arresting hook pushbutton. The hook then extends by gravity and torsion bar spring force. The torsion bar maintains a positive downward force on the hook while a self-contained hydraulic damping unit acts as a snubber to minimize hook bounce.

ARRESTING HOOK RELEASE BUTTON

14 The arresting hook release pushbutton placarded ARREST HOOK is on the lower left portion of the instrument panel, see Figure 1-1-5. Depression of the button releases the hook uplock and illuminates an integral button light. The hook cannot be retracted by the pilot but must be stowed manually to the locked position when the button light will extinguish.

CAUTION

- With the arresting hook ground safety pin installed, pushing the arresting hook release button may cause the locking solenoid to burn out, making the arresting hook system inoperative.
- Nosewheel steering is not available while the two-position nose strut is in transit from the extended to the retracted position.

NOTE

With landing gear down and nosewheel strut in EXTEND position, release of arresting hook will allow strut to return to RETRACT (nonelevated) position to permit more favourable barrier engagement on aborted takeoff.

GROUND SAFETY PINS

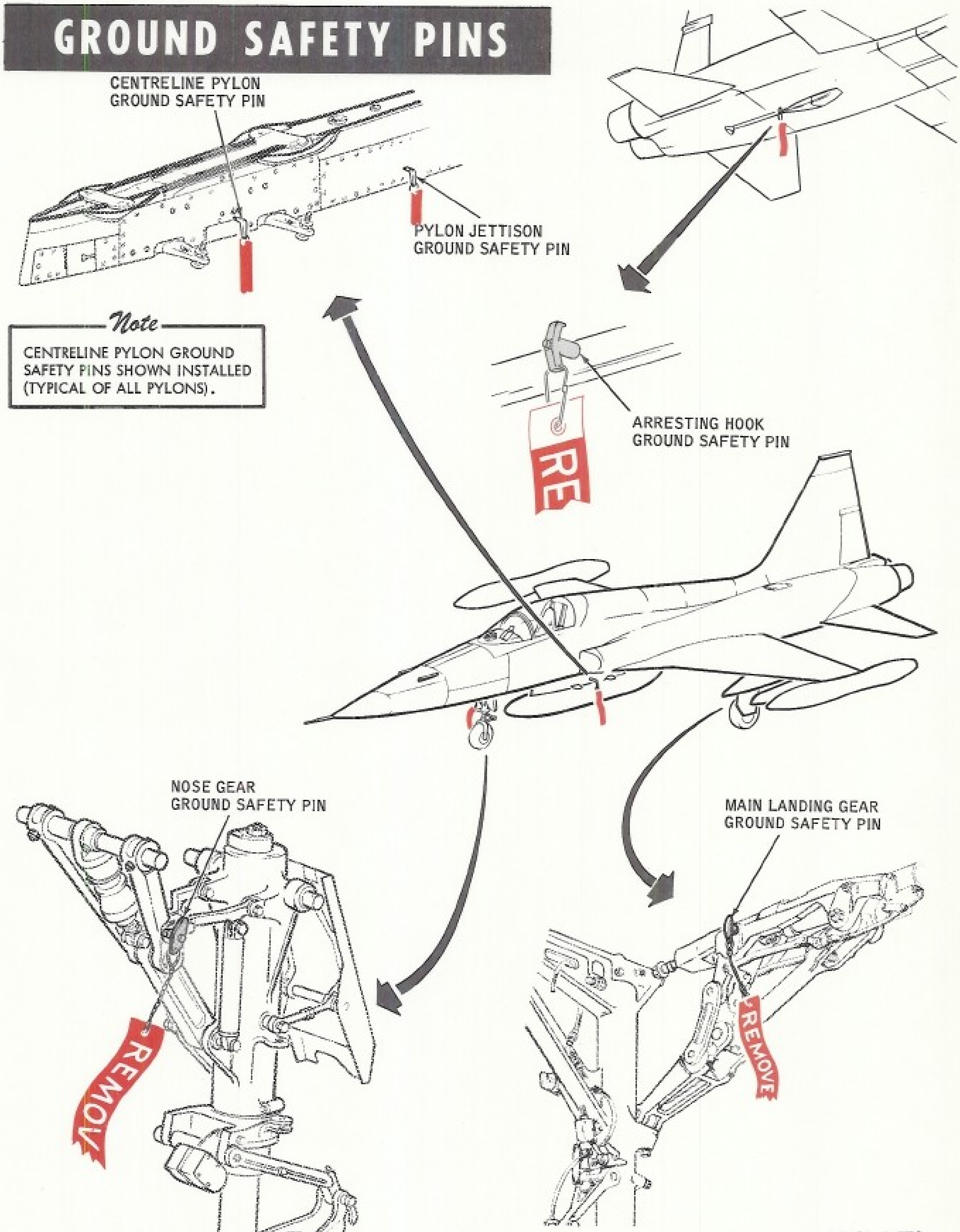


Figure 1-11-1

SECTION 12

INSTRUMENTS

GENERAL

1 The instruments covered herein are flight instruments which are electrically powered by the ac and/or dc electrical and pitot-static systems.

NOTE

For information regarding instruments that are an integral part of a particular system, refer to applicable Sections in this Part and in Part 2.

PITOT PRESSURE AND STATIC SYSTEMS

2 The pitot pressure and static systems supply both impact and static air pressure to the CADC, airspeed/mach indicator, and the airspeed compensator of the stability augmenter system. The altimeter and vertical velocity indicator receive only static pressure from the system.

PITOT-STATIC BOOM

3 A boom on the aircraft nose, see Figure 1-1-1, supplies impact pressure directly from the boom head into the aircraft pitot system. Static pressures are supplied through orifices on each side of the boom centreline within 5 inches of the boom tip.

AIRSPEED AND MACH NUMBER INDICATOR (AVU-9/A)

4 The AVU-9/A airspeed and mach number indicator, see Figure 1-12-1, on the instrument panel simultaneously displays airspeed in knots and in mach number. The

AIRSPEED/MACH INDICATOR (AVU-9/A)

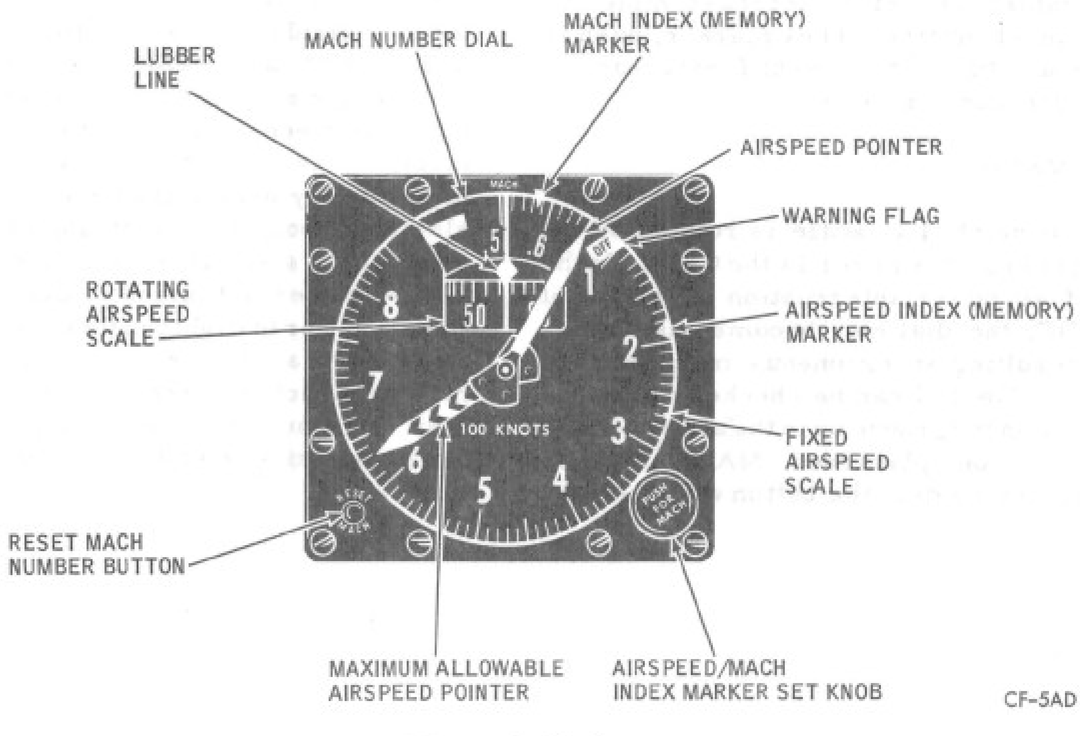


Figure 1-12-1

labelled markings on the dial face (fixed airspeed scale) of the instrument represent 100 to 850 knots indicated airspeed in 10-knot increments. The rotating airspeed scale within the dial face displays values less than 100 knots, is geared to the airspeed pointer, and indicates the airspeed in 2-knot increments with labelled markings from 0 to 90 knots. The mach number subdial is a servo repeater type, actuated by the central air data computer (CADC). The mach number is displayed in 0.02 increments and labelled in tenths from .5 to 2.2 mach. The yellow and black striped maximum allowable airspeed pointer moves with altitude to continuously show the maximum allowable airspeed for the aircraft. An OFF flag will appear in the event of ac power failure or failure of the CADC.

AIRSPEED AND MACH NUMBER SETTING INDICES

5 Two triangular index markers, one for the fixed airspeed scale and the other for the mach number scale, are provided to assist the pilot in setting airspeed reference (memory) markers. To set the indices, a single dual-function knob labelled PUSH FOR MACH is provided on the lower right corner of the indicator. To set the fixed airspeed scale index marker, turn the knob to the right to increase value; turn left to decrease value. To set the mach number index marker, push in the knob and turn either right for increase or left for decrease of value.

RESET MACH

6 The mach dial scale is rotated by the mach synchrotransmitter in the CADC. In the event of ac power interruption or failure of the CADC, the dial may become unsynchronized, resulting in erroneous mach number readout. The dial can be checked or resynchronized on the ground or in the air by pressing the button placarded MACH RESET. Pressing and holding the button will cause the dial to slew to 1.2 mach. When the reset button is released, the mach number subdial will then slew to the correct number (white arc below 0.5 mach if on ground and static).

COUNTER-DRUM-POINTER ALTIMETER (AAU-19/A TYPE)

7 A servo/pneumatic counter-drum-pointer (AAU-19/A type) altimeter, see Figure 1-12-2, on the instrument panel consists of a precision pressure altimeter combined with a servomotor and controlled by signal inputs from the CADC. The altimeter provides a continuous display of corrected altitude information as transmitted from the computer. Direct readout of altitude is accomplished by observing the numerals in the readout windows on the face of the instrument. Below an altitude of 10,000 feet, a diagonal white-striped, low-altitude warning marker will appear in the extreme left window dial. A barometric pressure set knob is provided to insert the desired altimeter setting in inches Hg. In case of power interruption longer than 3 seconds duration or system failure, the word standby, placarded STBY, will appear in the upper left portion of the instrument face, indicating that the altimeter has automatically reverted to standby operation (barometric altimeter) and uncorrected altitude is displayed. A function switch, placarded STBY and RESET, is a spring-loaded self-centering switch used to control the automatic or mechanical operation of the instrument. The function switch must be momentarily held in the RESET position when electrical power is first applied to the aircraft after starting engines and in the event of power interruption and subsequent restoration of power in flight to allow automatic operation of the altimeter. Mechanical operation may be selected by the pilot, while the altimeter is in automatic operation, by momentarily selecting the STBY position of the function switch.

COUNTER-DRUM-POINTER ALTIMETER

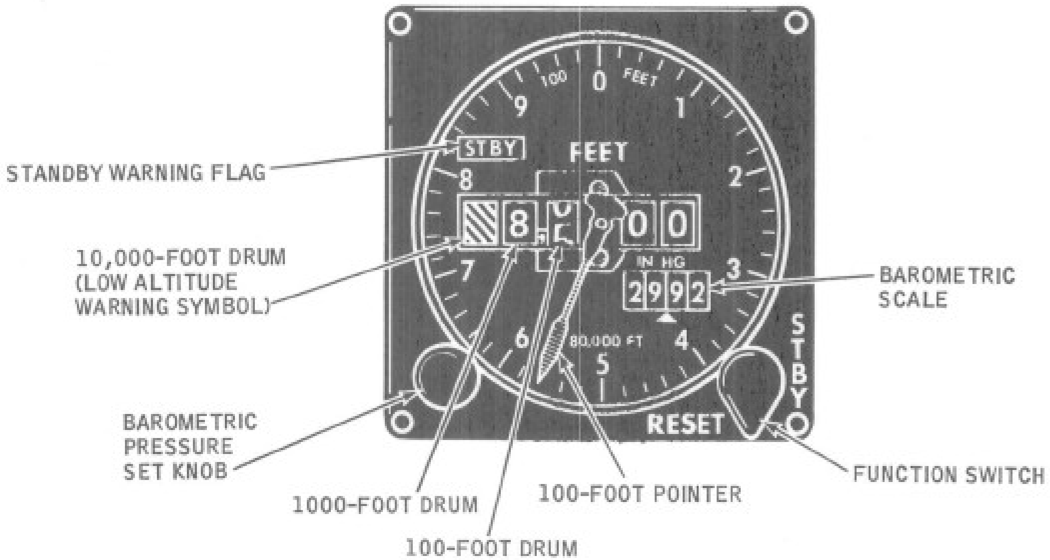


Figure 1-12-2

SECTION 13

CAUTION, WARNING AND INDICATOR LIGHT SYSTEM

GENERAL

1 Caution, warning, and indicator lights give visual indication of malfunction or the status of certain equipment and systems, and of failure or unsafe conditions in critical areas of the aircraft. The light systems consist of two red fire warning lights (one for each engine), gear unsafe red warning light, canopy unlocked red warning light, gear position indicator green lights, approach indexer lights, arresting hook light, master caution yellow light, and a caution light panel with 14 yellow caution lights for the individual systems. All caution, warning, and indicator lights operate on dc primary power in the bright mode and ac power in the dim mode. The dimming function is inoperable when the instrument lights are off.

MASTER CAUTION LIGHT

2 The master caution light on the instrument panel, see Figure 1-1-5, comes on and flashes continuously when a light on the console caution light panel comes on, and will go out when the console panel caution light goes out. However, the master caution light may be extinguished by pressing the master caution light. This press-to-reset feature rearms the light to provide warning of subsequent malfunctions.

CAUTION LIGHT PANEL

3 The caution light panel on the right console, see Figures 1-1-7 and 1-1-8, contains the following placarded caution lights:

LEFT GENERATOR	RIGHT GENERATOR
UTILITY HYDRAULIC	FLIGHT HYDRAULIC
EXT TANKS EMPTY	FUEL LOW
ENG ANTI-ICE ON	OXYGEN
LEFT FUEL PRESS	RIGHT FUEL PRESS
DUCT/WSHLD OVERHEAT	AIR DATA COMPUTER
SPARE	SPARE

Each caution light will remain on as long as the malfunction or system status is uncorrected and will not go out if the master caution light is rearmed.

WARNING TEST AND BRIGHT/DIM SWITCHES

4 Two switches placarded WARNING on the lighting control panel portion of the right console, see Figures 1-1-7 and 1-1-8, control the illumination of the caution, warning, and indicator light system. Holding the right-hand switch in the TEST position will activate all caution, warning, and indicator lights as well as test the landing gear audible warning signal, and the fire detector unit for each engine. The left-hand switch selects either BRIGHT (28 VDC) or DIM (14 VAC) mode of operation.

ENGINE FIRE WARNING LIGHTS

5 Two red fire warning lights, one for each engine, see Figure 1-1-5, are provided to warn of an overheat or fire condition in either engine compartment. When the fire detection system detects an overheat condition or fire, the warning light for the respective engine will illuminate. This light will remain illuminated until the condition is corrected. Should the fire or overheat condition recur, the light will again illuminate. Heat-resistant materials used in the engine bay plus protective blanketing and cooling airflow minimize the danger to adjacent structure in case of engine fire.

NOTE

The master caution light will not illuminate when engine fire warning lights are illuminated.

SECTION 14

CANOPY SYSTEM

GENERAL

1 The jettisonable, clamshell-type canopy is manually operated, counterbalanced throughout travel limits, and may be manually locked or unlocked by an internal control handle or an external control handle. Canopy drive mechanism is protected against excessive loads by a hydraulic canopy damper, which also restricts canopy opening and closing speeds during operation. An inflatable pressurization seal installed on the canopy is inflated only when the canopy is locked, the cockpit pressurization switch is in the CABIN PRESS position, and an engine is operating.

CAUTION

- The canopy should always be assisted through opening and closing cycle with hand pressure applied at the forward frame.
- Either the canopy external or internal handle is to be used to actuate the locking mechanism only. If the handle is used to raise or lower the canopy, damage may result to the mechanism.
- Stow external operating handle and close access door immediately after opening or closing canopy using external handle.

CANOPY WARNING LIGHT

2 A red canopy warning light, located on the instrument panel, illuminates and displays the legend CANOPY when the canopy is unlocked with ac or dc power available.

CANOPY CONTROLS

3 For location of canopy controls, see Figure 1-14-1. Canopy jettison controls are described in Section 15.

CANOPY CONTROLS

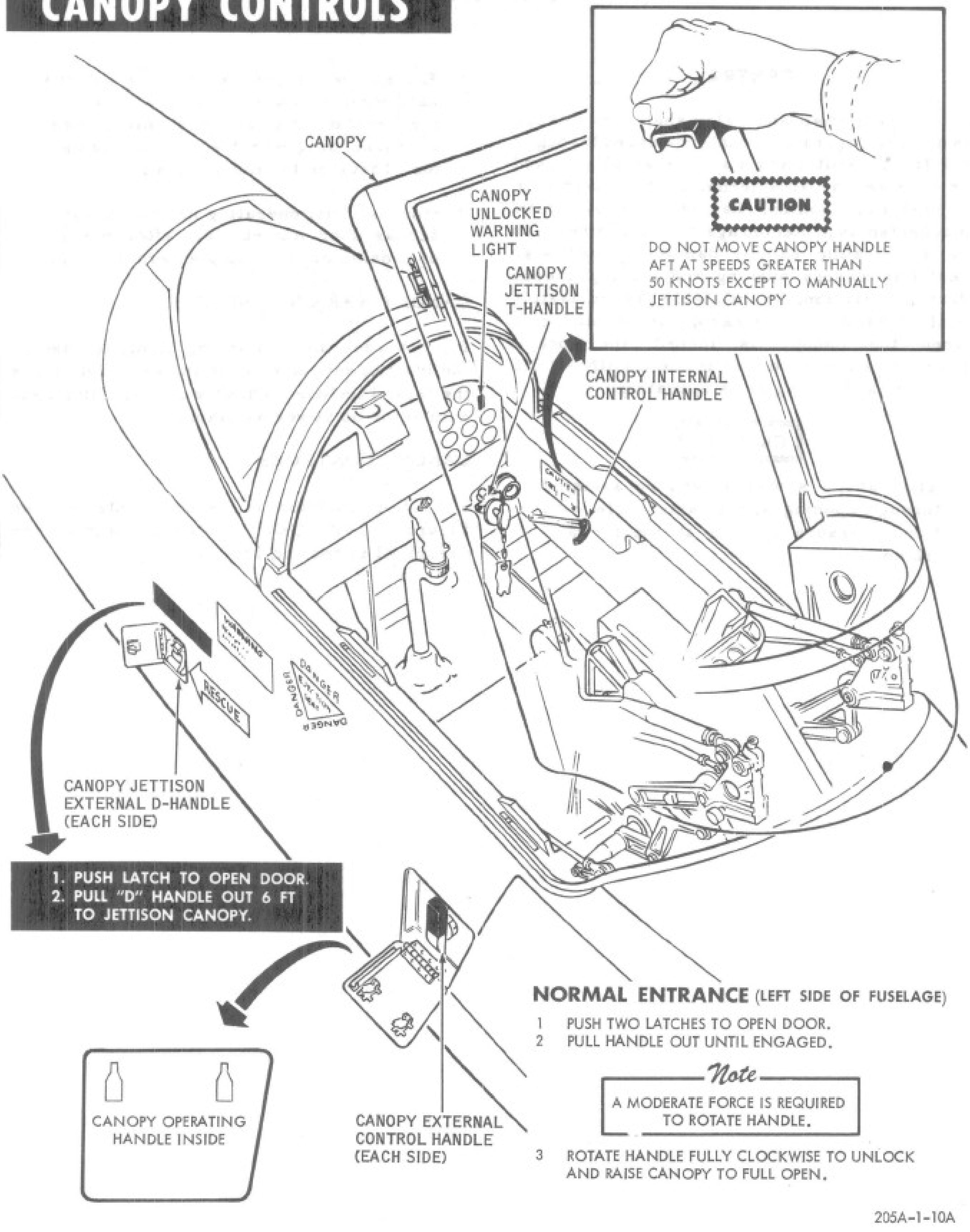


Figure 1-14-1

SECTION 15

ESCAPE SYSTEM

GENERAL

1 The escape system consists of jettisonable canopy, an ejection seat with automatic personal disconnect, automatic opening lap belt, man/seat separator, and the automatic barostat parachute with arming lanyard. Seat ejection automatically initiates UHF and IFF/SIF emergency signals with ac and dc power available.

CANOPY JETTISON

2 The canopy can be independently jettisoned from within the cockpit by pulling the T-handle on the right vertical control panel, see Figure 1-1-6. A safety pin, installed at the T-handle, prevents jettison during ground operations. A spring clip latch, which acts as an additional safety device, blocks inadvertent actuation of the jettison T-handle when the safety pin is removed for flight. External D-handles, see Figure 1-14-1, are used primarily for rescue purposes. The canopy jettison system will function with the canopy open or closed and locked.

CANOPY JETTISON AND SEAT EJECTION

3 Jettison of the canopy and seat ejection are initiated by raising either handgrip, see Figure 1-15-1, on the mechanically interconnected seat legbraces. This action exposes the catapult firing triggers and automatically locks the shoulder harness. Squeezing either trigger jettisons the canopy, and seat ejection occurs 0.3 seconds later. Accompanying this action, the seat adjuster power cable and personal connects are disconnected, the calfguard is lowered into position, and the automatic lap belt initiator (incorporating 1-second time delay) is actuated. After the seat has left the cockpit, the lap belt initiator fires, to open the lap belt and simultaneously actuate the worm-gear wind-up reel of the man/seat separator. The open lap belt releases the shoulder harness straps. The man/seat separator strap, routed down the back of the seat and under the survival kit, is drawn taut to separate the pilot from the seat. As the occupant is separated from the seat, the parachute arming lanyard, attached to the right lap belt, is withdrawn to allow automatic parachute deployment at a predetermined pressure level (altitude).

EJECTION SEAT

4 The cockpit is equipped with a rocket catapult ejection seat, see Figure 1-15-1, consisting of a seat adjusting unit and control switch, shoulder harness, inertial reel and locking lever, lap belt, calfguard, headrest, canopy piercer, legbraces, chaff dispenser, catapult firing triggers, jettison initiator, survival kit case, man/seat separator system, and personal lead quick-disconnect block assembly. The seat adjustment switch on the right leg-brace controls the electrical actuator unit, allowing seat adjustment of 5 inches from the fully lowered to the fully raised position. The seat will eject through the canopy if canopy jettison malfunctions.

WARNING

Do not disconnect the retention strap from the oxygen hose. The strap gives the straight downward pull required to disconnect the oxygen hose during man/seat separation.

LEGBRACES

5 Mechanically interconnected legbraces, see Figure 1-15-1, terminating in handgrips, are attached to the seat. Upward travel of either handgrip through the first 10 degrees releases the downlock on both legbraces. The legbraces are held in the raised position by an uplock. If legbraces have been raised, they cannot be lowered to the stowed position without removing the seat from the aircraft.

EJECTION SEAT

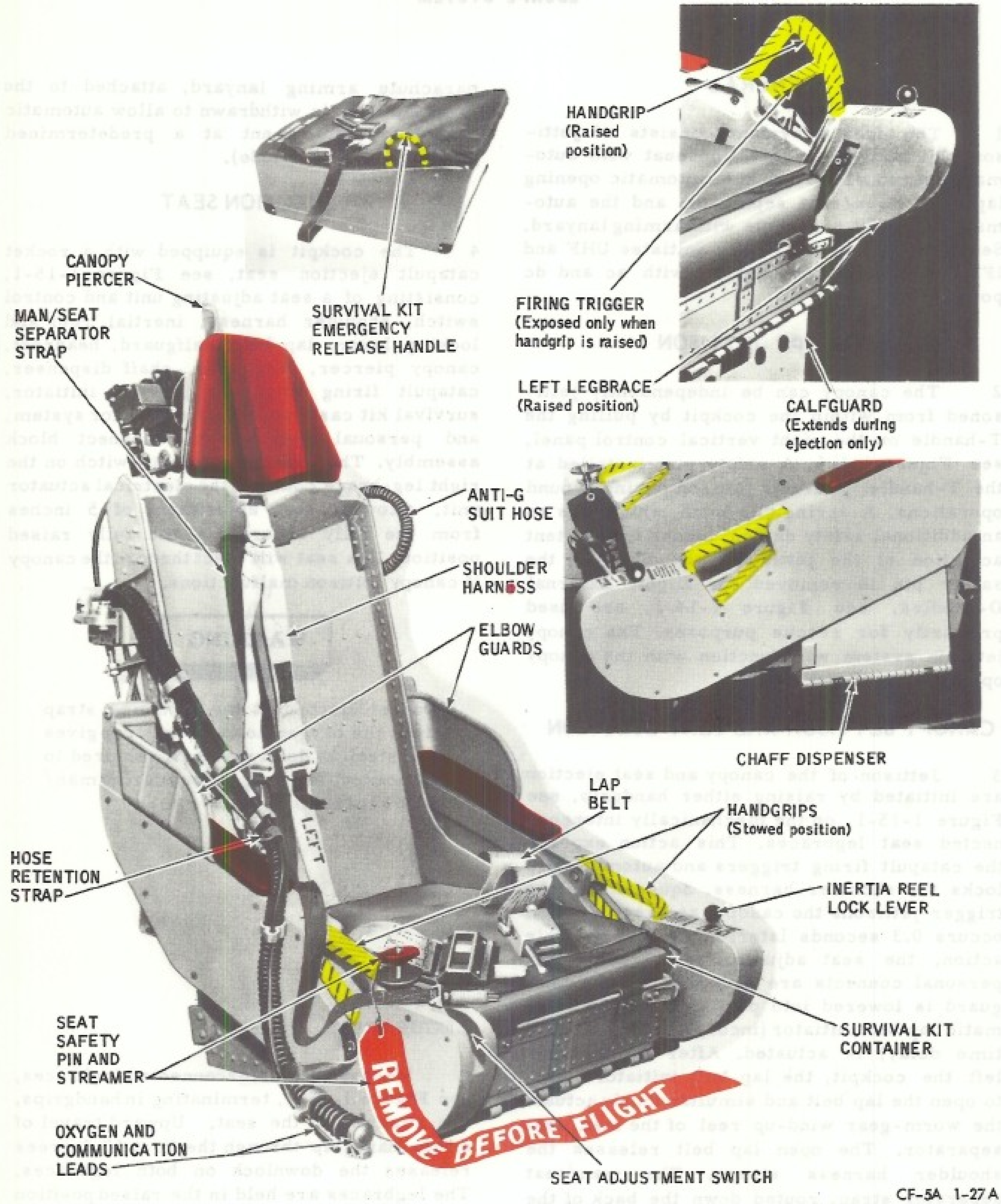


Figure 1-15-1

CATAPULT FIRING TRIGGERS

6 The catapult firing triggers, see Figure 1-15-1, are locked in the stowed position when the legbraces are down and are exposed when the legbraces are raised.

EJECTION SEAT AND CANOPY JETTISON T-HANDLE SAFETY PINS

7 The ejection seat safety pin, see Figure 1-15-1, when inserted, holds the right leg-brace down, preventing inadvertent seat ejection. One streamer is attached between the ejection seat safety pin and the canopy jettison T-handle safety pin.

AUTOMATIC-OPENING

8 DELETED included when

CHAFF DISPENSER

9 A chaff dispenser is installed on the left legbrace of the ejection seat for radar identification of the aircrew during bailout. Automatically actuated during seat-crew separation, the chaff is diffused into the air surrounding the crewmember, facilitating radar tracking. The chaff is retained in the dispenser by a spring-loaded door, which is held in the closed position by a cable acting as a latch pin. When the man/seat separator is actuated during ejection sequence, the cable is withdrawn and the door is opened to release the chaff.

LOW-ALTITUDE ESCAPE EQUIPMENT

10 The automatic features of the escape system are augmented by a 1.0- (-0, +1/4) second automatic opening barostat equipped parachute (below 15,000 feet altitude above sea level) for low-altitude ejection. The parachute arming lanyard is connected to an MK-10A barometric release mechanism in the parachute pack. When the arming lanyard cable is withdrawn, activating the barostat, the release mechanism pulls the parachute pack retaining pins. This action opens the parachute pack and the spring-loaded pilot chute is released and, in turn, deploys the main parachute. Refer to Part 3, Section 2, for low-altitude ejection airspeed and altitude parameters.

SURVIVAL KIT

11 The survival kit, see Figure 1-15-1, is attached to the parachute harness by web straps, and the survival kit lanyard is attached to either the life jacket (if worn) or the upper parachute harness, see Figure 2-1-2. The contents of the kit can be varied, dependent upon local command desires and environmental requirements, and may include a life raft. In the event of ejection from the aircraft, the survival kit is deployed by pulling the alternate black and yellow striped emergency release handle. The kit will then release from the parachute harness lower attachments and will remain suspended from the lanyard approximately 20 feet below the pilot.

SECTION 16

AIR-CONDITIONING AND PRESSURIZATION SYSTEM

GENERAL

1 Functions of the aircraft air-conditioning unit are to control cockpit air temperature, provide canopy and windshield defogging, pressurize hydraulic reservoirs and external fuel tanks, inflate canopy and windshield seals, operate anti-G suit, and cool electrical equipment. In addition, the air-conditioning unit provides [redacted] photoreconnaissance camera compartment cooling and defog (refer to Section 21 for system description). Air from the eighth stage of the compressor section of each engine is used to operate all the above functions; however, one engine is capable of operating the complete system. Check valves within the system prevent air bleedoff to an inoperative engine. Air is routed through a heat exchanger, cooling turbine, and water separator before entering the cockpit area. The degree of heating or cooling is regulated by the position of the temperature control valve. A pressure regulator automatically maintains the cockpit pressure differential schedule as illustrated in Figure 1-16-1. Cockpit pressure altitude is indicated on the cabin altitude indicator on the instrument panel, see Figure 1-16-2. A pressure safety valve is incorporated in the system to automatically protect the cockpit from excessively high or low pressures.

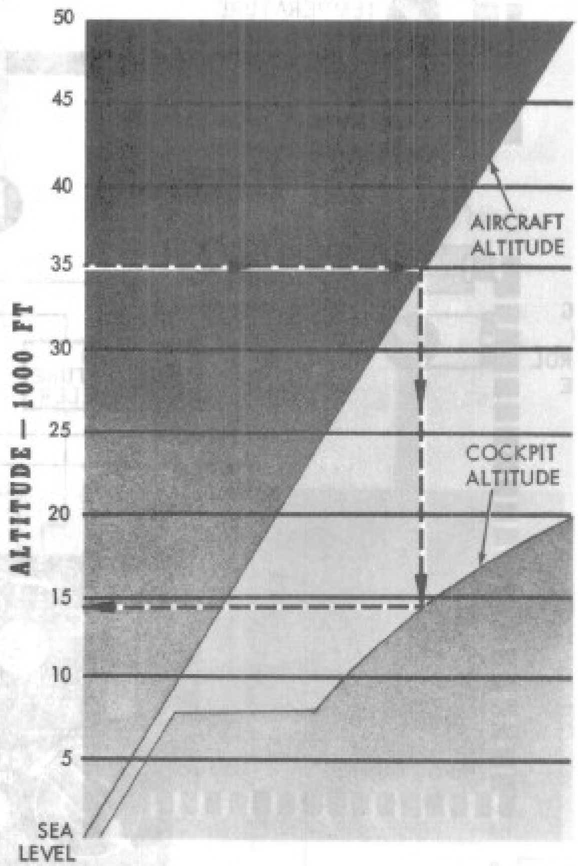
CAUTION

To prevent moisture from being blown onto the caution light panel when operating in a hot, humid environment, the cabin air outlet forward of the right console should be turned closed or the flow of air directed toward the extreme right side of the cockpit.

COCKPIT PRESSURIZATION SWITCH

2 The cockpit pressurization switch, placarded CABIN PRESS., on the right vertical control panel, see Figure 1-16-2, has two positions, RAM DUMP and CABIN PRESS. With the switch at CABIN PRESS., the cockpit is pressurized. Selection of RAM DUMP closes the system shutoff valve, opens the pressure safety valve, and opens the cabin ram air valve to allow ram air to enter the cockpit.

COCKPIT PRESSURIZATION SCHEDULE



EXAMPLE – REFER TO DASH LINE.

AIRCRAFT ALTITUDE OF 35,000 FEET EQUALS COCKPIT ALTITUDE OF APPROXIMATELY 14,500 FEET.

Figure 1-16-1 CF-5AD 1-73

COCKPIT AIR-CONDITIONING AND PRESSURIZATION SYSTEM

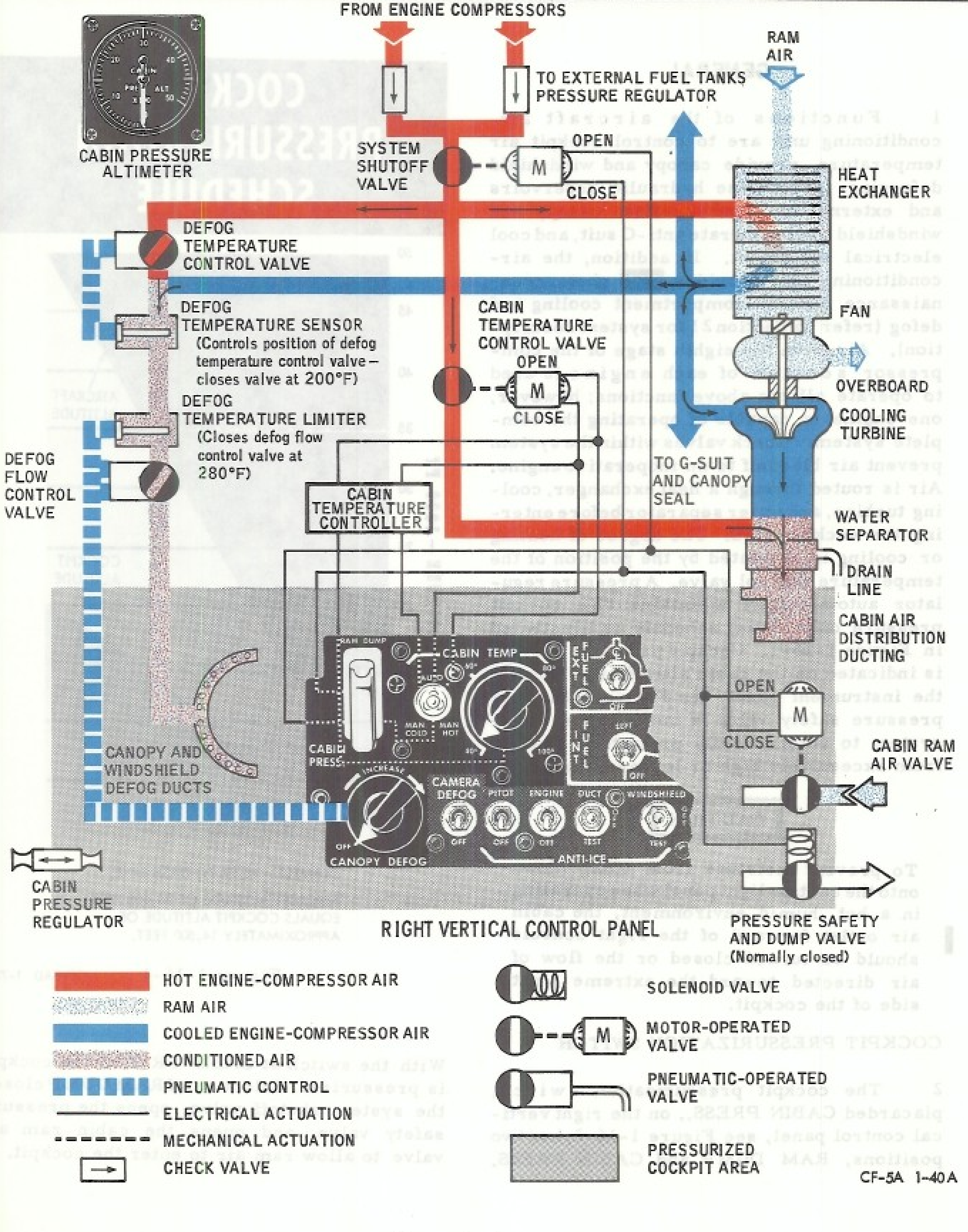


Figure 1-16-2

NOTE

With the switch in RAM DUMP position, pressurization to the external tanks is lost, making it impossible to transfer external fuel.

COCKPIT TEMPERATURE CONTROL

3 Cockpit temperature may be controlled automatically or manually. Placing the temperature control switch, see Figure 1-16-2, at AUTO and selecting the desired temperature setting with the temperature selector knob allows the air-conditioning unit to automatically mix the air to maintain the selected cockpit temperature. Manually positioning the switch to MAN. COLD or MAN. HOT allows the pilot to position the temperature control valve and control cockpit temperature should the automatic feature malfunction.

NOTE

When repositioning the temperature control switch between placarded positions, pause at intermediate OFF position for 1 second to allow relay to function.

CANOPY AND WINDSHIELD DEFOG

4 The canopy and windshield are defogged by directing air from the heat exchanger at 200° to 280°F through ducting to the canopy and windshield surfaces to prevent icing or fogging. The defog control knob on the right vertical control panel, see Figure 1-16-2, regulates the amount of airflow through the defog control valve. Defogging air temperature is independent of the temperature selected by the cabin temperature selector knob.

ANTI-G SUIT SYSTEM

5 The anti-G suit is supplied air pressure by the cockpit air-conditioning and pressurization system. Air is routed through an anti-G suit pressure-regulating valve to the anti-G suit attachment fitting. The flexible hose from the regulating valve to the attachment fitting passes through a quick-disconnect fitting on the upper left side of the ejection seat to allow automatic disconnect upon ejection.

