McDonnell-Douglas F4H-1F

Flight Manual

01-245FDA-1   1962

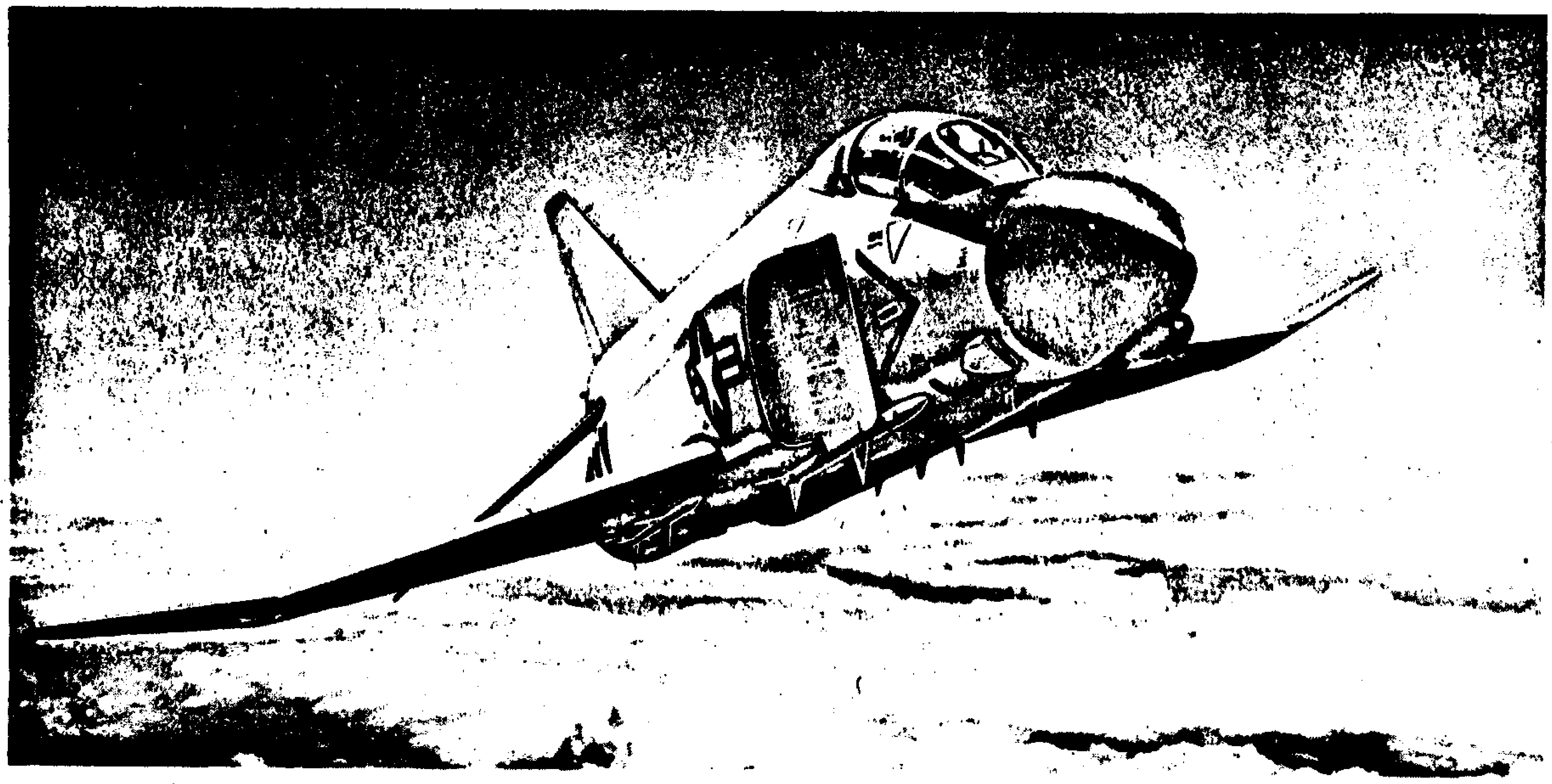
NAVWEPS 01-245FDA-1

Flight Manual

NAVY MODEL

F4H-1F
AIRCRAFT

THIS PUBLICATION IS INCOMPLETE WITHOUT CONFIDENTIAL SUPPLEMENT
NAVWEPS 01-245FDA-1A



PUBLISHED BY DIRECTION OF THE CHIEF OF THE BUREAU OF NAVAL WEAPONS

15 March 1962
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LIST OF REVISED PAGES ISSUED

INSERT LATEST REVISED PAGES. DESTROY SUPERSEDED PAGES.

NOTE: The portion of the text affected by the current revision is indicated by a vertical line in the outer margins of the page.

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* The asterisk indicates pages revised, added or deleted by the current revision.

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For listing of available material and details of distribution see Naval Aeronautics Publications Index NavWeps 00-500.

NAVWEPS 01-245FDA-1
28 NOVEMBER 1962

FLIGHT HANDBOOK INTERIM REVISION No. 11

Navy Model F-4A (F4H-1F) Aircraft

PUBLISHED BY DIRECTION OF THE CHIEF OF THE BUREAU OF NAVAL WEAPONS

Of paramount interest to pilots. To be read by all pilots operating these aircraft

1. Cancellation: BuWeps message 292047 Z of August 1962.

2. Purpose: To provide instructions and procedures in the event of illumination of generator warning light.

3. The following changes are made to Flight Handbook, Navy Model F4H-1F aircraft, NAVWEPS 01-245FDA-1 dated 15 March 1962 Revised 15 September 1962.

   Change Section III, Page 98 under single generator failure to read:

   3. Generator switch off
   4. Secure engine if practicable
   5. Land as soon as practicable

END

FLIGHT HANDBOOK INTERIM REVISION No. 10

Navy Model F-4A (F4H-1F) **Aircraft**

PUBLISHED BY DIRECTION OF THE CHIEF OF THE BUREAU OF NAVAL WEAPONS

Of paramount interest to pilots. To be read by all pilots operating these aircraft

1. CANCELLATION. This interim revision cancels message Interim Revision No. 9 of 19 September 1962 (191606Z).

2. PURPOSE. To set forth high speed-low altitude information to be observed during the operation of F-4A (F4H-1F) aircraft.

3. The following changes are made to the Flight Manual, NAVWEPS 01-245FDA-1 of 15 March 1962, revised 15 Sep. 1962, for Navy Model F-4A (F4H-1F) Aircraft.

a. Section VI, page 145, under the paragraph entitled HIGH SPEED FLIGHT AT LOW ALTITUDE:

(1) Delete the first sentence and substitute the following:

The stabilator sensitivity is very pronounced in the transonic region at low altitudes. With stability augmentation inoperative at aft C.G. locations this sensitivity contributes to undesirable and potentially hazardous flight characteristics.

(2) Add the following CAUTION on page 146 just prior to paragraph entitled SUPERSONIC FLIGHT:

CAUTION

With stability augmentation "OFF" at C.G. locations aft of 32.0% MAC, recent Navy tests have shown that an undamped longitudinal oscillation can occur at speeds above 350 knots CAS below 10,000 feet altitude with the STICK FREE. An abrupt input such as that obtained when suddenly releasing the stick in an accelerated maneuver, or the transient obtained when the stability augmentation monitors "OFF" in some accelerated flight conditions, will initiate the undamped motion. The entire longitudinal control system, including the stick, moves during this oscillation, which has a frequency of about ½ cycle per second. This stick free, undamped oscillation may increase the possibility of inadvertently obtaining dangerous pilot induced oscillations (PIO's) if the pilot attempts to damp the motion.

The problem is being investigated and at present appears to be related to control system dynamics. Available information indicates that:

a. To stop the oscillation the stick must be held firmly in the trim position. Attempts to stop the airplane motion by applying compensating longitudinal control inputs may lead to PIO's with increasing amplitudes. On the other hand, releasing the stick will not stop the motion but will result in an undamped oscillation. The characteristic is different from the usual PIO's where corrective action normally requires immediate release of the stick.

b. Because of the aforementioned control system dificulties, observe the CAUTION under AIRSPEED LIMITATIONS, Section V, NAVWEPS 01-245FDA-1A.

END

NAVWEPS FORM 5600/4A (3-61)

NAVWEPS 01-245FDA-1
7 SEPTEMBER 1962

FLIGHT HANDBOOK INTERIM REVISION No. 8

Navy Model F4H - 1F Aircraft

PUBLISHED BY DIRECTION OF THE CHIEF OF THE BUREAU OF NAVAL WEAPONS

Of paramount interest to pilots. To be read by all pilots operating these aircraft

1. CANCELLATION: This interim revision cancels Message Interim Revision No. 7 (292048Z) 29 Aug. 62.

2. PURPOSE: The purpose of this interim revision is to prevent damage to the utility hydraulic system reservoir due to incorrect cockpit procedures involving emergency actuation of the landing gear and the wing flaps.

3. INSTRUCTION: The following changes are made to the Flight Handbook, NAVWEPS 01-245FDA-1, revised 15 June 1962 for the Navy Model F4H-1F Aircraft.

   a. Section III Page 101 Landing Gear System Emergency Operation

   Delete:

   "There are no provisions for emergency retraction of the landing gear in flight."

   Replace with the following:

   "Do not retract the landing gear following an emergency extension.

   CAUTION

   If the landing gear is inadvertently extended in flight by emergency pneumatic pressure, it must be left in the extended position until post-flight servicing. If retraction in flight is attempted, rupture of the utility reservoir will probably occur with subsequent loss of the utility hydraulic system."

   b. Section III Page 102 Wing Flaps System Emergency Operation

   Delete:

   "There are no provisions for emergency retraction of the flaps in flight."

   Replace with the following:

   "Do not retract the flaps following an emergency extension.

   CAUTION

   If the flaps are inadvertently extended in flight by emergency pneumatic pressure, they must be left in the extended position until post-flight servicing. If retraction in flight is attempted, rupture of the utility reservoir will probably occur with subsequent loss of the utility hydraulic system."

END

INTERIM REVISION SUMMARY

The following Interim Revisions have been either canceled or incorporated in this Flight Manual:

Canceled or Previously Incorporated	*Incorporated in This Revision on Pages Indicated*	
No. 1, 2, 3 and 4	No. 5	Page 62

INTERIM REVISIONS OUTSTANDING: (to be maintained by custodian of Flight Manual)

Number	*Date*	*Purpose*

AIRCRAFT SERVICE CHANGE SUMMARY

The following ASC's are of direct interest to the crewmembers and are noted throughout the Flight Manual:

ASC NO.	TITLE	SECTION
84	Electrical Fuel Transfer System Circuitry Change (Provides for Fuel Transfer into No. 5 Fuselage Tank)	I

TABLE OF CONTENTS

* REFER TO CONFIDENTIAL SUPPLEMENT NAVWEPS 01-245FDA-1A

FM-FDA-TCD

!DRAWROF

THE F4H-1F FLIGHT MANUAL WILL ASSURE THAT YOU ARE PROCEEDING IN THE RIGHT DIRECTION....

SCOPE This manual and its classified supplement NAVWEPS 01-245FDA-1A comprises the Flight Manual. The manufacturer and the NAVY have presented this information in a form best suited for your use. The manual contains all the information necessary for safe and efficient operation of the F4H-1F. These instructions do not teach basic flight principles, but are designed to provide you with a general knowledge of the airplane, its flight characteristics, and specific normal and emergency operating procedures. Your flying experience is recognized, and elementary instructions have been avoided.

SOUND JUDGEMENT The instructions in this manual are designed to provide for the needs of a crew inexperienced in the operation of this airplane. This book provides the best possible operating instructions under most circumstances, but it is a poor substitute for sound judgement. Multiple emergencies, adverse weather, terrain, etc., may require modification of the procedures contained herein.

STANDARDIZATION Once you have learned to use one Flight Manual, you will know how to use them all - closely guarded standardization assures that the scope and arrangement of all Flight Manuals are identical.

ARRANGEMENT The manual has been divided into nine sections and an appendix. The objective of this subdivision is to make it easy both to read the book straight through when it is first received and thereafter to use it as a reference manual. Section I is a detailed description and discussion of the airplane and systems (including emergency equipment that is not part of auxiliary equipment) which are essential for flight. Section II covers operational instruction sequence from approach to airplane to leaving airplane after flight. Section III covers concise procedures for any emergency (except those connected with auxiliary equipment) that could reasonably be expected to be encountered. Section IV covers the description of, and procedures for operating both under normal and emergency conditions, all equipment not essential for flying the airplane. A portion of this section is contained in the confidential supplement. Section V covers all operating limitations and restrictions that must be observed during flight. This section is contained in the confidential supplement. Section VI contains general flight characteristics as based on flight tests, which are peculiar to the airplane and which may be either advantageous or dangerous. This section is contained in the confidential supplement. Section VII covers additional information on special characteristics and factors pertaining to airplane systems under various conditions. Section VIII contains specific operational instructions and duties for the crew member. Section IX includes procedures and techniques for proper operation under instrument flight and extreme weather conditions. Appendix I contains performance data for preflight and in-flight planning, and is presented in the confidential supplement.

YOUR RESPONSIBILITY These Flight Manuals are constantly maintained current through an extremely active revision program. Frequent conferences with operating personnel and constant review of UR's, accident reports, flight test reports, etc., assure inclusion of the latest data in these manuals. In this regard, it is essential that you do your part. If you find anything you don't like about the book, let us know right away. We cannot correct an error whose existence is unknown to us.

HOW TO GET COPIES If you want to be sure of getting your manuals on time, order them before you need them. Early ordering will assure that enough copies are printed to cover your requirements. Order your publications using form NAVWEPS-140, refer to Flight Manual "A" page to obtain address of nearest supply point. Make sure to establish some system that will rapidly get the manuals, interim revisions, and pocket check list to the flight crews once they are received in your squadron.

FLIGHT MANUAL INTERIM REVISIONS Interim revisions are issued as a rapid means of promulgating operating limitations, restrictions and other vital operating instructions for specific model aircraft pending incorporation of such instructions into the Flight Manual by Regular Revision. Interim Revisions are identified as to which Flight Manual they are written against by using the same number as the Flight Manual along with the model of aircraft affected. Interim Revisions written against a Flight Manual will start with number one, and will be assigned consecutively in order of issue throughout the life of the Flight Manual. It is important that you remain aware of the status of all Interim Revisions. The interim revision summary in the front of the Flight Manual list the Interim Revisions that have been cancelled or have been incorporated in the Flight Manual.

POCKET CHECK LIST A pocket check list has been published to provide, in abbreviated form, essential information for flight crews engaged in the operation of the aircraft. The pocket check list (NAVWEPS 01-245FDA-1B) is not distributed automatically with the Flight Manual, and therefore, should be ordered as a separate publication. The check list is printed on cardboard stock with tab pages, and is designed to be fastened to the pilots knee pad.

WARNINGS, CAUTIONS, AND NOTES For your information, the following definitions apply to the "Warnings", "Cautions", and "Notes" found throughout the manual:

WARNING

Operating procedures, practices, etc., which will result in personnel injury or loss of life if not carefully followed.

CAUTION

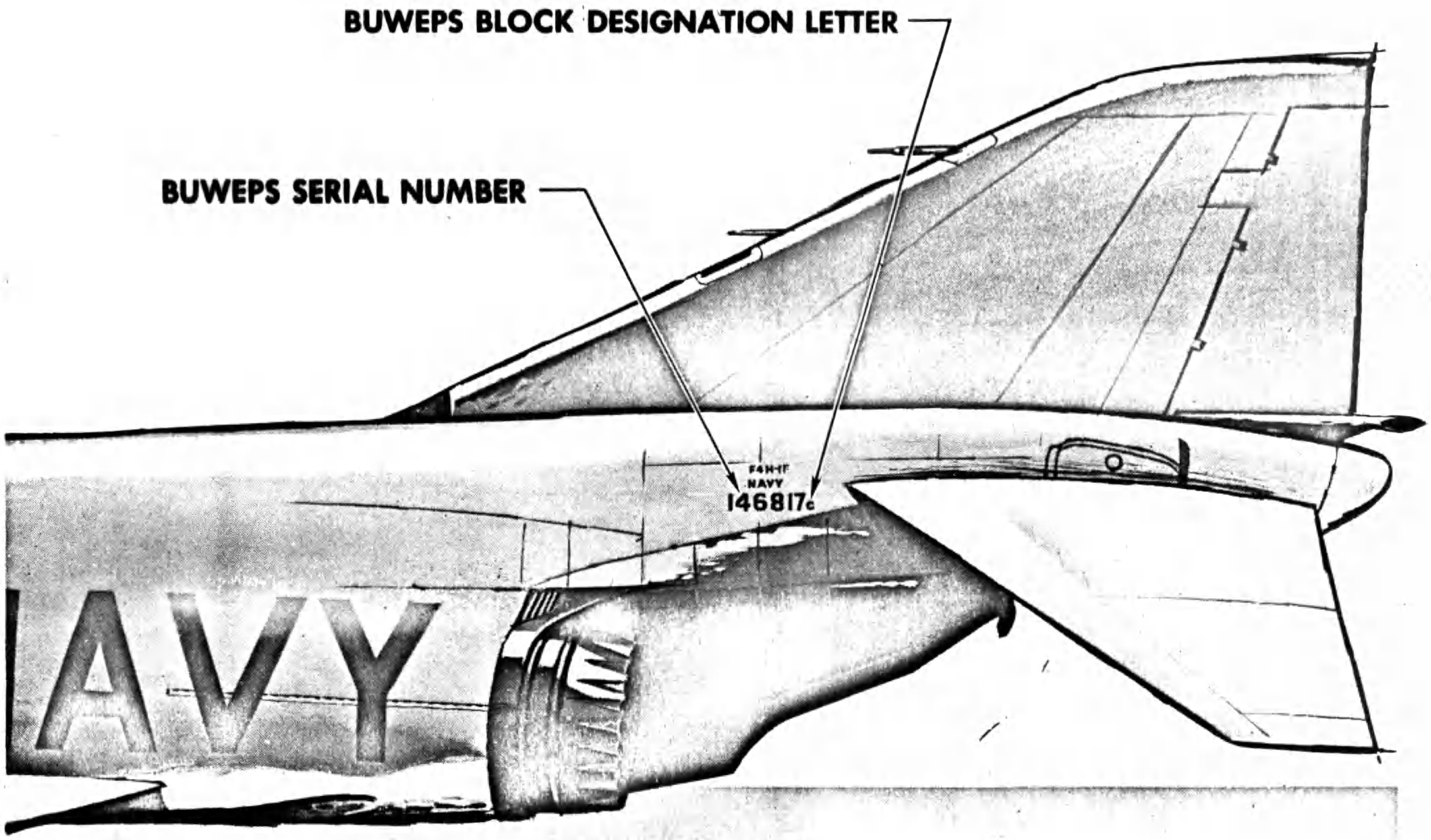
Operating procedures, practices, etc., which if not strictly observed will result in damage to equipment.

Note

An operating procedure, condition, etc., which it is essential to emphasize.

REVISION SYMBOL Revised text is indicated by a black vertical line in the outer margin of the page. The revision symbol indicates changes made in the current revision.

BLOCK NUMBERS



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BLOCK 2 b (11)

145307b thru 145317b

BLOCK 3 c (5)

146817c thru 146821c

BLOCK 4 d (10)

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BLOCK 5 e (14)

148262e thru 148275e

Note

FLIGHT HANDBOOK COVERAGE OF AIRPLANES 145307b THRU 148275e ONLY.

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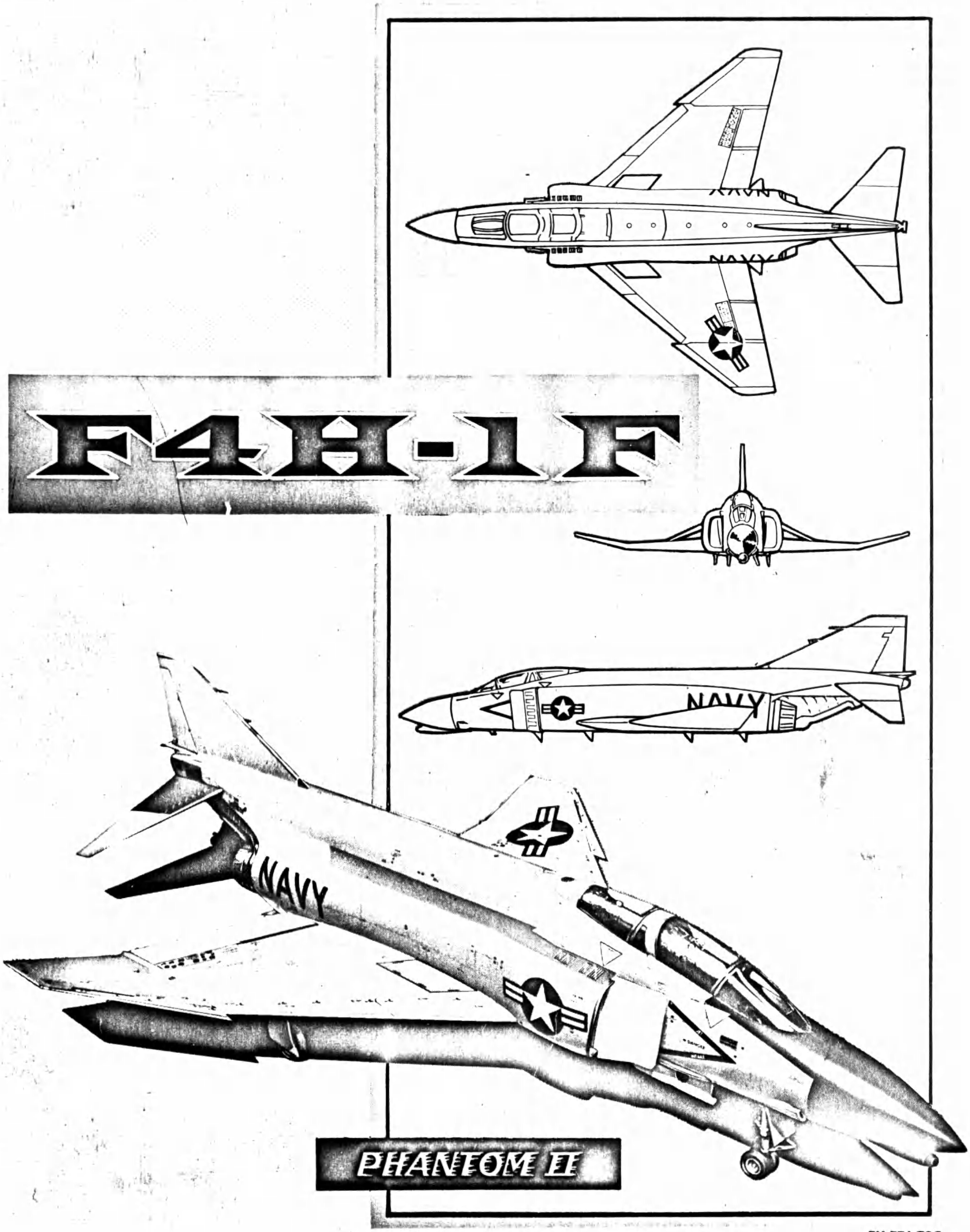

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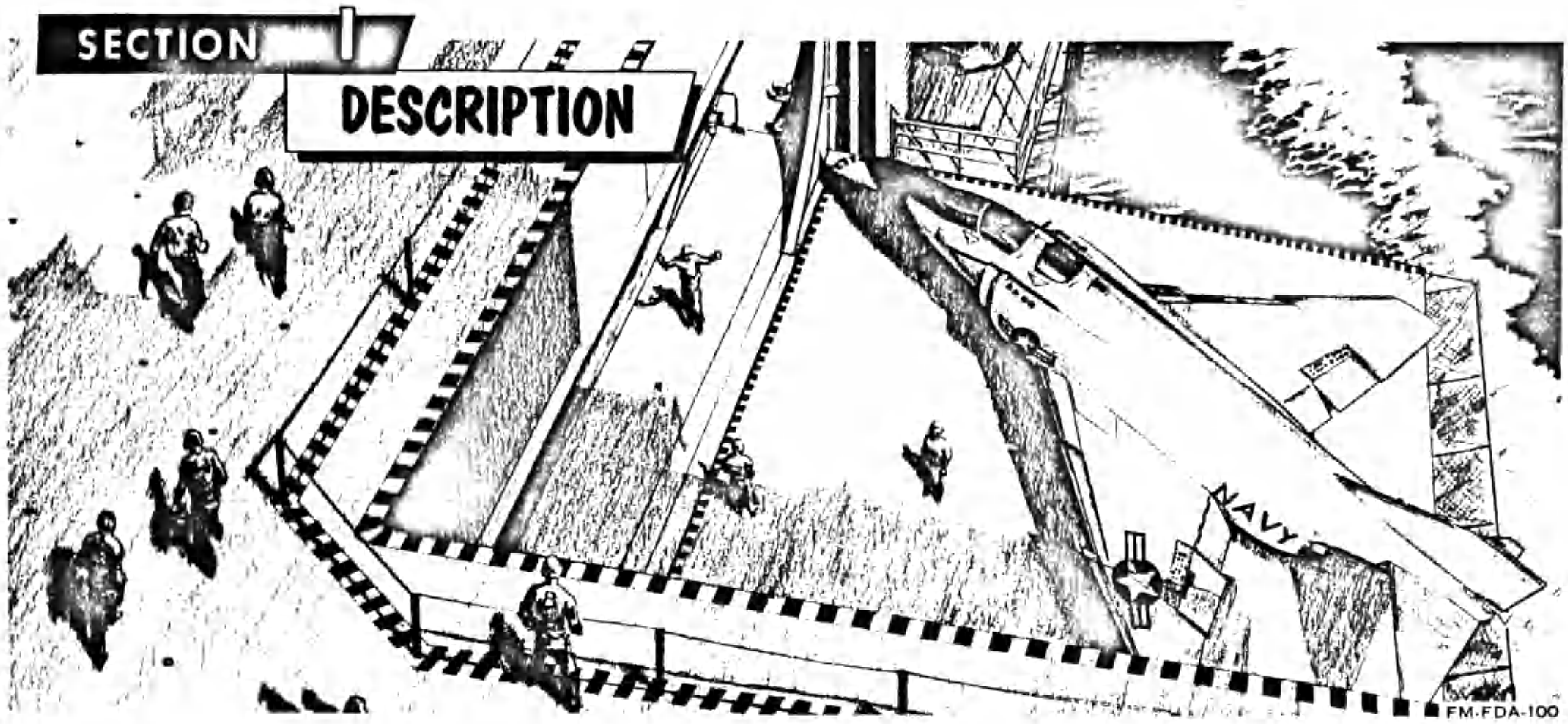


AIRPLANE

The F4H-1F airplane is a supersonic, two place, twin engine, jet-propelled all-weather fighter built by the McDonnell Aircraft Corporation. The airplane is designed for intermediate and long range high altitude interceptions using missiles as the principal armament, and for long range attack missions to deliver special weapons. The radar observer has control over the airborne missile control system, counter-electronic countermeasures, and communications and navigation. A retractable in-flight refueling probe and added electronic equipment such as an infrared detection unit, central air data computer and ground position indicator increases the overall effectiveness of the F4H-1F weapons system. The airplane is powered by two axial flow General Electric J79-GE-2 engines. This power plant is a high pressure ratio, single rotor, turbojet engine with a seventeen stage, variable stator compressor and an afterburner with a variable area, aerodynamic, convergent-divergent exhaust nozzle. The airplane features a low mounted swept-back wing with dihedral at the wing tips, and a one piece horizontal stabilator with cathedral (negative dihedral) mounted low on the aft fuselage. The wings have hydraulically operated leading and trailing edge flaps, ailerons, spoilers and speed brakes. All the control surfaces are positioned by irreversible hydraulic actuators to provide desired control effectiveness throughout the entire speed range.

AIRPLANE DIMENSIONS

The overall dimensions of the airplane are as follows:

Wing Span (Wings Spread)	38 feet 5 inches
(Wings Folded)	27 feet 6 1/2 inches
Length	56 feet 0 inches
Length (airplanes 145309b, 145313b, and 146817c thru 148275e)	58 feet 3 inches
Height (To Top of Fin)	16 feet 3 inches

ARMAMENT

The armament is equipped for carrying and launching four Sparrow III guided missiles. The missiles are carried on four independent launchers which are located on the bottom of the fuselage, two launchers located in the forward fuselage and two launchers located in the center fuselage. In airplanes 145313b, 146817c thru 148275e, provisions are also made to carry missiles on wing stations. The missiles are automatically controlled and fired by the Aero 1A Aircraft Missile Control System.

ENGINE

The airplane is powered by two General Electric J79-GE-2 engines, a lightweight (approximately 4000 pounds), high thrust, axial flow turbojet engine equipped with an afterburner for thrust augmentation. At Military power the engine develops 10,350 pounds thrust and with complete afterburner, total thrust is 16,150 pounds. The engine features a variable stator (first six stages), a 17-stage compressor, 10 annular through-flow combustion chambers, a three-stage turbine, a variable area exhaust nozzle and modulated afterburner thrust augmentation. An impingement type starter, supplied with air from an external auxiliary power unit, is used to crank the engine during starting. During operation, air enters the inlet of the engine and is directed into the compressor rotor by the variable inlet guide vanes located in the compressor front frame. As it is compressed, the air is forced back through the compressor rear frame into the combustion chamber liners. Fuel nozzles, projecting into the combustion chamber liners, eject a fuel spray which mixes with the compressed air. Ignition is provided by a spark plug located in the No. 4 combustion cham-

GENERAL ARRANGEMENT

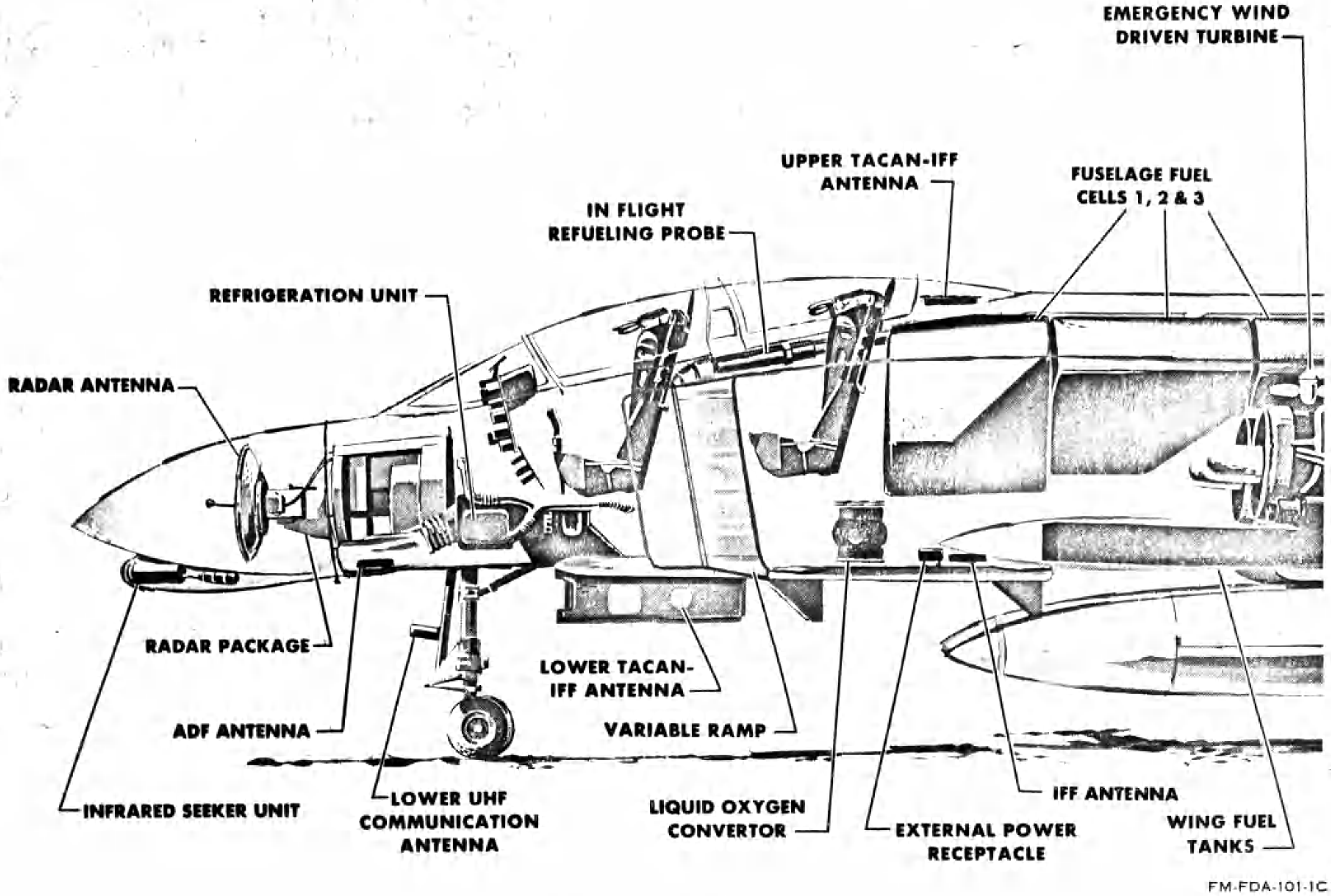


Figure 1-1 (Sheet 1)

ber, the remaining nine combustion cans are ignited through cross fire tubes. The gases resulting from combustion flow from the combustion chambers into the turbine. The three turbine wheels move as a unit on a common shaft, which is directly splined to the compressor rotor. After passing through the turbine section, the exhaust gases flow into the afterburner where their flow is stabilized and then ejected through the variable exhaust nozzle. Additional fuel may be injected into hot exhaust gases for afterburner combustion, producing considerable thrust augmentation. The engine oil system is a dry sump type completely contained on the engine. The oil system is used for lubricating purposes and to hydraulically actuate the primary and secondary exhaust nozzles. The variable stator vanes and the inlet guide vanes are interconnected externally through a system of half-rings, drag links, and bell cranks, and are positioned by two actuators which use high pressure engine fuel as a hydraulic fluid. The variable stator system positions the inlet guide vanes and the first six stages of the variable stator vanes to control airflow through the engine compressor. Control of compressor airflow greatly reduces problems of compressor stall by improving airflow characteristics at critical engine-airplane speeds, especially during burst acceleration and deceleration. An increase in rpm or decrease in compressor inlet temperature results in a signal from the main fuel control to open the stator blades to allow a greater airflow. A decrease in rpm or increase in compressor inlet temperature positions the stator blades to reduce inlet airflow. The variable nozzle system is hydromechanically controlled and schedules nozzle area by positioning the nozzle opening to obtain optimum thrust in regard to altitude and airspeed conditions, keeping within the safe design limits of the engine. The main function of the variable exhaust nozzles is to control the operating temperature of the engine as governed by the engine amplifier during Military and Maximum engine operation. Air bled from the 17th stage of the compressor is passed through a heat exchanger and is used for auxiliary equipment cooling, cockpit pressurization, air conditioning, and also supplies the air for the aircraft pneumatic system and the central air data computer unit.

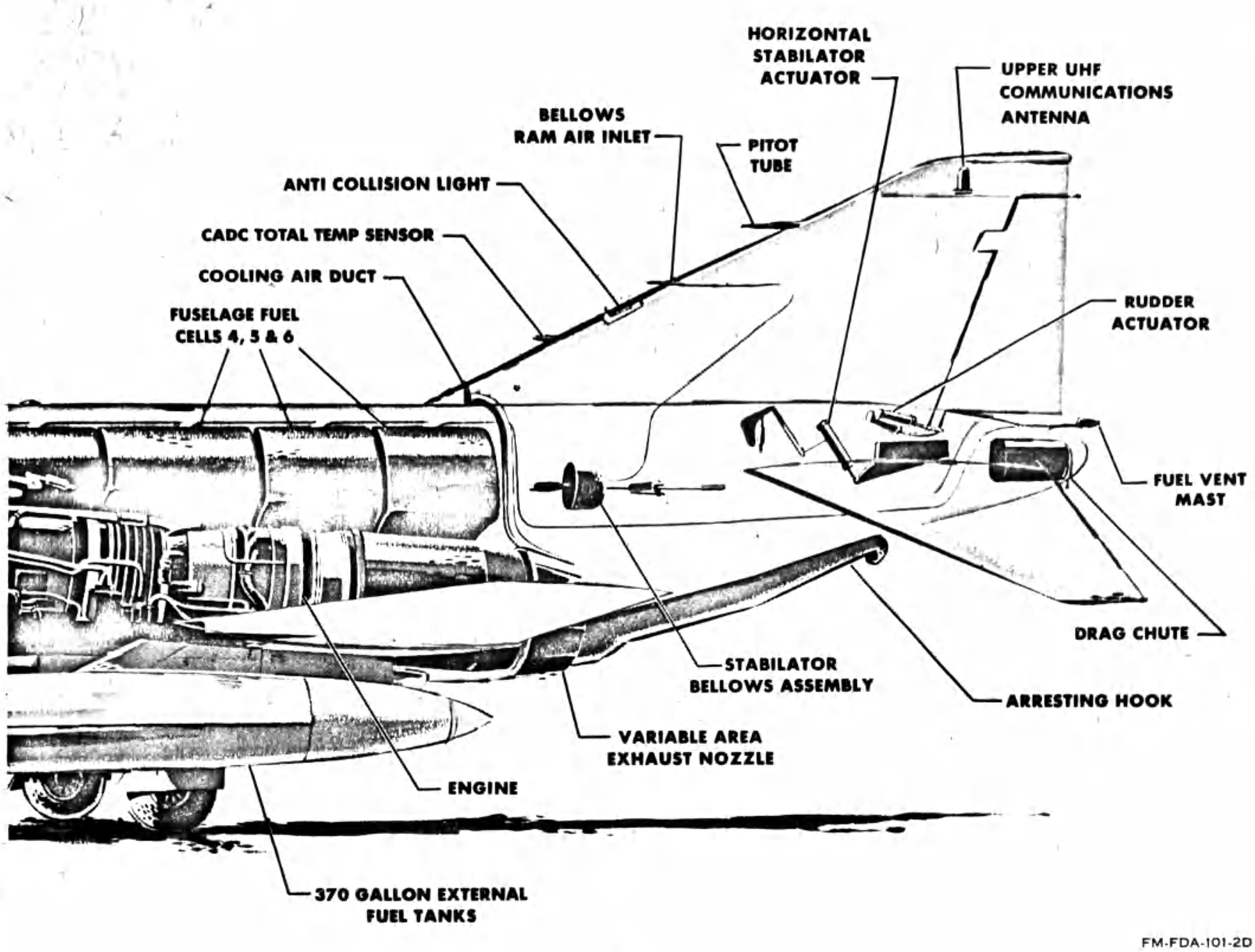


Figure 1-1 (Sheet 2)

ENGINE FUEL CONTROL SYSTEM

The fuel control system (figure 1-2) for each engine is complete in itself and the two systems are identical. For simplicity of discussion only one system or engine shall be discussed. The engine fuel control system transports fuel from the engine fuel inlet to the combustion chambers. This fuel is discharged in the proper state of atomization for complete burning. Varying engine power settings and conditions demand changes in fuel flow; therefore the engine fuel control system must also control fuel flow to obtain maximum engine efficiency within the design limits of the engine. Only the engine fuel system is discussed in the following paragraphs. The afterburner fuel system is discussed separately in this section.

Fuel Pump Unit

One engine-driven fuel pump of the parallel type is used on each engine. The pump consists of two gear-type, equal capacity, pumping elements mounted in an integral housing and driven by a common shaft. A centrifugal pumping element ahead of and in series with the two positive displacement gear pumping elements is incorporated in the pump and acts as a boost element at the fuel inlet, thus permitting operation with low inlet pressure. The pump shaft contains a shear section for each of the gear elements. Failure of either shear section will not cause a failure or a reduction in pump output of the remaining section. With the failure of one gear element of the fuel pump the engine will continue to operate normally except in low altitude high speed operation where a reduction in the power setting may be necessary. A relief valve is provided and is set at 950 psi above engine boost pressure. Fuel pump flow depends only on engine rpm, and the pump will deliver 48-56 gpm at 950 psi at maximum engine rpm.

Engine Fuel Controller

The main fuel control is a hydromechanical computer which uses engine fuel as the hydraulic controlling medium. The control performs the following functions: Provides engine speed control by regulating main fuel flow; provides fuel surge protection; limits turbine

ENGINE FUEL SYSTEM

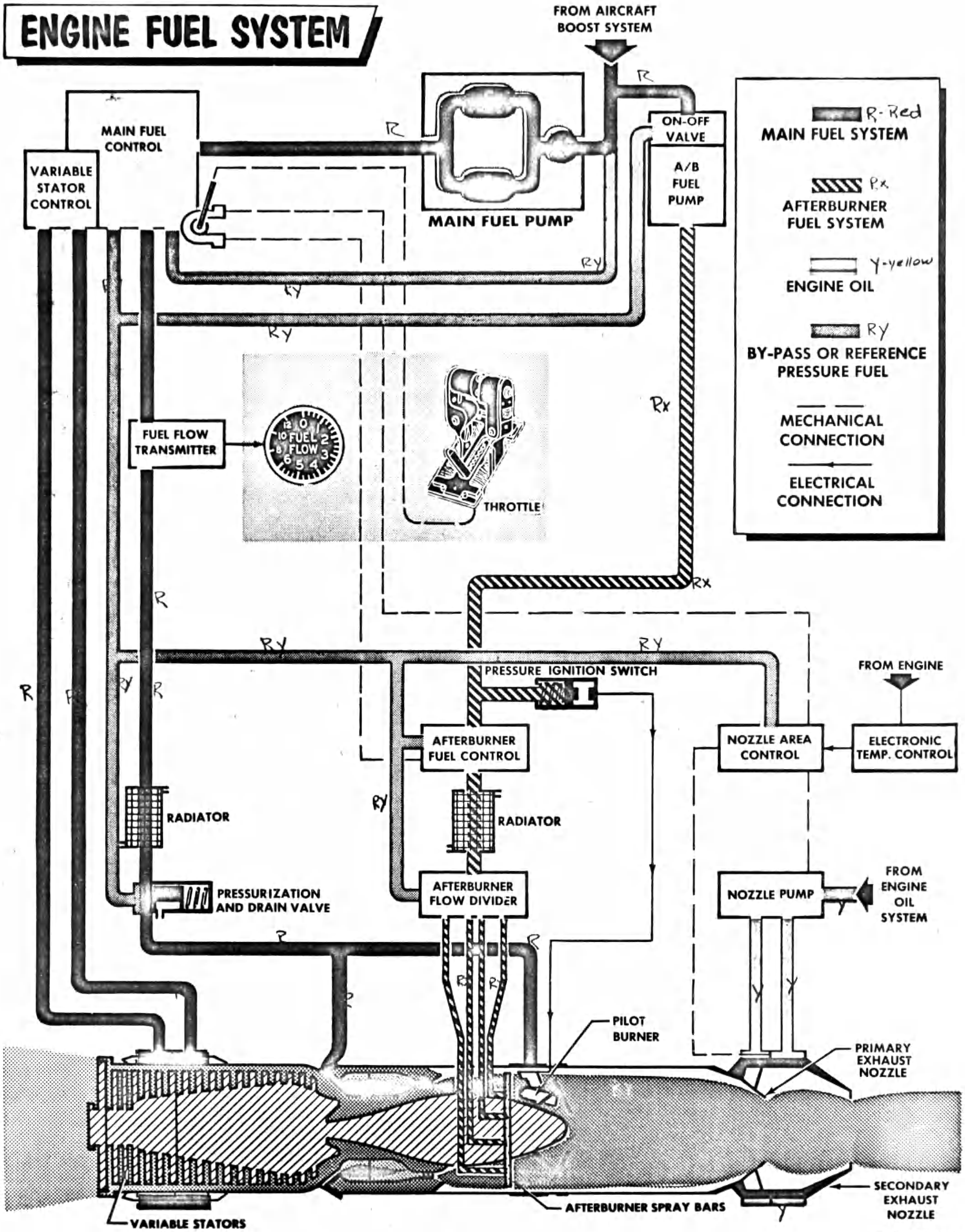


Figure 1-2

inlet temperature to a safe value; provides a positive fuel shutoff; schedules variable stator vane angle to control airflow into the engine and provides a hydraulic signal to initiate afterburning operation. High pressure fuel from the pump is delivered to the bypass valve and the metering valve in the main fuel control. The bypass valve senses the pressure differential across the metering valve, and maintains this pressure differential at a predetermined value by bypassing varying amounts of fuel back to the main fuel pump inlet. The metering valve is positioned by various operating signals and meters fuel to the engine as a result of these integrated signals. From the metering valve fuel flows through the main fuel control cutoff valve. The cutoff valve cuts off fuel flow at engine shutdown. Fuel passing through the open cutoff valve flows through a fuel-oil cooler which effects a transference of heat from the scavenge oil to the fuel. From the fuel-oil cooler, the fuel flows into the pressurizing and drain valve. This valve maintains the fuel pressure at the minimum acceptable for all operations and also drains the fuel manifold at engine shutdown. Fuel flows from the pressurizing and drain valve into the fuel manifold. High pressure metered fuel is fed from the manifold into the 10 flow-divider type fuel nozzles. The fuel nozzles discharge an atomized fuel spray into the combustion chambers. Here the fuel is mixed with air and burned to produce engine thrust.

Fuel Oil Radiator

Fuel metered from the main fuel control passes through the cooler tubes of the fuel-oil radiator and then to the fuel nozzles. The fuel serves as the coolant for the scavenge oil which flows around the radiator tubes. The radiator incorporates a bypass valve to regulate the flow and temperature of the oil and fuel. There are two fuel-oil radiators incorporated on the engine, one utilizes normal engine fuel flow as a coolant whereas the other radiator uses afterburner fuel flow as a coolant. Both fuel-oil radiators serve the same purpose and their operation is the same.

Fuel Pressurization and Drain Valve

The fuel pressurization and drain valve prevents fuel flow to the engine until sufficient fuel pressure is attained in the main fuel control to operate the servo assemblies which are used to compute the fuel flow schedules. It also drains the fuel manifold at engine shutdown to prevent post shutdown fires, but keeps the upstream portion of the system primed to permit faster starts. The pressurization and drain valve is pressure operated. When the fuel pressure differential across the pressurizing valve drops below 80 psi, the pressurizing valve closes cutting off fuel flow to the engine and the drain valve opens to drain the fuel manifold.

Fuel Nozzles

A flow-divider type fuel nozzle in each inner combustion chamber liner delivers metered fuel in the proper state of atomization for maximum burning into the compressor discharge air entering the combustion chamber. The nozzles produce a uniformly distributed, cone shaped, hollow fuel spray upon application of pressure at the nozzle inlets. High velocity compressor air is directed around the nozzle by an air shroud to provide a cooling action around the nozzle orifice and to reduce carbon deposits.

THROTTLES

A throttle (figure 1-3) for each engine is located on the left console of the forward cockpit. Mechanical linkage and teleflex cables transmit mechanical motion from the throttle to those accessories requiring coordination to obtain the degree of thrust desired. Movement of the throttle is transmitted by mechanical linkage to the main fuel control. The main fuel control unit incorporates a throttle booster which reduces the amount of effort needed to move the throttles. The boost power is supplied by fuel from the engine driven fuel pump. Teleflex cables from the main fuel control link the nozzle area control, afterburner fuel control and throttle angle switch to throttle movement so that fuel flow and nozzle area are compatible throughout the full range of engine operation. A friction adjusting lever is mounted between the throttles which permits adjustment of throttle friction to suit individual requirements. The throttle mechanism is a "gear shift" type. Included on the throttles are the ignition buttons (one for each engine on the applicable throttle), speed brake switch and microphone button on the right throttle, and master lights control switch on the outboard side of the left throttle. Limit switches which control the main fuel shutoff valves are built into the throttle quadrant. Advancing the throttle from OFF to IDLE (with the engine master switch ON) will actuate electrical switches which will open the main fuel shutoff valve corresponding with the throttle moved. With further advancement of the throttle from IDLE to MIL, engine thrust will increase proportionally. At the MIL position of the throttles, the engine should be delivering its rated Military power. Afterburner light-off can be initiated anywhere within the afterburner modulation range by shifting the throttles outboard and moving forward toward the MAX position. Movement of the throttles from IDLE to OFF will actuate a switch which will close the main fuel shutoff stopping fuel flow to the engine. Throttle movement through the cutouts is as follows: To move throttles from OFF to IDLE, push forward and then shift throttles inboard. To move from MIL to MAX shift throttles outboard, throttles can now be moved forward in the afterburner range.

ENGINE MASTER SWITCHES

The toggle lock type two-position engine master switches (figure 1-4) are located on the left console in the pilot's cockpit on the inboard engine control panel. Placing the switch in the ON position will direct power to the fuel booster pumps and fuel transfer pumps. The circuits for the fuel shutoff valves, which are normally operated by the throttles, are such that either valve will be closed when its respective engine master switch is placed OFF regardless of the throttle position.

THROTTLES

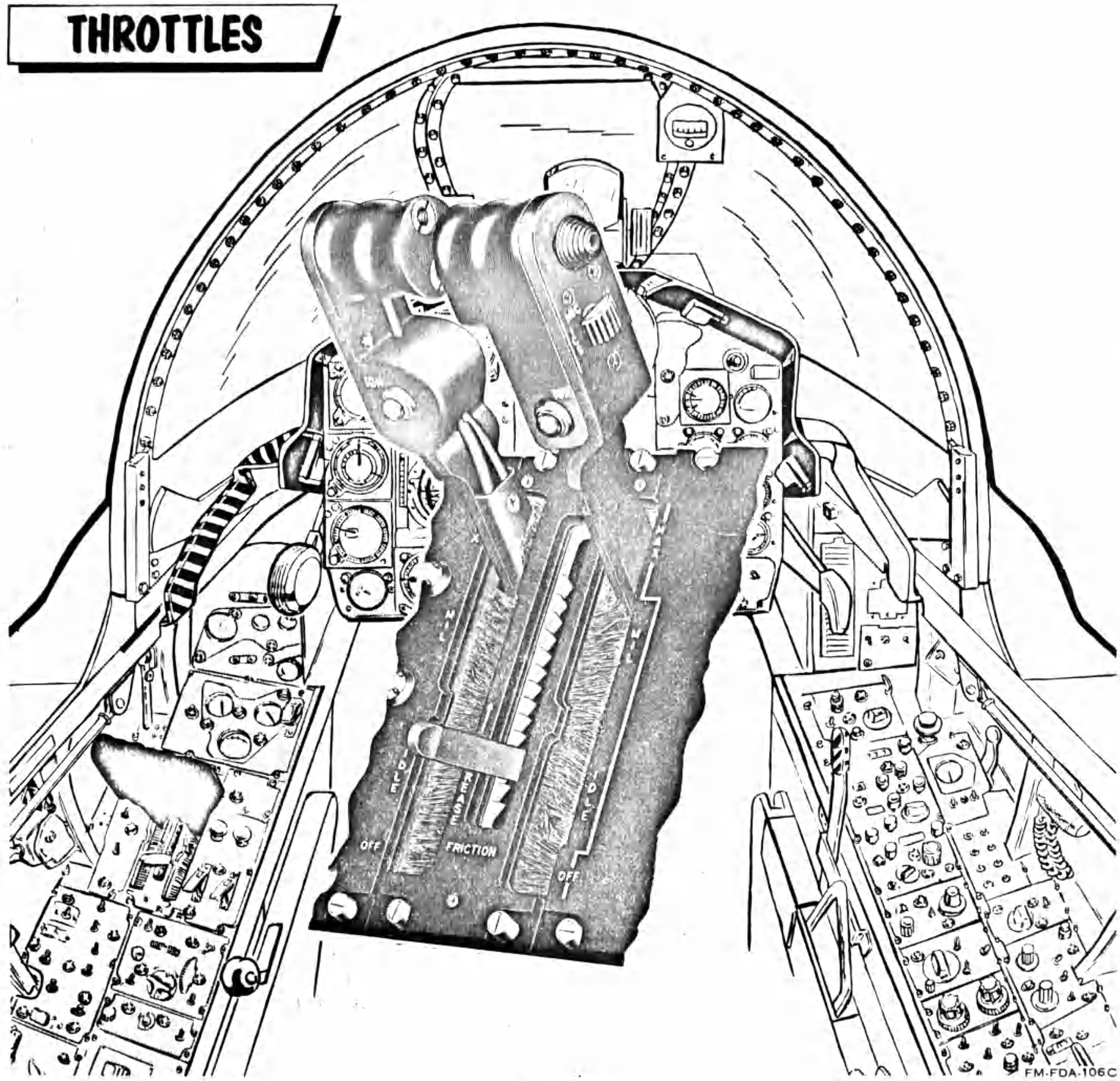


Figure 1-3

Catapult Hold-back Handles

Handles secured to the pilot's cockpit structure and located above the MIL throttle detent and MAX throttle detent may be hinged upward to line up with the throttle grips at the MIL and MAX throttle positions. The grips and handle may then be held together during catapulting to prevent inadvertently throttling back. The handles are automatically stowed when released.

ENGINE FUEL FLOW INDICATORS

The engine fuel flow indicators (figure 1-17) are mounted on the right side of the pilot's instrument panel. The fuel flow indicating system indicates the amount of main fuel system flow, in pounds per hour, of fuel the engines are using at a particular power setting. The rate of fuel flow is shown in 1000 pounds per hour by a pointer moving over a scale calibrated from 0 to 12. The flow is measured by transmitters mounted on the engines which receive power from the essential 28 volt a-c bus. Afterburner fuel flow bypasses the fuel flow transmitters and therefore, A/B fuel flow is not shown on the indicators.

TACHOMETERS

The electric tachometer system is composed of two tachometer indicators (figure 1-17) mounted on the pilot's instrument panel and one engine-driven tacho-

ENGINE CONTROL PANELS

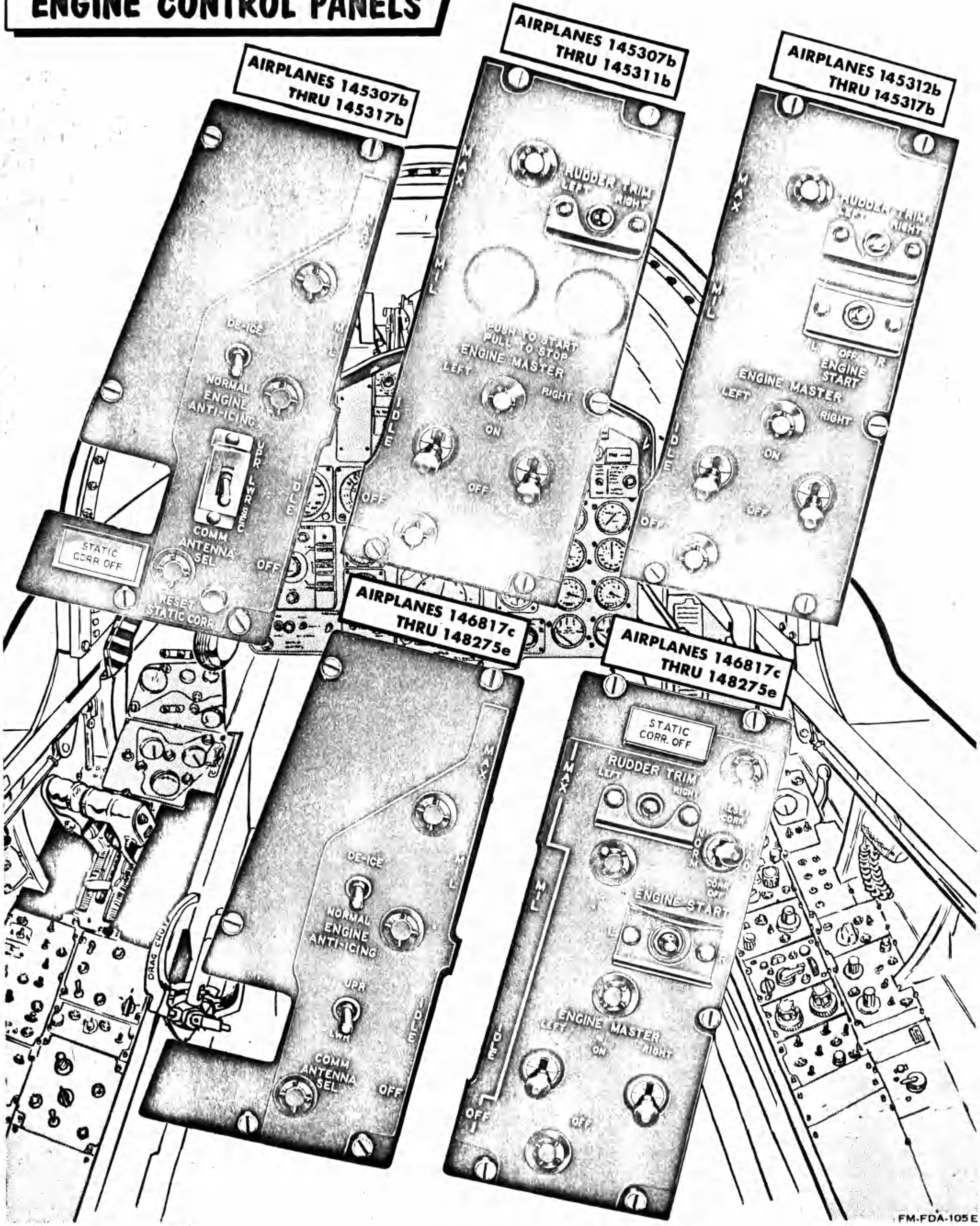


Figure 1-4

meter generator mounted on each engine. The system is completely self-contained in that it requires no external source of power. The tachometer generator develops a poly-phase alternating current which is used to indicate percentage of maximum engine rpm. The indicator dials are calibrated from 0 to 100; beyond 100% the tachometer reading is interpolated. Each indicator includes two pointers, a large one operating on the 0 to 100 scale and a small one operating on a separate scale calibrated from 0 to 10.