ANTI-G SUIT VALVE

6 The anti-G suit pressure-regulating valve is outboard of the left ejection seat rail in the cockpit, see Figures 1-1-3 and 1-1-4. The valve regulates air pressure to the anti-G suit and permits automatic inflation of the suit when positive G is encountered. The valve operates automatically and begins to function at about 1.75-G, exerting an increasing pressure as the G-load is increased. When the acceleration decreases below the valve opening G-setting, the valve closes and the suit deflates. A test button on top of the valve permits checking valve operation while the engine is running. Depressing the button will test the valve by allowing pressure to inflate the G-suit bladder.

[illegible]

[illegible]

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SECTION 17

ANTI-ICING SYSTEMS

ENGINE ANTI-ICING SYSTEM

1 Bleed air from the eighth stage of the compressors is used as anti-ice control for the stationary inlet guide vanes, variable vanes, and the bullet nose fairing of each engine. System components include control switch, see Figure 1-17-1, ENGINE ANTI-ICE ON caution lights, and a solenoid-controlled, pneumatically operated shutoff valve, along with ducting, for each engine. During normal engine operation, engine anti-icing air is supplied regardless of switch selection until engines reach 60 to 65% rpm. At higher rpm, when the anti-ice switch is selected to ENGINE, the caution light will illuminate and air will be directed to the inlet areas. A 9% loss in military thrust, and a 6.5% loss in maximum thrust may be expected with engine anti-ice operating.

NOTE

- The master caution light on the instrument panel will come on simultaneously with the activation of the ENG ANTI-ICE ON caution light.
- Press the master caution light to rearm circuit for subsequent malfunction warning.

DUCT ANTI-ICING SYSTEM

2 The engine inlet duct anti-icing system consists of heating elements (blankets) in the lip of each engine air inlet duct. A switch within the ANTI-ICE portion of the right vertical control panel has three placarded positions: DUCT, OFF, and TEST, see Figure 1-17-1. Placing the switch in the DUCT position provides ac power to the heating elements of each engine air inlet duct lip to eliminate or prevent ice formation during flight in icing conditions. The spring-loaded TEST position is used for ground crew check purposes only. The left duct heating element is powered by the left generator, and the right duct heating element is powered by the right generator. If a generator or an engine fails during flight, the heating element of the failed side will be automatically powered by interlock transfer of ac power to the operating generator of the opposite side. Temperature sensors are incorporated in the system to detect an overheat condition. Illumination of the DUCT/WSHLD OVERHEAT caution light and the master caution light indicates that either the windshield anti-icing system, if on, has overheated (see following system description) or that either one or both of the blanket temperature limit sensors has reached approximately 200°F and one or both engine air inlet duct anti-ice blankets are overheated. If the DUCT/WSHLD OVERHEAT caution light comes on, place the duct anti-ice switch in the OFF position.

CAUTION

Prolonged use of system during ground operations can cause overheating.

NOTE

- Due to the combined systems caution light warning, the activation of the light may have been caused by an overheat condition in the windshield system.
- To isolate the system at fault, it will be necessary to place the adjacent windshield anti-ice switch to OFF (if on) and place each system switch on individually to ascertain which system caused light illumination.
- If duct anti-icing system caused light illumination, place switch at OFF.

ANTI-ICE SYSTEMS

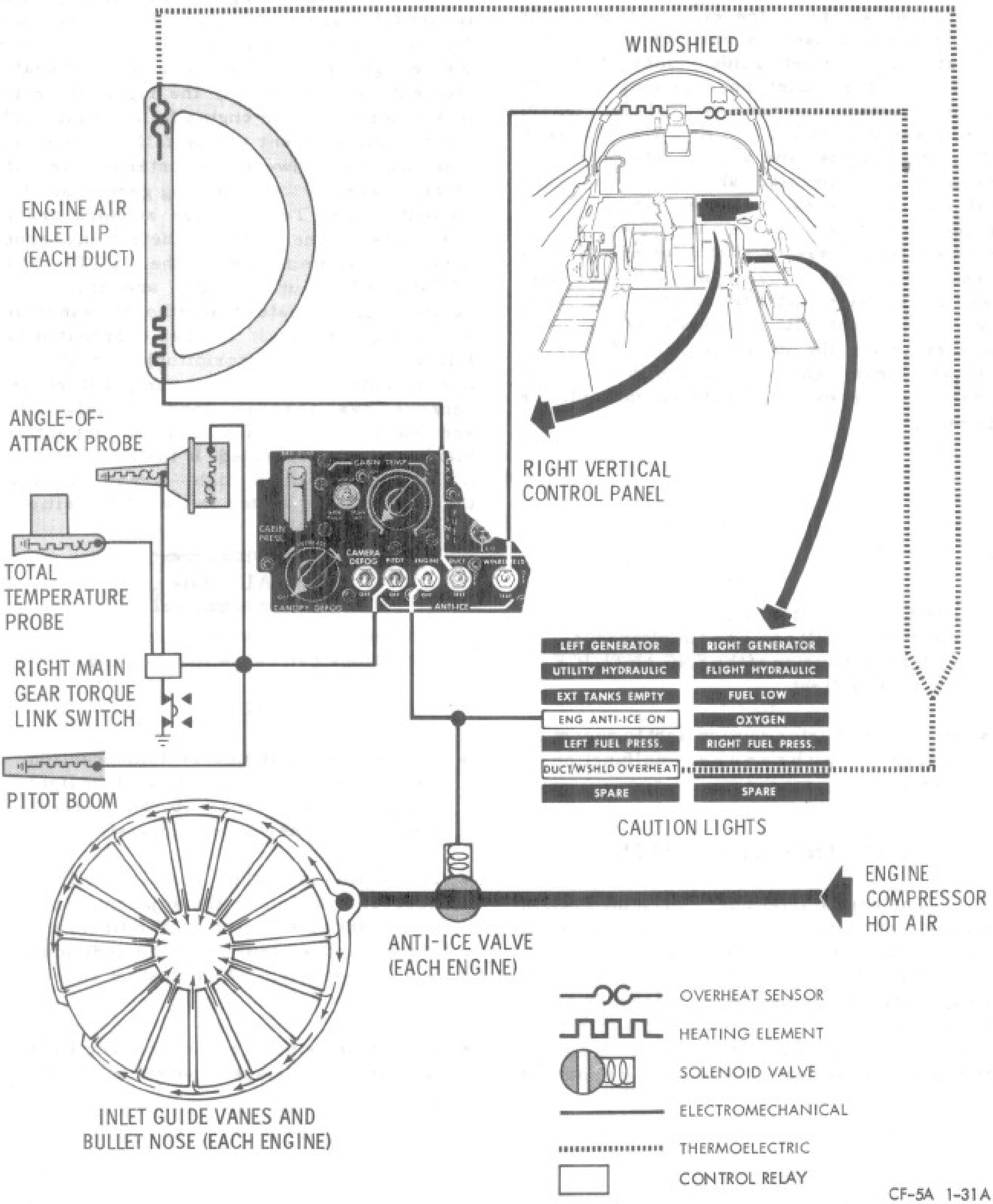


Figure 1-17-1

WINDSHIELD ANTI-ICING SYSTEM

3 The windshield anti-ice system incorporates a transparent electrically-conductive heating layer imbedded in the windshield glass. A three-position switch placarded WINDSHIELD, OFF, and TEST is located within the ANTI-ICE portion of the right vertical control panel, see Figure 1-17-1. Placing the switch to the WINDSHIELD position provides ac power to the conductive heating layer during flight to eliminate or prevent ice formation during flight in icing conditions. The spring-loaded TEST position is used for ground crew check purposes only. The system is powered by the right generator. However, in the event of generator failure of either generator with the duct anti-icing system also operating, the windshield anti-icing system will automatically shut off. The capacity of one generator is not sufficient during interlock transfer of ac power to operate the windshield anti-icing system with fuel boost pumps operating and the duct anti-ice system energized. To restore windshield anti-ice, if desired, it is necessary to turn off either the duct anti-ice system or both fuel boost pumps.

NOTE

Refer to Part 1, Section 3, and Part 4, Section 1, for fuel boost pump inoperative limitations.

4 Temperature sensors incorporated within the system will detect an overheat condition. Illumination of the DUCT/WSHLD OVERHEAT caution light on the caution light panel indicates that either the duct anti-icing system, if on, has overheated (see previous system description), or that the windshield temperature has reached approximately 180°F. If the caution light comes on, place the windshield anti-ice switch in the OFF position.

CAUTION

Prolonged use of system during ground operations can cause overheating.

NOTE

- Due to the combined systems caution light warning, the activation of the light may have been caused by an overheat condition in the duct anti-ice system.
- To isolate the system at fault, it will be necessary to place the adjacent duct anti-ice switch to OFF (if on) and place each system switch on individually to ascertain which system caused light illumination.
- If windshield anti-icing system caused light illumination, place switch at OFF.

PITOT BOOM ANTI-ICING SYSTEM

GENERAL

5 The pitot boom is deiced by an electric heating element within the boom. Pitot heat is controlled by manual selection of the pitot heat switch in the cockpit.

PITOT SWITCH

6 The two-position pitot heat switch, placarded PITOT and OFF, is located within the ANTI-ICE portion of the right vertical control panel, see Figure 1-17-1. Placing the switch in the PITOT position will cause the pitot boom heating element to supply heat to the boom.

CAUTION

Prolonged use of pitot heat on the ground may cause element to be overheated and malfunction.

ANGLE-OF-ATTACK PROBE ANTI-ICING SYSTEM

7 The angle-of-attack probe is deiced by two heating element arrangements; one element heats the probe body section and is thermostatically controlled; the other element

heats the probe when the weight of the aircraft is off the main gear torque link switch. The angle-of-attack body section heat is applied whenever the pitot heat switch is placed in the PITOT position.

TOTAL TEMPERATURE PROBE HEATER

8 A total temperature probe is provided on the lower nose section of the fuselage, see Figure 1-1-1, for obtaining ambient temperature signals to the central air data computer. A heating element and overtemp sensor within the probe, see Figure 1-17-1, is automatically activated when the weight of the aircraft is off the main gear torque link switch.

SECTION 18
COMMUNICATION AND NAVIGATION SYSTEMS

UHF COMMAND RADIO AN/ARC-507

GENERAL

1 The UHF command radio receiver-transmitter, located in the aft equipment compartment behind the ejection seat, provides voice transmission within the 225.00 to 399.95-megacycle range. Twenty operating frequencies can be preset in the set prior to flight. In addition, 3500 frequencies can be manually selected in steps of 0.5 megacycle without disturbing any of the preselected frequencies. The receiver-transmitter is automatically tuned after a channel or frequency change. An automatic direction finder (UHF/ADF) system is combined with the command UHF to provide bearing and track information on transmitting units. A fixed guard channel frequency of 243.00 megacycles is provided for emergency operation.

UHF COMMAND RADIO CONTROL PANEL (C-5210/ARC-507)

2 The UHF command radio control panel, see Figure 1-18-1, is positioned on the pedestal. A four-position rotary selector switch for controlling all primary input power to the equipment and for selecting the mode of transmission and reception is located on the right side of the panel. The switch positions are as follows:

(a) OFF.

(b) T/R for operation on selected frequency.

(c) T/R + G for operation on selected frequency and monitor guard reception.

(d) ADF for automatic direction finder operation on frequency selected.

NOTE

Allow approximately 10 seconds for warmup of receiver-transmitter when initially turning radio on.

3 A 3-position rotary selector switch allows rapid selection of desired frequencies as follows:

(a) PRESET CHAN position for preselected frequencies.

(b) MAN position for manually selected frequencies.

(c) GD XMIT position for operation on fixed guard position.

4 With the PRESET CHAN position selected, rotation of the knurled rotary type knob above the selected position marker (white line) in either direction will select channels 1 through 20. The channel number will appear in the round window in the upper left portion of the panel. A frequency card attached to the top of the panel is provided to record the frequencies preselected on each channel.

5 When the MAN position is selected, the digital readout of the frequency number in the window at the centre portion of the panel becomes the operating frequency. Three knurled rotary knobs below the window control the selection of the desired frequency. Each group of digits controlled by a knob is outlined by a white marker line leading to each knob. Turning the knobs in either direction will change the frequency.

6 In order to transmit and receive on the preset guard frequency, it is necessary to select GD XMIT on the left rotary selector switch and position the right rotary selector switch to T/R + G.

7 Rotation of the control knob placarded VOL increases (clockwise) or decreases (counterclockwise) audio volume.

8 A squelch disable (SQ DISABLE) switch is provided to allow the pilot to override the squelch in event a weak audio signal is being received. The switch should be positioned at OFF during normal operation of the receiver.

ANTENNA SELECT SWITCH

9 An antenna switch on the left console panel, see Figure 1-18-1, placarded ANT.,

COMMUNICATION AND NAVIGATION SYSTEMS CONTROLS

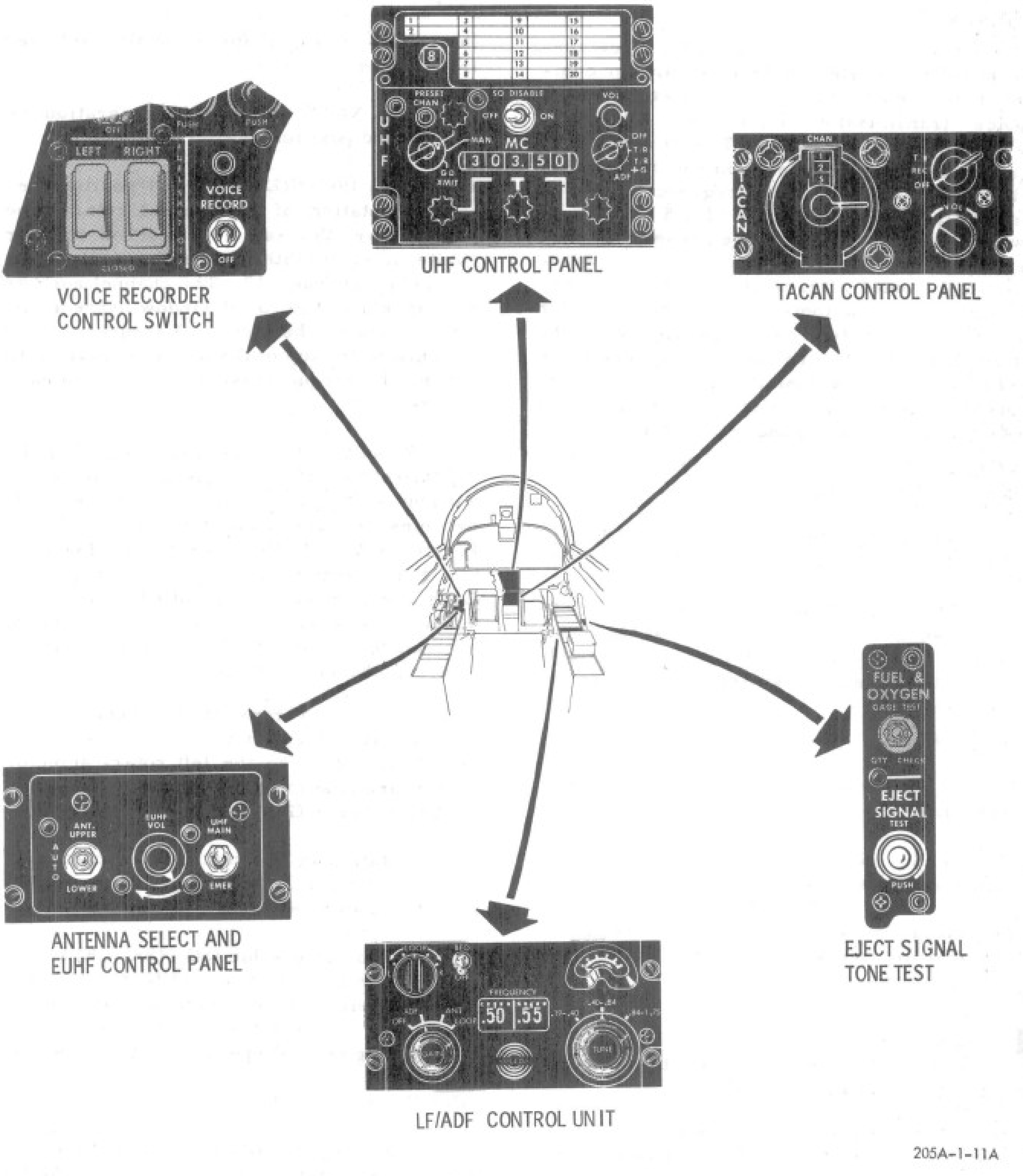


Figure 1-18-1

has three selective positions: UPPER, AUTO, and LOWER. In the UPPER position, only the antenna in the vertical stabilizer is used for UHF communication; the LOWER position selects the antenna on the underside of the fuselage nose. The AUTO position provides automatic antenna selection (upper or lower), depending on signal strength.

EMERGENCY TONE SIGNAL

10 An emergency tone modulation signal is automatically transmitted on the UHF command radio guard frequency when the seat is ejected from the aircraft. Separation of the personal leads quick-disconnect upon ejection will cause the UHF radio to automatically channel and lock on the emergency frequency (243.00 mc) and to transmit a continuous undulating distress signal tone. The tone signal source may be checked by pressing the button placarded EJECT SIG TEST on the panel outboard of the right console panel, see Figure 1-18-1.

CAUTION

- Do not press eject signal test button unless prior agreement with controlling agencies is established.
- Inadvertent operation may cause unnecessary alert of air/sea rescue units.

INTERCOM SYSTEM (AN/AIC-18)

11 A quick-disconnect receptacle is located on the underside of the fuselage left wing fillet to provide intercommunication with a ground crewman. Operation is automatic when the ground crewman connects his headset and microphone into the receptacle with electrical power applied to the aircraft.

VOICE RECORDER

GENERAL

12 The voice recorder, consisting of a recording and tape magazine section, is located in the left forward equipment compartment and provides pilot voice recording of any conversation he desires. Recording time for periods of up to 1 hour duration may be random (on-off) or continuous and is controlled by the pilot through the use of a control switch in the cockpit.

VOICE RECORDER CONTROL SWITCH

13 The voice recorder control switch on the left vertical control panel, see Figures 1-1-6 and 1-18-1, has two placarded positions: VOICE RECORD and OFF. With the switch in the VOICE RECORD position, the green light above the switch will come on and the pilot's voice will be recorded on the magnetic tape whenever he speaks into his microphone. If voice recording is not desired, the switch must be placed in the OFF position and the green light will not be illuminated.

EMERGENCY UHF RADIO TRANSCEIVER

GENERAL

14 The emergency UHF (EUHF) radio transceiver set is installed in the aft equipment compartment. The radio is provided for use as an alternate emergency voice communication method, primarily in the event the command UHF radio is rendered inoperative. The EUHF transceiver is powered by the 28-volt dc battery bus and utilizes a selector switch and volume control knob for operation. The emergency frequency of 243.00 megacycles is preset in the radio.

EUHF CONTROLS

15 The EUHF controls are adjacent to the UHF antenna select switch on the left console, see Figure 1-18-1. The switch placarded UHF has two selective positions: MAIN and EMER. With the switch at MAIN and electrical power applied to the aircraft, the EUHF radio is maintained in a standby condition. Placing the switch in the EMER position will cause the EUHF radio to become fully operational. The emergency radio, when activated, is completely independent of command radio guard channel operation and utilizes the UHF/IFF antenna for operation. Voice transmission is accomplished by using the microphone switch on the throttle. The volume control knob placarded EUHF VOL increases (clockwise) and decreases (counterclockwise) audio receiver volume.

TACAN AN/ARN-65

GENERAL

16 The AN/ARN-65 is the airborne portion of the short range navigation system called TACAN (Tactical Air Navigation). A UHF transmitter-receiver with associated indicators provides reception of bearing, distance, and tone identity information to show the location of the aircraft with respect to a specifically constructed surface navigation beacon. A total of 126 channels are provided and spaced 1 megacycle apart. Distance measuring pulses are transmitted by the radio set and returned by a ground beacon station. The receiver is automatically tuned to receive on a corresponding channel. This system enables the aircraft to obtain continuous indications of distance and bearing to any selected TACAN station within range. Bearing is displayed on the IDS-7 indicator on the instrument panel, see Figure 1-1-5. If TACAN is inoperative or bearing signals are weak or unreliable, the IDS-7 distance counter will display a black and yellow striped shutter and an orange background will appear in the deviation validity window.

TACAN CONTROL PANEL

17 The TACAN control panel on the pedestal, see Figure 1-18-1, incorporates a channel selector placarded CHAN, a volume control knob, and a function selector knob with 3 positions: OFF, REC, and T/R. In the OFF position, electrical power to the system is off. In the REC position, the set will receive bearing and audio identity signals only; distance will not be displayed because the transmitter portion of the set is not operating. In the T/R position, both the transmitter and receiver are in operation; the system will receive and display both distance and bearing of the station selected, refer to Paragraphs 28 and 36, following.

TACAN OPERATION

18 With the function selector knob at either REC or T/R, select the channel desired by turning the channel selector. Turning the serrated portion of the knob will select the first two digits of the channel frequency; turning the lever portion of the knob will select the last digit. Volume control is increased by rotating the volume control knob clockwise.

IFF/SIF CONTROL PANEL

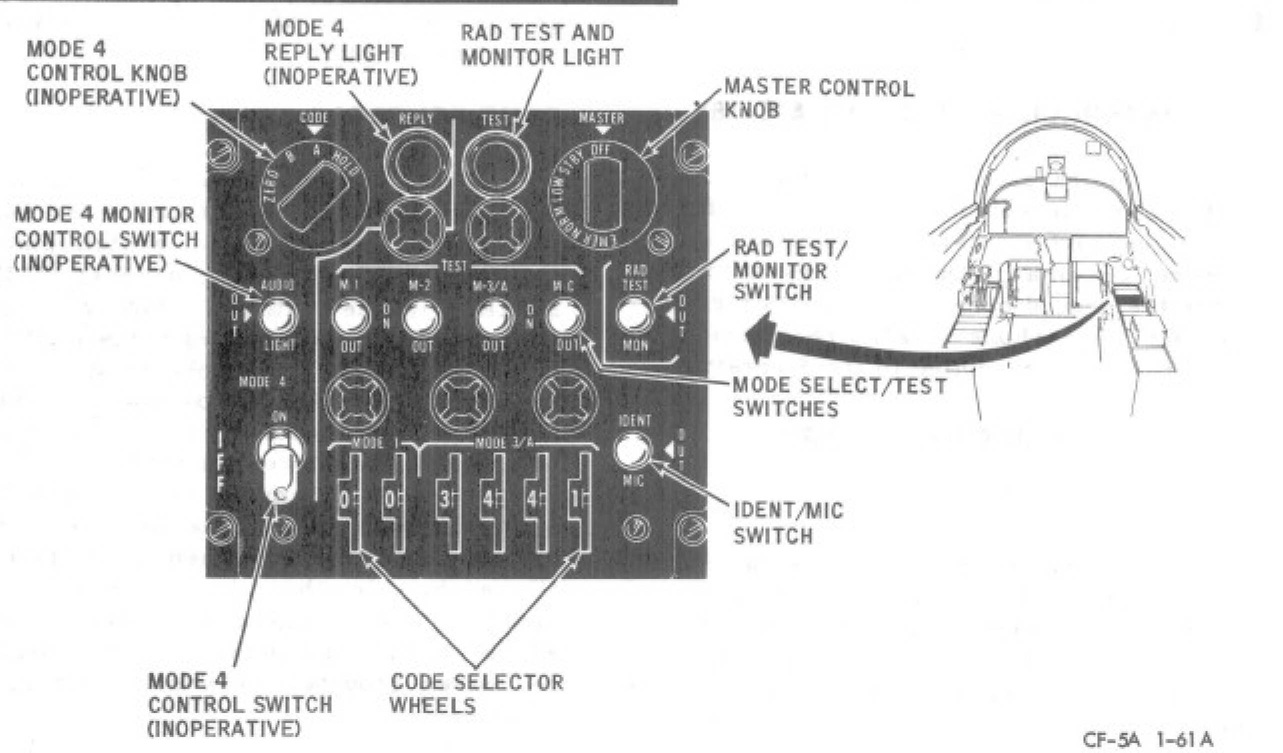


Figure 1-18-2

UHF/ADF RADIO COMPASS (AN/ARA-50)

18A The ultra high frequency/automatic direction finder (UHF/ADF) provides bearing information on transmitters operating in the frequency range of the UHF command radio. With the rotary selector knob on the command radio panel at ADF, see Figure 1-18-1, signals are received via the UHF/ADF directional antenna located aft of the nose gear doors. The information is displayed on the integrated display of situation indicator, refer to Paragraphs 28 and 36, following.

LF/ADF RADIO COMPASS

GENERAL

18B The low frequency/automatic direction finder system (LF/ADF) is primarily a backup navigational aid for the TACAN and UHF/ADF systems. The system provides bearing information on low-frequency transmitting stations, and can also be used to monitor low-frequency and broadcast band transmissions of flight and weather information. Principal system components are receiver, loop antenna (directional), sense antenna, and control unit. Bearing information is displayed on the integrated display of situation indicator, refer to Paragraphs 28 and 36, following, while audio signals are fed through the pilot's personal lead jack.

CONTROL UNIT

18C The control unit, see Figure 1-18-1, is located on the right console and incorporates mode switch/gain control, band switch/tuning control, tuning meter, frequency dial, BFO switch, and LOOP L-R switch. Switch functions follow:

(a) The larger knob of the mode switch/gain control may be selected OFF, ADF, ANT, LOOP; the smaller knob increases gain when turned clockwise.

(1) ADF - For automatic directional antenna alignment.

(2) ANT - For audio reception using sense antenna.

(3) LOOP - For manual directional antenna alignment.

(b) The larger knob of the band switch/tuning control selects frequency bands of .19-.40, .40-.80, and .80-1.75 mcs for display in the frequency dial window; the smaller knob is used to rotate the frequency dial and provide a visual indication of correct tuning.

(c) The BFO switch controls the operation of the beat frequency oscillator which is used to indicate off-station tuning. The tuning meter is a visual aid, and registers a gain, when retuning.

(d) The LOOP L-R switch may be used for directional antenna alignment when the automatic ADF mode malfunctions. Movement of the antenna is instinctive with L or R movement of the switch.

IFF/SIF (AN/APX-77)

GENERAL

19 The IFF/SIF (AN/APX-77) is an airborne pulse-type transponder which enables the aircraft to automatically identify itself through selective identification (Selective Identification Feature - SIF) when challenged by surface or airborne radar equipment capable of interrogation. The set can also identify the aircraft in which it is installed as a friendly aircraft within a group of specific friendly aircraft. Supplementary purposes are to provide momentary identification of position and altitude on request by Mark XII IFF/SIF equipped radar stations. In operation, the set receives coded interrogation signals and automatically transmits correct coded reply responses; it is not capable of performing interrogations of other stations. Operation is possible in any one of five modes, with the capability of I/P (Identification of Position) and emergency identification. The modes of operation have the following significance: Mode 1 - Security Identity, Mode 2 - Personal Identity, Mode 3/A - Air Traffic Identity, Mode 4 - Not Used, Mode C - Altitude Interrogation. The equipment consists of an IFF control panel, see Figure 1-18-2, on the right console, a transponder (transmitter-receiver), an airborne test set/in-flight monitor, and an antenna switching unit (lobing switch) in the forward equipment compartment for the upper (aft) and lower (forward) antenna positions. The lobing switch rapidly transfers contact of the transmitter-receiver from one antenna to the other. This

constant alternation eliminates blank spots in the antenna pattern caused by either aircraft structure and/or line of sight station to station. The receiver is sensitive to all signals within its frequency range; however, only those signals meeting the complete predetermined requirements of the code being used will be recognized and answered. Mode 2 code settings are set into the transmitter-receiver on the ground and thus are fixed for any one flight. All other codes are set up at the control panel and can be turned on or off. Replies to Modes 1, 2, 3/A, and C interrogations, as well as to I/P and emergency replies, are shown on the ground station equipment. In the case of the more complicated SIF codes, ground stations will use a plan position indicator (PPI) and letter symbol indicator to decode and indicate supplementary information, such as specific identification and location, and flight or aircraft conditions. The Mode C code reply response, programmed by input from the central air data computer (CADC), is inoperative. An optional low-power setting provision restricts sensitivity so that replies are made only to local interrogations. In the event of ejection from the aircraft, the ejection triggering system will automatically cause the IFF transponder to immediately transmit on emergency regardless of the position selected by the master control knob on the IFF/SIF control panel.

IFF/SIF MASTER CONTROL KNOB

20 The IFF/SIF master control knob, see Figure 1-18-2, has five positions marked OFF, STBY, LOW, NORM and EMER. When positioned to STBY, the equipment is turned on and warmed up but will not transmit. When positioned to LOW, only local (strong) interrogations are recognized and answered. When positioned to NORM, full range recognition and reply occurs. Transmitted power from the IFF system is the same for both the LOW and NORM positions. The knob must be pulled out and turned to position it to EMER. When the knob is positioned at EMER, an emergency-indicating pulse group is transmitted each time a Mode 1 or Mode 3/A (code 7700) interrogation is recognized.

IDENTIFICATION-OF-POSITION (I/P) SWITCH

21 The IDENT/MIC switch, see Figure 1-18-2, is used to control transmission of identification-of-position (I/P) pulse groups. The switch has three positions marked MIC, OUT, and IDENT. When the switch is momentarily held in the spring-loaded IDENT position, the I/P timer is energized for approximately 20 seconds. If a Mode 1 or 3/A interrogation is recognized within this 20-second period, I/P replies will be made. When the switch is placed in the MIC position, the I/P pulse group will be transmitted in reply to a Mode 1 or 3/A interrogation whenever the microphone switch is pressed. Placing the switch to the OUT position prevents transmission of I/P groups.

MODE SELECT/TEST SWITCHES

22 The four mode select/test switches used by the pilot are marked TEST, ON, and OUT. The switches are labelled M-1, M-2, M-3/A, and M-C and are grouped under the TEST heading, see Figure 1-18-2. The OUT position for each switch disables the transmitter-receiver for the mode selected. The ON position enables each respective switch to activate the modes selected. If more than one switch is placed at ON, the transmitter-receiver will reply to interrogations for all modes selected. The switches are spring-loaded to the ON position from the TEST position. The TEST positions are used for functional testing of the systems and the TEST light will come on while the test is being performed. The lights are press-to-test and turn-to-dim type lamps.

NOTE

The light placarded REPLY is not used (Mode 4 operation inoperative).

RADIATION TEST/MONITOR SWITCH

23 The radiation test/monitor switch, see Figure 1-18-2, has three placarded positions: RAD TEST, OUT, and MON. The switch is

spring-loaded for momentary contact in the RAD TEST position and will return to the OUT position when released. The RAD TEST position is used by ground service personnel to energize the test mode feature of the transponder during tests with AN/UPM-92 or similar equipment. Airborne testing, at present, is not feasible until suitable ground equipment is installed in the communications systems of ground facilities. With the switch in the MON position, the light placarded TEST will illuminate whenever a signal response to an interrogation occurs. With the switch in the OUT position, the signal response to an interrogation occurs; however, the light will not illuminate. Normal position for flight is MON.

CODE SELECTOR KNOBS

24 Two sets of thumb actuated code selector wheels, see Figure 1-18-2, are provided to set Mode 1 and Mode 3/A codes. The set of wheels labelled MODE 1 selects 32 different codes. The set of wheels labelled MODE 3/A selects 4096 codes. Each wheel is placarded with the digits 0 to 7 and can be seen recessed in the windows on the face of the control panel.

PRIMARY GYRO HEADING AND ATTITUDE REFERENCE SYSTEM (GHARS)

GENERAL

25 The primary gyro heading and attitude reference system (GHARS) provides simultaneous outputs of grid heading, magnetic heading, and pitch and roll and consists of a gyroscopic assembly and an electronics unit in the aft fuselage equipment compartment and a gyro controller on the right console, see Figures 1-1-7 and 1-1-8. Magnetic reference for the system is provided by a magnetic azimuth detector. A compensator is provided to permit compensation of the detector for magnetic deviation effects. The detector and compensator are installed in the dorsal area of the fuselage. In addition, the system incorporates a compass repeater amplifier to resolve heading information into coordinates for vertical gyro alignment. Grivation slaving cutout during turns is provided by a rate switching gyro. The compass repeater amplifier and rate switching gyro are also housed in the aft fuselage equipment compartment. All displays of information are

GYRO CONTROLLER PANEL

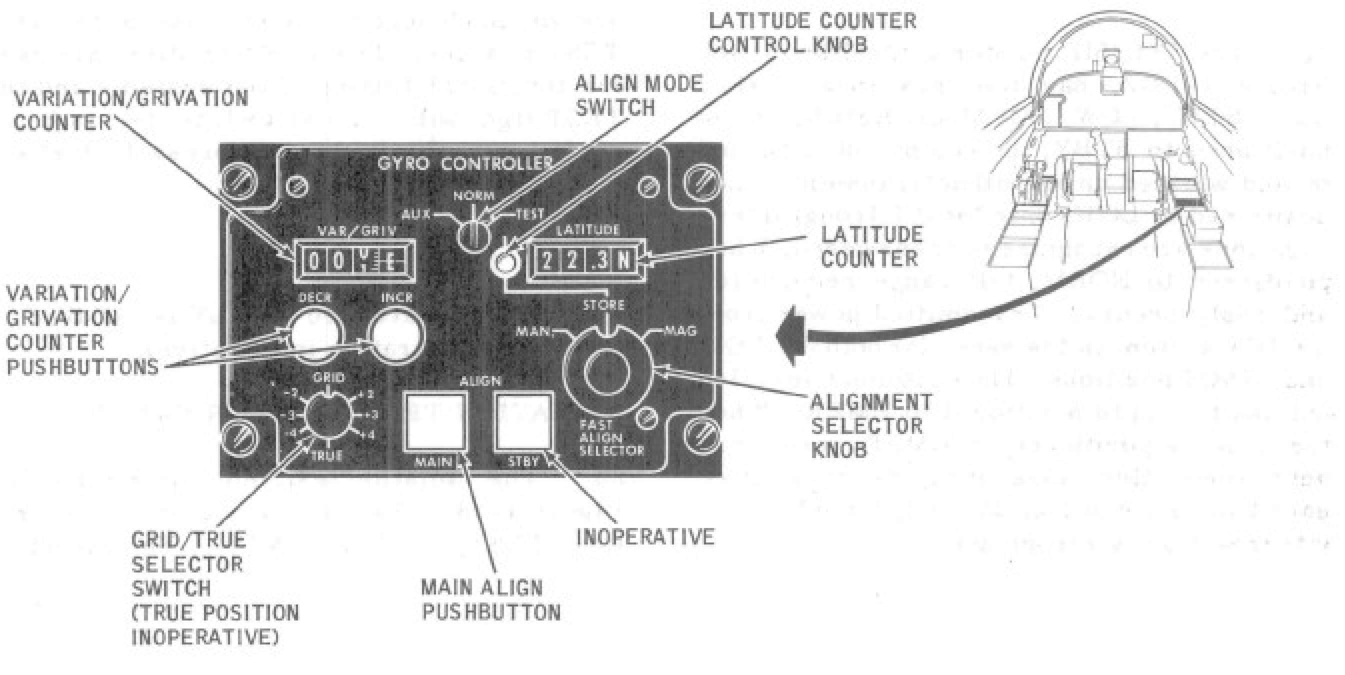


Figure 1-18-3

shown on the integrated display of situation indicator (IDS-7) and the attitude indicator on the instrument panel, see Figure 1-1-5.

GYRO CONTROLLER PANEL

26 The gyro controller panel, see Figure 1-18-3, provides the means of control of the GHARS. The GHARS will come on whenever ac electrical power is applied to the system. In particular the controller provides the following functions:

(a) Manual and automatic setting of variation or grivation. Two pushbuttons are provided to increase or decrease the setting.

(b) Manual and automatic setting of latitude.

(c) Platform fine alignment combined with variation or grivation, resetting at a rate of approximately 4 degrees per minute when airborne and at a rate of approximately 50 degrees per minute when on the ground, by momentarily pressing the increase (INCR) or decrease (DECR) pushbuttons. On the ground with the landing gear switch actuated (weight of aircraft on landing gear), only the variation or grivation is controllable; the directional gyro cannot be torqued.

(d) Selection of platform alignment to either manual (MAN), STORE, or magnetic (MAG) ± variation or ± grivation. MAG is the normal setting for flight. MAN is not used during flight unless the actual magnetic or grid heading is known. STORE is not used during flight.

(e) Displays variation or grivation as a digital readout. Variation is displayed when the platform is aligned to a true meridian (inoperative). Grivation is displayed when platform is aligned to a grid meridian (grivation is the grid variation between grid north and magnetic north).

(f) A manual control for selection of true or grid mode. The true mode is inoperative; therefore, the control should be placed in the grid position.

(g) A switch provides auxiliary (AUX), normal (NORM), or TEST functions. The TEST position is not used during flight.

(h) Provides digital readout of latitude which is manually set to the local latitude before takeoff.

(j) Manually operated pushbutton switch to allow fast alignment of the directional gyro of the platform to the reference selected by the MAN/STORE/MAG switch.

(k) The standby (STDBY) ALIGN position is inoperative.

SYSTEM OPERATION

27 The GHARS performs the functions of an all-attitude flight reference system, providing continuous pitch, roll, and heading data for the attitude indicator and IDS-7 indicator. To operate the system, accomplish the following:

(a) AUX/NORM/TEST switch should be in the NORM position.

(1) The switch selects the mode of alignment but does not command the alignment. After the first cycle of rapid alignment is completed (fast alignment is automatic and occurs when power is initially applied to the system to accomplish initial heading alignment and vertical erection) and if the alignment mode is to be changed, a new mode is selected and the MAIN ALIGN pushbutton is momentarily pressed. The MAIN ALIGN mode initiates realignment of the system gyro to the mode selected by the MAN/STORE/MAG switch. When the MAIN ALIGN pushbutton is pressed a 30 (±5)-second holding relay is activated and the pushbutton, incorporating an amber light, will illuminate.

(2) The AUX position is used only when ground test of the system reveals a vertical erection is out of tolerance and a mission must be flown prior to service of the system.

(3) TEST position is used only by ground servicing personnel. AUX/NORM/TEST modes are selected by a screwdriver operated rotary switch.

(b) GRID/TRUE selector switch should be in the GRID position.