EXHAUST GAS TEMPERATURE INDICATORS

The exhaust gas temperature indicators (figure 1-17) are mounted on the pilot's instrument panel. The scale range on the indicators is 0 to 10 with the reading multiplied by 100 degrees centigrade. The system indicates the temperature of the exhaust gas as it leaves the turbine unit during engine operation. Twelve dual loop thermocouples are installed on each engine extending from the aft edge of the aft stator case, aft to the turbine frame, and then circumferentially around the engine for approximately 160 degrees. The dual loop thermocouples are connected in parallel and the millivoltages produced by one of the sets of dual loop thermocouples is directed to an amplifier for temperature limiting. The millivoltages produced by the other set of thermocouples is directed to the cockpit indicator. The indicator is a null-seeking potentiometer type. It balances a thermocouple voltage against a constant voltage source with a small servo simultaneously balancing a bridge circuit and operating the indicator pointers. Power to operate the exhaust temperature gages comes from the essential 115 volt a-c bus.

OIL PRESSURE INDICATORS

The oil pressure indicators (figure 1-17) are located on the pedestal panel. The scale range on the indicators is 0 to 10 with reading multiplied by 10. The oil pressure indication system senses oil pressure downstream of the main lube pump in the main lube discharge line. The oil pressure transmitters are powered by the essential 26 volt a-c bus.

VARIABLE AREA EXHAUST NOZZLE

Two sets of cylindrical nozzles, operating together, make up the variable area exhaust nozzle system. The primary nozzle, hinged to the aft end of the tail pipe, controls the convergent portion of the nozzle, while the secondary nozzle, hinged to a support ring, controls the divergent portion of the nozzle. The two sets of nozzles are linked together and maintain a scheduled area and spacing ratio which is infinitely variable between full open and full closed. The nozzles are regulated by the nozzle area control. Movement of the nozzles is accomplished automatically by four synchronized hydraulic actuators. The exhaust gases leave the primary flaps at sonic velocity and are accelerated to supersonic velocity by the controlled expansion of the gases. Control of this expansion is provided by the cushioning effect of the secondary airflow through the annular passage between the two sets of nozzles.

Exhaust Nozzle Control Unit

Throttle position, nozzle position feedback, and exhaust gas temperature are the parameters utilized to schedule the correct nozzle area. During engine operation in the sub-military region, nozzle area is primarily a function of throttle angle and nozzle position feedback. The nozzle is scheduled full open at "idle" and the area is decreased as the throttle is advanced toward the military position. However, during a rapid throttle burst from below 88% rpm to 91% rpm, the main fuel control generates a hydraulic "off-speed" signal which is delivered to the nozzle area control. This signal prevents the primary nozzle from closing down past a preset position, permitting a rapid increase in engine rpm.

Exhaust Nozzle Position Indicators

Exhaust nozzle position indicators (figure 1-17) which show the exit area of the exhaust nozzle, are located on the pilot's instrument panel. The instruments are placarded jet nozzle position and are calibrated from CLOSE to OPEN in 1/4 increments. The nozzle position indicators enable the pilot to make a comparison of nozzle position between engines, and is also used to establish a relationship between nozzle position and exhaust gas temperature and nozzle position and throttle settings. Power for the instrument is derived from the essential 28 volt d-c bus.

Note

Spasmodic exhaust nozzle operation shall be logged on the yellow sheet (Op NAV Form 3760 -2).

ENGINE AIR INDUCTION SYSTEM

There are two independent and identical air induction systems, one for each engine. The component units are the fixed and variable ramps, which make up the primary air system, and a variable bypass bellmouth, and an auxiliary air door which make up the secondary air system.

Variable Duct Ramp System

The variable duct ramp system is utilized to provide primary air at optimum subsonic airflow to the compressor face throughout an extremely wide range of aircraft speeds. The ramp assembly consists of a fixed forward ramp and two variable ramps. The forward variable ramp is perforated to allow boundary layer air to be bled off and exhausted overboard (figure 1-5). The aft variable ramp is solid and has a stabilizing wedge attached to the trailing edge. This wedge causes a slight turbulence in the boundary layer air causing it to mix with the free stream air in the inlet duct and also assists in eliminating "duct buzz". The Central Air Data Computer supplies a temperature corrected Mach signal to a utility hydraulic system servo unit which positions the ramps for optimum airflow at high Mach numbers. The inlet ducts have a fixed forward ramp angle of 5° and a variable ramp angle of 4° to 10° relative to the fixed forward ramp.

ENGINE AIR DUCTS

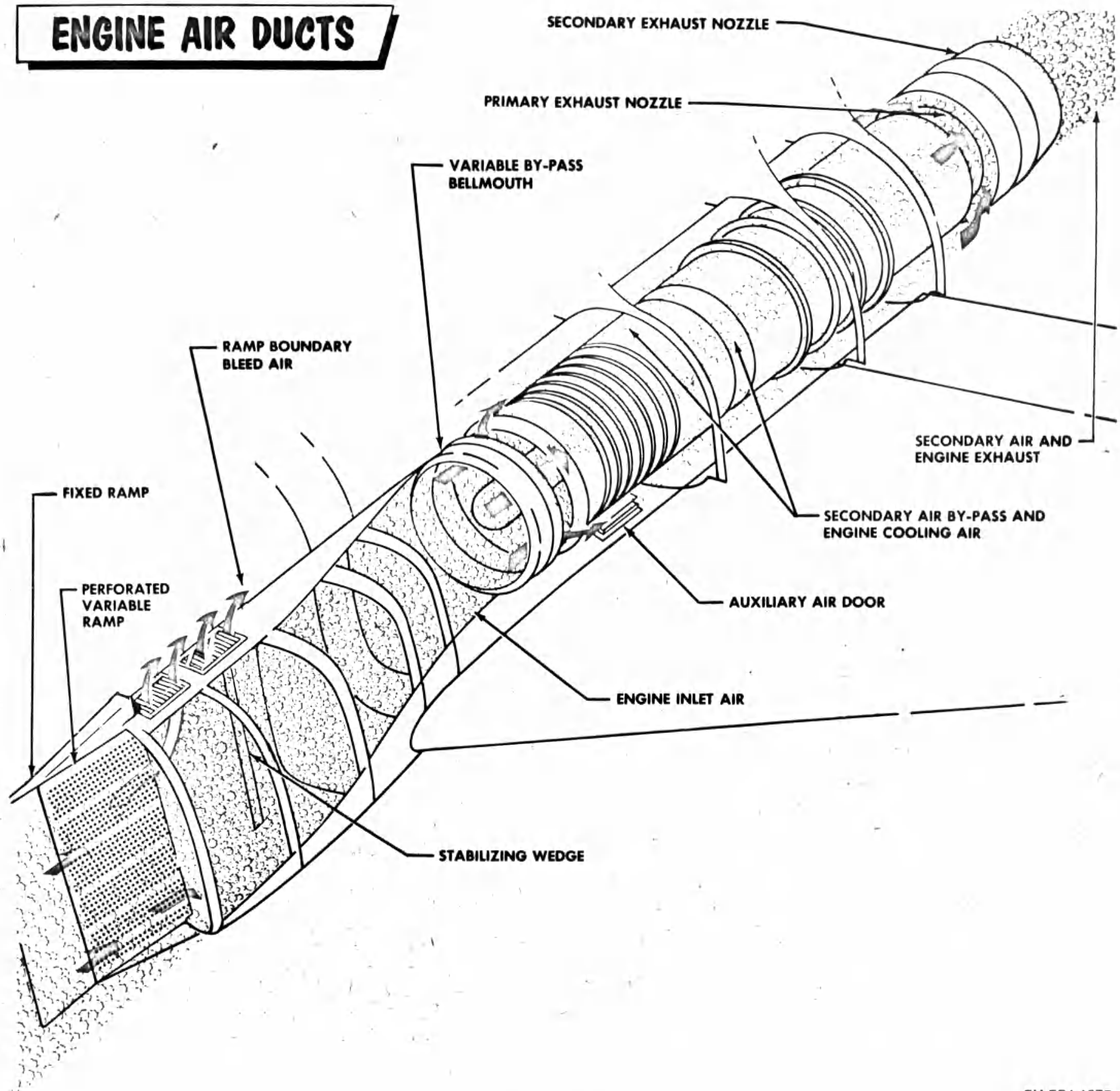


Figure 1-5

Variable Bypass Bellmouth

The variable bypass bellmouth is an automatic unit which allows excess induction air from the compressor face to flow into the engine compartment (figure 1-5). Air diverted in this fashion is referred to as the Secondary Air System. The variable bellmouth is a perforated ring located between the airplane duct structure and the engine compressor inlet. When the bellmouth is closed, a limited amount of bypass air flows into the engine compartment through the perforations in the bypass bellmouth and the engine air-oil cooler bleed. The bypass bellmouth controller senses the optimum airflow for induction into the engine. When this airflow is exceeded, the controller signals a utility system hydraulic actuator which opens the bypass until the optimum airflow to the engine is established.

Auxiliary Air Door

The auxiliary air doors are opened to relieve excessive engine compartment pressures (13.3 psig or above) or to provide additional airflow for engine cooling during low speed or ground engine operation. The doors are opened by a utility system hydraulic actuator when the landing gear handle is placed in the DOWN position. In flight (landing gear handle UP), the actuating cylinder is held closed by utility hydraulic pressure, however, engine compartment pressure of 13.3 psig, or over will actuate the door open until the overpressure is relieved.

Auxiliary Air Door Warning Lights

The auxiliary air door warning lights are intended to notify the pilot when the auxiliary air doors are not in the position called for by the gear handle. The lights will illuminate if the landing gear handle is up and the doors are open, or if the landing gear handle is down and the doors are closed. When the warning light(s) illuminates the following precautions should be observed:

1. If gear handle is up:
   a. No afterburner operation.
   b. Decrease airspeed to below 400 knots CAS.
   c. Avoid extended periods at high power settings.
2. If gear handle is down:
   a. Land or retract gear.
      Extended periods of low speed, high power setting flight with doors closed may cause overheating in the engine compartment.

STARTING SYSTEM—AIR TURBINE

(Airplanes 145307b thru 145311b)

Pneumatic starters, which require an external source of compressed air, are installed on each engine. The receptacles for connecting the air supply lines (one receptacle for each engine) are located on the bottom of the fuselage inboard of each main gear well. Starting air to the engine starter is controlled by an electrically operated valve that permits air from a ground turbine compressor to actuate the engine starter. The starter is disengaged automatically at 3400 rpm (45.5% rpm). The air turbine starters require a 3.3:1 pressure ratio gas turbine compressor starting unit.

STARTING SYSTEM–IMPINGEMENT

(Airplanes 145312b thru 148275e)

The impingement starting system consists of an assembly of ducting and valves which are airframe mounted and a manifold assembly which is mounted on the turbine frame of the engine. The single receptacle for connecting the air supply line is located on the bottom left side of the fuselage aft of the main gear wheel well. Air from the external source is directed to the left or right selector valve which will distribute the air to either the left or right engine, depending on cockpit selection. The engine manifold assembly distributes the starting air to seven (7) impingement nozzles, which direct the air against the second stage turbine blades of the turbine wheel.

Engine Start Switch (Buttons)

The starter buttons (figure 1-4) are located on the left console in the pilot's cockpit just inboard of the throttles. The buttons receive power from the 28 volt d-c starter cart bus. With APU air connected to the left engine and electrical a-c lines connected to the airplane, pressing the left engine starter button energizes the left starter air valve open. Air flows to the left starter turbine and engine cranking is initiated. The starter button once depressed may be released since it is held "on" by a holding solenoid. At approximately 45.4% rpm during a normal start, the engine should be turning at a self-sustaining speed so that the starter button pops up and closes the starter air valve, stopping air to the starter. Before starting the right engine, the APU air must be connected to the right engine; there is no common connection to start both engines. Effective airplanes 145312b thru 148275e, a starter switch is used in lieu of the starter buttons. The starter switch is a three-position switch with switch positions of L, OFF, and R. Selecting the L position energizes the left selector valve open and permits air to flow to the left engine impingement nozzles. Selecting the R position energizes the right selector valve open and permits air to flow to the right engine impingement nozzles. The OFF position closes both selector valves stopping airflow to the engine. For impingement starting, a 5.5:1 pressure ratio gas turbine compressor starting unit is required.

IGNITION SYSTEM

The ignition system consists of an ignition button (figure 1-4) on each throttle, a low voltage, high energy ignition unit on the engine, a spark plug in No. 4 combustion chamber and the necessary wiring. The main ignition system produces sparks which ignite the atomized fuel-air mixture in the No. 4 combustion can. The remaining nine combustion cans are ignited through the cross fire tubes.

Ignition Buttons

The ignition buttons (figure 1-3) are spring-loaded push button type switches located on each throttle directly below the throttle grip. The switches control the 115 volt a-c ignition system, which receives its power from the essential 115 volt a-c bus. Depressing the ignition button causes the spark plug to discharge, igniting the fuel-air mixture as the throttle is moved from OFF to IDLE during engine start. The spark plugs will fire only while the ignition button is depressed. The ignition duty cycle is 2 min. on, 3 min. off, 2 min. on, and 23 min. off. The ignition circuits are completed anytime aircraft power is on, and ignition button is depressed.

AFTERBURNER SYSTEM

The engine is equipped with an afterburner, where additional fuel may be injected into the hot exhaust gases for afterburner combustion, producing considerable thrust augmentation. The main components of the afterburner system (figure 1-2) are the afterburner fuel pump, afterburner fuel control, afterburner fuel manifol and spray bars, and the torch igniter.

AFTERBURNER FUEL SYSTEM

The afterburner fuel system (figure 1-2) provides the fuel for augmentation of the thrust produced by the engine. A separate afterburner constant pressure drop variable fuel control, meters the afterburner fuel, and ignition is by a separate A/B ignition system. In operation, the airframe boost pump supplies fuel to the inlet of the afterburner pump. The pump supplies fuel to the check and vent valve which will open under pressure and supply fuel to the rest of the system. Fuel passing through the check and vent valve continues to the afterburner fuel control. A small amount of fuel is piped to the flow divider/selector valve. Fuel entering the afterburner fuel control is metered by the fuel control and then passes through the afterburner fuel oil cooler to the inlet of the flow divider/selector valve. Here fuel is ported to the various sectors of the spray bars to obtain the best spray pattern for the condition of afterburning required.

Afterburner Fuel Pump

The afterburner fuel pump is an engine-driven centrifugal pump. It operates continuously, but discharges fuel to the afterburner fuel system only when the inlet to the pump is open. To open the inlet to the afterburner fuel pump, the pilot must move the throttle into the afterburner modulation range and engine speed must be sufficiently high (above 92% rpm) to support combustion.

Afterburner Fuel Control

The afterburner fuel control is linked mechanically to the main fuel control through the use of teleflex cabling. Any movement of the throttle moves the main fuel control teleflex and also moves the teleflex to the afterburner fuel control. Fuel entering the afterburner fuel control is metered by the fuel control in response to throttle movement and changes in compressor discharge pressure. The control varies fuel flow between the minimum necessary for afterburner combustion for any flight condition and the maximum fuel flow allowable at the flight condition. The afterburner fuel control is designed to hold a constant pressure drop across an orifice while the area of that orifice is varied in accordance with throttle position and compressor discharge pressure.

Afterburner Fuel Spray Bars

The flow divider and selector valve assembly distributes the fuel to the spray bars which release successive patterns of atomized fuel into the afterburner combustion chamber. The 20 dual spray bars are divided into four sections; primary sector, secondary sector, primary uniform and secondary uniform. When the throttle is first advanced into the afterburner position, the primary sector will light up. Further advancement will cause the secondary sector to light up. When the throttle is advanced still further, a distinct jump in thrust occurs; this is a result of the primary uniform sector lighting off. As the throttle is advanced to the maximum afterburner position, the secondary uniform sector lights off; this is full afterburner operation.

AFTERBURNER IGNITION SYSTEM

Afterburner ignition is controlled by a normally open automatic switch. When the throttle setting calls for afterburner operation afterburner fuel pressure closes the switch activating the system. The afterburner ignition system develops the arc necessary to ignite a fuel-air mixture in the torch igniter. The pressure switch controls the electrical circuits to the 115 volt a-c afterburner ignition system. In operation, fuel from the downstream side of the pressurizing and drain valve passes through a filter then enters the torch igniter orifice. From the torch igniter orifice the fuel passes through a check valve and enters the torch igniter (pilot burner). The orifice provides the right fuel flow for any flight condition, assuring positive torch igniter operation. The pilot burner operates exactly as an inner combustion can in a conventional jet engine; compressor discharge air enters the can and is mixed with atomized fuel from the torch igniter fuel nozzle. The mixture is ignited by the igniter plug and burns with an intense flame. This flame is allowed to escape through the open end of the burner. The main afterburner fuel flow is ignited by the flame emanating from the torch igniter. During afterburner operation, the afterburner igniter plug will fire continuously to assure positive ignition of the torch igniter.

OIL SYSTEM

The dry sump, full scavenging oil system is completely contained on the engine. Its purpose is to supply filtered lubrication oil to cool and prevent wear of the engine bearing, gears, and other rubbing surfaces. It filters and cools lubricating oil and returns the oil to the engine mounted oil tank for re-use. The 6.3 gallon oil tank stores the oil reserve and furnishes oil to the main lube and hydraulic pump and constant speed drive. The main lube pump furnishes oil under positive pressure to the lubrication and hydraulic system. The lube element of the main lube and hydraulic pump delivers oil at 20-85 psi through a filter and then into the main lube line. This line splits into three branches. A portion of the oil in the first branch lubricates the No. 1 bearing and the front gearbox. The remaining oil lubricates the gears in the transfer gearbox and aft gearbox. Oil in the second branch of the main lube line lubricates the No. 2 and No. 3 bearings, while oil in the third branch goes to the orifice block and then to the lube pressure transmitter or the lube relief valve. A sump to collect the lubricating oil is located at each lubrication point. Three scavenge pumps return oil through the fuel oil cooler to the reservoir during all flight conditions. The hydraulic element of the main lube pump supplies oil to the nozzle pump to hydraulically actuate the variable nozzle. Refer to Engine Hydraulic Oil System, this section for further information.

ENGINE HYDRAULIC OIL SYSTEM

The actuating force necessary to change exhaust nozzle area is developed hydraulically from the engine oil system. Oil is fed from the engine oil tank to the hydraulic element of the main lube pump. A hydraulic oil filter element in series with the flow removes foreign matter and discharges to the nozzle pump. A check valve in the filter eliminates any possibility of reverse flow through the system. A pressure relief valve located at the inlet to the nozzle pump branches off to the scavenge line. It allows a bleed of oil to the scavenge oil return line when oil pressure at the nozzle pump inlet exceeds 150 psi. The nozzle area control output signal directs the operation of the nozzle pump. Discharge pressure from the pump is supplied to either the rod end or head end of the four nozzle actuators, opening or closing the variable nozzle.

AIRPLANE FUEL SYSTEM

The fuel system (figure 1-6) consists of six interconnected fuel cells in the fuselage, and two integral "wet wing" cells located in the wing torque boxes. Provisions are made for two externally mounted droppable wing tanks and a droppable fuselage centerline external tank, which is interchangeable with a refueling tanker external store (hereafter referred to as Buddy Tank). Provisions are also made for an in-flight refueling system. The function of fuselage cells 2, 3, 4, 5 and 6 is to keep tank number 1 supplied with fuel. See figure 1-9, Fuel Quantity Data Table for fuel specifications and quantities. An air pressure fuel transfer system is provided to transfer wing and external tank fuel to the fuselage cells. Hydraulic and electric transfer pumps plus gravity feed are utilized to transfer fuel from the fuselage cells to number one tank which is the engine feed tank. Single point ground pressure fueling at the rate of approximately 250 gallons per minute may be accomplished. Two point ground pressure fueling is available by using the in-flight refueling probe. Refer to Section IV, In-Flight and Ground Refueling Systems for further information. There are no gravity fueling or defueling provisions made for the internal or external fuel systems. Single point defueling is accomplished by using the single point fueling receptacle. All internal fuel cells incorporate capacitance type fuel gaging units which continuously indicate the total fuel quantity in pounds in all internal cells. The fuel system is equipped with refueling level control valves which are float type valves that shut off the pressure fueling when predetermined fuel levels are reached. All internal and external fuel tanks are pressurized in flight by regulated engine bleed air which is also utilized to transfer wing or external fuel to the fuselage cells or to dump wing fuel. The internal cells and external centerline tank or Buddy Tank are vented to a common manifold which dumps overboard under the aft end of the empennage. The external wing tanks are vented to the wing cell dump lines. With the Buddy Tank installed, the airplane becomes a tanker with the capabilities of transferring in flight a predetermined amount of its internal fuel supply (plus the Buddy Tank fuel supply) to a receiver airplane or return transfer from the Buddy Tank to its own internal fuel supply.

FUEL BOOST SYSTEM

Fuel is supplied to the engines during all flight attitudes by two submerged electric motor-driven centrifugal type boost pumps. The left pump is a two-speed type pump. During normal operation, the two-speed pump operates at high speed. In the event of a complete electrical failure or double engine failure, extending the wind-driven turbine automatically switches the two speed pump from high to low speed, thereby reducing a high amperage load and conserving electrical power and at the same time maintaining positive fuel pressure (8 psi minimum) at the engine inlet. The boost pumps are located in the engine feed (No. 1) tank. Both pumps are mounted on the bottom of the tank and provide for negative "g" requirements. Due to internal tank baffling and check valves, which trap approximately 133 gallons of fuel in the lower third of the tank during inverted flight, the boost pumps will always remain submerged and provide a constant fuel flow at 30 ± 2 psi to the engines. The two boost pumps will operate when either engine master switch is ON, provided a-c power is supplied to the system. The left main a-c bus supplies power for the right boost pump. Power to the left boost pump high speed relay is supplied by the right main 28 volt d-c bus. The left boost pump low speed relay draws its power from the essential 28 volt d-c bus. The power for operation of the left boost pump is supplied by the essential 115/200 volt a-c bus.

WARNING

In the event of a double engine failure and loss of electrical power, lowering the wind-driven turbine will automatically switch the left boost pump to low speed. The low speed boost pump plus gravity feed will supply enough fuel pressure to the engine driven fuel pumps to enable the engines to be started. Refer to Section III, Emergency Procedures, for additional information.

BOOST PUMP PRESSURE INDICATORS

The boost pump pressure indicators (figure 1-18) are mounted on the left console in the pilot's cockpit. The gage dials are calibrated from 1 to 10 with readings multiplied by 10. Pressure transmitters mounted on the airplane keel in the engine compartment measure pressure in the aircraft fuel system as it enters the engine fuel pump. This signal is transmitted via the essential 26 volt a-c bus to the indicators in the cockpit.

FUEL BOOST PUMP CHECK

Two momentary type check switches (figure 1-7) are located on the fuel control panel to check the operation of the fuel boost pumps. This check can be conducted only with external power applied and the master switches OFF, since either master switch in the ON position will operate both boost pumps. Placing either switch in the CHECK position operates the corresponding boost pump and opens the corresponding left or right engine shutoff valve allowing a pressure transmitter to pick up engine inlet pressure. This pressure is indicated on the applicable pressure gage for that pump. The pressure gages are located on the forward end of the pilot's left console. A below normal discharge pressure reading while a pump is being checked indicates a malfunction. The fuel boost pump check switches should be operated individually in order to obtain a correct reading.

FUEL TRANSFER SYSTEM

The electric transfer pumps located in fuselage cells 4 and 6 will commence transferring fuel to the engine feed tank with the selection of the ON position of either engine master switch, providing a-c power is supplied to the system. With external hydraulic power supplied to the airplane or an engine operating, the hydraulic transfer pumps in cells 4 and 6 will also be operating and delivering fuel to tank 1, and cell 2 via float type level control valves located in tank 1 and cell 2. The level control valves open to allow fuel from the transfer pumps to enter tank 1 and cell 2 when the fuel level in these cells drops below that of the floats. Cell No. 2 transfers to tank No. 1 by gravity only; cell 3 gravity feeds cell 4; and cell 5 gravity feeds to cell 6. All internal and external tanks and cells are pressurized when the landing gear control is in the gear up position or the wing transfer pressure switch is in the emergency position and an engine is running. Wing fuel is now capable of being transferred by regulated air pressure to fuselage tank 1, and cell 3. Wing fuel will not normally enter tank 1 unless the fuel level in the tank 1 drops low enough to permit the transfer level control valve to open. On airplanes 145307b thru 145312b and 145314b thru 145317b, wing fuel will not enter cell 3 until the total remaining fuselage fuel has decreased to 5700 pounds. On all other airplanes, wing fuel is transferred to fuselage cell 3 as soon as the internal wing tanks are pressurized, providing the refueling level control valve in tank 3 opens. This action constitutes the automatic wing fuel transfer feature. Wing fuel is not transferred to cell 5 in order to prevent an aft c.g. condition. Upon the incorporation of ASC 84, airplanes 146817c thru 148275e will transfer wing and external fuel to tanks 3 and 5 simultaneously. With an engine running fuel from the external tanks will commence transferring upon selection of the desired position (OUTB'D or CENTER) on the external transfer switch (figure 1-8), providing the landing gear handle is in the gear up position or the wing transfer pressure switch is in the emergency position.

TRANSFER SYSTEM CONTROLS

Internal Wing Transfer Switch

The internal wing transfer switch (figure 1-7) is a three-position toggle switch located in the fuel control panel on the left console in the pilot's cockpit. The switch positions are marked AUTO, MAN-OVERRIDE and STOP TRANS. The AUTO position will energize the refueling level control valves closed in cells 3 and 5 until the total remaining fuselage fuel has decreased to 5700 pounds. At the 5700 pound level, the refueling level control valve in No. 3 cell is de-energized open by a switch in the fuel quantity indicator allowing the entrance of wing fuel. Selecting the MAN-OVERRIDE position overrides a failure in the fuselage fuel indicator gage switch and allows wing fuel transfer such as would normally occur at the 5700 pound level. Selecting the STOP TRANS position of the switch closes the wings internal external fuel transfer valves thus preventing further transfer to the fuselage tanks and cells. Placing or leaving the external transfer switch in any position other than OFF will also prevent wing fuel from transferring to the fuselage tank and cells. The left main 28 volt d-c essential bus applies power to operate the fuel level indicator relays and the fuel level switch. Power to operate the fuel level control valves is supplied by the right main 28 volt d-c bus. Effective airplanes 145313b and 146817c thru 148275e, the internal wing transfer switch has only two position, normal and stop transfer. In the NORMAL position, internal wing fuel is transferred to the fuselage tanks as soon as the internal wing tanks are pressurized, providing the refueling level control valves are open. The STOP TRANS position operates the same as previously described.

Wing Transfer Pressure Switch

The wing transfer pressure switch (figure 1-7) is a two-position toggle switch located in the fuel control panel on the left console of the pilot's cockpit. The switch positions are marked NORMAL and EMER. When the landing gear handle is in the UP position and the wing transfer switch is in the NORMAL position, all internal and external tanks become pressurized by the pressure regulator valves being de-energized open and the pressure relief valves energized closed. This maintains 15 ± .5 psi air pressure in the wing and external tanks and 2 ± .5 psi in the fuselage tanks. Wing fuel will commence transferring, providing the external transfer switch is in the OFF position. Placing the switch in the EMER position performs the same functions as did the landing gear handle switch; all pressure regulators open and all pressure relief valves close; the tanks are thereby pressurized and ready to transfer. The left main 28 volt d-c bus supplies power for this circuit.

External Transfer Switch

The external transfer switch (figure 1-7) is a three-position toggle switch located in the fuel control panel on the left console of the pilot's cockpit. The switch positions are marked CENTER, OFF and OUTB'D. Upon the selection of the CENTER position, the internal wing tank shutoff valves are closed, the centerline tank fuel shutoff valve and the fuel shutoff valve located in the pressure fueling line are energized open, allowing fuel to transfer. Placing the switch in the OUTB'D position opens the left and right external tanks shutoff valves, and closes the internal wing tanks shutoff valves, thus external fuel commences to transfer. In airplanes 145307b thru 145312b and 145314b thru

AIRPLANE FUEL SYSTEM

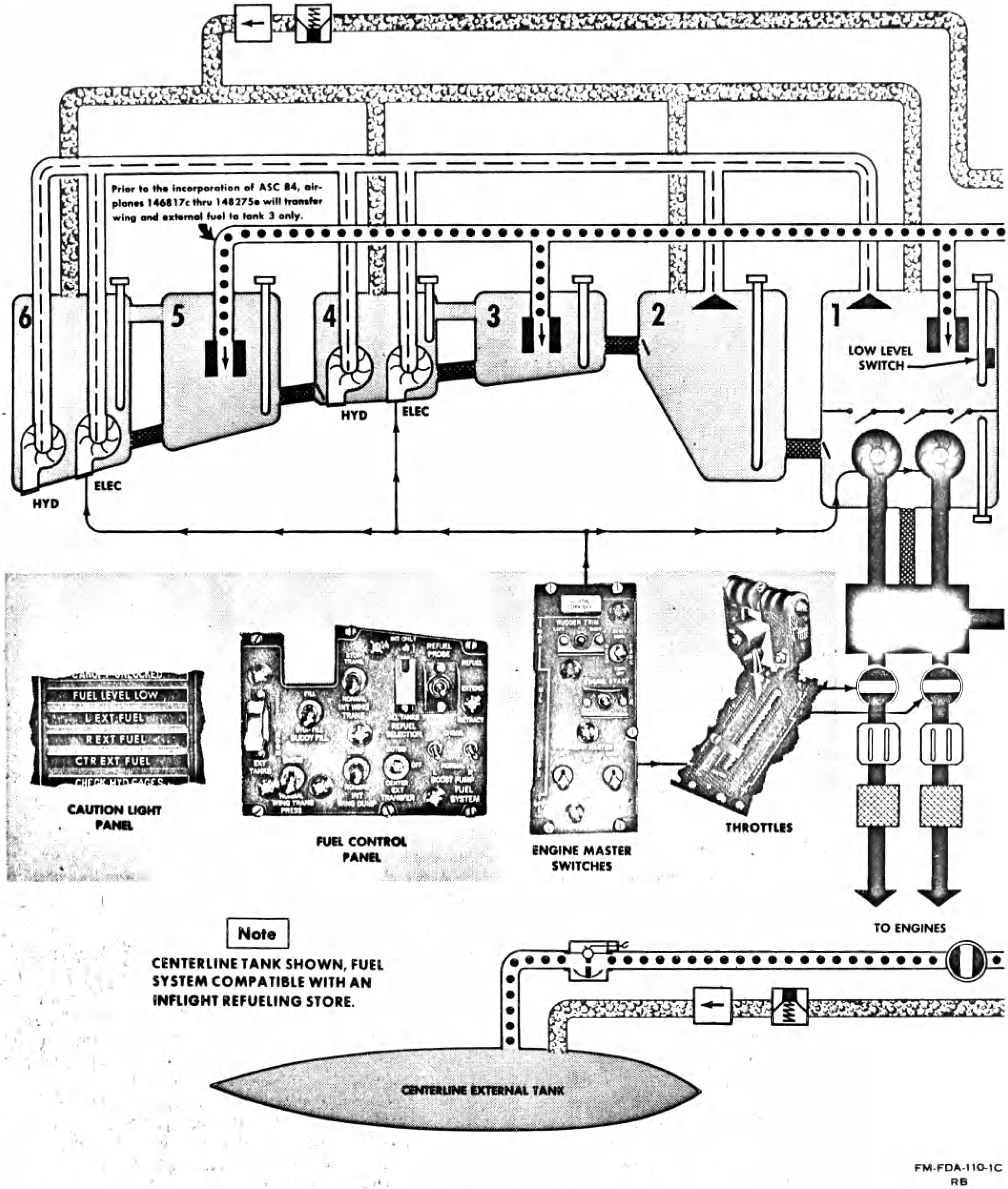


Figure 1-6 (Sheet 1)

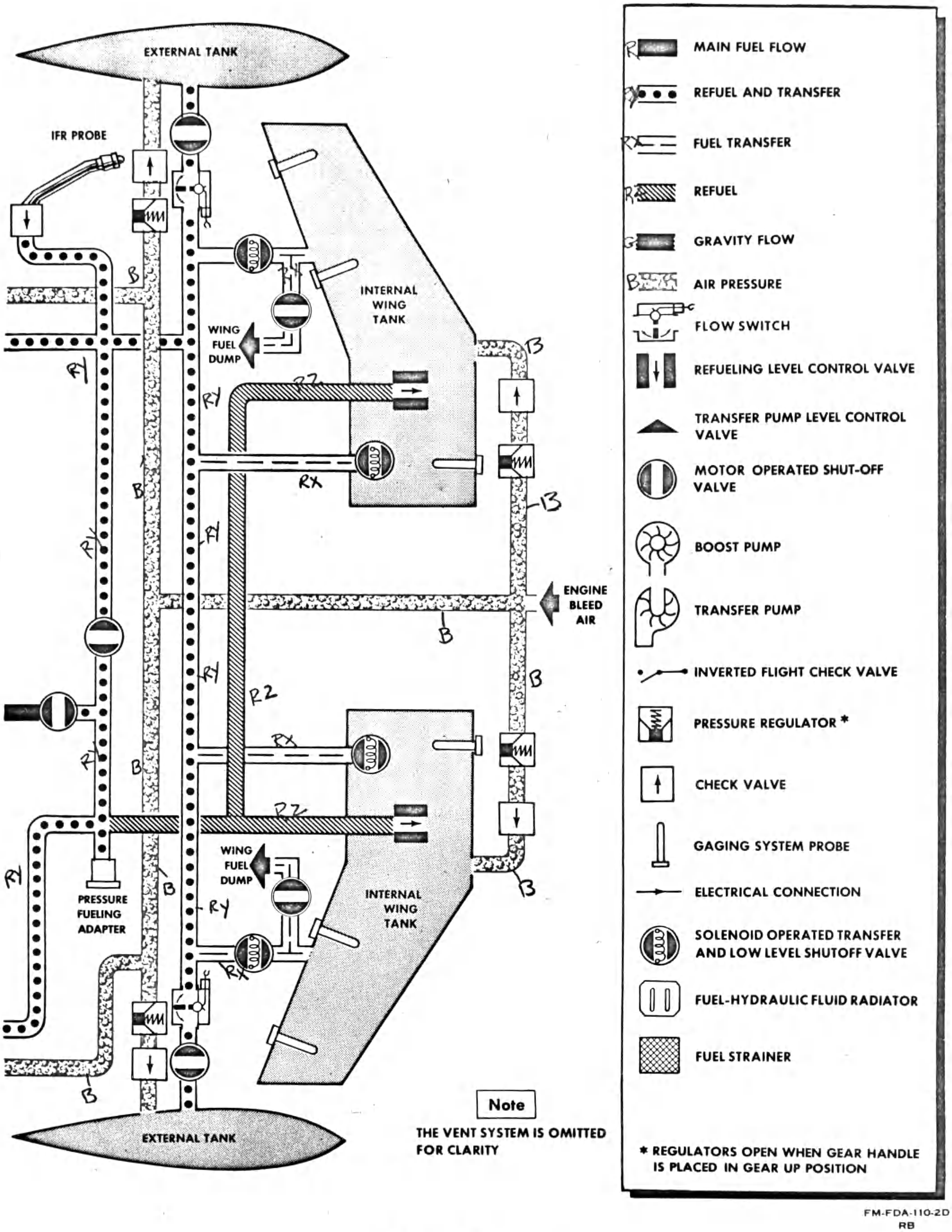


Figure 1-6 (Sheet 2)

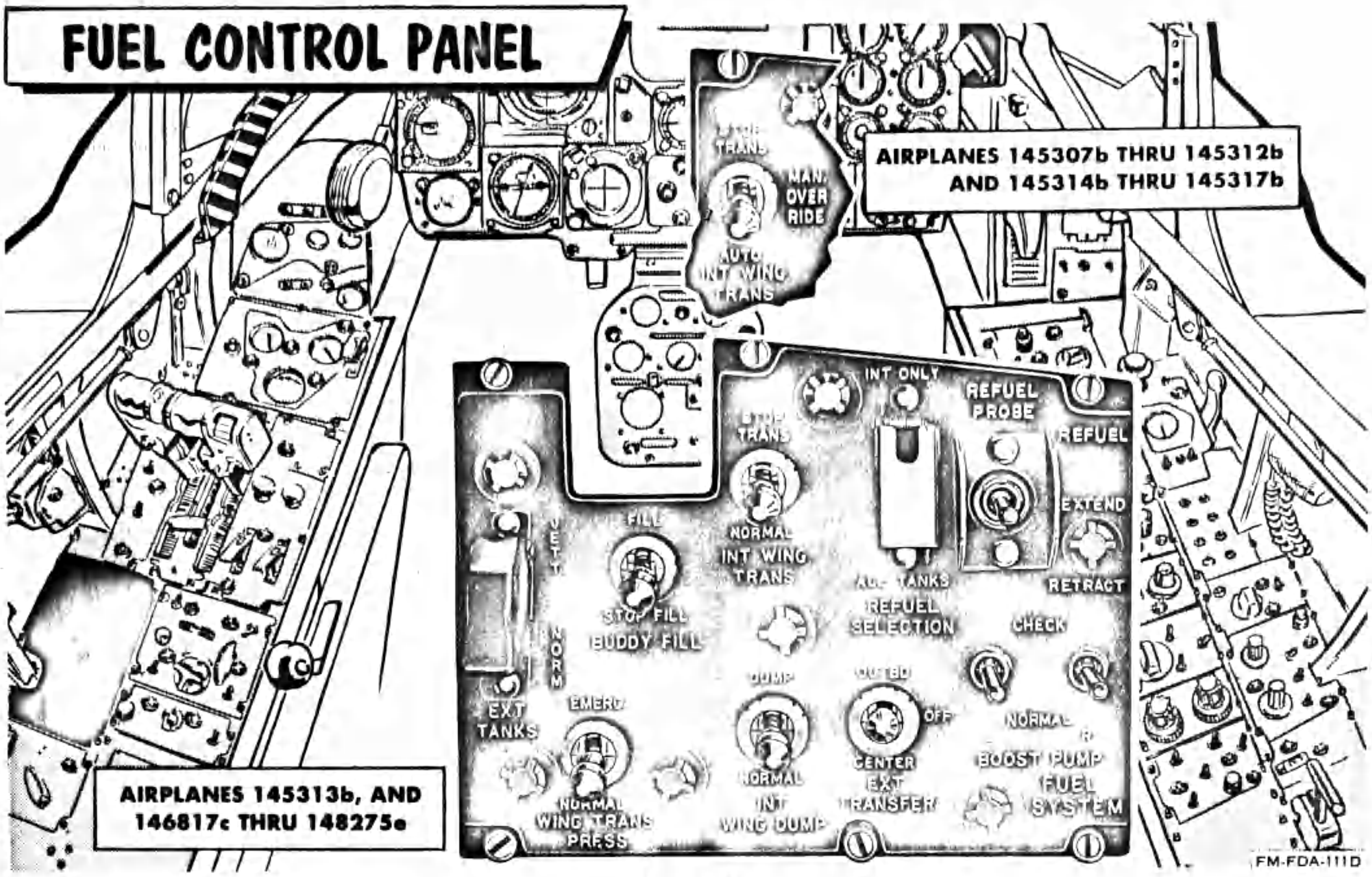


Figure 1-7

145317b if the fuel level in the fuselage cells is above the 5700 lb. level, the external tanks will transfer to fuselage cells 3 and 5. If the fuel level is below the 5700 lb. level, the external tanks will transfer fuel to fuselage cell 3 only, to prevent an aft c.g. condition from occurring. In airplanes 145313b and 146817c thru 148275e, external fuel will transfer only to fuselage cell 3 regardless of the amount of fuel in the fuselage cells. Fuel will never transfer to fuselage cell 5 except during refueling operations. Upon the incorporation of ASC 84, airplanes 146817c thru 148275e will transfer wing and external fuel to tanks 3 and 5 simultaneously. The right main 28 volt d-c bus supplies power to operate the centerline and external tanks pressurization and selector relays. Power to operate the external and centerline tanks fuel transfer valves is supplied by the right main 28 volt d-c bus.

CAUTION

To prevent drop tank collapse during high altitude descent with wheels down, place wing trans press switch to EMERG. Place wing trans press switch to NORMAL prior to landing.

Note

Wing fuel will not transfer if the external transfer switch is in any position other than OFF.

Emergency Fuel Transfer

There are no provisions for an emergency fuel transfer system on this airplane. With hydraulic and electric fuselage transfer pumps working simultaneously, and the utilization of air pressure for wing and external tanks transfer, the possibility of a complete fuel transfer system failure is highly improbable.

EXTERNAL TANK JETTISON SYSTEM

External Tank Jettison Switch

The external wing tanks can be jettisoned by simply selecting the JETT position on the fuel control panel (figure 1-7) located on the left console in the pilot's cockpit. The tanks can be jettisoned before or after the flow indicating light illuminates. Illumination of the flow light indicates flow has ceased and external tanks are empty. If the external transfer switch has been inadvertently left in either the OUTB'D or CENTER position and external tanks are not installed on the airplane, or the tanks have been jettisoned, the external wing tanks fuel shutoff valve will close and the switch will be ineffective, allowing wing fuel to transfer in its normal manner. The essential 28 volt d-c bus supplies power for the external wing tank jettison circuit.

CAUTION

The external wing tanks can be jettisoned by the external wing tank jettison switch any time power is on the airplane and the external tanks safety pins are removed. This circuit is not wired through the landing gear handle.

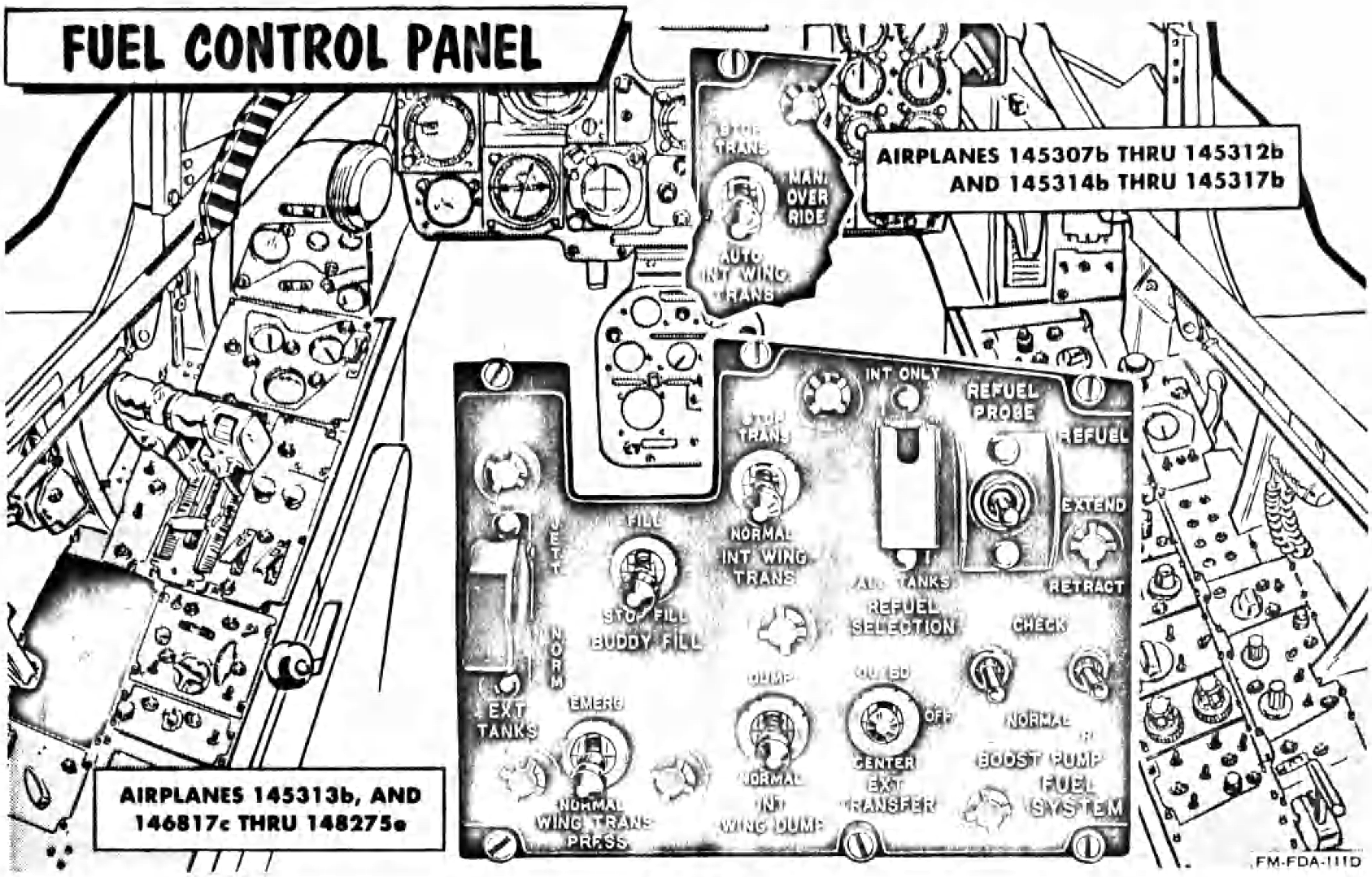


Figure 1-7

145317b if the fuel level in the fuselage cells is above the 5700 lb. level, the external tanks will transfer to fuselage cells 3 and 5. If the fuel level is below the 5700 lb. level, the external tanks will transfer fuel to fuselage cell 3 only, to prevent an aft c.g. condition from occurring. In airplanes 145313b and 146817c thru 148275e, external fuel will transfer only to fuselage cell 3 regardless of the amount of fuel in the fuselage cells. Fuel will never transfer to fuselage cell 5 except during refueling operations. Upon the incorporation of ASC 84, airplanes 146817c thru 148275e will transfer wing and external fuel to tanks 3 and 5 simultaneously. The right main 28 volt d-c bus supplies power to operate the centerline and external tanks pressurization and selector relays. Power to operate the external and centerline tanks fuel transfer valves is supplied by the right main 28 volt d-c bus.

CAUTION

To prevent drop tank collapse during high altitude descent with wheels down, place wing trans press switch to EMERG. Place wing trans press switch to NORMAL prior to landing.

Note

Wing fuel will not transfer if the external transfer switch is in any position other than OFF.

Emergency Fuel Transfer

There are no provisions for an emergency fuel transfer system on this airplane. With hydraulic and electric fuselage transfer pumps working simultaneously, and the utilization of air pressure for wing and external tanks transfer, the possibility of a complete fuel transfer system failure is highly improbable.

EXTERNAL TANK JETTISON SYSTEM

External Tank Jettison Switch

The external wing tanks can be jettisoned by simply selecting the JETT position on the fuel control panel (figure 1-7) located on the left console in the pilot's cockpit. The tanks can be jettisoned before or after the flow indicating light illuminates. Illumination of the flow light indicates flow has ceased and external tanks are empty. If the external transfer switch has been inadvertently left in either the OUTB'D or CENTER position and external tanks are not installed on the airplane, or the tanks have been jettisoned, the external wing tanks fuel shutoff valve will close and the switch will be ineffective, allowing wing fuel to transfer in its normal manner. The essential 28 volt d-c bus supplies power for the external wing tank jettison circuit.

CAUTION

The external wing tanks can be jettisoned by the external wing tank jettison switch any time power is on the airplane and the external tanks safety pins are removed. This circuit is not wired through the landing gear handle.

External Stores Emergency Release Button

The external stores emergency release button (figure 1-16) is located on the left vertical panel. This button when depressed, will jettison all external stores except missiles, carried on the airplane, providing the landing gear handle is up. The external tanks can be jettisoned before or after the flow indicating light illuminates. Illumination of the flow light indicates flow has ceased and external tanks are empty. If the external transfer switch is in the CENTER position at time of jettison, the centerline tanks fuel shutoff valve will close and the switch will be ineffective, allowing wing fuel to transfer in its normal manner. The external stores emergency release circuit receives power from the essential 28 volt d-c bus. Centerline external stores only may be jettisoned (providing the landing gear handle is up) by selecting the DIRECT position on the bomb control switch and then depressing the bomb release button on the control stick grip. Electrical power for bomb release circuit is provided by the right main 28 volt d-c bus.

PRESSURIZATION AND VENT SYSTEM

The pressurization and vent system provides regulated engine bleed air pressure to all internal and external tanks for pressurization fuel transfer and wing dump. The system also provides for relief of external tanks to prevent collapse of tanks during fast descents.

Wing Tank Pressurization and Vent

The wing cells and external tanks pressurization system utilizes pressure regulators and pressure relief valves which are set respectively at 15 ± .5 psi and 17.5 ± .5 psi. The wing cell pressure relief valves, providing fuel tank pressure and vacuum relief, dump into a common manifold which is vented overboard under the aft end of the empennage. The external wing tanks are vented through their pressure relief valves to the wing cell dump lines. The wing cells and external wing tanks are vented to the atmosphere when the landing gear is extended.

Fuselage Tank Pressurization and Vent

The fuselage tank pressure regulator in conjunction with the flow limiter and pressure relief vent valve will maintain regulated air pressure at 2 ±.5 psi and pressure relief at 3.5 ± .5 psi in the fuselage fuel tank and cells. The fuselage tank and cells and the Buddy Tank or centerline external tank are vented to the common fuel vent manifold and then dumped overboard through the fuselage pressure relief valve which also provides for fuselage fuel tank ventilation and vacuum relief. When the airplane is on the ground, all pressure relief valves are open venting all tanks to the atmosphere.

WING FUEL DUMP

Wing fuel may be dumped in flight at any time regardless of any other transfer position by selecting the DUMP position on the internal wing dump switch (figure 1-7). The two-position toggle switch marked NORMAL and DUMP is located on the fuel control panel on the left console of the pilot's cockpit. Selecting the DUMP position opens the left and right wing dump shutoff valves and closes the wing transfer and vent valves. The wing air regulator will open allowing the wing tank to remain pressurized and force fuel out the dump lines at the wing fold trailing edge. Air pressure will continue to bleed out the dump line until the internal wing dump switch is placed in the NORM position to close the dump valves. The right main 28 volt d-c bus supplies power for this circuit.

CAUTION

- Due to the fact that the internal wing dump switch will function with the engine master switch ON or OFF, and the landing gear UP or DOWN, wing fuel will be dumped ON THE DECK when internal wing dump switch is placed in the DUMP position and external power is applied to the airplane.

- To prevent drop tank collapse during descent with wheels down while dumping fuel, place wing trans press switch in EMERG. Place wing trans press switch to NORMAL prior to landing.

FUEL QUANTITY INDICATING SYSTEM

The fuel quantity indicating system is of the capacitance type and provides a reading in pounds of total internal fuel. The system components include the fuel quantity indicator, fuel check button and a fuel level low indicator light. There are thirteen fuel gaging units located throughout the internal tanks which register at the one cockpit fuel quantity gage. The system is powered by the essential 115 volt a-c bus.

Fuel Quantity Indicator

A combination (counter-pointer) fuel quantity indicator (figure 1-17) is located in upper right corner of the pilot's instrument panel. The counter unit of the gage continuously indicates the total usable fuel quantity (with readings multiplied by 10) in all internal tanks. The pointer portion of the indicator simultaneously indicates the total usable fuel quantity in the fuselage tanks only with readings multiplied by 1000. Power for the fuel quantity gage circuit is supplied by the essential 115 volt d-c bus.

WARNING

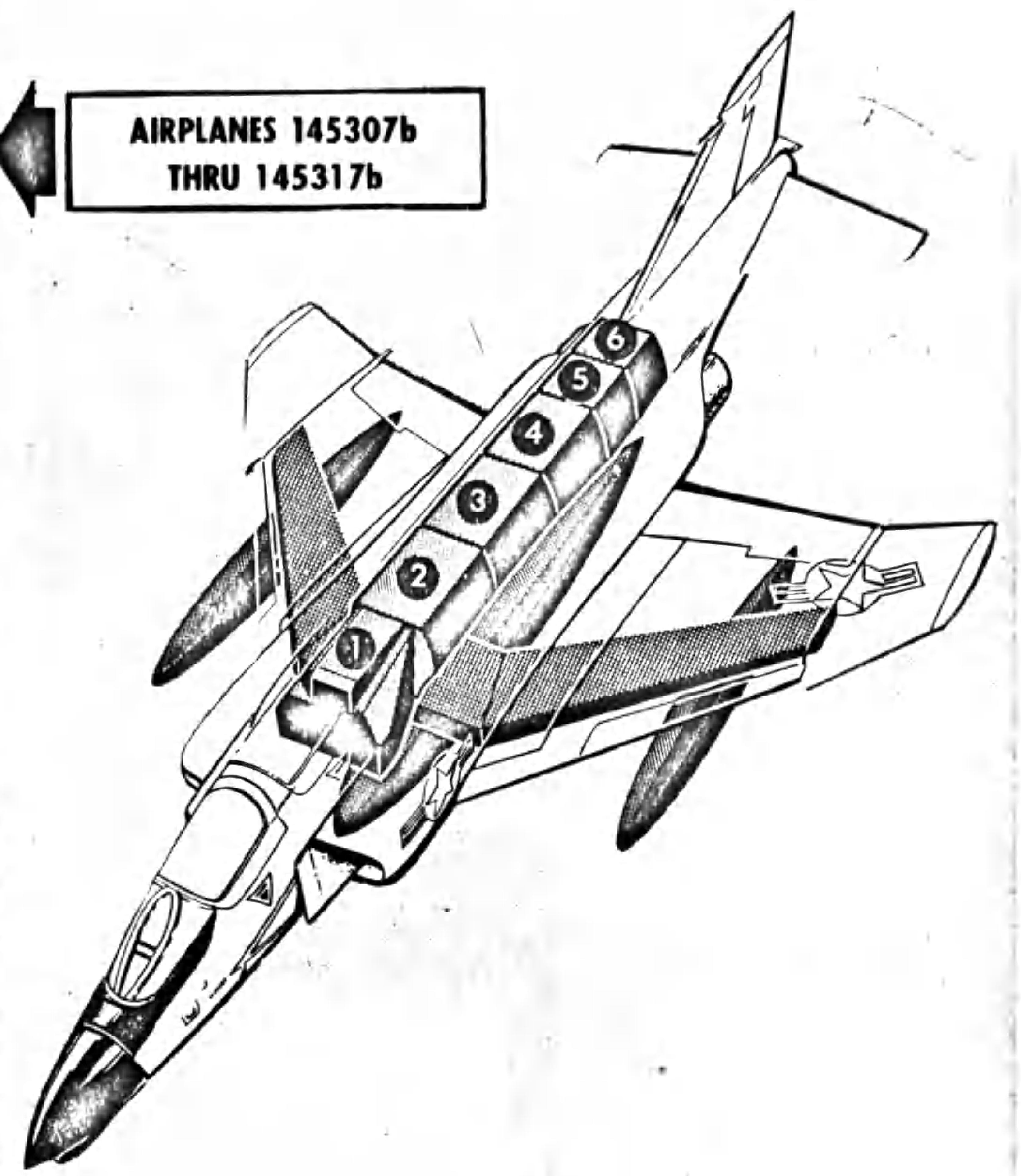
The fuel quantity gage may give unreliable (higher than actual) readings in the low end of the scale (0-1000 pounds). Therefore, the fuel level low warning light should be used as the primary indication of a low fuel state. Continued operation after illumination of this light should be cautiously considered.