RTW ON LAMBERT CONFORMAL MAP

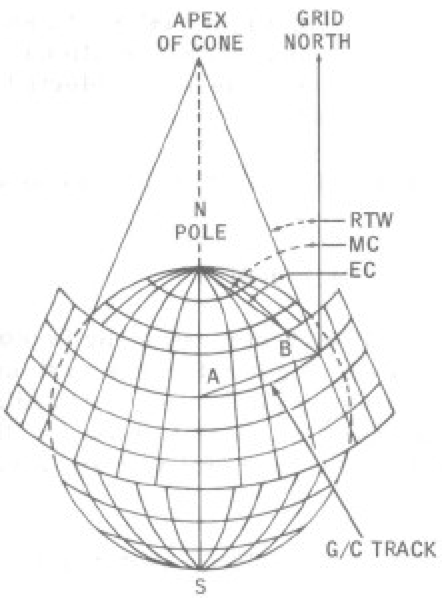


Figure 1-18-4

NOTE

TRUE position is inoperative.

(1) The GRID position corrects for earth's rate of movement only.

(2) The GRID ± 2, 3, or 4 positions (inoperative) correct for earth rate and residual transport wander (RTW).

a. RTW is the difference between earth convergency (EC) and map convergency (MC); however, it is expressed as a rate since the error depends also on the aircraft's speed. Therefore the average RTW rate is a function of earth convergency, map convergency, and elapsed time. This may be written as $EC - \frac{MC}{t}$, where t = the elapsed time, see Figure 1-18-4.

b. Constant gyro headings generate great circle tracks; however, for the Lambert Conformal Map, map convergency is the product of change in longitude and the constant of the cone, (0.748819) of the map. Thus, RTW is made solely a function of EC and MC; i.e., RTW = EC - MC.

c. RT = (change in longitude x sine of latitude) - (change in longitude x constant of the cone). Since the cone is constant over any one map and the sine of the latitude varies with the latitude of operation, the RTW will be negative below the parallel or origin and positive above it. RTW corrections are manually set on the GRID/TRUE control, the ranges are from -2 to -4 (negative) and from +2 to +4 (positive). The RTW compensation signals are applied to the directional gyro torquer.

(c) Set MAN/STORE/MAG switch to MAG.

(1) If MAN position is selected, gyro alignment is accomplished for the track selected in the track readout window of the IDS-7 indicator. The MAIN ALIGN pushbutton must be pressed if MAN position is selected.

(2) MAG position provides error signal for normal alignment to magnetic heading information from the magnetic detector, ± grivation. The magnetic angular difference is modified by the INCR/DECR pushbuttons.

(3) STORE position will allow storage of the last heading data flown, provided the aircraft is not moved after shutdown. Before turning off ac electrical power turn switch to STORE. When restarting aircraft, leave switch at STORE until rapid erection is complete (approximately 1 minute), then switch to MAG.

NOTE

- If ac power failure occurs or aircraft is moved after start with STORE position selected, the system will realign to a heading other than the desired stored heading.
- MAG position is the normal in-flight mode of operation.

(d) Set local area grivation by means of pushbuttons, DECR or INCR. Digital readout windows will display grivation east (E) or west (W).

(e) Set latitude readout for local latitude geographical position by turning knurled knob to left of window display. Turn knob clockwise to increase; counterclockwise to decrease.

(f) With aircraft ac power on, check roll, pitch, and heading indication on attitude indicator and IDS-7. Indicators should begin to centre, indicating rapid erection.

NOTE

Approximately 1 minute is required for instruments to be stabilized.

(g) After takeoff and when flying straight and level, pressing the MAIN ALIGN pushbutton will correct directional gyro alignment, when necessary.

INTEGRATED FLIGHT INSTRUMENTS

GENERAL

28 The attitude indicator, and the integrated display of situation indicator (IDS-7) display the horizontal, vertical, and directional situation as programmed from the GHARS. Additionally, the IDS-7 will display distance and/or bearing information based on TACAN, UHF/ADF, and LF/ADF inputs. Associated component controls for the distance and/or bearing displays are TACAN, command UHF, integrated display of situation indicator control panel (ICP-7), and the LF/ADF control unit.

ATTITUDE INDICATOR

29 The attitude indicator, see Figure 1-18-5, contains a sphere with the upper half painted gray and the lower half painted black. The gray area represents the sky and the black area represents the ground. At the junction of the gray and black is the horizon bar. General pitch attitude near level flight may be obtained by referencing the miniature aircraft against the sphere colour. Specific pitch attitude may be obtained by referencing the miniature aircraft against the attitude sphere pitch markings. These are short lines every 5 degrees of pitch, longer lines every 10 degrees of pitch, and numbered lines every 30 degrees of pitch. The pitch trim knob allows the attitude sphere to be adjusted to provide the desired pitch presentation relative to the miniature aircraft.

30 Bank Pointers. A bank pointer is provided at the top and bottom of the indicator. An angle-of-bank scale is located at the bottom of the face of the indicator, which is graduated in 10-degree increments up to 30 degrees and in 30-degree increments up to 90 degrees of bank.

31 Turn and Slip Indicator. One needle width deflection provides a 4-minute 360-degree turn.

32 Attitude Warning Flag. The attitude warning flag (OFF) will appear whenever electrical power to the system has failed or is interrupted. The flag will also appear during the initial application of electrical power for approximately 1 minute. The instrument is unreliable until the flag disappears.

WARNING

- There is no warning of attitude sphere malfunctions other than power failure.
- The attitude warning flag will not appear with a slight electrical power reduction or failure of other components within the system. Failure of certain components can result in erroneous or complete loss of pitch and bank presentations without a visible flag. It is imperative that the attitude indicator be cross-checked with other flight instruments when under actual or simulated instrument conditions.

INTEGRATED DISPLAY SITUATION INDICATOR (IDS-7)

33 The IDS-7 indicator, see Figure 1-18-5, provides the pilot with a view of the navigation situation as if he were above the aircraft looking down. The indicator presents a lighted display of the following output data:

(a) Magnetic heading displayed by means of a rotating heading card against a lubber line.

INTEGRATED FLIGHT INSTRUMENTS

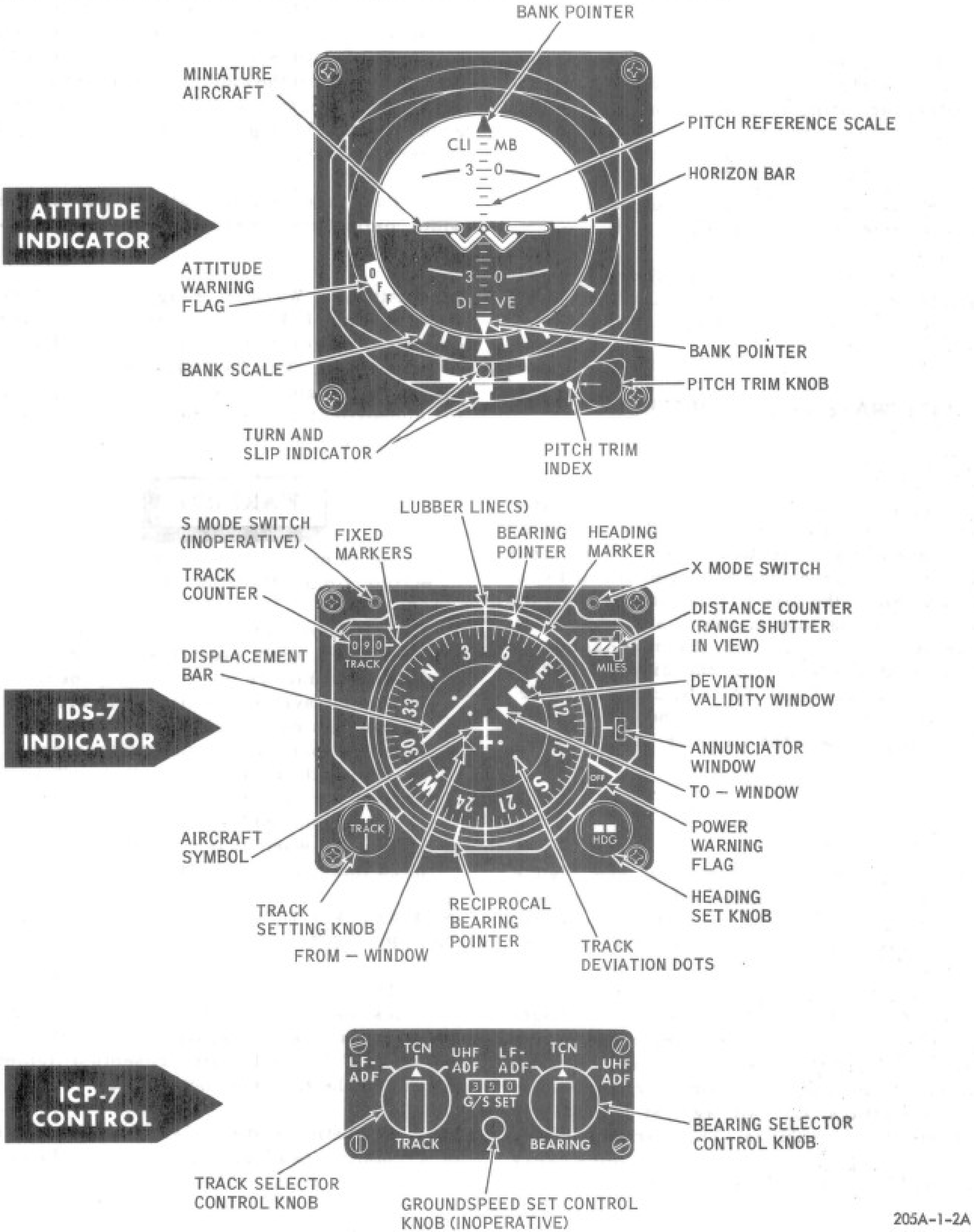


Figure 1-18-5

(b) Grid heading displayed by the heading marker against the rotating heading card only when the heading set knob is pulled out (white band on knob shaft displayed).

(c) Bearing is displayed by means of a rotating pointer on the outer periphery of the rotating card. Bearing is read against the heading card. Relative bearing is read relative to the lubber line.

(d) Track is displayed by use of track setting knob which sets the track in degrees in a 3-drum counter. Turning the knob clockwise will increase the numbers; turn knob counterclockwise for decrease. The track pointer, which rotates around the inner edge of the card, is synchronized with the heading card after it is set. The reciprocal end of the pointer is at the opposite end of the card.

(e) Displacement of the aircraft from the selected track is accomplished by the position of the displacement bar over a fixed scale of five dots. The centre dot is zero and each dot is spaced 5 degrees apart.

(f) Manual positioning of the heading index is displayed by means of the heading marker. With the heading set knob in the mid-position (white band on knob shaft not displayed and knob not pushed against spring-loaded tension) a clockwise rotation of the knob will displace the marker in a clockwise direction; counterclockwise rotation will displace the marker accordingly.

(g) Range to a selected TACAN station is displayed on a 3-drum counter in the upper right corner. The drum counters will display 0 to 999 nautical miles. The units drum has 1-mile and half-mile index lines.

(h) Range shutter of black and yellow stripes will appear over the miles scale whenever unreliable information is being received or when no range information is available.

(j) TO-FROM position from a selected navigation position line is shown by two white arrow shaped flags situated under the track pointer. The "TO" flag points in the direction of the pointer; the "FROM" flag points in the direction of the reciprocal of the track pointer.

(k) The electrical power OFF flag will show whenever ac or dc power failure occurs.

(m) Deviation validity display in the form of an orange rectangular flag under the track pointer will show whenever the deviation requires a correction.

(n) Magnetic synchronization is corrected whenever the annunciator window displays a warning flag with the letters "C" or "CC." To correct magnetic synchronization, push in and turn the knob placarded HDG clockwise for "C" and counterclockwise for "CC" until the letters disappear.

(p) The S Mode (gyro slaving) is inoperative.

(q) The X Mode switch is a "push-push" type switch. It should always be in (white band not showing) for normal system operation. If it is out (white band showing), the compass card will be inoperative since there is no secondary compass source in the aircraft.

INTEGRATED DISPLAY OF SITUATION INDICATOR CONTROL PANEL (ICP-7)

34 The ICP-7 control panel on the instrument panel, see Figures 1-1-5 and 1-18-5, provides the pilot with the means of controlling track and/or bearing indications as displayed on the IDS-7 indicator. The track selector control knob on the left portion of the control panel, placarded TRACK, and the bearing selector knob on the right portion, placarded BEARING, each allows selection of three placarded modes of operation: LF-ADF (low frequency radio automatic direction finder mode), TCN (TACAN), and UHF-ADF (command radio automatic direction finder mode). The G/S SET (groundspeed set) control knob and the digital readout window portion of the panel is inoperative.

35 TRACK knob selection of LF-ADF, TCN, or UHF-ADF will cause the displacement bar

on the IDS-7 to indicate displacement of the aircraft to the left or right of the track to the selected radio station. In addition, the TO or FROM window of the IDS-7 will indicate station position relative to aircraft heading.

36 BEARING knob selection of LF-ADF, TCN, or UHF-ADF will cause the bearing pointer on the IDS-7 to indicate bearing to the selected radio station.

STANDBY ATTITUDE INDICATOR

37 The standby attitude indicator on the instrument panel, see Figure 1-1-5, will provide an attitude indicating system if the GHARS integrated attitude indicator malfunctions. The standby indicator is powered by the right ac bus and a power adapter transformer provides 115-volt, 3-phase power to the indicator, which is a self-contained gyro instrument. Complete erection of the gyro should occur within approximately 5 minutes after ac power is applied to the aircraft. A caging knob is provided for manual erection of the gyro axis to the vertical position in relation to the instrument case. If necessary to cage the gyro during flight, the aircraft must be straight and level. Turn, acceleration, and deceleration errors are corrected by an erection mechanism at the rate of 3 to 6 degrees per minute. Pitch errors resulting from acceleration or deceleration will appear as a climb after an increase in airspeed and as a descent after a decrease in airspeed. Turn error should not exceed 5 degrees of pitch and/or bank after completion of a normal turn (errors in excess of 5 degrees may be expected after acrobatic manoeuvres).

NOTE

Manually cage standby indicator prior to flight.

38 On the sphere, the horizon is represented by a solid fluorescent line. The bank scale is located at the bottom of the indicator and a bank index (pointer) is located on the sphere. The bank scale is graduated with 10-degree increments up to 30 degrees and in 30-degree increments up to 90 degrees of bank. The fixed miniature aircraft is displayed as a "W" and the horizon line is adjusted to the aircraft for level flight condition by a knob on the lower left of the indicator. The instrument permits 360 degrees of rotation about the pitch and bank axes without tumbling the gyro. The horizon bar provides sensitive pitch reference near a level flight attitude. As the aircraft enters 27 degrees of pitch, up or down, the horizon bar remains at this limit and the sphere then becomes the new reference. The sphere is marked with the words CLIMB and DIVE, each followed by a bullseye. CLIMB or DIVE, when under the miniature aircraft, represents approximately 60 degrees of pitch. The bullseye is graduated at 70, 80, and 90 degrees of pitch. As the aircraft approaches vertical, as in a loop, the sphere begins to rotate 180 degrees (counterclockwise in a climb, and clockwise in a dive). The momentary rotation of the sphere is controlled precession and should not be confused with gyro tumbling.

39 An attitude warning flag appears if the instrument is not receiving adequate electrical power.

WARNING

The attitude warning flag is an indication of insufficient electrical power only. It does not appear with malfunctions of other components within the instrument.

EMERGENCY (MAGNETIC) COMPASS

40 The emergency compass on the right side of the cockpit windshield frame, see Figures 1-1-3 and 1-1-4, provides magnetic heading information.

ANGLE-OF-ATTACK INDICATING SYSTEM

GENERAL

41 The angle-of-attack indicating system provides an indication of the angular position of the wing chord in relation to the aircraft flight path. This indication is used for approach speed monitoring on this aircraft. The system includes a probe type transmitter on the left front fuselage side, see Figure 1-1-1;

an angle-of-attack indicator on the instrument panel, see Figure 1-1-5; and an approach indexer on the top of the instrument panel, see Figure 1-1-5. In flight, the probe will detect relative wind by means of differential air pressure through a series of slots at the front of the probe. Probe rotation moves three potentiometer wiper arms, producing electrical resistance variations which are signalled to the indicator on the instrument panel. Two electrical heaters, one in the probe and the other in the case (at the fuselage attachment), are provided for anti-ice protection. Refer to Section 17 of this Part for anti-ice operation.

ANGLE-OF-ATTACK INDICATOR

42 The angle-of-attack indicator is calibrated in arbitrary units from 0 to 30, which is equivalent to a range of -10 to +40 angular degrees of probe rotation. The fixed approach indexer on the indicator face is set at the 3-o'clock position and is set against the 20-increment marker, which is the optimum angle-of-attack setting for landing approaches in this aircraft. The remainder of the index markers are not used. The pointer moves with the probe selection of angle-of-attack. The power OFF warning flag will appear whenever electrical power is removed from the system.

ANGLE-OF-ATTACK INDEXER

43 The angle-of-attack indexer, which operates from switches within the angle-of-attack position indicator, presents angle-of-attack position information during a landing approach by illuminating symbol cutout lamps. The indexer consists of three red lamps arranged vertically. The low speed top symbol (V-shaped) lights when the angle-of-attack is above optimum for the approach; the on-speed centre symbol (doughnut-shaped) lights when at optimum angle; and the high speed symbol (inverted V-shape) lights at low angle-of-attack. The indexer lights will only come on when the landing gear is in the down position.

CENTRAL AIR DATA COMPUTER (CADC)

44 The central air data computer furnishes corrected static pressure and temperature to the following control systems: landing gear warning system, engine doors close, servo-altimeter, mach dial of airspeed-mach indicator, stability augmenter system, lead computing sight system, and velocity and height to the VGH recorder (when installed). The system consists basically of an electromechanical computer, which processes the raw data from the pitot-static system, and a temperature sensing probe. When the raw data reaches the computer, it is transformed into electrical signal outputs to the various components of the supplied systems. The CADC is equipped with a failure monitoring system which continually monitors the computing functions. Should a computing function fail, the master caution light will flash continuously on the instrument panel and the caution light panel on the right console will display the words "AIR DATA COMPUTER." Should a component fail affecting only the airspeed-mach indicator, the indicator will display a power OFF flag; or if the altimeter fails, the STDBY flag on the altimeter will come into view.

SECTION 19

LIGHTING EQUIPMENT

GENERAL

1 The lighting equipment provisions of the aircraft are categorized as exterior and interior lighting. Exterior lighting consists of the landing-taxi lights, position and rotary beacon lights, and formation lights. Interior lighting consists of individual blue-white instrument lights, cockpit floodlights, console and panel lighting, and a utility light.

EXTERIOR LIGHTS

LANDING-TAXI LIGHTS

2 The landing-taxi lights are controlled by a switch on the left vertical control panel, see Figure 1-1-6. The lights are electrically controlled, combination landing and taxi light assemblies which extend and retract. With the landing gear down and the position lights on, the lights will extend to the landing light position. Without the weight of the aircraft on the main gear, turning the landing-taxi light switch on illuminates a high intensity beam. When the weight of the aircraft is on the main gear and the position and landing-taxi light switches are on, a lower intensity beam will be illuminated and the light assemblies will retract to the taxi position. The landing-taxi light assemblies will retract automatically when the position lights are turned OFF or the landing gear is retracted. The landing-taxi lights are located in either side of the lower fuselage inboard of the air intakes. The position lights switch on the right console, see Figures 1-1-7 and 1-1-8, must be selected BRIGHT or DIM to enable the landing-taxi lights to be turned on.

POSITION AND ROTARY BEACON LIGHTS

3 The position lights consist of a red (left) light and a green (right) light on the outer section of the wings and white lights in the vertical stabilizer and lower left and lower right centre fuselage sections. The lights are controlled by the position light switch. The position light in the nose of each tip tank matches the color of, and is connected to, its related wing position light. The rotary beacon (red) light is mounted in the vertical stabilizer and is controlled by a switch on the right console, see Figures 1-1-7 and 1-1-8.

FORMATION LIGHTS

4 The formation lights consist of two white-frosted lights (right and left sides) individually flush mounted in the dorsal area of the fuselage to the rear of the cockpit, see Figure 1-1-1. The lights are controlled by the position light switch.

INTERIOR LIGHTS

5 Interior lighting consists of individual blue-white instrument lights, white floodlights, console and panel lights, and a utility light in the cockpit. The lights operate on ac power, and intensity is controlled by rheostat knobs on the right console, see Figures 1-1-7 and 1-1-8. The two white instrument floodlights in the cockpit automatically switch to dc battery power if ac power supply fails, but intensity control is lost. A utility light with integral rheostat control switch and extension cord is mounted on the right canopy rail.

[illegible]

SECTION 20

OXYGEN SYSTEM

GENERAL

1 The aircraft uses a liquid oxygen system to supply breathing oxygen to the pilot. An oxygen regulator on the right console controls the flow and pressure of the oxygen and distributes it in the proper proportions to the mask. The oxygen regulator contains a pressure indicator, a blinker type flow indicator, emergency flow lever, oxygen diluter lever, and a supply lever.

OXYGEN REGULATOR

2 A combination pressure breathing, diluter demand oxygen regulator, see Figure 1-20-1, is used in conjunction with the oxygen mask. The oxygen system is controlled by the supply, the diluter, and the emergency levers. Gaseous oxygen is supplied to the regulator in the range of 65 to 110 psi. The regulator reduces the oxygen pressure, mixes oxygen with air in varying amounts depending on the altitude and crewmember demand, and delivers it through a flexible tube to the oxygen mask. At high cockpit altitudes (approximately 40,000 feet and above), the regulator supplies positive pressure breathing. The crewmember receives a visual indication of system operation from the flow indicator and oxygen pressure gauge on the oxygen regulator panel.

3 Diluter Lever. The diluter lever, see Figure 1-20-1, has two positions, NORMAL OXYGEN and 100% OXYGEN. With the lever at NORMAL OXYGEN, the regulator mixes air with oxygen in varying amounts, according to altitude, and delivers it through a flexible hose to the oxygen mask. With the lever at 100% OXYGEN, the regulator delivers 100 percent oxygen regardless of altitude.

4 Emergency Lever. The emergency lever, see Figure 1-20-1, has three placarded positions, EMERGENCY, NORMAL, and TEST MASK. The lever should remain in the NORMAL position at all times unless an unscheduled pressure increase is required. Moving the lever to EMERGENCY provides continuous positive pressure of 100 percent oxygen to the mask. Holding the lever in the TEST MASK position provides pressure to test mask for leaks.

WARNING

When placing the emergency lever in EMERGENCY or TEST MASK position, it is mandatory that the oxygen mask be fitted to the face and not removed. Continuous use of positive pressure with a leaking oxygen mask or the mask removed for extended time periods will deplete the oxygen supply rapidly.

5 Supply Lever. The oxygen system supply lever, see Figure 1-20-1, placarded SUPPLY with positions ON and OFF is located on the right end of the regulator. When the lever is in the ON position, oxygen is permitted to flow through the regulator; when in the OFF position, flow is prevented.

6 Pressure Gauge and Flow Indicator. A pressure gauge and flow indicator (blinker), see Figure 1-20-1, are located on the oxygen regulator panel. The pressure gauge shows gaseous oxygen pressure in psi in the supply line, whether or not the system is in use. The flow indicator consists of an oblong opening on the face of the regulator panel and shows white and black alternately during the breathing cycle.

OXYGEN QUANTITY INDICATOR

7 An oxygen quantity indicator, see Figure 1-20-1, on the right vertical control panel indicates the quantity of oxygen in the converter and is calibrated from 0 to 5 litres. The indicator is of the capacitance type and accurately records the liquid oxygen supply in the converter. The indicator depends on ac power for its indication.

OXYGEN DURATION AND CONTROLS

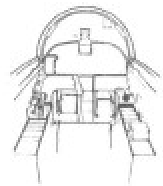
DURATION IN HOURS

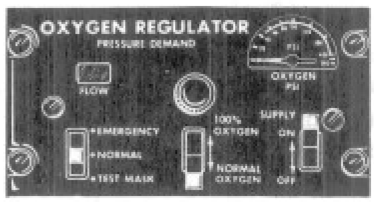
COCKPIT ALTITUDE—FEET							
35,000 & ABOVE	28	22	16	11	5.6	2.8	**EMERGENCY** DESCEND TO ALTITUDE NOT REQUIRING OXYGEN.
	28	22	16	11	5.6	2.8	
30,000	20	16	12	8.1	4.0	2.0	
	20	16	12	8.3	4.1	2.0	
25,000	15	12	9.4	6.2	3.1	1.5	
	19	15	11	7.8	3.9	1.9	
20,000	11	9.5	7.1	4.7	2.3	1.1	
	22	17	13	8.9	4.4	2.2	
15,000	9.5	7.6	5.7	3.8	1.9	0.9	
	27	21	16	10	5.4	2.7	
10,000	7.6	6.1	4.6	3.0	1.5	0.7	
	27	21	16	10	5.4	2.7	
LIQUID CONTENTS—LITERS	5	4	3	2	1	1/2	BELOW 1/2

- TOP FIGURES INDICATE DILUTER LEVER "100% OXYGEN".
- BOTTOM FIGURES INDICATE DILUTER LEVER "NORMAL OXYGEN".

CAUTION LIGHT

INDICATOR





REGULATOR

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Figure 1-20-1

OXYGEN LOW CAUTION LIGHT

8 The oxygen low caution light on the caution light panel, see Figures 1-1-7 and 1-1-8, displays OXYGEN and is powered by the dc bus. The light will come on, together with the master caution light, when the oxygen quantity indicator reads 0.5 litre or less.

NORMAL OPERATION OF OXYGEN SY

9 Normal operation is a

(a) Supply Leve

(b) Di —100% OXYGEN or NOR-MA as required.

Emergency Lever—NORMAL.

DELETED

EMERGENCY OPERATION OF OXY SYSTEM

10 If symptoms of hypo op, or smoke or fumes enter th t, proceed as follows:

(a) Push er lever forward to EMERGEN ion.

(b) o Part 3, Section 2 for emergency ilout bottle.

DELETED

SECTION 21

PHOTOGRAPHIC RECONNAISSANCE SYSTEM

GENERAL

1 The photographic reconnaissance (recon) camera control system (Type CCS-1) in aircraft configured as A/R provides for high-speed and low, medium, and high altitude photographic ground coverage. The camera system is especially adaptable to low altitude oblique photography at high airspeeds. The system consists of a self-contained removable camera nose divided into three tandem camera bays (forward, centre, and aft) by camera mounting brackets. The forward (upper) portion of the camera nose is hinged and is fastened to the aft (lower) portion by latches, permitting ease of access to the camera equipment during ground servicing. Three 70mm film magazine cameras with various lens combinations provide a selection of 11 camera arrangements (arrays) for optimum coverage of any selected target area. The forward camera bay is equipped with a single camera window and can accommodate one camera to provide forward oblique views. The centre camera bay is equipped with two windows and can accommodate one camera to provide either left or right oblique photo coverage. The aft camera bay is equipped with four windows and can accommodate one or two cameras to provide single-oblique, oblique and vertical, or split-vertical photo coverage. The camera nose also contains a reflected light level monitor with optical porthole type window, and defog and cooling ducts. An incident light level monitor is provided behind the pilot seat near the top of the ejection seat rails. A camera control unit, located within the forward equipment compartment, provides correct lens iris aperture and pulse sequencing for each camera. Within the defog and cooling ducting, which passes through the aircraft forward equipment compartment, are the system modulating and overtemperature sensors which control the air temperature within the recon nose. The camera control panel on the left console of the cockpit provides the controls for selection and operation of the cameras. Camera-operate lights on the instrument panel provide a display to indicate camera operation. The recon camera-operate button (nosewheel steering button) on the control stick grip provides optional remote operation of the cameras during flight when the landing gear is up.

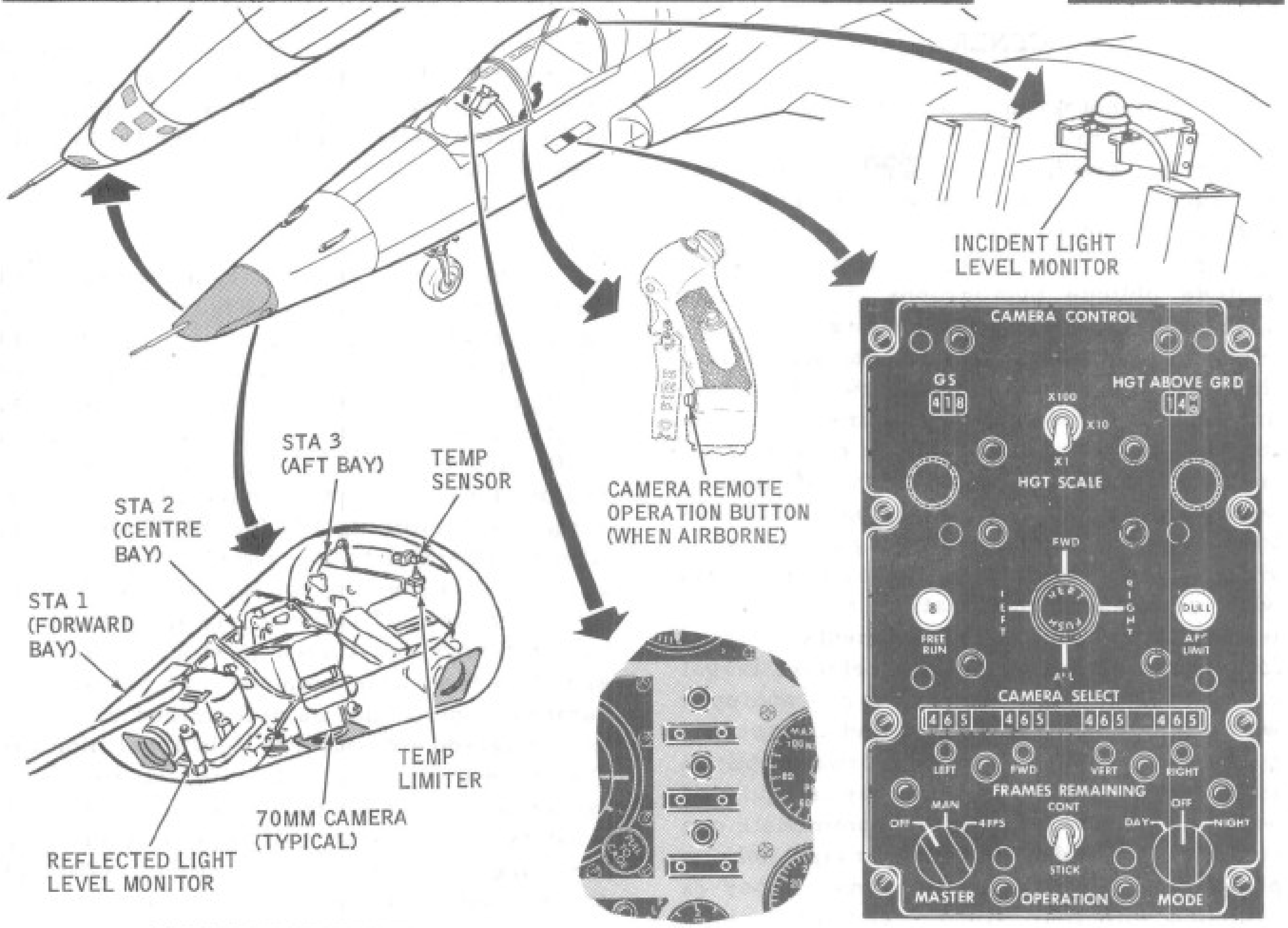
CAMERA

2 The 70mm sequential, pulse-operated still picture camera, see Figure 1-21-1, incorporates two shutter speed settings, 1/1500 and 1/3000 second. The camera can be equipped with one of three available focal length lenses: 1.75-inch (f2.8), 3.0-inch (f2.0), and 6.0-inch (f2.8). Each lens is equipped with an optical haze filter. A film magazine containing 100 feet of 70mm aerial film is attached to the camera body and is capable of providing a total of 500 frames (photographs). The lens iris diaphragm is automatically positioned (f-stop) to correspond to the value computed by the system camera control unit as determined by either the reflected or incident light level monitor Two 50-watt dc-powered heater elements attached to the camera body, controlled by thermostats, are automatically operated between the range of 0° to 20°C to permit efficient camera operation in extreme temperature.

CAMERA CONTROL UNIT (CCU-2)

3 The camera control unit (CCU), see Figure 1-21-1, within the forward equipment compartment is essentially two computers within a single unit. The aperture computer section furnishes the iris control signal to the diaphragm in the camera lens unit as determined by the ground preselected light level monitor setting for the type film rating (ASA exposure index), and furnishes the amber DULL indication to the camera control panel APC (aperture control) limit light. Preselection of either the reflected or incident light monitors is accomplished through the use of a switch provided on the CCU. When night mode is selected on the camera control panel, the camera lens unit(s) are set at wide open and the aperture computer section will cause the control panel APC limit light to illuminate.

RECONNAISSANCE CAMERA SYSTEM A/R



CAMERA ARRANGEMENT (NO. 3 SHOWN)

CAMERA OPERATE LIGHTS AND DATA CARD HOLDERS

CAMERA CONTROL PANEL

CAMERA ARRANGEMENT	STATION 1		STATION 2			STATION 3		
	FOCAL LENGTH — INCHES	DEPRESSION ANGLE — DEGREES	LH RH	FOCAL LENGTH — INCHES	DEPRESSION ANGLE — DEGREES	LH RH	FOCAL LENGTH — INCHES	DEPRESSION ANGLE — DEGREES
1	NONE	NONE	LH	3	56	RH LH	3 3	56 90
2	3	20	RH	3	15	LH	3	15
3	3 OR 6	20	LH	3	73	RH	3	73
4	3 OR 6	20	LH	1.75	65	RH	1.75	65
5	3 OR 6	20		NONE	NONE	LH RH	6 6	83 83
6	NONE	NONE		NONE	NONE	LH	3	90
7	NONE	NONE	LH	3	45		NONE	NONE
8	3 OR 6	20		ANY	ANY	RH	6	90
9	3 OR 6	20		ANY	ANY	RH	1.75	90
10	3 OR 6	20	RH	1.75	27	LH	1.75	27
11	3 OR 6	20	RH	6	5	LH	6	5

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Figure 1-21-1

4 The frame rate computer section computes the basic pulse rate for each of the selected cameras for optimum usage of each roll of film. Data inputs are groundspeed and height above ground which are manually entered in the digital readout windows on the camera control panel. Groundset inputs (FOA) are preselected into each channel of the computer and are based on mathematical formula computations for each camera position and desired photo overlap. The computed pulse rate is variable over a range from 6 pulses-per-second (pps) to 1 pulse every 30 seconds. If the computed pulse rate for any channel exceeds 6 pps, the computer will automatically provide a "free run" (8 pps) mode steady state signal as required by the associated camera. In addition, a blue FREE RUN indicator light with the numeral 8 showing will illuminate on the camera control panel.

LIGHT LEVEL MONITORS

5 The reflected light level monitor, located within the camera nose, and the incident light level monitor, located behind the ejection seat at the upper portion of the aft electronics compartment, see Figure 1-21-1, provide a continuous output voltage representative of light intensity to the aperture computer for control of the camera lens iris diaphragm. The reflected light level monitor averages existing light intensity and terrain reflectivity. The incident light level monitor measures proportionate light intensity of surrounding or incident light rays. Each monitor incorporates an adjustable iris in the front of the light sensor calibrated to accommodate a film exposure index number which is preset at the time of film magazine loading.

CAMERA CONTROL (MCA-3)

6 The camera control panel on the left console of A/R equipped aircraft, see Figure 1-21-1, provides the pilot with the following controls for camera operation: master switch, mode select switch, camera select switch, operation switch, frames remaining selector controls and frames remaining indicators for each camera position, velocity set control, height scale factor switch and height control. In addition, the control panel incorporates the APC limit light and a free run light. The functions of each control are as follows:

(a) Master Switch. The master switch is a three-position rotary switch with the following selections:

(1) OFF.

(2) MAN – Upon selection of the manual position a camera shutter speed of 1/3000 second is selected, and power is applied to the camera motor drive field coil and the camera control unit computer. The pulse rate computation in the computer will utilize pilot induced inputs of groundspeed and height above ground (absolute altitude).

(3) 4 FPS – Selection of the four frames-per-second position selects a camera shutter speed of 1/1500 second and the power supply is the same as for MAN position except that the pulse rate computation is switched off and the selected camera(s) will operate in a four-frames-per-second free running condition.

(b) Mode Select Switch. The mode select switch is a three-position rotary switch with the following selections:

(1) DAY – The system provides outputs to position each lens iris in either the bright, hazy, or dull aperture as computed by the CCU.

(2) OFF – The aperture computation is automatically switched off and the lens iris of each camera is automatically positioned in the fully open (night) aperture.

(3) NIGHT – The aperture computation remains off and each lens iris is positioned fully open (night). Camera synchronization pulse and the night illumination system (inoperative) signals are applied throughout the system.

(c) Camera Select Switch. The camera select switch, placarded CAMERA SELECT controls selection of camera positions by means of a "joystick" type switch. Placing the joystick in the following positions will select the appropriate camera operation.

(1) Centre – No camera selection.

(2) LEFT – Left facing camera (left oblique)

(3) RIGHT – Right facing camera (right oblique)

(4) FWD – Forward facing camera (forward oblique)

(5) ALL – All camera positions.

(6) VERT PUSH (stick depressed) – Camera mounted in vertical position is selected.

(d) Operation Switch. The operation switch is a two-position toggle switch placarded OPERATION. The two placarded positions operate as follows:

(1) STICK – In this position the cameras selected by the positions of the camera select switch operate only when the recon camera button (nosewheel steering button) on the control stick grip is pressed, with the landing gear in the up position.

(2) CONT – In continuous, the system will operate selected cameras until manually turned off (joystick centred and pulled out).

(e) Frames Remaining Reset Selectors. The frames remaining selector controls are grouped within the section of the camera control panel placarded FRAMES REMAINING. The selector controls consist of four counter-reset controls (one for each camera position: LEFT, FWD, VERT, and RIGHT) which are used in conjunction with setting the number of frames remaining in the digital readout windows for each position at the time the cameras are loaded with film.

(f) Frames Remaining Indicators. The number of frames remaining in each camera is indicated on a three-drum counter on the master camera control. Camera position controls on the CCU (computer) determine which of camera 1, camera 2, or camera 3 shall correspond to forward, vertical, left, and right. When SPLIT/TRI is selected, the applicable counter is the VERT frames remaining indicator.

(g) Velocity Set Control. The velocity set control is the knurled knob below the digital readout window placarded GS (groundspeed). Groundspeed is entered into the system by manually rotating the knob left (decrease); right (increase). The numerical value of groundspeed (range is 100 to 600 knots) will appear in the window.

(h) Height Set Control. The height set control is the knurled knob below the digital readout window placarded HGT ABOVE GRD (height above ground). Absolute altitude is entered into the system by manually rotating the knob left (decrease); right (increase). This control operates in conjunction with the height scale factor switch described below.