FUEL QUANTITY DATA TABLES

AIRPLANES 145307b THRU 148275e

AIRPLANES 145307b THRU 145317b

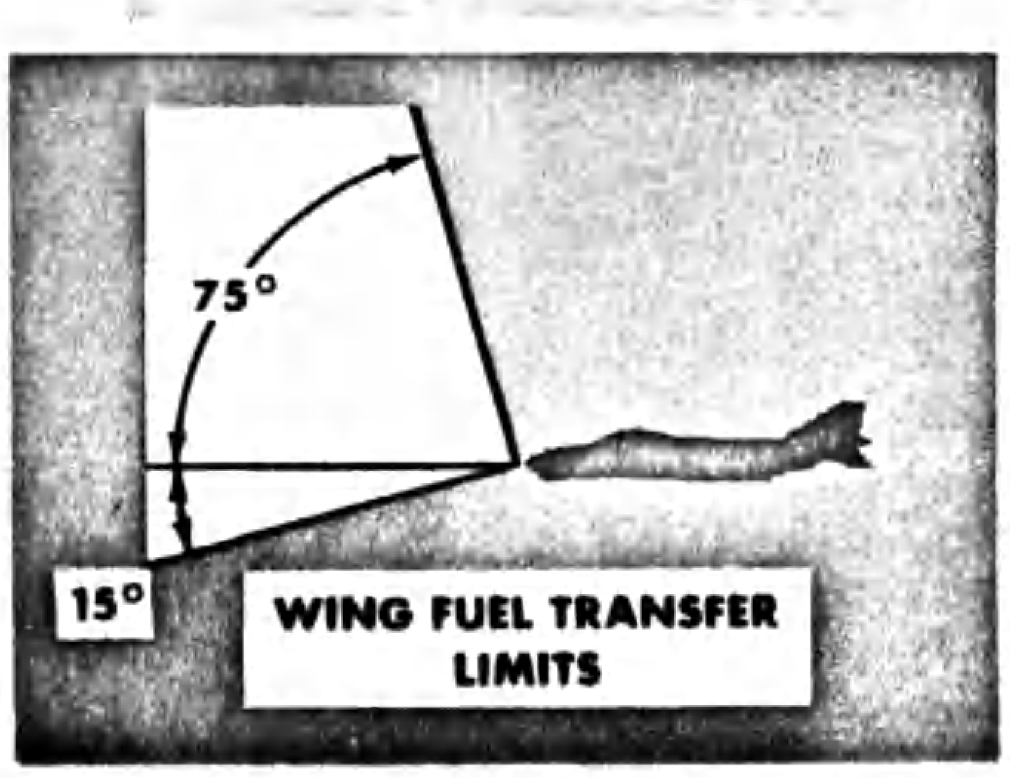
	FULLY SERVICED (GALLONS)	FULLY SERVICED (POUNDS)	USABLE FUEL (GALLONS)	USABLE FUEL (POUNDS)
FUSELAGE CELLS	1,336	9,085	1,315	8,942*
INTERNAL WING TANKS	650	4,420	642	4,366*
TOTAL INTERNAL	1,986	13,505	1,957	13,308*
EXTERNAL WING TANKS	744	5,059	740	5,032
INTERNAL FUEL PLUS EXT. WING TANKS	2,730	18,564	2,697	18,340
EXT. CENTER TANK	602	4,094	600	4,080
INTERNAL FUEL PLUS EXT. CENTER TANK	2,588	17,599	2,557	17,388
MAX. POSSIBLE FUEL TOTAL INTERNAL PLUS ALL EXTERNAL TANKS	3,332	22,658	3,297	22,420



AIRPLANES 146817c THRU 146819c AND 148255d THRU 148275e

	FULLY SERVICED (GALLONS)	FULLY SERVICED (POUNDS)	USABLE FUEL (GALLONS)	USABLE FUEL (POUNDS)
FUSELAGE CELLS	1,361	9,255	1,341	9,118*
INTERNAL WING TANKS	650	4,420	642	4,366*
TOTAL INTERNAL	2,011	13,675	1,983	13,484*
EXTERNAL WING TANKS	744	5,059	740	5,032
INTERNAL FUEL PLUS EXT. WING TANKS	2,755	18,734	2,723	18,516
EXT. CENTER TANK	602	4,094	600	4,080
INTERNAL FUEL PLUS EXT. CENTER TANK	2,613	17,769	2,583	17,564
MAX. POSSIBLE FUEL TOTAL INTERNAL PLUS ALL EXTERNAL TANKS	3,357	22,828	3,323	22,596

*GAGEABLE FUEL



Note

WEIGHTS ARE BASED ON 6.8 POUNDS PER GALLON FOR STANDARD DAY TEMPERATURE. JP-5 DENSITY VARIES FROM 6.68 TO 7.0 LB./GAL. IF ACCURACY IS REQUIRED FUEL DENSITY SHOULD BE MEASURED.

DATA DOES NOT APPLY TO AIRPLANES 146820c THRU 148254d. THESE AIRPLANES ARE DESIGNATED AS PROTOTYPES

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Figure 1-8

Note

There is a possibility that fuel quantity variations will be noted on the fuel quantity indicator during aircraft accelerations and decelerations. These variations are due to the high acceleration and deceleration rates which can be achieved with the airplane. Transient increases in fuel quantity readings may be noted during deceleration, and transient decreases in fuel quantity readings may be noted during acceleration.

Fuel Check Switch

The fuel check switch (figure 1-17) is located on the upper right corner of the pilot's instrument panel. This push button type switch is incorporated in the fuel indicator circuit to enable the pilot to periodically check the fuel indicator for correct operation. With the loss of the essential 115 volt a-c bus, the fuel quantity indicator will be inoperative; and the fuel indicator will remain at the indication at the time of power loss. When the pilot depresses the fuel check switch with power on the circuit, the pointer and counter will start moving toward zero. If the switch is held in long enough, the pointer and counter will go to zero and when released, return to its original reading. This is an indication that there is power to the circuit and that the indicator is functioning properly. If the fuel check switch is depressed and the pointer and counter do not move, this indicates a loss of the 115 volt a-c essential bus or a malfunction in the fuel quantity indicating system. Effective airplane 145313b, 146817c thru 148275e, the fuel check switch has been removed, and a feed tank check switch has been added. The two-position feed tank check switch, with switch positions of check and normal, gives the pilot the opportunity of checking the fuel quantity in the engine feed tank. When the switch is placed in the spring-loaded CHECK position, the sector portion and the counter portion of the fuel quantity gage will both read engine feed tank fuel quantity. Aside from the fact that the feed tank check switch affords an opportunity of checking feed tank fuel quantity, it is also an indication that there is power to the fuel quantity circuits and that the gage is functioning properly.

WARNING LIGHTS

In order to reduce the amount of instrument surveillance required of the pilot, warning lights have been incorporated and are grouped on the right vertical panel (figure 1-10) of the pilot's cockpit. In addition, a Master Caution light (figure 1-17) is located on the upper right corner of the pilot's instrument panel above the fuel quantity indicator. When a malfunction exists, the Master Caution light and a warning light on the panel will illuminate informing the pilot of the condition for which caution should be exercised. The pilot may extinguish the Master Caution light on the instrument panel by depressing a button (figure 1-10) located on the right vertical panel. When an additional caution condition exists, the Master Caution light and a warning light will again illuminate. The pilot need only watch the Master Caution light for an indication of caution condition and then refer to the warning lights panel which will inform him of the caution condition existing. Power is supplied to this system by the essential 26 volt a-c bus.

Fuel Level Low Warning Light

The "Fuel Level Low" warning light located on the right vertical panel (figure 1-10) illuminates when the combined usable fuel in the engine feed tank and tank No. 2 is reduced to approximately 1960 ± 200 lb. level (288 ± 30 gal.) for JP-5 fuel. The "Fuel Level Low" light will illuminate at the above fuel quantities only if the airplane is in a perfectly level attitude and moderate stabilized power settings are being used. However, due to the various attitudes and power settings required during a normal flight, the illumination of the "Fuel Level Low" warning light is not an accurate indication of the amount of fuel remaining in the No. 1 and No. 2 tanks. The illuminated light will only serve notice to the pilot that his fuel is low. In this system the unit which operates the low level warning light is a thermistor sensing switch which is located on the engine feed tank fuel quantity probe. When the fuel level in No. 1 and No. 2 tanks is above the sensor switch, the resistance of the reference thermistor (which is enclosed in an air filled capsule) is much less than the resistance of the sensing thermistor, causing an unbalance in the bridge circuit. The relay is energized and the "Fuel Level Low" warning light is out. When the fuel level drops below the sensor switch the thermistors are exposed to air, and resistances of the reference and sensing thermistors are equal, balancing the bridge circuit. The relay then becomes de-energized and allows current flow to the "Fuel Level Low" warning light.

Left and Right External Tank Fuel Lights

The "L. Ext Fuel" or "R. Ext Fuel" warning lights located on the warning lights panel (figure 1-10) are provided to indicate an empty left or right outboard external tank with OUTB'D position selected on the external transfer switch. One or the other external fuel indicator lights and the Master Caution light will illuminate simultaneously when fuel flow from one of the external wing tanks ceases. This will notify the pilot that the tank indicated is empty or flow is interrupted. Since external fuel transfer is intermittent rather than continuous, the "Master Caution" light and the "L. Ext Fuel" and the "R. Ext Fuel" warning lights will come on during a temporary halt of fuel flow. After the "Master Caution" light is cleared and the external tanks fuel warning lights have gone out, the lights will again illuminate approximately 10 to 30 seconds later during the next interruption of fuel flow. Intermittent external fuel transfer is desired since this means the transfer rate is greater than engine consumption and fuselage fuel is being maintained at its highest possible volume. The "L. Ext Fuel" or "R. Ext Fuel" warning lights will also illuminate when the tanks are full during refueling

ELECTRICAL SYSTEM

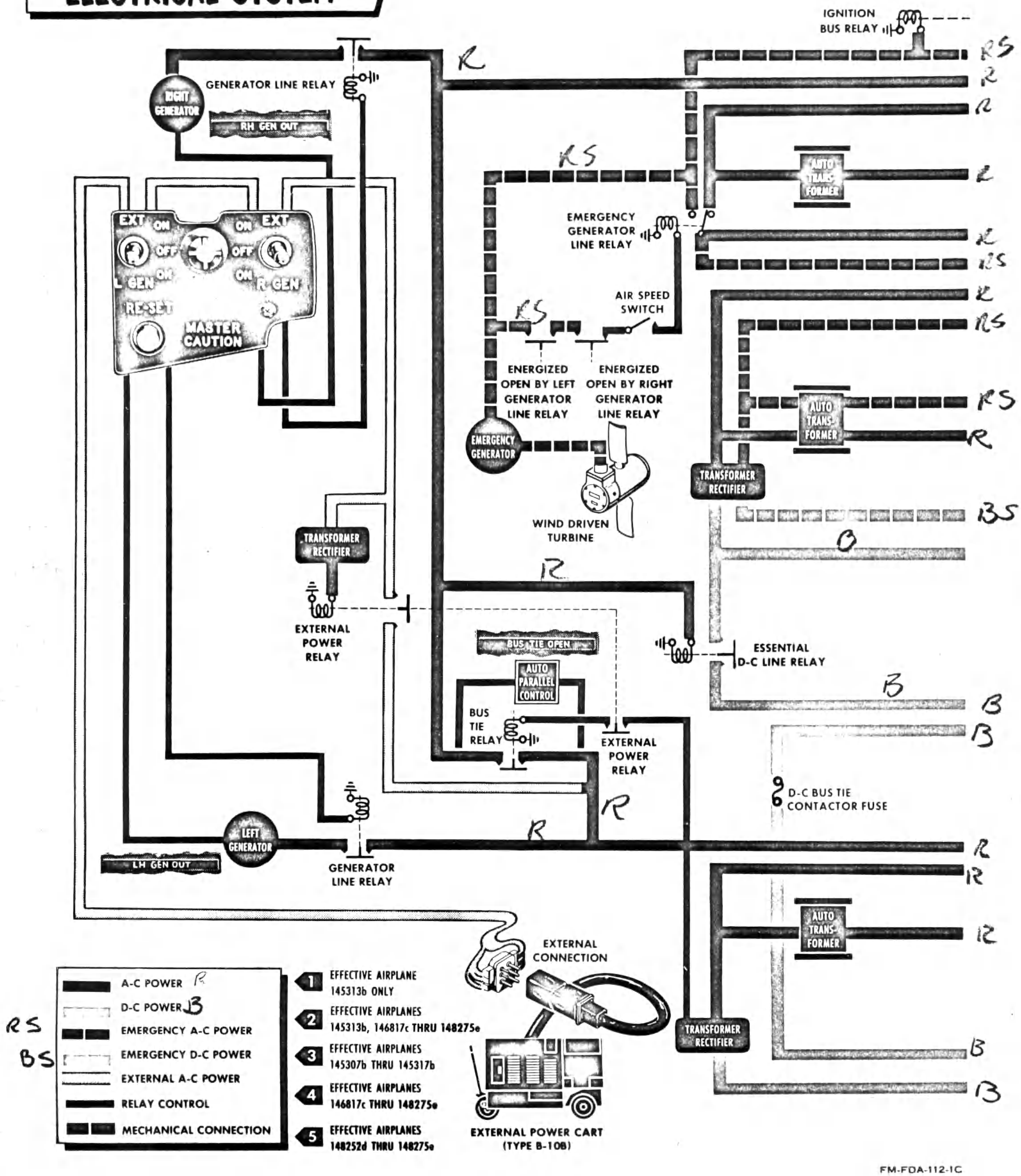

Figure 1-9 (Sheet 1)

IGNITION 115V A-C BUS

LEFT MAIN IGNITION
RIGHT MAIN IGNITION

RIGHT MAIN 115/200V A-C BUS

ADF
AFT COCKPIT INSTRUMENT LIGHTS ◄1
AILERON FEEL TRIM (POWER) ◄3
AILERON RUDDER INTERCONNECT (POWER)
ALTIMETER VIBRATOR (POWER)
AUTOPILOT
COCKPIT HEAT AND VENT (CONTROL)
ENGINE RAMPS (CONTROL)
NAVIGATION COMPUTER
NO. 6 TRANSFER PUMP (POWER)
OXYGEN GAGING SYSTEM
PITOT HEATERS
RADAR ALTIMETER SYSTEM (POWER)
RIGHT AUTO TRANSFORMER
SIDEWINDER
TACAN
UHF COMMUNICATIONS

RIGHT MAIN 26V A-C BUS

AFT COCKPIT INSTRUMENT LIGHTS (BRIGHT) ◄2
AILERON RUDDER INTERCONNECT (CONTROL)
AIRBORNE MISSILE CONTROL SYSTEM
ANTI-COLLISION LIGHTS
UTILITY LIGHTS
WHITE FLOODLIGHTS
WING AND TAIL LIGHTS (BRIGHT)

ESSENTIAL 115/200V A-C BUS

AFT COCKPIT INSTRUMENT LIGHTS ◄3
AILERON FEEL TRIM (POWER) ◄2
ESSENTIAL AUTO TRANSFORMER
EXHAUST GAS TEMPERATURE INDICATING SYSTEM
FORWARD COCKPIT INSTRUMENT LIGHTS
FUEL QUANTITY INDICATING SYSTEM
IFF
INTERCOM
LEFT FUEL BOOST PUMP (POWER)
LOFT BOMB RELEASE COMPUTER SET
UHF COMMUNICATIONS

ESSENTIAL 26V A-C BUS

AFT COCKPIT FLOODS (BRIGHT)
AILERON POSITION INDICATOR
CENTRAL AIR DATA COMPUTER
CAUTION LIGHT RESET
ENGINE FUEL FLOW INDICATING SYSTEM
ENGINE FUEL PRESSURE INDICATING SYSTEM
ENGINE OIL PRESSURE INDICATING SYSTEM
LOFT BOMB RELEASE COMPUTER SET
P.C. 1 HYDRAULIC PRESSURE INDICATING SYSTEM
PNEUMATIC PRESSURE INDICATING SYSTEM
RUDDER POSITION INDICATING SYSTEM
TAXI LIGHTS
WARNING LIGHTS (BRIGHT)

ESSENTIAL 28V D-C BUS

BUDDY TANK HOSE JETTISON
CENTRAL AIR DATA COMPUTER
EXTERNAL STORES EMERGENCY RELEASE
EXTERNAL WING TANK JETTISON (POWER)
IFF
INTERCOM
LANDING GEAR AND FLAP POSITION IND.
LATERAL AND LONGITUDINAL TRIM SYSTEM
LEFT FUEL BOOST PUMP, EMERGENCY (CONTROL)
LOFT BOMB RELEASE COMPUTER SET
MAIN FUEL CONTROL
MISSILE JETTISON (POWER)
NOZZLE POSITION INDICATING SYSTEM
STABILATOR FEEL TRIM (POWER)
UHF COMMUNICATIONS

RIGHT MAIN 28V D-C BUS

ADF
ANGLE-OF-ATTACK TRANSMITTER HEATER
ARMAMENT BUS (CONTROL)
AUTOPILOT
CENTERLINE STORE RELEASE (PICKLE BUTTON)
COCKPIT HEAT AND VENT (CONTROL)
EQUIPMENT COOLING (CONTROL)
EXTERIOR LIGHTS (CONTROL)
EXTERNAL WING FUEL TRANS (CONTROL)
FLAPS
INTERNAL WING DUMP ◄2
INTERNAL WING FUEL TRANSFER (CONTROL) ◄3
LANDING GEAR (CONTROL)
LEFT FUEL BOOST PUMP (CONTROL)
MISSILES
NAVIGATION COMPUTER
NO. 6 TRANSFER PUMP CONTROL
NOSE WHEEL STEERING (CONTROL) ◄5
RADAR ALTIMETER SYSTEM
RUDDER FEEL TRIM (CONTROL)
RUDDER TRIM SYSTEM
SIDEWINDER
SPEED BRAKE (CONTROL)
STABILATOR FEEL TRIM (POWER)
TACAN
UHF COMMUNICATIONS
UTILITY RECEPTACLE
WARNING LIGHTS CONTROL

LEFT MAIN 115/200V A-C BUS

AFTERBURNER IGNITION
AFT COCKPIT EQUIPMENT LIGHTS ◄5
AIRBORNE MISSILE CONTROL SYSTEM
ANTI-ICE (CONTROL)
BUDDY TANK FUEL PUMP (POWER)
BUDDY TANK HYDRAULIC PUMP (POWER)
EQUIPMENT COOLING (CONTROL)
FIRE DETECTOR SYSTEM
FORWARD COCKPIT CONSOLE LIGHTS
INFRARED COOLING
LEFT AUTO TRANSFORMER
NOSE WHEEL STEERING (POWER) ◄5
NO. 4 TRANSFER PUMP (POWER)
RIGHT FUEL BOOST PUMP (POWER)
SEAT ADJUSTMENT
SIDEWINDER
SPARROW III
WINDSHIELD TEMP SENSING

LEFT MAIN 26/14V A-C BUS

AFT COCKPIT INSTRUMENT LIGHTS (DIM) ◄2
AFT COCKPIT FLOODS (DIM)
APPROACH LIGHTS (POWER)
FORWARD COCKPIT RED CONSOLE FLOODS (DIM)
FORWARD COCKPIT RED INSTRUMENT LIGHTS (DIM)
FUSELAGE AND ANTI-COLLISION LIGHTS
P.C. 2 HYDRAULIC PRESSURE INDICATING SYSTEM
UTILITY HYDRAULIC PRESSURE INDICATING SYSTEM
WARNING LIGHTS (DIM)
WING AND TAIL LIGHTS (DIM)

LEFT MAIN 28V D-C BUS

AILERON RUDDER INTERCONNECT (CONTROL)
AIRBORNE MISSILE CONTROL SYSTEM
APPROACH LIGHTS (CONTROL)
ARRESTING HOOK (CONTROL)
AUXILIARY AIR DOORS (CONTROL)
BELLMOUTH (CONTROL)
BUDDY TANK (CONTROL)
EXTERNAL STORES EMERGENCY RELEASE
FUEL LEVEL LOW WARNING
EXTERNAL FUEL SHUTOFF VALVES (POWER)
IN-FLIGHT REFUELING PROBE (CONTROL)
INFRARED COOLING SOLENOID ◄4
INTERNAL WING DUMP ◄3
INTERNAL WING FUEL TRANSFER
MISSILES (POWER) ◄2
NO. 4 TRANSFER PUMP (CONTROL)
OUTBOARD TANK JETTISON
PNEUMATIC SYSTEM (CONTROL)
RADAR MISSILE FAIRING (POWER)
RAIN REMOVAL (CONTROL)
RIGHT BOOST PUMP (CONTROL)
SIDEWINDER
STABILATOR TRIM INDICATOR
STATISTICAL ACCELEROMETER
TRAILING EDGE BOUNDARY LAYER (CONTROL)
TURN AND SLIP INDICATOR ◄4
WING FOLD (CONTROL)

FM-FDA-112-2D
RB

Figure 1-9 (Sheet 1)

operation. Power is supplied to the indicator lights by the essential 26 volt a-c bus.

Note

When selecting the outboard or refuel positions on the fuel control panel, the "L. Ext Fuel", or "R. Ext Fuel" and "Master Caution" lights will illuminate any time fuel flow is less than 4.5 gpm.

Centerline External Tank Fuel Light

The Ctr Ext Fuel warning light (figure 1-10) is provided to indicate an empty centerline external tank with CENTER position selected on the external transfer switch. The "Ctr Ext Fuel" warning light and the "Master Caution" light illuminate simultaneously when fuel flow ceases. The "Ctr Ext Fuel" caution light will also illuminate when the tank is full during a refueling operation or when fuel flow stops during fuel transfer to the Buddy Tank. Power is supplied to the indicator light by the essential 26 volt a-c bus.

Note

When selecting the center, buddy fill or refuel positions on the fuel control panel, the "Ctr Ext Fuel" and "Master Caution" lights will illuminate any time fuel flow is less than 4.5 gpm.

ELECTRICAL POWER SUPPLY SYSTEM

The airplane electrical power supply system (figure 1-9) consists of an a-c generator power supply, a transformer-rectifier d-c power supply and the power distribution system. The a-c power supply distributes power to the left and right main 115/200 volt a-c bus, the left main 26/14 volt a-c bus, the essential 115/200 volt a-c bus, the essential 26 volt a-c bus, and the right main 26 volt a-c bus. The d-c power supply distributes power to the left and right main 28 volt d-c bus and the essential 28 volt d-c bus.

A-C ELECTRICAL POWER

The primary source of electrical power on the airplane is a 400 cycle, three-phase, 115/200 volt, parallel a-c power generating system. The system consists of two generators, two static exciter-regulators and two control panels. The two 20,000 volt-ampere a-c generators are located in the forward engine compartments and are mounted one on each engine. The generators are driven by constant speed drive units. The total system and each individual generator system is capable of initiating and delivering power to the load bus independently of any outside power source. Either generator may be manually disconnected from the main a-c bus and reset. The pilot's controls for the system consists of one generator control switch for each generator. These switches energize and de-energize the generators in the event of a protective trip. The reset operation is trip free. Each static exciter-regulator includes the excitation power and control systems for one generator. The electrical system is so designed that no one fault will cause a complete loss of electrical power. The 115/200 volt a-c main and essential buses are powered directly by the generators through the line contactors. Under normal conditions, the main and essential buses are connected together and are in effect, one common bus. In the event that one generator fails, the remaining generator will supply all the a-c power. One phase of the generator output powers an auto-transformer for the 26 volt a-c main and essential loads. The generator output powers a similar auto-transformer for 26 volt a-c main loads only. Two 60-ampere transformer-rectifiers are used to convert three-phase, 200/115 volt a-c power to 28 volt d-c power to accommodate the airplane d-c power requirements.

D-C ELECTRICAL POWER

All d-c power is supplied by two 60-ampere transformer-rectifiers. The input to one transformer rectifier is connected to the main left a-c bus and the other is connected to the essential a-c bus so that a failure of one a-c bus will not constitute a complete loss of d-c power. The outputs of the transformer-rectifiers are connected in parallel through a current limiter so that in the event of a failure of one of the transformer-rectifiers, the remaining transformer-rectifier will supply all the d-c power for the system. No one fault will cause a complete loss of d-c power.

Generator Control Switches

Two generator control switches (figure 1-10), one for each generator, are located on the pilot's right vertical panel outboard of the caution lights. The three-position toggle switches are labeled ON EXT, OFF, ON (left generator) and ON EXT, OFF, ON (right generator). With external power plugged into the airplane, both pilot controlled generator switches must be placed in the EXT ON position. This connects external 115 /200 volt a-c to the aircraft's 115/200 volt a-c main bus. Placing the switches to the left generator and right generator ON positions will connect the generators to the airplane buses. If the generator control panel has disconnected a generator due to a temporary malfunction, which has been corrected, the generator can be reconnected to the bus system by placing the switch to OFF, then returning it to ON.

External Electrical Power Receptacle

To provide adequate power for ground operation of electrical equipment, an external power receptacle (figures 1-23 and 7-1), is located on the bottom of the left air duct. The external power required is three-phase, 400 cycle, 115/200 volt a-c and it is distributed through the entire electrical system in the same manner as generator output.

Circuit Breakers

Most of the d-c circuits are protected by circuit breakers located in the aft cockpit. The circuit breakers in essential circuits are located on a panel (figure 1-19) on the outboard side of the right console

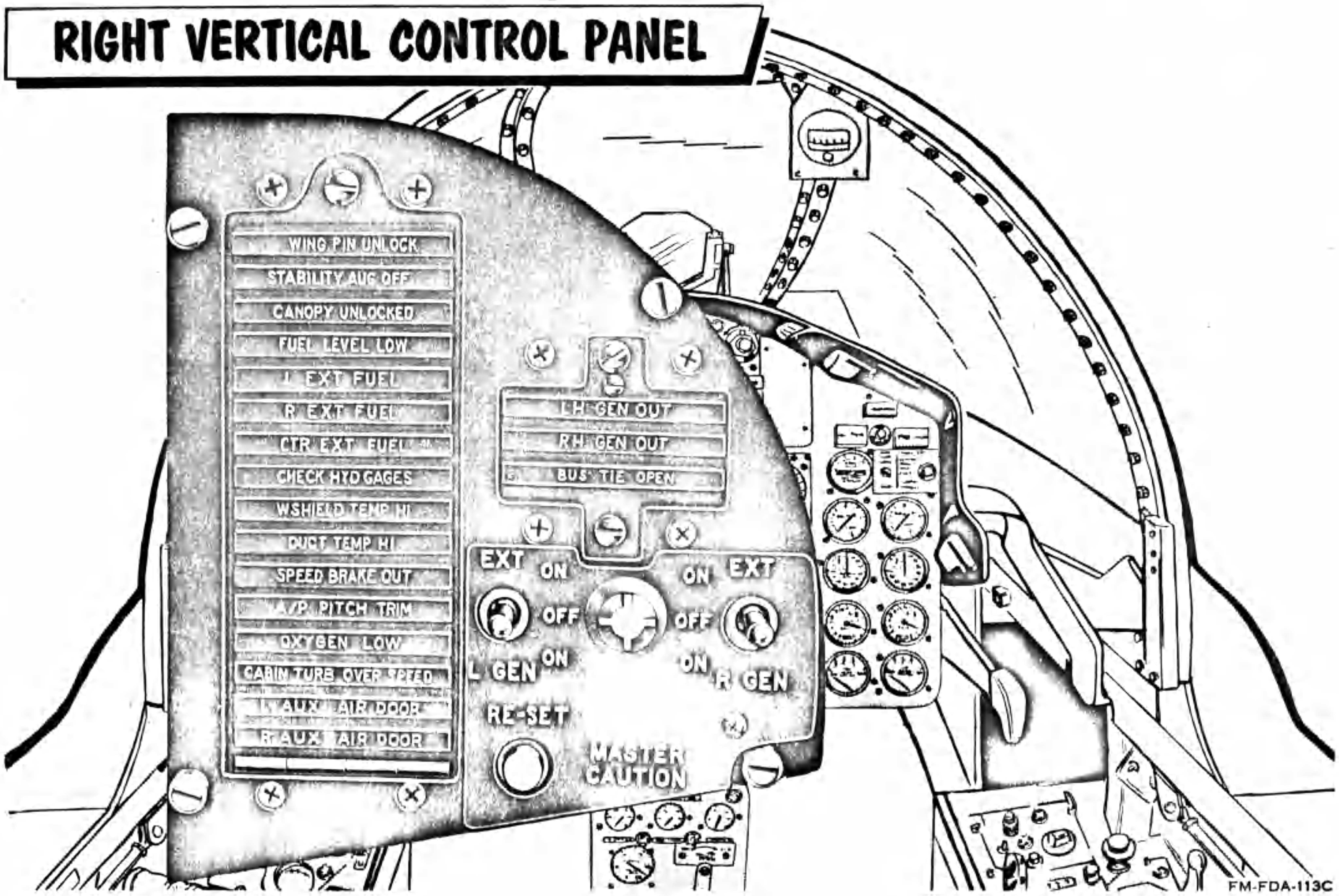

Figure 1-10

in the forward cockpit. The majority of the remaining circuit breakers are located on panels in the RO's cockpit.

Emergency Electrical System

In the event both engines fail resulting in complete loss of electrical power, a wind-driven turbine, which is extended into the airstream pneumatically, powers a hydraulic pump and a 3 KVA three-phase, 400 cycle emergency generator. When this generator comes up to speed and voltage, it operates an a-c relay which will connect the generator to the essential a-c and d-c buses. If either of the main generators is restored, the emergency generator automatically drops off the line and the wind-driven turbine can be retracted pneumatically. To extend wind-driven turbine, push DOWN on EMERG HYD PUMP handle (figure 1-18) located outboard of the left console. Pull UP on EMERG HYD PUMP handle to retract the wind-driven turbine.

Note

- Emergency wind-driven generator operational time shall be logged on the yellow sheet (Op NAV FORM 3760-2).

- A pressure switch in the emergency generator circuit will disconnect the emergency generator from the essential buses when the aircraft's airspeed drops below 195 knots. This will result in complete loss of electrical power but will allow all of the wind-driven turbine's power to be utilized in driving the emergency hydraulic pumps in order to maintain control of the airplane.

Generator Warning Lights

The L.H. and R.H. Gen Out warning lights and the Bus Tie Open warning light are located on a warning lights panel (figure 1-10) in the front cockpit. One of the generator warning lights will illuminate any time an applicable generator is not on the line when the generator switch is on. The pilot may attempt to reset the generator by placing generator switch to OFF and then placing it back to ON. No other corrective action can be taken unless both generators are off the line. In this case, all warning lights will be out until the emergency generator is brought into operation by extending the wind-driven turbine which will drive the emergency generator. To extend wind-driven turbine, push DOWN on the EMERG HYD PUMP handle (figure 1-18). With emergency generator operating, both a-c generator lights will be on, if the generator switches are on and the generators are turning. The "Bus Tie Open" warning light will illuminate when the generators are not in parallel. No corrective action is necessary since this does not affect the operation of the electrical system. The generator warning lights and the "Bus Tie Open" warning light receive their power from the essential 26 volt d-c warning lights bus.

HYDRAULIC POWER SUPPLY SYSTEM

Three independent 3000 psi closed center hydraulic systems are used to supply power to components of the airplane which require application of heavy force. The three systems comprise power control systems No. 1 and No. 2 and a utility system. The power control hydraulic systems supply power to the ailerons, spoilers and stabilator only. The utility system supplies power for the rest of the hydraulically operated systems.

POWER CONTROL HYDRAULIC SYSTEM NO. 1

Power control system No. 1 (figure 1-11) is powered by a 25 gpm (at 100% engine rpm) engine driven, variable delivery pump-mounted on the left engine. This system furnishes power to the dual aileron, spoiler and stabilator power cylinders. Fluid is supplied to the pump by power control system No. 1 reservoir that has a usable capacity of one gallon. It is an airless, pressure loaded, piston type reservoir that insures necessary positive pressure at the pump suction port regardless of altitude or flight attitude. In the event of a decrease in system pressure a 2000 psi accumulator is used to maintain pressure in the reservoir. Pressure from the pump flows through a relief valve which opens to return at pressures exceeding 3850 psi. An accumulator precharged to 1000 psi is utilized in the system to reduce pump pulsations and to supplement pump flow when system demands are in excess of pump output. A pressure transmitter for the cockpit No. 1 control hydraulic pressure gage is also located in the main pressure manifold. The system is cooled by an oil to fuel heat exchanger. The cooler is designed to maintain an acceptable bulk oil temperature during all phases of operation. Hydraulic power will be delivered to the power control system No. 1 by a wind-driven emergency hydraulic pump in the event of engine or pump failures.

POWER CONTROL HYDRAULIC SYSTEM NO. 2

Power control system No. 2 (figure 1-11) is powered by a 25 gpm (at 100% engine rpm) engine driven, variable delivery pump-mounted on the right engine. This system also furnishes power to the dual aileron, spoiler and stabilator power cylinders. The system reservoir, relief valve, accumulator, pressure transmitter and heat exchanger are identical and function the same with those in power control system No. 1. No wind-driven emergency hydraulic pump or an accumulator to pressurize the reservoir in the event of a decrease in system pressure are utilized in power control system No. 2.

Note

System No. 1 and No. 2 are independent of each other, but each system satisfactorily functions as an emergency system for the other.

UTILITY HYDRAULIC SYSTEM

The function of the utility hydraulic system (figure 1-12) is to supply fluid under pressure to the following subsystems:

- Lateral Control Servo
- Wheel Brakes
- Speed Brakes
- Landing Gear
- Wing Fold
- Flap (Leading and Trailing Edge)
- Variable Engine Air Duct Ramp
- Inflight Refueling
- Arresting Gear
- Rudder Power
- Engine Bellmouth
- Aileron and Rudder Flutter Dampers (fluid supplied from return lines)
- Engine Auxiliary Air Doors
- Air Compressor
- Rudder Feel and Rudder Actuators
- Fuel Transfer Pumps

A 2.5 gallon airless, pressure loaded, piston type reservoir supplies fluid to the utility system's engine driven variable delivery pumps. One pump is driven by the right engine and the other is driven by the left engine. The combined pumps deliver 30 gpm flow at 100% engine rpm. Pump pulsations are reduced by an accumulator which is precharged to 1000 psi. The accumulator also supplements flow when system demands are in excess of pump output. Hydraulic system pressure is transmitted electrically to a cockpit indicator by a system pressure transmitter located in the main pressure manifold. A relief valve limits system pressure to 3850 psi at full system flow. Hydraulic fluid radiators installed in the engine fuel feed lines remove heat from the hydraulic system by means of heat transfer from hydraulic fluid to fuel.

EMERGENCY HYDRAULIC POWER

An emergency hydraulic pump is located in the upper left side of the fuselage behind the aft canopy. The hydraulic pump is powered by a wind-driven turbine which is raised into the airstream pneumatically. Fluid under pressure flows from the pump through pressure regulators and check valves into the circuitry of power control system No. 1. In addition the wind-driven turbine drives an a-c generator. The wind-driven turbine extends up and out by pneumatic pressure and drives both the hydraulic pump and the a-c generator. Extending the turbine is accomplished by pushing DOWN on the EMERG HYD PUMP handle (figure 1-) which is located on the left console below the canopy sill immediately outboard of the fuel control panel in the pilot's cockpit. Pull UP on the EMERG HYD PUMP handle to retract the wind-driven turbine. Power control hydraulic systems No. 1 and No. 2 will also function as an emergency system for each other in the event one of the systems should fail.

Note

In the event there is a failure in one system, the other system will remain in operation and continue to supply power to the dual power cylinders. This will have no effect upon stick movement and control response but will limit full travel of the lateral control and stabilator systems at high airspeeds.

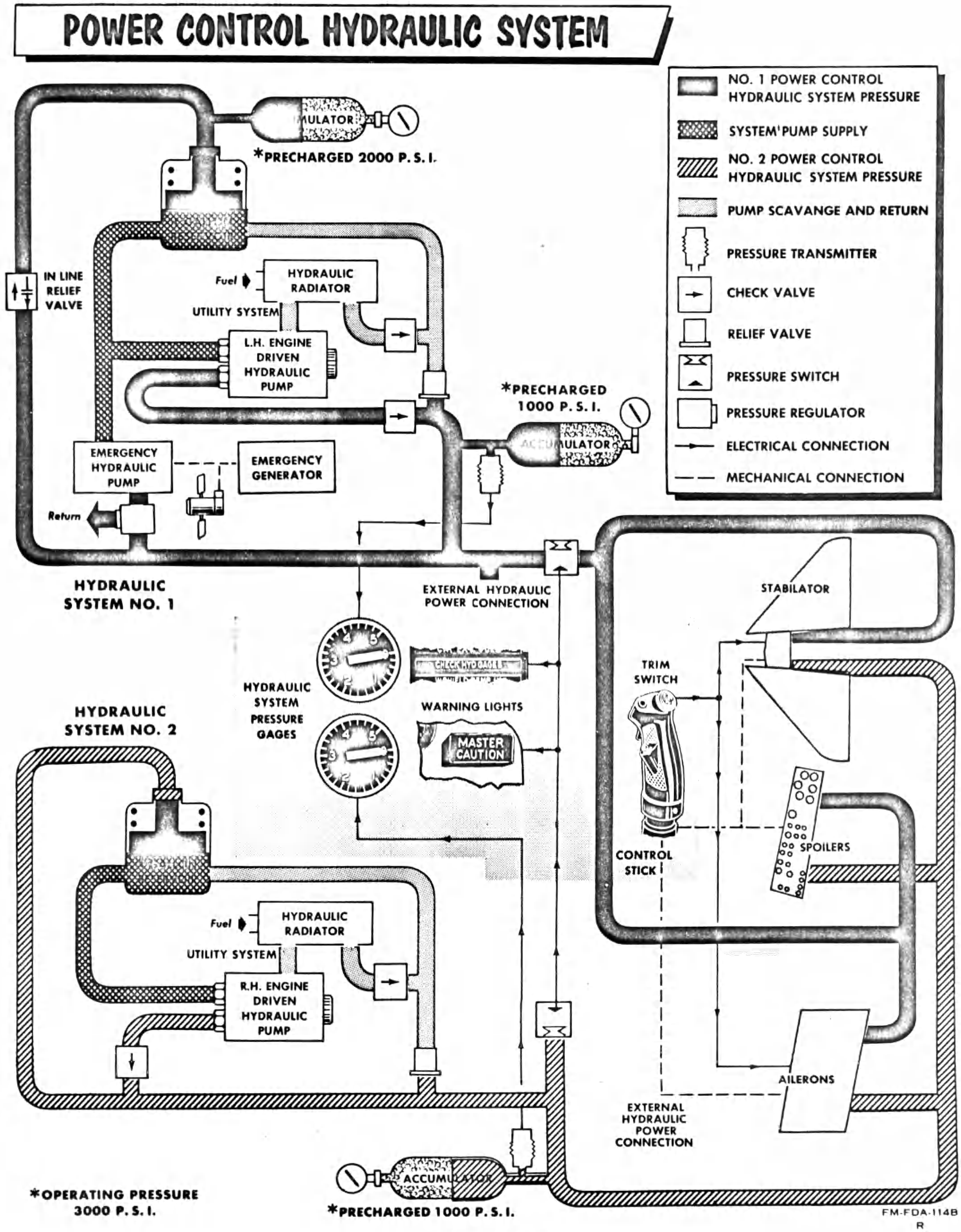
POWER CONTROL HYDRAULIC SYSTEM
NO. 1 POWER CONTROL HYDRAULIC SYSTEM PRESSURE
SYSTEM PUMP SUPPLY
NO. 2 POWER CONTROL HYDRAULIC SYSTEM PRESSURE
PUMP SCAVANGE AND RETURN
PRESSURE TRANSMITTER
CHECK VALVE
RELIEF VALVE
PRESSURE SWITCH
PRESSURE REGULATOR
ELECTRICAL CONNECTION
MECHANICAL CONNECTION
*PRECHARGED 2000 P.S.I.
IN LINE RELIEF VALVE
Fuel
HYDRAULIC RADIATOR
UTILITY SYSTEM
L.H. ENGINE DRIVEN HYDRAULIC PUMP
*PRECHARGED 1000 P.S.I.
ACCUMULATOR
EMERGENCY HYDRAULIC PUMP
EMERGENCY GENERATOR
Return
HYDRAULIC SYSTEM NO. 1
EXTERNAL HYDRAULIC POWER CONNECTION
STABILATOR
TRIM SWITCH
CHECK HYD GAGES
WARNING LIGHTS
MASTER CAUTION
HYDRAULIC SYSTEM PRESSURE GAGES
HYDRAULIC SYSTEM NO. 2
CONTROL STICK
SPOILERS
HYDRAULIC RADIATOR
R.H. ENGINE DRIVEN HYDRAULIC PUMP
AILERONS
EXTERNAL HYDRAULIC POWER CONNECTION
ACCUMULATOR
*OPERATING PRESSURE 3000 P.S.I.
*PRECHARGED 1000 P.S.I.
FM-FDA-114B R

Figure 1-11

UTILITY HYDRAULIC SYSTEM

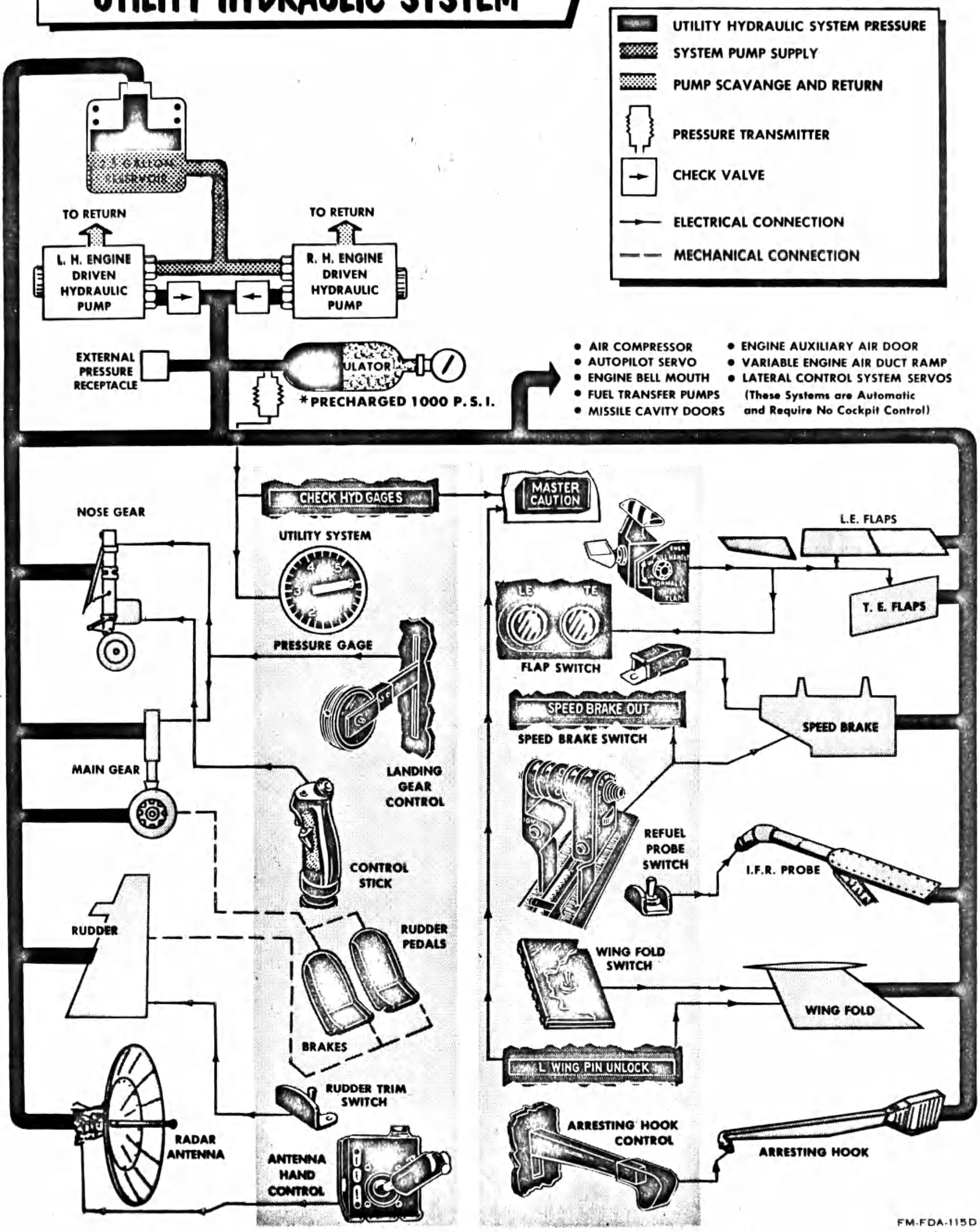

Figure 1-12

HYDRAULIC PRESSURE INDICATORS

The hydraulic pressure indicators consist of three individual indicators (figure 1-17) and three pressure transmitters. The indicators are mounted on the pedestal panel and are marked No. 1 control, No. 2 control and utility. The three transmitters are located in the two engine accessory compartments. The three indicators are identical having a range of 0 to 5000 pounds with the calibration being 0 to 5 and the reading being multiplied by 1000. The three systems normally will have a pressure range of 2750 to 3250 pounds while the airplane is on the ground and hydraulically operated equipment is not functioning. Electrical power for control system No. 2 indicator and the utility indicator is supplied by the left main 26 volt a-c bus. Control system No. 1 indicator receives power from the essential 26 volt a-c bus.

HYDRAULIC SYSTEMS WARNING LIGHT

An amber "Check Hyd Gages" warning light on the warning lights panel (figure 1-10) will illuminate when pressure in the system is decreased to 1500 ± 50 psi. The single light serves both power control systems No. 1 and No. 2. The purpose of the light is to warn the pilot that pressure in one of the systems (power control No. 1 or No. 2) has decreased for some reason and he should check the hydraulic system pressure gages for further information. The warning light will be extinguished when the increasing pressure reaches 1750 psi. Effective airplanes 146817c thru 148275e, the "Check Hyd Gages" warning light also monitors the two hydraulic pumps in the utility hydraulic system. Power for the warning light circuit is supplied by the essential 26 volt a-c bus.

PNEUMATIC POWER SUPPLY SYSTEM

The function of the 3000 psi pneumatic system (figure 1-13) is to provide high pressure air for operation of the following eight normal and emergency subsystems:

EMERGENCY SYSTEMS	NORMAL SYSTEMS
Canopy	Canopy
Landing Gear Extension	Nose Gear Strut Extension
Wheel Brakes	Wind-Driven Turbine Extension
Flaps	
Flooding Doors	

Air supplied from the 17th stage of the engine compressors is used for maintenance of a constant inlet pressure at any altitude and is compressed to 3000 psi by the hydraulically operated air compressor and maintains the air bottles at 3000 psi. A solenoid-operated shutoff valve in the hydraulic line to the compressor motor is controlled by a pneumatic pressure switch cutting hydraulic pressure to the compressor when air pressure is 3000 ± 100 psi and opening the hydraulic line when air pressure drops to 2700 ± 100 psi. The air compressor discharges into a manifold which contains the air drying equipment, ground charging valve, relief valve, ground charging pressure gage and a pressure transmitter for the cockpit gage. The manifold then branches to individual air bottles with check valves installed in each of the pneumatic operated systems. The air bottles are installed in the component systems to preclude loss of the system due to engine or pneumatic pump failure. The branch systems are isolated from return flow to the manifold by the check valves. Reverse flow into the compressor is prevented during ground charging by a check valve in the manifold between the compressor and ground charging valve.

PNEUMATIC PRESSURE INDICATOR

The pneumatic pressure indicator (figure 1-17) is mounted on the pedestal panel and operates in conjunction with the pneumatic pressure transmitter which is located in the engine accessories compartment. The indicator has a range of 0 to 5000 pounds with calibrations of 0 to 50 and readings multiplied by 100. The normal system pressure reading on the cockpit indicators ranges from 2650-3300 psi.

FLIGHT CONTROL SYSTEM

The airplane flight controls (figure 1-11) consist of the stabilator, ailerons, spoilers and rudder. All control surfaces except the rudder are actuated by irreversible hydraulic dual power control cylinders. The rudder is actuated by a conventional irreversible power cylinder. Artificial feel systems provide the pilot with simulated stick and pedal forces due to the lack of aerodynamic "feedback" forces from the power cylinders. Viscous dampers are used in the aileron, rudder and stabilator systems. All feel systems have trim actuators which are controlled by the pilot. Cockpit indicators on the left vertical panel in the pilot's cockpit show stabilator trim, aileron and rudder positions. Secondary controls are leading edge flaps, trailing edge flaps and wing mounted speed brakes.

Note

Airplanes 148262e thru 148275e can be converted to dual control airplanes. Refer to Dual Control Configuration, Section IV.

LATERAL CONTROL SYSTEM

The lateral control system which utilizes power control systems No. 1 and No. 2 is a unique aileron-spoiler combination with both surfaces operating in all flight ranges. The use of spoilers for lateral control partially alleviates torsional loading of the wing during high speed rolling maneuvers, and the aileron-spoiler combination increases the roll rate. The ailerons travel downward only, from the full trail position to 30° down. Upward travel is limited to only 1° from full trail. Travel of the spoilers is 45° upward from a flush contour position on the upper wing surface. The control linkage is such that if the aileron valve is actuated in one wing, the spoiler valve is actuated in the

PNEUMATIC SYSTEM

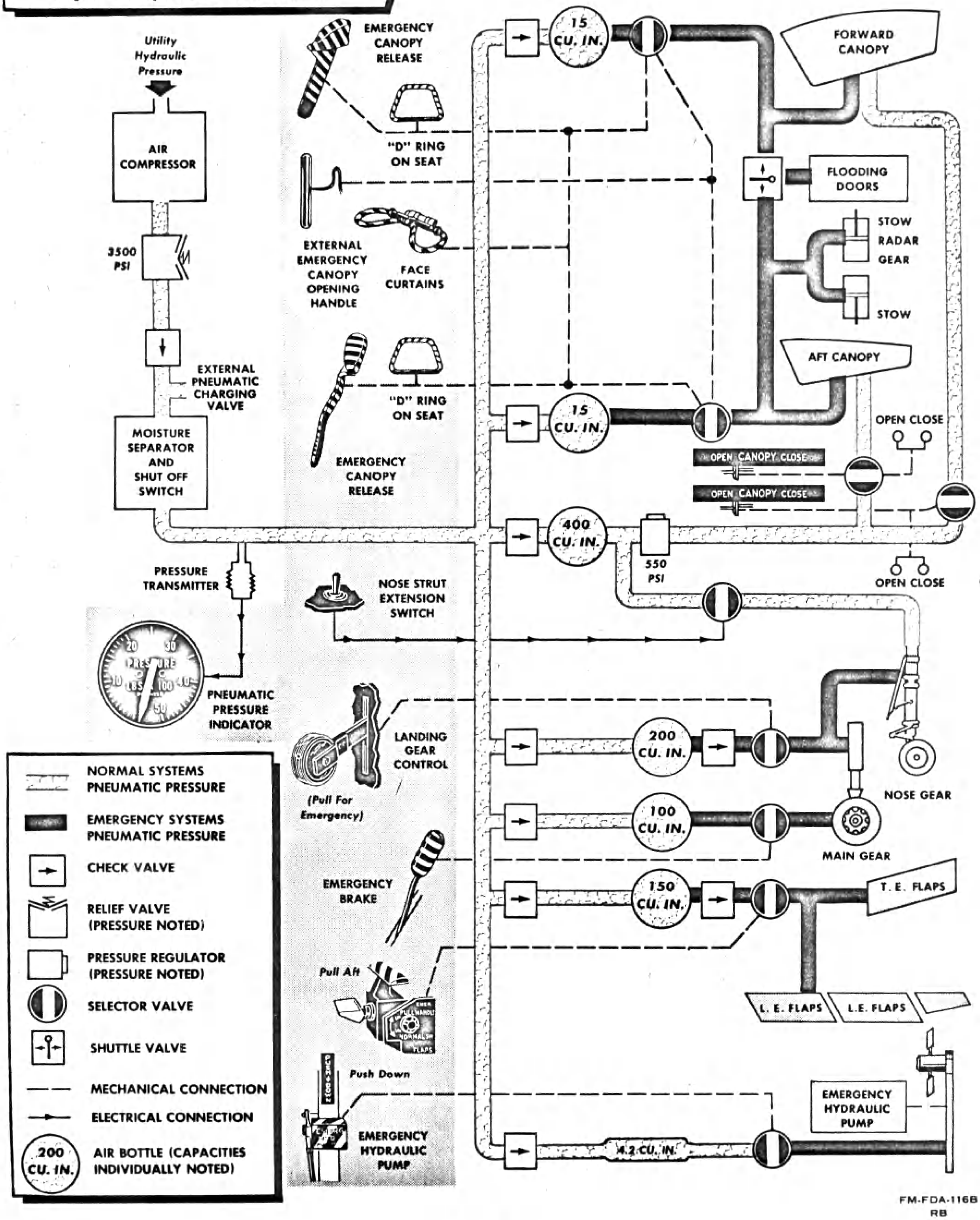

Figure 1-13

other. The lateral control system components consist of the stick, push pull rods, aileron dual power control cylinders, spoiler dual servo valves, spoiler tandem cylinders, dampers, feel trim cartridge and autopilot servos. The push rod system includes a spring cartridge in each wing which acts as a push rod during normal operation. If any jamming occurs in the linkage, servo valves or cylinders that the cartridge normally actuates, the spring cartridge will deflect under pilot effort allowing operation of the other lateral system. A self-serviced hydraulic damper attached to the aileron back-up structure is designed to be used as an upstop for the aileron as well as a flutter damper. The use of dual hydraulic cylinders allows simultaneous use of both power control hydraulic systems previously mentioned. In the event of a single failure, the other system will supply sufficient power for control.

Aileron Control

The aileron dual power control cylinders utilize pressure from both No. 1 and No. 2 basic power control systems. Each cylinder has four pistons tied together by a yoke at the rod ends. An integral dual tandem servo valve directs fluid as required to each set of pistons. The two outer pistons are supplied hydraulic pressure by one of the power control systems while the two inner pistons are supplied by the other power control system. This arrangement provides symmetrical loading on the yoke with failure of one system. Valves and pistons supplied by the No. 1 power control system are hydraulically independent of the valves and pistons supplied by the No. 2 power control system.

Aileron-Rudder Interconnect

An aileron-rudder interconnect (ARI) system is incorporated on the airplane to automatically coordinate turns made at airspeeds below 225 knots. This is done by providing rudder displacement as a function of aileron displacement. The ARI system operates in conjunction with the automatic flight control system to provide full 15° rudder authority. Components of the ARI system include the ARI control amplifier, a 10° servo actuator, an airspeed pressure switch, and the aileron transducers. The yaw servo amplifier and the force transducer of the autopilot system are also used. The rudder integrated actuator is utilized for part of the rudder movement. For autopilot information, refer to Section IV.

Spoiler Control

The spoiler dual servo valves direct fluid to the two pistons of each spoiler cylinder, one section directing fluid from the No. 1 power control system to one of the pistons and the other section directing fluid from the No. 2 power control system to the other piston. The spoiler tandem cylinders are dual cylinders utilizing pressure from both the No. 1 and the No. 2 power control systems. Each cylinder has two pistons on a common rod. Pistons in each cylinder are hydraulically independent of each other. The dual system hydraulic cylinders provide spoiler power in case of failure of one of the two hydraulic systems.

Lateral Control Feel and Trim

An artificial feel trim cartridge joined in tandem with a screw jack is attached to the bell crank prior to where spoiler linkage and aileron linkage are separated from the common pilot input effort. The two feel trim cartridges are synchronized by flex shafting and powered by an electric actuator. Forty percent trim authority is designed into the system. A hydraulic servo in each wing, attached in series with the linkage through a walking beam, provides autopilot operation of the lateral system. Refer to Autopilot, Section IV.

STABILATOR SYSTEM

The stabilator system consists of the control stick, push-pull rods, power cylinder valve and an integrated power control cylinder. The stabilator is a single unit horizontal tail which is actuated by an irreversible power control cylinder. Movement of the control stick in the cockpit is translated into stabilator deflection by means of a combination push-pull rod and cable system to the power cylinder. Hydraulic pressure to the actuator is supplied by power control systems No. 1 and No. 2 (figure 1-11). Two hydraulic autopilot servos and a hydraulic damper are integrated into the power cylinder valve. Autopilot operation of the stabilator will move the control stick fore and aft. Refer to Autopilot, Section IV. A "bob" weight is included in the longitudinal control system linkage to increase stick forces as "g" forces increase.

Stabilator Feel and Trim

Artificial feel is provided by a dynamic pressure bellows. The force from this bellows with a variable moment arm is balanced by a spring of nearly constant force on a variable moment arm. As the velocity of the aircraft increases, the bellows force increases necessitating a change in trim which decreases the bellows force moment arm and increases the spring moment arm. The relationship of these two arms is changed with respect to a point of rotation by an electric trim actuator which is operated by a trim button on the control stick. A viscous damper in the bellows linkage increases stick forces with rapid motion of the stick. This feature tends to prevent exceeding structural limitations inadvertently. A spring cartridge is incorporated in the control linkage which enables the pilot to override the feel system in the event of a malfunction. A bellows diaphragm failure in the stabilator feel system would be indicated to the pilot by a mild pitch down tendency which he would be able to correct.

Stabilator Trim Indicator

The stabilator trim indicator (figure 1-16) is located on the left vertical panel in the pilot's cockpit. It is directly controlled by a transmitter which is integral with the stabilator feel trim actuator. The left 28 volt d-c bus supplies power for this indicator.

RUDDER CONTROL SYSTEM

The utility hydraulic system (figure 1-12) is used to supply hydraulic pressure to the rudder power control and feel trim cylinders. Movement of adjustable position rudder pedals in the cockpit is translated into rudder deflection by means of a combination push-pull rod and cable system to the power cylinder. The power cylinder is of the irreversible type with integral servo valve. Artificial feel is supplied to the pedals through linkage from an artificial feel trim system. A double acting viscous linear damper and a rotary damper are coupled directly to the rudder to dampen rudder flutter. It is possible for the pilot to have limited authority over the rudder in case of failure of the hydraulic system which supplies the rudder power cylinder and rudder feel system. This is done by the internal pressure operated bypass valve in the power control cylinder which opens with loss of system pressure and allows fluid to pass from one side of the piston to the other. Thus, the cylinder acts as another "link" in the rudder system and the rudder can be moved by pilot effort. Total amount of rudder available would be dependent upon air loads on the rudder. A hydraulic servo for yaw damping and autopilot operation is integrated into the rudder power cylinder. Operation of the autopilot servo does not move the rudder pedals. For autopilot information, refer to Section IV.

Rudder Feel and Trim

Artificial feel is supplied to the pedals through the linkage from an artificial feel trim system attached in parallel to the linkage in the aft fuselage. A hydraulic cylinder with system pressure to both sides of the differential area piston provides a pedal force of four pounds per degree of rudder deflection below 235 ± 10 knots during acceleration and below 220 ± 5 knots during deceleration. Above 240 knots, a pressure switch in the pitot static system cuts off hydraulic system pressure to the low area side of the hydraulic cylinder piston and the pedal force becomes 18 pounds per degree of rudder deflection. Use of a trim switch on the console, in conjunction with an electric actuator in the feel trim linkage, removes loads from the pedals after the pilot has positioned the rudder to a desired flight attitude. Normal trim range is ± 5 degrees of rudder deflection.

Rudder Trim Switch

The rudder trim switch (figure 1-18) is located in the forward cockpit on the left console forward of the engine start buttons. This switch controls the trim actuator in the rudder feel system which trims the airplane directionally.

Rudder Position Indicator

The rudder position indicator (figure 1-16) is located on the left vertical panel in the pilot's cockpit. The indicator is controlled by a transmitter which is mechanically coupled to the rudder. This indicator operates from the essential 26 volt a-c bus.

RUDDER PEDALS

The rudder pedals are conventional type suspended units which are coupled to the rudder push-pull rods system by individual screw jacks. The screw jacks provide adjustment of the rudder pedals for pilot comfort and are adjusted simultaneously by turning a crank on the pedestal. The pedals are also coupled to the power brake valves so that toe pressure on the pedal will apply the brakes. Effective airplanes 148252d thru 148275e, the rudder pedals are used to control the nose wheel steering unit when the nose steer button on the cockpit stick is depressed.

STALL WARNING VIBRATOR

A stall warning vibrator is mounted on the left rudder pedal to warn the pilot of approaching stall conditions. The vibrator consists of an electrical motor which drives an eccentric weight. Rotation of the weight causes the rudder pedals to vibrate, warning the pilot of an impending stall. A sufficient margin exists to allow the pilot to return to the proper flight attitude by normal reaction to the warning. The vibrator is electrically connected to a switch in the angle-of-attack indicator. The switch to activate the rudder pedal vibrator is set at 20.5 units on airplanes 145307b thru 145312b and 145314b thru 145317b. Airplane 145313b and airplanes 146817c thru 148275e have a switch setting of 22.3 units.

CONTROL STICK

The control stick is mounted in a yoke which permits both left and right and fore and aft motion. The control grip is a quick removable assembly containing four switches, a trigger switch for missiles, a bomb and centerline stores button located in the upper left side of the grip and a nose gear steering button located on the left side and halfway down the grip. A trim switch which operates both the aileron feel trim and the stabilator feel trim actuators is located at the top of the grip.

WING FLAPS

Two position leading edge flaps and three-position trailing edge flaps that utilize basic utility hydraulic system pressure are incorporated on the airplane. The leading edge flaps are mounted on the inboard, center and outer wing panels. Trailing edge flaps are mounted on the inboard portion of the wing adjacent to the fuselage. Each flap has its own hydraulic actuator. The leading edge flaps are locked in the retracted position by overcenter linkages. Trailing edge flaps are locked in the retracted position by internal locks in the cylinders. A check valve is provided as an integral part of the selector valve to prevent unlocking of overcenter mechanisms and internal locks by back pressure in the return lines. A flow divider is provided to synchronize the trailing edge flaps. There is no synchronization between leading edge flaps or between leading and trailing edge flaps. The flap actuators are connected with hydraulic lines to a common manifold solenoid

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WING FLAPS

Two position leading edge flaps and three-position trailing edge flaps that utilize basic utility hydraulic system pressure are incorporated on the airplane. The leading edge flaps are mounted on the inboard, center and outer wing panels. Trailing edge flaps are mounted on the inboard portion of the wing adjacent to the fuselage. Each flap has its own hydraulic actuator. The leading edge flaps are locked in the retracted position by overcenter linkages. Trailing edge flaps are locked in the retracted position by internal locks in the cylinders. A check valve is provided as an integral part of the selector valve to prevent unlocking of overcenter mechanisms and internal locks by back pressure in the return lines. A flow divider is provided to synchronize the trailing edge flaps. There is no synchronization between leading edge flaps or between leading and trailing edge flaps. The flap actuators are connected with hydraulic lines to a common manifold solenoid

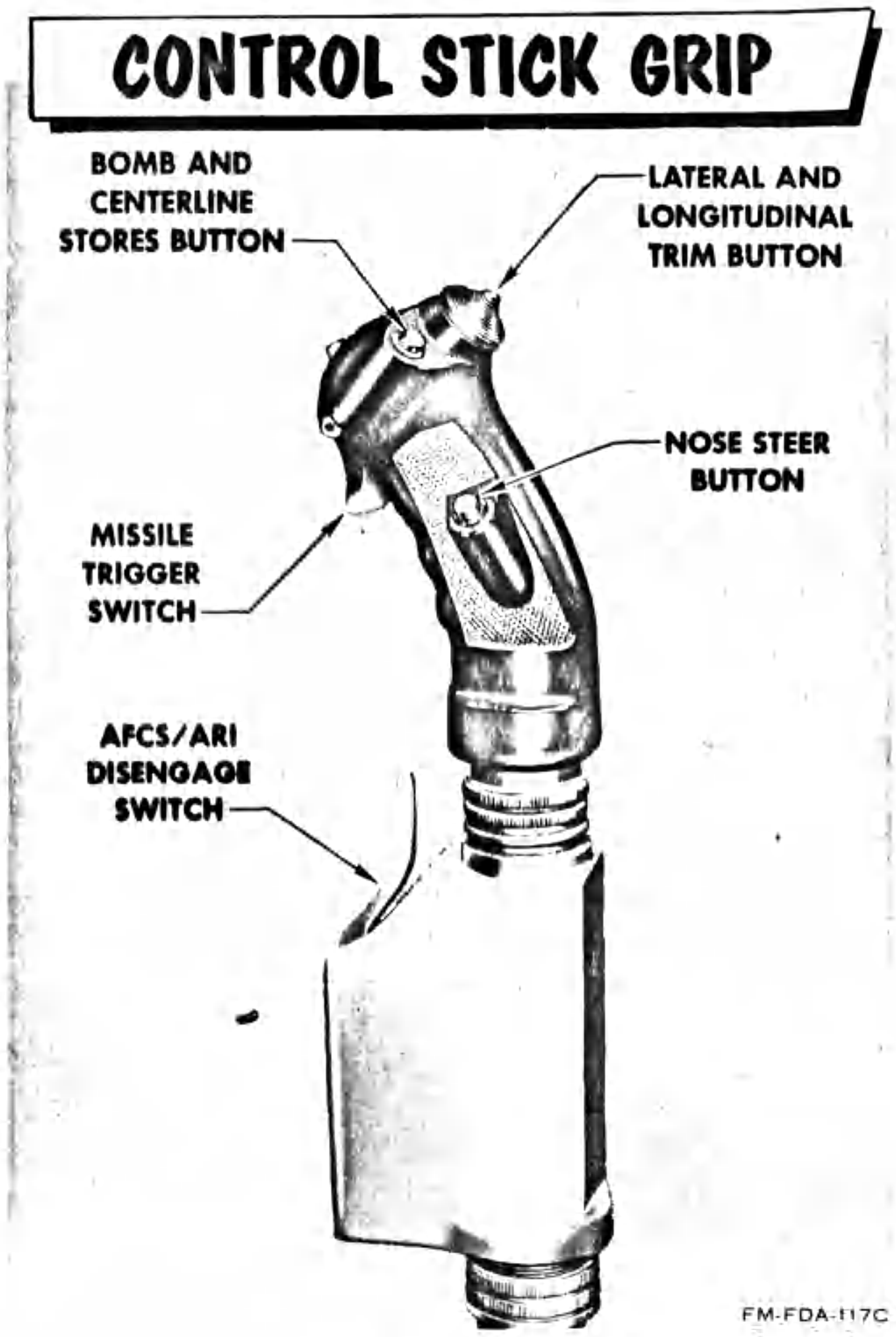

Figure 1-14

selector valve. The selector valve is operated by an electrical switch in the cockpit. When the switch is pushed down it simultaneously actuates the leading edge flaps and trailing edge flaps. Should the cockpit switch inadvertently be left in the down position, the leading edge flaps and trailing edge flaps will retract when the airplane reaches an airspeed of 200 ± 5 knots on airplanes 145307b thru 145312b and 230 ± 5 knots on all other airplanes. This is accomplished through an airspeed pressure switch which operates the common manifold solenoid selector valve. Should the cockpit switch continue to remain in the down position, the flaps will extend when the airplane reaches an airspeed of 15 ± 5 knots less than the speed at which they retracted. Flap extension and retraction will be accomplished within six seconds.

WING FLAP SWITCH

The leading and trailing edge flap switch (figure 1-15) is located on the wing flap control panel which is mounted above the left console outboard of the throttles. The three-position toggle switch is marked UP, 1/2, and DN and is shaped like an airfoil for ease of identification. Selecting the 1/2 position will move the center and outboard leading edge flaps to the full down position (60°); it will move the inboard leading edge flaps to the full down position which is 30°; and it will move the trailing edge flaps 1/2 (30°) down. Selection of the DN position will move the trailing edge flaps to the fully extended position (60°). Selecting the 1/2 position after the flaps have been fully extended will raise the trailing edge flaps to the 1/2 (30°) position. Placing the flap switch in the UP position will simultaneously return all the flaps to the fully retracted position. There is no individual selecting of flaps. Power is supplied to the switch from the right main 28 volt d-c bus.

Emergency Flap Extension

Emergency extension of the flaps is accomplished pneumatically by high pressure air from emergency air storage bottles. The emergency flap extension handle is mounted on the wing flap control panel (figure 1-15) and is marked EMERG. Lowering the flaps pneumatically is accomplished by pulling the emergency handle AFT. Actuation of the emergency flap extension system will extend all flaps to the full down position. The handle is airfoil shaped and painted in black and yellow stripes for ease of identification. The air bottles contain air sufficient for only one extension of the flaps.

Flap Position Indicator

The leading edge and trailing edge flap indicators (figure 1-16) are located on the left vertical panel in the pilot's cockpit. The indicators work in conjunction with position switches on the leading and trailing edge

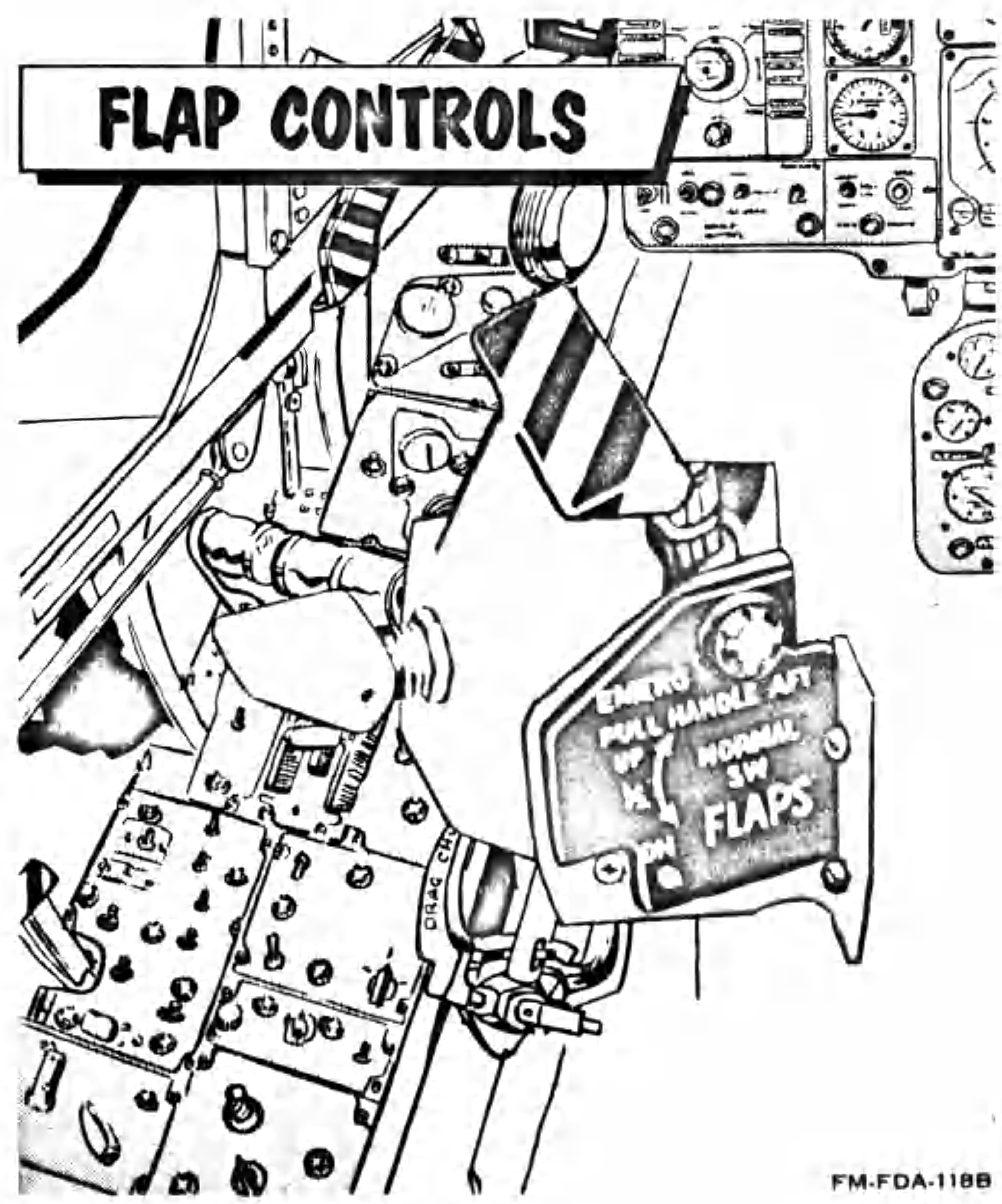

Figure 1-15

flaps. The position of the flaps is indicated by drum dials viewed through cutouts in the instrument panel. With flaps up, the word UP will appear on the indicators; flaps in transient will be indicated by a barber pole; half flaps will be indicated by the fraction 1/2 appearing on the drum dial for the trailing edge flaps only; with flaps down, the letters DWN will appear on the indicators. The indicator circuit draws its power from the essential 28 volt d-c bus.

Landing Gear Handle Warning Light

The landing gear warning light is located in the landing gear control handle (figure 1-18). The light will flash when the flaps are down and the gear handle is in the gear up position. The same light, which operates on essential 26 volt a-c power, also serves as a landing gear indicator. Refer to Landing Gear, this section. In airplanes 145313b and 146817c thru 148275e, an additional light, marked "wheels" is located on the upper left corner of the main instrument panel. This light will flash whenever the flaps are down and the gear is up, and replaces the flashing light that was in the landing gear handle. A light that indicates landing gear in transit is still located in the landing gear handle.

BOUNDARY LAYER CONTROL SYSTEM

The boundary layer control system utilizes air bled from the 17th stage of the engine compressor. This air passes through ducts attached to the rigid part of the wing between leading edge flaps and the spar and between the trailing edge flap and the flap closure beam. Slots along the ducts behind the outboard and center panel leading edge flaps and in front of the trailing edge flaps direct laminar air over the wing and flaps when the flaps have deflected sufficiently to expose the slots. The high temperature and high velocity laminar air directed over the wings and flaps will delay flow separation over the airfoil, hence reducing turbulence and drag. This results in a lower stall speed and therefore a reduction of landing speed. Leading edge BLC is operative in the 1/2 or full flap position. Trailing edge BLC is operative only when the flaps are in the full down position.

BLC MALFUNCTION WARNING LIGHT

A "BLC Malfunction" warning light (figure 1-10) is located on the right console just below the caution lights panel. The purpose of the warning light is to indicate a BLC valve malfunction in the flaps up condition. When any one of the four BLC valves is not fully closed, and the flaps are up, the "BLC Malfunction" warning light will illuminate. It must be remembered that the illumination of the "BLC Malfunction" warning light only indicates that a BLC valve has failed to close when the flaps are up. No indication is provided for a completely inoperative system, nor is there an indication provided for a BLC valve failing to open when the flaps are down.

SPEED BRAKES

The hydraulically operated speed brakes are mounted on the underside of the inboard wing panels and are hinged on the forward side permitting the brakes to open downward. The speed brakes are controlled from a switch on the throttle grip and may be positioned at any point in their travel. Due to the construction of the selector valve, the speed brakes will not close following a hydraulic pressure failure unless the emergency speed brake switch located on the left console is placed in the RETRACT position. This de-energizes a solenoid bypass valve to block the speed brake open and speed brake close lines from the selector valve and connects both sides of the cylinders to return. Air loads will then close the speed brakes to the trail position. The basic utility hydraulic system is used to operate the speed brakes.

SPEED BRAKE SWITCH

The speed brake switch (figure 1-3) on the throttle grip has three positions: IN, STOP and OUT. The STOP position is the normal position of the switch. Only the OUT position of the switch is momentary. Placing the switch in the OUT position will operate the speed brakes toward the extend position. When the switch is released, it will return to the STOP position. The STOP position de-energizes the selector valve and blocks all ports giving a hydraulic lock for holding the speed brakes in any desired position. Selecting the IN position of the switch will close the speed brakes flush with the wing. The speed brakes will take four seconds to fully open and four seconds to fully close. The switch receives power from the right main 28 volt d-c bus.

Note

The STOP position of the speed brake switch may not hold the speed brakes completely closed. This will be noted by the illumination of the "Speed Brake Out" light. If this occurs, position the speed brake switch to IN and leave in that position.

OVERRIDE SAFETY SWITCH

A deck operated, two-position, safety switch marked SAFE and NORMAL is located in the right wheel well under a panel marked SWITCHES, Ground Fueling Check, and Speed Brake Disabling. The switch is provided for overriding the cockpit switch that controls the selector valve. The SAFE position of the switch extends the speed brakes and prevents energizing the selector valve from the cockpit. This switch is provided for protection of personnel working in the speed brake well. The NORMAL position of the override switch permits normal speed brake operation.

Note

Speed brake safety switch should be checked for NORMAL position prior to flight. If the switch is in the SAFE position, speed brakes will be inoperative.

SPEED BRAKE OUT LIGHT

An amber "Speed Brake Out" indicator light (figure 1-10) located on the caution lights panel will illuminate when either or both of the speed brakes are not fully closed.

Note

Speed Brake Out light does not light the Master Caution light.

LANDING GEAR SYSTEM

The airplane is equipped with fully retractable tricycle landing gear which are completely covered by flush doors when retracted. The gear is electrically controlled by the right main 28 volt d-c bus and hydraulically actuated by the basic utility system. The airplane is not equipped with a tail skid. Accidental retraction of the landing gear when the airplane is on the ground is prevented by safety switches on the main gear torque scissors, and ground safety locks.

MAIN GEAR

Each main gear is hydraulically retracted and extended. As the main gear retracts, the struts are mechanically compressed. They automatically return to the normally extended position when the gear is extended. The gear is locked down by an internal finger type latch in the side brace actuator. The main gear retracts inboard and is enclosed by fairing doors that are flush with the underside of the wing. The gear is locked up by a hydraulically actuated mechanism. All main gear doors remain open when the gear is extended.

NOSE GEAR

The nose gear is hydraulically retracted and extended. The gear is locked in the down position by an integral down lock mechanism within the gear actuating cylinder. A spring-loaded overcenter mechanism is used to lock the gear in the UP position. The gear is locked down by an internal finger type latch in the side brace actuator. The nose gear retracts aft into the fuselage and is covered by mechanically operated doors that close flush with the underside of the fuselage. The forward door is attached to the nose gear strut, and closes with retraction; the aft door is operated and latched closed by the gear uplatch mechanism. The nose gear is equipped with dual nose wheels, a shimmy damper and a self-centering mechanism. The nose gear is steered by differential braking of the main gear wheels.

LANDING GEAR CONTROL HANDLE

Operation of the landing gear is controlled by a handle (figure 1-18) at the left side of the main instrument panel. The handle has a wheel shaped knob for ease of identification. Placing the handle in the UP or DOWN position energizes a solenoid valve to connect system pressure to the landing gear. Placing the handle in the gear UP position will energize switches in the fuel tank vent and pressurization, jettison and armament circuits. A red warning light is located in the landing gear control handle knob. This light comes on whenever the control handle is moved to retract or extend the gear and it will remain on until the gear completes its cycle.

Emergency Landing Gear Control

A 200 cu. in. air bottle provides sufficient compressed air to extend the gear pneumatically in the event of a hydraulic system failure. Pulling the landing gear control handle full AFT operates an air valve which directs 3000 psi compressed air to open all gear doors, release the uplocks and extend all gear.

LANDING GEAR POSITION INDICATORS

The landing gear position indicators (figure 1-16) are located on the left vertical panel in the front cockpit. The indicators operate in conjunction with position switches on the landing gear. The position of the landing gear wheels is indicated by drum dials viewed through cutouts in the instrument panel. With gear up, the word UP will appear on the three indicators; gear in transient will be indicated by a barber pole; and with gear down, a picture of a wheel will be seen through the cutouts. The indicator circuit receives its power from the essential 28 volt d-c bus.

NOSE STRUT CATAPULT EXTENSION

A switch in the left main gear wheel well operates a solenoid valve which controls the inflation of the catapult extension chamber of the strut. Compressed air from the basic pneumatic system is supplied to a cavity above a floating piston in the nose gear shock strut upper cylinder. This air forces the piston down causing the strut to fully extend and giving the desired airplane attitude for catapulting. Retraction of the gear will cause the catapult extension chamber in the strut to be deflated, and the strut returns to the normal configuration. The catapult extension chamber may also be deflated by momentarily placing the cockpit generator switches to the OFF position.

WARNING

Airplane pneumatic system must be fully charged (2750 psi minimum), before extending nose gear. Insufficient pneumatic pressure may allow the strut to bottom out causing damage to the strut or fuselage structure. It also results in an improper airplane attitude for catapulting.

LEFT VERTICAL PANEL

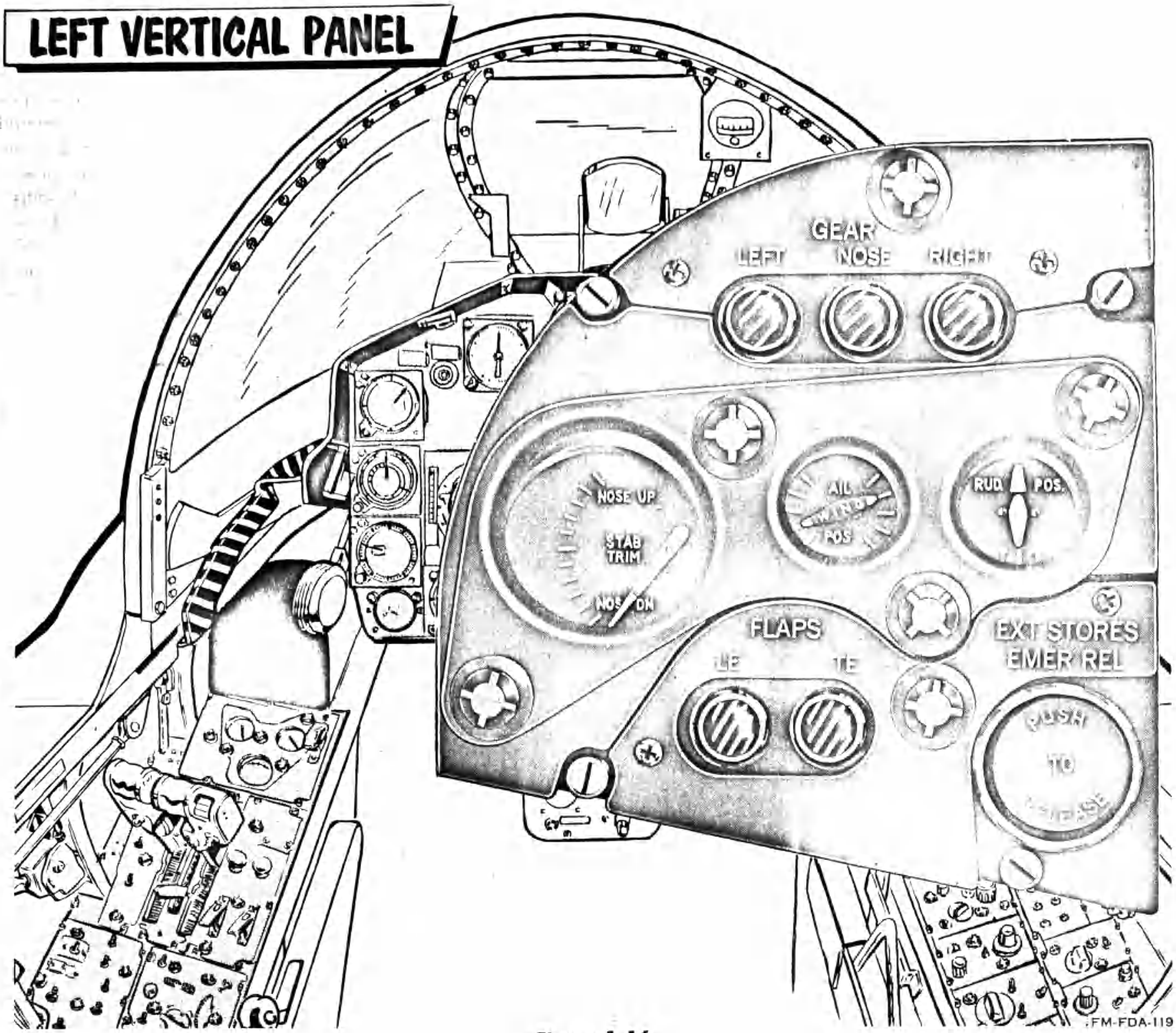

Figure 1-16

NOSE WHEEL STEERING

Effective airplanes 148252d thru 148275e, and all other airplanes upon the completion of the F4H Drive-In Modernization Program, an electrically controlled, hydraulically operated actuated nose wheel steering system has been installed in the aircraft. The steering actuator is located on the nose wheel strut and geared to the strut torque collar. It performs the work of both steering and damping. A bypass valve in the steer-damper manifold directs hydraulic fluid to a vane type rotary actuator which can be pressurized on either side as directed. For the damping mode, the bypass valve traps the fluid in the rotary actuator and channels it through damping orifices which absorb energy. When the stick grip nose steering button (figure 1-14) is held down, the main gear strut is compressed and the nose gear is down and locked, the system is energized and steering is affected by rudder pedal movement. The limit of the nose wheel steering system is 70° on each side of center.

WING FOLD SYSTEM

Each outer wing panel is folded upward and inboard by a conventional hydraulic actuator that receives hydraulic pressure from the utility hydraulic system, via the landing gear down lines. A hand operated locking system is installed in the airplane to lock wing pins in hinge fittings when wings are spread. A flat flush mounted control lever (figure 1-19) located on the right console in the pilot's cockpit is connected by push rods and push-pull cables to a pin locking device in the wing fold area. Pulling UP on the lever unlocks wing pins, extends warning flags on the upper wing surfaces, illuminates an amber L. Wing Pin Unlock and R. Wing Pin Unlock warning lights in both cockpits and energizes the wing fold and spread switch. Wing fold is actually accomplished by a two-position toggle switch that is located underneath the wing pin release lever and is exposed when the lever is raised. The switch is marked FOLD and SPREAD. As an added safety precaution, the wing fold hydraulic circuit receives its hydraulic pressure from the landing gear down pressure line; this will prevent pressurizing the wing fold circuit when the landing gear is UP. The wing fold and spread switch receives power from the left main 28 volt d-c bus. The warning lights are operated off of the essential 26 volt a-c bus. On some airplanes the wing pins are released by pulling up on a control handle that juts out of the wing fold panel. The wing fold switch on these airplanes is located immediately forward of the pin pull handle. The wing pin pull handle and wing fold switch operates the same controls as the previously described wing pin unlock lever and wing fold switch. When folding or spreading the outer panels, observe the following precautions:

1. Jury struts removed.
2. Do not fold or spread wings broadside of the blast of an aircraft's engines.
3. Do not fold or spread wings in winds over 60 knots.

FOLD WINGS

To fold the wings, proceed as follows:

1. Pull UP on wing pin lock lever.
2. Place wing fold switch in FOLD position.

Note

Install jury struts after folding wings.

SPREAD WINGS

To spread wings, proceed as follows:

1. Remove jury struts.
2. Place wing fold switch in the SPREAD position.
3. After panel has spread and pins have extended, push DOWN on the wing pin lock lever.

Note

- Red warning flags, which are attached to the pinlocks will be flush with wing skin if pinlocks are fully inserted. The warning flags will extend above wing surface inboard of the wing fold line when pinlocks are not inserted.
- Check L.H. and R.H. wing unlock warning lights. When wing lock pin is fully inserted, warning lights should be extinguished.

CAUTION

- Ashore, whenever aircraft are parked or towed with wings folded, jury struts will be installed. Taxiing with wings folded and jury struts not installed will be held to a minimum.
- Afloat, jury struts will be installed at all times wings are folded.

WHEEL BRAKE SYSTEM

The main landing gear wheels are equipped with power-manually operated brakes. Two power brake valves are located in the nose wheel well and each is operated in a conventional manner from linkage attached to the rudder pedals. The brake control valves are power operated rather than a power boost type and assure positive brake feel to the pilot. Excessive pedal travel and pumping of the brakes in order to obtain a firm pedal is eliminated since the fluid supply to the wheel cylinders is virtually unlimited. This brake system provides differential wheel brake pressures. The brake valve incorporates both a power phase and a manual phase of operation. With no system hydraulic pressure available, the manual portion of the valve is capable of furnishing flow and pressure to accomplish a deceleration of the airplane and is also capable of securing the airplane during a deck roll of 8°. The brake system utilizes utility hydraulic system pressure.

EMERGENCY BRAKE HANDLE

In the event of hydraulic pressure failure, an emergency air system is provided to accomplish maximum braking. Emergency pressure is provided by a 100 cubic inch air bottle charged to 3000 psi. The emergency brake handle (figure 1-19) located beside the right armrest is painted in black and yellow stripes for ease of identification. It is hand-operated and spring-loaded to the off or brake released position. It is a power brake valve of conventional design and meters air pressure to completely independent pneumatic brake cylinders in proportion to applied pilot effort. The emergency pneumatic brake system does not provide a differential wheel brake pressure, however, by applying the manual portion of the hydraulic brakes in conjunction with the emergency air brakes, differential braking can be accomplished. Hydraulic and pneumatic systems are entirely separate; actuation of the emergency pneumatic system will not introduce air into the hydraulic system.

DRAG CHUTE SYSTEM

The airplane is equipped with a 16 foot ring slot type parachute which is deployed by the pilot after touchdown (when airspeed is below 200 knots IAS) to aid in reducing landing roll distances. The drag chute may also be utilized for spin recovery. The chute is carried in a compartment within the empennage at the base of the vertical stabilator. The chute is pulled into the airstream by a pilot chute when the spring-loaded compartment door is released. The design of the attaching mechanism is such that should the compartment door open inadvertently, without operating the cockpit control handle, the chute will be released and fall free of the airplane. The drag chute is retained to the airplane structure upon normal deployment. There is no "breakaway" fitting within the attaching mechanism.

DRAG CHUTE HANDLE

The drag chute is deployed by means of a control handle located along side of the left console. A cable joins the handle, the release and jettison mechanism, and the door latch mechanism. Rotating the handle aft without depressing the button on the handle, releases the door latch mechanism. The spring-loaded actuator then opens the drag chute door, and at the same time the hook lock is positioned over the drag chute attach ring. The spring-loaded pilot chute pops out, opens, and pulls out the drag chute. The drag chute is jettisoned by depressing the button and pulling aft on the handle to clear the detent and then by pushing the handle forward and releasing it. The release and jettison mechanism then returns to the normal position, permitting the drag chute to pull free.

ARRESTING GEAR

A conventional arresting hook and actuating cylinder is utilized to pneumatically extend and hydraulically retract the arresting gear. The hook is actuated by one combination shock absorbing actuating cylinder. The arresting gear consists of an arresting hook, a combination hydraulic actuator and dashpot, an up-latch mechanism, and a lever in the cockpit.

ARRESTING GEAR CONTROL

The arresting gear is controlled by a handle (figure 1-19) located at the right of the main instrument panel of the forward cockpit. When the control handle is pushed down, the latch is released and the arresting gear is extended by gravity and compressed air in the cylinder. Placing the arresting gear handle in the UP position actuates a switch which energizes a solenoid valve which directs hydraulic pressure to the cylinder to raise the arresting gear. A red warning light inside the control handle will come on when the control handle is placed in the down position and will remain on until the arresting gear is fully extended. The warning light will not illuminate again until the system is recycled. Electrical power for the gear is supplied by the left main 28 volt d-c bus.

APPROACH LIGHT SYSTEM

An approach light system with red-amber-green indications is used to show the angle of attack of the airplane on approach. The system operates in conjunction with the angle-of-attack system with the approach lights being operated by switches in the angle-of-attack indicator. The system is automatically energized when the arresting gear is down and the landing gear is down and locked. Two switches in the indicator unit determine which one of the three lights will be lit. The approach lights will burn steadily without the arresting gear being extended by momentarily holding the arresting gear bypass switch in the BYPASS position. This switch is accessible on the ground in the left gear wheel

ANGLE OF ATTACK UNITS	INDEXER SYMBOL	AIRPLANE ANGLE OF ATTACK	APPROACH LIGHT COLOR
1 18.4-30 2 20.3-30		HIGH (SLOW)	GREEN
1 17.8-18.4 2 19.7-20.3		SLIGHTLY HIGH (SLIGHTLY SLOW)	AMBER
1 16.8-17.8 2 18.7-19.7		OPTIMUM (ON SPEED)	AMBER
1 16.2-16.8 2 18.1-18.7		SLIGHTLY LOW (SLIGHTLY FAST)	AMBER
1 0-16.2 2 0-18.1		LOW (FAST)	RED

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1 Airplanes 145312b and 145314b thru 145317b

2 Airplanes 145307b, thru 145311b and 145313b, thru 148275e

well. One complete recycling of the arresting gear will reset the bypass switch to the NORMAL position.

COCKPIT ANGLE-OF-ATTACK INDEXER

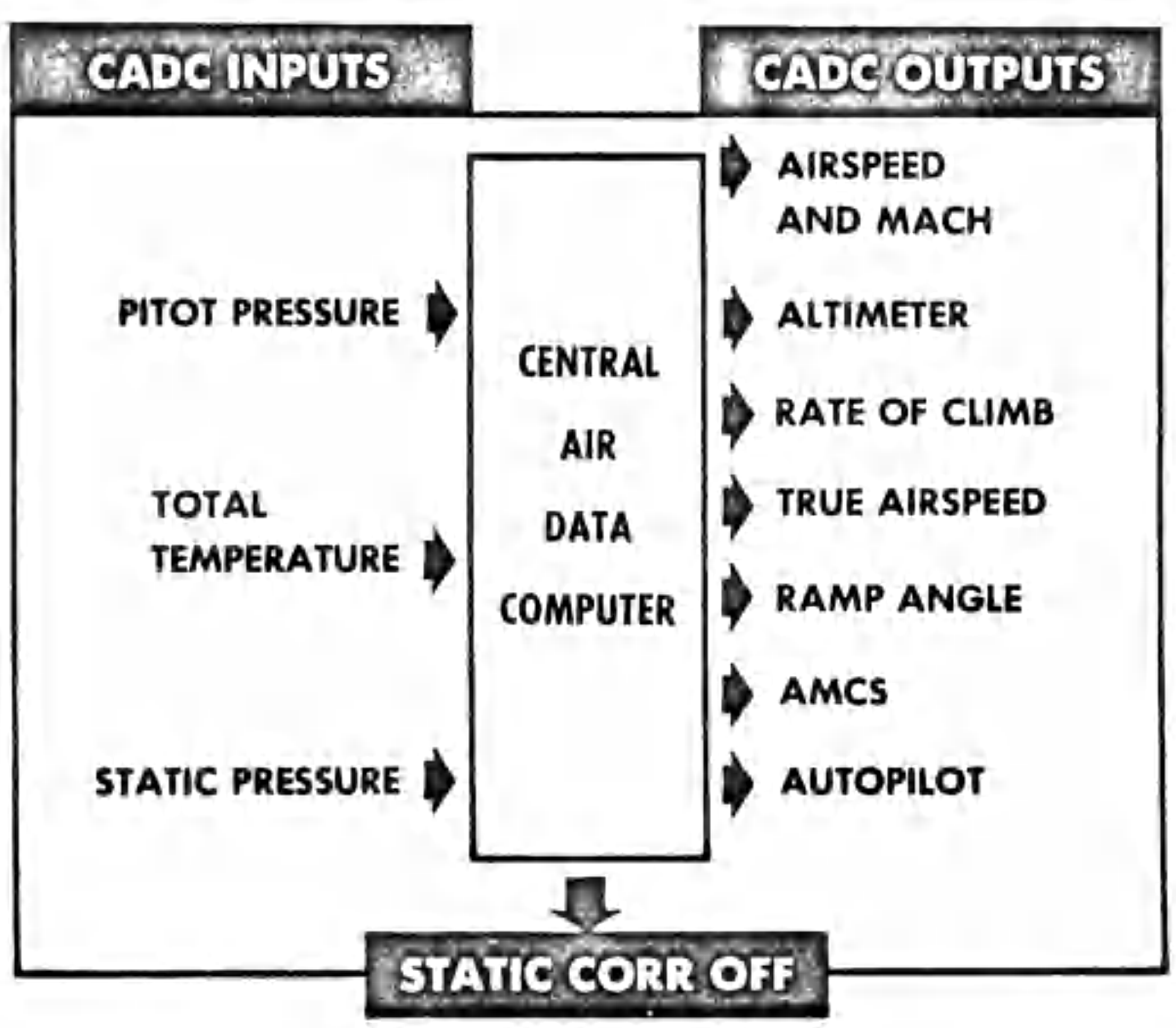
An angle-of-attack indexer is installed on the left side of the windshield in the pilot's cockpit. On some airplanes, two angle-of-attack indexers are installed, one on each side of the windshield. The indexer(s) present angle of attack information to the pilot while making a landing approach, by illuminating symbolic cutouts. The indexer(s) are energized by cam operated switches in the angle-of-attack indicator. At very low angles of attack an inverted "V" is illuminated and warns the pilot to increase his angle of attack. At slightly low angles of attack both the inverted "V" and circular symbols are illuminated. At optimum angles of attack only the circular symbol is illuminated. At very high angles of attack a "V" symbol is illuminated, warning the pilot to decrease his angle of attack. When approaching at slightly high angles of attack both the "V" and circular symbols are illuminated. The indexer information corresponds with the approach light assembly information.

PITOT-STATIC SYSTEM

A conventional pitot static system is used in the airplane with a single pitot tube located near the top of the vertical fin and two static ports located one on each side of the fuselage immediately aft of the radome. The purpose of the pitot static system is to supply both impact (pitot) and atmospheric (static) pressure to various instruments and system components. The pitot static system is composed of two separate systems. The pitot system obtains its pressure through one source and the static system through two sources. Both pressures may be utilized by the same instruments but at no time do the pressures intermingle. The heating element of the pitot tube is pilot operated by the pitot heat switch located on a panel on the pilot's right console (figure 1-19). Both pitot and static pressures are supplied to airspeed pressure switches that retract the flaps at 230 ± 5 knots, cut out the emergency generator at 195 ± 5 knots, and actuates the rudder feel system and aileron rudder interconnect (ARI) system at 240 ± 5 knots accelerating and 220 ± 5 knots decelerating. The pitot and static pressures are also directed to the central air data computer where they are calibrated and corrected (static pressure only) and then sent out to the various instruments and systems requiring pitot static pressures.

CENTRAL AIR DATA COMPUTER SYSTEM

The Central Air Data Computer receives three inputs from sensors which measure atmospheric conditions through which the airplane is flying. These inputs are impact air pressure, static air pressure, and total temperatures which are utilized by the Central Air Data Computer to provide corrected information to various systems, and indicators. The inputs to the CADC are supplied by the pitot tube and the total temperature sensor which are located on the vertical fin and by two static ports which are located immediately forward of the nose wheel well. As obtained from the sensing instruments, these measurements are in error by predictable amounts. The reading of an instrument measuring static pressure, for example, will deviate from the true static pressure, because of its location in the airstream moving over the surface of the aircraft, and the error introduced will vary with the airspeed. These indications from the measuring instruments are continuously corrected by passage through computing elements, so that the outputs are true indications of the required data. Output indications are presented as visual indications on panel-mounted instruments and as electrical signals (or resistance variations) for transmission to associated equipment in the aircraft. One type of indication, static pressure, is also presented as an actual air pressure which may be supplied to other equipment. The CADC system receives operating power from the 115 volt a-c essential bus, 28 volt a-c essential bus, 28 volt d-c essential bus, and 17th stage engine bleed air.



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CADC SYSTEM FAILURE

The instruments and/or systems utilizing the outputs from the CADC will be inoperative or in error if a failure or an interruption occurs in the following systems: essential a-c power supply, essential d-c power supply, or engine bleed air system. An interruption or failure in any of the above systems will illuminate the "Static Corr Off" warning light.

Static Pressure Compensator Warning Light

The static pressure compensator warning light is located on the outboard engine control panel (figure 1-18). This light, when illuminated, informs the pilot that his instrument readings may be in error due to the lack of a static pressure correction. Static pressure error (without compensation) is appreciable in the transonic and supersonic region. When the "Static Corr Off" light illuminates it is accomplished by a rapid change in the altimeter reading since the altimeter must respond to an instantaneous change in pressure being supplied to the instrument. The magnitude of the change various with Mach number. A static pressure reset button is located beside the warning light. This button, when depressed, resets the compensator and also breaks the circuit to the warning light. When the button is released, the warning light should be extinguished indicating that the instruments are reading correctly. If the warning light remains on, the compensator has not been reset, and the instruments are still in error. The static pressure compensator must be reset after the engines have been started, prior to each flight. The static pressure compensator warning light and a static pressure correction switch are located on the inboard engine control panel. The static pressure correction switch has positions of RESET CORR. NORM and CORR OFF. In the event the warning light illuminates, indicating a loss of static pressure correction, the RESET CORR position of the switch resets the compensator and breaks the circuit to the warning light. The CORR OFF position removes static pressure correction from the instruments. The static pressure compensator must be reset after the engines have been started prior to each flight.

Note

An engine must be operating in order to reset the static pressure compensator.

Note

In the event the "Static Corr Off" light illuminates in flight and cannot be reset, the variable ramp may not function properly and engine stalls may possibly be encountered in the high Mach number range. If practicable avoid Mach numbers above 1.5.

INSTRUMENTS

Most of the instruments are electrically operated by power from the electrical system, see figure 1-9. Some instruments, such as the accelerometer, are self-contained and do not require external power.

TRUE AIRSPEED INDICATOR

A true airspeed indicator (figure 1-17) is located on the pilot's and radar observer's instrument panels. The airspeed is indicated by a small counter which rotates to show a row of numbers through a window on the indicator face. The airspeed indicator is read directly in knots TAS; the range of the instrument is from 0 to 1500 knots to the nearest knot. The true airspeed outputs are produced from the signal from the total temperature sensor by routing this signal through a potentiometer driven by one of the Mach number function cams. Thus, Mach number is translated into true airspeed.

ALTIMETER

An altimeter (figure 1-17) located on the pilot's and radar observer's instrument panel is designed to indicate the altitude of the airplane above sea level. This unit is of the counter pointer type which displays the whole thousands numbers in a counter window and indicates the increments of the whole number with a pointer which rotates on the face of the instrument. The pointer scale is graduated in 50 feet units with major 100 feet scale divisions from 1 to 10. The range of the altimeter is 0 to 80,000 feet. An adjustable barometric scale is provided so that the altimeter may be set to sea level pressure. This scale range is from 28.10 to 31.00 inches mercury.

AIRSPEED AND MACH NUMBER INDICATOR

The combination airspeed and Mach number indicator (figure 1-17) shows airspeed readings at low speeds and includes Mach number readings at high speeds. Both readings are provided by a single pointer moving over a fixed airspeed scale, graduated from 80 to 850 knots, and a rotatable Mach number scale graduated from Mach .4 to Mach 2.5. A movable "bug" is included as a landing speed reference and can be positioned by the knob on the face of the instrument. The same knob can position another "bug" on the Mach number scale for maximum indicated airspeed reference. The airspeed indicator pointer and the Mach number scale are synchronized so that a proper relationship between the two is assured throughout all altitude changes. Thus, at sea level and under standard conditions, the pointer will indicate Mach 1 at approximately 660 knots. Under the same conditions, but at 50,000 feet, if the same true airspeed is maintained the pointer will indicate approximately 292 knots and a Mach number of 1.15.

ANGLE-OF-ATTACK INDICATOR

This indicator (figure 1-17) measures the angle of attack of the airplane by reflecting the direction of airflow relative to the fuselage. This is accomplished by means of a probe protruding through the fuselage skin. Airstream direction is sensed by means of a pair of parallel slots in the rotating probe protruding horizontally into the airstream. When the airplane changes its angle of attack, pressure becomes greater in one slot than the other. The pressure differential sensing diaphragm opens or closes the control pressure regulator and engine bleed air pressure is increased or decreased to move the probe actuating diaphragm and align the probe slots with the airstream. Probe rotation moves the potentiometer wiper arms a proportional amount, producing resistance variations which are the signals sent to the angle of attack indicator. The angle of attack indicator is calibrated from 0 to 30 in arbitrary units equivalent to a range of -10 to +40 angular degrees of rotation of the probe. Pointers are provided and are set at the following cruise, stall and approach optimum angle-of-attack reference points:

Airplanes 145312b, and 145314b thru 145317b

Cruise	4.7 units
Stall	30.0 units
Approach	17.3 units

Airplanes 145307b thru 145311b, 145313b and 146817c thru 148275e.

Cruise	7.9 units
Stall	30.0 units
Approach	19.2 units

The angle-of-attack indicator also contains a switch that actuates the pedal shaker at 20.5 units on airplanes 145312b and 145314b thru 145317b and at 22.3 units on airplanes 145307b thru 145311b, 145313b, and 146817c thru 148275e. When the indicator is inoperative, the word OFF shows in a small window on the face of the dial. Refer to NAVWEPS 01-245FDA-1A, Section VI, Angle-of-Attack Conversion for optimum angle-of-attack units for various airspeeds and gross weights.

Note

The angle-of-attack indicating system is unreliable at engine rpm's lower than 80% with flaps extended, and 73% with flaps retracted. This unreliability is caused by insufficient engine bleed air at low rpm's to operate the airstream direction transducer.

AN/AJB-3 ATTITUDE DIRECTOR INDICATOR

The AN/AJB-3 all attitude, 3 axis, stabilized platform indicating system (figure 1-17) has two distinct functions. First it furnishes the pilot with all the necessary information to perform low altitude bombing maneuvers. Refer to Section IV for detailed information concerning the low altitude bombing system. Secondly, the AN/AJB-3 system functions as a pitch and roll stabilized directional gyro compass system. The system also provides heading, pitch and bank signals to the autopilot and navigation computer. The display sphere is painted to give the illusion of visual flight. Heading is marked off in five-degree increments along the horizontal and in 30 degree increments at high or low pitch angles. Bank angle scale around the periphery is graduated in 10 degree increments up to 30 degrees. When the airplane is flying at pitch angles above or below 70 degrees where changes in bank angle actually represents changes in airplane heading, the bank angle graduations automatically disappear from the indicator bezel so that the pilot will think of aileron control in terms of its effect of changing airplane heading rather than in terms of bank angle. The bank graduations automatically return when the airplane pitch or dive angle drops below approximately 70 degrees. Use of a stabilized platform instead of conventional gyro compass and horizon gyro, provides unlimited pitch, roll axis maneuvering without gyro tumbling or discontinuity in pitch axis indications during an Immelmann turn. It also eliminates large heading errors at high pitch and bank angles, or during continuous turns. Power for the operation of this system is supplied by the essential 115 volt a-c bus, the essential 26 volt a-c bus and the essential 28 volt d-c bus. A pitch trim knob is provided on the indicator for the pilot to center the horizon bar in relation to the fixed miniature airplane.

VERTICAL VELOCITY INDICATOR

A vertical velocity indicator (figure 1-17) is located on the pilot's instrument panel. The indicator shows the rate of ascent or descent of the airplane and is so sensitive that it can register a rate of gain or loss of altitude which would be too small to cause a noticeable change in the altimeter reading. The upper half of the indicator face is graduated in 500 foot units from 0 to 6,000 feet with 100 foot scale divisions from 0 to 1,000 feet. The upper half of the instrument indicates rate of climb in thousands of feet per minute. The lower half of the indicator face is identical to the upper half except that it would indicate rate of descent in thousands of feet per minute. The vertical velocity indicator is connected to the static pressure system of the airplane and measures the change in atmospheric pressure as the airplane climbs or descends.

TURN-AND-SLIP INDICATOR

A conventional turn-and-slip indicator (figure 1-17) on the main instrument panel operates off the low pressure pneumatic system. The dial of the turn indicator is plain having only the neutral mark and an index mark on each side of neutral. This index indicates a turn of 180° in one minute when the pointer is coinciding with the index. Effective airplanes 145313b, 146817c thru 148275e, the bank inclinometer and rate of turn needle are incorporated in the ADI (figure 4-17). The rate of turn needle measures turn rate only around the vertical axis of the airplane. Therefore, increased bank angle will not necessarily be reflected in increased needle deflection. Although the actual turn rate of the airplane increases, the turn rate about the vertical axis of the airplane does not. Consequently, do not depend too strongly on the turn needle for establishing turn rate during IFR flying since, the rate of turn needle should never indicate a single needle width deflection in coordinated flight. Single needle width turns can only be accomplished during low speed flight due to increased turn rate for a given bank angle, however, beyond a certain point needle deflection will not continue to increase with bank angle. The index indicates a turn of 180° in one minute when the pointer is coinciding with the index.

ACCELEROMETER

An accelerometer (figure 1-17) with three indicating hands, registers and records positive and negative "g" loads. One hand moves in the direction of the "G" load being applied while the other two, one for positive "g" loads and one for negative "g" loads, follow the indicating pointer to its maximum travel. The recording pointers remain at the respective maximum travel positions of the "g's" being applied, thus providing a record of maximum "g" loads encountered. Depressing the push to reset knob at the lower left corner of the instrument, allows the recording pointers to return to the normal (1 g) position.

Note

Accelerometers may read as much as 1/2 "g" low; possibly lower if the pull-in rate is high.

STANDBY COMPASS

A conventional magnetic compass mounted on the cockpit windshield frame is provided for navigation in event of instrument or electrical malfunction. Compass cards are located above the canopy sill on the right side of the cockpit.

EMERGENCY EQUIPMENT

ENGINE FIRE AND OVERHEAT DETECTOR SYSTEM

Fire Warning Lights

Two fire warning lights (figure 1-17), one for each fire warning system, are located on the upper portion of the main instrument panel. Along with the lights, each system consists of a control unit, and a series of continuous type sensing elements, which are routed throughout the engine compartments. The lights are energized when a temperature of approximately 765°F occurs in the engine compartment. This excessive temperature causes the sensing element to ground out and unbalance a bridge network in the detector control unit. The unbalancing of this bridge circuit energizes the fire warning light for the engine which is in an overtemperature condition. On some airplanes an aft fuselage overheat warning system has been installed, and on these airplanes a combination Fire and Overheat warning light is used. The aft fuselage overheat warning systems are separate (but similar) to the fire-warning systems. The sensing elements are routed vertically in recesses provided in the skin fairing of the keel. These recesses are located approximately opposite the aft end of the secondary engine nozzle fingers. The energizing temperature is the same as for the fire warning systems, approximately 1050°F. Illumination of the Fire or Overheat warning lights warns the pilot to initiate emergency procedures. In general, illumination of either the Fire or Aft Fuselage Overheat warning lights signals the pilot to reduce power immediately on the affected engine to a setting of military or lower. Afterburning must never be used if the Overheat warning light illuminates. Unless the warning is false, it means a safety of flight condition exists, such as an open engine compartment door or a damaged engine nozzle. Either of these conditions can lead to loss of flight control if afterburning is used.

Note

Illumination of "Fire" or "Overht" warning lights shall be logged on the yellow sheet (Op NAV FORM 3760-2).

Fire Detector Check Switch

The warning lights may be tested by momentarily placing the warning lights switch (figure 4-13) in the TEST position; this will only check the continuity of the bulbs. By depressing the fire detector check switch (figure 1-17), the pilot checks the fire warning circuit from the bulb to the sensing element. The fire warning light circuit receives power from the left main 115/200 volt a-c bus.

CANOPIES

Each cockpit area is enclosed by a separate transparent acrylic plastic clam shell type canopy. The canopies are hinged aft of each cockpit enclosure and open approximately 53°. Canopy operation, both normal and emergency, is accomplished pneumatically. Clean dry air at 3000 psi is supplied by the basic pneumatic system and is reduced by a pressure regulator to 550 +30, -0 psi for use in the normal canopy system. Individual manual controls are provided for each cockpit. In addition, external operation is provided for each cockpit by means of individual push buttons on the exterior of the fuselage. In normal operation the canopy operating time is set at 5 seconds; this is controlled by pneumatic restrictors incorporated in the system. Each cockpit employs an inflatable canopy seal to seal the canopies for cockpit pressurization. The canopy seals are automatically inflated and deflated upon opening and closing of the canopies.

EXTERNAL CANOPY CONTROLS

External Canopy Buttons

The forward cockpit external canopy control push buttons are located on the left side of the airplane just forward of the engine air intake duct. The aft cockpit external canopy control push buttons are located on the left side of the airplane just above the engine air intake ducts between the forward and aft canopies. The forward and aft canopy push buttons operate independently of each other but their functions are the same. Push OPEN button to open the canopy; push CLOSE button to close the canopy. The push buttons operate the same valves as the internal canopy controls.

External Manual Unlock Handles

The forward canopy external manual unlock handle is located on the left side of the fuselage below the aft end of the canopy. The aft canopy external manual unlock handle is located in the same position below the aft canopy. Each handle operates independently of the other but their functions are the same. Operating a push-type latch causes the handle to pop out about 1-3/4 inches. A 63° counterclockwise rotation of the handle unlocks the canopy down-lock mechanism and permits the canopy to be lifted open manually. The aft canopy manual unlock handle operates the same as that of the forward canopy except that the handle rotation is clockwise.

INTERNAL CANOPY CONTROLS

Canopy Control Handle

The forward cockpit canopy control handle (figure 1-18) is located on the left side of the cockpit above the flap control panel and just below the canopy sill. The aft cockpit canopy control handle is located in the same position in the aft cockpit. Each canopy control operates independently of the other but their functions are the same. Pull control handle aft to OPEN the canopy; push forward to CLOSE the canopy. Returning the control handle to the neutral position when opening or closing the canopy allows the canopy to be stopped at an intermediate position.