(j) Height Scale Factor Switch. The height scale factor switch, placarded HGT SCALE, is a three-position toggle switch. The switch is operated in conjunction with the height set control described above and provides the pilot the capability of setting height in one of three ranges. The switch selections are:

(1) X1 - Represents range from 100 to 1000 feet.

(2) X10 – Represents range from 1000 to 10,000 feet.

(3) X100 – Represents range from 10,000 to 100,000 feet.

- EXAMPLE -

HGT ABOVE GRD	HGT SCALE	ABSOLUTE ALTITUDE
145	X1	145 Feet
145	X10	1450 Feet
145	X100	14,500 Feet

7 Operation of the blue FREE RUN (8) light and the amber APC LIMIT (DULL) light were covered in the preceding paragraphs. To amplify the operation of the lights, the following applies:

(a) APC Limit Light. The word DULL will appear when the light is illuminated, indicating that the ambient light intensity has dropped beyond the lower level of the dull exposure range of the film. During DAY operation, the picture will not be effective. During night operation with the camera mode set at NIGHT, the light will illuminate; however, photoflash illumination of the target will produce effective pictures commensurate with film rating, lens aperture, and target light intensity.

(b) Free Run Light. The camera free run light with the numeral 8 is illuminated when one or more of the three channels in the pulse rate computation supplies a free running mode signal to its associated camera.

CAMERA OPERATE LIGHTS

8 Three recon camera operate lights on the instrument panel, see Figure 1-21-1, permit visual indication of camera operation without diverting attention to the frames-remaining counters on the camera control panel. The three blue lights are arranged vertically and represent camera position as installed in the camera nose. Beneath each light a data card holder is provided to accommodate insertion of camera data at the time the recon cameras are installed. The respective camera operate lights will illuminate as each camera is operated.

CAMERA AREA COVERAGE

9 Ground coverage for each of the eleven camera arrangements is graphically illustrated in Figure 1-21-2, Sheets 1 through 11. The camera arrangement number and position in the recon nose compartment, and fields of view are included in each illustration. Refer to Figure 1-21-1 for camera lens and depression angle installations.

DEFINITIONS

10 The following definitions are listed for convenience and interpretation:

(a) ABSOLUTE ALTITUDE – actual altitude over terrain (or water).

(b) ARRAY – two or more camera grouped together.

(c) NADIR – a point on the terrain directly beneath the aircraft and 180 degrees directly opposite the zenith.

(d) OBLIQUE – camera axis between horizon and vertical. A high oblique will include the horizon in camera field of view. A low oblique will not include the horizon in field of view. Oblique views permit photo of target without flying directly over the target.

(e) SPLIT-TRIMETROGON – actually can be either a split vertical or a trimetrogon arrangement as defined below:

(1) Split Vertical – two cameras, each inclined slightly to the left and right of vertical and covering a narrow band of terrain in overlap. Covers nearly twice the area covered by a single camera.

(2) Trimetrogon – overlapping lateral coverage from left to right using two oblique and one vertical camera arrangement.

(f) PULSE-OPERATED CAMERA – A preselected electrical interval impulse which snaps the shutter (exposes film) and advances film to the next frame.

CAMERA NOSE COMPARTMENT AIR-CONDITIONING SYSTEM

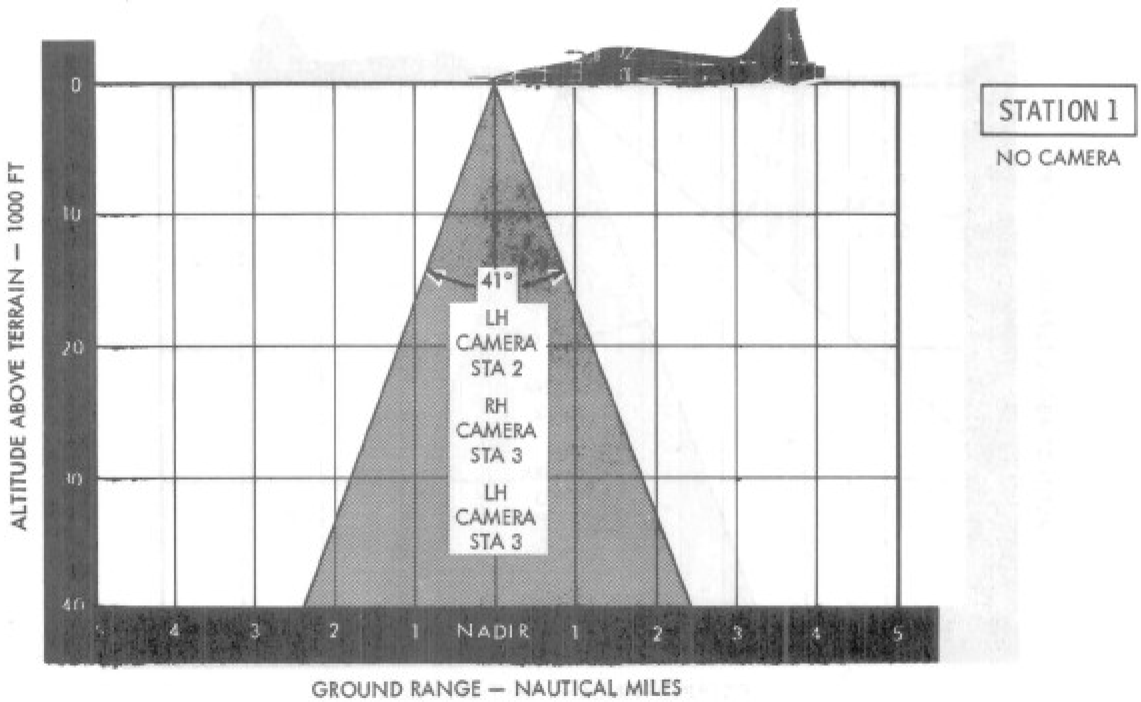
11 The camera nose compartment air-conditioning system controls the temperature within the camera nose and directs the flow of heated air used to defog the optical camera windows. The defog ducts and outlets in the camera nose are connected to the forward fuselage ducting when the camera nose section is attached to the airframe. When the camera nose section is removed from A/R configured aircraft, the forward end of the defog ducting within the forward fuselage is capped off. Hot air from the eighth stage compressor section of each engine is cooled by cold ram air in a heat exchanger and mixed with bypassed hot air, the amount of which is controlled by a modulating valve. A pressure regulating valve within the system regulates the air pressure at approximately 20 psig. The air then passes through a flow control venturi and then into the defog ducts and window outlets of the camera nose. Three camera windows out of a total of seven are

defogged at any one time, as selected by the particular camera arrangement when the cameras are loaded before flight. A minimum of three nozzles are opened even if only one camera is installed in order to prevent overpressurization within the ducting. An additional duct along the top centreline of the nose carries a portion of the cabin discharge air into the forward portion of the camera nose compartment. An air temperature sensor probe and an air temperature limiter probe are located on the aft bulkhead of the camera nose compartment. The temperature sensor controls the position of the modulating valve and thus adjusts the defog air temperature to control the mixed air temperature in the nose. The temperature limiter senses compartment overtemperature and shuts off the defog air when the temperature exceeds 110°F. Compartment cooling is provided by the cabin discharge air, which is on continuously. Compartment temperature is maintained between 50° and 90°F, except for occasional transients to 120°F on an extremely hot day during low level photo missions. System operation is entirely automatic when the two-position camera defog switch on the cockpit right vertical control panel is placed in the CAMERA DEFOG position, see Figure 1-21-3.

CAMERA AREA COVERAGE

ARRANGEMENT NO. 1

A/R



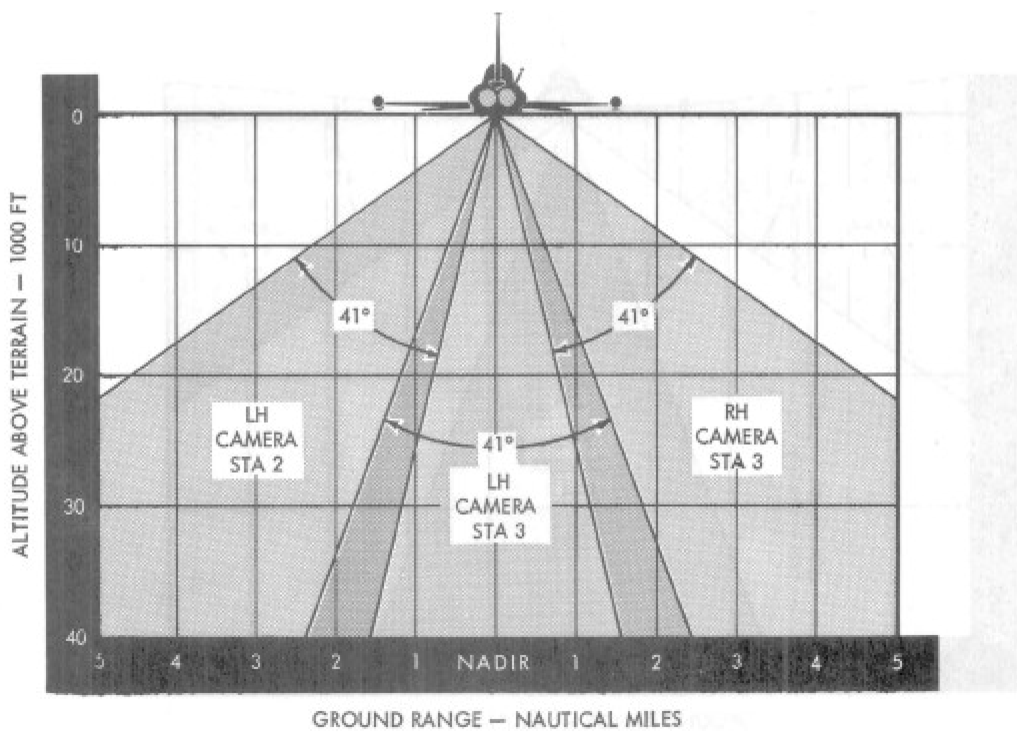


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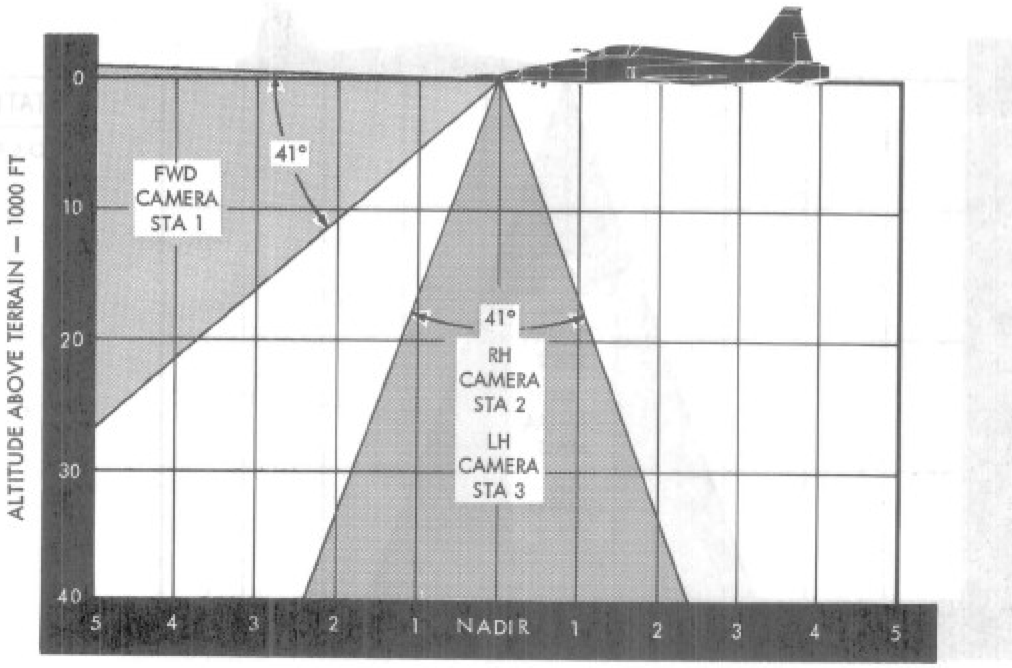
Figure 1-21-2 (Sheet 1 of 11)

CAMERA AREA COVERAGE

ARRANGEMENT NO. 2 A/R



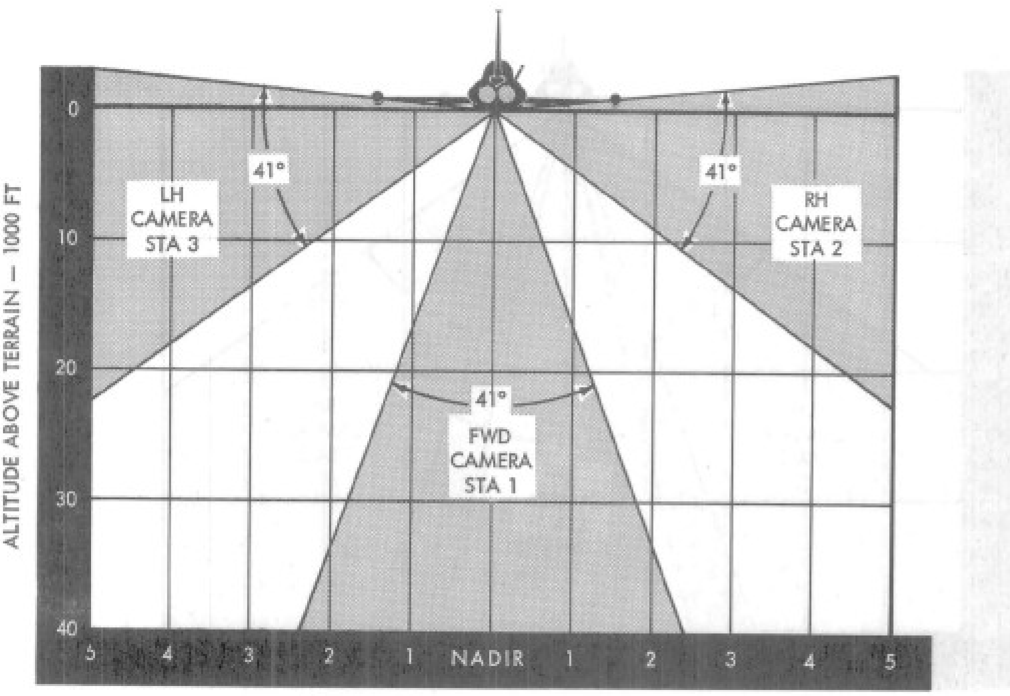


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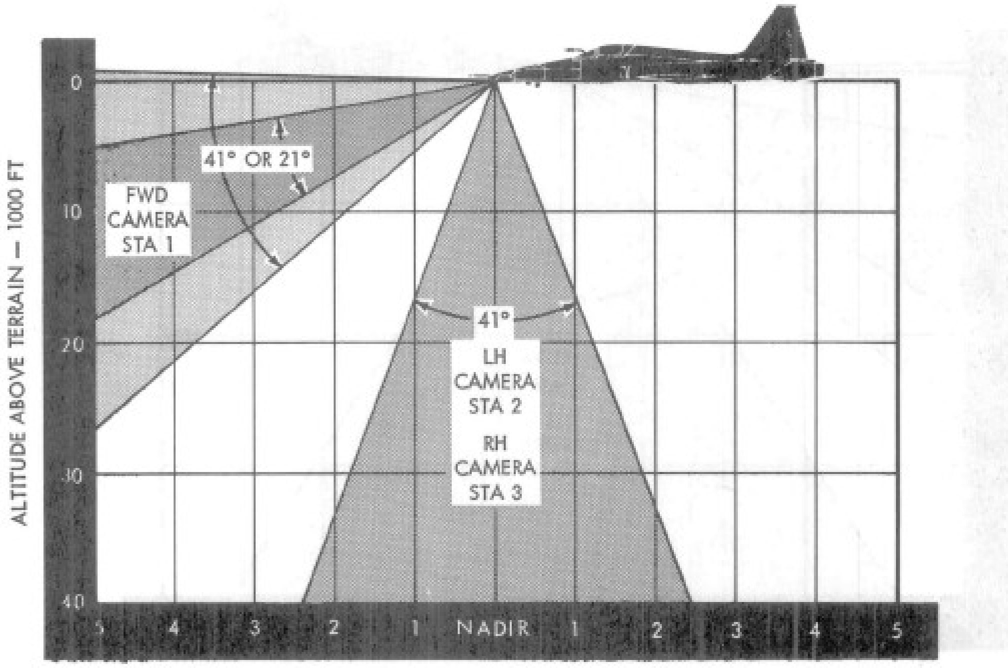
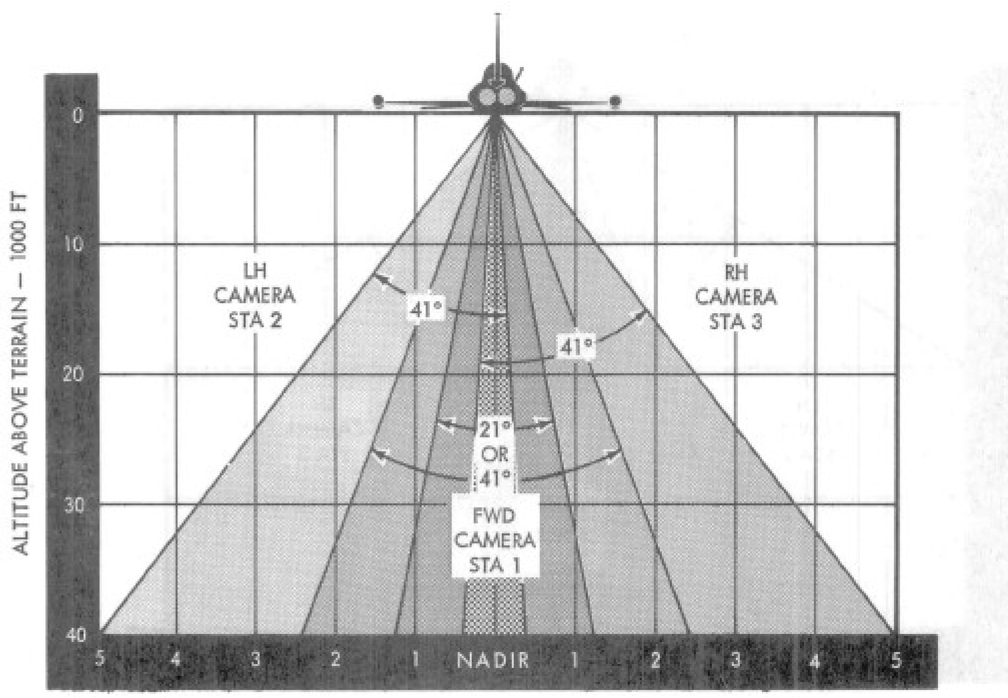
Figure 1-21-2 (Sheet 2 of 11)

CAMERA AREA COVERAGE ARRANGEMENT NO. 3 A/R



GROUND RANGE — NAUTICAL MILES



GROUND RANGE — NAUTICAL MILES

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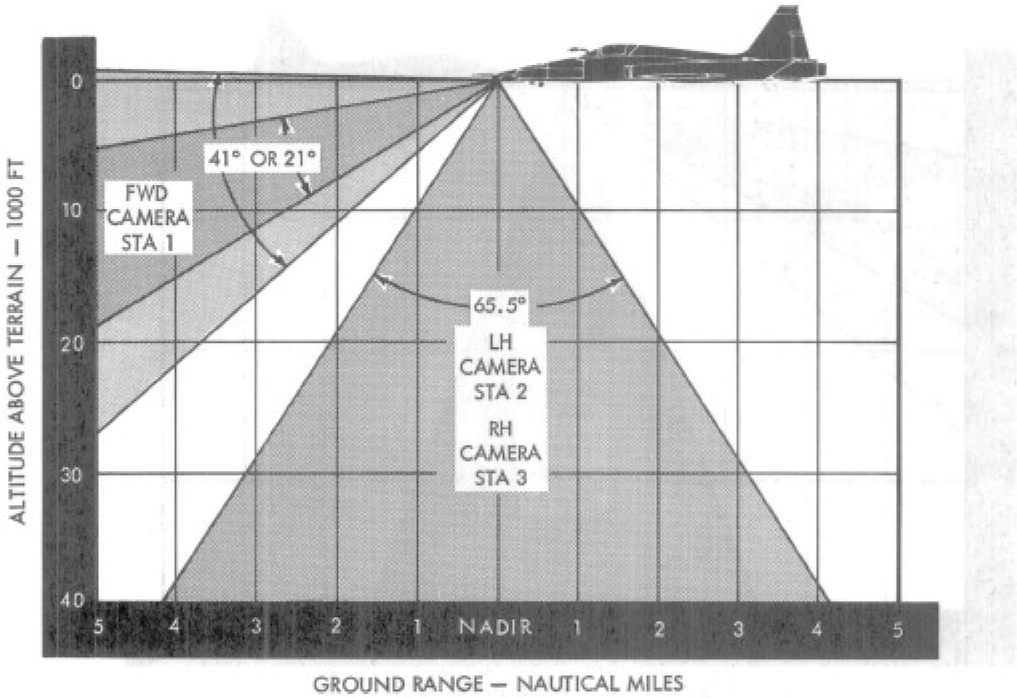
Figure 1-21-2 (Sheet 3 of 11)

CAMERA AREA COVERAGE

ARRANGEMENT NO. 4 A/R



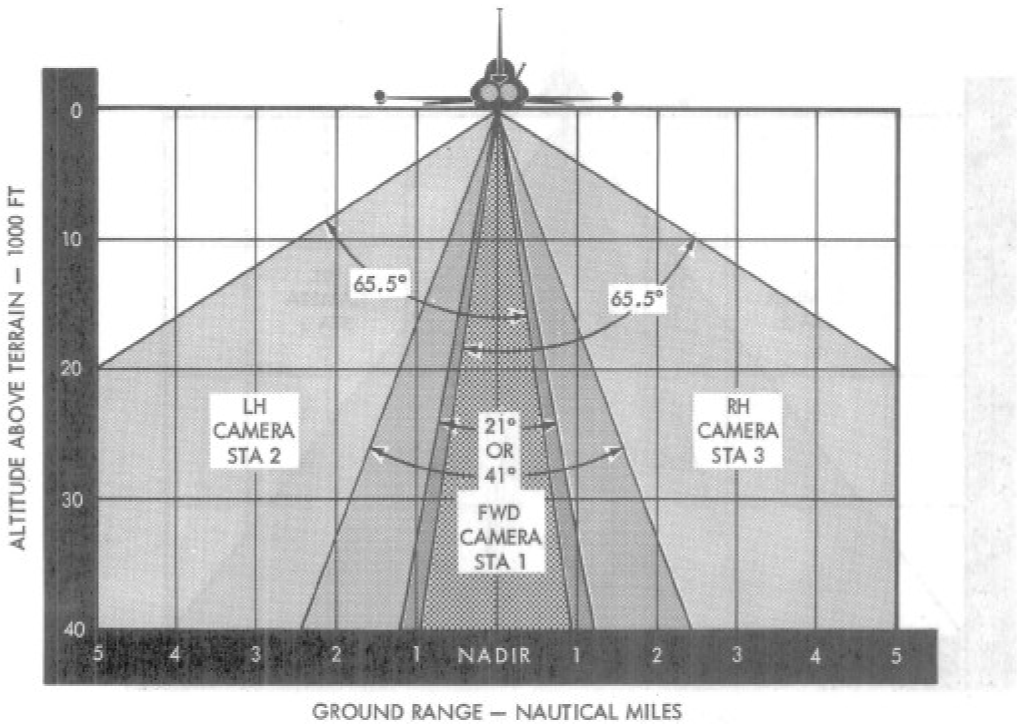


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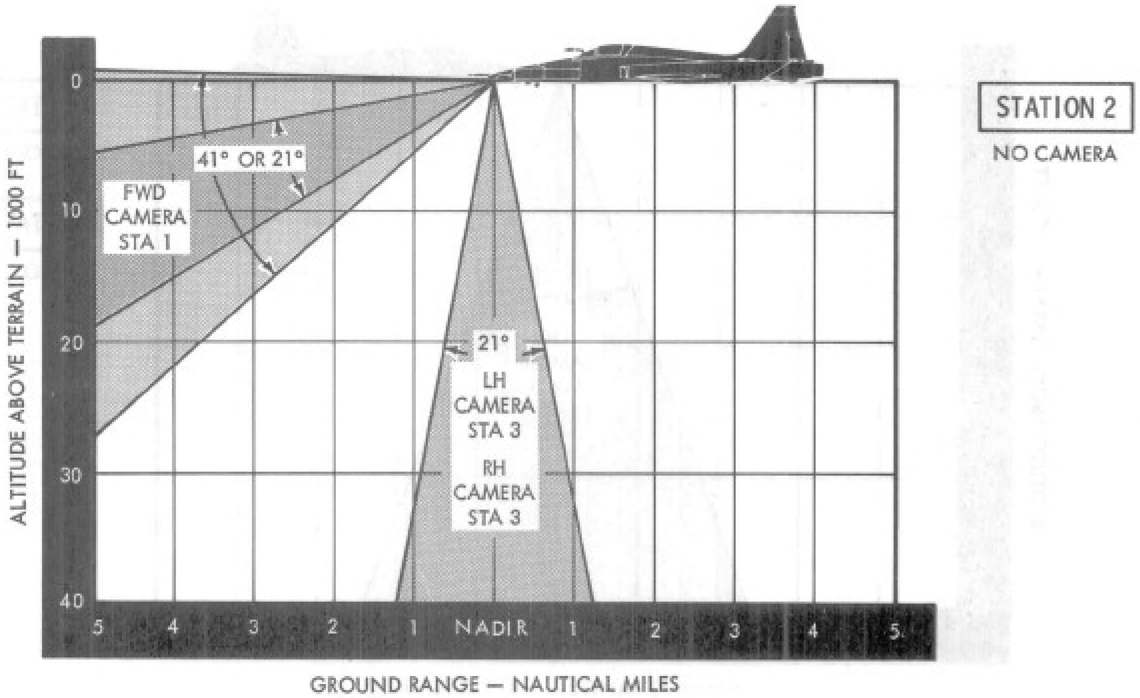
Figure 1-21-2 (Sheet 4 of 11)

CAMERA AREA COVERAGE ARRANGEMENT NO. 5 A/R



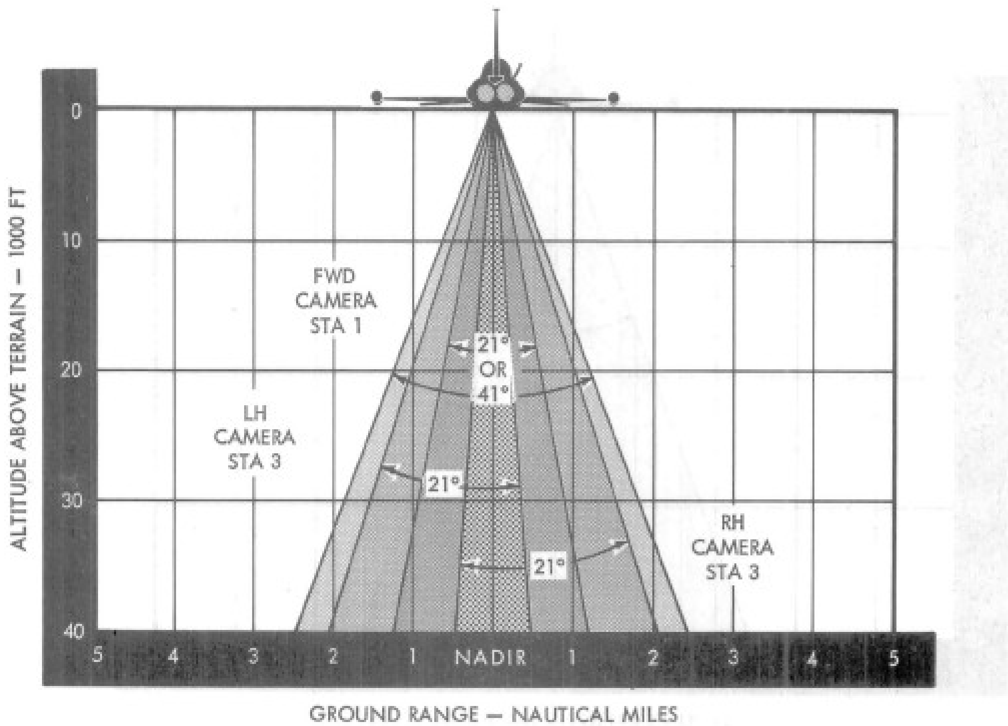


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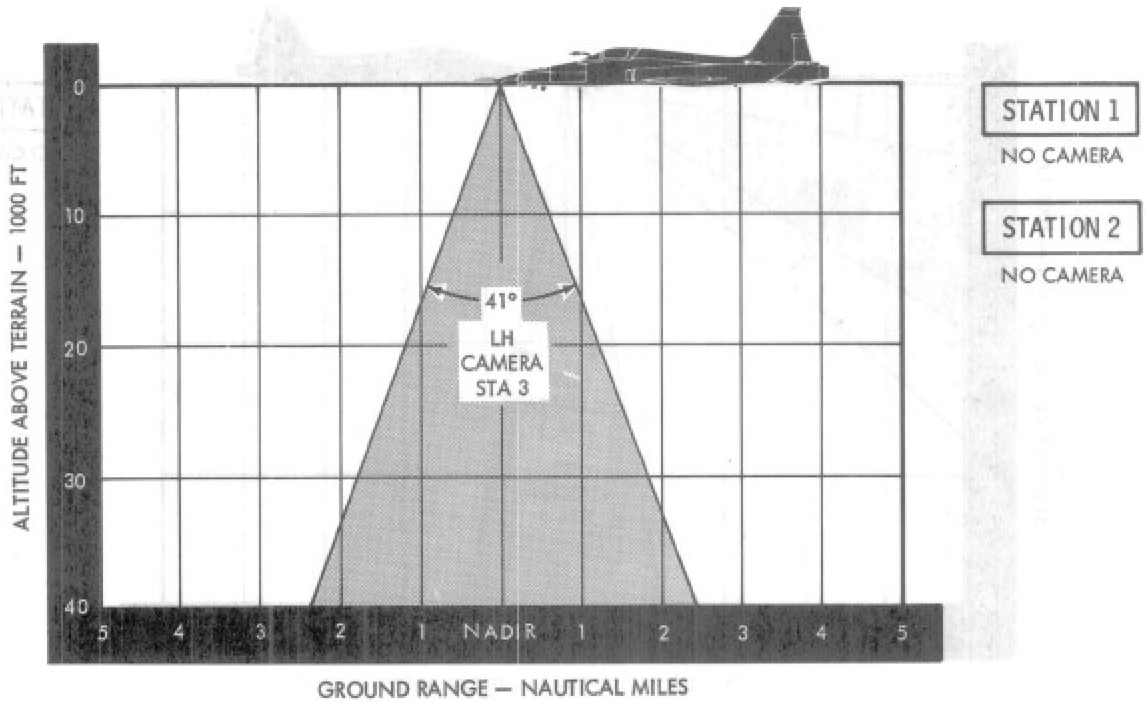
Figure 1-21-2 (Sheet 5 of 11)

CAMERA AREA COVERAGE

ARRANGEMENT NO. 6 **A/R**



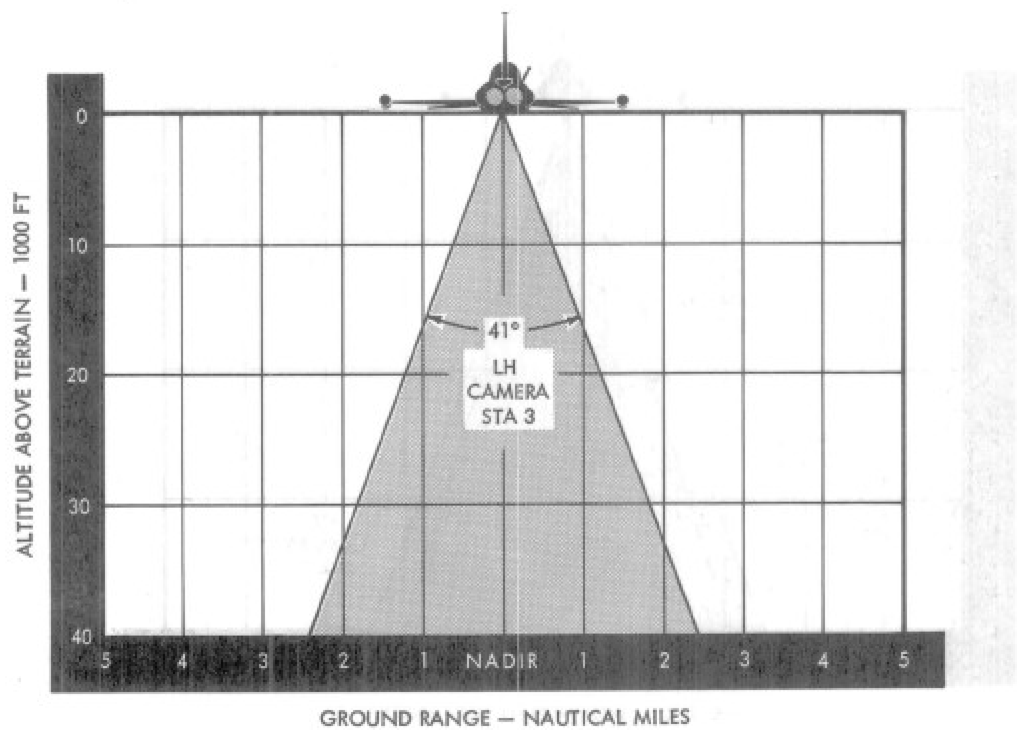


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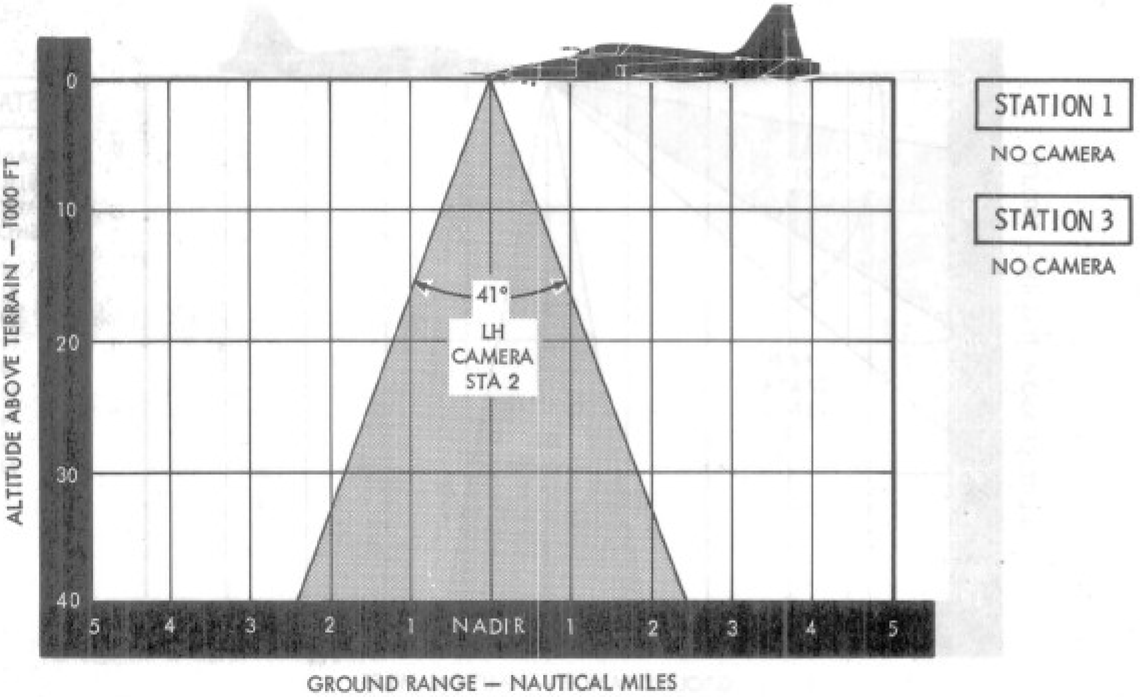
Figure 1-21-2 (Sheet 6 of 11)

CAMERA AREA COVERAGE ARRANGEMENT NO. 7 A/R



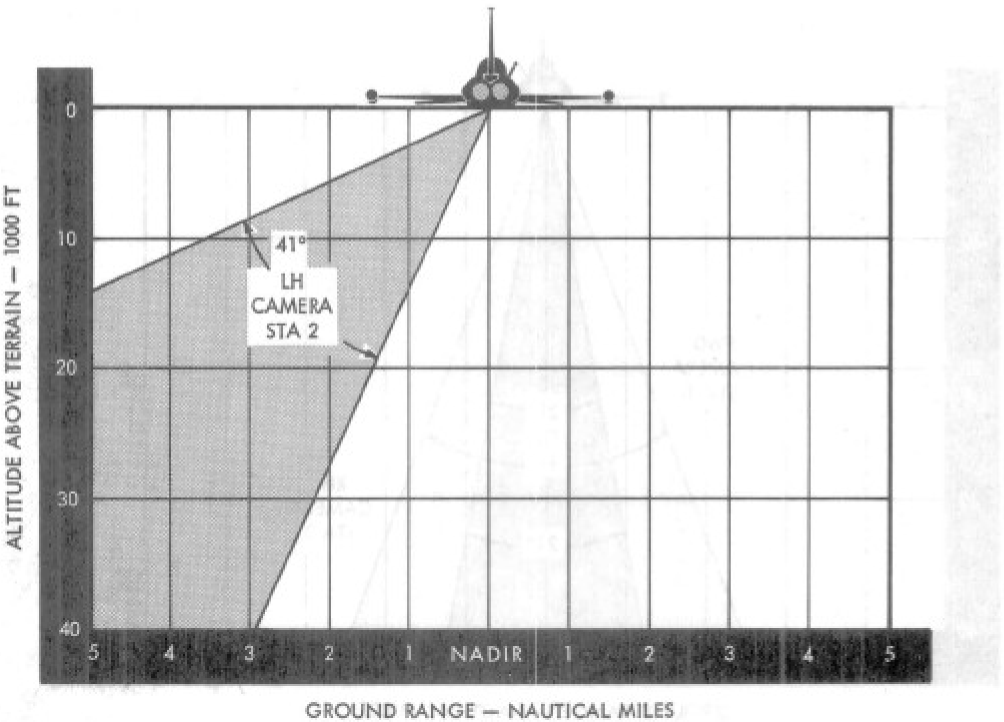
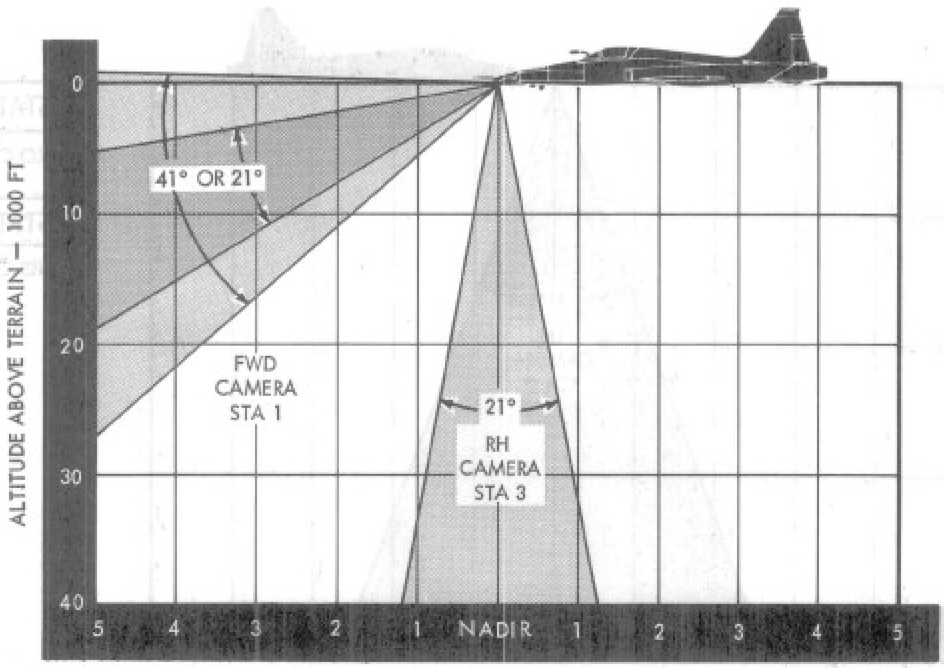