Manual Canopy Unlock Handles

The forward cockpit manual canopy unlock handle (figure 1-19) is located on the right side of the cockpit above the arresting gear control. The aft cockpit manual canopy unlock handle is located in the same position in the aft cockpit. Each cockpit manual canopy unlock handle operate independently of the other but their functions are the same. The handle when pulled aft breaks the canopy downlock linkage, releasing the downlocks so that the canopy may then be pushed open to permit exit from the cockpit. Before manual unlocking of the canopy, the normal control lever must be placed in the open position. The manual unlock handle is used in the event that the airplane pneumatic systems are depleted.

Note

When the canopy has been opened manually, it should be held open while entering or leaving the cockpit, since without power the actuator may not hold the canopy in the open position.

Canopy Unlocked Warning Light

An amber Canopy Unlocked warning light (figure 1-19) located in the forward and aft cockpits is used to notify the crew that a canopy is unlocked. The pilot's canopy unlocked warning light will illuminate when either canopy should happen to be unlocked. The observer's warning light will illuminate only when his canopy is unlocked. Power for the canopy unlocked warning light is supplied by the essential 26 volt a-c warning light bus.

Note

The canopy is designed to remain in the full open position up to 60 knots and to separate from the airplane at approximately 100 knots.

CANOPY EMERGENCY SYSTEM

Clean dry air at 3000 psi is supplied from the basic pneumatic system to air bottles which furnish air for both canopy emergency systems. After leaving the air bottles, the system pressure is reduced to 1375 ± 100 psi in order to minimize the possibility of structural damage upon emergency opening of the canopy. Each canopy has individual manual control valves that operate independently of each other to direct the flow of air to the actuating cylinders in each cockpit. These valves can be operated by pulling on the face curtain pull handles, alternate ejection handle ("D" ring), or by actuating the emergency canopy release lever. In addition, one emergency control is provided on the fuselage to open both canopies simultaneously for ground crew rescue purposes. The canopy emergency system also actuates the forward and aft cockpit flooding doors in order to reduce the time required to equalize the cockpit pressure under water in the event the airplane ditches. Design time for deck operation or testing of canopy emergency system is 2 seconds. Inflight operation of the canopy emergency system takes 1 second. Design time for under-water pressure equalization of the cockpit is 7 seconds.

EXTERNAL EMERGENCY CANOPY CONTROL

Emergency Jettison Lanyard

The external emergency canopy jettison lanyard is located on the lower left side of the fuselage just forward of the aft nose gear door. Access to the handle is through a small door hinged at the forward end with a push latch release. Pulling the handle through a cable system operates the emergency canopy valve. This control simultaneously operates both the forward and aft canopy emergency systems and is intended for use by the ground crew for rescue purposes.

INTERNAL EMERGENCY CANOPY CONTROLS

Emergency Canopy Release Handle

The forward cockpit emergency canopy release handle (figure 1-18) is located on the left side of the cockpit above the landing gear control. The handle is placarded emergency canopy release and is painted with black and yellow stripes for ease of identification. The aft cockpit emergency canopy release handle is located in the same position and is placarded and painted the same as the forward cockpit lever. Each canopy emergency release handle operates independently of the other but their functions are the same. The handle, when pulled aft, operates the emergency canopy valve which directs 1375 ± 100 psi to the canopy actuator and cockpit flooding doors actuators. When the aft emergency canopy release handle is pulled aft, the action is identical to that in the forward cockpit. In addition, operation of the handle stows both the rear cockpit flight indicator and radar set control to provide more room for exit from the cockpit, either by ejection or climbing out under emergency conditions. The emergency canopy release handle is primarily used for ditching.

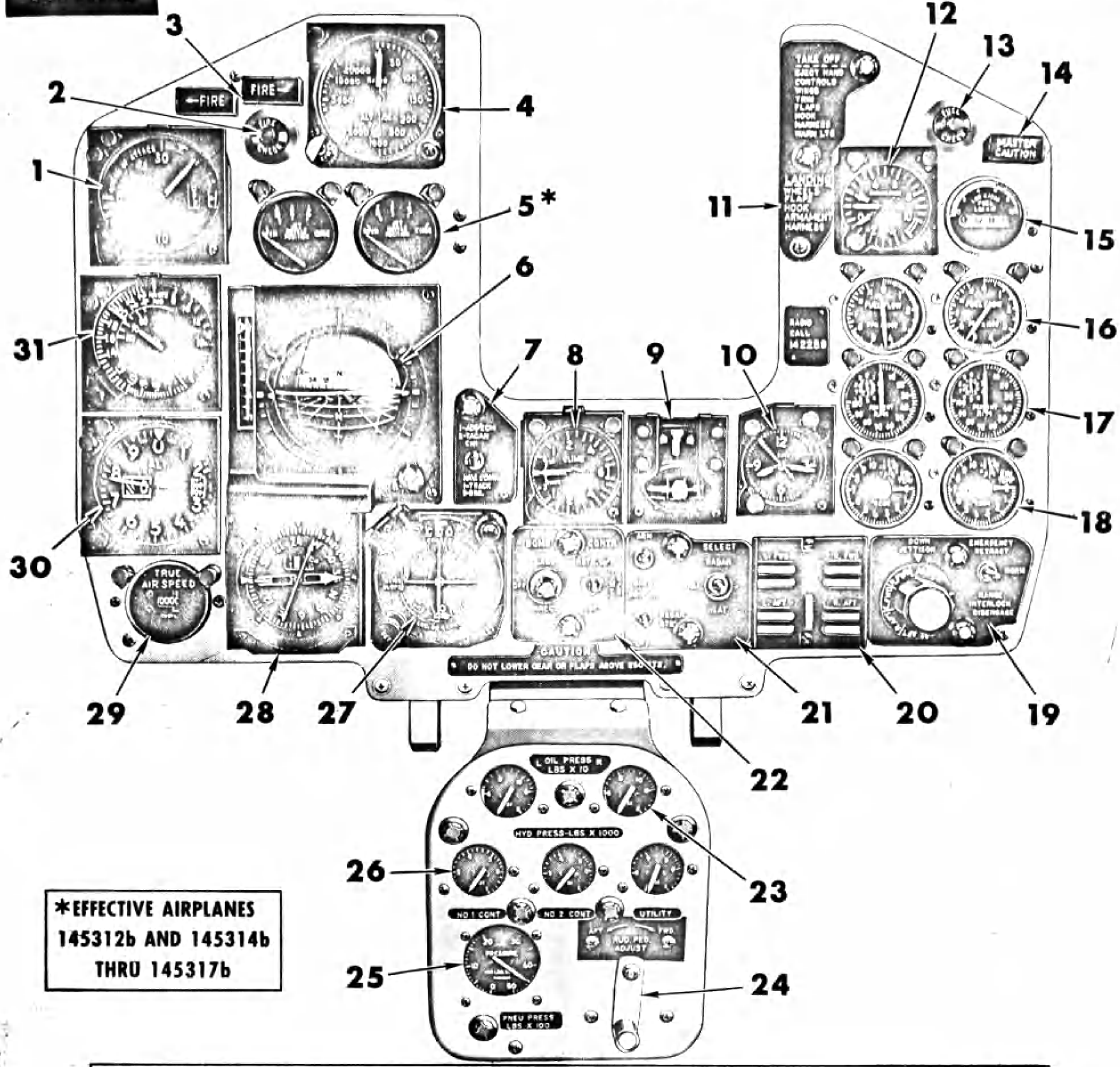
Emergency Canopy Release (Ejection Seat)

The seat ejection handles are used to jettison the canopy prior to ejection. A downward pull on the face curtain handle, or an upward pull on the alternate ejection handle ("D" ring) will jettison the canopy. Refer to Ejection Seats, this section.

PILOT'S INSTRUMENT PANEL

TYPICAL

EFFECTIVE AIRPLANES 145307b THRU 145312b AND 145314b THRU 145317B (PRIOR TO THE F4H DRIVE-IN MODERNIZATION PROGRAM).



*EFFECTIVE AIRPLANES 145312b AND 145314b THRU 145317b

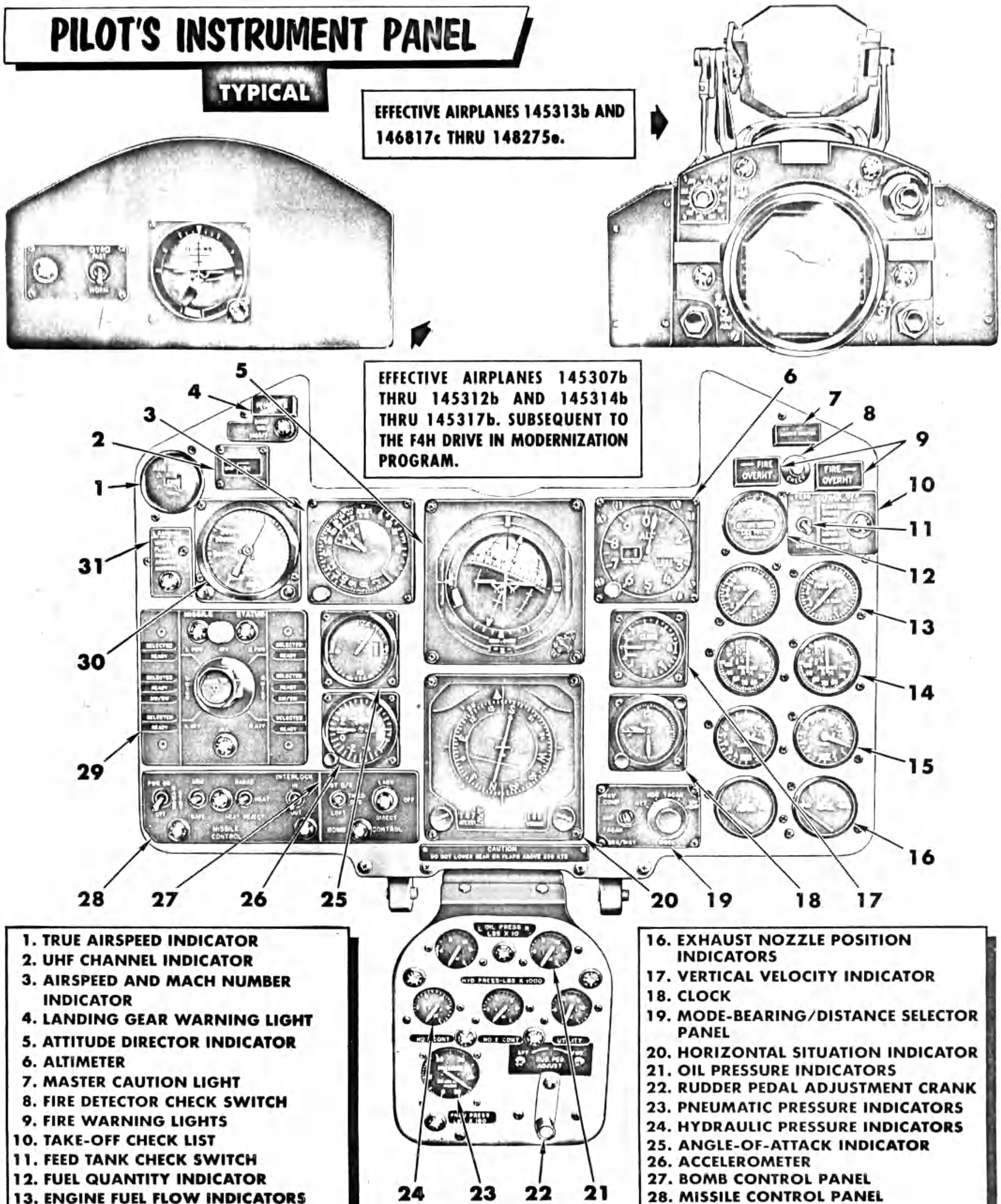
1. ANGLE OF ATTACK INDICATOR
2. FIRE DETECTOR CHECK SWITCH
3. FIRE WARNING LIGHTS
4. RADIO ALTIMETER
5. EXHAUST NOZZLE POSITION INDICATORS*
6. AN-AJB-3 ATTITUDE INDICATOR
7. NAVIGATION FUNCTION SWITCH
8. VERTICAL VELOCITY INDICATOR
9. TURN AND SLIP INDICATOR
10. CLOCK
11. TAKE-OFF AND LANDING CHECK LIST
12. ACCELEROMETER
13. FUEL CHECK SWITCH
14. MASTER CAUTION LIGHT
15. FUEL QUANTITY INDICATOR
16. ENGINE FUEL FLOW INDICATORS
17. TACHOMETERS
18. EXHAUST GAS TEMPERATURE INDICATORS
19. MISSILE JETTISON PANEL
20. ARMAMENT LIGHTS PANEL
21. MISSILE CONTROL PANEL
22. BOMB CONTROL PANEL
23. OIL PRESSURE INDICATORS
24. RUDDER PEDAL ADJUSTMENT CRANK
25. PNEUMATIC PRESSURE INDICATORS
26. HYDRAULIC PRESSURE INDICATORS
27. ID-387/ARN COURSE INDICATOR
28. BEARING DISTANCE HEADING INDICATOR
29. TRUE AIRSPEED INDICATOR
30. ALTIMETER
31. AIRSPEED AND MACH NUMBER INDICATOR

FM-FDA-120-1E

Figure 1-17 (Sheet 1)

PILOT'S INSTRUMENT PANEL

TYPICAL



1. TRUE AIRSPEED INDICATOR
2. UHF CHANNEL INDICATOR
3. AIRSPEED AND MACH NUMBER INDICATOR
4. LANDING GEAR WARNING LIGHT
5. ATTITUDE DIRECTOR INDICATOR
6. ALTIMETER
7. MASTER CAUTION LIGHT
8. FIRE DETECTOR CHECK SWITCH
9. FIRE WARNING LIGHTS
10. TAKE-OFF CHECK LIST
11. FEED TANK CHECK SWITCH
12. FUEL QUANTITY INDICATOR
13. ENGINE FUEL FLOW INDICATORS
14. TACHOMETERS
15. EXHAUST GAS TEMPERATURE INDICATORS
16. EXHAUST NOZZLE POSITION INDICATORS
17. VERTICAL VELOCITY INDICATOR
18. CLOCK
19. MODE-BEARING/DISTANCE SELECTOR PANEL
20. HORIZONTAL SITUATION INDICATOR
21. OIL PRESSURE INDICATORS
22. RUDDER PEDAL ADJUSTMENT CRANK
23. PNEUMATIC PRESSURE INDICATORS
24. HYDRAULIC PRESSURE INDICATORS
25. ANGLE-OF-ATTACK INDICATOR
26. ACCELEROMETER
27. BOMB CONTROL PANEL
28. MISSILE CONTROL PANEL
29. MISSILE STATUS PANEL
30. RADIO ALTIMETER
31. LANDING CHECK LIST

FM-FDA-120-2D

Figure 1-17 (Sheet 2)

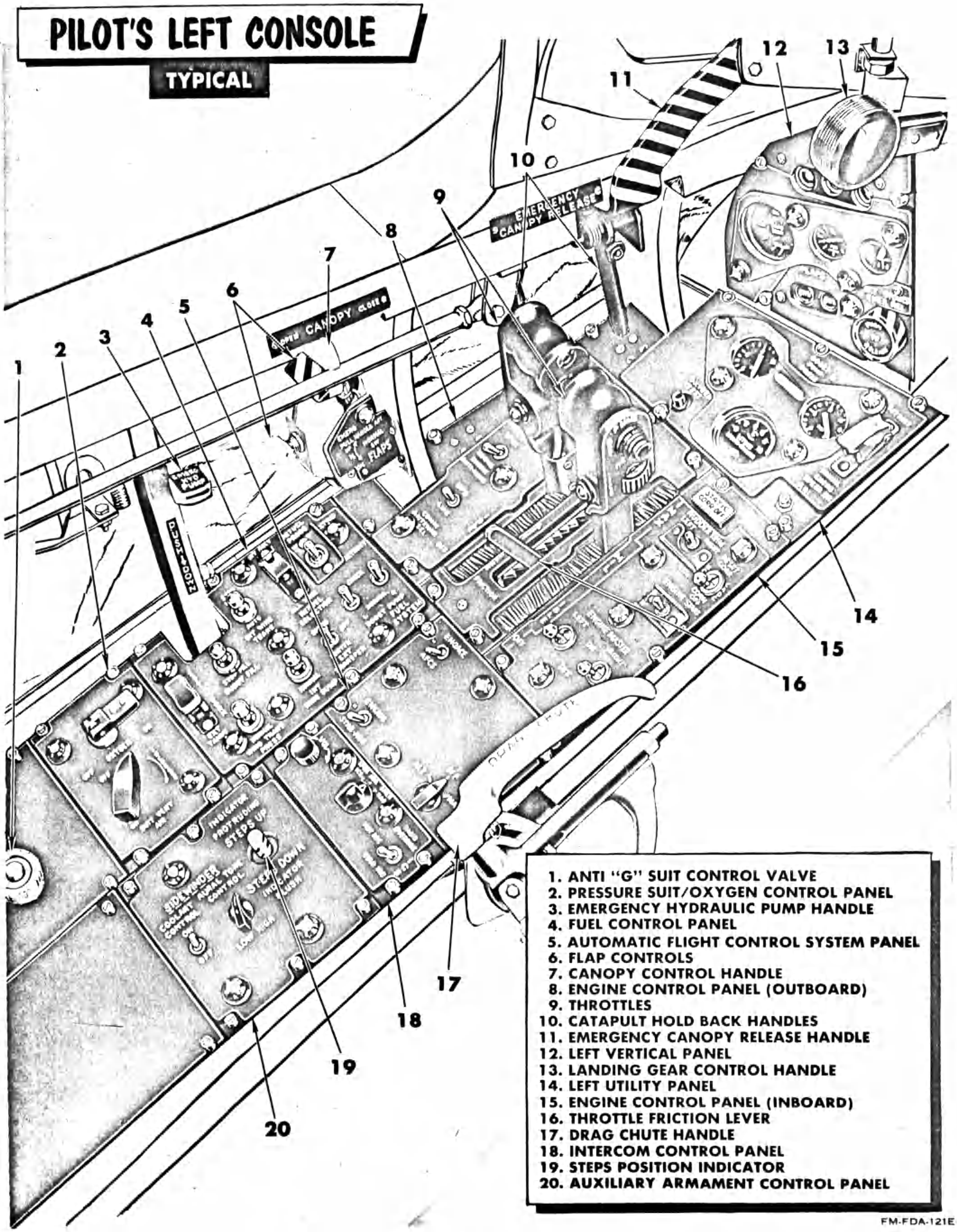
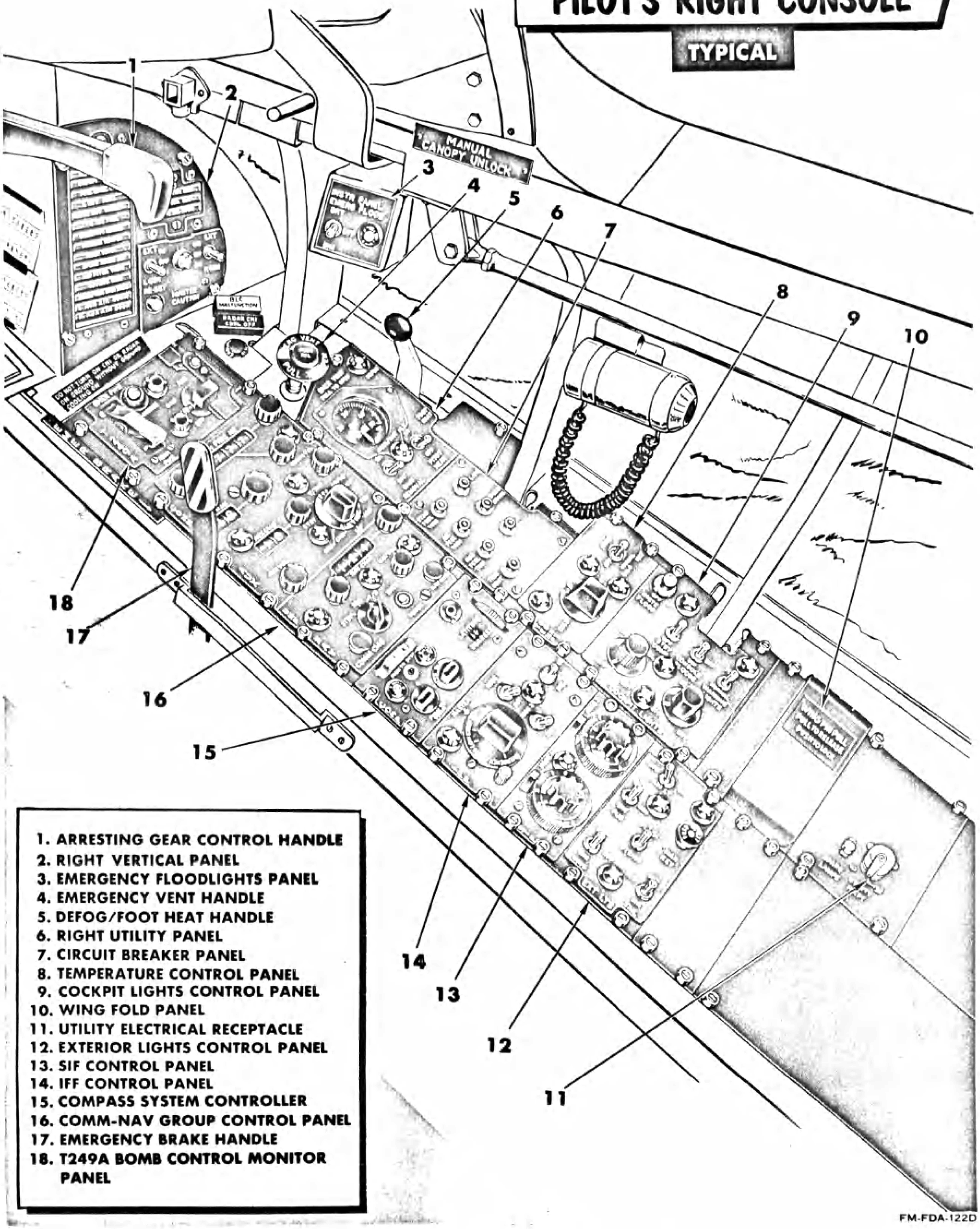

PILOT'S LEFT CONSOLE
TYPICAL
EMERGENCY CANOPY RELEASE
CANOPY
PUSH DOWN
FLAPS
STEPS UP
STEPS DOWN
SIDEWINDER COOLANT CONTROL
1. ANTI "G" SUIT CONTROL VALVE
2. PRESSURE SUIT/OXYGEN CONTROL PANEL
3. EMERGENCY HYDRAULIC PUMP HANDLE
4. FUEL CONTROL PANEL
5. AUTOMATIC FLIGHT CONTROL SYSTEM PANEL
6. FLAP CONTROLS
7. CANOPY CONTROL HANDLE
8. ENGINE CONTROL PANEL (OUTBOARD)
9. THROTTLES
10. CATAPULT HOLD BACK HANDLES
11. EMERGENCY CANOPY RELEASE HANDLE
12. LEFT VERTICAL PANEL
13. LANDING GEAR CONTROL HANDLE
14. LEFT UTILITY PANEL
15. ENGINE CONTROL PANEL (INBOARD)
16. THROTTLE FRICTION LEVER
17. DRAG CHUTE HANDLE
18. INTERCOM CONTROL PANEL
19. STEPS POSITION INDICATOR
20. AUXILIARY ARMAMENT CONTROL PANEL
FM-FDA-121E

Figure 1-18

PILOT'S RIGHT CONSOLE

TYPICAL



1. ARRESTING GEAR CONTROL HANDLE
2. RIGHT VERTICAL PANEL
3. EMERGENCY FLOODLIGHTS PANEL
4. EMERGENCY VENT HANDLE
5. DEFOG/FOOT HEAT HANDLE
6. RIGHT UTILITY PANEL
7. CIRCUIT BREAKER PANEL
8. TEMPERATURE CONTROL PANEL
9. COCKPIT LIGHTS CONTROL PANEL
10. WING FOLD PANEL
11. UTILITY ELECTRICAL RECEPTACLE
12. EXTERIOR LIGHTS CONTROL PANEL
13. SIF CONTROL PANEL
14. IFF CONTROL PANEL
15. COMPASS SYSTEM CONTROLLER
16. COMM-NAV GROUP CONTROL PANEL
17. EMERGENCY BRAKE HANDLE
18. T249A BOMB CONTROL MONITOR PANEL

Figure 1-19

EJECTION SEATS

The airplane ejection seats are designed to give the crewmember both low level and high altitude escape capability. The seat will provide safe escape at airspeeds from 130 to 350 knots between the altitudes of 50 and 1000 feet. It will also provide safe escape at airspeeds up to 550 kts above 1,000 feet. Ground level ejection is made possible by the use of a high velocity, triple cartridge charge catapult gun, and by use of two stabilizing drogue chutes which stabilize the seat and reduce height loss to a minimum. During ejection, the occupant is held in the seat by body harness and leg restraining systems. Time delay mechanisms automatically release the occupant after a safe deceleration and a safe altitude has been reached. All features are discussed in greater detail as this description progresses. The seat catapult uses cartridges which provide sufficient force to insure ejection of the seat and occupant clear of the airplane tail structure. The seat catapult is also used as a guide rail for the seat by means of channels fitted on either side of the catapult barrel. Power to eject the seat and occupant is obtained from one primary and two auxiliary cartridges. Pulling either ejection handle to the fullest extent draws a wedge-shaped sear out of the firing pin, allowing the spring-loaded pin to strike the percussion primer of the primary charge. Expanding gases drive the intermediate and inner tubes upward. As the tubes travel upward ports to the auxiliary cartridges are uncovered and ignited. The seat is controlled after ejection by two drogue chutes which are deployed automatically. The smaller drogue chute (pilot chute) is drawn from the pack by a metal piston fired from a drogue gun one-half second after seat ejection. The pilot drogue chute when withdrawn from the pack, tows the main drogue parachute out of its container. The main chute, when deployed, stabilizes the seat, tilts it to a horizontal position, and slows down the seat. This action ensures deceleration approximately in line with the seat axis, and the horizontal flight path of the seat reduces height loss to a minimum. If ejection is made at a high altitude, a barostatic control attached to the seat delays separation of the occupant from the seat and prevents the crewmember's parachute from opening. This permits the occupant and the seat to descend quickly through the cold rarified atmosphere to a more tolerable altitude (10,000 +3000 -000 ft.) where the automatic mechanism will separate the occupant from the seat and deploy the personal chute. At very high ejection speeds the opening of the personal parachute is delayed by a switch and the crewmember remains strapped in the seat, which is steadied by the controller drogue chute and main drogue parachute until the seat has decelerated to a safe speed (4.5 "g"). His legs are held in place by a restraining harness which keeps them from flailing in the airstream. The seat is built to accommodate the Martin Baker parachute system and a seat pack type survival kit. A composite disconnect is attached to the left rear corner of the seat pack. The composite disconnect houses the quick-disconnect plugs for the pressure suit, anti-G system, suit ventilation, oxygen and electrical lines for communication purposes. By pulling up on the composite disconnect release knob, the unit can be quickly disconnected from the seat. When the crewmember is ejected, bailout oxygen is automatically supplied.

DROGUE GUN

The drogue gun is mounted on the left side of the ejection seat headrest and is used to extract the controller drogue from its container 1/2 second after ejection. Upon ejection, a trip rod fixed to the airplane structure pulls a sear from the drogue gun to initiate the 1/2 second time delay. After the time delay has elapsed, a cartridge is fired and the resultant gas pressures propel a piston out of the drogue gun barrel. Attached to this piston is a lanyard which pulls the controller drogue parachute from its container. When deployed, the controller drogue pulls the stabilizer drogue (main drogue) from its container.

TIME RELEASE MECHANISM

The time release mechanism is located on the right side of the ejection seat headrest. Its function is to delay deployment of the personnel parachute and separation of the occupant from the seat until the occupant has descended from the cold rarified upper atmosphere and/or has slowed enough to prevent excessive opening shock of the personnel parachute. The time release mechanism is armed upon ejection by a trip rod secured to the airplane. Initiation of the timing sequence follows immediately, providing the altitude is below 10,000 feet and the deceleration rate of the seat is less than 4.5 g's. Initiation is delayed until these conditions are met. One and one half seconds after initiation, the time release mechanism releases the drogues from the scissors shackle allowing the personnel parachute to be pulled from its container. At the same time, it unlocks the harness and leg restraint lines to allow the occupant to be pulled from the seat when the personnel parachute deploys.

SEAT POSITIONING SWITCH

The pilot's and RIO ejection seat may be adjusted vertically only. Fore and aft seat positioning is compensated for by adjustable rudder pedals. Vertical seat positioning is accomplished by actuating a momentary contact switch (figure 1-21) located on the right forward side of the seat bucket. The pilot's seat can be adjusted (up or down) in flight through a total distance of 6 inches. The RIO seat is adjustable in flight through a total of 5 1/4 inches. It is not necessary to adjust the seat height to any certain position before ejection. Electrical power is supplied to the seat by the left main 115 volt a-c bus.

FACE CURTAIN EJECTION HANDLE

A face curtain ejection handle (figure 1-21) is provided for normal seat ejection. The ejection handle is located at the top of the seat, projecting forward and providing a grip surface for the crew member. When ejection is desired a forward and downward pull on the

EJECTION SEATS

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The pilot's and RIO ejection seat may be adjusted vertically only. Fore and aft seat positioning is compensated for by adjustable rudder pedals. Vertical seat positioning is accomplished by actuating a momentary contact switch (figure 1-21) located on the right forward side of the seat bucket. The pilot's seat can be adjusted (up or down) in flight through a total distance of 6 inches. The RIO seat is adjustable in flight through a total of 5 1/4 inches. It is not necessary to adjust the seat height to any certain position before ejection. Electrical power is supplied to the seat by the left main 115 volt a-c bus.

FACE CURTAIN EJECTION HANDLE

A face curtain ejection handle (figure 1-21) is provided for normal seat ejection. The ejection handle is located at the top of the seat, projecting forward and providing a grip surface for the crew member. When ejection is desired a forward and downward pull on the handle will fire an initiator, and the expanding gases from the initiator operate the emergency canopy valve which directs 1375 ± 100 psi to the canopy actuator and cockpit flooding doors actuators. The canopy will open and air loads will separate the canopy from the cockpit and remove an interlock from the seat firing mechanism. After the canopy has jettisoned the handle can be pulled the remainder of the way, until the full travel is reached. This pulls the face curtain over the face and removes the wedge-shaped sear from the ejection gun firing head, firing the main charge and ejecting the seat. Pulling the face curtain on the RO's ejection seat will also stow the RO's flight indicator and radar set control to provide more room for ejection from the cockpit. The interlock in the firing mechanism of both seats prevents seat ejection before the canopy is jettisoned.

SECONDARY EJECTION HANDLE

The secondary ejection handle (figure 1-21) is located on the seat bucket between the crew member's legs. This handle works in the same manner as the face curtain pull handle, i.e., an upward pull on the handle will jettison the canopy, which will remove the interlock and allow the handle to be pulled the remainder of the way until full travel is reached and the seat is ejected. Pulling the secondary handle on the RO's seat will also stow the flight indicator and the radar set control.

SHOULDER HARNESS LOCKING REEL

The shoulder harness locking reel mechanism is part of the seat center cross member and is approximately at shoulder level. It is used to hold the occupant's upper torso against the seat back during all flight conditions and the ejection cycle. The locking reel consists of a strap wound on a spring-loaded reel, a snubbing unit, and an upper harness release pin. The strap wound around a spring-loaded retraction reel passes through a snubbing unit, and then through the occupant's upper harness roller fitting. The eye of the strap then returns back to the seat and through the rings of the two straps, securing the personnel parachute to the seat, and is then anchored by the upper harness release pin. The snubbing unit prevents any forward movement of the strap unless release is effected by pushing forward on the release handle. During ejection, upon actuation of the time release mechanism, the upper harness release pin releases the eye fitting, thereby freeing the occupant's upper harness and the parachute securing straps.

Locking Reel Release Handle

The locking reel release handle (figure 1-21) is located on the left side of the seat bucket. It is spring-loaded aft, which is the normal and locked position of the locking assembly. The forward position of the handle unlocks the reel assembly. When the handle is pushed forward and held, the reel assembly will permit normal forward and aft movement of the pilot's shoulders. When the handle is released, it will spring aft to the locked position and the reel will prevent forward movement. Should the pilot or crewmember be leaning away from his seat when he releases the handle and locks the reel and then moves or bounces back toward the seat, the reel assembly automatically takes up the slack and locks the reel in successive locked positions.

Shoulder Harness Inertia Reel

Effective airplanes 146817c thru 148275e, a conventional inertia reel replaces the locking reel. This reel may be locked to preclude any movement, and will also automatically lock when the seat and occupant are subjected to the high "g" forces which would be associated with sudden stoppage or ejection.

Inertia Reel Handle

Although it is mounted in the same place, the inertia reel handle does not function in the same manner as the locking reel release handle. The inertia reel handle is mechanically linked to a two-position cam and must be mechanically moved to the unlocked (aft) position or the locked (forward) position.

Emergency Harness Release Handle

The emergency harness release handle (figure 1-21) is located on the right side of the seat bucket. The handle is used by the occupant to manually separate from the seat in the event the automatic time release mechanism fails or in the event of ditching. Pulling aft on the handle operates a system of linkages which pulls the pins retaining the lap belt harnessing, pulls the pin from the shoulder harnessing, and releases the leg restraint lines. This handle performs the same functions as the time release mechanism. To actuate the handle, squeeze the trigger and pull up and aft on the handle. Effective airplanes 146817c thru 148275e, the emergency harness release handle when actuated will also perform the duties previously accomplished by the drogue lanyard release knob. The handle is protected by a trigger that must be squeezed before the handle may be pulled. When the handle is pulled, the lap belt harnessing, shoulder harnessing, and leg restraint lines will be released. In addition, a cartridge is fired, and the gas pressures liberated are piped to a guillotine which severs the line which connects the drogue chute to the personnel parachute. The guillotine is located on the upper left side of the seat back.

LEG RESTRAINERS

A leg restraint assembly is provided on the seat to hold the occupant's legs in place and to prevent them from flailing during ejection. The leg restraint assembly consists of garters worn by the crewmember, leg restraint lines with lock pins, snubber unit, and shear fitting secured to the floor. The garters are strapped on to the leg just below the knee. The leg

PRESSURE SUIT AND HARNESSING

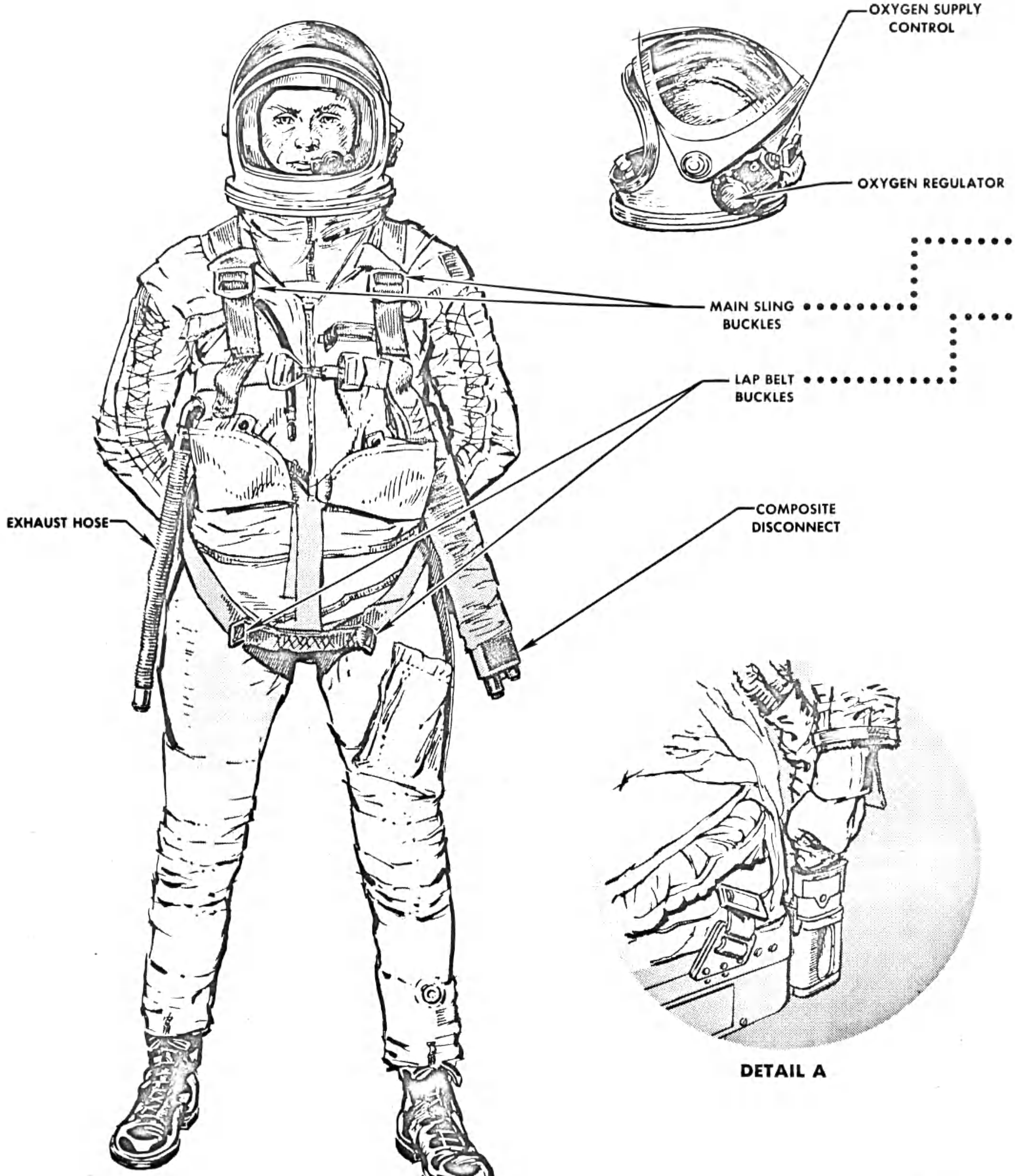


Figure 1-20

EJECTION SEAT

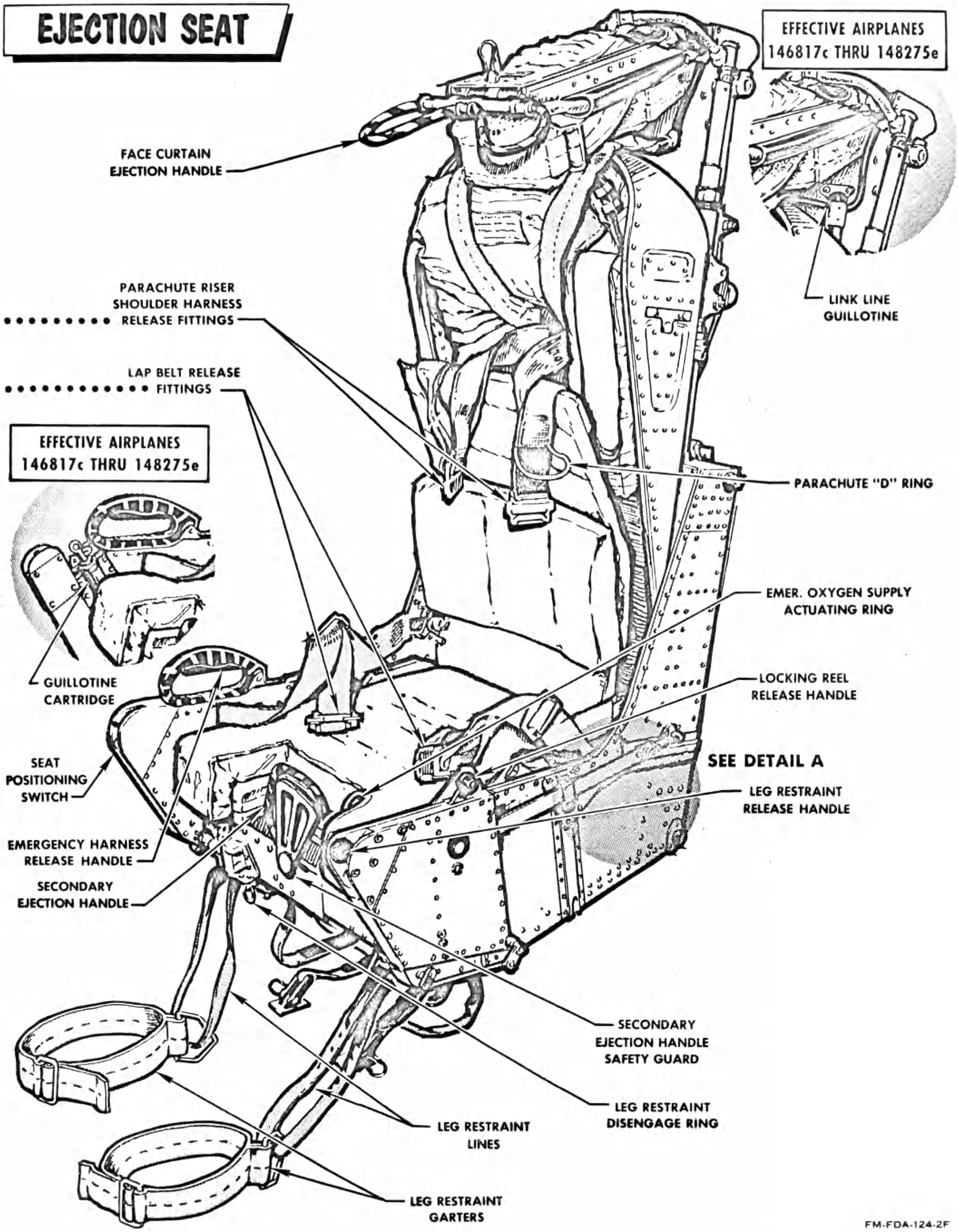


Figure 1-21

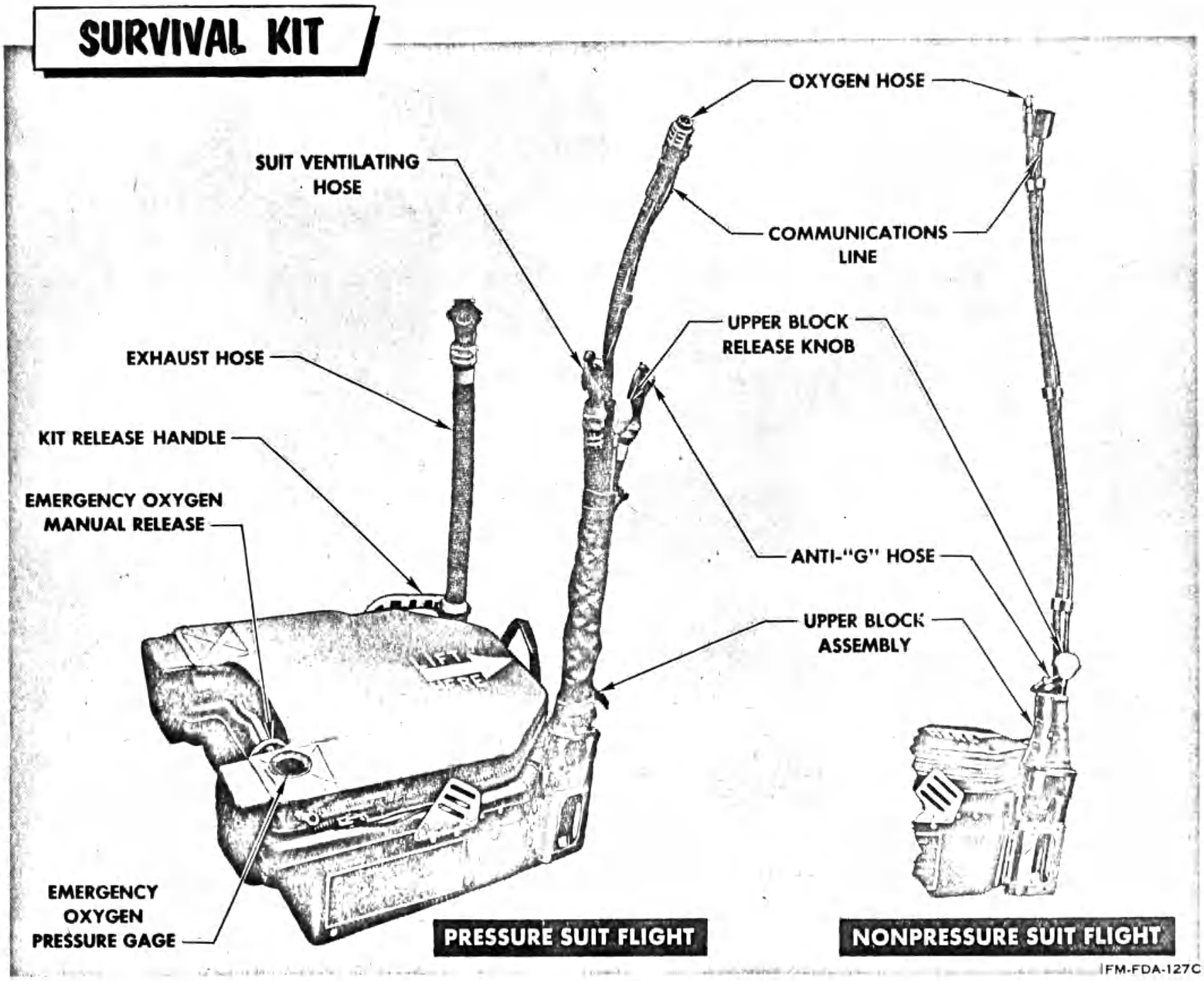


Figure 1-22

lines running from the shear fitting beneath the seat, passes through the snubber unit, through the garter, and then the lock pin on the leg lines, plugs into the leg lock mechanism on the front of the seat pan. When the seat is ejected, the slack in the leg restraint line is taken up by the upward travel of the seat, pulling the occupant's legs to the front face of the seat pan. When all the slack has been removed in the leg restraint lines, the tension of the line will cause the shear fitting to fail. The occupant's legs will be firmly held against the seat pan by the snubbing unit until the harness is released and the occupant is separated from the seat. Leg restraint disengage rings (figure 1-21) located on the face of the seat pan are provided to adjust the amount of slack in the leg restraint lines. This slack may be adjusted by the occupant by pulling out on the appropriate finger ring. This allows more restraint line to be pulled out to provide sufficient slack. To take in excess slack, the occupant need only reach under the seat bucket and pull in the excess restraint line through the snubber unit.

Leg Restraint Release Handle

The leg restraint release handle (figure 1-21) is located on the left forward side of the seat bucket. When the handle is moved to the aft (unlocked) position, the lock pins on the leg lines are released from the leg lock mechanism. This allows the occupant to thread the leg lines back through the garter, enabling him to leave the seat without removing the garters.

CREWMEMBERS INTEGRATED HARNESS

The integrated harness (figure 1-20) is a vest like garment worn over the summer flying suit or a series of web straps worn over the pressure suit. The harness, when used with the C/MBEU/615/PA integrated type parachute, takes the place of a lap belt and shoulder harness. Both of the harness configurations have four buckles for attaching the parachute to the crewmember. The lower two buckles, when connected to the lap belt release fittings, which in turn is fastened to the seat, serves as the lap belt. The upper two buckles, when connected to the parachute riser-shoulder harness release fittings, which in turn is fastened to the locking reel assembly, serves as the shoulder harness. The integrated harness eliminates the need for the crewmember to wear his parachute to and from the airplane, and it also eliminates a separate lap belt and shoulder harness with its inherent limited restraint capabilities.

SURVIVAL KIT

A modified PK-2 survival kit is packed within a two piece fiberglas container (figure 1-22) which, in turn, is attached to the occupant by strap-harnessing. The content of the survival kit is the same as for a normal issue PK-2, but the packing arrangement has been changed to suit the requirements of the container. The following is a list of contents of this kit:

Note

> The emergency provisions included in the PK-2 survival kit are subject to local option and may be altered at the discretion of the area commander.

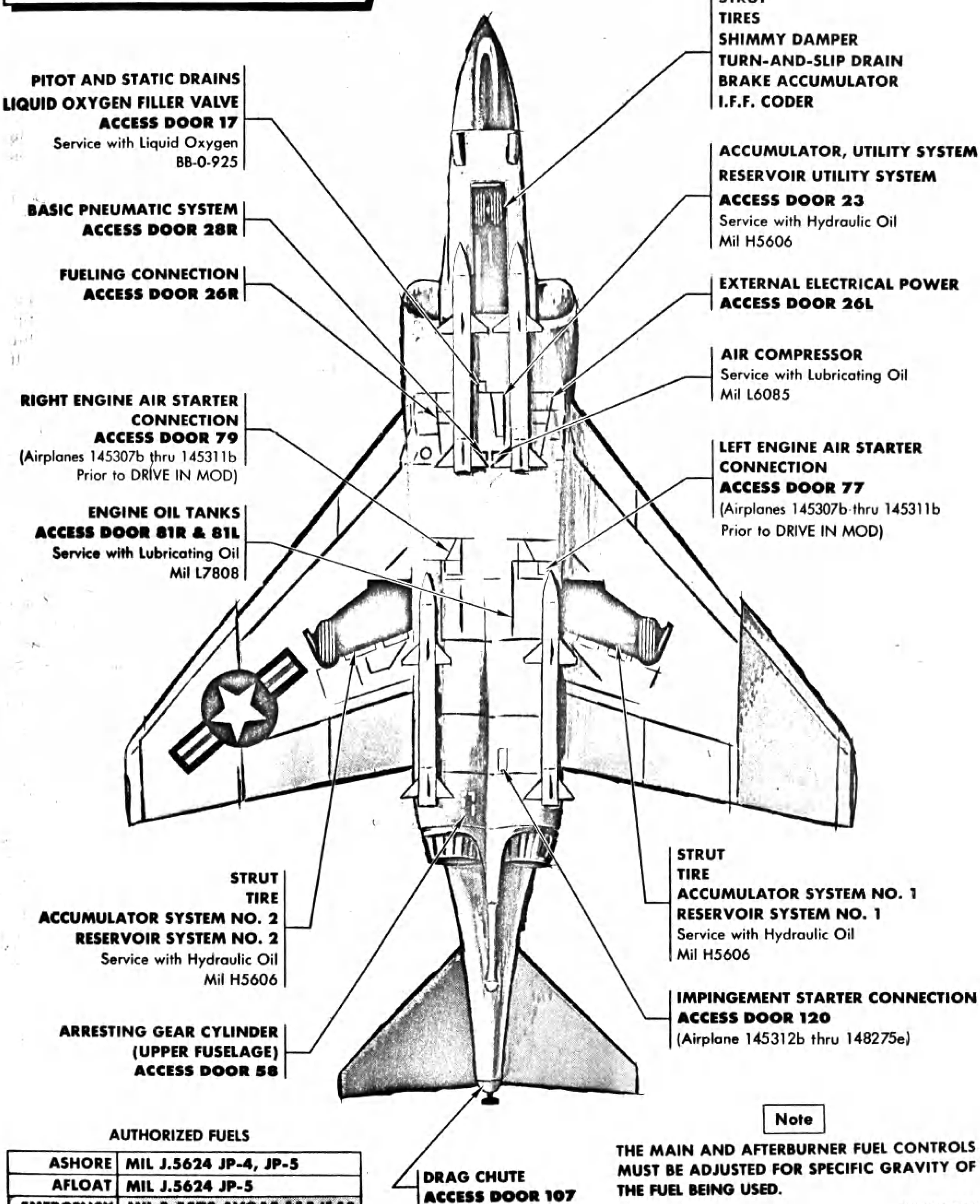
Pararaft with inflation bottle, sleeve type sea anchor and lanyard.
Metallic radar reflector assembly (disassembled).
Solar distillation unit.
De-salter kit (tablets).
Water storage bag.
Signal mirror.
Bailing sponge.
50 feet of nylon line.
2 packs dye marker.
Poncho with reflective surface.
Canned rations.
Can of sunburn ointment.
Emergency code instruction sheets.

All items except the pararaft and its associated gear are packed in a zipper enclosed bag which is attached to the pararaft by a lanyard. Both the pararaft and bag are packed into the survival kit container and the pararaft is attached by a lanyard to the upper half of the container. Aside from the PK-2 equipment packed in the lower half of the container, the upper half contains the emergency oxygen, and pressure suit controller. A receptacle for plugging in the composite disconnect is located in the left rear corner of the survival kit container, and a kit release handle is located on the right rear side of the survival kit container. Pulling up on the kit release handle unlocks the container and actuates the pararaft inflation bottle. The inflating pararaft separates the container; the upper half of the container with emergency oxygen and suit controller remains with the crewmember, and the lower half of the container with pararaft and survival equipment floats free on the end of the lanyard that is attached to the upper half of the container. In the event of an ejection, the pararaft could be inflated prior to entering the water since the kit release handle is accessible while still in the parachute harnessing, and all survival equipment is secured to the crewmember and to each other by lanyards.

AUXILIARY EQUIPMENT

Information concerning the following operational equipment is supplied in Section IV of this handbook: Cockpit air conditioning and pressurization system, cockpit defrosting and anti-icing system, engine anti-icing system, communication and associated electronic equipment, lighting equipment, oxygen system, autopilot, navigation equipment, radar system, armament equipment, refueling systems, and miscellaneous equipment.

SERVICING POINTS



AUTHORIZED FUELS

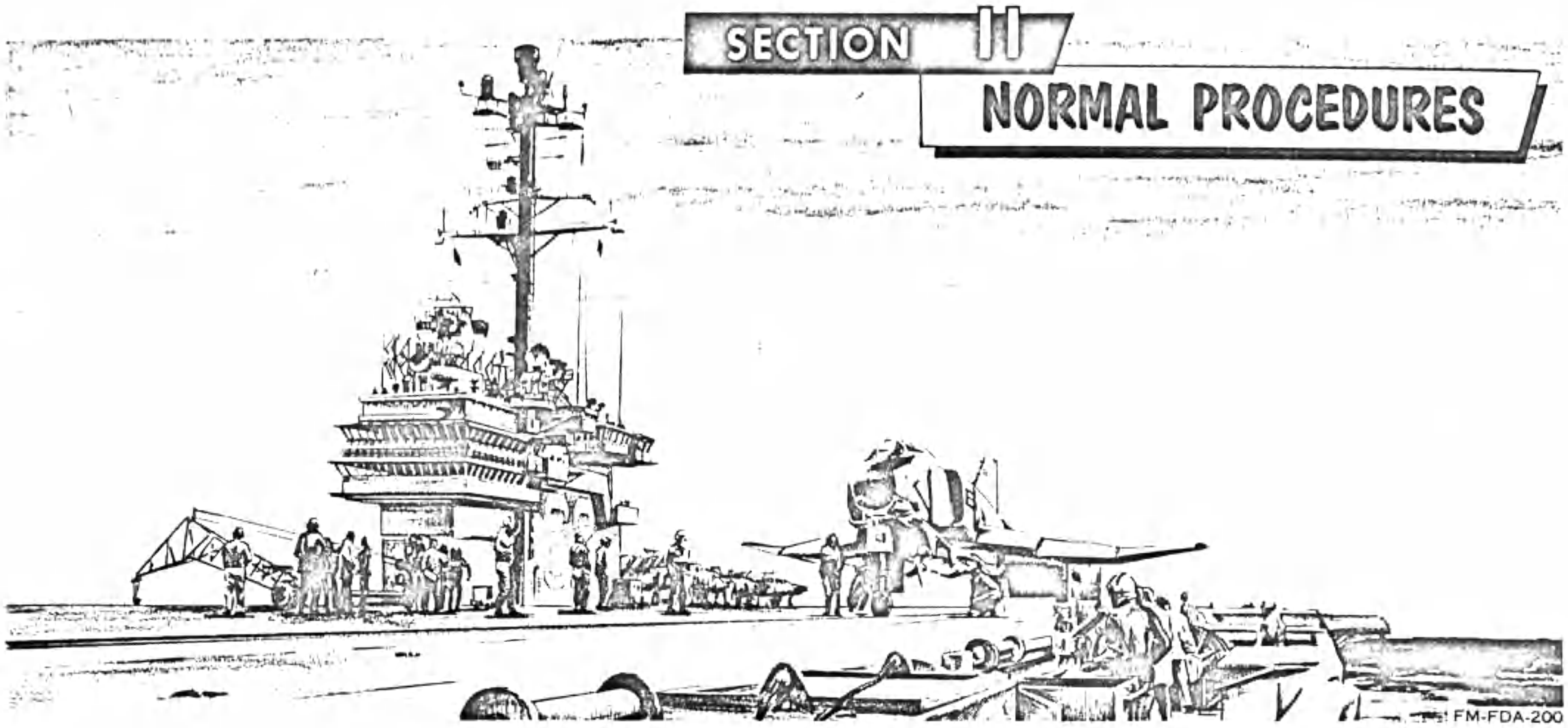
ASHORE	MIL J.5624 JP-4, JP-5
AFLOAT	MIL J.5624 JP-5
EMERGENCY	MIL F-5572 AVGAS 115/145

Note

THE MAIN AND AFTERBURNER FUEL CONTROLS MUST BE ADJUSTED FOR SPECIFIC GRAVITY OF THE FUEL BEING USED.