Figure 1-21-2 (Sheet 7 of 11)

CAMERA AREA COVERAGE ARRANGEMENT NO. 8 A/R



STATION 2

SEE ANY DESIRED SIMILAR STATION ARRANGEMENT AS SHOWN IN:

NO. 1
NO. 2
NO. 3
NO. 4
NO. 7
NO. 8
NO. 9
NO. 10
NO. 11

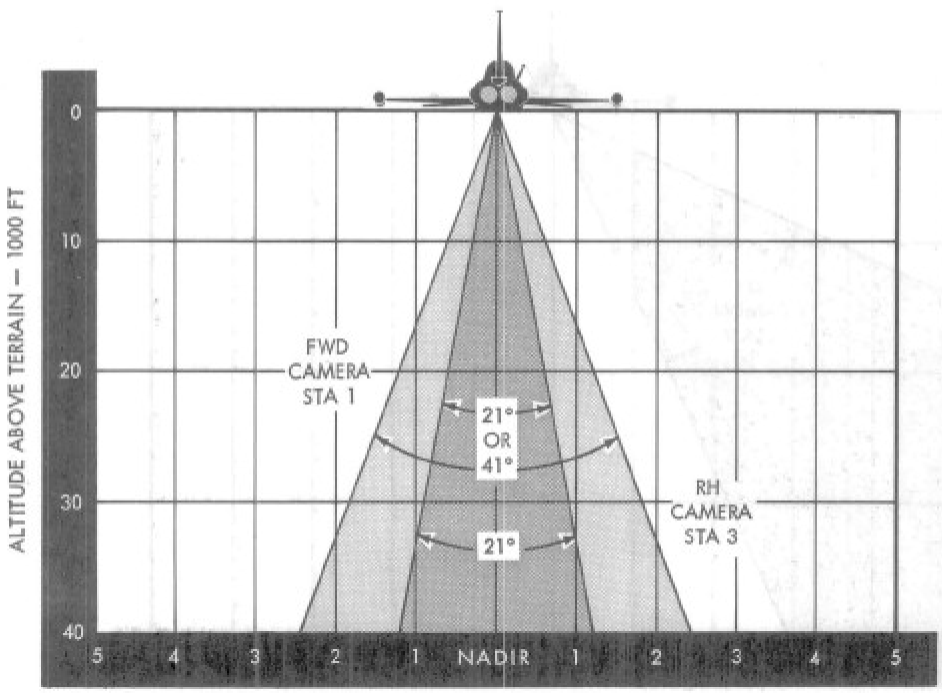


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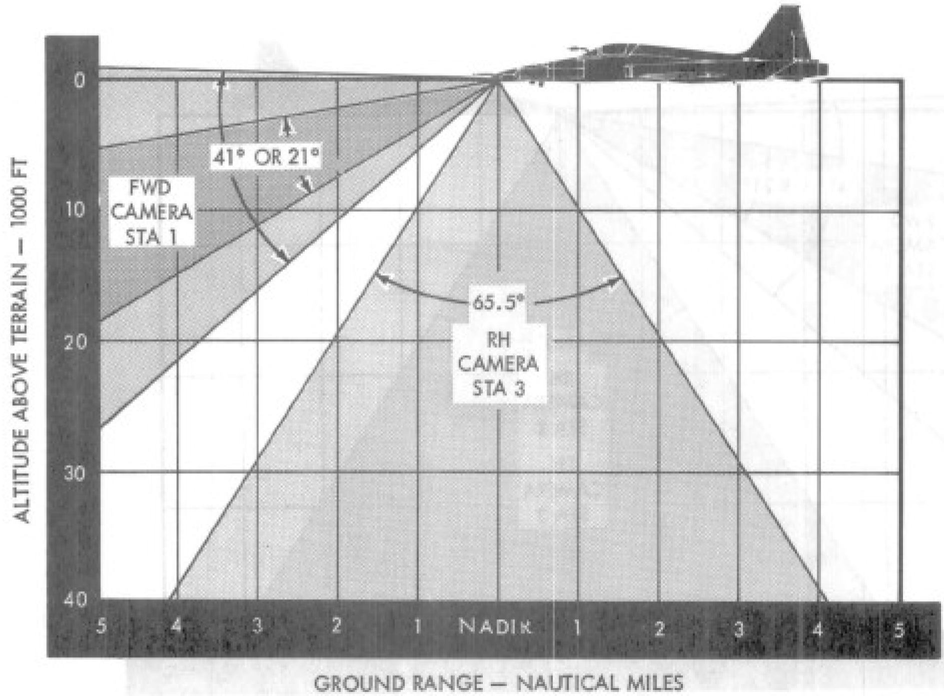
Figure 1-21-2 (Sheet 8 of 11)

CAMERA AREA COVERAGE

ARRANGEMENT NO. 9

A/R



STATION 2

SEE ANY DESIRED SIMILAR STATION ARRANGEMENT AS SHOWN IN:

NO. 1
NO. 2
NO. 3
NO. 4
NO. 7
NO. 8
NO. 9
NO. 10
NO. 11

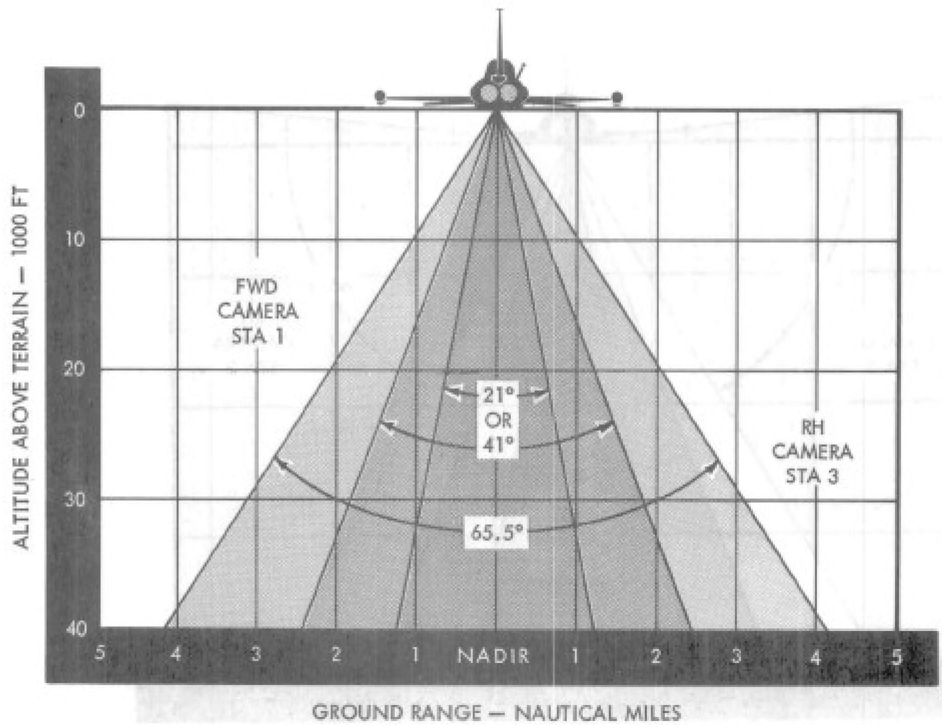


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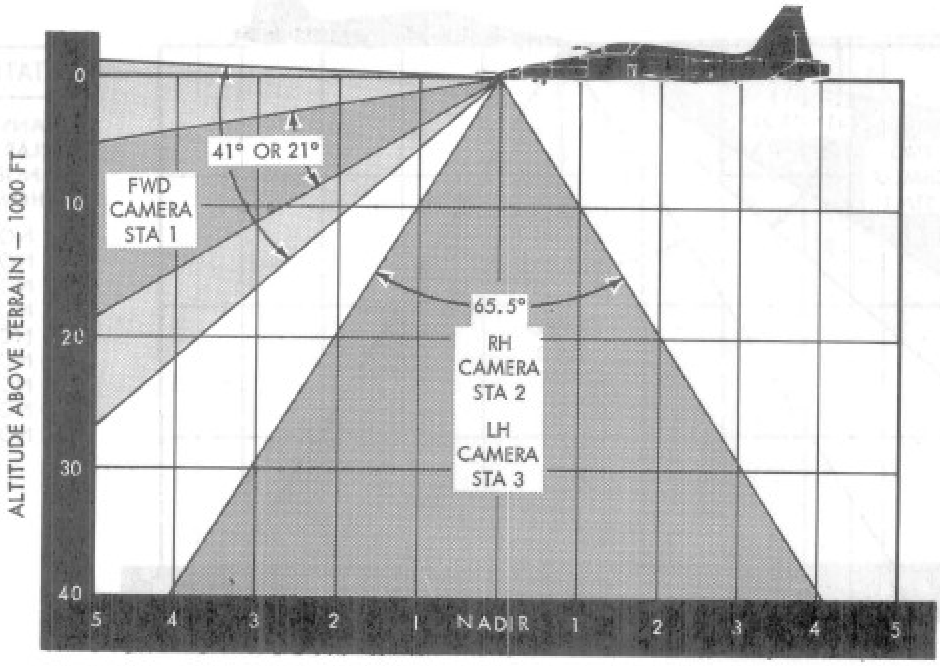
Figure 1-21-2 (Sheet 9 of 11)

CAMERA AREA COVERAGE

ARRANGEMENT NO.10 **A/R**



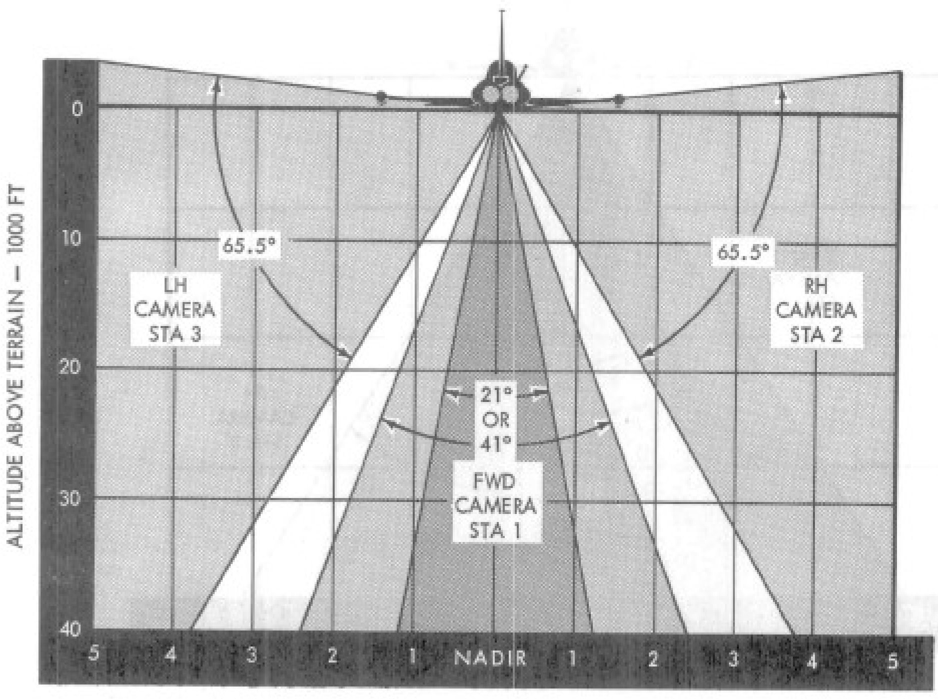


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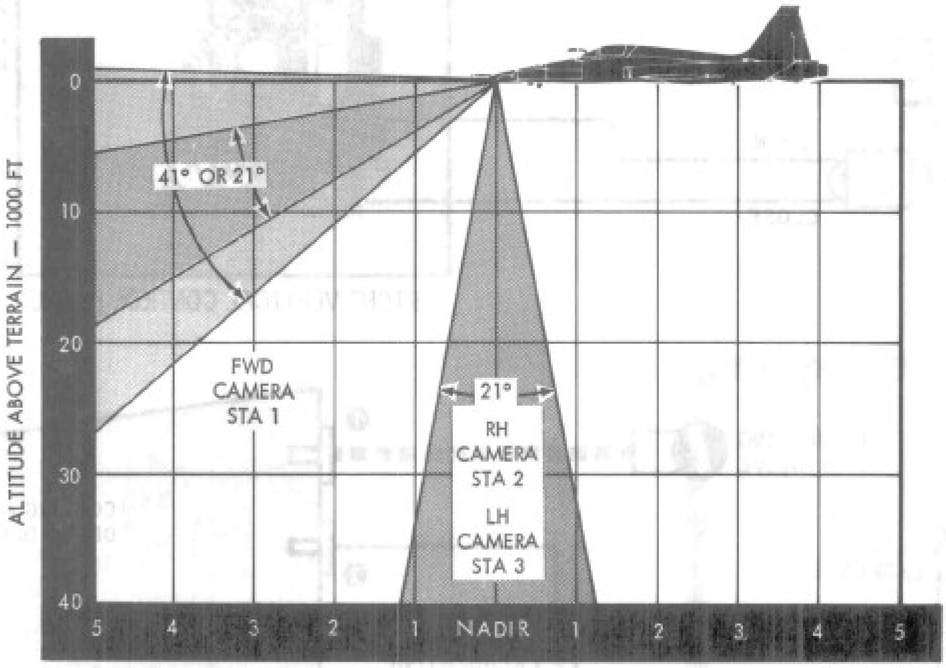
Figure 1-21-2 (Sheet 10 of 11)

CAMERA AREA COVERAGE

ARRANGEMENT NO. 11

A/R



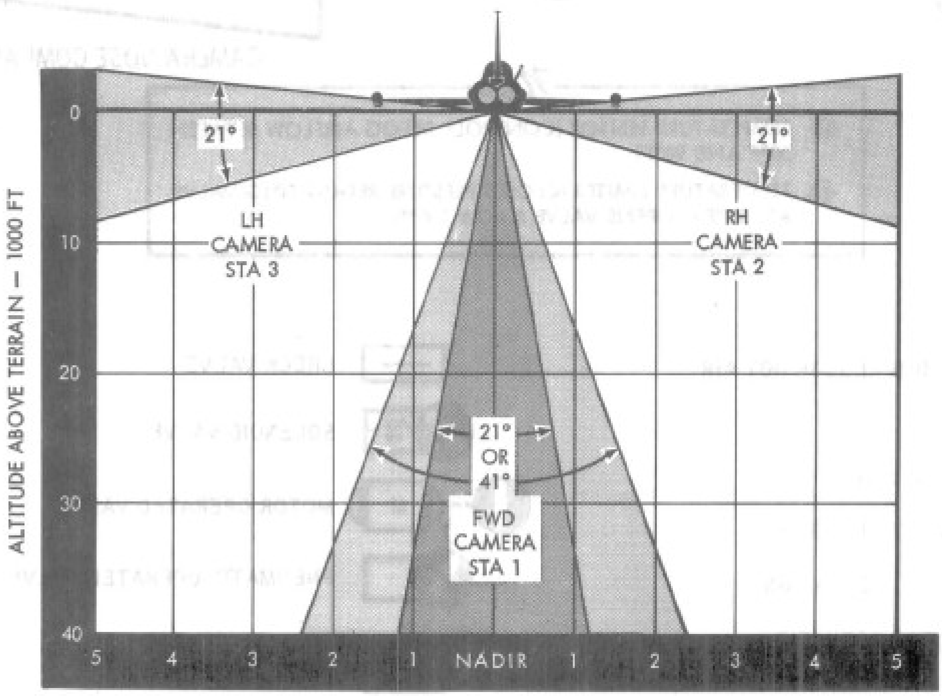


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Figure 1-21-2 (Sheet 11 of 11)

CAMERA COMPARTMENT AIR-CONDITIONING SYSTEM A/R

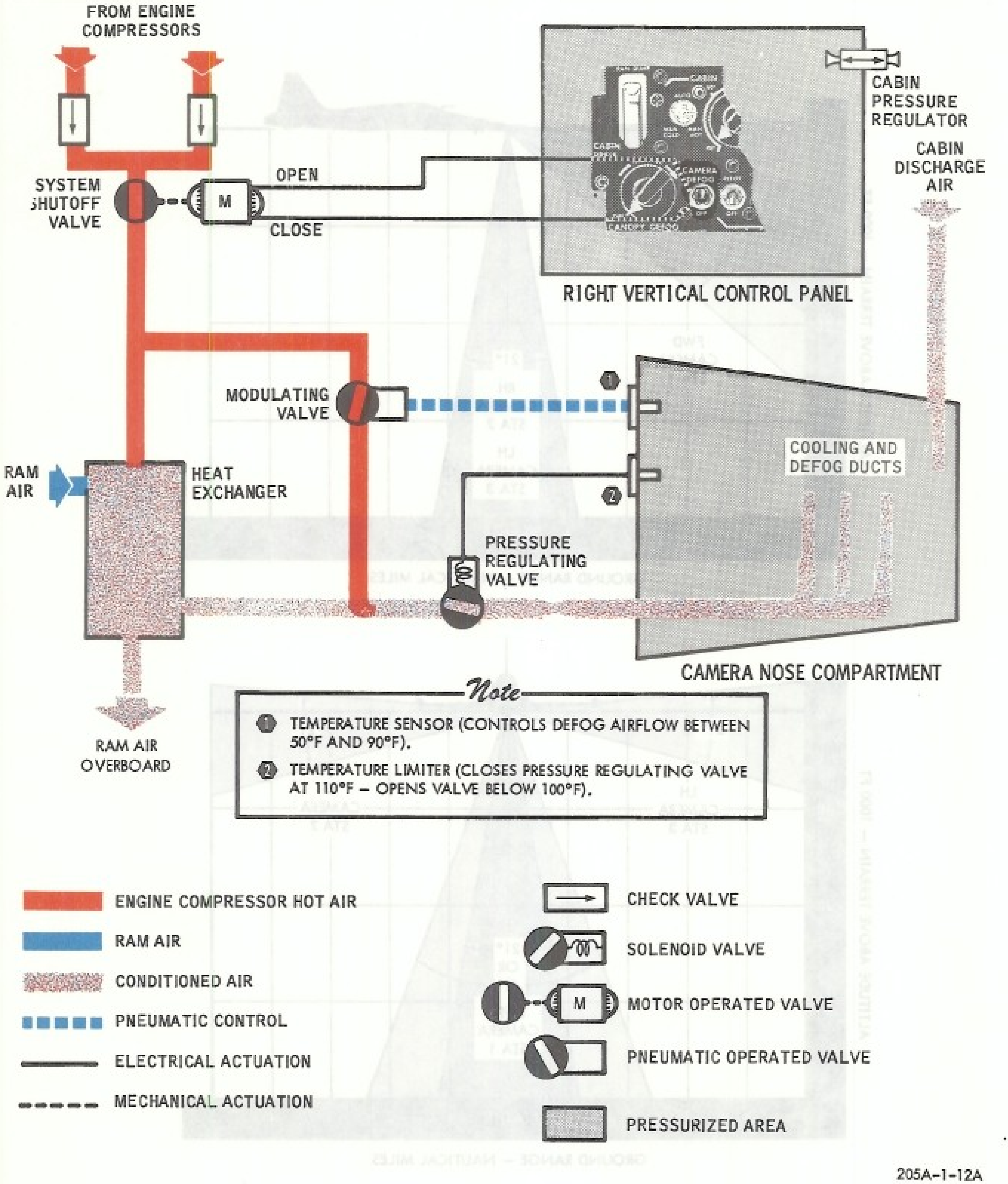

Figure 1-21-3

SECTION 22

MISCELLANEOUS EQUIPMENT

REAR VIEW MIRRORS

1 Two adjustable side-mounted rear vision mirrors are installed on the cockpit windshield frame, see Figures 1-1-3 and 1-1-4.

MAP CASE

2 A map case is provided on the right console panel, see Figures 1-1-7 and 1-1-8.

STOWAGE BOX

3 A stowage box for storage of ground safety pins is provided on the left console panel of (A), see Figure 1-1-7. A stowage box is not provided on A/R .

PART 2

HANDLING

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TABLE OF CONTENTS (Cont'd)

TABLE OF CONTENTS (Cont'd)

PART 2

HANDLING

SECTION 1

NORMAL PROCEDURES

BEFORE EXTERIOR INSPECTION

1 Before starting the exterior inspection, perform the following:

(a) Check Form CF337 for aircraft status.

(b) Check external power off.

(c) Unlock and open canopy.

2 Perform the following cockpit checks:

(a) Battery Switch - OFF.

(b) Seat and Canopy Safety Pins - Installed.

(c) Landing Gear Lever - LG DOWN.

(d) Armament Switches - OFF and SAFE.

(e) EXT. STORES JETTISON Handle - In; Safety Pin - Installed.

(f) Reference Publications - Check.

EXTERIOR INSPECTION

3 See Figure 2-1-1.

AFTER ENTERING AIRCRAFT

4 After entering aircraft, proceed as follows:

WARNING

If any safety wire is broken, do not fly aircraft until cleared by maintenance personnel.

(a) Retractable Step - Stowed.

(b) Lap Belt, Shoulder Harness and Personal Equipment - Fasten.

NOTE

Refer to Figure 2-1-2, Sheets 1 and 2, for proper strap-in procedure.

LEFT CONSOLE

5 Check left console as follows:

(a) Circuit Breakers - In.

A/R (b) Master Camera Control Switch - OFF.

(c) Pitch and Yaw Damper Switches - OFF.

(d) Rudder Trim Knob - Centred.

(e) Wing Flap Lever - UP.

(f) Throttles - OFF.

(g) Speed Brake Switch - Off (centred).

(h) Nose Strut Switch - RETRACT.

(j) Engine Doors Switch - CLOSED.

(k) Aerial Refuel Light Switch - OFF.

(m) Antenna Switch - As Desired.

(n) Emergency UHF Controls - As Required.

LEFT VERTICAL CONTROL PANEL

6 Check left vertical control panel as follows:

(a) Fuel Shutoff Switches - LEFT, RIGHT (Guarded Position).

(b) Voice Recorder Switch - OFF.

EXTERIOR INSPECTION

DURING THE EXTERIOR INSPECTION, THE AIRCRAFT SHOULD BE CHECKED FOR GENERAL CONDITION, WHEELS CHOCKED, ACCESS DOORS AND FILLER CAPS SECURED, GROUND SAFETY GUARDS AND COVERS REMOVED, AND FOR HYDRAULIC, OIL, AND FUEL LEAKS, AS WELL AS FOR THE FOLLOWING SPECIFIC ITEMS:

TAIL SECTION

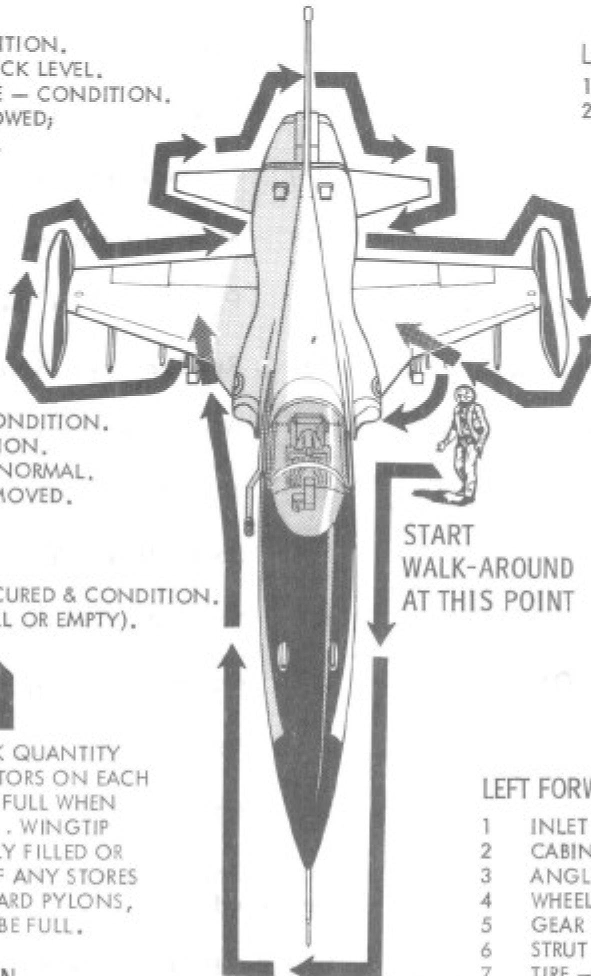
1 EMPENNAGE — CONDITION.
2 TAILPIPES — DENTS, CRACKS, FUEL.
3 DRAG CHUTE COMPARTMENT DOOR — SECURED.

RIGHT AFT SECTION

1 ENGINE DOOR — CONDITION.
2 HYDRAULIC FLUID — CHECK LEVEL.
3 UNDERSIDE OF FUSELAGE — CONDITION.
4 ARRESTING HOOK — STOWED; SAFETY PIN — REMOVED.

LEFT AFT SECTION

1 HYDRAULIC FLUID — CHECK LEVEL.
2 ENGINE DOOR — CONDITION.



LEFT CENTRE SECTION

1 WING — CONDITION.
2 TIP TANK — CHECK (FULL OR EMPTY).
3 EXTERNAL STORES — SECURED AND CONDITION.
4 TIRE — CONDITION.
5 STRUT — EXTENSION.
6 GEAR SAFETY PIN — REMOVED.
7 WHEEL WELL — CONDITION.
8 SPEED BRAKE WELL — CONDITION.

RIGHT CENTRE SECTION

1 SPEED BRAKE WELL — CONDITION.
2 WHEEL WELL — CONDITION.
3 GEAR DOOR SWITCH — NORMAL.
4 GEAR SAFETY PIN — REMOVED.
5 STRUT — EXTENSION.
6 TIRE — CONDITION.
7 WING — CONDITION.
8 EXTERNAL STORES — SECURED & CONDITION.
9 TIP TANK — CHECK (FULL OR EMPTY).

WARNING

CHECK WINGTIP FUEL TANK QUANTITY INDICATORS - TWO INDICATORS ON EACH TIP TANK SHOW THE WORD FULL WHEN TANK COMPLETELY FILLED . WINGTIP TANKS MUST BE COMPLETELY FILLED OR DRAINED BEFORE FLIGHT. IF ANY STORES ARE CARRIED ON THE INBOARD PYLONS, THE WINGTIP TANKS MUST BE FULL.

LEFT FORWARD SECTION

1 INLET DUCT — CLEAR, CONDITION.
2 CABIN PRESSURIZATION STATIC PORT — CLEAR.
3 ANGLE-OF-ATTACK PROBE — CONDITION.
4 WHEEL WELL — CONDITION.
5 GEAR SAFETY PIN — REMOVED.
6 STRUT — EXTENSION.
7 TIRE — CONDITION.
A/R 8 RECON CAMERA NOSE — SECURED AND CONDITION (WHEN CONFIGURED).
9 PITOT TUBE AND STATIC PORTS — CLEAR.

RIGHT FORWARD SECTION

1 AIR REFUELING PROBE (IF INSTALLED) — CONDITION.
2 CABIN PRESSURIZATION STATIC PORT — CLEAR.
3 INLET DUCT — CLEAR, CONDITION.

Note

PYLON AND STORE SAFETY PINS, WHEN INSTALLED, SHOULD BE REMOVED IN ARMING AREA.

205A-1-13A

Figure 2-1-1

(c) Landing and Taxi Light Switch - OFF.

(d) Landing Gear Alternate Release Handle - In.

(e) AIM-9B Volume Control (Inoperative).

(f) Armament Controls and Switches - OFF and SAFE.

(g) External Stores Jettison T-Handle - In; Safety Pin - Installed.

INSTRUMENT PANEL

7 Check instrument panel as follows:

(a) Landing Gear Lever - LG DOWN.

(b) Drag Chute Handle - In.

(c) Airspeed-Mach Indicator - Adjust.

(d) Clock - Set.

(e) Vertical Velocity Indicator - Check Needle at 0.

(f) Altimeter - Set.

(g) Accelerometer - Set.

RIGHT VERTICAL CONTROL PANEL

8 Check right vertical control panel as follows:

(a) Cabin Pressurization Switch - CABIN PRESS (Guarded Position).

(b) Canopy Defog - OFF.

(c) Cabin Temp Switch - AUTO.

(d) Cabin Temp Control Knob - As Required.

(e) Camera Defog Switch - OFF.

(f) Pitot Anti-Ice Switch - OFF.

(g) Engine Anti-Ice Switch - OFF.

(h) Duct Anti-Ice Switch - OFF.

(j) Windshield Anti-Ice Switch - OFF.

(k) External Fuel Switches - OFF.

(m) Boost Pump Switches - LEFT, RIGHT.

(n) Crossfeed Switch - OFF.

(p) Aerial Refuel Switch - OFF.

(q) Canopy Jettison T-Handle - In; Safety Pin - Installed.

(r) Battery Switch - ON.

(s) Generator Switches - ON.

(t) Oxygen - 2 Litres Minimum, refer to Paragraph 9(e), following.

(u) External Power - Connect (if necessary).

(v) Seat (if ac Power on) - Adjust.

RIGHT CONSOLE

9 Check right console as follows:

(a) Oxygen Regulator Controls - NORMAL, NORMAL OXYGEN, ON.

(b) Oxygen System and Pressure Gauge - Check (65 - 110 psi).

(c) IFF/SIF - OFF.

(d) Gyro Controller controls, position as follows:

(1) Fast Align Selector - MAG.

(2) Var/Griv - Local Grivation.

(3) Latitude - Local Latitude.

(4) Grid/True Selector Switch - GRID Position.

(e) Fuel and Oxygen Check Switch - GAGE TEST and QTY CHECK (gauges zero if ac power connected; quantity registers if battery power only available).

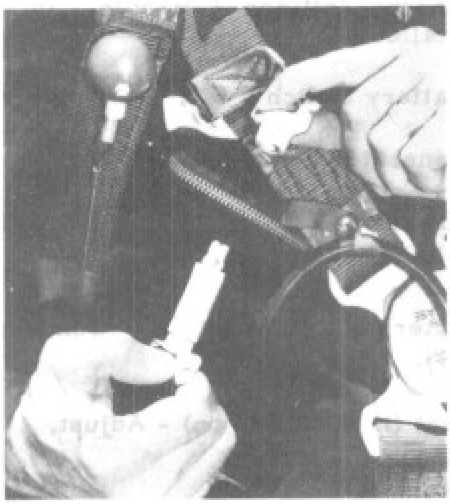
(eA) LF/ADF Controls - OFF.

(f) Cockpit Lighting Controls - As Desired.

STRAP-IN PROCEDURE

WARNING

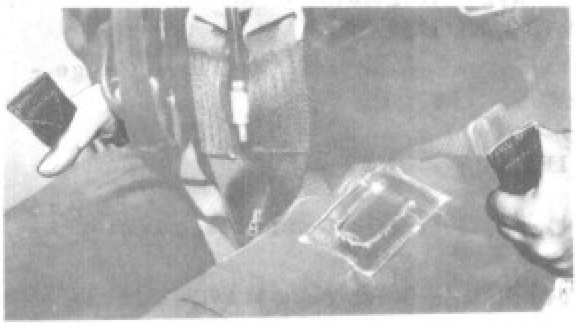
FAILURE TO ATTACH STRAPS IN PROPER SEQUENCE MAY PREVENT SEPARATION FROM EJECTION SEAT AFTER EJECTION.



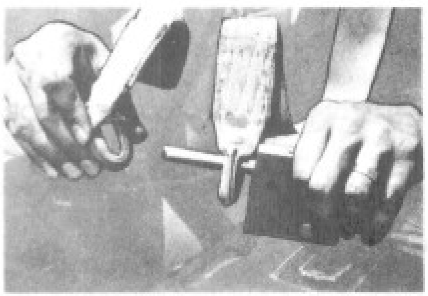
1 SURVIVAL KIT LANYARD – ATTACH TO PARACHUTE HARNESS OR LIFE JACKET (IF WORN).

2 SURVIVAL KIT TO LOWER RIGHT PARACHUTE HARNESS – ATTACH.

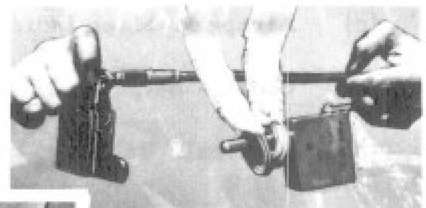
3 SURVIVAL KIT TO LOWER LEFT PARACHUTE HARNESS – ATTACH.



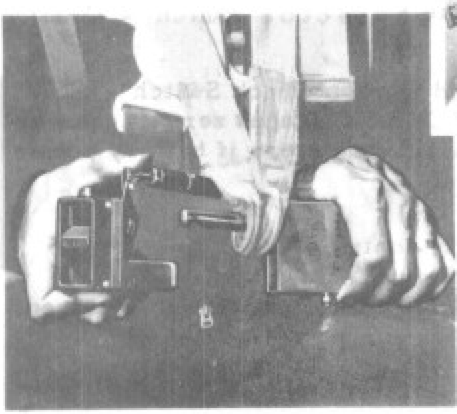
4 SURVIVAL KIT WEB STRAPS – TIGHTEN.



5 LEFT AND RIGHT SHOULDER HARNESS STRAP LOOPS – FASTEN ON LAP BELT.



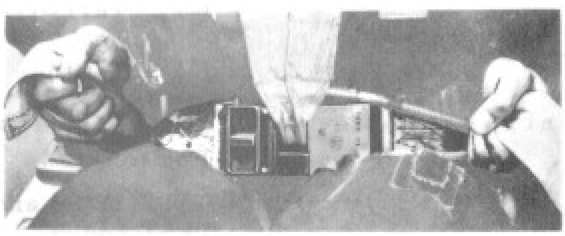
6 PARACHUTE ARMING CABLE KEY – ROUTE UNDER SHOULDER HARNESS STRAPS AND INSERT INTO KEY SLOT OF RIGHT LAP BELT.



7 LAP BELT – FASTEN.

Note

PRESS LAP BELT FITTINGS TOGETHER UNTIL CLICK IS HEARD.



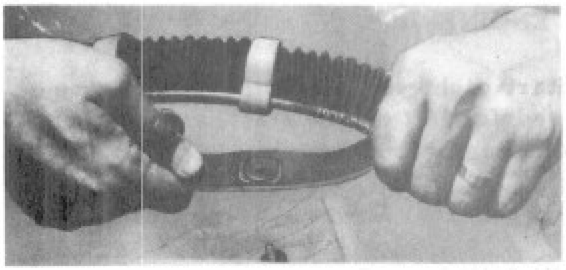
8 LAP BELT – TIGHTEN.

9 SHOULDER HARNESS – ADJUST UNTIL YOKE TOUCHES BACK OF NECK.

CF-5AD 1-36(1)A

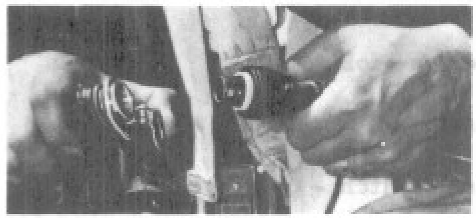
Figure 2-1-2 (Sheet 1 of 2)

STRAP-IN PROCEDURE (CONTINUED)

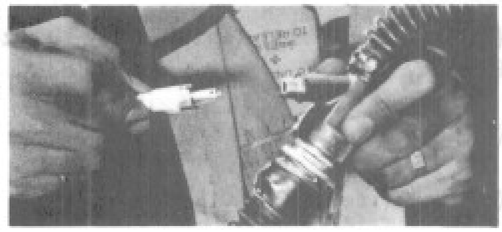


⑩ OXYGEN HOSE — ATTACH TO LAP BELT DOT FASTENER.

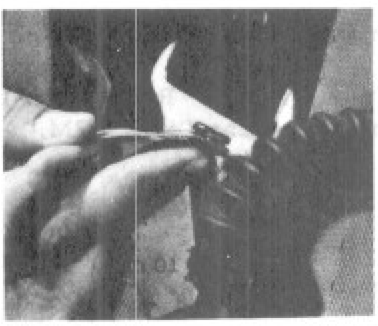
⑪ ANTI-G SUIT HOSE — CONNECT.



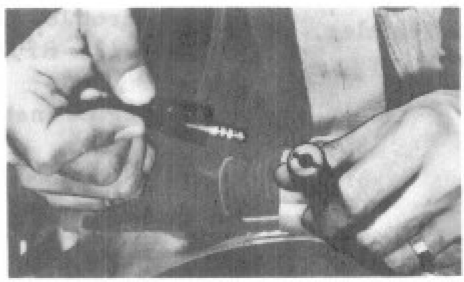
⑫ WITH HELMET ON, CONNECT (TYPE MC-3A CONNECTOR) MASK OXYGEN HOSE TO SEAT OXYGEN HOSE.



⑬ BAILOUT BOTTLE OXYGEN HOSE— ROUTE UNDER RIGHT SHOULDER HARNESS STRAP AND CONNECT TO MC-3A CONNECTOR.