FM-FDA-126D

Figure 1-23



Note

Since the CADC automatically compensates for static source error to provide a direct reading of calibrated airspeed, all airspeed values within this section will be quoted as CAS. In the event of illumination of the "Static Corr Off" warning light, quoted airspeeds should be corrected for static source error. Refer to Airspeed Position Error Correction charts in Part I of the Appendix.

PREPARATION FOR FLIGHT

Even though this is a two-place airplane the responsibility for the thorough accomplishment of the airplane preflight is primarily the pilot's. However, when both crewmembers are present, the airplane preflight should be made together following an established procedure so as not to overlook any preflight items regardless of responsibility. It is further recommended that a thorough briefing is necessary concerning emergency procedures, crew coordination and cooperation throughout the planned flight.

Note

Dual control airplanes do not require any special interior check or special flight precautions or techniques.

PERSONAL FLYING EQUIPMENT

The flying equipment listed below will be worn on every flight (OPNAV Inst. 3710.7):

1. Currently modified anti-buffet helmet.
2. Oxygen mask.
3. Anti "g" suit.
4. Fire retardant high visibility flight suit.
5. Ankle high laced shoes or boots.
6. MK-3C life vest.
7. Integrated torso harness.
8. Sheath knife.
9. Pistol with tracer ammunition.
10. Flight gloves.
11. Identification tags.
12. Exposure suit (or full pressure suit) on all overwater flights when the water temperature is 59 °F or below, or OAT is 32 °F or below, or the combined air/water temperature is 120 °F or below.
13. Approved personal survival kit.
14. Other survival equipment appropriate to the climate of the area.
15. Full pressure suit of all flights above 50,000 ft., MSL.
16. Flashlight (for all night flights).

All survival equipment will be secured in a manner so that it is easily accessible and will not be lost during ejection or descent.

PREFLIGHT PLANNING

Refer to Appendix I of this publication to determine fuel consumption, correct airspeed, power settings and altitude for the intended flight mission.

BEFORE EXTERIOR INSPECTION

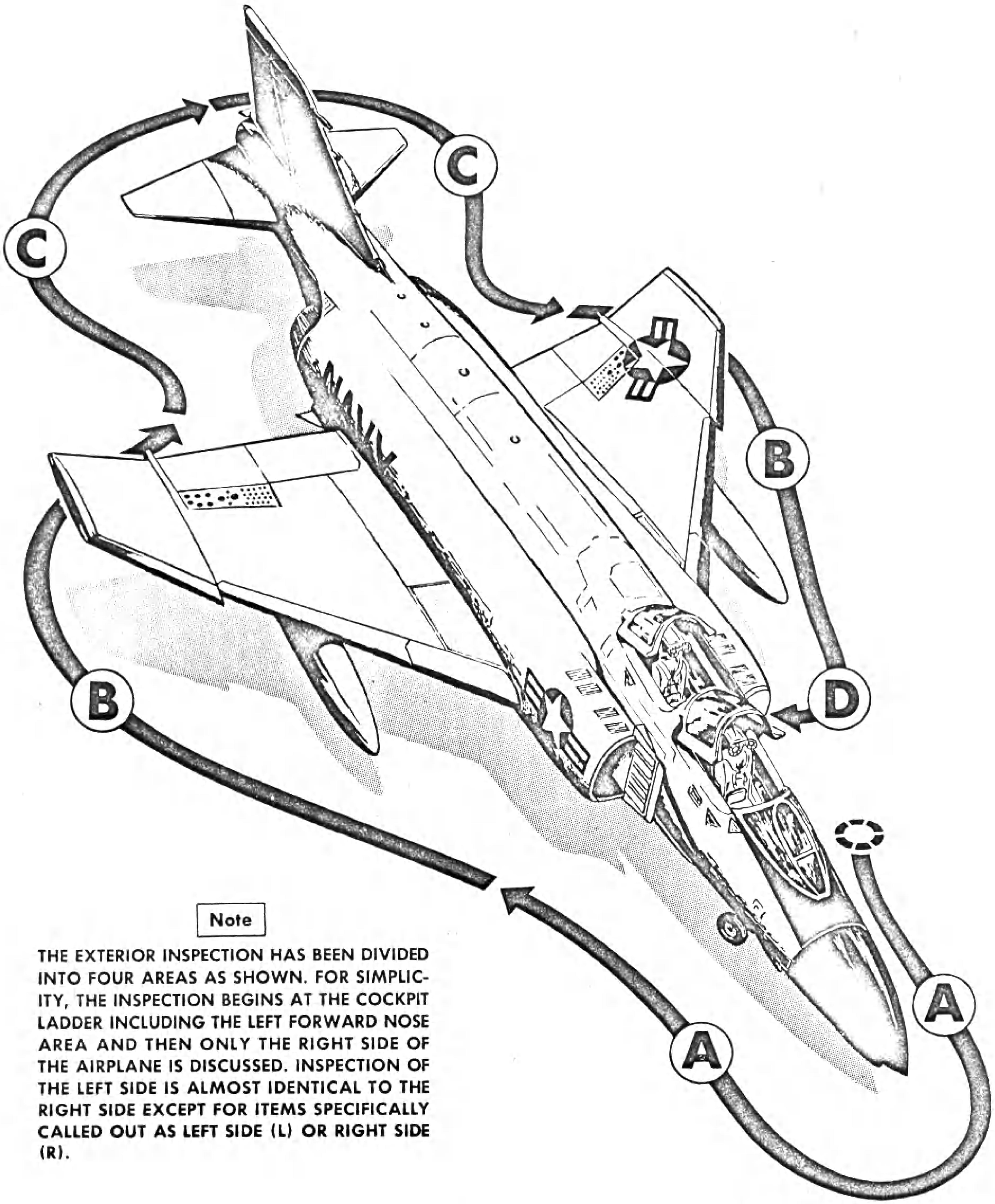
Log Book - CHECK
Check that log book indicates that seat catapult is loaded with personnel cartridge.

OPNAV 3760-2 (White Sheet) - CHECK
Check "white sheet" to determine flight status and servicing.

EXTERIOR INSPECTION

Perform exterior inspection as outlined in figure 2-1.

EXTERIOR INSPECTION



Note

THE EXTERIOR INSPECTION HAS BEEN DIVIDED INTO FOUR AREAS AS SHOWN. FOR SIMPLICITY, THE INSPECTION BEGINS AT THE COCKPIT LADDER INCLUDING THE LEFT FORWARD NOSE AREA AND THEN ONLY THE RIGHT SIDE OF THE AIRPLANE IS DISCUSSED. INSPECTION OF THE LEFT SIDE IS ALMOST IDENTICAL TO THE RIGHT SIDE EXCEPT FOR ITEMS SPECIFICALLY CALLED OUT AS LEFT SIDE (L) OR RIGHT SIDE (R).

FDA-202-1B

Figure 2-1 (Sheet 1)

(A) NOSE

1. GENERAL AREA

A. REFRIGERATION UNIT INTAKE CLEAR
B. RADOME SECURE

2. NOSE GEAR

A. TIRE CONDITION, INFLATION
B. SHIMMY DAMPER SERVICED
C. STRUT CONDITION, PROPER INFLATION
D. GROUND LOCK REMOVED
E. GEAR DOORS SECURE
F. APPROACH LIGHTS SECURE
G. WHEEL WELL CONDITION
H. EMERGENCY FLAP PRESSURE GAGE–2750 PSI TO 3100 PSI
I. WIND DRIVEN TURBINE PRESSURE GAGE–2750 PSI TO 3100 PSI
J. EMERGENCY LANDING GEAR PRESSURE GAGE–2750 PSI TO 3100 PSI
K. EMERGENCY BRAKE AIR PRESSURE GAGE–2750 PSI TO 3100 PSI
L. WHEEL BRAKE ACCUMULATOR PRESSURE GAGE 1500 ± 50 PSI

3. FORWARD FUSELAGE

A. ANGLE OF ATTACK PROBE COVER REMOVED (R)
B. PROBE SECURE (R)
C. ACCESS DOORS SECURE
D. INTAKE DUCT COVER REMOVED

(B) CENTER FUSELAGE AND WING

1. GENERAL AREA

A. CONDITION OF WING AND CENTER FUSELAGE
B. ACCESS DOORS SECURE

2. WING

A. WING FLAPS AND CONTROL SURFACES CHECK
B. EXTERNAL TANKS SECURE (IF CARRIED)
C. WING FOLD JURY STRUT REMOVED
D. NAVIGATION AND JOIN UP LIGHTS SECURE
E. AIR TURBINE DOOR SECURE L (TOP)

(C) AFT FUSELAGE

1. GENERAL AREA

A. GENERAL CONDITION
B. ACCESS DOORS SECURE
C. PITOT COVER REMOVED
D. COLLISION LIGHT SECURE
E. ENGINE ACCESS DOORS (96 L&R) SECURE
F. NOZZLE COVER REMOVED, NOZZLE CONDITION
G. ARRESTING HOOK UPLOCK REMOVED
H. ARRESTING HOOK CONDITION, SECURE
I. STABILATOR AND RUDDER CHECK
J. NAVIGATION LIGHT SECURE
K. DRAG CHUTE DOOR SECURE

(D) UNDERSIDE OF FUSELAGE

1. GENERAL AREA

A. GENERAL CONDITION
B. EMERGENCY CANOPY AIR GAGES (ACCESS DOOR #17) 2750 PSI TO 3100 PSI
C. OXYGEN FILLER CAP SECURE, VALVE IN BUILD-UP POSITION (ACCESS DOOR #17)
D. PNEUMATIC SYSTEM AIR GAGE–2750 PSI TO 3100 PSI (ACCESS DOOR #28R)
E. UTILITY SYSTEM ACCUMULATOR AIR GAGE 1000 ± 50 PSI (ACCESS DOOR #23)

2. MAIN GEAR AND GEAR WELL

A. WHEELS CHOCKED
B. TIRE CONDITION, INFLATION
C. STRUT CONDITION, PROPER INFLATION
D. GROUND LOCK REMOVED
E. GEAR DOORS SECURE
F. SPEED BRAKE SAFETY SWITCH–NORMAL, GROUND FUELING SWITCH OFF (R)
G. #2 P.C.S. ACCUMULATOR PRESSURE GAGE– 1000 ± 50 PSI(R)
H. #1 P.C.S. ACCUMULATOR PRESSURE GAGE– 1000 ± 50 PSI (L)
I. #1 P.C.S. RESERVOIR ACCUMULATOR PRESSURE GAGE– 2000 ± 50 PSI (L)
J. SPEED BRAKES CONDITION, GROUND LOCKS REMOVED

FDA-202-2D

Figure 2-1 (Sheet 2)

MINIMUM CREW REQUIREMENTS

The minimum crew for all flights except special flights directly concerned with research, development, evaluation and ferry shall be the pilot and RIO, or two pilots.

FLIGHT RESTRICTIONS

Refer to Section V for detailed airplane and engine limitations.

WEIGHT AND BALANCE

Refer to Section V for weight limitations. For loading information, refer to Handbook of Weight and Balance Data, AN 01-1B-40. Before each flight, check for the following:

1. Check take-off and anticipated landing gross weight and balance.
2. Check Form F for weight and balance clearance.

CHECK LISTS

The check lists in figure 2-2 are reproductions of the TAKE-OFF and LANDING check lists located on the pilot's and radar observer's instrument panels, and SOLO FLIGHT check list located on the right side of the radar observer's cockpit. These check lists are only used as a reminder and list, in general terms only what to check; the pilot must be thoroughly familiar with the procedures outlined in this handbook so as to know how these items should be checked.

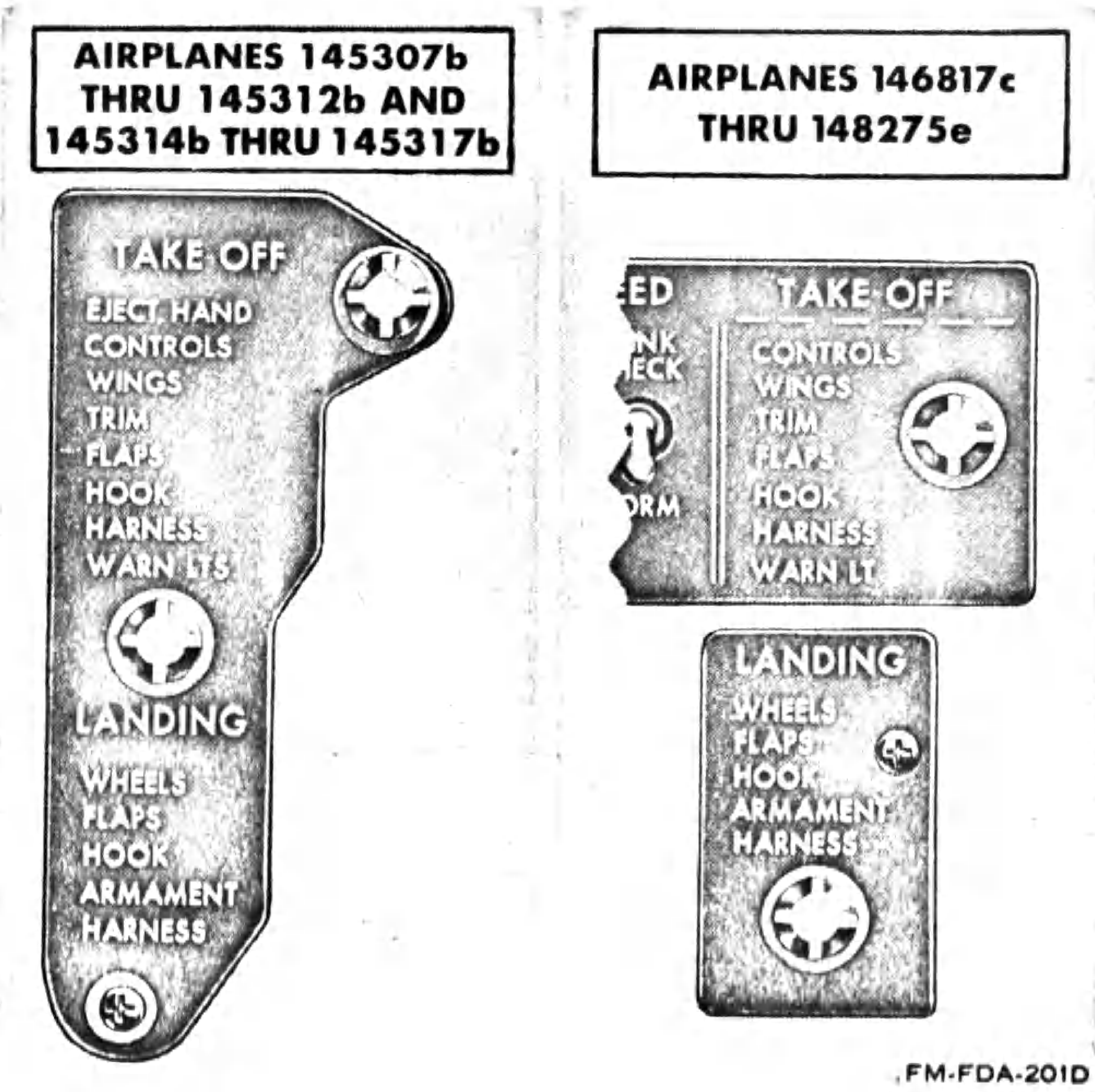


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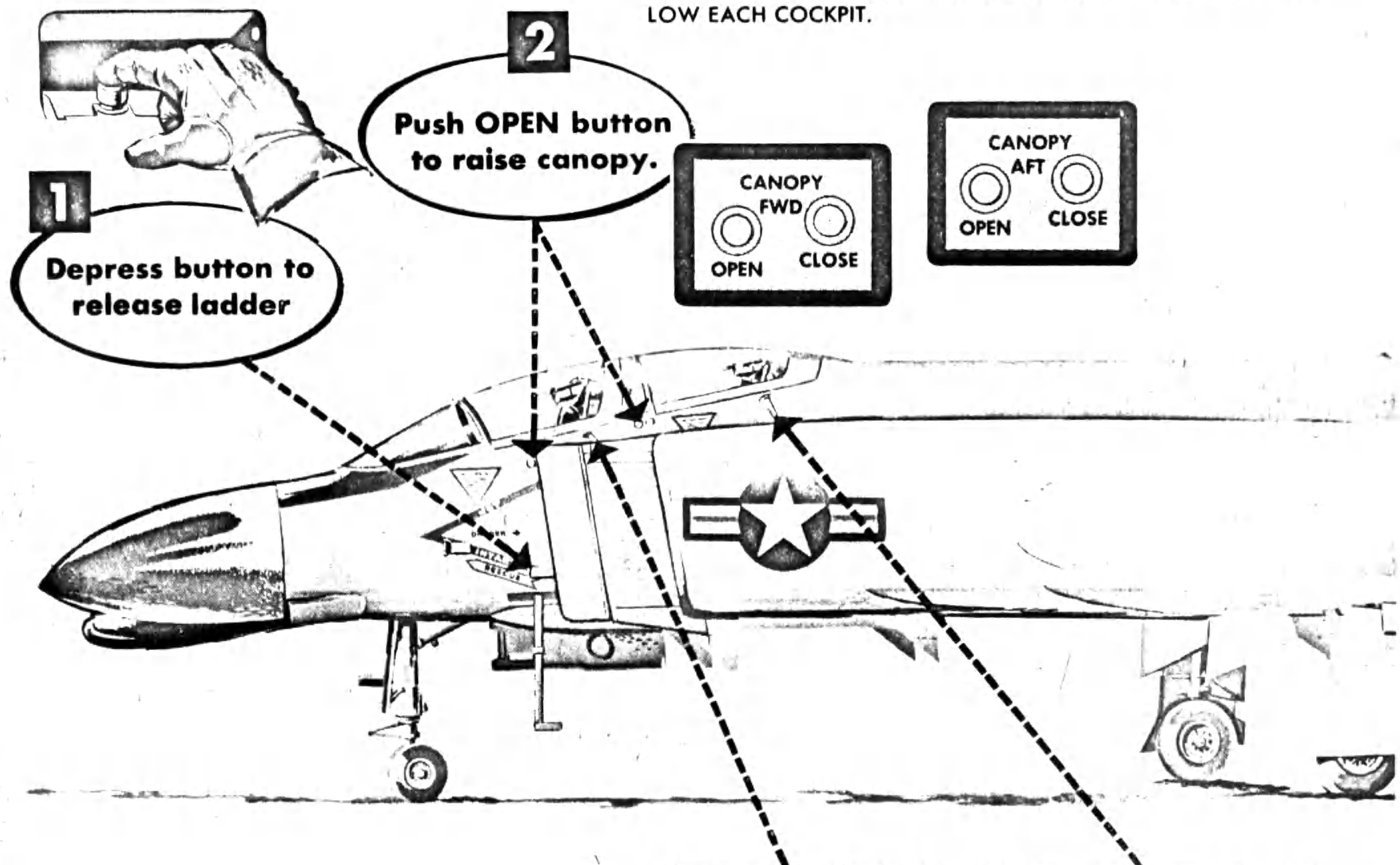
Figure 2-2

ACCESS TO COCKPIT

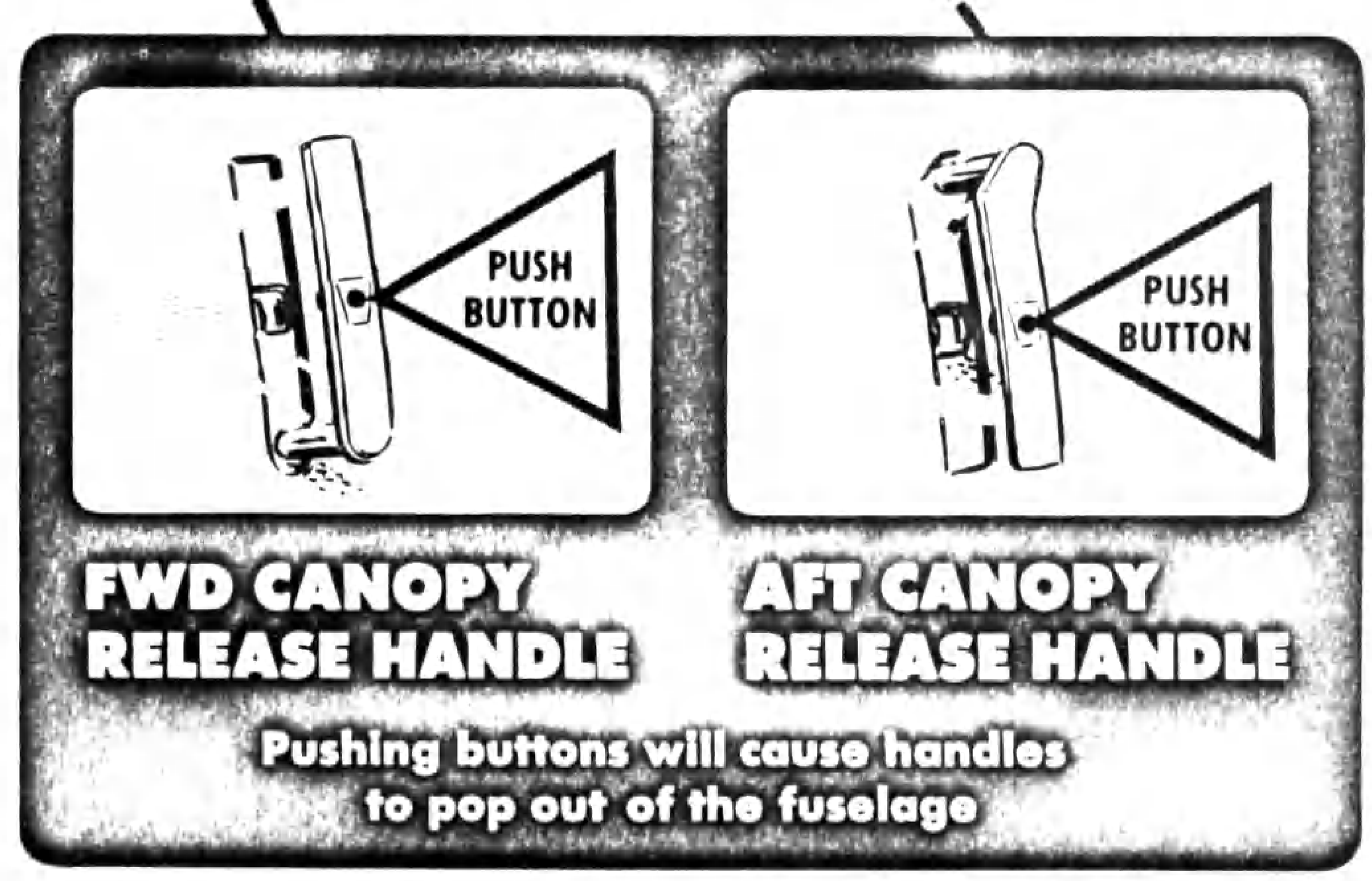
Refer to figure 2-3 for normal entry into cockpits.

ACCESS TO COCKPIT

NORMAL ACCESS TO THE COCKPIT IS MADE VIA A BUILT-IN BOARDING LADDER AND TWO KICK-IN STEPS LOCATED ON THE LEFT SIDE OF THE FORWARD FUSELAGE. THE BOARDING LADDER RELEASE BUTTON IS LOCATED IN THE BOTTOM KICK-IN STEP. THE CANOPY OPERATING BUTTONS ARE LOCATED BELOW EACH COCKPIT.



TO MANUALLY OPEN THE CANOPIES, PUSH THE **OPEN** BUTTONS ON EACH CANOPY CONTROL. DEPRESS PUSH TYPE LATCH ON THE MANUAL RELEASE HANDLES. ROTATE THE **FORWARD** CANOPY MANUAL RELEASE HANDLE **COUNTERCLOCKWISE**. ROTATE THE **AFT** CANOPY MANUAL RELEASE HANDLE **CLOCKWISE**. AFTER THE CANOPIES HAVE BEEN UNLOCKED THEY CAN BE LIFTED OPEN MANUALLY.



CAUTION

When raised manually, the canopy must be held or propped up in the open position.

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Figure 2-3

SEAT AND CANOPY SAFETY PINS AND INSPECTION POINTS

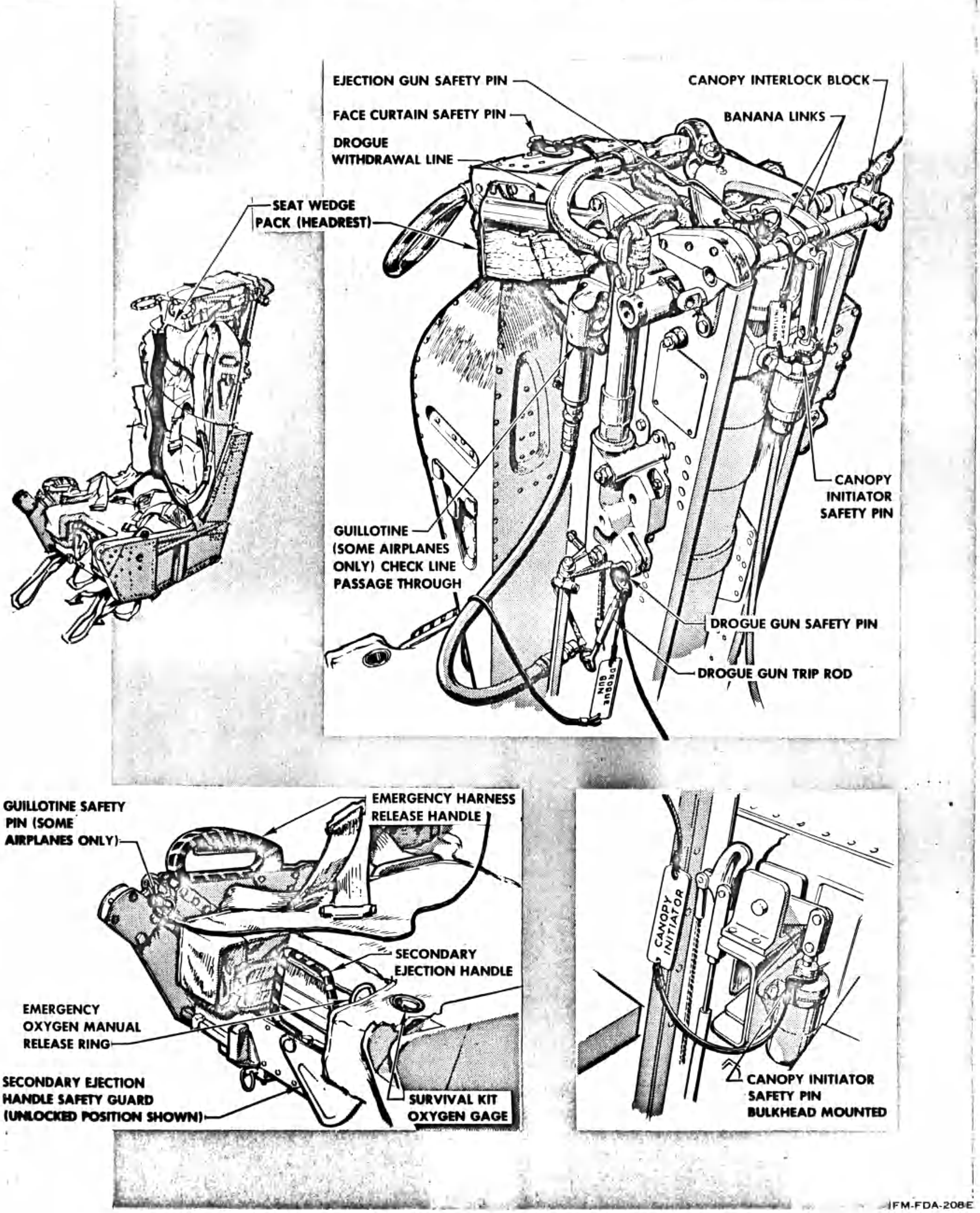


Figure 2-4

AFT COCKPIT INTERIOR CHECK FOR SOLO FLIGHT

1. Seat and canopy safety pins - INSTALLED
2. Circuit breakers - IN

Note

Two circuit breaker panels are located on the port side of the cockpit. These circuit breakers control much of the electrical equipment that is essential to flight and are inaccessible to the pilot while in flight.

3. Navigation computer function switch - OFF
4. Radar master switch - OFF
5. Pressure suit vent air valve - OFF
6. Oxygen supply lever - OFF
7. Cockpit light switches - OFF
8. Seat harness - STOWED
9. All loose gear - STOWED
10. Canopy - LOCKED

BEFORE ENTERING COCKPIT

1. Harness assembly - CHECK
   Check emergency harness release handle down. Check if pin is installed at reel assembly. Check pins that secure lap belt and survival kit to seat are in place.
2. Composite disconnect lower block - CHECK
   Check lower block locking indicator (yellow metal tab) on bottom of lower block is tight.
3. Canopy interlock block - CHECK
   Check canopy interlock block in place, and interlock line secured to canopy.
4. Drogue withdrawal line - CHECK
   Check drogue withdrawal line (in wire braid sleeve) passes over and lays on top of all other lines.
5. Scissor shackle tie-down thread - CHECK
   Check that scissor shackle tie-down thread passes below the flap securing pin, and does NOT pass through the loop of the drogue flap.
6. Banana links - CHECK
   Check links attached to sear pin.
7. Seat latch nut - CHECK
   Check seat latch nut finger tight and safety wired.
8. Parachute withdrawal line quick disconnect - CHECK
   Check connection of parachute withdrawal line quick disconnect.
9. Seat wedge pack (headrest) - CHECK
   Check seat wedge pack (headrest) has approximately one (1) inch side play.
10. Drogue gun trip rod - CHECK
    Check drogue gun trip rod on left side of seat secure to airplane structure.
11. Safety banner - REMOVED
    Check safety banner and all five safety pins - face curtain, ejection gun, canopy initiator (seat mounted), drogue gun, canopy initiator (bulkhead mounted) - and both dust covers (composite disconnect, exhaust port), are removed.

WARNING

- When removing or checking for the removal of the face curtain safety pin, make sure that the safety pin shank has been removed from the hole. The safety pin collar has been known to separate from the pin shank upon attempted safety pin removal, leaving the pin shank in the hole and the face curtain safetied.

- Do NOT pull down on the face curtain ejection handle. Seat and canopy ejection systems are fully armed when safety pins are removed.

CONTINUED ON NEXT PAGE

BEFORE ENTERING COCKPIT CONTINUED

12. Secondary ejection handle safety guard - CHECK
    Check alternate ejection handle safety guard in front of the seat bucket is in the vertical (up) position.
13. Survival kit oxygen gage - CHECK
    Check survival kit oxygen gage within limits.
14. Landing gear handle - DOWN
    Manually check that landing gear handle is DOWN.
15. Generator control switches - OFF

INTERIOR CHECK

1. Time delay mechanism trip rod - CHECK
   Check time delay mechanism trip rod on right side of seat secured to airplane structure.
2. Composite disconnect intermediate block - CHECK
   Check yellow tubing projects from oxygen connector on top of intermediate block.
3. Composite disconnect - INSERTED and LOCKED
   Check upper block properly inserted and locked into intermediate block by exerting an upward pull on block assembly after composite disconnect release knob is locked to cable housing.

Note

The composite disconnect should be carefully inserted with a downward force parallel to the seat ejection plane. After the composite disconnect is fully inserted, push down on composite disconnect release knob to lock knob to cable housing and prevent the release knob from laying over and dangling.

4. Pressure suit lines - CONNECTED
   Check pressure suit lines properly connected to the survival kit.

WARNING

When the pressure suit is being worn without an anti-G garment, the anti-G hose must not be connected and the corresponding port on the pressure suit must be capped. Explosive decompression will result upon ejection if the anti-G hose is connected. In the event of ejection over water or ditching, the water tight integrity of the pressure suit will be nullified.

5. Pressure suit vent air valve - ON
   Adjust air flow as desired.
6. Oxygen supply lever - ON
   Check for normal flow.

WARNING

Do not pull emergency oxygen manual release prior to actual use. If the emergency oxygen manual release is actuated prior to intended use, the pressure reducer manifold may not prevent emergency oxygen from flowing to the suit controller, and/or oxygen regulator. In the event this happens, the crewmember has no way of knowing how much, or if any emergency oxygen remains and has no way of replenishing the depleted supply.

CONTINUED ON NEXT PAGE

INTERIOR CHECK CONTINUED

7. Leg restraint lines - CONNECTED
   Pass leg lines through garters and plug into seat pan.

WARNING

It is imperative that leg restraint system be hooked up at all times during flight to ensure legs are pulled aft upon ejection. This will prevent leg injury and enhance seat stability by preventing legs from flailing following ejection. An unhooked leg restraint system necessitates pulling legs aft against seat to preclude hitting canopy bow. This action will cause spine to flex and will increase the possibility of spinal injury during ejection.

8. Harnessing - FASTENED
   Check parachute riser-shoulder harness release fittings fastened to main sling buckles, check lap belt release fittings fastened to lap belt buckles. Check locking reel operation.

CAUTION

Make sure that the automatic harness release assembly is securely fastened to the seat. The pins must be in their proper receptacles, one pin on each side of the bucket seat and one pin on the harness locking reel assembly. The emergency harness release handle must be down.

9. Rudder pedals - ADJUST
   Adjust rudder pedals by use of hand crank located on the pedestal panel.
10. Stick grip - CHECK
    Check stick grip firmly attached.
11. Intercom control panel - CHECK
    a. Volume selector knob - AS DESIRED
    b. Function selector switch - HOT MIC
12. Fuel control panel - CHECK
    a. External tank jettison switch - NORMAL, guard closed.
    b. Buddy fill switch - STOP FILL
    c. Internal wing transfer switch - NORMAL (some airplanes - AUTO)
    d. Refuel selection switch - ALL TANKS, guard closed.
    e. Refuel probe switch - RETRACT
    f. Boost pump check switches - NORMAL
    g. External transfer switch - OFF - (ON if tanks are carried)
    h. Internal wing dump switch - NORMAL
    i. Wing transfer pressure - NORMAL
13. Emergency hydraulic pump lever - IN PLACE AND SECURE
14. Autopilot control panel - CHECK
    a. Damper (Stab Aug) switch - OFF or STAB AUG
    b. Autopilot (AFCS) switch - OFF or AFCS
    c. Altitude switch - OFF or ALT
    d. Mach switch - OFF or MACH
15. Wing flap switch - UP
16. Wing flap emergency pull handle - UP
17. Communication antenna selector switch - SET AS REQUIRED
18. Engine anti-icing switch - NORMAL
19. Throttles - CHECK
    Check throttles for full OFF.
20. Master lights switch - OFF
21. Speed brake switch - IN
22. Throttle friction lever - SET AS DESIRED
23. Engine master switches - OFF
24. Engine start switch - NEUTRAL
25. Emergency speed brake switch - GUARD DOWN

CONTINUED ON NEXT PAGE

INTERIOR CHECK CONTINUED

26. Liquid oxygen gage - CHECK
    Check oxygen quantity is sufficient for flight.
27. Emergency canopy release lever - IN PLACE AND SECURE
28. Drag chute handle - IN PLACE AND SECURE
29. Landing gear handle - DOWN
    Manually check landing gear handle down.
30. Missile power switch - OFF
31. Arming switch - SAFE
32. Bomb master switch - OFF
33. Missile jettison selector switch - OFF
34. Accelerometer - SET
35. Altimeter - SET
36. Vertical velocity indicator - CHECK
37. Clock - SET
38. Arresting gear control handle - UP
39. Generator control switches - OFF
40. T-249A power switch - OFF
41. Emergency vent knob - IN
42. Rain removal switch - OFF
43. Pitot heat switch - OFF
44. Communications function selector switch - STANDBY
45. TACAN function selector switch - STANDBY
46. Circuit breakers - IN
47. Cockpit temperature control panel - CHECK
    a. Heat knob - SET AS DESIRED
    b. Temperature control switch - AUTO
48. Instrument panel emergency floodlights switch - OFF
49. IFF master knob - OFF
50. Cockpit lights - OFF
51. Exterior lights - OFF
52. Spare lamps - CHECK
53. Flashlight, charts and reference material - CHECK

WITH EXTERNAL ELECTRICAL POWER CONNECTED

CAUTION

Do not place generator control switch to the EXT position until external power has been connected and has had time to reach rated voltage and frequency.

54. Generator control switches - EXT ON
55. Seat - ADJUST
56. Interphone system - CHECK
    Check the interphone system between radar observer and ground crew.
57. Boost pumps and engine fuel shutoff valve - CHECK
    Observe boost pump pressure indicators while actuating boost pump check switches one at a time. Normal pressure on side being checked indicates engine fuel shutoff valve open and boost pump running. Concurrent pressure on other indicator indicates other valve faulty (not properly closed). Lack of pressure on side being checked indicates faulty valve (not properly open) or pump inoperative.
58. Flap position indicator - UP
    Check flap position indicator corresponds with flap position.
59. Landing gear indicators - CHECK
    Check landing gear position indicators indicate gear down.
60. Fire warning light - CHECK
    Depress the fire check button and note fire warning lights illuminated
61. Fuel quantity gage - CHECK
    Actuate fuel gage test button (some airplanes) and note needle dropping toward zero or actuate feed tank check switch (other airplanes) and check fuel gage reads feed tank fuel (approx. 1,900 lbs.).

CONTINUED ON NEXT PAGE

INTERIOR CHECK CONTINUED

62. Fuel quantity - CHECK
    Check fuel quantity indicators against known fuel quantity.
63. Master lights switch - ON
64. Warning lights - CHECK
    Depress warning light test switch and note master caution light, warning lights panel, arresting hook warning light and landing gear warning lights are illuminated. Check warning lights dimming circuit by holding warning lights test button depressed and rotating instrument panel lights control knob from OFF to BRIGHT. Warning lights should dim and revert to bright when knob is returned to OFF.
65. Cockpit lights - CHECK
    If conditions warrant, check operation of all cockpit lights and adjust desired brilliance.
66. Exterior lights - CHECK
    Check navigation lights in STEADY, FLASH, BRIGHT and DIM positions. Check anti-collision and position lights. Receive acknowledgement from ground crew that these lights are operational.
67. Radio equipment - CHECK
    Check communication and navigation equipment for proper operation and frequency.
68. Flight instruments - CHECK
69. ADI pitch indicator knob - SET FOR TAKE-OFF
70. Compass system controller - SET
    a. Latitude compensator - SET
    b. Mode switch - SLAVED AND SYNCHRONIZED
    c. Sync. button - PUSH
    d. Sync. indicator - CHECK
71. IFF master selector switch - STDBY

INTERIOR CHECK-NIGHT FLIGHTS

1. Master lights switch - ON
2. Cockpit lights - AS DESIRED
3. Exterior lights - AS REQUIRED
   Check navigation, join-up, anti-collision and fuselage lights in the BRIGHT, DIM, STEADY, and FLASH positions.
4. Availability and operation of flashlights - CHECK

BEFORE STARTING ENGINES

1. Wheels - CHOCKED
2. Fire bottle - MANNED
3. Intake and exhaust areas - CLEARED
4. Intercom contact with ground crew - ESTABLISHED
5. External air supply - CONNECTED AND PRESSURE UP (75 PSI)

WARNING

- Suction at the intake is sufficient to kill or severely injure personnel drawn into or pulled suddenly against the duct.
- Danger areas aft of the airplane are created by high exhaust temperature and velocities. The danger increases with afterburner operation. See figure 2-5.

Whenever practicable, start and run up engines on paved surfaces to minimize the possibility of foreign objects being drawn into the compressor with resultant engine damage. Start the engines with the nose into or at right angles to the wind as exhaust temperatures may aggravated by tail wind.

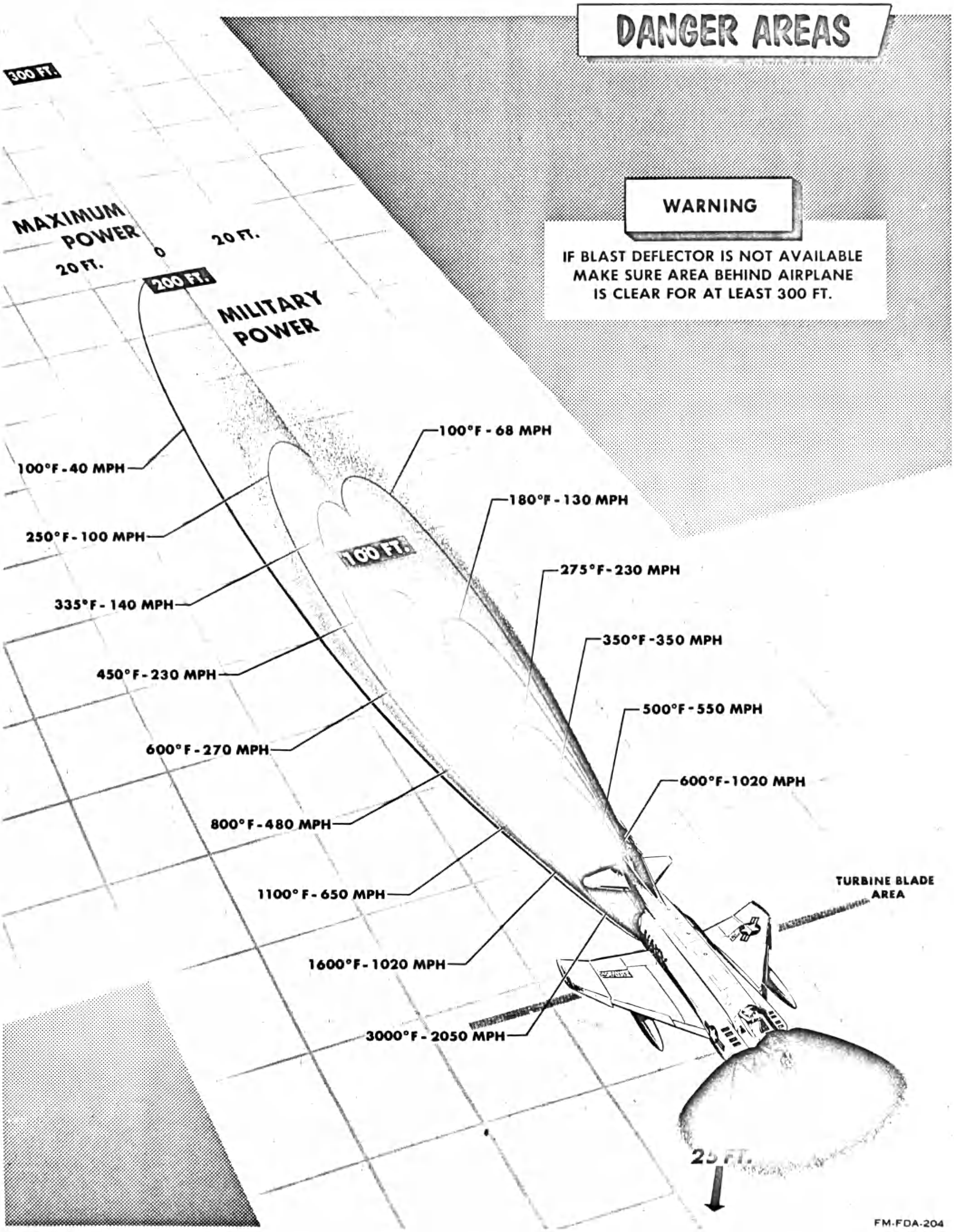

DANGER AREAS
WARNING
IF BLAST DEFLECTOR IS NOT AVAILABLE MAKE SURE AREA BEHIND AIRPLANE IS CLEAR FOR AT LEAST 300 FT.
300 FT.
MAXIMUM POWER
20 FT.
0
20 FT.
200 FT.
MILITARY POWER
100 FT.
100°F - 68 MPH
180°F - 130 MPH
275°F - 230 MPH
350°F - 350 MPH
500°F - 550 MPH
600°F - 1020 MPH
100°F - 40 MPH
250°F - 100 MPH
335°F - 140 MPH
450°F - 230 MPH
600°F - 270 MPH
800°F - 480 MPH
1100°F - 650 MPH
1600°F - 1020 MPH
3000°F - 2050 MPH
TURBINE BLADE AREA
FM-FDA-204
R

Figure 2-5

STARTING ENGINES

Note

The following procedure establishes the right (No. 2) engine as being started first. This procedure was adopted in order to ascertain that both utility hydraulic system pumps are operating. The right engine pump delivers approximately 2800 psi at idle rpm, and the left engine pump delivers approximately 3000 psi at idle rpm. Therefore the single needle utility hydraulic system indicator cannot be used to determine pump operation unless the right (No. 2) engine is started first.

CAUTION

- Do not start engines with flaps extended. With flaps extended, the BLC ducts are open, and the loss of engine bleed air while attempting to start the engines may result in a hot or false start.
- Effective airplanes 145307b thru 145311b, (prior to the F4H Drive-in Modernization Program) electrical connection from the airplane starter circuit external receptacle to the gas turbine ground power unit is to be made prior to starting to insure automatic shutoff of the air supply at the correct starter cutout speed. Hand signals or ground controlled starts are not recommended for initiation and/or shutoff of the ground air supply. Time delay inherent in this method will result in overspeed of the starter, causing progressive damage and/or failure. Failure of a starter due to overspeed could be hazardous to operating personnel. If automatic shutoff of the air supply cannot be achieved due to a malfunction of equipment, pilots and crewmen are cautioned that the starter air supply must be manually shutoff at a speed not to exceed 47% engine rpm. The control valve in the gas turbine ground power unit will not shutoff the air supply automatically unless the unit is connected electrically to the airplane starting circuit.

1. Throttles - OFF
2. External compressed air source - CHECK
   Check proper external air source is connected to starter system and that sufficient air pressure is available. Airplane 145311b requires a 3.3:1 pressure ratio gas turbine compressor starter unit, GTC 85 or equivalent. Airplanes 145312b thru 148275e require a 5.5:1 pressure ratio gas turbine compressor starter unit, RCPP-105 or equivalent. Check to insure connection of gas turbine ground power supply remote control cable.
3. Right engine master switch - ON
4. Engine start switch - RIGHT

CAUTION

If there is no indication of engine rpm within 15 seconds, or no indication of oil pressure within 30 seconds after start cycle begins, shut down immediately and investigate.

CONTINUED ON NEXT PAGE

STARTING ENGINES CONTINUED

5. At 11% rpm, right engine ignition button - DEPRESS
   At approximately 11% rpm, depress right engine ignition button and simultaneously advance the throttle to idle.

Note

The engine usually fires at approximately 13.5 to 16% rpm with a fuel flow of 500 to 800 pph.

CAUTION

If the engine does not light off, by the time fuel flow reaches 500-800 pph or within 15 seconds after fuel flow or pressure is indicated, chop throttle to full OFF position, release ignition button and disengage starter switch.

6. Release ignition button when light-off is indicated by a sudden increase in EGT.

CAUTION

- If engine does not continue to accelerate after light-off, discontinue start.
- The engine start buttons on Airplane 145311b will stay down when depressed. When the engine reaches 47%, the starter should be de-energized to prevent overspeeding and disintegration of the starter turbine.

7. Start switch - NEUTRAL
   When the engine is operating at a self-sustaining rpm (usually about 45%) move the starter switch to the neutral position.
8. Exhaust temperature gage - CHECK WITHIN LIMITS
   Exhaust temperature should not exceed the maximum starting temperature limits during transition period of idle range.

CAUTION

- At no time should EGT exceed 1000°C nor should the starting temperature exceed maximum starting limits.
- With only one engine in operation, do not move control stick (surface controls) excessively. If the stick is moved excessively with hydraulic pressure on only one side of the tandem power cylinders, the fluid that is in the other side of the cylinder is forced back through the return line to the reservoir, filling the reservoir, and forcing the excess fluid overboard. The seals within the tandem power cylinders may also be damaged due to the ingesting and expelling of air and lack of lubrication. The power control hydraulic systems must be reserviced and checked.

Note

After the engine reaches idle rpm and stabilizes, the EGT will recede to a temperature of approximately 320° to 400°C.

CONTINUED ON NEXT PAGE

STARTING ENGINES CONTINUED

9. Fuel flow indicator - CHECK
   Fuel flow will indicate 500 to 800 lbs. per hour during starting and 1100 to 1300 lbs. per hour at idle rpm.

Note

Fuel consumed while starting engines is approximately 65 pounds.

CAUTION

If fuel flow is appreciably less than 500 lbs. per hour, a false start will likely result. If fuel flow is in excess of 800 lbs. per hour, a hot start will likely result. Refer to Section V, Engine Limitations.

10. Oil pressure gage - CHECK
    Check oil pressure 20 psi minimum at idle rpm.

Note

- With the right engine started, the No. 2 power control hydraulic system pressure gage and the utility hydraulic system pressure gage should read within normal. The hydraulic pressure warning light will remain illuminated until the other engine is started and all four hydraulic pumps (P.C. No. 1, P.C. No. 2, and utility) are operating properly.

- In the event the throttles cannot be returned OFF, the engine may be shut down from any throttle setting by placing the respective master switch in the OFF position. This will close the corresponding fuel shutoff valve, thus depriving the engine of fuel.

11. After any wet start or false start, allow one minute or longer for the combustion system to drain before starting the engine.
12. Start left engine as per items 1 thru 11.
13. Generator control switches - GEN ON
    Position both generator control switches from the EXT ON position to GEN ON position after both engines have reached idle speed, and check that generator warning lights are extinguished. If bus tie light illuminates, cycle one generator control switch to extinguish light.

Note

- Non-start or abnormal starts shall be logged on the yellow sheet (OPNAV FORM 3760-2).

- Fuel consumption at idle rpm is approximately 42 ppm.

ENGINE GROUND OPERATION

After satisfactory starts are accomplished, the engines do not require any warm-up time prior to placing the throttles in any position.

WARNING

Do not lower flaps when wings are folded, and engines are running. With wings folded, a boundary layer control duct is exposed and the hot engine bleed air may cause injury to ground personnel.

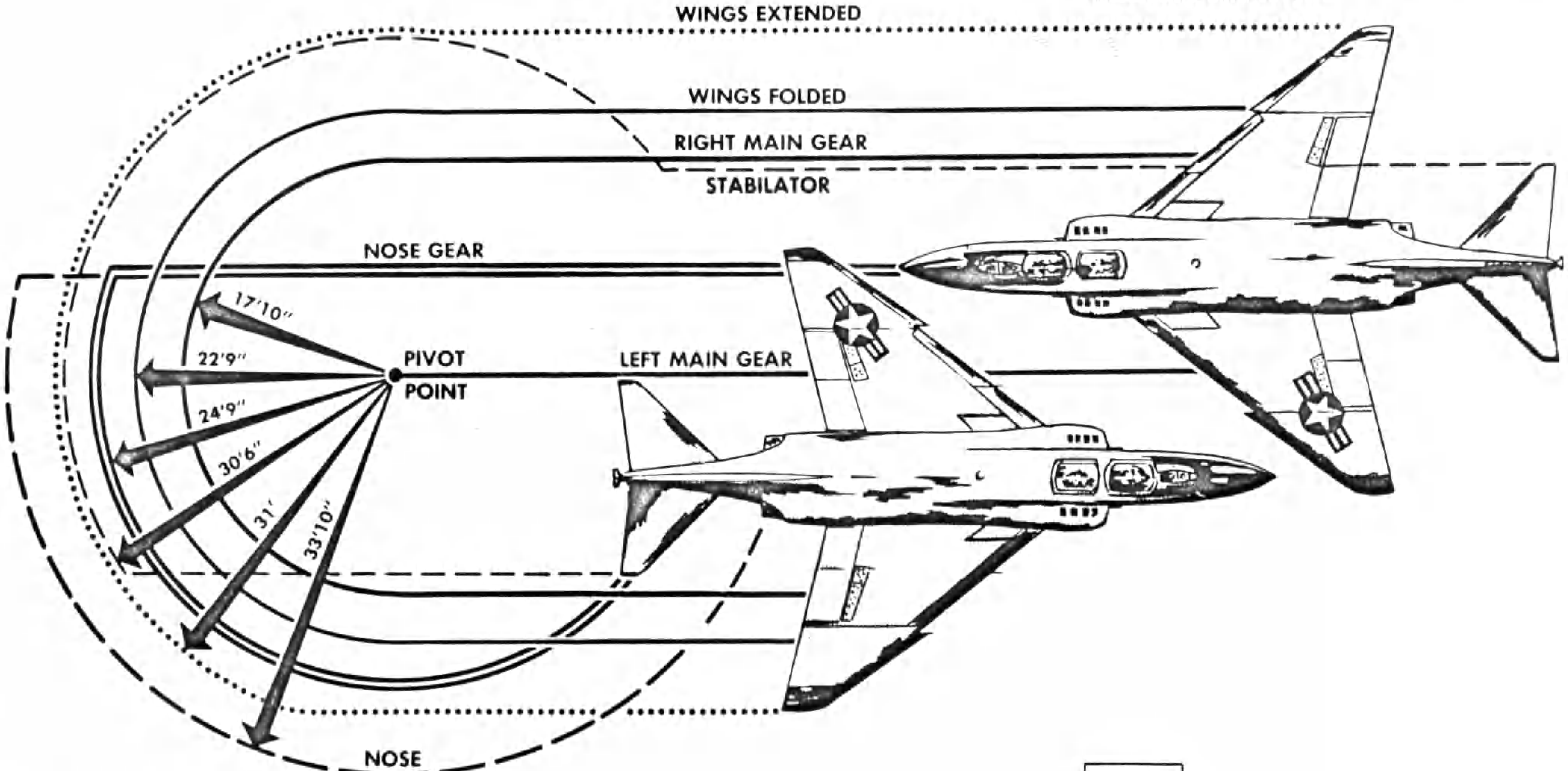
MINIMUM TURNING RADIUS AND GROUND CLEARANCE

AIRCRAFT BEING TAXIED

Note

UNDER HIGH GROSS WEIGHT CONDITIONS, THE TURN RADIUS SHOULD BE INCREASED TO RELIEVE SIDE LOADS ON THE MAIN GEAR AND TIRES.

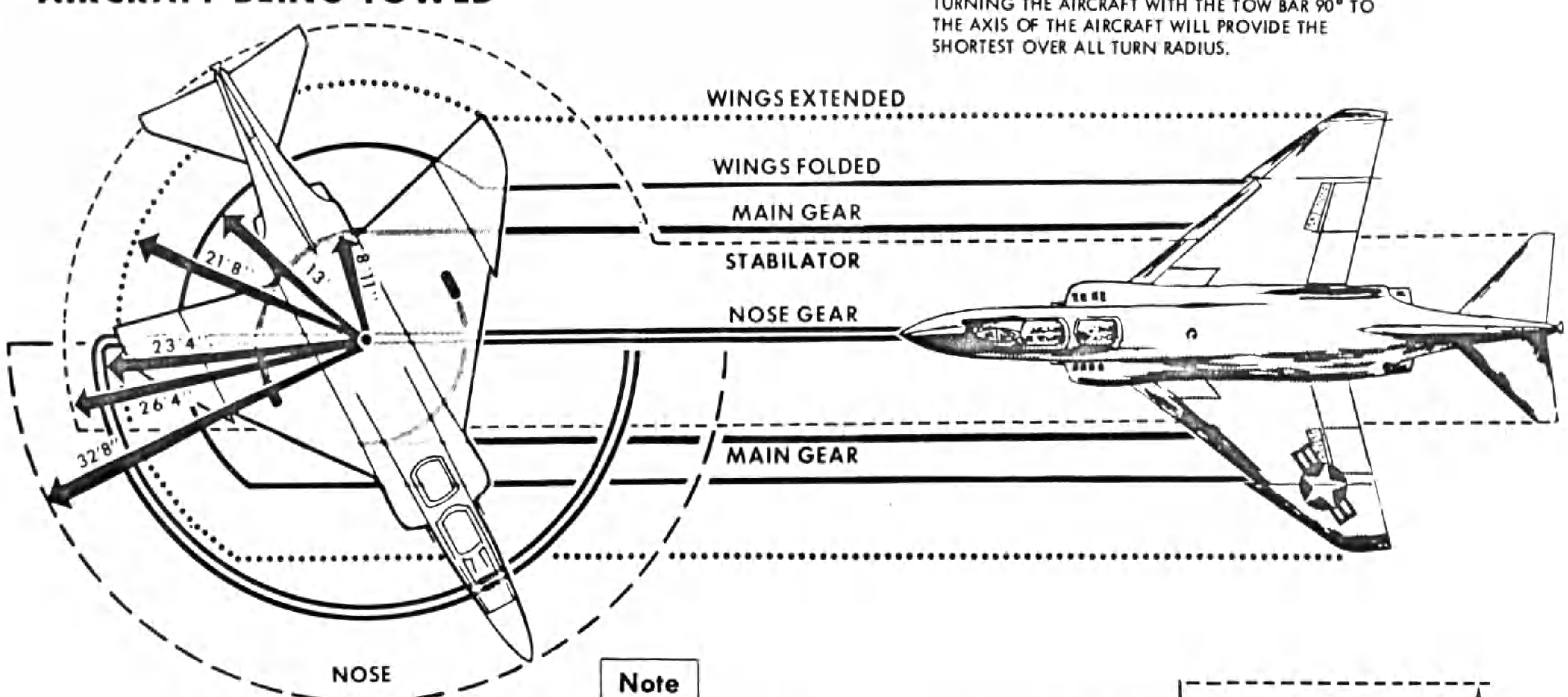
IF THE SITUATION WARRANTS THE AIRCRAFT CAN BE PIVOTED AROUND THE GEAR BY LOCKING THE APPLICABLE BRAKE, HOWEVER DOING SO SCUFFS THE LOCKED TIRE EXCESSIVELY.



AIRCRAFT BEING TOWED

Note

TURNING THE AIRCRAFT WITH THE TOW BAR 90° TO THE AXIS OF THE AIRCRAFT WILL PROVIDE THE SHORTEST OVER ALL TURN RADIUS.



Note

DO NOT ATTEMPT TO TURN THE AIRCRAFT WITH THE NOSE STRUT FULLY EXTENDED.

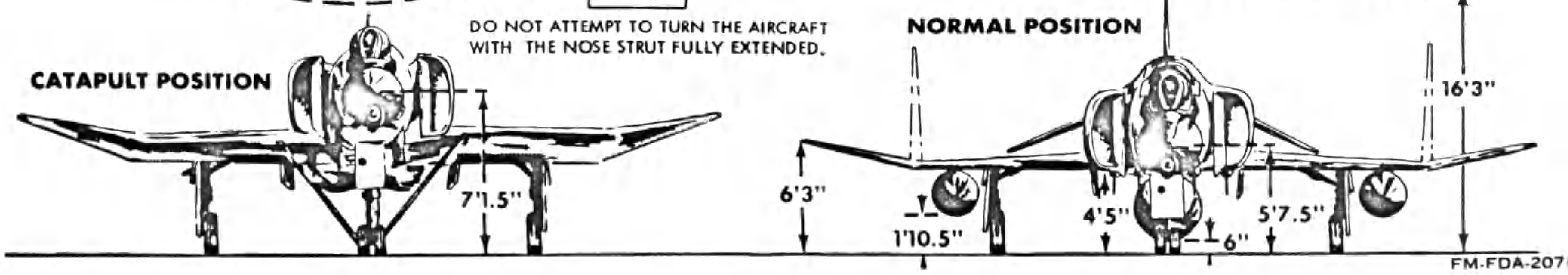


Figure 2-6

BEFORE TAXIING

1. External power sources - DISCONNECT
   Have starting external electrical and compressed air sources disconnected; receive confirmation from ground crew that external electrical and compressed air sources are clear of airplane.
2. Area clear - CHECK
   Visually check area and receive confirmation from ground crew that area is clear.
3. Wings - SPREAD AND LOCKED
   After wings have been spread, manually push wing pin lock handle down, check that wing pin unlocked warning lights are extinguished.
4. Speed brakes - CYCLE
   Open and close speed brakes and check operation. Obtain ground crew signal that speed brakes are fully closed. Observe that warning light is out.
5. Wing flaps - CYCLE, CHECK INDICATORS
   Cycle flaps and check that flap indicators indicate flap positions. Observe ground crew signal that flaps are cycling properly.
6. Refuel probe - CYCLE
   Cycle refuel probe, check for operation, and visible damage to nozzle.
7. Flight control surfaces and power control systems - CHECK TRAVEL - NOTE PRESSURE DROP
   Cycle controls, check corresponding movement of control stick and control surfaces. While cycling controls, observe pressure drop on both power control hydraulic systems. This insures that both power control systems are supplying power to ailerons, spoilers, and stabilator.

Note

When making the above check, pull stick full aft and release gently, stick should return to its forward position. Initiate stick aft movement and release, the stick should stop and return to its forward position. There should be NO TENDENCY for the stick to motor which means that the stick continues to move in the direction initiated without further application of pilot effort.

8. Stab Aug (Damper) switch - ENGAGE
9. AFCS Reliability - CHECK
   a. Full NOSE UP trim.
   b. Grasp control column below control stick grip and neutralize.
   c. Wait 3 seconds.
   d. AFCS - ENGAGE
   e. Check for hard forward control stick jump, visually check control surfaces for hard over movement.
   f. If either of the above conditions exist DO NOT USE AFCS.
10. Automatic flight control system - ENGAGE, CHECK, AND DISENGAGE
    a. Stab Aug (Damper) switch - ENGAGE
    b. AFCS (Autopilot) switch - ENGAGE

CAUTION

Check for hard over control surface movement when autopilot is engaged. If present, a system malfunction exists which could cause violent maneuvers in flight upon engagement of autopilot.

    c. Altitude switch - ENGAGE
       Move control stick forward; switch should move to OFF. RE-ENGAGE altitude switch and move stick aft; switch should again move to OFF.
    d. Mach switch - ENGAGE
       Move control stick forward; switch should move to OFF. RE-ENGAGE Mach switch and move stick aft; switch should again move to OFF.

CONTINUED ON NEXT PAGE

BEFORE TAXIING CONTINUED

e. Disengage lever - DEPRESS

Depress disengage lever on control stick; check that all switches on flight control panel are OFF.

11. Static pressure compensator - RESET
12. Trim switches - CYCLE AND SET FOR TAKE-OFF

    Check stick trim button and rudder trim switch for operation. Check indicators correspond to switch movement. Set for take-off indication.
13. Ladder - RETRACT

    Have ground crew retract boarding ladder. Check indicator to assure retraction.

"All data deleted from page 69."

TAXIING

1. Taxi area - CLEAR
   Check taxi area clear of personnel and other obstructions and check jet blast area before applying taxi power. Advise radar observer of intention to taxi.
2. Chocks - REMOVE
   Give ground crew positive visual signal for the removal of chocks.

CAUTION

When chocks are pulled, have feet on brakes since there is no parking brake on the airplane.

3. Canopy - CLOSE
   Before closing canopy, check that canopy sill is clear. Advise radar operator to close his canopy. Check that canopy unlocked warning light is extinguished.

WARNING

Closing the canopy on an arm or hand may result in serious injury.

CAUTION

If canopy does not close immediately after placing control lever in the closed position, return control lever to the OPEN position before attempting to close canopy again. This is to prevent canopy from closing violently if canopy was being held open by fast taxi speeds or high wind velocities.

4. Brakes - CHECK
   After initial roll, apply brakes and check operation.
5. Taxi - LOWEST PRACTICAL RPM
   Once the airplane is moving, taxi at the lowest practical rpm necessary to maintain forward motion to minimize brake heat and wear.
6. Nose wheel steering - ENGAGE
   Engage nose wheel steering to minimize brake wear.

CAUTION

- With the canopy open, keep speed below 60 knots to prevent damage to canopy operating mechanism.
- Maintain a minimum distance of 125 feet from the exhaust blast of another airplane that is operating at military thrust and 200 feet from the exhaust blast of an airplane that is operating at maximum thrust to prevent damage to an open canopy.

Note

Under high gross weight conditions, the turn radius should be increased to relieve side loads on the main gear, wheels, and tires.

7. Flight instruments - CHECK FOR CORRECT INDICATIONS
   Check flight instruments for correct operation while taxiing.
8. Minimize taxi time to decrease fuel consumption.

Note

Fuel consumption during taxiing at idle rpm is approximately 42 ppm.

BEFORE TAKE-OFF

After taxiing to the run-up area, allow the aircraft to roll straight ahead to assure nose gear alignment. Apply brakes and complete the following checks:

PRE-TAKE-OFF AIRCRAFT CHECK

1. Integrated harness - CHECK
   Check that integrated harness is properly secured to the parachute.
2. Inertia reel handle - LOCKED
3. Personal equipment leads - CHECK
   Check composite disconnect in place and secure. Check all equipment leads properly attached and that they create no obstruction to movement.
4. Canopies - CLOSED - Warning Light - OUT
5. Speed brakes - IN - Warning Light - OUT
6. Arresting gear control handle - UP
   Manually check arresting gear control handle to assure that arresting gear is up.
7. Wings - SPREAD AND LOCKED
   Check that wings are fully spread and locked. Wing pin unlocked warning lights extinguished.
8. Take-off trim - NEUTRAL
   Check trim indicators.
9. Flight controls - CHECK FOR FREE MOVEMENT
   Check flight controls for freedom of movement and that no equipment interferes with stick movement.
10. Alternate ejection handle safety guard - ROTATE CLEAR
    Rotate safety guard to the horizontal position.

PRE-TAKE-OFF ENGINE CHECK

CAUTION

- Check the engines individually because the engines develop enough thrust on a cold day to cause the airplane to skid if both engines are run up together and maximum braking is applied.
- Do not operate engine(s) at MIL power or above with the flaps down for a period longer than one minute. Operation for longer periods of time will cause the BLC compressor bleed air to overheat the leading edge wing rib.

1. Brakes - HOLD
   Apply and hold a firm steady brake pressure.
2. Engines - ALTERNATELY CHECK TO MIL POWER
   Move throttle to MIL stop, allow engine rpm to stabilize at 103.5%; observe that the EGT, nozzle position, fuel flow, oil pressure, and hydraulic pressure gages are within proper operating ranges.

Note

- Spasmodic exhaust nozzle operation shall be logged on the yellow sheet (OPNAV FORM 3760-2).
- Fuel consumed during pre-take-off engine check is approximately 100 pounds.

3. Wing flaps - 1/2 DOWN
   Check wing flap indicator in the 1/2 position.

TAKE-OFF (ASHORE)

1. Tower - REQUEST CLEARANCE
   Request tower clearance on the active runway.
2. On reaching active runway - LINE UP AND HOLD
3. Throttles - 85%
   Advance throttles to approximately 85%, allow to stabilize.
4. Flight and engine instruments - MONITOR
   Monitor all flight and engine instruments for correct indication.
5. Warning lights - CHECK
   Check all warning lights extinguished.
6. Obtain a ready to take-off clearance from the radar observer.
7. Brakes - RELEASE
   Release brakes evenly.
8. Throttles - MIL (MAX if required)
   If afterburner power is required, move throttles outboard and full forward to MAX (afterburner) position.
9. Maintain directional control until the rudder becomes effective at approximately 70 knots IAS.
   a. Nose wheel steering - ENGAGE (Airplanes 148252d thru 148275e).
   b. Use light differential braking (all previous airplanes).

CAUTION

Disengage nose wheel steering before aircraft lift off to insure proper nose wheel centering and nose gear retraction.

10. Rotate airplane at applicable take-off airspeed.
    a. 1/2 flaps 135-140 knots CAS.
    b. Full flaps 130-135 knots CAS.

Note

The "Aux Air Door" caution lights and the "Master Caution" light may illuminate momentarily when the landing gear is being raised.

TAKE-OFF TECHNIQUE

Take runway and, when in position, roll forward slightly to align nose wheel. Pump brakes to insure positive pressure. Perform all checks as outlined. Three flap configurations are available for take-off; flaps up, one-half, and full trailing edge flap. Half flaps is recommended over full flap for normal field use due to a reduction in drag, increased power (less bleed air for BLC), only a small loss of lateral control, increased stabilator effectiveness, proper configuration in case of single engine failure, and less trim change while cleaning up. Full flaps provide a gain towards minimum speed liftoff, and are necessary for shipboard catapult launches. When ready, take-off roll may be started with engines at idle or brakes can be applied until 90% rpm is reached on each engine. After the take-off roll has begun, and temperature and engine speed have been checked out, move both throttles into the afterburner detent and continue smoothly to max thrust. With the J79-2 engine, three small bursts of increased power will be felt from each engine as the afterburner sectors light off. Airspeed will probably have reached 60-70 knots IAS as the afterburners light off and the rudder will then be effective for heading control. Very light braking or nose wheel steering (if installed) can be used for initial heading control. If one afterburner fails to light, sufficient directional control is available with rudders to continue the take-off with asymmetric power. The nose strut can be extended at 110-115 knots, with slight back pressure. Moderate back stick is then required to raise the nose wheel and liftoff at 145 knots. If full flaps are used the same technique applies, and a 10 kt. reduction in airspeed can be gained by leaving the nose down until reaching 125-130 kts. at which time full aft stick will rotate the airplane to a flying attitude. Ease off the back stick when a good climb path is established. No problem should be encountered with pitch or roll sensitivity during liftoff. When the airplane is definitely airborne raise the landing gear and immediately afterward start the flaps up. If flap retraction is forgotten, an airspeed switch will start flaps up at 225 kts. Transition to clean configuration will require little stick force or attitude changes. For a clean take-off extend the nose strut at 130 kts. and liftoff at 160 kts. Pitch control will be more sensitive than with flaps down. Initial climb acceleration is very rapid, and the pilot must be prepared to scan instruments quickly.

MINIMUM RUN TAKE-OFF

A minimum run take-off in this airplane is the same as a normal afterburner take-off.

CROSSWIND TAKE-OFF

1. Release brakes evenly. Maintain directional control by use of brake during initial part of roll, or by nose gear steering if installed.

CAUTION

Do not ride or keep pressure on brakes during crosswind take-off. Brakes should be used sparingly to prevent overheating. Excessive braking will increase take-off run.

2. Rudder control becomes effective at approximately 70 knots CAS.
3. Hold nose gear on runway until flying speed is reached.
4. Pull aircraft off at approximately 150 knots CAS.

WARNING

In order to counteract airplane drift at lift-off, do not attempt an immediate wing low attitude. The pilot cannot properly judge the wing tip ground clearance on a swept wing type aircraft.

HEAVY GROSS WEIGHT TAKE-OFF

A heavy gross weight take-off is accomplished in the same manner as a normal take-off with the following exceptions and additions: It is recommended that all heavy gross weight take-offs be made with afterburner. The possibility of a main landing gear tire failure increases with an extended take-off ground roll under heavy gross weight conditions. Nose gear lift-off speed and take-off speed must be increased appropriately for heavy gross weight conditions. In the event of an aborted take-off, it must be remembered that stopping distance is greatly increased under heavy gross weight conditions.

TAKE-OFF (CATAPULT LAUNCH)

Prior to catapult holdback hook up:

1. Pneumatic system pressure gage - CHECK (2750 PSI MINIMUM)
   Check pneumatic system pressure gage for minimum pressure of 2750 psi prior to nose gear strut extension.
2. Nose gear strut - ASCERTAIN FULLY EXTENDED PRIOR TO BRIDLE HOOK UP

After catapult holdback hookup, prior to bridle hookup and tensioning:

1. Brakes - RELEASE

After bridle hookup and tensioning:

1. Pneumatic system pressure gage - CHECK (1450 to 1580 PSI)
   Check pneumatic system pressure between 1450 to 1580 psi after nose strut extension.
2. Wing flap switch - FULL DOWN
3. TRIM - SET
   a. Pitch trim
      (1) 0-10 Knots above normal end airspeed - 4 UNITS NOSE UP
      (2) 10-20 Knots above normal end airspeed - 2 UNITS NOSE UP
   b. Lateral trim - NEUTRAL
   c. Directional trim - NEUTRAL
4. Throttles - 85%
   Advance throttles to 85%, allow to stabilize.
5. Flight and engine instruments - MONITOR
   Monitor all flight and engine instruments for correct indications.
6. Warning lights - CHECK
   Check all warning lights extinguished.
7. Obtain a ready for take-off clearance from the radar observer.
8. Throttle - MIL (MAX IF REQUIRED)
   If afterburner power is required move throttles outboard and full forward to MAX (afterburner position).
9. Throttle friction lever - ADJUST

CONTINUED ON NEXT PAGE

TAKE-OFF (CATAPULT LAUNCH) CONTINUED

10. Catapult holdback handle - SET
    Raise catapult holdback handle (MIL or MAX) and grasp holdback handle and throttles.
11. Assume catapult launching position.
    a. Head firmly back against headrest.
    b. Stick held firmly, arm braced.
    c. Feet on lower portion of rudder pedals.
12. Catapult officer - SIGNAL READY FOR LAUNCHING

CAUTION

Do not cycle the generator control switches after nose strut extension. Momentary loss of power to the strut extension solenoid valve will allow the nose gear to return to the normal position. Excessive nose gear loads, poor launch attitudes and possible shedding of the bridle during the catapult stroke could result.

AFTER TAKE-OFF

When airplane is definitely airborne:

1. Landing gear - UP
   Place landing gear handle in the UP position; check indicators and warning light in gear handle, and instrument panel.

CAUTION

Landing gear should be completely up and locked before gear limit airspeed is reached; otherwise excessive air loads may damage gear mechanisms and prevent subsequent operation.

2. Wing flaps - UP
   Retract flaps before reaching flap limit airspeed. Check flap indicator.
3. Accelerate to climb schedule.

CLIMB

1. Trim for climb.
2. Antenna selector switch - LWR
3. Cabin/suit altimeter - CHECK

CRUISE

Cruise control data for various gross weights and several configurations are contained in Appendix I.

WARNING

The fuel quantity gage may give unreliable (higher than actual) readings in the low end of the scale (0-1000 pounds). Therefore, the fuel level low warning light should be used as the primary indication of a low fuel state (1960 ± 100). Continued operation after illumination of this light should be cautiously considered.

CAUTION

Prior to zoom climbs, and/or flights above 60,000 feet turn all CNI, and AMCS equipment OFF. Flight above 60,000 feet with CNI, and AMCS equipment on will result in damage to equipment.

Note

There is a possibility that fuel quantity variations will be noted on the fuel quantity indicator during aircraft accelerations and decelerations. These variations are due to the high acceleration and deceleration rates which can be achieved with the airplane. Transient increases in fuel quantity readings may be noted during deceleration, and transient decreases in fuel quantity readings may be noted during acceleration.

FLIGHT CHARACTERISTICS

Refer to Section VI for information regarding flight characteristics.

DESCENT

1. In all descents, the pilot should not exceed any airframe limitations. See Limitations, Section V.

Note

Circumstances may arise which require a fast descent from high altitude. This may be accomplished by chopping the throttles to idle and by extending the speed brakes.

2. Prior to all descents from altitude, actuate defogging to prevent possible fogging or icing on the windscreen and canopy.
3. Since rapid descents cannot always be anticipated, the maximum comfortable interior temperature should be maintained. This will aid in defrosting the windshield.

BEFORE LANDING

1. Mission jettison selector switch - OFF
2. Missile power switch - OFF
3. Mode select switch - OFF
4. Fuel quantity - CHECK
5. Hydraulic pressure gages - CHECK
   Check all three hydraulic pressure gages for normal indication.
6. Integrated harness - CHECK
   Check integrated harness tightened.
7. Autopilot switch - OFF
8. Speed brakes - AS NECESSARY
9. Enter traffic pattern.
   Adjust power to maintain pattern speed and altitude. Refer to Figure 2-7 for complete traffic procedures.

LANDING PATTERN

1. Inertia reel handle - LOCKED
2. Speed brakes - RETRACT
3. Landing gear - DOWN AND LOCKED

Note

The "Aux Air Door", and the "Check Hyd. Gages" caution lights and the "Master Caution" light may illuminate momentarily when the landing gear is being lowered.

4. Wing flaps - FULL DOWN

CAUTION

Do not lower landing gear in turns or pull-ups, as "g's" encountered may damage the landing gear mechanism.

WARNING

To prevent tire failure and possible damage to the airplane and occupants do not touchdown with any pressure on the brake pedals.

LANDING (FIELD)

NORMAL LANDING

For a normal field landing with a gross weight of 31,000 lbs., fly the pattern as illustrated in figure 2-7. Enter the pattern using the speed brakes and throttles as necessary. On the downwind leg, power should be reduced to obtain gear down speed (250 knots CAS), the gear and flaps should be extended, speed brakes retracted (to decrease buffet) and a mild rate of descent should be started. Landing gear and flap extension requires negligible attitude and trim change. Quite a bit of noise and buffet will come from the nose gear well when the gear is extended, however, this will slowly disappear when approach speeds are reached. By the time the gear is down and locked, airspeed should indicate about 200 knots. Roll into the base leg with a moderate rate of descent and maintain about 160-165 knots CAS. When gear and flaps are checked and reported on the base leg, check aileron rudder interconnect (ARI) by watching rudder position response to aileron movements. When the 90° position is reached, airspeed should be down to approximately 10-15 knots above recommended approach airspeed. This will give a comfortable control for completing the turn onto final. Use the angle of attack indexer and maintain the "on speed" indication. Maintaining a final approach speed of approximately 88% rpm, and a 2 1/2°-3° glide slope angle will provide a mild rate of descent (approximately 700 fpm). Plan to touchdown within the first 1000 feet of the runway whenever possible. Do not attempt to flare or chop power prior to crossing the end of the runway as the sudden loss of lift would cause the airplane to settle immediately. Upon touchdown retard throttles to idle and deploy drag chute. The nose will drop almost immediately due to CG and stabilator location. When the nose wheel is on the deck use differential braking or nose wheel steering if installed.

LANDING ROLL

Since little aerodynamic braking will be available, raise the flaps to increase weight on the main gear and start light braking action at about 100 kts. The airplane is very clean on landing and even with fairly low residual thrust it will want to roll down the runway with little deceleration. Exercise caution while using the brakes until you get the feel of them. They are fully powered rather than boosted and there is not much feel at the pedals. The tire pressures are very high and they will break loose and skid with heavy applications. Do not deploy the drag chute prior to touchdown, or landing will be accomplished much shorter and harder than expected. The drag chute will substantially reduce landing roll and keep taxi speeds down returning to the ramp.

CROSSWIND LANDING

Carefully compensate for crosswinds in the traffic pattern to guard against undershooting or overshooting the final turn. On final approach, use wing low or crab method to maintain course. Maintain normal approach speed aligning airplane with the runway just prior to flare-out. After touchdown, lower nose gear to the runway as soon as possible and apply forward stick, and engage nose wheel steering. With the nose gear held off, the airplane's "weathervane" effect may not be correctable. If required, the drag chute may be deployed but only after the nose gear is on the runway. Under crosswind conditions where effective crosswinds are in excess of 20 knots, use of drag chute intensifies the weathervane effect. If "weathervane" effect is severely intensified with the drag chute deployed, jettison the drag chute.

HEAVY GROSS WEIGHT LANDING

As landing gross weight increases, the landing pattern should be expanded and approach and touchdown speeds should be increased accordingly. Follow procedures outlined in Landing Pattern Diagram, figure 2-7, adding 2 knots to quoted airspeeds for each 1000 pounds over normal landing gross weight, i.e., an airplane with 6000 pounds of fuel remaining, final approach speed would be 141 knots CAS.

WET RUNWAY LANDING

This information will be supplied when available.

LANDING (CARRIER)

Approach the ship upwind approximately 1-3/4 miles abeam the starboard side and at 500 feet altitude. Lower hook, disengage autopilot, and turn armament switches off. Break at 500 feet approximately 1/2 mile past the bow of the ship. Reduce airspeed to 230 knots CAS using throttle and speed brakes (if necessary) and lower gear. Turn downwind, and lower flaps at 210 knots CAS, retract speed brakes if extended and perform landing check list. Maintain 500 feet altitude and an on-speed index indication on the downwind leg. Notify MLO when landing check list is complete. The mirror approach pattern is initiated by a 30° angle of bank turn from the downwind leg 4 to 5 seconds after passing the stern of the ship. Hold 500 feet, and an on-speed index indication during turn to intercept glide slope. Pick up "meatball" in the turn on to final to aid in the interception of the proper glide slope altitude, and to facilitate "line-up" as early as possible. After rollout from base turn, reduce thrust 4% to 5% rpm to obtain initial glide slope rate of descent. Maintain on-speed indexer indication, and keep "meatball" and deck centerline centered.

WAVE-OFF

The decision to go-around should be made as early as possible. If decision is made to go-around, proceed as follows:

1. Throttles - MIL or MAX (if required)
   Advance throttles to MIL or AFTERBURNER as required to check sink rate.

CAUTION

Due to excessively high fuel consumption, afterburner should be used only in emergencies if a low fuel state exists.

2. Landing gear - UP
   Retract landing gear only after adequate flying speed is attained as touchdown may be necessary during go-around.
3. Wing flaps - UP
4. Clear runway.

LANDING

1. Upon touchdown, throttles - IDLE
2. Ease nose gear to runway.
3. Maintain directional control during rollout.
   a. Nose wheel steering - ENGAGE (Airplanes 148252d thru 148275e)
   b. Use light differential braking (all previous airplanes)
4. Drag chute - DEPLOY
5. Wing flaps - UP
6. Brakes - APPLY
7. Alternate ejection handle safety guard - UP
   Rotate the safety guard to the UP position.

After clearing runway:

8. Drag chute - RELEASE IN DESIGNATED AREA

CAUTION

- If there has been indication of a hydraulic system failure, DO NOT attempt to shut down engines until wheel chocks are in place and landing gear ground locks installed.
- If canopies are open during taxiing, do not exceed canopy open limits of 60 knots.

FIELD LANDING PATTERN

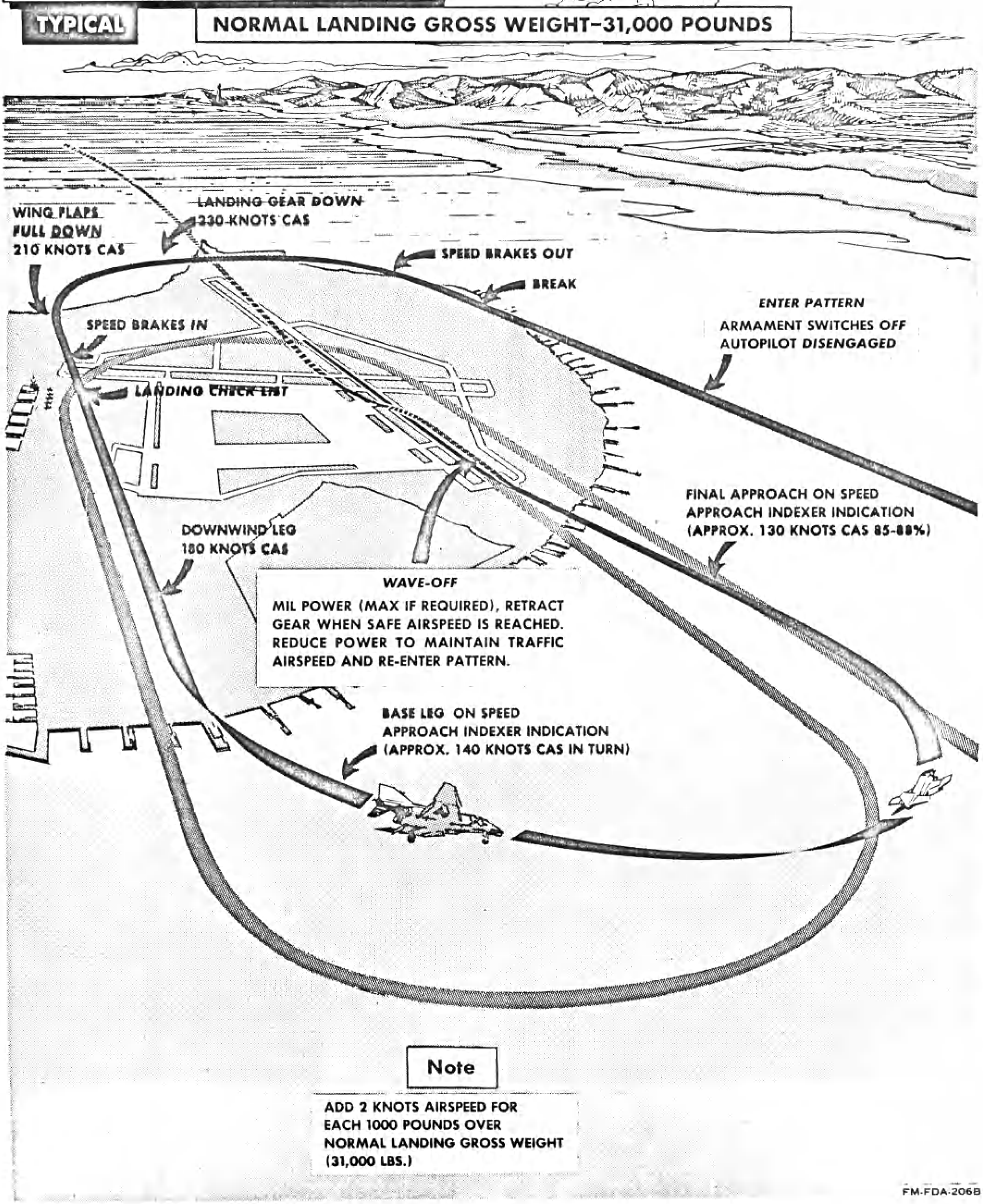


Figure 2-7

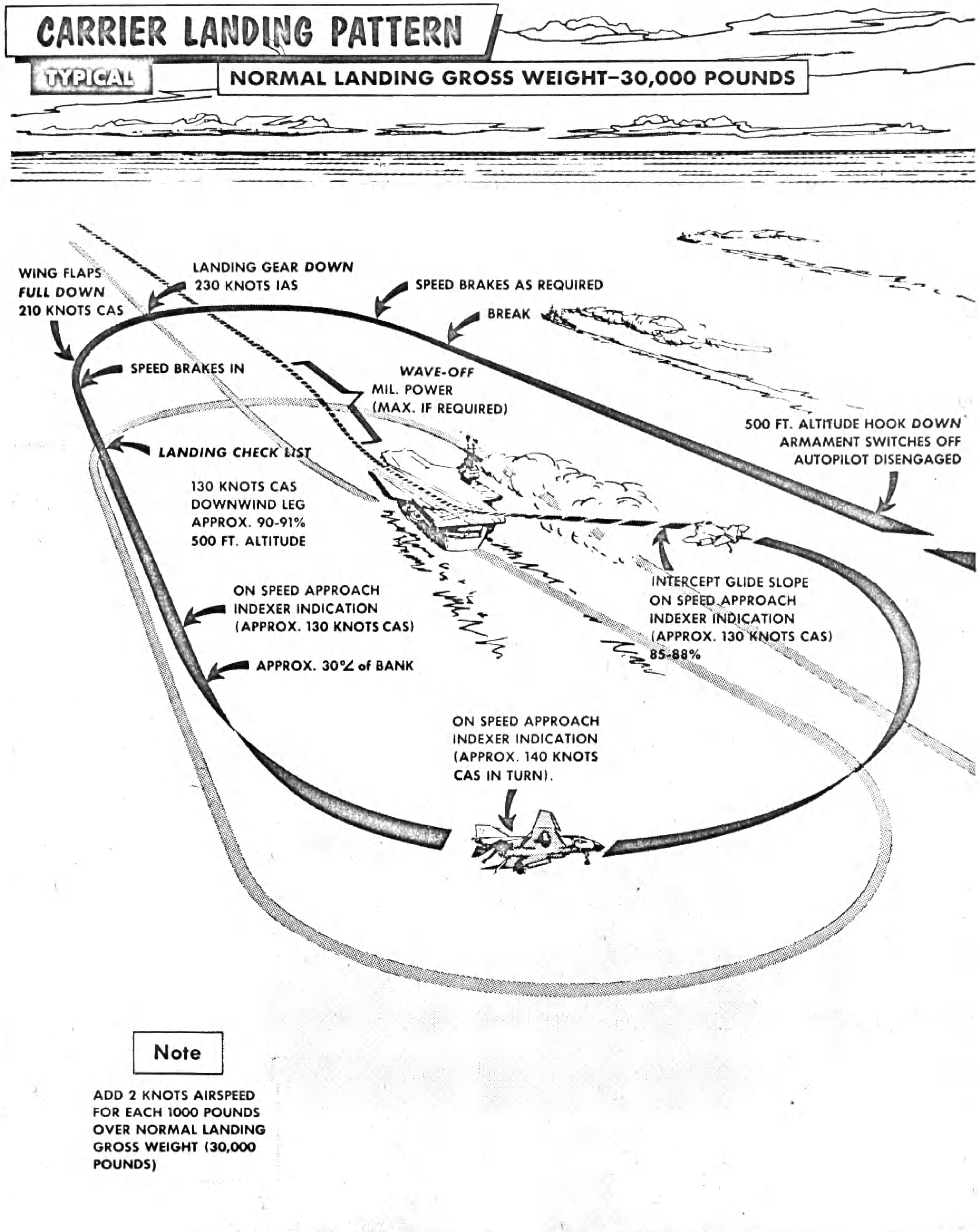

CARRIER LANDING PATTERN
TYPICAL
NORMAL LANDING GROSS WEIGHT–30,000 POUNDS
WING FLAPS FULL DOWN 210 KNOTS CAS
LANDING GEAR DOWN 230 KNOTS IAS
SPEED BRAKES AS REQUIRED
BREAK
SPEED BRAKES IN
WAVE-OFF MIL. POWER (MAX. IF REQUIRED)
500 FT. ALTITUDE HOOK DOWN
ARMAMENT SWITCHES OFF
AUTOPILOT DISENGAGED
LANDING CHECK LIST
130 KNOTS CAS
DOWNWIND LEG
APPROX. 90-91%
500 FT. ALTITUDE
ON SPEED APPROACH INDEXER INDICATION (APPROX. 130 KNOTS CAS)
APPROX. 30°∠ of BANK
INTERCEPT GLIDE SLOPE ON SPEED APPROACH INDEXER INDICATION (APPROX. 130 KNOTS CAS) 85-88%
ON SPEED APPROACH INDEXER INDICATION (APPROX. 140 KNOTS CAS IN TURN).
Note
ADD 2 KNOTS AIRSPEED FOR EACH 1000 POUNDS OVER NORMAL LANDING GROSS WEIGHT (30,000 POUNDS)
FM-FDA-209A

Figure 2-8

ENGINE SHUTDOWN

Note

In order to check accessory malfunctions of the individual engines that may not be apparent when both engines are operating, shutdown the engines individually. The engine that was started first should be shutdown first.

1. Throttles - IDLE
   It is recommended but not required that the engines be operated in IDLE for 3 to 5 minutes to allow engine temperatures to stabilize (rollout and taxi time may be included).

Note

Carrier landings may require that the engine be shut down almost immediately after touchdown from high power settings. If engines are shutdown prior to recommended idle time, make notation on yellow sheet.

2. Wheels - CHOCKED
   Get signal from ground crew that wheels are chocked before releasing brakes.
3. Right engine - SHUTDOWN
   a. Throttles - OFF
   b. Engine master switches - OFF
   c. Generator switches - OFF

Note

During coast down, listen for unusual engine noises, and note deceleration times are normal.

CAUTION

With only one engine in operation, do not move control stick (surface controls) excessively. If the stick is moved excessively with hydraulic pressure on only one side of the tandem power cylinders, the fluid that is in the other side of the cylinder is forced back through the pressure line to the reservoir, filling the reservoir, and forcing the excess fluid overboard. The seals within the tandem power cylinders may also be damaged due to the ingesting and expelling of air and lack of lubrication. The power control hydraulic systems must be reserviced and checked.

WARNING

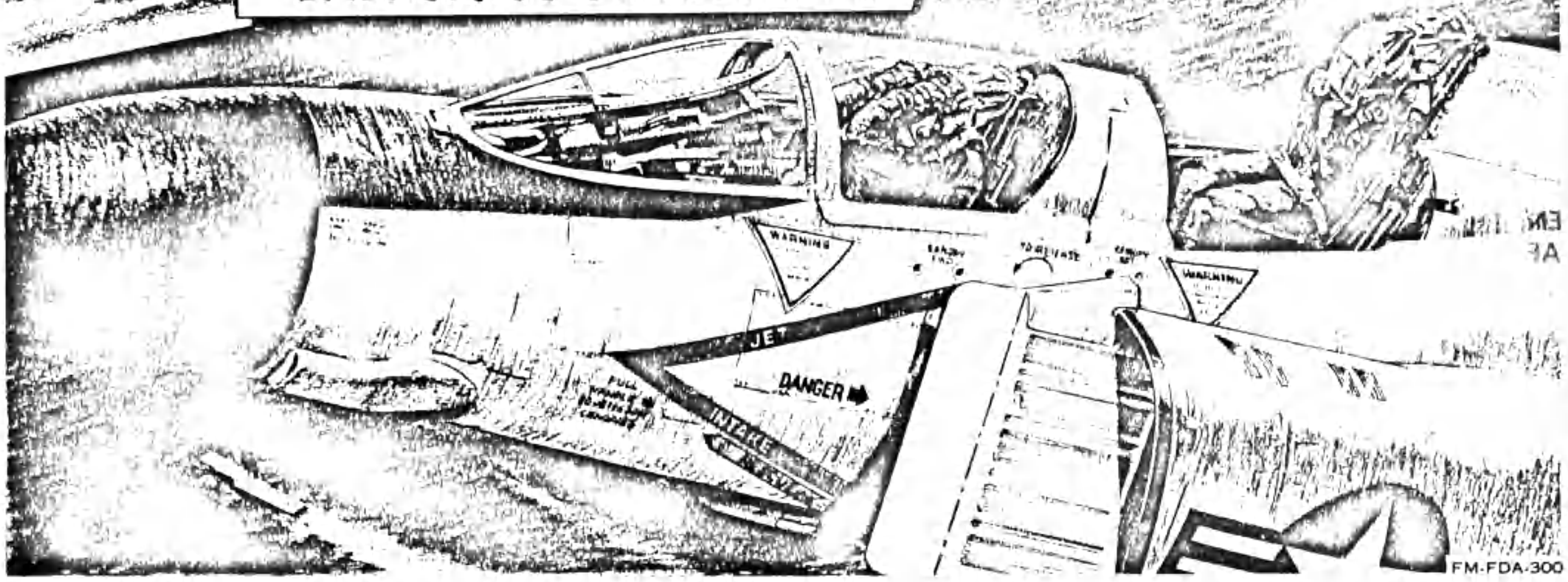
When electrical power is removed from the airplane, the auxiliary air doors will close violently.

BEFORE LEAVING AIRPLANE

1. All switches, levers, and personal equipment - OFF or DISCONNECTED
2. Install plastic guard on composite disconnect.
3. Alternate ejection handle safety guard - UP
   Check the safety guard in the vertical position.
4. Safety banner - INSTALLED
   Install the canopy and ejection seat safety pins and dust covers.
5. Landing gear ground locks - INSTALLED
   Check that the landing gear ground locks have been properly installed.
6. Flight forms - COMPLETED

SECTION III

EMERGENCY PROCEDURES



FM-FDA-300

Note

Since the CADC automatically compensates for static source error to provide a direct reading of calibrated airspeed, all airspeed values within this Section will be quoted as CAS. In the event of illumination of the "Static Corr Off" warning light, quoted airspeeds should be corrected for static source error. Refer to Airspeed Position Error Correction charts in Part 1 of Appendix.

ENGINE FAILURE

Jet engine failures in most cases will be caused by improper fuel scheduling due to malfunction of the fuel control system or incorrect techniques used during certain critical flight conditions. Engine instruments often provide indications of fuel control system failures before the engine actually stops. If engine failure is due to a malfunction of the fuel control system or improper operating technique, an air start can usually be accomplished, providing time and altitude permit. If engine failure can be attributed to some obvious mechanical failure within the engine proper, DO NOT attempt to restart the engine.

SINGLE ENGINE FLIGHT CHARACTERISTICS

Single-engine flight characteristics are essentially the same as normal flight characteristics due to proximity of the thrust lines to the center of the airplane. With one engine inoperative, slight rudder deflection is required to prevent yaw toward the failed engine. Thus, good control is assured in the single-engine range. Minimum single-engine control speed varies with gross weight, flap setting, and landing gear position. The aircraft design is such that no one system (flight control, pneumatic, electrical, etc.) is dependent on a specific engine. Thus, loss of an engine will not result in a loss of a complete system.

ENGINE FAILURE DURING TAKE-OFF BEFORE LEAVING GROUND

If an engine fails before leaving the ground, the continuation of take-off is dependent on length of remaining runway, gross weight, airspeed, field elevation and ambient temperature.

Note

- During take-off using Military power, where take-off will not be aborted, immediately advance operating engine to Maximum power and follow Engine Failure During Flight Procedures, this section, as soon as possible.
- If single-engine failure occurs using Maximum power and take-off will not be aborted, immediately retard "dead" engine throttle from afterburning range and follow Engine Failure During Flight procedures, this section, as soon as possible.

If the decision to stop is made, proceed as follows:

1. Throttles - IDLE
2. Drag chute - DEPLOY
3. Wheel brakes - APPLY
   Supplementary braking can be obtained with the emergency braking system.
4. Notify R.O. of existing emergency and intended action.
5. Request arresting gear if necessary.
6. Wing flaps - UP

If an arrested roll out is to be made:

7. Arresting gear - DOWN
   Extend the arresting gear shortly before anticipated engagement.
8. Aim for the center of the wire.

Note

- The hydraulic brakes will remain operative for approximately 10 full brake applications after the loss of the utility hydraulic system.
- It is possible to use differential hydraulic braking in conjunction with metered pneumatic emergency braking.

ENGINE FAILURE DURING TAKE-OFF AFTER LEAVING GROUND

If an engine fails immediately after take-off, lateral and directional control of the airplane can be maintained if airspeed remains above stalling speed. However, the ability to maintain altitude or to climb depends upon gross weight and air density. Since the pilot's reactions will depend upon the conditions mentioned above, after take-off, and at critical airspeeds with heavy gross weight, the pilot must complete the following if level flight cannot be maintained.

1. Operating engine - MAX (AFTERBURNER)
2. Landing gear - UP
3. External stores - JETTISON (if necessary)
   If altitude cannot be maintained, jettison the external stores.
4. Continue straight ahead.
5. Wing flaps - UP
6. Failed engine - SHUTDOWN

If failure is other than mechanical:

7. ATTEMPT AIRSTART
8. Land as soon as possible.

ENGINE FAILURE DURING FLIGHT

In the event of an engine failure, perform the following:

1. Positively determine which engine has failed.
2. Throttle - OFF

If failure is other than mechanical:

3. ATTEMPT AIRSTART

CAUTION

In order to reduce the possibility of engine seizure, initiation of engine restart should not be prolonged following flameout. If one or both engines flameout, do not delay the airstart attempt in order to set up windmill rpm.

If failure is mechanical:

4. Engine - SHUTDOWN
   a. Engine master switch - OFF
5. Land as soon as possible.

DOUBLE ENGINE FAILURE DURING FLIGHT

The possibility of a double engine failure is highly remote. However, if such a situation should occur, proceed as follows:

1. Reduce airspeed to 515 knots CAS or Mach 1.1 whichever is lower.
2. Wind driven turbine - EXTEND
   Extend the wind driven turbine to operate the left fuel boost pump at low speed. This will supply enough fuel to either engine for an air start.

CAUTION

Maintain airspeed above 195 knots CAS to prevent the emergency generator pressure switch from disconnecting the emergency generator from the essential buses. This would result in a complete loss of electrical power with the exception of engine ignition.

3. Throttles - OFF
4. Either engine - ATTEMPT AIRSTART (45,900 feet or below)
   To provide maximum fuel flow from an airstart, retain the throttle of the remaining engine in the OFF position.

CAUTION

In order to reduce the possibility of engine seizure, initiation of engine restart should not be prolonged following flameout. If one or both engines flame out, do not delay the airstart attempt in order to set up windmill rpm.

If an airstart has not been accomplished within 30 seconds:

5. Throttle - OFF
6. Remaining engine - ATTEMPT AIRSTART

If neither engine can be started before reaching minimum safe ejection altitude, and the precautionary emergency approach procedure cannot be set up:

7. Notify RIO of existing emergency and intended action.
8. EJECT

AIRSTARTS

In general, airstart capability is increased by higher airspeeds and lower altitudes. However, airstarts can be made over a wide range of airspeeds, as high as approximately 48,000 feet with JP-4 fuel or 40,000 feet with JP-5 fuel.

CAUTION

In order to reduce the possibility of engine seizure, initiation of engine restart should not be prolonged following flameout. If one or both engines flame out, do not delay the airstart attempt in order to set up windmill rpm.

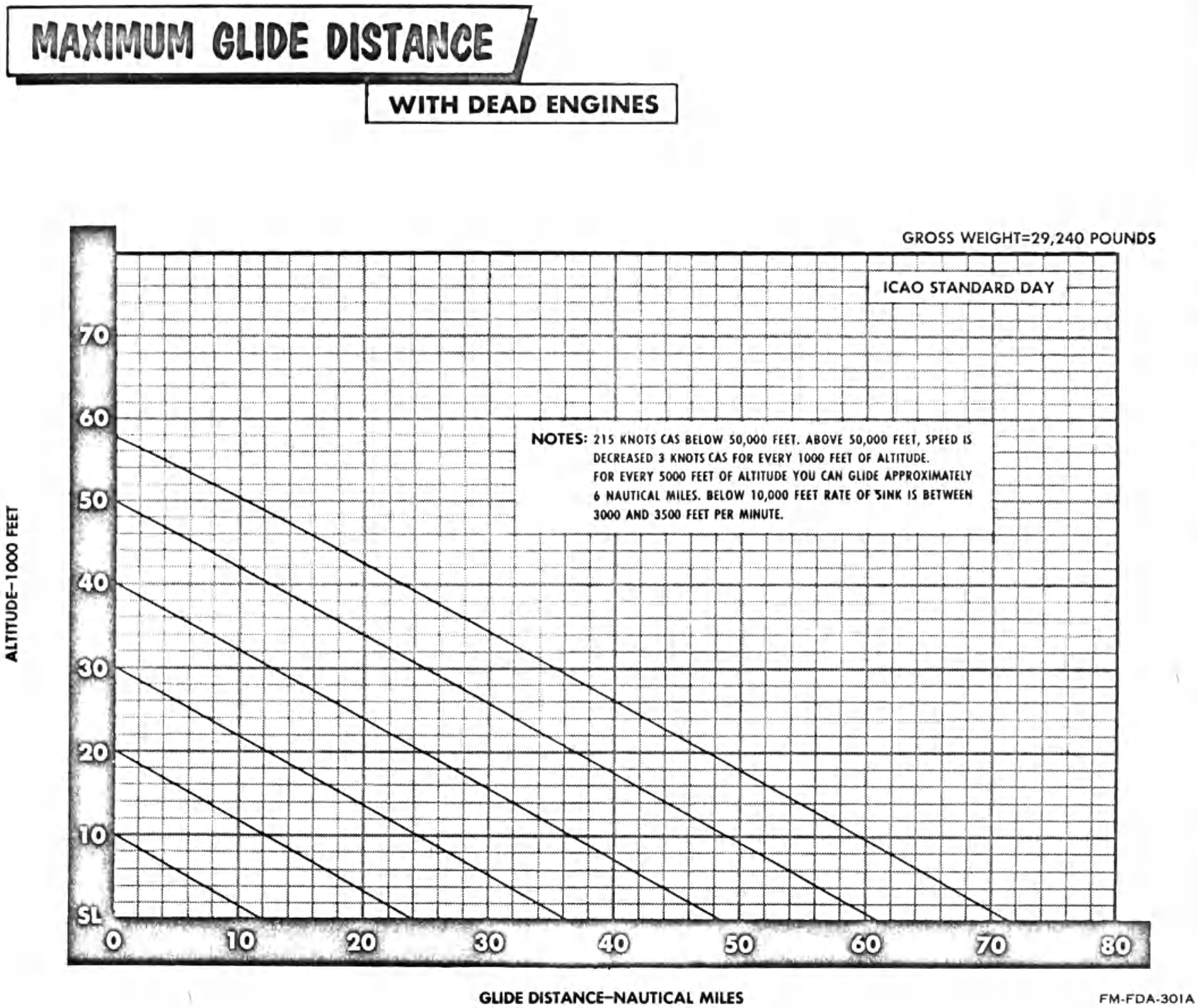


Figure 3-1

1. Establish a normal flight attitude.
2. Engine rpm - 12% MINIMUM
3. Ignition button - DEPRESS
4. Throttle - ADVANCE TO IDLE
5. A start is indicated by a rapid increase in EGT followed by an increase in RPM.
6. After the engine starts, adjust throttle as required.

In the event:

1. Light-off does not occur within 30 seconds after throttle is advanced;
2. The engine does not continue to accelerate after light-off;
3. The exhaust temperature exceeds maximum limitations;
4. The oil pressure does not attain the minimum limit at idle rpm.

retard the throttle to OFF to discontinue start. Allow engine to windmill approximately 30 seconds minimum prior to next attempted start.

Note

For windmilling restrictions (not applicable under emergency shutdown conditions), see Section V, NAVWEPS 01-245FDA-1A.

GLIDE DISTANCE

The Glide Distance Chart (figure 3-1) provides the pilot with glide distances he may attain with both engines windmilling from a given altitude and at a given airspeed. This airspeed will provide near maximum glide distance capability of the aircraft and will provide engine windmilling speed capable of keeping hydraulic pressure within safe limits.

SINGLE ENGINE LANDINGS

A single engine landing is basically the same as a normal landing (figure 2-7) except that the pattern is expanded to avoid steep turns and the final approach speeds are increased for better lateral control.

CAUTION

Stability augmentation will frequently monitor off during an engine failure or shutdown. Since lateral-directional control is marginal during the final stages of the approach without stability augmentation, re-engage the stability augmentation switch.

1. Turn off all non-essential electrical equipment.
2. Reduce gross weight to 34,000 lbs. or less (32,000 lbs. optimum).
3. Jettison external stores as required.
4. Cycle afterburner of operating engine to insure rapid light-off for possible afterburner operation.
5. Stab Aug switch - ENGAGE
6. Wind driven turbine - EXTEND
7. Wing flaps - 1/2

CAUTION

- Full flaps are not recommended during a single engine approach and landing since the engine bleed air that would be utilized for the trailing edge BLC system deprives the engine of fully rated thrust. In the 1/2 flaps configuration, trailing edge BLC is inoperative and the operating engine can deliver fully rated thrust if a wave-off is necessary.

- In the event the generator on the operating engine is lost, the flaps solenoid operated selector valves will revert to a full trail position, and the airstream will blow the flaps UP.

8. Landing gear - DOWN
9. Arresting hook - DOWN (Carrier)
10. Retrim airplane.
11. Abeam position 1-1/2-2 NM; 1000 ft. altitude; 150 knots CAS.
12. Maintain angle of attack of 20-21 units unless forward field of view is reduced and line up is affected. If line up becomes critical slowly reduce angle of attack (to prevent rapid sink rate) to 19.5 units. Corresponding airspeeds for 34,000 lb. airplane are as follows:
    a. 20-21 units - 146 knots CAS
    b. 19.5 units - 150 knots CAS

In the event of wave-off:

13. Operating engine - MAX (AFTERBURNER)

In the event of field landing:

14. Make normal touchdown.
15. Drag chute - DEPLOY

In the event of carrier landing:

16. Upon main gear touchdown:
    a. Operating engine - MAX (AFTERBURNER)
17. Upon arrestment:
    a. Operating engine - IDLE
    b. Observe flight deck crew signals.

LANDING WITH BOTH ENGINES INOPERATIVE

If a "dead stick" landing on a prepared surface is feasible, use Precautionary Emergency Approach Procedure, see figure 3-2. To practice the precautionary emergency approach procedure, extend speed brakes and adjust throttles to a power setting of 78% rpm.

FIRE

ENGINE FIRE DURING START

1. Throttles - OFF
2. Engine master switches - OFF
3. Continue to crank engine for 20 sec.
4. External compressed air units-DISCONNECTED
5. Leave airplane as quickly as possible.

ENGINE FIRE DURING TAKE-OFF

Before Airborne

If either fire warning light illuminated during take-off roll, it is preferable to abort immediately if sufficient runway is available to stop safely. Refer to Critical Field Length Charts.

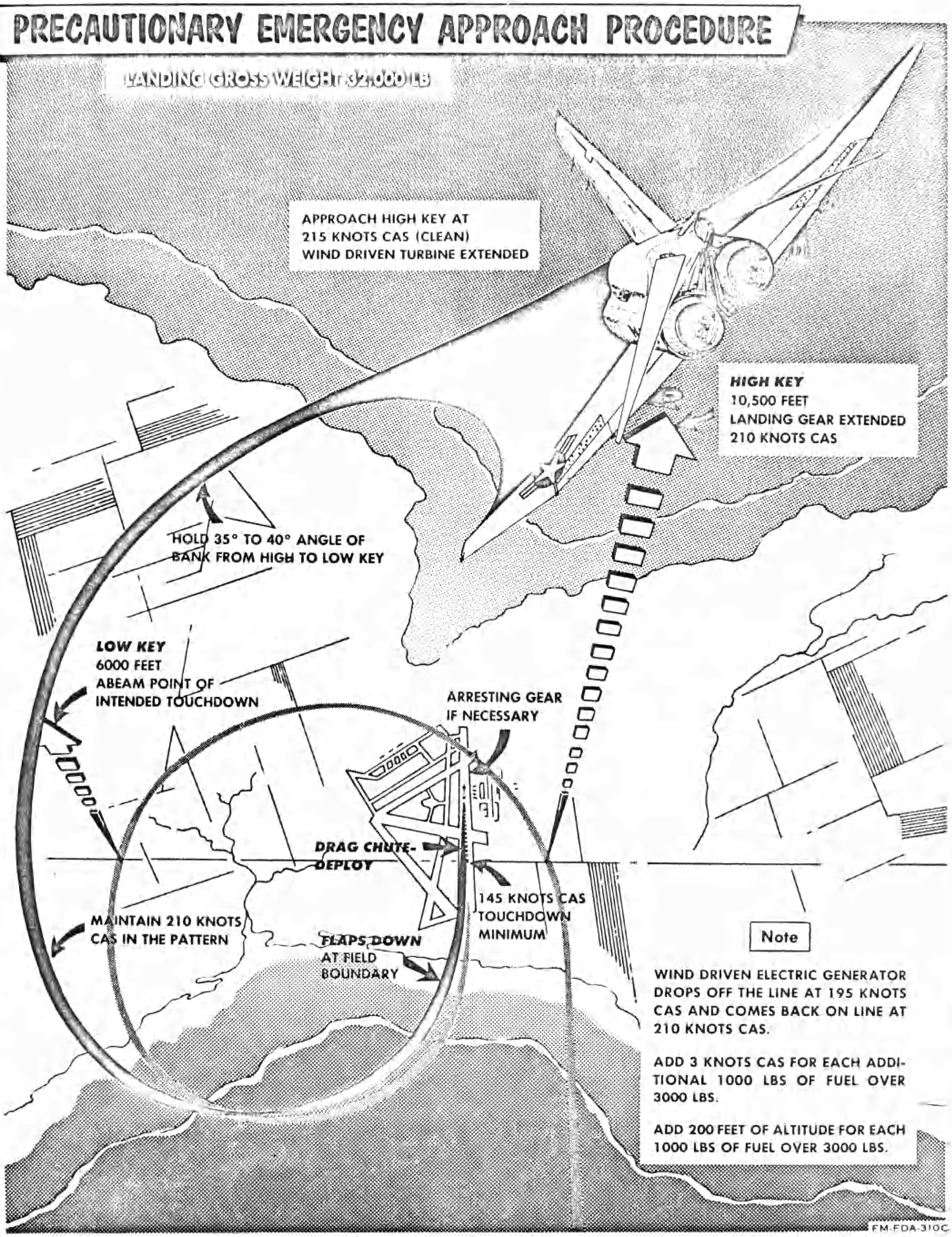
1. Throttles - OFF
2. Drag chute - DEPLOY
3. Wheel brakes - APPLY
4. Notify R.I.O. of existing emergency and intended action.
5. Wing flaps - UP (if electrical power is available)
6. Arresting gear - DOWN (if wire engagement is anticipated).

Note

- The hydraulic brakes remain operative after loss of the hydraulic power system for approximately 10 brake applications.

- If normal hydraulic braking is inadequate, use emergency brake system. Refer to Brake System Emergency Operation, this section.

PRECAUTIONARY EMERGENCY APPROACH PROCEDURE



Note

WIND DRIVEN ELECTRIC GENERATOR DROPS OFF THE LINE AT 195 KNOTS CAS AND COMES BACK ON LINE AT 210 KNOTS CAS.

ADD 3 KNOTS CAS FOR EACH ADDITIONAL 1000 LBS OF FUEL OVER 3000 LBS.

ADD 200 FEET OF ALTITUDE FOR EACH 1000 LBS OF FUEL OVER 3000 LBS.

Figure 3-2

Note

- It is possible to use differential hydraulic braking in conjunction with metered pneumatic emergency braking.