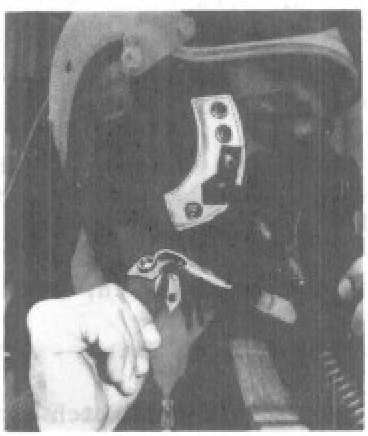


⑭ OXYGEN MASK RETAINING HOSE STRAP — CONNECT TO PARACHUTE HARNESS SNAP FASTENER.



⑮ MIKE CORD — CONNECT.

⑯ OXYGEN MASK — ADJUST TO FIT FACE.



⑰ HELMET CHIN STRAP — FASTEN.

CF-5AD 1-36(2) B

Figure 2-1-2 (Sheet 2 of 2)

(g) Warning Test Switch - TEST.

(h) Position (exterior) Lights - As Required.

(j) Rotating Beacon - BEACON.

(k) Circuit Breakers - In.

PEDESTAL

9A Check the pedestal as follows:

(a) UHF and TACAN Radios - OFF.

(b) Gunsight Control Switch - OFF.

(c) Rudder Pedal Adjustment - As required.

(d) Circuit Breakers - In.

BEFORE STARTING ENGINES

10 Before starting engines, make sure that danger areas shown in Figure 2-1-3 are clear of personnel, aircraft, and vehicles. Refer to Part 4, Section 1, for starting exhaust gas temperature limits.

STARTING ENGINES

LEFT ENGINE

11 Start left engine first using the following procedure:

(a) External Air - Apply.

(b) At 12% RPM, Start Button - Press.

(c) Throttle - Advance to IDLE.

CAUTION

If ignition does not occur before fuel flow reaches 360 pounds per hour, retard throttle to OFF. When all fuel and vapours have been purged from the engine, signal ground crew to turn off airflow. Wait at least 2 minutes to permit fuel to drain before attempting another start.

(d) Check that EGT, engine RPM and oil pressure are within idle operating limits.

CAUTION

If EGT indication reaches 850°C, abort start to preclude exceeding transient temperature limits.

RIGHT ENGINE

12 Start right engine, using same procedure used for left engine start, and continue with following procedure:

(a) All External Power - Disconnect.

CROSSBLEED START

13 To start the right engine using the crossbleed start system, use the following procedure:

(a) Throttle (Left Engine) - Advance to 85% to 90% RPM.

(b) Right Start Button - Press.

(c) Throttle (Right Engine) - Advance to IDLE at 12% RPM.

(d) Engine Instruments - Check.

(e) Throttle (Left Engine) - Retard to IDLE after right engine reaches idle RPM.

BEFORE TAXIING

14 Check the following before taxiing:

(a) Battery Switch - Check ON.

(b) Circuit Breakers - Check.

(c) Damper Switches - PITCH, YAW (On Positions).

(cA) Move both damper switches OFF and back to on. Check that no horizontal stabilizer or rudder movement occurs.

(d) Pitch Cutoff Switch, check as follows:

(1) Pitch Cutoff Switch - Actuate.

(2) Pitch Damper Switch - Moves to OFF.

(3) Pitch Damper Switch - PITCH; Check that horizontal stabilizer does not move.

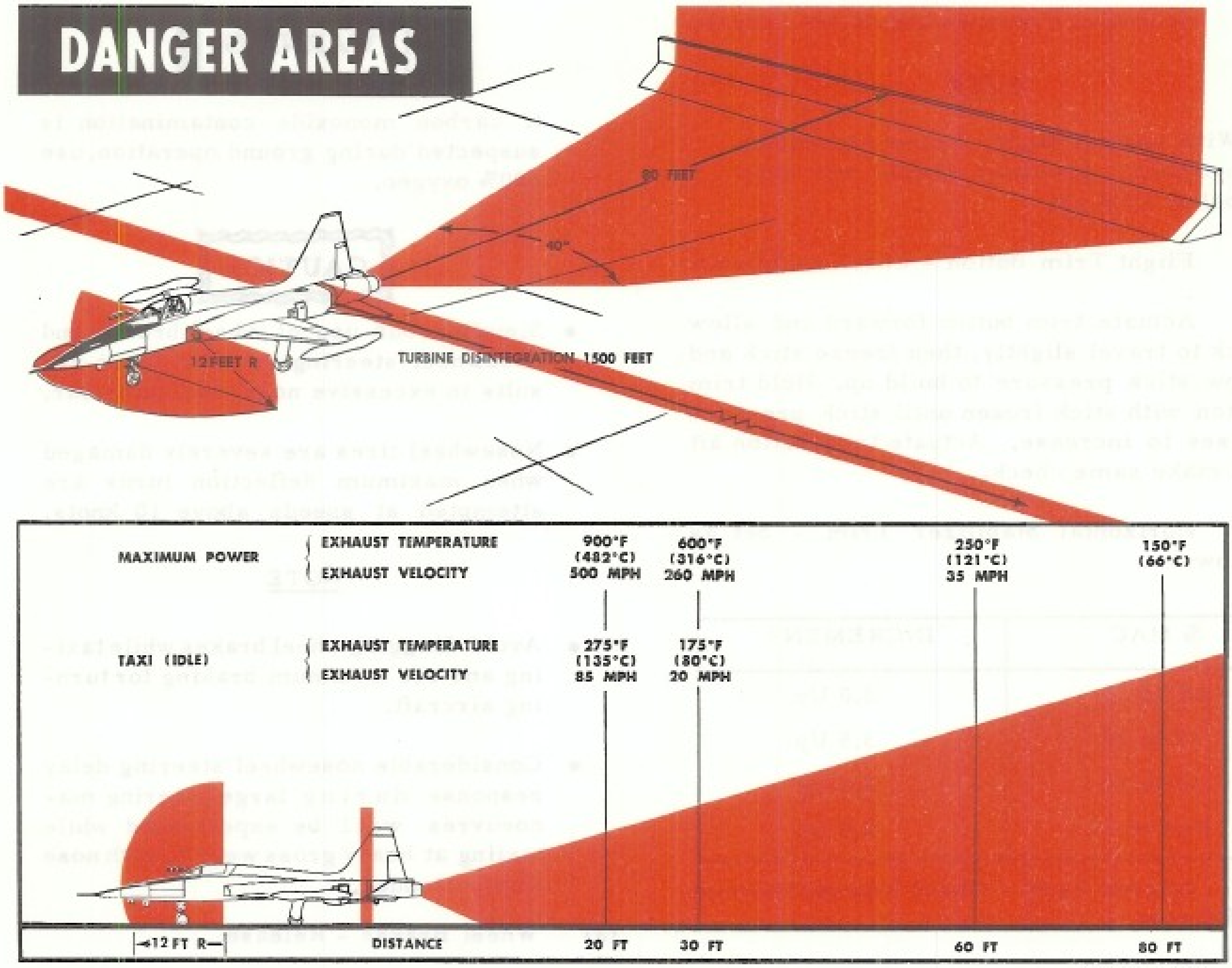


Figure 2-1-3

(4) If horizontal stabilizer moved, Pitch Damper Switch - OFF.

(e) Speed Brake - In (switch forward).

(1) Check that horizontal stabilizer trailing edge moves up as speed brake moves in.

(f) Wing Flap Lever - FULL.

(1) Check for trailing-edge movement of horizontal stabilizer as flaps are actuated. Trailing edge moves down as flaps extend. Verify horizontal stabilizer position with ground crew.

(g) Attitude Indicator - Check.

(h) Standby Attitude Indicator - Cage and Uncage.

(j) ICP-7 - Check Mode (As Desired).

(k) Horizontal Situation Indicator - Check.

(m) Altimeter - Reset.

(n) Radios and Navigational Aids - Switch On and Check as Required.

(p) External Fuel - As Required.

(q) Generator Transfer - Check as follows:

(1) Position left generator switch to OFF and observe that the OFF flag is not visible on either EGT indicator. Return switch to ON position and repeat procedure for right generator.

(r) Oxygen Regulator - Check.

(s) IFF/SIF - STBY.

(t) Flight Controls - Check.

(u) Hydraulic Pressure Indicators - Check.

NOTE

With normal flight control movement, hydraulic pressure should not drop below 1500 psi.

(v) Flight Trim Button - Check as follows:

(1) Actuate trim button forward and allow stick to travel slightly, then freeze stick and allow stick pressure to build up. Hold trim button with stick frozen until stick pressure ceases to increase. Actuate trim button aft and make same check.

(w) Horizontal Stabilizer Trim - Set as follows:

% MAC	INCREMENT
Aft of 20	3.0 Up
17 to 20	3.5 Up
12 to 17	4.5 Up
Fwd of 12	5.0 Up

(x) Aileron Trim - Check Neutral Position as follows:

(1) With control stick centered, check for neutral aileron position by visually sighting trailing edge of ailerons flush with trailing edge of wings.

(y) Seat, Stores Jettison, and Canopy Safety Pins - Remove and Stow.

(z) Wheel Brakes - Apply Heavy Pressure.

NOTE

Heavy pressure application to both brake pedals will set automatic brake adjusters and maintain minimum pedal travel for proper braking efficiency.

(aa) Wheel Chocks - Removed.

TAXIING

15 Perform the following during taxi operation:

WARNING

If carbon monoxide contamination is suspected during ground operation, use 100% oxygen.

CAUTION

- Simultaneous use of wheel brakes and nosewheel steering to effect turns results in excessive nosewheel tire wear.
- Nosewheel tires are severely damaged when maximum deflection turns are attempted at speeds above 10 knots.

NOTE

- Avoid dragging wheel brakes while taxiing and use minimum braking for turning aircraft.
- Considerable nosewheel steering delay response during large steering manoeuvres will be experienced while taxiing at heavy gross weights with nose strut extended.

(a) Wheel Brakes - Release.

(b) Nosewheel Steering - Engage as follows:

(1) As aircraft moves from parked position, press and hold nosewheel steering button on control stick grip and initiate desired turn by applying rudder. Aircraft may be taxied without the use of nosewheel steering.

(c) Flight Instruments - Check.

(d) TACAN - Check.

BEFORE TAKEOFF

16 Perform the following checks immediately before takeoff:

H -	Hood (Canopy)	- Closed and Locked, Light Out.
	Harness	- Tight and Locked.
	Hydraulics	- Check.

T -	Trims	- Set.
	Temperature and Pressures	- Check.
F -	Fuel Boost Pumps	- On.
	Crossfeed	- OFF.
	Quantity	- Check.
	Flaps	- Full and check indicator.
S -	Speed Brakes	- In.
O -	Oxygen Quantity	- Check.
	Blinker	- Check.
	Regulator Controls	- Check.
S -	Battery Switch	- ON.
	Pitch and Yaw Dampers	- On.
	Nose Strut	- EXTEND.
	Engine Door	- OPEN and check indicator.
A/R	Camera Defog	- On.
	Pitot Heat	- On.
	Engine Anti-ice	- As Required.
	Duct Anti-ice	- As Required.
	Windshield Anti-ice	- As Required.
	IFF/SIF	- As Required.
W -	Warning and Caution Lights	- Out.
	Seat, Canopy and Stores Jettison Safety Pins	- Removed and Stowed.

TAKEOFF

17 Normal Takeoff procedure is as follows:

NOTE

The following Takeoff procedures will provide the results shown in the Takeoff charts in Part 4, Section 3.

(a) Wheel Brakes - Apply.

(b) Throttles - MIL.

NOTE

Fuel flow is increased 200 to 300 pph/eng with engine doors open.

(c) Engine Instruments - Check.

(d) Nosewheel Steering - As Required.

CAUTION

- Do not exceed 65 knots with nosewheel steering engaged, except when required for control in emergency situations.
- If nosewheel shimmy is encountered before reaching 65 knots, disengage nosewheel steering.

(e) Wheel Brakes - Release.

(f) Throttles - MAX as follows:

(1) Advance throttles from MIL to MAX AB without pausing at any intermediate AB power setting. Monitor EGT, nozzle position, and RPM as throttles are advanced to MAX AB. After engine operation has stabilized in MAX AB, intermediate AB power may be selected.

WARNING

If MIL power is used for takeoff, ground roll and total takeoff distance will be greatly increased.

NOTE

Unless operational requirements dictate, MAX power will be used for all takeoffs.

(g) Aft Stick Pressure - Apply.

NOTE

Aft stick pressure should be initiated approximately 5 knots below computed takeoff speed. Nosewheel liftoff will not occur immediately after aft stick movement but requires a short time interval, depending on the gross weight and CG position of the aircraft. The heavier the aircraft and the further forward the CG position, the longer will be the time interval required for nosewheel liftoff.

If aft stick movement is initiated too early, the resultant increased drag will reduce acceleration before rotation speed is reached and will lengthen the takeoff distance. Refer to Part 4, Section 3, for takeoff speeds.

CROSSWIND TAKEOFF

18 In addition to the procedures used in a normal takeoff, be prepared to counteract aircraft drift at liftoff by crabbing into the wind.

AFTER TAKEOFF – CLIMB

19 Use the following procedures:

(a) Gear - LG UP, when definitely airborne. Check red light in lever extinguishes.

(b) Flaps - UP, when positive rate of ascent is established. Check indicator.

(c) Throttles - As Desired, as recommended climb speed is reached.

(d) Engine Door Position Indicator - Check CLOSE.

NOTE

If indicator indicates OPEN after accelerating through 245 to 265 KIAS, manually position Engine Doors Switch to CLOSED and check indicator for CLOSE indication.

(e) External Fuel - As Required.

(f) Oxygen - Check Flow.

(g) IFF/SIF - As Required.

(h) Cockpit Pressurization - Check.

(j) Canopy Defog - As Required.

(k) Damper Switches - Check.

NOTE

If pitch damper switch has become disengaged, do not reengage below 5000 feet. Refer to Part 4, Section 1, for engagement and disengagement limits.

(m) Altimeter - Set.

CRUISE

20 For cruise control data at various gross weights, refer to Part 4, Sections 5 and 6, and for cruise flight characteristics, refer to Part 2, Section 5.

FUEL BALANCING

21 Perform fuel balancing as follows:

(a) Fuel Quantity Indicators - Check:

(1) Position Fuel and Oxygen Check switch at GAGE TEST and check that fuel quantity indicator pointers move counterclockwise.

(b) Crossfeed Switch - CROSSFEED.

(c) Boost Pump (on side of lower fuel quantity) - OFF.

(d) Boost Pump - On (when fuel quantities balanced).

(e) Crossfeed Switch - OFF.

CAUTION

Pilot must remember to turn crossfeed switch off after fuel balancing operation to prevent unbalancing toward other internal system.

DESCENT

22 Perform the following, either before or during descent:

(a) Canopy Defog - As Required.

NOTE

Canopy and windshield defogging should be initiated before descent from altitude in sufficient time to allow heating of transparent surfaces. Failure to do so will result in fogging of these surfaces at lower altitudes.

(b) Duct, Windshield, Engine, and Pitot Anti-Ice - As Required.

NOTE

Anti-ice heat should be applied for descent into known or suspected icing conditions.

(c) Oxygen Quantity and Pressure - Check.

(d) Oxygen - Check Flow and Controls.

(e) Altimeter - Set.

BEFORE LANDING

23 Perform the following before landing checks:

(a) Crossfeed - Discontinue.

(b) Hydraulic Systems - Check Pressure.

(c) Gear - LG DOWN. Check three green lights.

(d) Flaps - FULL. Check indicator.

(e) Engine Doors - OPEN. Check indicator.

LANDING

24 The landing procedures shown in Figure 2-1-3 and the following discussions will produce the results shown in the Landing Charts in Part 4, Section 8.

NORMAL LANDING

25 The recommended normal landing pattern procedures are shown in Figure 2-1-4. For a normal landing, maintain precise airspeed control throughout the final approach and touchdown. If runway length and conditions permit, aerodynamic braking may be used to conserve brakes and tires. To accomplish aerodynamic braking, the landing attitude angle, after touchdown, should be increased until the nose is level with the horizon. The stick position should be moved aft gradually until full back stick is obtained. If drag chute is to be used, lower the nosewheel to the runway before deploying the chute. Counteract aircraft yawing with rudder, nosewheel steering (below 65 KIAS), and wheel braking as necessary. Refer to Part 4, Section 1, for landing gear sink rate limitations and to Part 4, Section 8, for landing airspeeds and distances. The landing distances presented in the appendix are based on a 3-degree approach slope, lowering the nosewheel to the runway immediately after touchdown, and use of heavy braking. More precise longitudinal control for landing is obtained when trim is set forward of full aft trim.

MINIMUM RUN LANDING

26 To accomplish a minimum run landing (shortest obtainable stopping distance), maintain normal final approach and touchdown speeds for existing aircraft gross weight and CG condition. After touchdown, if a crosswind is not a factor, lower nosewheel immediately and deploy drag chute. Apply maximum wheel braking without skidding tires. If a crosswind is a factor, maintain a nose-high attitude to obtain maximum aerodynamic braking until speed is decreased, then lower nosewheel and deploy drag chute. Apply maximum wheel braking, using nosewheel steering for directional control. Landing distances shown in Part 4, Section 8, are based on the use of the above procedures.

HEAVY-WEIGHT LANDING

27 When making a heavy-weight landing, proper technique in the application of power and flight controls is essential. Final approach and touchdown speeds must be increased accordingly. The pilot should obtain from the appendix the recommended final approach and touchdown speeds for the maximum gross weight landing anticipated before each flight. Having determined these speeds, a normal traffic pattern should be flown.

LANDING PATTERN (TYPICAL)

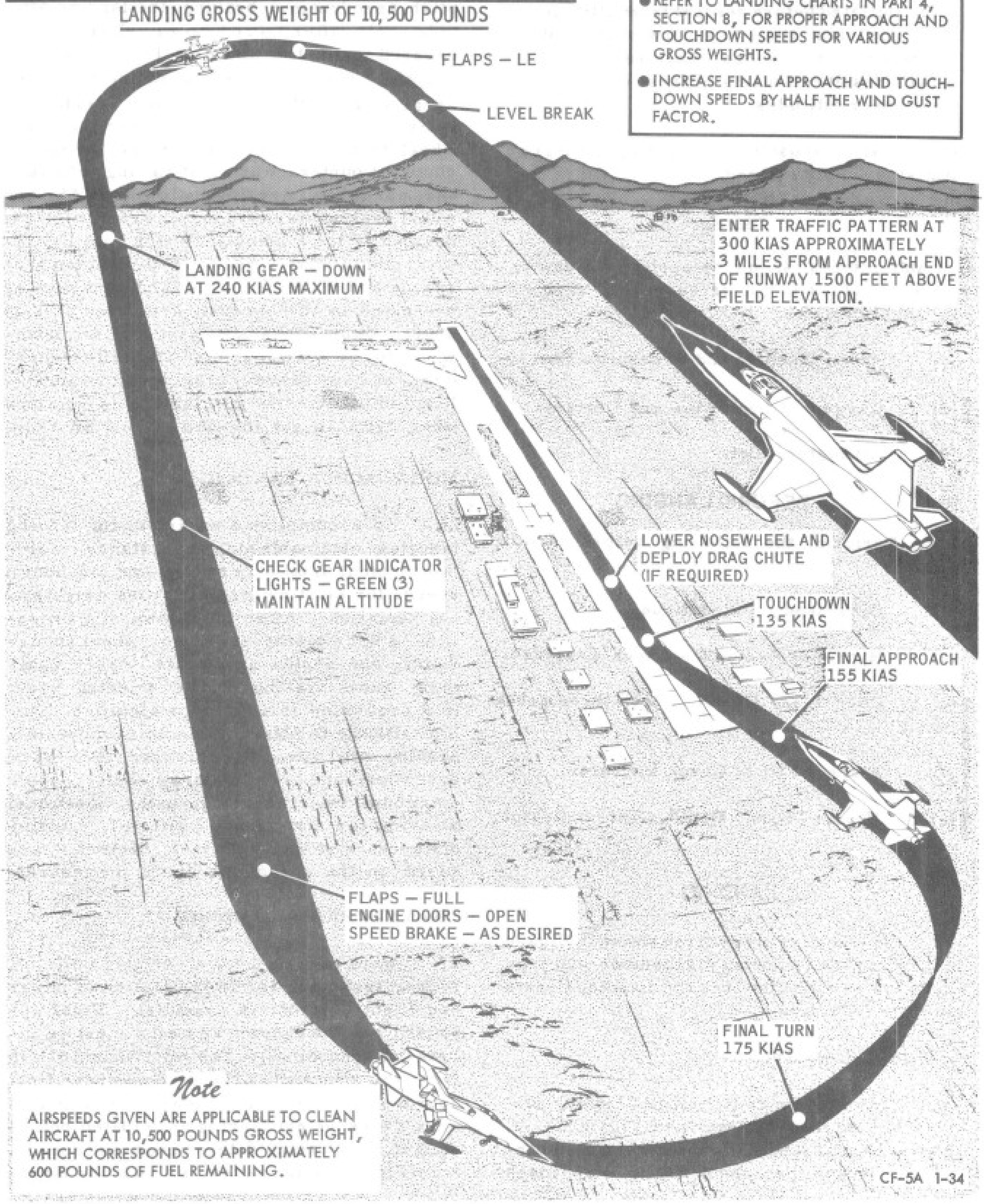


Figure 2-1-4

Care should be taken to control the rate of sink to touchdown, using power as necessary. Full stall landings at any gross weight are not recommended.

CROSSWIND LANDING

28 The procedure for landing in a crosswind is the same as for normal landings except for drift correction. On final approach, counteract drift by crabbing into the wind, maintaining flight path alignment with the runway. The crab should be held until touchdown. In crosswinds above 15 knots, touchdown should be planned for the upwind side of the runway. The wings must be level at touchdown. After touchdown, maintain directional control of the aircraft with the rudder. Do not increase the landing attitude angle at a rate which will cause the aircraft to become airborne and drift across the runway. Drift will create a high probability of tire damage. Do not lower the nose immediately after touchdown, as this will produce a compression of the downwind strut causing the aircraft to weathervane into the wind. If use of drag chute is required, lower the nosewheel to the runway before deploying the chute and be prepared to counteract weathervaning tendency with nosewheel steering and wheel braking. If directional control cannot be maintained, jettison the drag chute immediately.

WET OR SLIPPERY RUNWAY LANDING

29 When landing on a wet or slippery runway, normal landing procedures should be used. Since landing ground roll distances are significantly increased on a wet or slippery runway, the landing pattern should be planned so that touchdown is well within the first third of the runway. If a crosswind is not a factor, lower the nosewheel to the runway immediately after touchdown and deploy drag chute, maintaining directional control with rudder and brake as necessary. Should a crosswind be a factor, hold the nose as high as possible after touchdown without becoming airborne to obtain maximum aerodynamic braking, and hold this attitude until speed is reduced. After nosewheel is lowered, brakes should be applied cautiously. Avoid locking the brakes. Hydroplaning and/or tire skidding on a wet or icy runway will increase stopping distance and can easily result in loss of directional control. Every effort should be made to remain in the center of the runway should barrier engagement become necessary. Taxi with caution as nosewheel steering can be relatively ineffective under wet or slippery runway conditions. See Figure 4-8-4 for the runway condition reading.

TOUCH-AND-GO LANDING

30 To make touch-and-go landing, use the procedures for normal landing followed by a normal go-around.

USE OF WHEEL BRAKES

31 To prevent skidding the tires, extreme care must be exercised when applying wheel brakes immediately after touchdown, at high landing speeds and/or heavy gross weights, or whenever there is considerable lift on the wing. Heavy brake pressure will lock the wheels more easily immediately after touchdow than when the same pressure is applied after the full weight of the aircraft is on the tires. A wheel locked in this manner immediately after touchdown will not become unlocked as aircraft load increases as long as brake pressure is maintained.

32 If maximum braking is required, decrease lift as much as possible by lowering the nosewheel to the runway and raising flaps before applying wheel brakes. This will improve braking action by increasing the load on the tires and thus increase the frictional force between tires and runway.

33 If excessive use of wheel brakes during taxi and landing operation results in brake overheating, or at the first indication of brake malfunction, stop the aircraft on taxiway in a suitable location. The aircraft should not be taxied into a crowded parking area. Overheated wheels and brakes must be cooled before the aircraft is towed or taxied. In extreme overheat cases, heat buildup can cause the wheel and tire to fail with explosive force or to burn if proper cooling does not occur. Taxiing at low speed to obtain air-cooling of overheated brakes will not reduce temperatures adequately and can even cause additional heat buildup.

34 Use the following chart as a guide for action required regarding care of wheel brakes in the event of an aborted takeoff, heavyweight landing, drag chute deployment, nondeployment, or failure at various aircraft gross weights, or excessive use of wheel brakes.

NOTE

The data in the following table is based on commencing brake application at stall speed or maximum refusal speed for weights noted. Drag chute is assumed deployed at 165 KIAS.

ACFT Gross Weight	Drag Chute	Action Required
Up to 15,000 lb Up to 18,000 lb	No Yes	None; follow recommended flight interval for braking in Part 4, Section 1.
15,000 to 18,000 lb Above 18,000 lb Above 18,000 lb	No Yes No	Taxi clear of congested area, let brakes cool, inspect tires and brakes, and make appropriate entry in Form CF 337.

GO-AROUND

35 The decision to go around should be made as soon as possible. Use the following procedures:

(a) Throttles - MIL (MAX if necessary).

(b) Speed Brake - In (switch forward).

(c) Gear - LG UP, when positive rate of climb is established.

(d) Flaps - As Required.

36 A short, closed pattern go-around at 12,000 pounds gross weight with tiptanks and five pylons, using two engines and military thrust for climb, requires 220 pounds of fuel. Fuel consumption increases approximately 18 pounds for every 1000-pound increase in gross weight above 12,000 pounds.

AFTER LANDING - CLEAR OF RUNWAY

37 The after landing - clear of runway procedures are as follows:

(a) Drag Chute - Jettison, if deployed.

CAUTION

Do not allow the chute to collapse as the risers will be burned while resting on the hot tail section.

NOTE

Do not attempt to taxi with the drag chute deployed.

(b) Cabin Altimeter - Check.

NOTE

If altimeter reading is below field elevation, position cabin pressurization switch to RAM DUMP before opening canopy.

(c) Flaps - UP (barber pole indication).

(d) Engine Doors - CLOSED. Check indicator.

(e) Speed Brake - Out. Return switch to centre.

(f) Cabin Temp - 100° (Full Hot) If Required.

(g) Duct, Windshield, Engine, and Pitot Anti-Ice - OFF.

(h) Canopy Defog - OFF.

A/R (j) Camera Defog - OFF.

(k) IFF/SIF - OFF.

ENGINE SHUTDOWN

38 Shut down the engines as follows:

(a) Wheel Brakes - Hold until wheel chocks in place.

(b) All Unguarded Switches (except battery and generator switches) - OFF.

(c) Throttles - OFF.

(d) Battery and Generator Switches - OFF.

(e) Oxygen Supply Lever - OFF.

BEFORE LEAVING AIRCRAFT

39 Before leaving the aircraft, perform the following:

(a) Seat, Stores Jettison, and Canopy Safety Pins - Installed.

SECTION 2

ENGINE SYSTEM OPERATION

COMPRESSOR STALL

1 A compressor stall is an aerodynamic interruption of airflow through the compressor. Factors which can increase the stall sensitivity of an engine are foreign object damage, high angles of attack at low airspeeds and high altitudes, abrupt yaw impulses at low airspeeds (below 150 KIAS), temperature distortion, and ice formation on the aircraft inlet ducts or engine inlet guide vanes. Compressor stalls can also be caused by component malfunctions, engine rigging out-of-limits, throttle bursts to military or maximum AB power at high altitude and low airspeed, and hot gas ingestion. The types of stalls which may occur are discussed below.

TAKEOFF AND LOW ALTITUDE—HIGH AIRSPEED COMPRESSOR STALL

2 Compressor stall may occur on takeoff with AB initiation, or at low altitude and very high airspeed when in military or AB operation. The stall is recognized by a "pop" or "bang" followed by an audible "buzzing" sound and vibration, accompanied by a rapid RPM drop and high EGT. The stall should be cleared as soon as possible to prevent engine damage by overtemperature. This type of compressor stall can normally be recovered by a rapid throttle reduction to IDLE, and flameout usually does not occur.

HIGH ALTITUDE—LOW AIRSPEED COMPRESSOR STALL

3 Compressor stall at high altitude and low airspeed typically occurs during throttle bursts to military or AB power. The stall is recognized by an audible "chug" or "pop," accompanied by rapidly unwinding RPM and decreasing EGT. Flameout may result from this type of stall. A rapid throttle reduction to IDLE and immediate actuation of the engine start button will often recover the engine and prevent flameout.

FLAMEOUT

4 Engine flameout may occur as a result of certain engine component malfunctions; engine compressor stall, fuel starvation, fuel contamination (water), fuel icing, engine inlet guide vane icing, and from unusual aircraft attitudes outside the aircraft flight envelope. Improper recovery from an unusual aircraft attitude such as a high-attitude stall (Part 2, Section 5) can produce an abrupt yaw at low airspeed resulting in engine compressor stall and flameout.

VARIABLE INLET GUIDE VANES

5 Variable inlet guide vanes and bleed valves have been incorporated in the engine to reduce the possibility of a compressor stall throughout the operating range of the engine. The system operates automatically as a function of engine RPM and inlet temperature.

T_2 HEATER SYSTEM

6 A T_2 heater system is provided for the inlet temperature sensor system on each engine. Eighth-stage compressor bleed air is directed on the sensor to improve response rate of the engine inlet guide vane and bleed valve systems during periods of rapid increase in inlet temperature. An air temperature probe, mounted on the fuselage dorsal, activates the system when a total temperature of 120°F or above is reached. The system compensates for rapid changes in temperature, i.e., weapons delivery dive, by increasing the response capability of the inlet guide vane system and thus providing adequate protection against compressor blade instability and compressor stall. A slight decrease in engine rpm during system operation at MIL or MAX power will be observed.

EXHAUST GAS TEMPERATURE AND RPM CUTBACK

7 Engine susceptibility to stall is minimized by an automatic reduction of engine RPM when compressor inlet temperature (T_2) falls below -25°C, thus preventing excessive airflow through engine. The exhaust gas temperature is also reduced by a control signal from the compressor inlet temperature sensor (T_2) to the turbine exit temperature amplifier (T_5). As a result, RPM and EGT indications may be below normal operating limits at MIL and MAX power.

ENGINE OIL PRESSURE

8 Abnormal engine oil pressure indications frequently are an early indication of some engine trouble. The engine oil pressure indicator is marked for normal operating conditions on the ground or in the air (20-50 psi); however, it should be noted that idle rpm may produce only a 5-psi indication while military power setting may cause an indication as high as 55 psi, which is within operating limits. These indications are acceptable provided they return within the 20 to 50 psi range after engine oil has warmed and has had sufficient time to circulate. Of primary concern to the pilot is that a sudden change of 10 psi in the oil pressure indication at any stabilized RPM setting necessitates that engine be shut down and inspected.

VARIABLE NOZZLE OPERATION

9 The afterburner (AB) variable exhaust nozzle opening is controlled by throttle position and turbine exit temperature. During normal advance of the throttle, nozzle opening decreases until approximately 90% indicated RPM is reached; the nozzle then stops closing and stays at approximately 30% nozzle open until RPM is approximately 97%. At this point, control of nozzle area is transferred from the throttle to the turbine exit temperature amplifier (T_5). As the engine accelerates above approximately 97% RPM, the T_5 amplifier modulates nozzle opening to maintain approximately 670°C EGT. During rapid throttle advance from IDLE to MIL, nozzle opening is mechanically controlled by the throttle until nozzle opening is approximately 45%. The nozzle remains at this opening until the engine accelerates to the speed established by throttle position; the T_5 amplifier then assumes control of the nozzle. During ground operation at MIL power, nozzle opening should be approximately 15%. As throttle is advanced into AB range, indicated nozzle opening will increase from approximately 35% at minimum AB to approximately 85% at maximum AB. Should the T_5 system fail during military or AB operation, retard the throttle to maintain EGT within limits.

SECTION 3

FUEL MANAGEMENT

INTERNAL

1 At all altitudes, the internal fuel systems automatically supply fuel to the engines when throttles are advanced from OFF and the boost pumps are operating. With boost pumps not operating, MAX throttle fuel is not guaranteed above 6000 feet pressure altitude and, at lower throttle settings, fuel starvation may result above 25,000 feet.

1A The fuel quantity indicator indicates internal fuel only and should be monitored to maintain the two systems within 200 pounds of each other when operating at or near the aft CG limit. This is necessary to ensure the aircraft is kept within the CG limits.

1B A decrease in range will result from just a 500-pound differential in the internal fuel systems. This is due to the increased drag resulting from horizontal tail deflection required to trim the aircraft. Maintaining the two systems within this parameter is accomplished by using the fuel balancing procedures discussed in Part 2, Section 1.

EXTERNAL

2 External fuel may be carried in two non-jettisonable wingtip tanks, two jettisonable inboard pylon tanks and one jettisonable centre-line pylon tank, see Figure 1-3-2 for capacities. External fuel is transferred by air pressure applied to the external tanks from the air-conditioning system, causing fuel to flow into both internal systems as needed. Overfilling is prevented by a fuel level control valve in each system. Fuel transfer valves are electrically controlled by three toggle switches on the right vertical control panel in the front cockpit. The placarded position (CL, PYLONS, TIPS) is the on position. Edge lighting is provided for positive tank selection at night. Placing any of these switches on with an engine operating and the cockpit pressurization switch at CABIN PRESS will pressurize all external tanks. If the cockpit pressurization switch is placed at RAM DUMP, or the aerial refuel switch at AERIAL REFUEL, pressure will not be available for fuel transfer.

2A External tank groups should be selected individually and in proper sequence, refer to Paragraph 4, following. When transfer of the selected group is completed the EXT TANKS EMPTY caution on the caution light panel and the master caution light will illuminate. The pilot should monitor the fuel quantity indicator for decrease in quantity before selecting another tank group to confirm that all fuel from selected tanks has been transferred. The next tank group should be turned on before turning off the empty tank group to prevent depressurizing the system. Failure to turn off the switch of the empty tank group prevents any further light indication of fuel transfer completion and allows the external tanks empty light to remain on.

NOTE

> When transferring fuel from the inboard pylon tanks, the pilot must verify complete transfer of fuel before selecting wingtip fuel, because of restricted airspeed, see Figure 4-1-5, with other than residual fuel in the inboard fuel tanks and empty tip tanks. Completion can be verified by a time check, the external tanks empty light on, and a decrease in the quantity indication of the internal systems.

CROSSFEED OPERATION

3 Either internal fuel system can supply both engines or both systems can supply either engine through an electrically operated crossfeed valve and proper positioning of boost pump switches. With the crossfeed switch in CROSSFEED position, the crossfeed valve is opened, providing the pilot the capability of balancing the internal fuel supply or

of operating one engine from both systems during single-engine operation. When conducting crossfeed operation at altitude, the crossfeed switch should always be placed in CROSSFEED position before turning the fuel boost pump switch OFF (low side) to preclude engine flameout. When fuel balancing is completed, boost pumps should be turned on and crossfeed turned OFF. Use of crossfeed with low fuel indicated in either internal system should be attempted only if a level or slight nose-up aircraft attitude can be maintained. This is necessary to preclude uncovering the boost pump inlets, allowing air to enter the fuel supply lines causing engine flameout.

NOTE

- With one boost pump inoperative, fuel may be crossfed from the system with the operative fuel boost pump.
- Approximately 10 minutes after operating on one engine, with both fuel boost pumps in operation, a low fuel pressure warning light will appear on the inoperative engine side; however, normal fuel flow from both systems is available to the operating engine.

FUEL SEQUENCING

4 The sequence of external fuel transfer when all tanks are installed is wing pylons, centreline pylon, then wingtip. In any configuration of tanks or weapons, wingtip fuel should normally be transferred after other external fuel has been used and/or stores released from inboard pylons. To prevent wingtip tank flutter when the wingtip position is selected, all fuel must be transferred before turning the external fuel tip switch off.

FUEL VENTING

5 Fuel may be lost overboard in any attitude due to failure of the fuel level shutoff valves while transferring fuel. If this occurs, discontinue all fuel transfer until fuel quantity indicators indicate less than total capacity. Refill internal supply from external tanks, as necessary, by turning desired external fuel selector switches on; turn external fuel selector switches OFF when fuel quantity indicator indicates near full. Fuel may be lost overboard in a climb due to failure of the fuel vent check valves. If this occurs, level aircraft immediately and do not climb to a higher altitude until the internal fuel quantities have been reduced.

SECTION 4

AIR REFUELING PROCEDURE

GENERAL

1 The procedures recommended herein are those required for air refueling from KC-130F tanker aircraft, when the air refuel probe is installed on the aircraft. Refer to Part 4, Section 10, for air refueling performance data.

2 Optimum airspeed for refueling is 215 KCAS with flaps at FULL position. Full flaps provide a lower angle-of-attack, improved stability and control, and require less power at high gross weights.

KC-130F DESCRIPTION

3 The KC-130F tanker is equipped with two paradrogues on extendible hoses located within pylon-mounted underwing pods outboard of the engines and approximately 10 feet inboard from the wingtips. Refueling is possible at any hose length between 20 and 80 feet. Refueling rate is 300 gallons per minute per station. Prior to receiver engagement, the drogues trail with the hoses fully extended to 85 feet. During refueling, a hydraulic reel control system maintains hose tension to eliminate whip. Fuel flow is automatically initiated following hookup as the receiver moves forward and the hose length decreases to 80 feet. If the hose length decreases to 20 feet or increases past 80 feet, the fuel flow is automatically cut off. Normal disengagement is accomplished by slowing the receiver aircraft and moving straight back until the hose reels full out and the drogue detaches from the probe.

ENGAGEMENT PROCEDURE

4 The following procedures will ensure a successful engagement.

(a) If external fuel tanks are to be refueled, position aerial refueling switch to AERIAL REFUEL and select desired external fuel tanks.

(b) Stabilize the receiver aircraft speed and align the probe 10 to 15 feet behind the drogue.

(c) Establish a reference point dead-ahead on the tanker and apply power to establish a 5-knot steady closing rate.

(d) Maintain sight of the drogue with peripheral vision but do not attempt to chase the drogue.

(e) Adjust power after engagement to maintain 40 to 50 feet of hose extended.

(f) Maintain station on the tanker slightly stepped down and with the hose extended straight aft from the refueling pod.

POWER MANAGEMENT

5 The use of afterburner (AB) in the minimum AB position normally will be required to maintain station when refueling to full capacity in the ferry configuration at 20,000 feet. The method of power management for heavy configurations is as follows.

(a) Place the left throttle in the minimum AB position while approaching the tanker.

(b) Regulate closure during final approach and engagement with the right throttle only.

(c) Continue to advance the right throttle to MIL power as gross weight increases during fuel transfer.

(d) Modulate the left AB power if more power is required with additional fuel transfer.

(e) If it becomes necessary to use AB on both engines to maintain or regain station, the throttle positions should be synchronized.

6 The throttle combinations selected allow use of the radio microphone button at all times.

CAUTION

Do not use speed brake during engagement or disengagement. Use of speed brake will cause excessive trim changes.

DISENGAGEMENT PROCEDURE

7 Fuel venting from the external tanks and the vertical tail will occur when the respective systems are full. To disengage from the drogue, gradually reduce power and slowly back away from the tanker. Air refueling probe will disengage when the tanker refueling hose is fully extended.

NOTE

- When disengaged, ensure that aerial refueling and external fuel transfer switches are positioned to OFF.
- Resume normal fuel management procedures.

SECTION 5
FLIGHT CHARACTERISTICS

INTRODUCTION

1 The aircraft is designed for high speeds at all altitudes. The area rule fuselage minimizes the effect of mach number. The wing does not have geometric dihedral, since sufficient aerodynamic dihedral effect is supplied by the moderate sweepback (25 degrees) of the wing. Large tail surfaces provide adequate longitudinal and directional stability for all angles of attack and mach numbers. However, a two-axis stability augmentation system is provided to improve handling characteristics.

STALLS

GENERAL

2 A 1-G stall at 0 degrees of bank, see Figure 2-5-1, Sheets 1 and 2, is not accompanied by any abrupt aircraft motion. The stall condition is preceded by heavy, low-speed buffet and moderate wing roll. Complete lateral control is available to well below stall speed. The actual stall is accompanied by a very high sink rate. Low speed buffet is most severe with flaps fully extended.

STALL RECOVERY

3 Low-attitude stalls can be terminated by an increase of power and a decrease in stick back pressure. It is not necessary to allow the nose to pitch down during a 1-G stall recovery.

ACCELERATED STALLS

4 Subsonic accelerated stalls, see Figure 2-5-1, Sheets 1 and 2, are similar to 1-G stalls except the stall is not accompanied by wing roll.

HIGH-ATTITUDE STALLS

5 A high-attitude stall is defined as a manoeuvre terminating at a positive pitch angle of 75 degrees or less, resulting in airspeeds as low as zero.

NOTE

As an airspeed of 100 knots is approached, smoothly utilize controls to correct minor rolling or pitching motions to minimize oscillations during recovery and reduce power to no less than 80% RPM. Stalls in excess of 75-degree pitch attitude may result in poststall gyrations.

WARNING

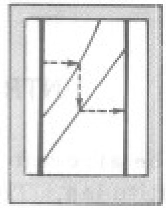
Maximum rates and deflections of the controls (either momentary or sustained) will force the aircraft into either a poststall gyration or a spin with possible pilot disorientation. Intentional stalls at pitch angles greater than 75 degrees are prohibited because of possible dual engine flameout.

6 Following stall, the aircraft will pitch toward the horizon at approximately 0 G's until a diving attitude is achieved and flying speed is regained. Dependent upon the nose-down pitching rate, the aircraft may rotate beyond the nose-down vertical attitude and develop a pendulous yawing and rolling motion as airspeed is regained.

POSTSTALL GYRATIONS

7 Poststall gyrations are characterized by high pitching rates, large yawing angles, and abrupt oscillatory motions about all three axes of the aircraft or an outside spiraling (recognized by increasing roll rate, negative load factor, and airspeed) or a level inverted position with very little yawing motion. These motions can occur either in the erect or inverted positions; however, they usually are associated with a high rate pitch-over, progressing from a high attitude condition through the nose-down vertical position to inverted flight. The recovery phase of a

STALL SPEED CHART
FLAPS AND GEAR DOWN



DATA BASIS
- ESTIMATED
- ALL CONFIGURATIONS
- POWER-OFF (IDLE THRUST)

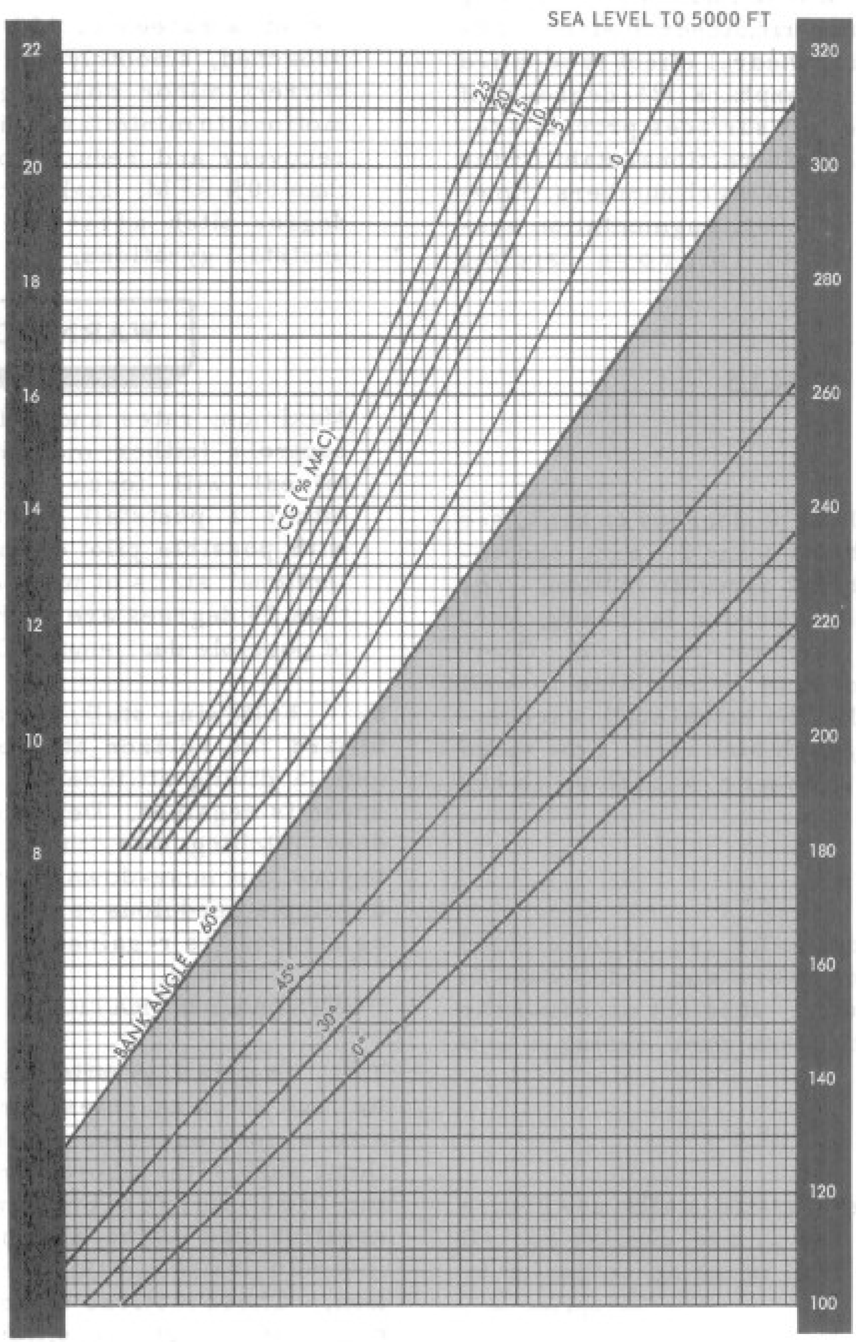


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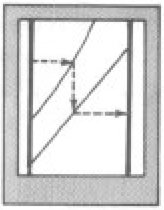
Figure 2-5-1 (Sheet 1 of 2)

STALL SPEED CHART
FLAPS AND GEAR UP

DATA BASIS
- ESTIMATED
- ALL CONFIGURATIONS
- POWER-OFF (IDLE THRUST)



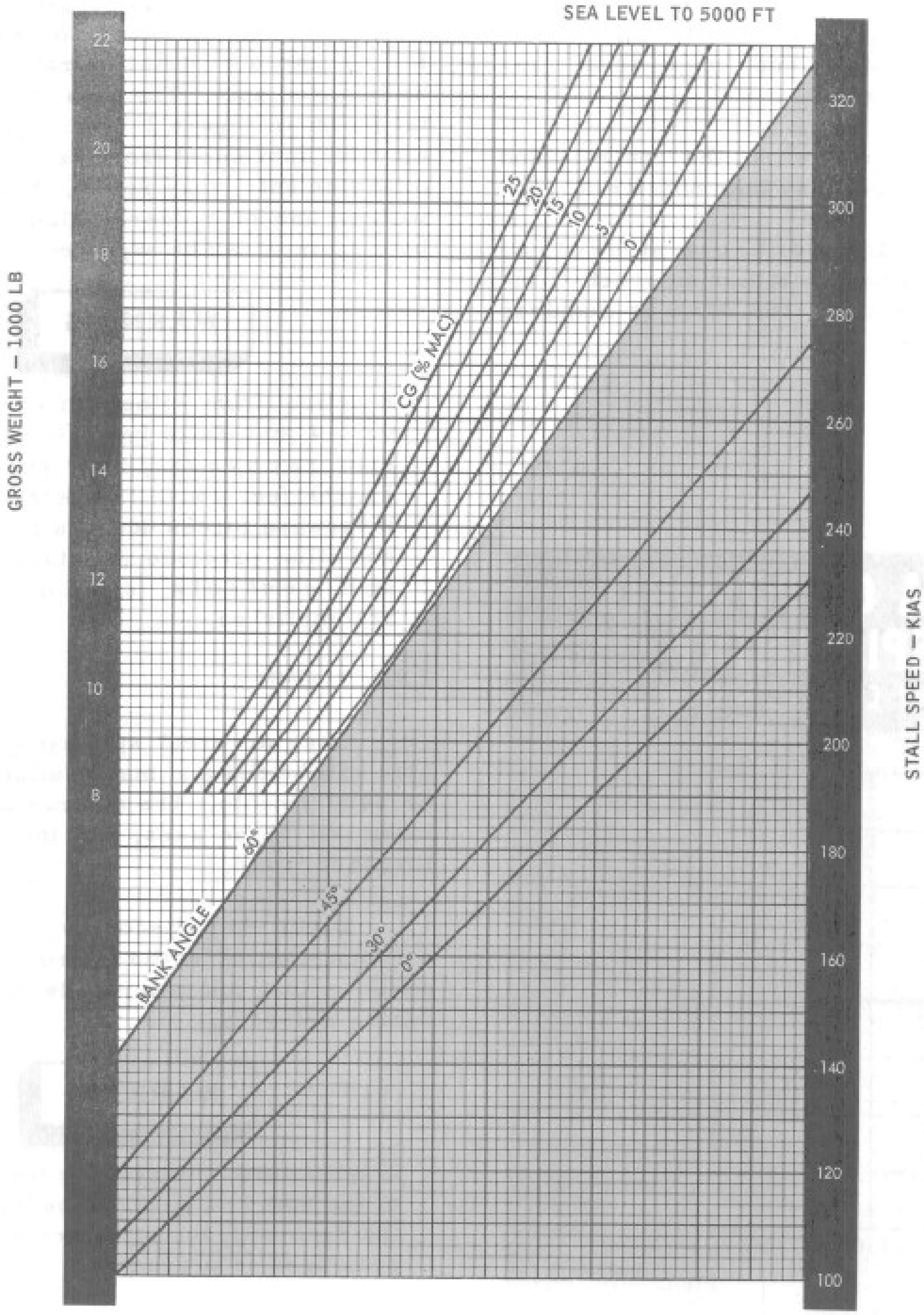


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Figure 2-5-1 (Sheet 2 of 2)

poststall gyration is usually characterized by either a large pendulous yawing and rolling motion or a sustained rolling condition combined with increasing airspeed. The addition of external wing and centreline stores will result in more severe poststall gyrations.

HIGH-ATTITUDE STALL AND POSTSTALL GYRATION RECOVERY

8 Recovery from a high-attitude stall or a poststall gyration manoeuvre is accomplished as follows:

(a) Smoothly neutralize the controls and adjust power to not less than 80% RPM.

(b) Maintain the controls at neutral ("hands-off") when in either an erect or inverted attitude until 150 KIAS is acquired during the dive.

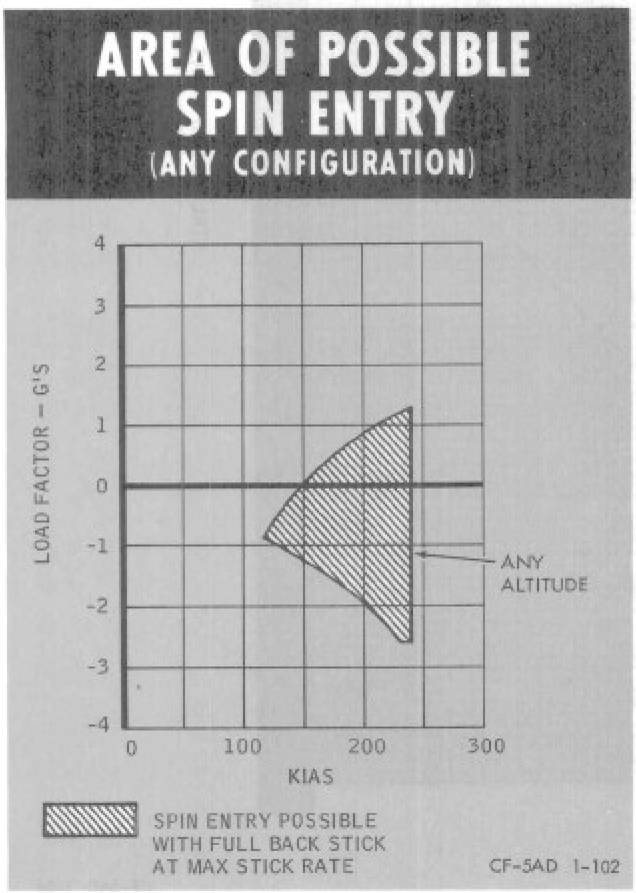

Figure 2-5-2

(c) Smoothly apply controls to recover the aircraft.

SPINS

GENERAL

9 Intentional spins are prohibited. The aircraft exhibits a high degree of resistance to spin entry. However, the aircraft can be forced into a poststall gyration or an erect or inverted spin. The area of possible entry is shown in Figure 2-5-2. Avoid abrupt full aft stick movement from the conditions shown or a spin entry may result. Entry can occur without use of rudder. Flameout of one or both engines can be expected.

WARNING

Ejection from either an erect or inverted spin is to be accomplished if a spin recovery is not completed by 15,000 feet above the terrain. Also, if forward forces due to a flat spinning condition preclude maintaining anti-spin controls, ejection should be accomplished.

ERECT SPIN

10 Once an erect spin has developed, the spin will be flat and may be either oscillatory or very smooth. The aircraft may oscillate about all three axes, and the pilot will be forced forward.

11 Erect Spin Recovery. The primary antispin control is the aileron, and it is imperative that full aileron deflection be held during recovery.

WARNING

If full aileron deflection in the direction of the spin is not maintained throughout the recovery, the spin recovery may be prolonged or prevented.

12 Immediately upon recognition of the direction of rotation, use the following procedures:

(a) Control Stick - Full aileron in the direction of the spin (use both hands) and as much aft stick as possible without sacrificing aileron.

(b) Rudder - Full opposite.

(c) Do not change gear, flaps, and speed brake positions during recovery.

(d) Neutralize controls after recovery.

NOTE

Recovery from the spin is normally abrupt and may be followed by some spiraling during the resultant dive.

INVERTED SPIN

13 An inverted spin is very oscillatory about all axes.

14 Inverted Spin Recovery. Immediately upon experiencing an inverted spin, use the following procedure:

(a) All flight controls - Neutralize.

WARNING

Maintain controls in neutral position throughout the spin recovery. Any aileron or rudder deflection can induce a transition to an erect spin.

EXTERNAL STORES

15 Do not jettison external stores until aircraft is under control.

DRAG CHUTE

16 Do not use the drag chute as a spin recovery device because of the following undetermined factors:

(a) Deployment.

(b) Effectiveness.

(c) Ejection seat-to-chute clearance.

(d) Structural failure.

FLIGHT CONTROLS

RUDDER

17 The rudder is able to generate large roll rates. Caution should be used when applying abrupt aileron plus rudder together in the same direction, as large angles of attack with large sideslip angles and load factors in excess of limits can be generated. The heavily loaded configurations during rudder applications below 240 KCAS will require aileron deflections in excess of the aileron spring stop limit to hold wings near level. This may lead to lateral/directional oscillations which will be adequately damped when both controls are neutralized. Rudder reversals (full rudder input in one direction, then in the other direction) for airspeeds below 250 KCAS will create large excursions of sideslip angles and corresponding roll rates. The application of opposite aileron will always be sufficient to stop the induced roll rate after the rudder is neutralized.

SPEED BRAKE

18 The speed brake system effectively controls airspeed in dives or when a rapid speed reduction is desired; however, pitch transients will occur during speed brake operation. The pitch transients may be in either a nose-up or nose-down direction, depending on flight conditions.

CAUTION

When operating speed brake with wing stores on the aircraft above 0.80 IMN and below 15,000 feet altitude, a nose-up trim change on extension will be produced and increased aircraft sensitivity will be apparent. Load factor increments as high as 2.5G can be experienced. Because of aircraft sensitivity, the pilot should exercise caution in attempting precise attitude control. If pitch oscillations occur, control can be established by releasing stick or holding stick fixed. Incremental extension and retraction of the speed brake will reduce abruptness of the trim change.

SPEED PROFILE MAXIMUM THRUST
(2) WINGTIP TANKS STANDARD DAY

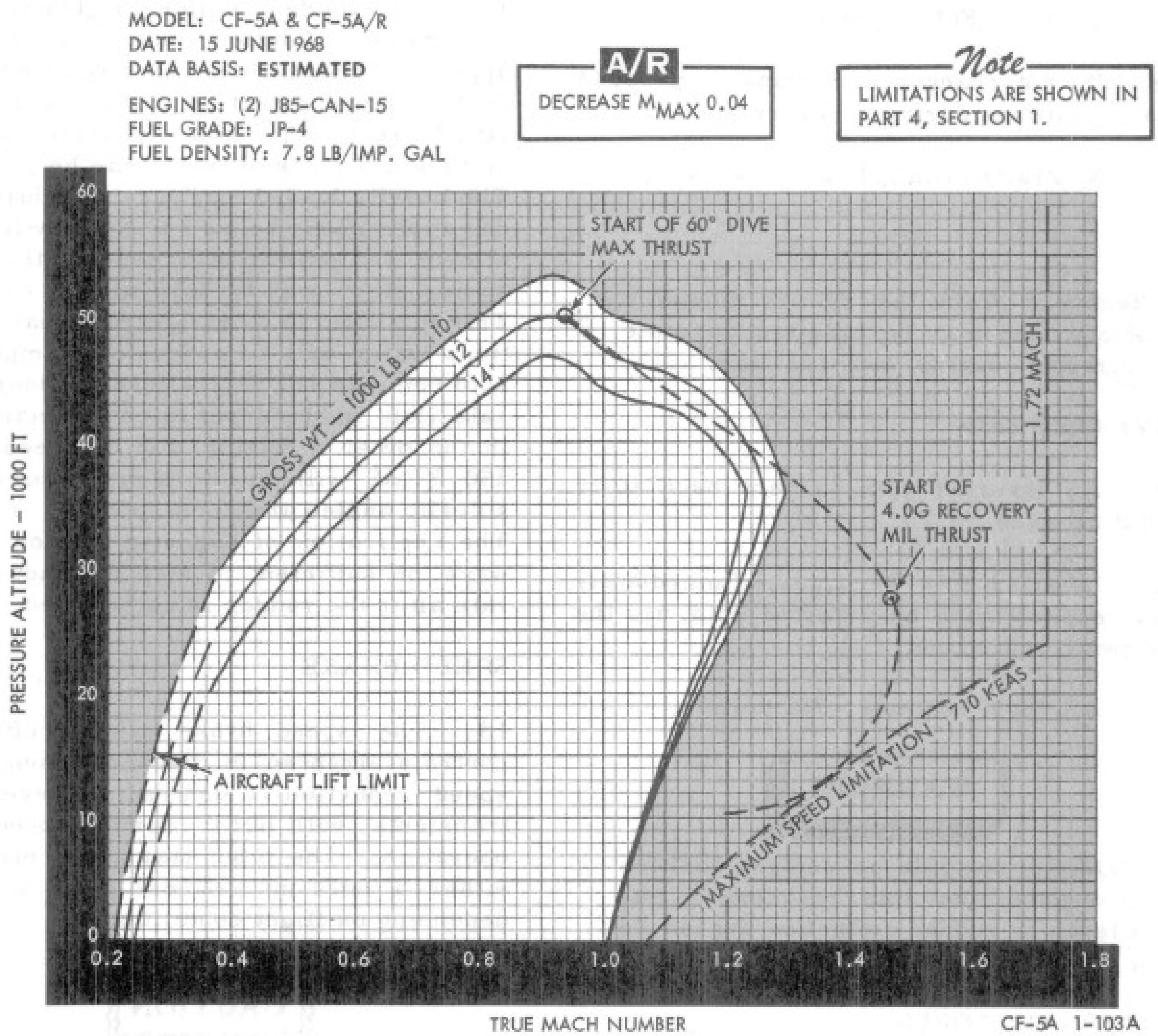

Figure 2-5-3

SPEED PROFILES

19 The speed profile chart shown in Figure 2-5-3 is for unaccelerated level flight conditions. Maximum thrust is maintained in the dive to the start of pullout at which time the thrust of both engines is reduced to military thrust. The lift limits are based on maximum lift conditions at the prevailing mach numbers.

MANOEUVERING FLIGHT

STICK FORCES

20 Stick forces per G are at a minimum in the range of 0.80 to 0.95 mach. When manoeuvering near these mach numbers, be careful not to exceed the allowable load factor by overcontrol.

BOBWEIGHT EFFECT, G-OVERSHOOT

21 The bobweight incorporated in the pitch control system increases the stick forces under G-loads. Since the pilot feels little effect of the bobweight until the aircraft responds to the stick movement, G-overshoots may occur if aft stick is applied too abruptly.

CAUTION

Abrupt forward or aft deflection or "pulsing" of the stick in the mach range 0.80 to 0.95 may result in overshoot of the limit load factor. With pitch augmenter off, aircraft sensitivity about trim and while manoeuvering will be more apparent.

ROLLS

22 Roll rates obtainable in this aircraft with full aileron deflection (spring stop limit) are extremely high and could cause the pilot to become disoriented. Caution should be exercised when using rudder in conjunction with aileron applications during rapid roll or turn entry. Rapid inputs of both rudder and at least half aileron can cause large load factor excursions during the manoeuvre.

DIVING CHARACTERISTICS

HIGH SPEED DIVE

23 Acceleration in a dive is very rapid. Steep dives at high speeds at low altitudes should be avoided because of the large altitude loss during recovery, see Figure 2-5-4. Should a high-speed, steep dive be entered at low altitude, the speed brake should be extended immediately. Use of speed brake does not restrict G's attainable.

FLIGHT WITH EXTERNAL STORES

SYMMETRICAL LOADS

24 Loading of external stores may affect aircraft aerodynamic characteristics or the CG, or both. Loading of centreline stores has a negligible aerodynamic effect on longitudinal stability, while inboard and outboard stores have slight to moderate aerodynamic effect. Centreline stores cause a forward shift in the CG, while inboard stores cause a slight aft CG shift. The CG shift with outboard stores is negligible.

NOTE

Salvo of 4 or 5 BLU-1/B bombs or simultaneous release of the outboard BLU-1/B bombs will cause the aircraft to pitch down approximately 1.0 G. This pitch change will be more pronounced with pitch damper disengaged.

ASYMMETRICAL LOADS

25 The major cause for lateral control corrections with asymmetric store configurations is the weight imbalance; thus the effects are more noticeable at lower airspeeds. Rudder trim is sufficient to trim out yaw produced by asymmetric loads; however, aileron trim may have to be supported by stick forces at low speeds in some cases of asymmetric loads, such as when landing with a full wing pylon tank on one side and opposite pylon tank empty.

26 Should a landing with an extreme asymmetrical loading condition be necessary, it is recommended that the pilot, while at a safe altitude, perform a simulated landing approach at the planned airspeed, to become familiar with the aircraft flight characteristics under this loading configuration. A very flat, straight-in approach, with very little flare, should be made to accomplish a smooth touchdown. The approach and landing should be carefully planned and executed, considering the runway length and increased approach speed. With gear and flaps down add 15 KIAS to computed final approach speed for single asymmetry or 70 KIAS for double asymmetry. It is recommended that asymmetric external stores be jettisoned prior to landing in a strong or gusty crosswind. Landing with asymmetric external stores may result in decreased lateral control of the aircraft.

DIVE RECOVERY CHART

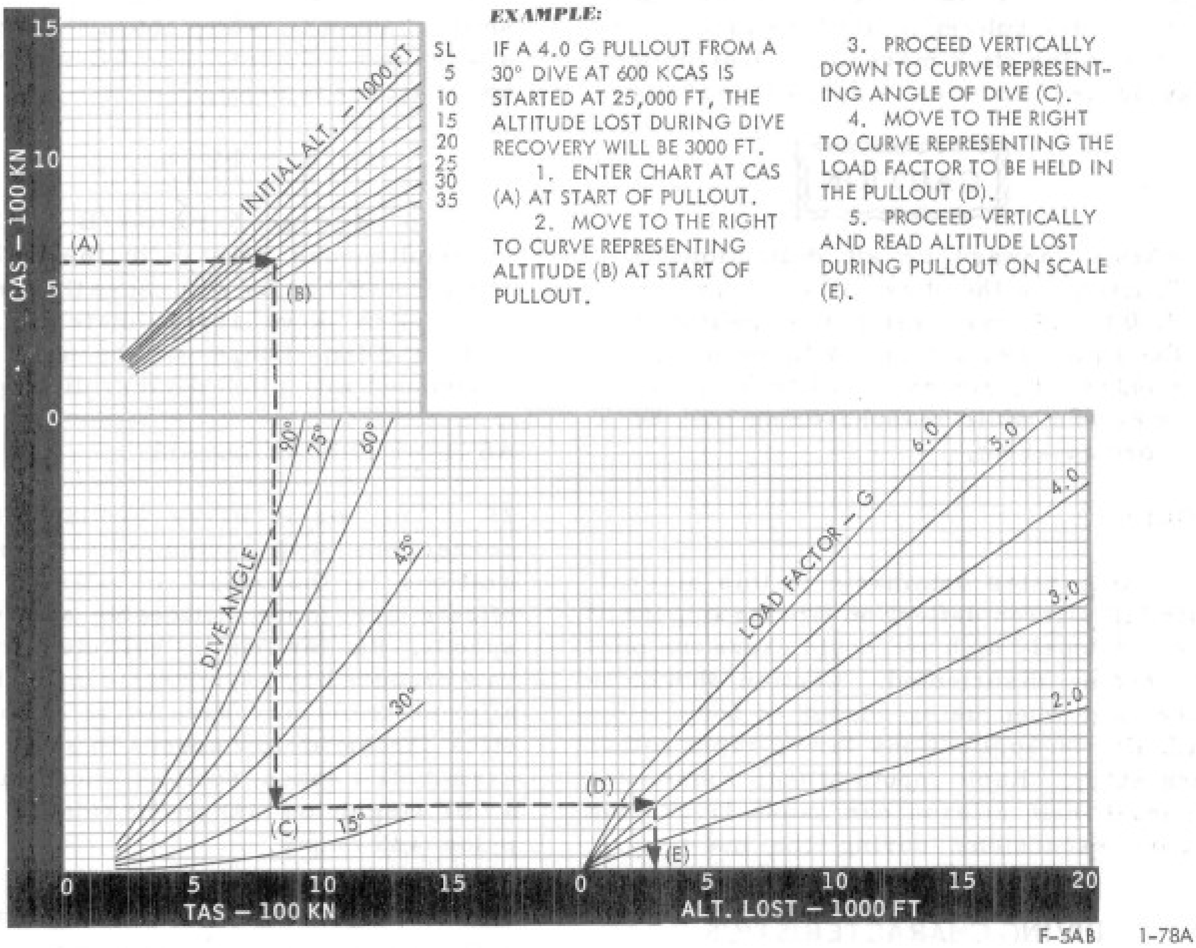

Figure 2-5-4

LANDING WITH EXTREME FORWARD CENTRE OF GRAVITY

27 Landing with a full tank or heavy weapon on the centreline and a full load of ammunition is critical, due to an extreme forward centre-of-gravity position. At forward CG positions with flaps down, the maximum available horizontal stabilizer travel will be limited if approach and touchdown airspeeds are allowed to decrease below the recommended airspeeds.

NOTE

At forward CG positions, landing pattern airspeeds below recommended may result in a nose over and loss of altitude.

SECTION 6
ALL WEATHER OPERATION

INSTRUMENT FLIGHT PROCEDURES

GENERAL

1 Except for some repetition necessary for emphasis or thought continuity, this Part contains only those procedures required in instrument flight that differ from, or are in addition to, the normal operating procedures given in Section 1.

INSTRUMENT TAKEOFF

2 For an instrument takeoff, perform all normal pretakeoff checks, and turn on pitot heat, engine anti-ice, windshield anti-ice, and duct anti-ice switches, if necessary. Takeoff distances should be computed to allow for thrust loss when engine anti-ice system is in operation. Check the horizontal situation indicator for proper heading, and align the index marker on the attitude indicator to a 3-degree nose-low indication. Allowances must be made if aircraft is not on a level surface. This setting should give an approximate level flight indication for intermediate altitude level-offs during departures and at normal cruise conditions. Use normal takeoff procedures. Whenever visibility permits, runway features and lights should be used as an aid to maintain proper headings. Apply aft stick pressure at the recommended aft stick speed to change the pitch attitude from a 3-degree nose-low to an 8-degree nose-high indication at lift-off. Allow the aircraft to fly off the runway and when the vertical velocity indicator and altimeter indicate a definite climb, retract the landing gear. Raise wing flaps after gear has been retracted and aircraft is at a safe altitude and airspeed.

INSTRUMENT CLIMB

3 Approaching 300 KIAS in takeoff attitude, retard throttles to MIL. Maintain a 2 to 5-degree climb indication and at least a 1000-fpm climb until reaching recommended climb schedule. A slow airspeed and/or low rate of climb may be required to comply with departure procedures. For this type climb, reduce power below MIL as required. Power settings between 90% and 95% RPM will provide comfortable climb rates at 300 KIAS for intermediate altitude level-offs. MAX thrust instrument climbs require extremely high pitch angles and are not normally used for instrument departures. If conditions require a MAX thrust climb, maintain a 2 to 5-degree climb indication until approaching recommended climb airspeed/mach; then adjust pitch to maintain climb schedule.

HOLDING PATTERNS

4 Hold at 300 KIAS above 14,000 feet. At or below 14,000 feet, hold at 275 KIAS. To descend in holding patterns, reduce power and maintain holding airspeed in descent. Speed brake may be used for holding pattern descents if higher rates of descent are desired. Fuel consumption in holding patterns between 20,000 and 40,000 feet averages approximately 1500 pounds per hour.

PENETRATION DESCENT

5 To enter a penetration descent, reduce both throttles to 80% RPM and lower the nose approximately 10 degrees on the attitude sphere. Extend speed brake (if required) at 280 KIAS and maintain speed by adjusting pitch as required. Initiate the level-off from a penetration descent 1000 feet or more above the desired altitude by decreasing the pitch attitude approximately one-half. Use normal lead point for level-off at the desired altitude. The speed brake may be left extended or retracted, as required, to obtain the desired airspeed for the approach.

TACAN APPROACH

6 A typical TACAN penetration and approach is shown in Figure 2-6-1. Aircraft configuration and airspeed will depend upon the type of approach being made. Airspeed inbound to the final approach fix should be 220 KIAS with LE flaps. At or prior to the final approach fix, achieve the recommended approach configuration and computed airspeed. For a straight-in approach, maintain 155 KIAS plus weight correction, with gear down and full flaps. For a circling approach, maintain 175 KIAS plus weight correction, with gear down and full flaps. Use full flaps for landing.

RADAR APPROACH

7 A typical radar approach is shown in Figure 2-6-2. Limit all bank angles to 30 degrees. Airspeed on base leg (or 10 miles out if making a straight-in approach) should be 175 KIAS plus weight correction with landing gear down and full flaps. Prior to reaching the glide slope, lower the landing gear and full flaps, and adjust airspeed to 155 KIAS plus weight correction. Maintain the recommended airspeed down the glide slope until a safe landing is assured.

MISSED-APPROACH PROCEDURE

8 To accomplish a missed approach, advance throttles to MIL, retract speed brake (if extended) as power is applied, and rotate the aircraft to a normal instrument takeoff attitude. Retract landing gear, position flaps to LE, and accelerate to 220 KIAS. Reduce power to approximately 95% RPM and climb at 220 KIAS to missed approach altitude.

SINGLE-ENGINE APPROACH

9 A typical radar approach on single-engine is shown in Figure 2-6-3. Delay lowering landing gear until just before glide path. MAX thrust should be used on single-engine approaches if necessary.

SINGLE-ENGINE MISSED APPROACH

10 Use MAX thrust for single-engine missed approach. Landing gear should be retracted as soon as practicable. Position flaps to LE at 165 KIAS, and accelerate to 220 KIAS with MAX thrust.

ICE AND RAIN

GENERAL

11 Anti-icing equipment for wings and empennage is not provided. Each aircraft is provided with engine, duct, and windshield anti-ice systems and, in addition, canopy defog and pitot heat for adverse weather operation. Icing conditions which may be encountered are trace, light, moderate, and heavy. Moderate and heavy icing, particularly, can cause rapid buildup of ice on aircraft surfaces, greatly affecting performance. Short duration climbs and descents may be made through light icing conditions.

WARNING

The aircraft should not be flown in moderate or heavy icing conditions. If any icing is encountered, leave the area of icing condition as soon as possible.

12 When icing conditions are anticipated, the engine, duct, and windshield anti-ice switches should be placed in the on position; pitot heat should be switched on and the canopy defog rheostat switch turned to full increase.

ENGINE ICING

13 Engine inlet guide vane icing may occur when ambient temperature is below 40°F and either the temperature-dewpoint spread is less than 5° or visible moisture is present. Under these conditions and when icing conditions are anticipated, the engine anti-ice switch should be immediately placed in the ENGINE position. This action ensures continuing anti-icing action.

TACAN PENETRATION AND APPROACH (TYPICAL)

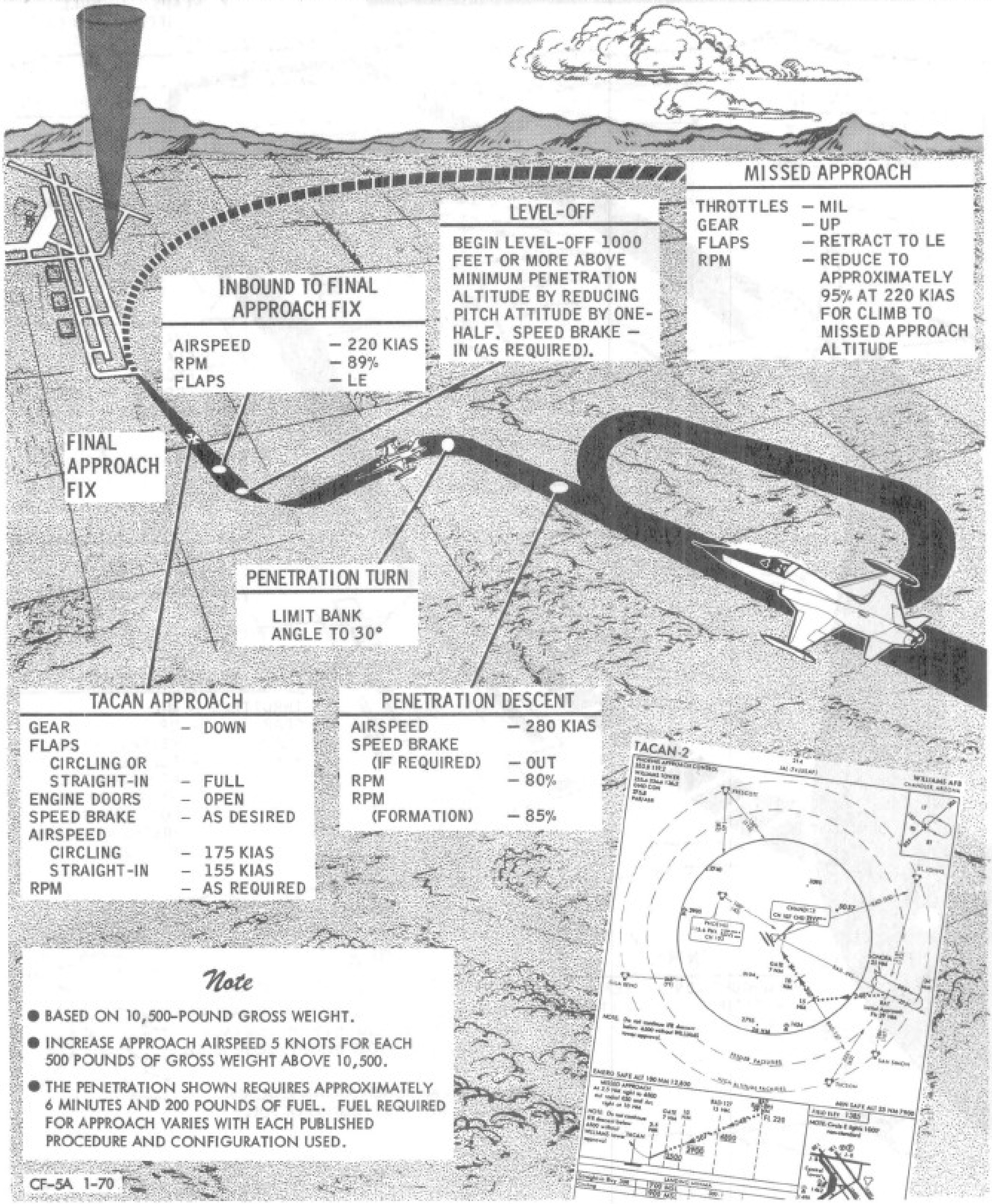

Figure 2-6-1

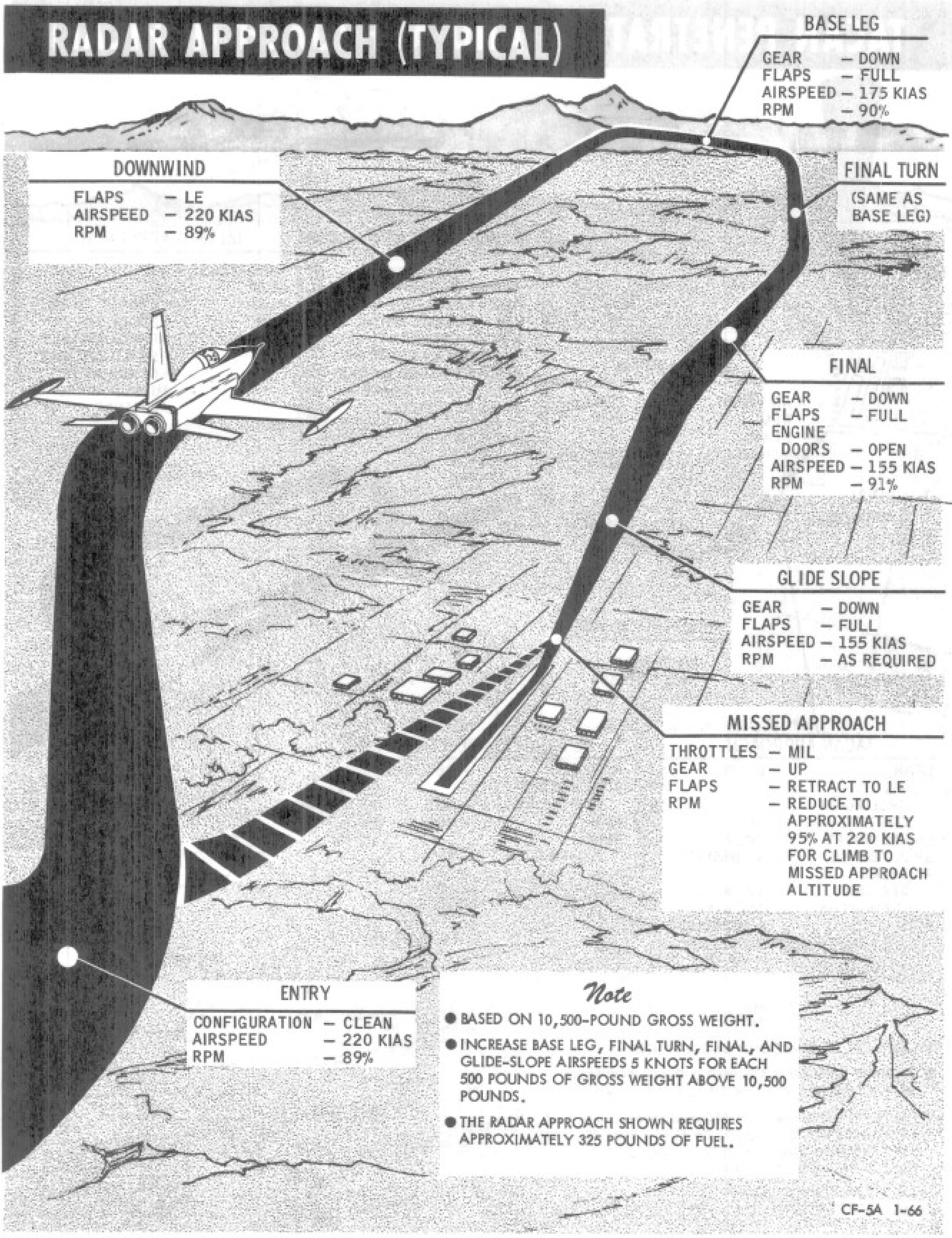
RADAR APPROACH (TYPICAL)
BASE LEG
GEAR – DOWN
FLAPS – FULL
AIRSPEED – 175 KIAS
RPM – 90%
DOWNWIND
FLAPS – LE
AIRSPEED – 220 KIAS
RPM – 89%
FINAL TURN
(SAME AS BASE LEG)
FINAL
GEAR – DOWN
FLAPS – FULL
ENGINE DOORS – OPEN
AIRSPEED – 155 KIAS
RPM – 91%
GLIDE SLOPE
GEAR – DOWN
FLAPS – FULL
AIRSPEED – 155 KIAS
RPM – AS REQUIRED
MISSED APPROACH
THROTTLES – MIL
GEAR – UP
FLAPS – RETRACT TO LE
RPM – REDUCE TO APPROXIMATELY 95% AT 220 KIAS FOR CLIMB TO MISSED APPROACH ALTITUDE
ENTRY
CONFIGURATION – CLEAN
AIRSPEED – 220 KIAS
RPM – 89%
Note
● BASED ON 10,500-POUND GROSS WEIGHT.
● INCREASE BASE LEG, FINAL TURN, FINAL, AND GLIDE-SLOPE AIRSPEEDS 5 KNOTS FOR EACH 500 POUNDS OF GROSS WEIGHT ABOVE 10,500 POUNDS.
● THE RADAR APPROACH SHOWN REQUIRES APPROXIMATELY 325 POUNDS OF FUEL.
CF-5A 1-66

Figure 2-6-2

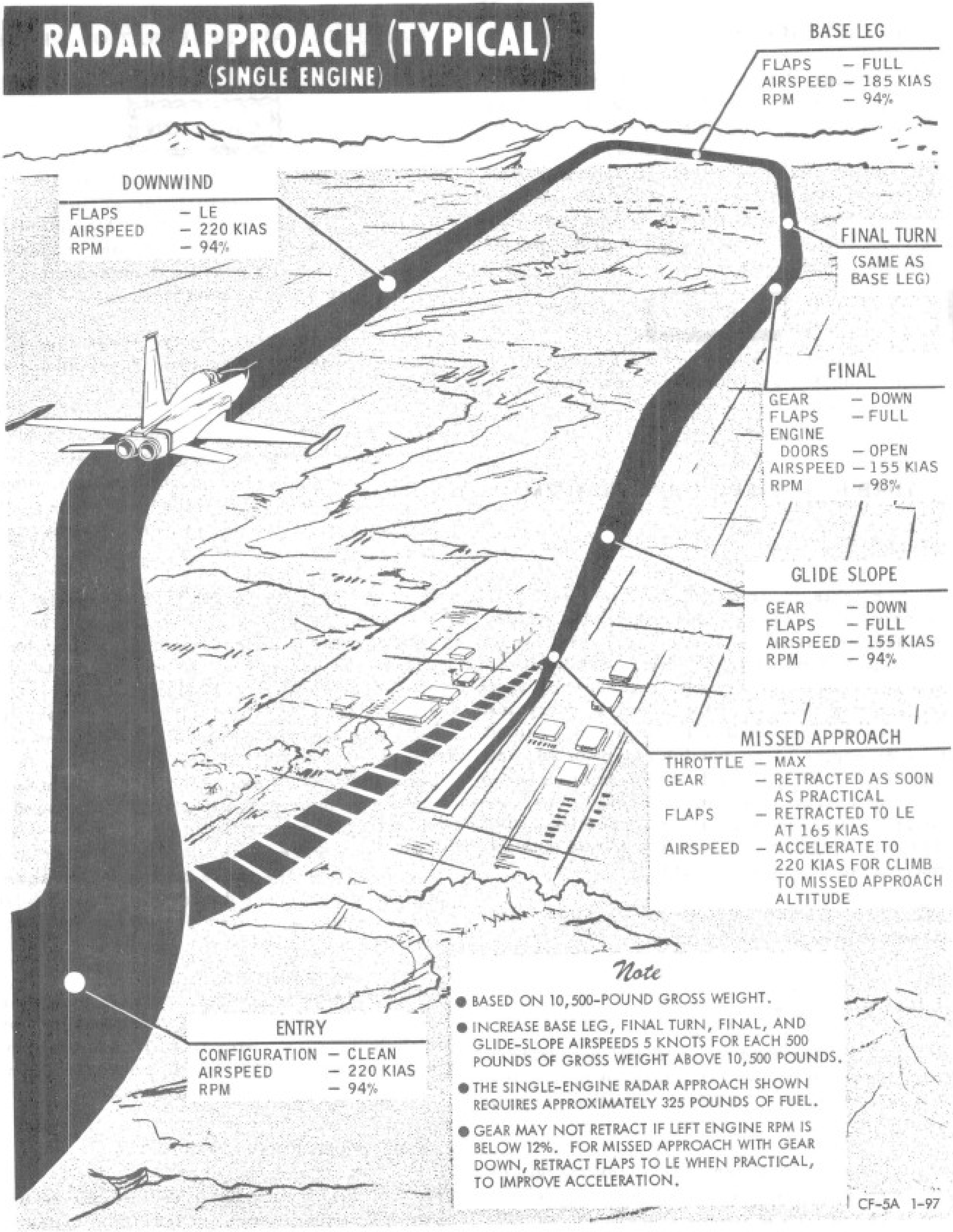
RADAR APPROACH (TYPICAL)
(SINGLE ENGINE)
BASE LEG
FLAPS – FULL
AIRSPEED – 185 KIAS
RPM – 94%
DOWNWIND
FLAPS – LE
AIRSPEED – 220 KIAS
RPM – 94%
FINAL TURN
(SAME AS BASE LEG)
FINAL
GEAR – DOWN
FLAPS – FULL
ENGINE DOORS – OPEN
AIRSPEED – 155 KIAS
RPM – 98%
GLIDE SLOPE
GEAR – DOWN
FLAPS – FULL
AIRSPEED – 155 KIAS
RPM – 94%
MISSED APPROACH
THROTTLE – MAX
GEAR – RETRACTED AS SOON AS PRACTICAL
FLAPS – RETRACTED TO LE AT 165 KIAS
AIRSPEED – ACCELERATE TO 220 KIAS FOR CLIMB TO MISSED APPROACH ALTITUDE
ENTRY
CONFIGURATION – CLEAN
AIRSPEED – 220 KIAS
RPM – 94%
Note
● BASED ON 10,500-POUND GROSS WEIGHT.
● INCREASE BASE LEG, FINAL TURN, FINAL, AND GLIDE-SLOPE AIRSPEEDS 5 KNOTS FOR EACH 500 POUNDS OF GROSS WEIGHT ABOVE 10,500 POUNDS.
● THE SINGLE-ENGINE RADAR APPROACH SHOWN REQUIRES APPROXIMATELY 325 POUNDS OF FUEL.
● GEAR MAY NOT RETRACT IF LEFT ENGINE RPM IS BELOW 12%. FOR MISSED APPROACH WITH GEAR DOWN, RETRACT FLAPS TO LE WHEN PRACTICAL, TO IMPROVE ACCELERATION.
CF-5A 1-97

Figure 2-6-3

INLET DUCT ICING

14 Engine damage may occur if ice is allowed to accumulate on the engine inlet duct lips. Ingestion of accumulated ice into an engine may be evidenced by a jar or noise in the engine and may result in damage to the inlet guide vanes and first-stage compressor blades. Engine instrument indications may remain normal, even though engine damage has been experienced.

WARNING

To prevent possibility of ice buildup on inlet duct lips, duct anti-ice switch should be turned on prior to encountering known icing conditions.

TURBULENCE AND THUNDERSTORMS

GENERAL

15 Flights in areas of turbulent air, hailstorms, and thunderstoms should be avoided, if possible, because of the increased danger of engine flameout and high probability of damage to airframe and components from impact ice, hail, and lightning. If entry into adverse weather conditions cannot be avoided, turn on engine, duct, windshield, pitot anti-ice, and canopy defog prior to penetration. Monitor EGT and engine tachometer indicators continuously to allow for timely corrective action. The following factors, singly or in combination, may cause engine flameouts:

(a) Penetration of cumulus buildups with associated high liquid content.

(b) Engine icing of either nose cowl or inlet guide vanes.

(c) Turbulence associated with penetration can result in angles of attack of plus 9 degrees or more, causing marginal engine performance.

(d) Above 40,000 feet, the surge margin of the engine is reduced and there is poor air distribution across the face of the compressor.

TURBULENT AIR PENETRATION PROCEDURES

CAUTION

Flights through thunderstorms or other areas of extreme turbulence must be avoided whenever possible. Maximum use of weather forecast and radar facilities to help avoid thunderstorms and turbulence is essential.

16 If flight through these areas cannot be avoided, the following procedures should be followed:

(a) Airspeed - Adjust power to establish 300 KIAS. Trim the aircraft for level flight at this speed. Severe turbulence will cause large and rapid variations in airspeed. Do not change thrust except for extreme airspeed variations.

(b) Attitude - The attitude of the aircraft becomes the primary reference in areas of extreme turbulence. Both pitch and bank should be controlled by reference to the attitude indicator. Establish the aircraft attitude for level flight and trim for that attitude and airspeed. Do not change trim settings. Maintain control as near neutral as possible to avoid exaggerating the flight conditions by overcontrolling. Do not use sudden or extreme control inputs. Extreme gusts will cause large attitude changes, but smooth and moderate use of the horizontal stabilizer will reestablish the desired attitude. Do not overcontrol with the rudder; attempt to maintain neutral rudder at all times.

(c) Altitude - Severe vertical gusts will cause appreciable altitude variations. Allow altitude to vary. Sacrifice altitude to maintain the desired attitude. Do not chase altimeter and vertical velocity indications.

PENETRATION SPEED

17 If flight through turbulent air is unavoidable, the recommended "best penetration speed" is 300 KIAS.

WARNING

If inadvertent flight in turbulence and thunderstorms is experienced, do not exceed 400 KIAS.

CAUTION

Flying in turbulence or hail may increase inlet distortion. At higher altitudes this distortion can result in engine surge and possible flameout. However, normal air restarts may be accomplished as outlined in Part 3, Emergency Handling.

NIGHT FLYING

18 This aircraft is adequately equipped with lighting and instrumentation and presents no problem in night flying.

NOTE

The red rotary beacon light should be turned off in the vicinity of clouds or before entering cloud formation to prevent spatial disorientation.

COLD WEATHER OPERATION

GENERAL

19 Most cold weather operating difficulties are encountered on the ground. The following instructions are to be used with the normal procedures given in Part 2 when cold weather aircraft operation is necessary.

BEFORE ENTERING AIRCRAFT

20 Remove all protective covers and duct plugs; check to see that all surfaces, ducts, struts, drains, and vents are free of all snow, ice, and frost. Brush off all light snow and frost. Remove all ice and encrusted snow, either by a direct flow of air from a portable ground heater or by using deicing fluid. Remove light frost from the windshield and canopy with a clean soft rag.

WARNING

- Depending on the weight of snow and ice accumulated, takeoff distances and climb-out performance can be seriously affected. The roughness and distribution of the ice and snow could vary stall speeds and characteristics to an extremely dangerous degree. Loss of an engine shortly after takeoff is a serious enough problem without the added (and avoidable) hazard of snow and ice on the wings. In view of the unpredictable and unsafe effects of such a practice, the ice and snow must be removed before flight is attempted.
- Ensure that water does not accumulate in control hinge areas or other critical areas where refreezing may cause damage or binding.

CAUTION

To avoid damage to aircraft surfaces, do not permit ice to be chipped or scraped away.

21 Check the fuel tank vents on the vertical stabilizer for freedom from ice. Inspect aircraft carefully for fuel and hydraulic leaks caused by contraction of fittings or by shrinkage of packings. Inspect area behind aircraft to ensure that water or snow will not be blown onto personnel and equipment during engine start.

ENGINE START

22 Use external power for starting to conserve the battery. No preheat or special starting procedures are required. Turn on cockpit heat and canopy defog system, as required, immediately after engine start.

NOTE

At very cold ambient temperatures (approximately -30°F and below), delay use of canopy defog until engines are in operating range, since only a cold air blast is produced at idle speeds prior to warmup.

WARMUP AND GROUND CHECK

23 After engine start, oil pressure indications above 55 psi will be observed. As the oil warms up, pressure should reduce to within operating limits. To decrease oil warmup time, the engine may be operated at MIL power until pressure is within limits. If oil pressure does not decrease within 6 minutes at MIL power, the engine should be shut down. Slightly lower idle speeds are to be expected with cold engines and a small advancement of throttles may be necessary to place the generators on the line. When engines are sufficiently warmed up, check flight controls, speed brake, and aileron trim for proper operation. Cycle flight controls 4 to 6 times. Check hydraulic pressure, control reaction, and operation of all instruments.

TAXIING

24 Nosewheel steering effectiveness is reduced when taxiing on ice and hard packed snow. A combination of nosewheel steering and wheel braking should be used for directional control. The nosewheel will skid sideways easily, increasing the possibility of tire damage. If conditions permit, taxi with one engine at idle and the other at high rpm (70% to 80%) to provide more heat for the cockpit and canopy and windshield defrosting. However, reduced speeds will generally be necessary when taxiing over the uneven snow and ice covered surfaces common in low temperature environments. Increase the normal interval between aircraft, both to ensure a safe stopping distance and to prevent icing of aircraft surfaces from melted snow and ice caused by the jet blast of the preceding aircraft. Minimize taxi time to conserve fuel and reduce the amount of ice fog generated by the engines. If bare spots exist through the snow, skidding onto them should be avoided.

WARNING

Make sure all instruments have warmed up sufficiently to ensure normal operation. Check for sluggish instruments while taxiing.

TAKEOFF

25 Use normal throttle procedures during takeoff.

SCRAMBLE TAKEOFF

26 When the temperature is 32°F or below and operational requirements dictate, it is permissible to take off when a decreasing indication in oil pressure has been established and pressure indications have decreased to 95 psi or below. If operating at military power or in afterburner, the oil pressure should decrease to normal operating limits within approximately 6 minutes. If the pressure does not return to normal within the time limit, the throttle should be retarded as required to decrease the pressure to an acceptable limit. If lowering the power setting does not decrease the oil pressure within limits, it is cause for engine shutdown.

LANDING

27 Use minimum run landing techniques given in Section 1. When landing on runways that have patches of dry surface, avoid locking the wheels. If the aircraft starts to skid, release brakes until recovery from skid is accomplished.

ENGINE SHUTDOWN

28 Use normal engine shutdown procedure.

BEFORE LEAVING AIRCRAFT

29 The canopy should be fully closed on aircraft parked outdoors to prevent the entry of blowing snow caused by operation of other aircraft or from natural conditions.

HOT WEATHER AND DESERT OPERATION

GENERAL

30 Operation in hot weather and desert requires that precautions be taken to protect

the aircraft from damage caused by high temperatures, dust, and sand. Care must be taken to prevent the entrance of sand into aircraft parts and systems such as the engines, fuel system, pitot-static system, etc. All filters should be checked more frequently than under normal conditions. Plastic and rubber segments of the aircraft should be protected both from high temperatures and from blowing sand. Canopy covers should be left off to prevent sand from accumulating between the cover and the canopy and acting as an abrasive on the plastic canopy. With a canopy closed, cockpit damage may result when ambient temperature is above 110°F. Canopy should be opened in advance of flight to reduce cockpit temperature for comfort. Desert and hot weather operation require that in addition to normal procedures, the following precautions be observed.

ENTERING AIRCRAFT

31 During preflight inspection and upon entering aircraft, it is recommended that light flying gloves be worn since aircraft surfaces are extremely hot in high ambient temperatures.

TAKEOFF

32 Use normal takeoff procedures and be alert for gusts and wind shifts near the ground. At high gross weights (approximately 19,000 lb), the interval between application of aft-stick pressure and takeoff will be noticeably longer.

DURING FLIGHT

33 The canopy defog system should be operated at the highest flow possible (consistent with pilot's comfort) for 10 minutes prior to descent from high altitude flight to preheat the transparent surfaces and prevent the formation of frost or fog during descent.

APPROACH AND LANDING

34 Monitor airspeed closely to ensure that recommended approach and touchdown airspeeds are maintained; high ambient temperatures cause speed relative to the ground to be higher than normal. Anticipate a long landing roll due to higher ground speed at touchdown.

SECTION 7

WEAPONS SYSTEM – DESCRIPTION AND OPERATING PROCEDURES

GENERAL

1 The aircraft is equipped with a lead computing optical sight system to assist in weapons delivery of a variety of external armament and for aiming the two 20mm guns. Five jettisonable pylons, at two outboard and two inboard wing stations and a fuselage centreline station, are capable of carrying multiple configurations of general purpose and practice bombs, fire bombs, and rocket launchers. Provision for carriage of wingtip missiles is incorporated into the electrical armament system of the aircraft, although the aircraft is not equipped with missile launcher rails.

AUTHORIZED CONFIGURATIONS

2 See Figure 4-1-4 for the external weapon and fuel tank configurations authorized to be carried on the aircraft.

OPTICAL SIGHT SYSTEM

GENERAL

3 The optical sight system, type AWQ-501, consists of the sight head, sight control panel, sight computer, and the sight twist grip. The system is a computing gyro gun sight providing a predicted sight line for aiming various weapons during air-to-air and air-to-ground modes of attack. Predicted sight lines are available for air-to-air gun and missile attacks and for air-to-ground gun, bomb, and rocket attacks.

SIGHT HEAD

4 The optical system of the sight head consists of a moving convex mirror mounted on a gyro rotor, a fixed mirror, collimating lens and the reflector glass mounted on top of the sight head. The reticle display consists of a 2-mil aiming dot (pipper) and two reticle arcs symmetrically located on either side of the pipper. The distance apart and length of the two reticle arcs are controlled by the span setting control and the twist grip on the left throttle. The position of the pipper, which is roll stabilized, in each mode of attack provides the required lead angle, gravity drop, and drift corrections. The sight head contains the following four controls and indicators:

(a) Span control, which allows wing span settings from 20 to 120 feet for stadiametric ranging in the guns air-to-air mode.

(b) Dimmer control for adjusting the illumination of the reticle display.

(c) Lamp change-over switch for switching to standby reticle display illuminating lamp if the main illuminating lamp fails.

(d) Range dial to provide direct read-off of range from 500 to 3000 feet during stadiametric ranging in the guns air-to-air mode.

SIGHT CONTROL PANEL

5 The sight control panel on the pedestal contains the mode selector, drift control, and depression control.

MODE SELECTOR

6 The mode selector is a 7-position rotary switch with an OFF position and the following mode selections:

GA	Guns	Air-to-Air
M	Missiles	Air-to-Air
G	Guns	Air-to-Ground
R	Rockets	Air-to-Ground
B	Bombs	Air-to-Ground
S	Manual (Sight Line)	

7 Positioning the mode selector from OFF to any one of the six mode positions electrically energizes the sight computer and the gyro motor in the sight head. The function of each mode selection is as follows:

(a) GA (Guns air-to-air) provides aiming, tracking, and ranging capability for lead pursuit attacks initiated from above, co-altitude, or below the target, at firing ranges from 600 to 2500 feet. The sight line prediction angle compensates for gravity drop, range, altitude, and the relative motion of the target and the attacking aircraft.

(b) M (Missile air-to-air) provides a fixed sight for aiming missiles in a pure pursuit attack.

(c) G (Guns air-to-ground) provides a computed predicted sight line allowing for gravity drop, at firing ranges from 1000 to 4000 feet. A wind correction is automatically applied when the pipper is kept on the target during the attack.

(d) R (Rockets air-to-ground) provides a computed predicted sight line allowing for gravity drop, true airspeed, pitch, and drift at fixed firing range of 3000 feet.

(e) B (Bombs air-to-ground) provides a computed predicted sight line allowing for gravity drop, true airspeed, pitch, and drift at all release ranges from 2500 to 7500 feet manually preset on the sight computer.

(f) S (Manual sight line) provides manual control of the predicted sight line by setting the drift and depression controls.

DRIFT CONTROL

8 The drift control provides the capability of setting ±5° drift when the mode selector is set at either R (rockets), B (bombs), or S (manual) modes. The AUTO position on the control is inoperative.

DEPRESSION CONTROL

9 The depression control allows selection of sight line depression angles from 0 to 200 mils when the mode selector is at S (manual).

SIGHT COMPUTER

10 The sight computer houses the computing portions of the sight system. The computer receives true airspeed and altitude data from the air data computer, aircraft pitch and roll axis information from the GHARS system, and pilot controlled range inputs. The range, reticle, and pitch and roll servos of the computer, along with the altitude and true airspeed data, provide a corrected prediction sight line for the various attack modes. The computer contains a range selector which allows a preset range adjustment of 2500 to 7500 feet for the bombing mode.

SIGHT TWIST GRIP

11 A sight twist grip on the outboard side of the left throttle provides manual ranging by the pilot while tracking in the GA air-to-air mode. The twist grip is provided with two stop positions which limit rotation of the grip to 90° of travel. Rotating the twist grip adjusts the reticle display arcs, drives the range and reticle servos in the computer, and drives the range dial on the sight head for direct range read-off.

ARMAMENT CONTROLS

GENERAL

12 Armament controls are shown in Figure 2-7-1. Controls consist of the following:

(a) Optical sight and controls.

(b) Left vertical armament control panel, containing the following:

(1) AIM-9 volume control switch (inoperative).

(2) Guns and camera switch.

(3) Bombs arm switch.

(4) External stores armament mode rotary selector switch.

(5) Emergency all-jettison pushbutton.

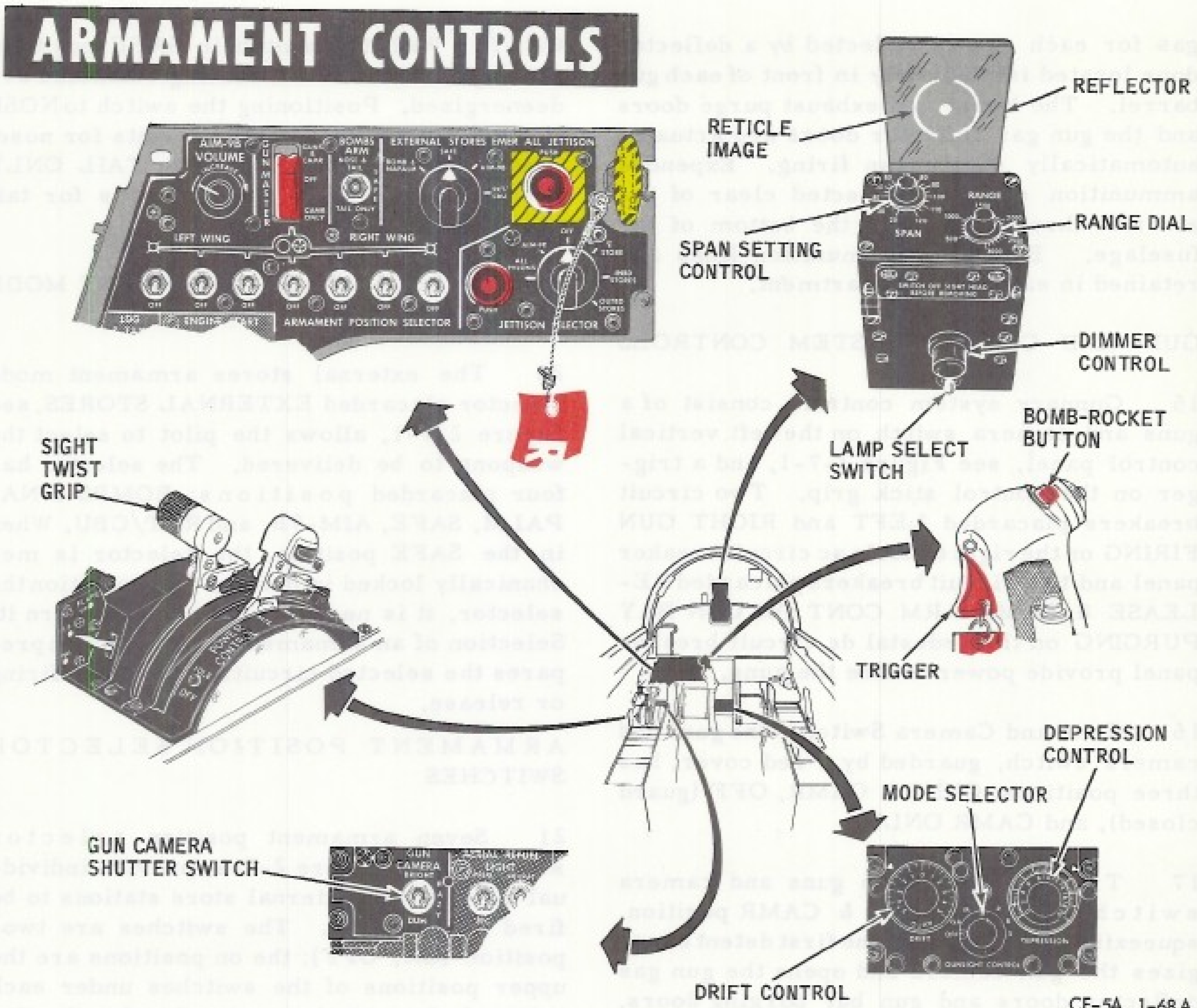

Figure 2-7-1

(6) External stores jettison T-handle (thermal battery power).

(7) Armament position selector switches.

(8) Jettison selector and pushbutton.

(c) Control stick grip containing bomb-rocket button and trigger.

GUN CAMERA SYSTEM

13 An N-9 16mm motion picture camera is mounted above the instrument panel forward and to the right of the optical sight. Before takeoff, the desired frame speed, overrun time, and lens diaphragm (f-stop) setting, are preset in the camera. A camera shutter switch, see Figure 2-7-1, on the left console forward of the throttle quadrant allows the pilot to override the preset lens diaphragm setting by repositioning the setting to BRIGHT, HAZY, or DULL to suit prevailing light conditions.

GUNNERY SYSTEM

14 Two M-39A2 or M-39A3 20mm guns are mounted in the upper nose section of the aircraft. The maximum number of rounds loaded for each gun is 280 rounds, providing 11.2 seconds of continuous gun fire at a rate of 1500 rounds per minute. The guns are manually charged prior to takeoff. During the gun firing sequence, both gun bays are purged of explosive gases by ram air intake and exhaust doors on each side of the nosewheel well. The retractable doors open when the trigger is squeezed to the first detent, remain open at trigger second detent while the guns are firing, and remain open for approximately 10 seconds after the trigger is released. Gun

gas for each gun is deflected by a deflector door located immediately in front of each gun barrel. The intake and exhaust purge doors and the gun gas deflector doors are actuated automatically during gun firing. Expended ammunition cases are ejected clear of the aircraft through tubes at the bottom of the fuselage. Expended ammunition links are retained in each gun compartment.

GUN AND CAMERA SYSTEM CONTROLS

15 Gunnery system controls consist of a guns and camera switch on the left vertical control panel, see Figure 2-7-1, and a trigger on the control stick grip. Two circuit breakers placarded LEFT and RIGHT GUN FIRING on the right console ac circuit breaker panel and two circuit breakers placarded RELEASE & BOMB ARM CONT and GUN BAY PURGING on the pedestal dc circuit breaker panel provide power to fire the guns.

16 Guns and Camera Switch. The guns and camera switch, guarded by a red cover, has three positions: GUNS & CAMR, OFF (guard closed), and CAMR ONLY.

17 Trigger. When the guns and camera switch is in the GUNS & CAMR position, squeezing the trigger to the first detent energizes the gun camera and opens the gun gas deflector doors and gun bay purging doors. At the second (fully squeezed) trigger detent, gun camera operation continues and the guns fire. With the switch in CAMR ONLY position, squeezing the trigger to the first detent will energize the gun camera.

BOMB-ROCKET BUTTON

18 The bomb-rocket button on the stick grip is the normal release control for bombs, missiles, and rockets.

BOMBS ARM SWITCH

19 The bombs arming switch on the left vertical control panel has three positions placarded SAFE, NOSE & TAIL, and TAIL ONLY. With the switch in SAFE position, arming circuits to all arming solenoids are deenergized. Positioning the switch to NOSE & TAIL prepares arming circuits for nose, centre, and tail arming. The TAIL ONLY position prepares arming circuits for tail arming only.

EXTERNAL STORES ARMAMENT MODE SELECTOR

20 The external stores armament mode selector placarded EXTERNAL STORES, see Figure 2-7-1, allows the pilot to select the weapons to be delivered. The selector has four placarded positions: BOMBS & NAPALM, SAFE, AIM-9B, and RKT/CBU. When in the SAFE position, the selector is mechanically locked in detent. To reposition the selector, it is necessary to push and turn it. Selection of an armament mode position prepares the selected circuits for weapons firing or release.

ARMAMENT POSITION SELECTOR SWITCHES

21 Seven armament position selector switches, see Figure 2-7-1, provide individual selection of external store stations to be fired or released. The switches are two-position (ON, OFF); the on positions are the upper positions of the switches under each external store station grouped under the aircraft silhouette on the panel. The wingtip positions are provided for wingtip missile firing, when equipped.

AIM-9B MISSILE VOLUME CONTROL

22 Inoperative.

EMERGENCY ALL JETTISON PUSHBUTTON AND EXTERNAL STORES JETTISON T-HANDLE

23 For operation of the emergency all jettison pushbutton and external stores jettison T-handle controls, refer to Part 1, Section 4. See Figure 2-7-1 for locations of controls.

SECTION 8
PHOTOGRAPHIC RECONNAISSANCE SYSTEM OPERATION

GENERAL

1 The procedures herein cover the operation of the camera control (MCA-3) as a part of the camera control system (CCS-1). Refer to Part 1, Section 21, for complete system description.

PREFLIGHT OPERATION

2 The following functions shall be checked by ground crew (or pilot) prior to commencing mission. See Figure 2-8-1 for location of camera control panel switches.

(a) Master Switch - OFF.

(b) Operation Switch - STICK.

(c) Mode Switch - OFF.

(d) Frames Remaining Counters - Push and turn counter reset knobs until 500 frames appears in appropriate digital readout window(s).

(e) Camera Select - OFF (joystick centred and pulled out).

(f) Groundspeed - As Required.

(g) Height Scale Switch - As Required.

(h) Height Above Ground - As Required.

FLIGHT OPERATION

3 Prior to entering target area, recheck and set switches and controls for mission requirements.

(a) Groundspeed - Set.

(b) Height Scale - As Required.

(c) Height Above Ground - Set.

NOTE

The height scale switch should not be changed when the camera(s) are operating.

(d) Master Switch - MAN or 4FPS (as required).

(e) Mode Switch - DAY or NIGHT (as appropriate).

NOTE

- If subliminal light condition exists, the APC LIMIT lamp will be illuminated, indicating marginal light for mission.
- Refer to Flight Procedure With APC LIMIT Light On, following paragraph.

(f) Joystick - Select necessary cameras.

(1) With the operation switch in the STICK position, cameras will operate by pressing the nosewheel steering button on the control stick grip, see Figure 1-8-1. The cameras will operate only as long as this switch is pressed.

(2) With the operation switch in continuous (CONT) position the cameras will operate continuously.

(g) If any of the selected cameras are in the 8FPS mode, the FREE RUN (blue 8) lamp will be illuminated.

(h) Camera operation is indicated by illumination of the camera operate blue lights on the instrument panel, see Figure 1-21-1.

4 Alternate Flight Operation With APC Limit Light On:

(a) Select master switch to 4FPS. This results in a full stop increase in exposure since

CAMERA CONTROL PANEL A/R

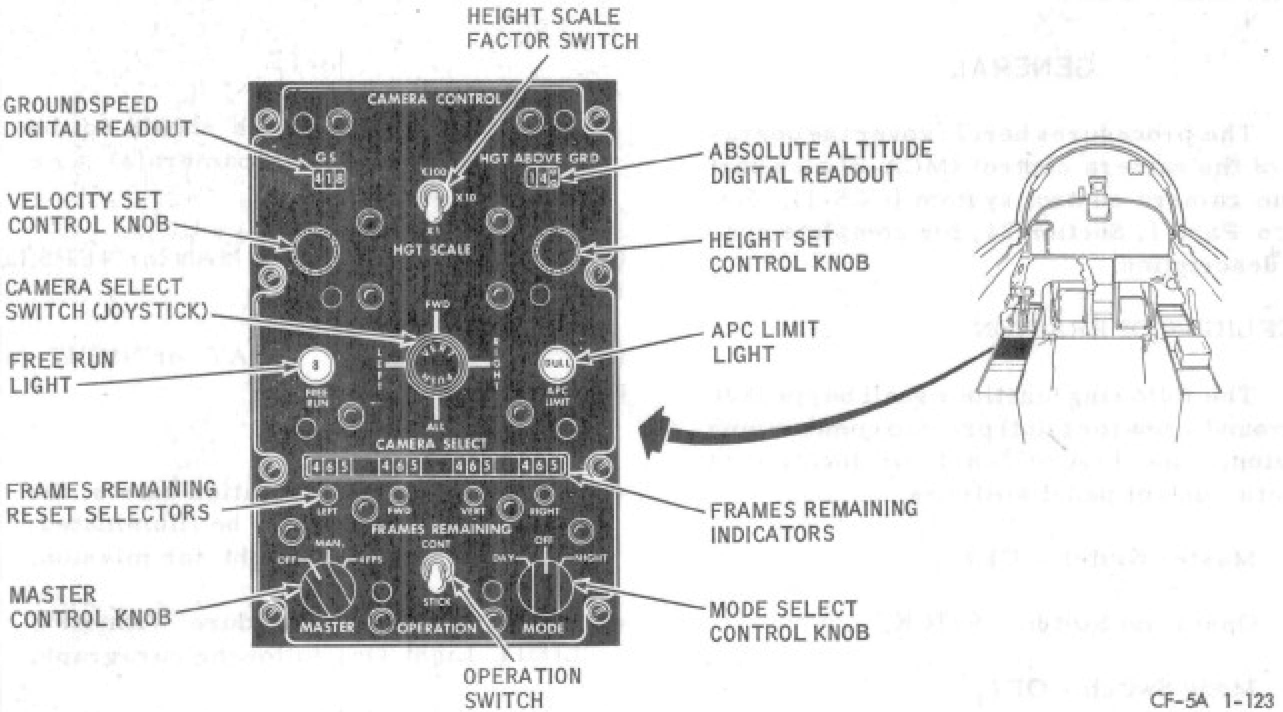

Figure 2-8-1

the shutter speed is reduced from 1/3000 to 1/1500 second; however, all selected cameras will run at 4FPS; or:

(b) Select OFF position on mode select switch. This results in a full stop increase in exposure for the 1.75-inch lens and the 6.0-inch lens, and two full stops with the 3.0-inch lens. Automatic pulsing and shutter speed of 1/3000 second is still maintained with this setting.

(c) In extreme low light conditions, position master switch to 4FPS and mode select switch to OFF.

SHUTDOWN OPERATION

5 To secure camera operation at end of film roll or mission, use the following steps:

(a) Camera Select Joystick - Centred and Out position.

(b) Master Switch - OFF.

(c) Operation Switch - STICK.

(d) Mode Switch - OFF.