7. Aim airplane at the center of the wire.
8. As soon as stopped - LEAVE AIRPLANE

After Airborne

1. Normally operating engine - MAX (AFTERBURNER)
2. Throttle (engine indicating fire) - OFF
3. Engine master switch - OFF
4. External tanks - JETTISON (if necessary to maintain altitude)
5. Landing gear - UP
6. Wing flaps - UP
   Retract wing flaps when safe altitude and airspeed are attained.
7. If warning light remains on and fire is apparent - EJECT
8. If warning light goes out and no fire is apparent, land as soon as possible.

WARNING

Make a single engine landing, DO NOT attempt to restart engine.

9. If fire warning light remains on and no fire is apparent, climb to a safe altitude and investigate.
   Climb to a safe ejection altitude and determine positively that no fire exists. If there is evidence of a fire - EJECT. If there is no fire make a single engine landing. DO NOT attempt to restart the engine.

Note

Illumination of "Fire" or "Overheat" warning light shall be logged on the yellow sheet (OPNAV FORM 3760-2).

FIRE WARNING DURING FLIGHT

1. Engine indicating fire - THROTTLE BACK
   Throttle back on the engine indicating fire to reduce EGT. If warning light remains lighted;
   a. Throttle - OFF
   b. Engine master switch - OFF
2. If fire warning light goes out and there is no evidence of continuing fire, make a single engine landing as soon as possible.
3. If fire warning light remains illuminated and evidence of fire is apparent, EJECT.
4. If fire warning light remains illuminated and fire is not apparent, proceed to a safe ejection altitude and investigate thoroughly. If there is no evidence of fire, make a single engine landing as soon as possible. DO NOT attempt to restart the engine.

ENGINE FIRE DURING SHUTDOWN

If external compressed air source and external electrical power is hooked up or readily available:

1. Throttles - OFF
2. Engine master switches - OFF

When external power is connected to airplane:

3. Generator control switches - EXT ON
4. Engine start switch - LEFT or RIGHT (engine indicating fire)
5. Motor engine until fire is extinguished.

If external compressed air source and external electrical power is not available:

1. Throttles - OFF
2. Engine master switches - OFF
3. As soon as possible - LEAVE AIRPLANE

ELECTRICAL FIRE

1. Generator control switches - OFF
2. All electrical switches - OFF
3. When fire subsides, generator control switches - ON

If fire still persists when generator control switches are turned on:

4. Emergency wind driven turbine - EXTEND
5. Generator control switches - OFF
   This will supply power to the essential buses only.
6. Land as soon as possible.

If no fire is apparent when generator control switches are turned on:

7. Individually reposition electrical equipment switches to ON beginning with the most essential equipment first.
8. If malfunctioning item is found, turn electric power switch OFF and pull applicable circuit breaker.

ELIMINATION OF SMOKE AND FUMES

1. Descend to below 34,000 feet.
2. Emergency ventilating knob - PULL
3. If step 2 fails to clear cockpit - JETTISON CANOPY

EJECTION

Escape from the airplane in flight should be made with the ejection seat, and it is an established procedure that the radar observer eject first.

BEFORE EJECTION SEQUENCE

TYPICAL BOTH COCKPITS

IF TIME AND CONDITIONS PERMIT

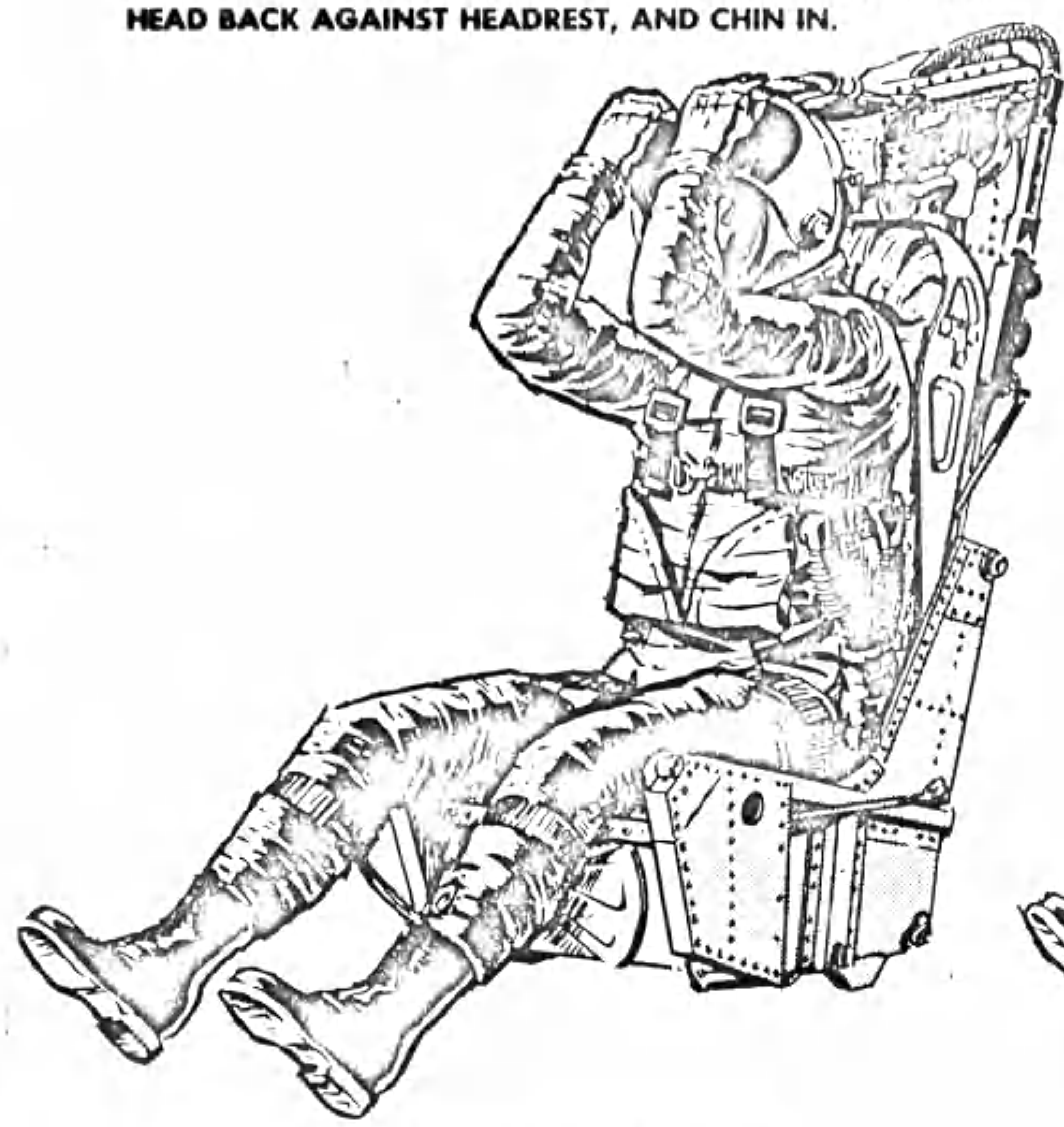
- SLOW AIRCRAFT AS MUCH AS POSSIBLE.
- STOW ALL LOOSE EQUIPMENT.
- PULL EMERGENCY VENTILATING KNOB TO MINIMIZE DECOMPRESSION EFFECTS WHEN JETTISONING CANOPY.

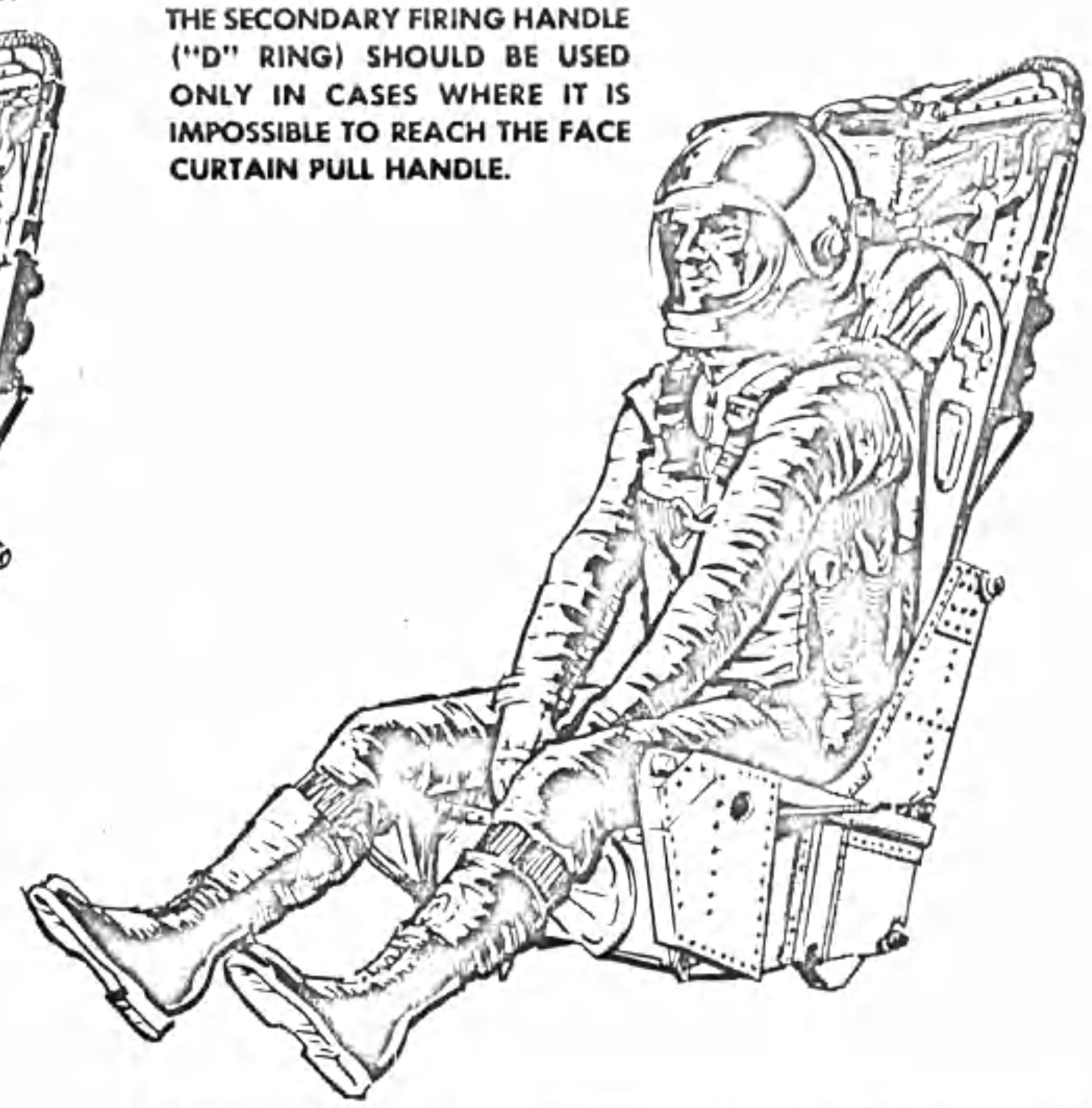
1. ASSUME PROPER EJECTION POSITION

ADJUST SEAT POSITION SO EYES WILL BE AT, OR BELOW GUNSIGHT LEVEL. BRACE THIGHS ON SEAT CUSHION, LEGS EXTENDED. SIT ERECT, BUTTOCKS BACK, SPINE STRAIGHT. HEAD BACK AGAINST HEADREST, AND CHIN IN.

Note

THE SECONDARY FIRING HANDLE ("D" RING) SHOULD BE USED ONLY IN CASES WHERE IT IS IMPOSSIBLE TO REACH THE FACE CURTAIN PULL HANDLE.





2. FACE CURTAIN HANDLE—PULL

REACH OVERHEAD, WITH PALMS AFT KEEPING ELBOWS TOGETHER, GRASP FACE CURTAIN HANDLE, PULL FACE CURTAIN AND MAINTAIN DOWNWARD FORCE UNTIL STOP IS ENCOUNTERED. WHEN CANOPY JETTISONS, CONTINUE PULLING FACE CURTAIN UNTIL FULL TRAVEL IS REACHED.

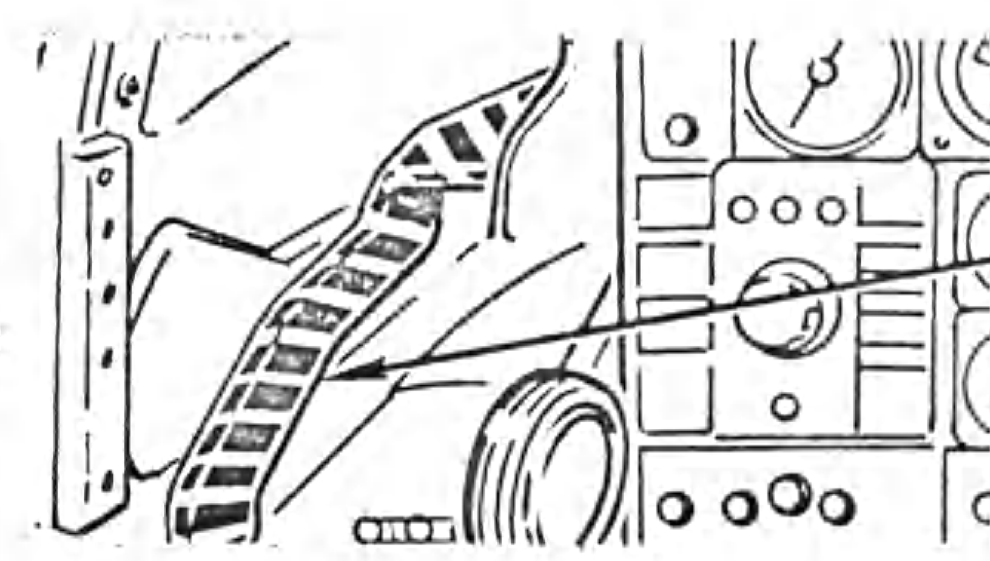
2. SECONDARY EJECTION HANDLE—PULL

POSITION ARMS BETWEEN LEGS WITH PALMS AFT, GRASP SECONDARY EJECTION HANDLE ("D" RING) AND PULL UP UNTIL STOP IS ENCOUNTERED. WHEN CANOPY JETTISONS, CONTINUE PULLING UP ON SECONDARY EJECTION HANDLE UNTIL FULL TRAVEL IS REACHED.

IF CANOPY FAILS TO JETTISON

- EMERGENCY CANOPY RELEASE HANDLE—PULL

IF CANOPY FAILS TO JETTISON, RELAX GRIP ON FACE CURTAIN HANDLE, AND WHILE HOLDING HANDLE WITH ONE HAND, PULL EMERGENCY CANOPY RELEASE HANDLE. WHEN CANOPY JETTISONS, AGAIN GRASP FACE CURTAIN HANDLE WITH BOTH HANDS AND PULL UNTIL FULL TRAVEL IS REACHED.



IF CANOPY STILL FAILS TO JETTISON

- CANOPY CONTROL HANDLE—OPEN
- MANUAL CANOPY UNLOCK HANDLE—PULL

WHEN AIR LOADS SEPARATE CANOPY, AGAIN GRASP FACE CURTAIN HANDLE WITH BOTH HANDS AND PULL UNTIL FULL TRAVEL IS REACHED.

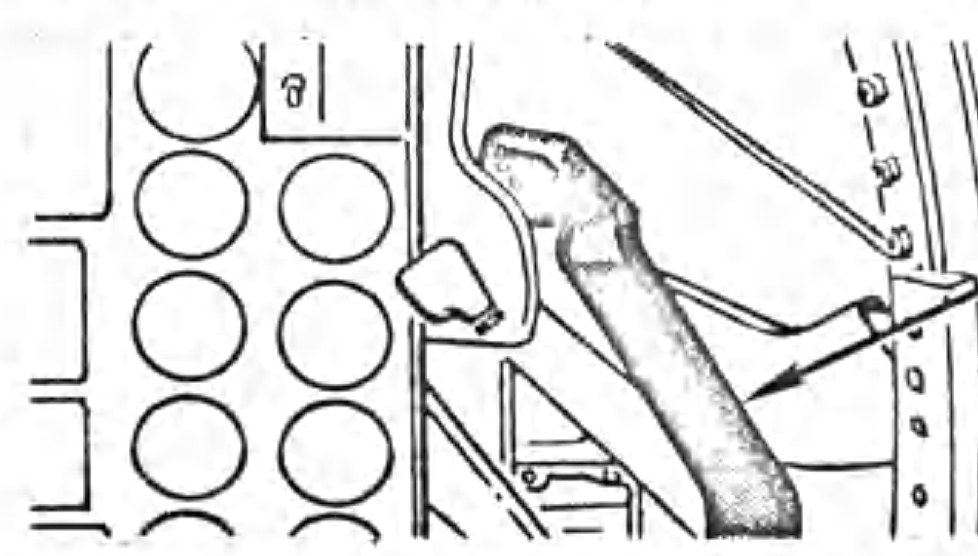


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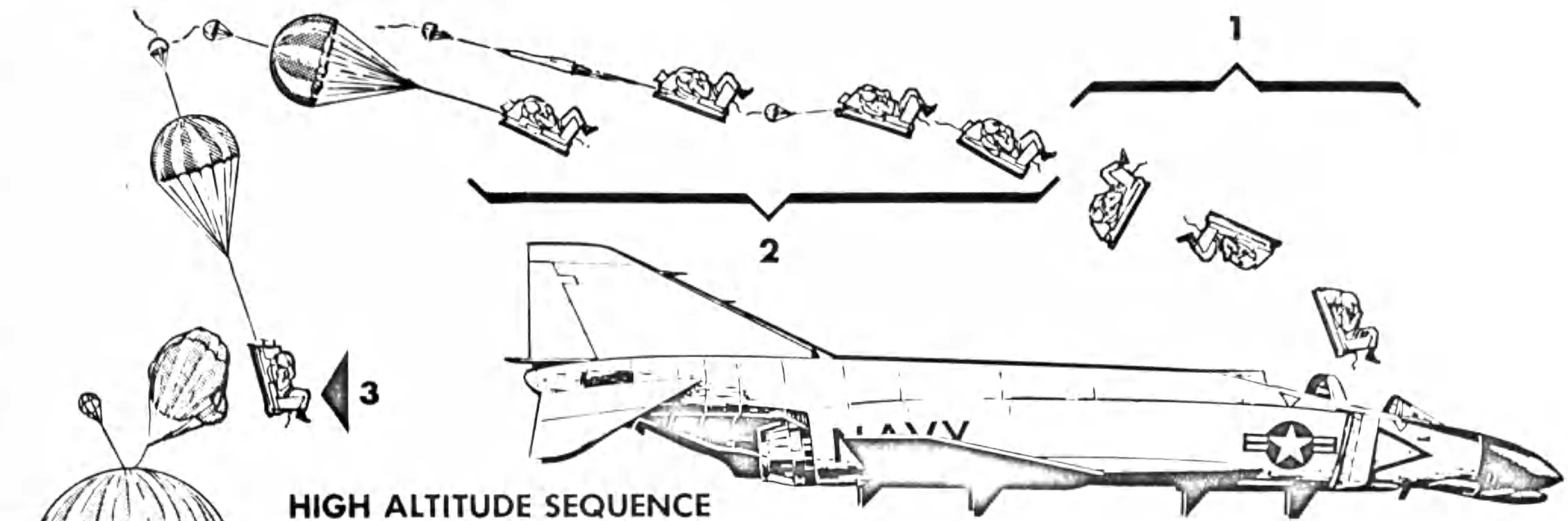
Figure 3-3

AFTER EJECTION SEQUENCE



HIGH ALTITUDE SEQUENCE

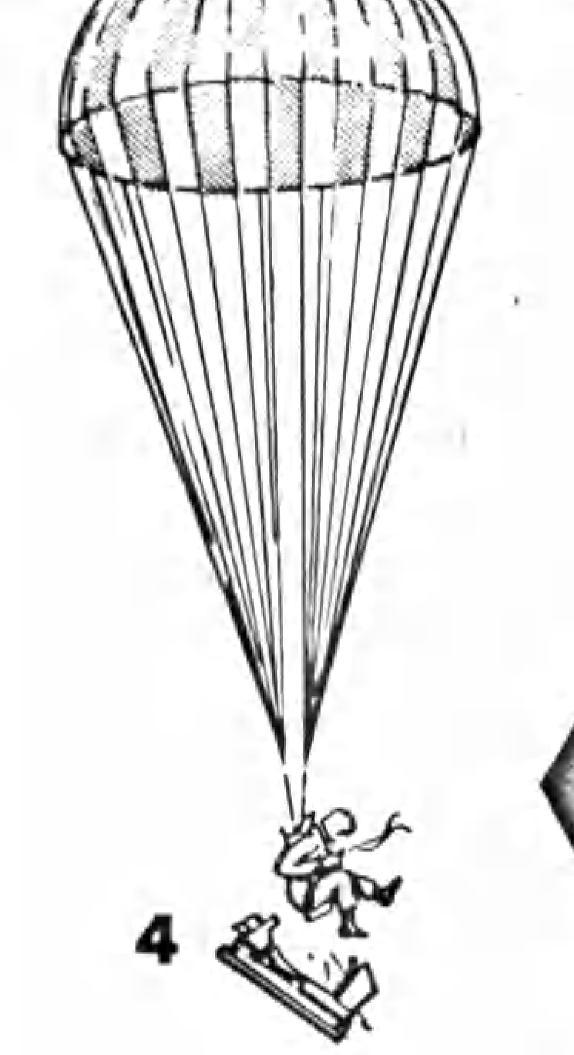
PULL FACE CURTAIN TO INITIATE THE EJECTION SEQUENCE, CANOPY JETTISONS AND PULLS INTERLOCK BLOCK, PERMITTING CURTAIN TO EXTEND FULLY AND FIRE THE CATAPULT.

1 SEAT IS PROPELLED UP GUIDE RAIL. OCCUPANT'S LEGS ARE RESTRAINED, EMERGENCY OXYGEN IS ACTUATED, TIME RELEASE MECHANISM AND DROGUE GUN ARE TRIPPED, AND EMERGENCY IFF IS ACTUATED.

2 DROGUE GUN FIRES APPROX. ½ SEC. AFTER EJECTION, DEPLOYS CONTROLLER DROGUE, WHICH IN TURN, DEPLOYS STABILIZER DROGUE. SEAT IS STABILIZED AND DECELERATED BY DROGUE CHUTES.

3 SEAT AND OCCUPANT DESCEND RAPIDLY THRU UPPER ATMOSPHERE. WHEN AN ALTITUDE OF APPROX. 10,000 FT. IS REACHED, THE BAROSTAT RELEASES THE ESCAPEMENT MECHANISM, WHICH IN TURN, ACTUATES TO RELEASE THE OCCUPANT'S HARNESSING, LEG RESTRAINT LINES, AND CHUTE RESTRAINT STRAPS. THE DROGUE CHUTES PULL THE LINK LINE TO DEPLOY THE PERSONNEL PARACHUTE.

10,000 FT IF NECESSARY PROCEED WITH



4 OCCUPANT IS HELD TO SEAT BY STICKER CLIPS UNTIL OPENING SHOCK OF PARACHUTE SNAPS SEAT FROM HIM PERMITTING NORMAL DESCENT.

LOW ALTITUDE SEQUENCE

SAME AS CORRESPONDING STEPS **1-2** IN HIGH ALTITUDE SEQUENCE EXCEPT

3 APPROX. 1½ SEC. AFTER EJECTION THE ESCAPEMENT MECHANISM ACTUATES TO RELEASE THE OCCUPANT'S HARNESSING, LEG RESTRAINT LINES, AND CHUTE RESTRAINT STRAPS. THE DROGUE CHUTES PULL THE LINK LINE TO DEPLOY THE PERSONNEL PARACHUTE.

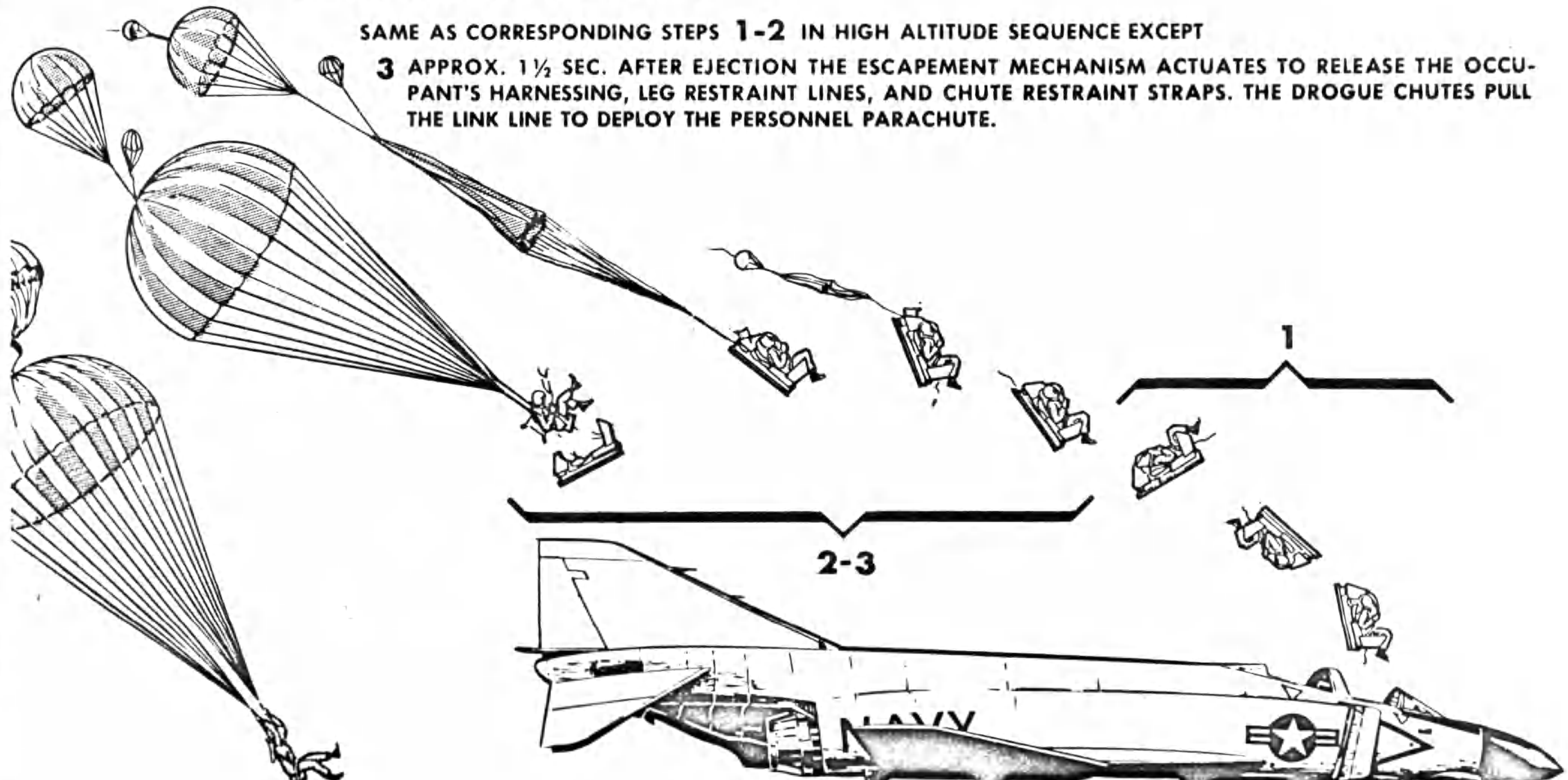


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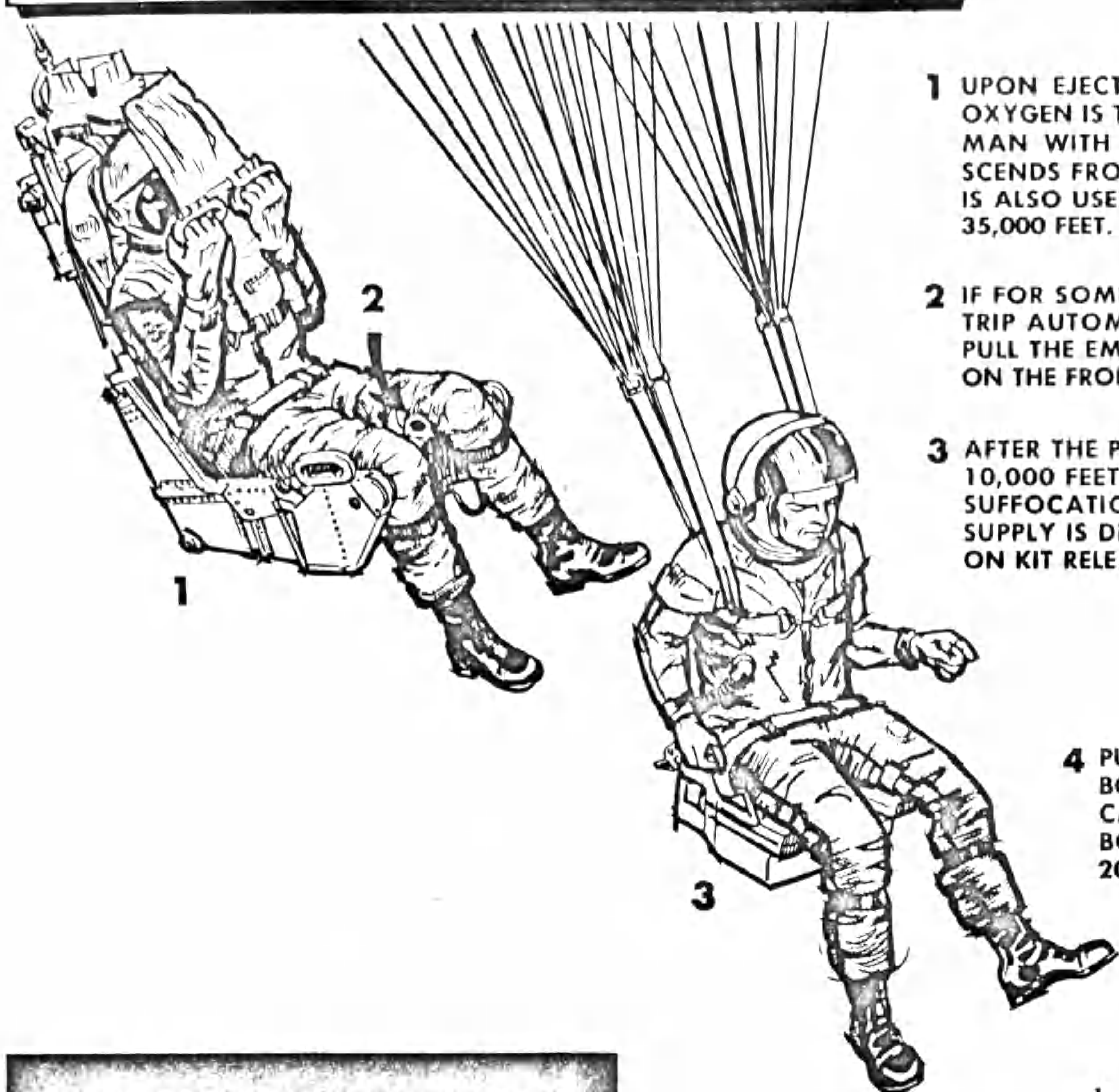
Figure 3-4

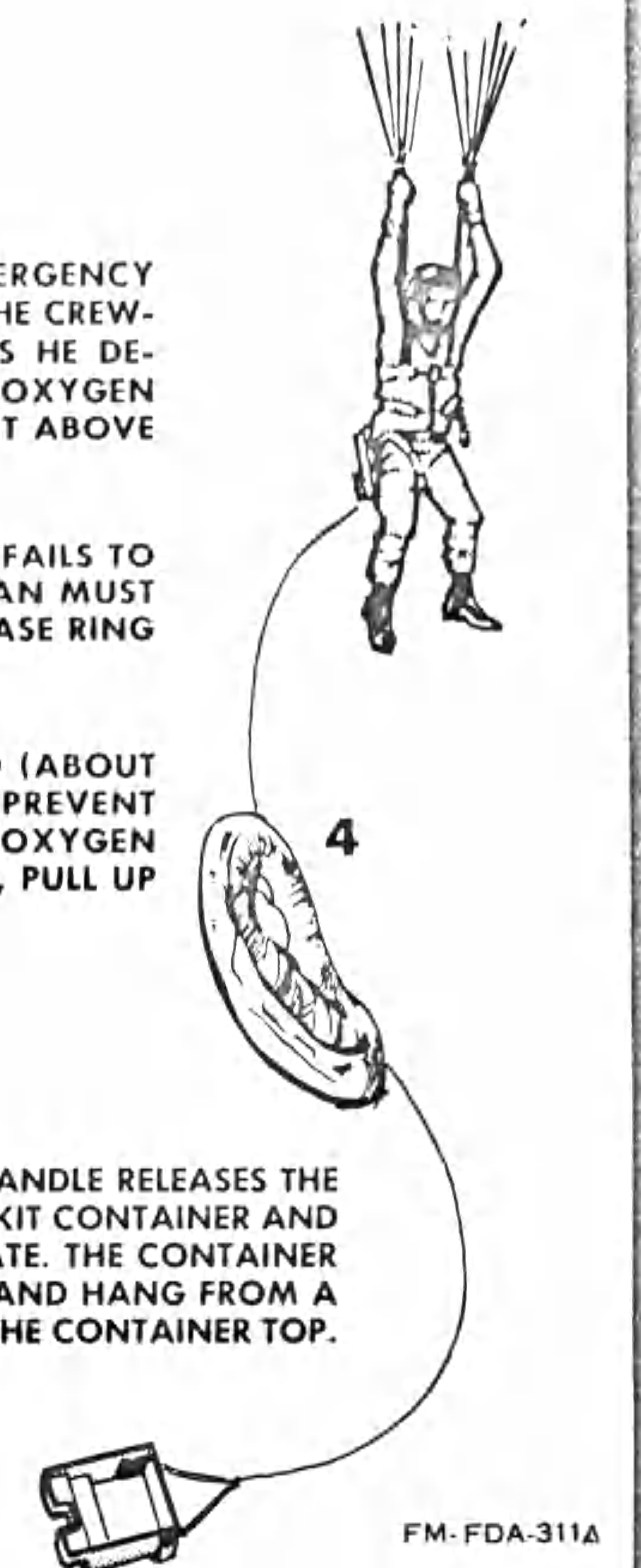
SURVIVAL KIT DEPLOYMENT



1 UPON EJECTION, SURVIVAL KIT EMERGENCY OXYGEN IS TRIPPED. THIS PROVIDES THE CREWMAN WITH BREATHING OXYGEN AS HE DESCENDS FROM HIGH ALTITUDES. THE OXYGEN IS ALSO USED TO PRESSURIZE THE SUIT ABOVE 35,000 FEET.

2 IF FOR SOME REASON THE OXYGEN FAILS TO TRIP AUTOMATICALLY, THE CREWMAN MUST PULL THE EMERGENCY OXYGEN RELEASE RING ON THE FRONT OF THE SURVIVAL KIT.

3 AFTER THE PARACHUTE HAS OPENED (ABOUT 10,000 FEET) OPEN FACE VISOR TO PREVENT SUFFOCATION WHEN EMERGENCY OXYGEN SUPPLY IS DEPLETED. IF OVER WATER, PULL UP ON KIT RELEASE HANDLE.



4 PULLING THE KIT RELEASE HANDLE RELEASES THE BOTTOM OF THE SURVIVAL KIT CONTAINER AND CAUSES THE RAFT TO INFLATE. THE CONTAINER BOTTOM AND RAFT DROP AND HANG FROM A 20 FOOT LINE SECURED TO THE CONTAINER TOP.

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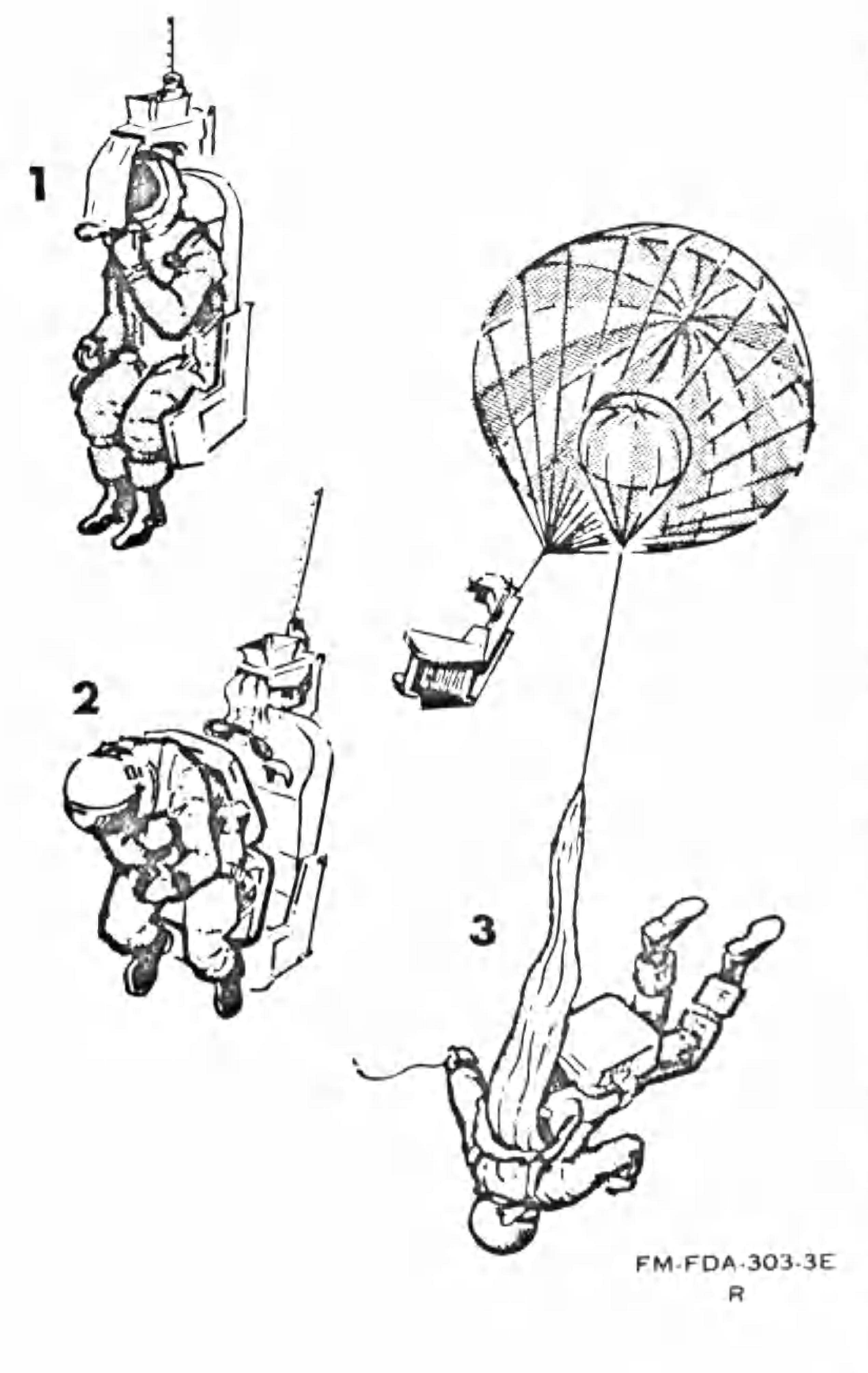
MANUAL SEPARATION



SHOULD THE TIME RELEASE MECHANISM FAIL TO OPERATE AUTOMATICALLY, THE OCCUPANT WOULD MANUALLY SEPARATE FROM THE SEAT AS FOLLOWS:

1 ACTUATE EMERGENCY HARNESS RELEASE HANDLE ON RIGHT SIDE OF SEAT TO ITS FULL AFT POSITION. THIS ACTION WILL RELEASE THE RESTRAINT HARNESS AND LEG RESTRAINT CORD, AND ON AIRPLANES 146817c THRU 148275e, A CARTRIDGE ACTUATED GUILLOTINE WILL SEVER THE LINK LINE BETWEEN THE PERSONNEL CHUTE AND DROGUE CHUTE. ON AIRPLANES PRIOR TO 146817c, A STATIC LINE DISCONNECT WILL AUTOMATICALLY SEPARATE THE PERSONNEL CHUTE FROM THE DROGUE CHUTE LINK LINE DURING SEPARATION FROM THE SEAT.

2 PUSH FREE OF STICKER CLIPS AND CLEAR OF THE SEAT.

3 PULL PARACHUTE RIPCORD "D" RING (LOCATED ON LEFT SHOULDER) AND MAKE A NORMAL PARACHUTE DESCENT TO THE DECK.

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Figure 3-5

The study and analysis of escape techniques by means of the ejection seat reveals that:

1. Ejection at airspeeds ranging from stall speed to 525 knots CAS results in relatively minor forces being exerted on the body, thus reducing injury hazard.
2. Appreciable forces are exerted on the body when ejection is performed at airspeeds of 525 to 600 knots CAS rendering escape more hazardous.
3. At speeds above 600 knots CAS, ejection is extremely hazardous because of excessive forces on the body.

Ejection at low altitudes is facilitated by pulling the nose of the airplane above the horizon ("zoom up" maneuver). This maneuver affects the trajectory of the ejection seat providing a greater increase in altitude than if ejection is performed in a level flight attitude.

This gain in altitude will increase the time available for separation from the seat and deployment of parachute. Ejection should not be delayed when the aircraft is in a descending attitude and cannot be leveled out. When circumstances permit, slow the airplane down prior to ejection to reduce the forces exerted on the body. The emergency harness release handle should **never** be actuated before ejection for the following reasons:

1. Actuating the emergency harness release handle creates a hazard to survival during uncontrollable flight, since negative "g" forces may prevent the crew from assuming the correct ejection position. A full understanding of the particular situation must be established between crew members so that there is no mistaken or time consuming activity.
2. Actuating the emergency harness release handle creates a hazard to survival if the pilot decides that he has insufficient altitude for ejection and is required to proceed with a forced landing. Once the emergency harness release handle has been pulled, the lap belt shoulder harnessing is released and cannot be refastened in flight.
3. Actuating the emergency harness release handle prior to ejection causes the occupant to separate from the seat immediately after ejection, and severe shock loads will be imposed on the body.

HIGH ALTITUDE EJECTION

For a high altitude ejection, the basic ejection procedure is applicable. Furthermore, the "zoom up" maneuver is still useful to slow the airplane to a safer ejection speed or provide more time and glide distance as long as an immediate ejection is not mandatory.

EJECTION SEAT FAILURE (BAILOUT)

In the event that the canopy has been jettisoned but the ejection seat fails, proceed as follows:

1. Reduce speed as much as possible.
2. IFF - EMERG
3. Emergency harness release handle - PULL
   Pull up on the emergency harness release handle on the right side of the seat bucket to disconnect the parachute harness and the leg restraint harness from the seat. Effective airplanes 146817c thru 148275e, this handle also fires the cartridge actuated guillotine which severs the link line between the drogue chute and the personnel parachute.
4. Full nose down trim while holding airplane level.
5. Reach over shoulders and grasp parachute from the horseshoe fitting on the seat.

CAUTION

If the parachute risers are not grasped, and the chute pulled off the horseshoe, the chute will not clear the protrusion at the top of the seat when attempting to leave.

6. Roll inverted, push stick forward and push sharply to fall clear of the airplane.
7. When clear of airplane and at 10,000 feet or below, parachute "D" ring - PULL

SURVIVAL KIT DEPLOYMENT

During a parachute descent the crewmember may desire to jettison the survival kit, or hold it by its carrying strap to be dropped just prior to ground contact. The reason for jettisoning or hand carrying the survival kit is to reduce the bulk and weight of the crewmember, which will minimize the possibility of injuries occurring upon ground impact due to the survival kits weight and normal carrying position. It would probably be to the crewmembers advantage to jettison or hand carry the survival kit when making a parachute descent over land. In the event of a night ejection over land, the survival kit should be removed from its normal carrying position, however, if it cannot positively be determined that the crewmember will touch down on land, the survival kit should be hand carried and dropped just prior to impact. In the event there is insufficient time to jettison or hand carry the survival kit, the life raft with the survival kit should be released. In all over water ejections, the survival kit should be left in its normal carrying position and the life raft should be released during parachute descent. To jettison or hand carry survival kit, proceed as follows:

1. Open face visor or disconnect oxygen mask.
2. Composite disconnect release knob - PULL
3. Release left lap belt release fitting.
4. Take hold of survival kit lifting handle with right hand.
5. Release right lap belt release fitting.

EJECTION ALTITUDE REQUIREMENTS

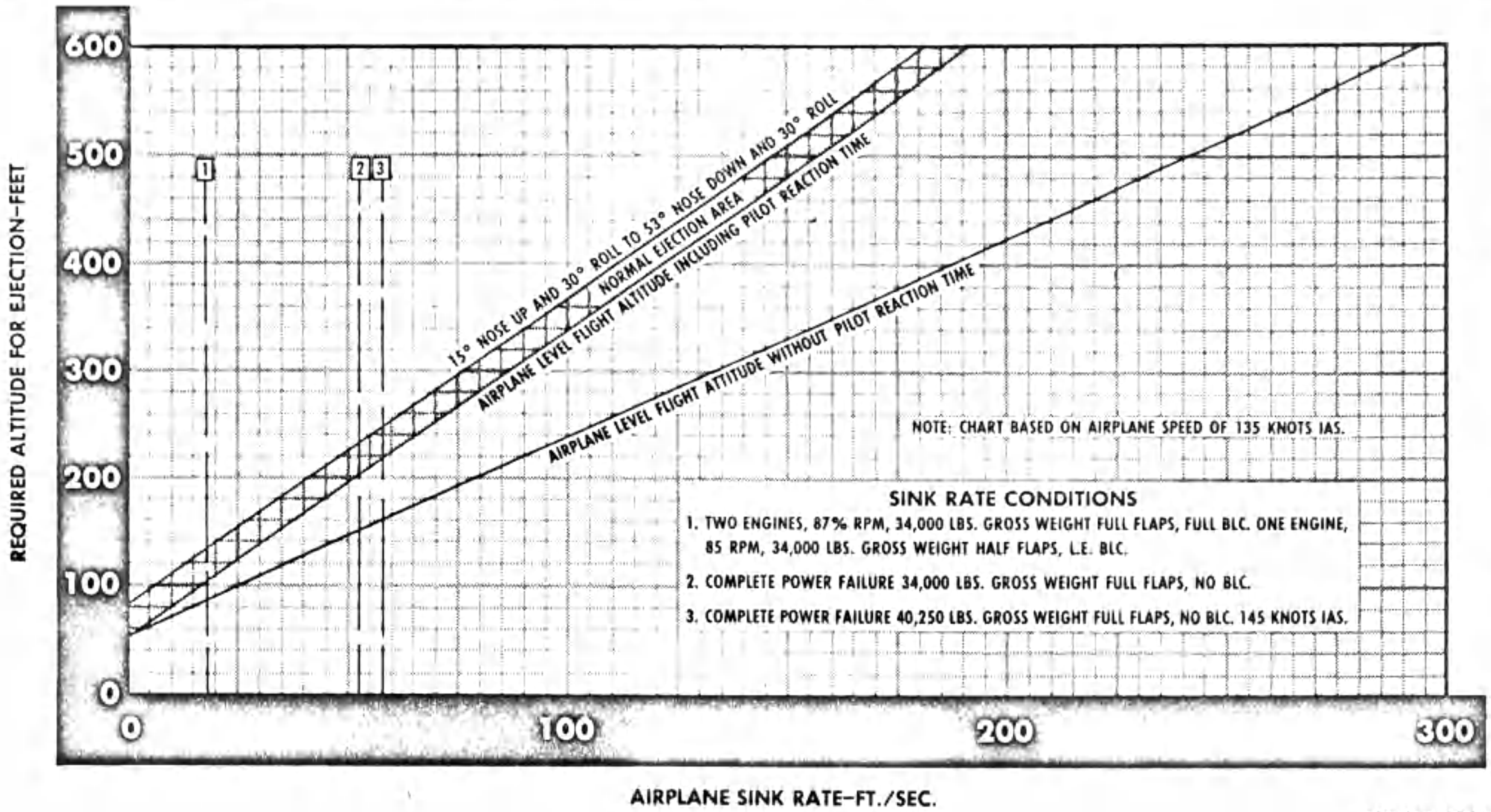

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Figure 3-6

LANDING EMERGENCIES

FORCED LANDING

WARNING

All forced landings shall be made with the landing gear extended, regardless of terrain. A greater injury hazard is present whenever emergency landings are made with the landing gear retracted. Increased airspeed or nose high angle of impact during landings with landing gear retracted is common practice and contributes greatly to pilot injury and damage to the airplane. This nose high attitude causes the airplane to "slap" the ground on impact, subjecting the pilot to possible spinal injury. Less airplane damage will result with the gear extended.

It is recommended that a gear-up landing NOT be attempted with this aircraft; the crew should EJECT. If a forced landing is unavoidable, proceed as follows:

1. Wind driven turbine - EXTEND
   Extend the wind driven turbine if needed to supplement the power control hydraulic or electrical power. Both systems will be inoperative if the engines are windmilling.
2. If time and conditions permit, dump or burn excess fuel.
3. Notify R.O. of existing emergency and intended action.
4. Harness inertia reel handle - LOCKED
5. Canopies - JETTISON (forward canopy first)
   The aft canopy should be jettisoned last to preclude the possibility of the forward canopy entering the aft cockpit when jettisoned.
6. Armament - JETTISON
7. Landing gear - DOWN AND LOCKED
8. Wing flaps - DOWN
9. External tanks - RETAIN IF EMPTY
   Empty external tanks should be retained to absorb the shock of landing.
10. Make normal approach.

Upon touchdown:

11. Drag chute - DEPLOY
12. Engines - SHUTDOWN
    a. Throttles - OFF
    b. Engine master switches - OFF
13. Generator control switches - OFF
14. As soon as stopped - CLEAR AIRPLANE

LANDING GEAR UNSAFE

An unsafe gear indication does not necessarily constitute an emergency. The unsafe indication could be caused by a malfunction with the indicating system or the result of incorrect gear lowering procedure coupled with a low pressure condition of the utility hydraulic system. Upon initial detection of unsafe gear indication, proceed as follows:

Emergency Extension and Locking

1. Airspeed - 250 knots CAS OR BELOW

If utility hydraulic pressure is within limits:

2. Landing gear - RECYCLE
3. Check gear indicates down and locked.

If unsafe condition still exists:

4. Landing gear circuit breaker - PULL
5. Landing gear handle - DOWN-PULL AFT
   When the landing gear handle is down, pull aft to actuate the emergency pneumatic system.
6. Check gear indicates down and locked.

If unsafe condition still exists:

7. Yaw airplane to assist in locking main gear.
8. Check gear indicators and have gear visually checked by another airplane or by tower "fly-by".

Landing With Gear Up or Unsafe

The following factors should be considered before landing:

Availability of mid field or short field arresting gear near the approach end of the runway.
Crosswind effect.
Refer to Crosswind Landing, Section II.
Utility hydraulic system failure.
With a utility system failure, normal rudder control, normal wing flaps, and normal braking will not be available.

1. Request foam on runway if desired.
2. Armament - JETTISON
3. Lighten airplane.
   Lighten airplane by dumping or burning excess fuel.
4. External tanks - RETAIN IF EMPTY
5. Notify R.O. of existing emergency and intended action.
6. Inertia reel handle - LOCKED (if installed)
7. Make normal approach.

If one main gear is up or unsafe:

8. Land on side of runway opposite failed or unsafe gear.

If nose gear or all gear is up or unsafe:

9. Land in center of runway.
10. At 120 knots IAS - EASE NOSE TO RUNWAY.
11. Drag chute - DEPLOY
12. Brakes - APPLIED
13. If gear is up or has collapsed - SHUTDOWN ENGINES
14. As soon as stopped - LEAVE AIRPLANE
15. If gear has not collapsed, roll straight ahead until stopped, keep engines running until crash crew has installed ground locks.

WARNING

Do not attempt arrestment if nose gear is up and main gear is extended.

ARRESTED LANDING

1. Request foam if desired.
2. Notify RIO of existing emergency and intended action.
3. External tanks - RETAIN IF EMPTY
4. Arresting gear - DOWN
5. Plan to engage wire shortly after touchdown.
6. Aim for center of wire.
7. Drag chute - DEPLOY
8. Release brakes 100 feet prior to engaging wire.
9. Flaps - RETRACT
10. Apply slightly aft stick.

After engagement,

11. If gear is up or has collapsed - SHUTDOWN ENGINES
12. As soon as stopped - LEAVE AIRPLANE
13. If gear has not collpased, roll straight ahead until stopped, keep engines running until crash crew has installed ground locks.

BLOWN TIRE

A situation may occur when the pilot must land with a blown tire, or the tire may rupture during ground roll. A blown tire at high speed will require immediate control action to keep the aircraft aligned with the runway. Proceed as follows:

1. Make a normal final approach.
2. Land on side of runway opposite blown tire.
3. Touchdown with weight on undamaged tire.
4. Drag chute - DEPLOY
5. Use light opposite braking and nose wheel steering to slow aircraft and maintain alignment.

CAUTION

If possible, do not shutdown engines until adequate fire fighting equipment is available. The damaged wheel will be either on fire or very hot and fuel drained overboard after engine shutdown could contact the hot wheel causing a fire.

NO-FLAPS LANDING

A no flaps landing is basically the same as a normal landing (figure 2-7) except that the pattern is expanded to avoid steep turns; the downwind, base leg, and final approach speeds are increased 22 knots to provide adequate lateral control.

1. Fly a wide normal pattern.
2. Wing flap lever - RETRACT

3. Establish at least a one mile low angle straight in approach.
4. Maintain applicable approach speed.

Total Fuel Remaining Lbs.	Approximate Final Approach Speed Kts. CAS
3000	157
4500	160
6000	163
7500	166
9000	169
10,500	172
12,000	175

5. Maintain a mild rate of descent.
6. Fly the aircraft down to the runway.
   Do not flare the aircraft or "chop" power prior to crossing the end of the runway.
7. Make normal touchdown and roll out.

BOUNDARY LAYER CONTROL SYSTEM FAILURES

A boundary layer control (BLC) system failure will effect the handling characteristics and approach speeds of the airplane. This (BLC) system failure usually will not effect the complete BLC system, but rather a portion of the system, and will probably be of one of the following variations:

a. Trailing edge BLC inoperative on one side.
b. Leading edge BLC inoperative on one side.
c. Leading and trailing edge BLC inoperative on the same side.

The BLC failure will probably occur prior to, or in the transition to flaps down during a landing approach, with the result being an asymmetric BLC condition. The asymmetric BLC condition has been found to be safe and easily controllable even with both leading edge and trailing edge BLC inoperative on the same side. A more detailed description of the characteristics and safe handling procedures for the above mentioned failures follow below:

Trailing Edge BLC Inoperative on One Side

This condition is characterized by a moderate roll when the flaps reach full down. Trim requirements vary only slightly as speed is reduced. Fly a normal "on speed" angle of attack. This will increase airspeed 5 to 7 knots. Increasing the angle of attack in order to fly normal airspeeds will provide satisfactory lateral control, but the increased angle of attack impairs the view over the nose of the airplane. If the above failure occurs proceed as follows:

1. Airplane - RETRIM
2. Fly normal "on speed" angle of attack.
3. Increase minimum approach speed 4 knots.

Leading Edge BLC Inoperative on One Side

This condition is characterized by little or no lateral retrim requirements when flaps are first lowered, followed by increased trim requirements as airspeed is reduced. Fly "on speed" angle of attack and/or an airspeed six knot higher than normal for the gross weight involved. Use caution on flaring since increasing angle of attack will lower rolling power and increase lateral control required for BLC off. Maintain a constant attitude to touchdown if possible. If the above failure occurs proceed as follows:

1. Airplane - RETRIM
2. Fly normal "on speed" angle of attack.
3. Increase minimum approach speed 6 knots.

Leading and Trailing Edge BLC Inoperative on the Same Side

The initial flap down lateral trim requirements of the trailing edge is first apparent followed by increased trim as airspeed decreases. Full lateral trim will be required well above approach airspeed. Minimum approach airspeed should be increased 18 knots to 145 knots CAS at 30,000 lbs. gross weight. Mild buffet will be noticeable at all airspeeds. If the above failure occurs proceed as follows:

1. Airplane - RETRIM
2. Increase minimum approach speed 18 knots.

There is no reason to raise flaps and land at 155-160 knots to avoid an asymmetrical BLC condition. In general for either type failure, a 5-10 knots increase will give normal angle of attack readings and satisfactory approach control.

BOUNDARY LAYER CONTROL SYSTEM MALFUNCTION

A boundary layer control system malfunction will be indicated by the illumination of the "BLC Malfunction" warning light. The only type of malfunction indicated by the light will be a BLC valve stuck open when the flaps are up. There is a possibility of rectifying this type of malfunction by reducing the engine power settings. The lower engine rpm will reduce the amount of engine bleed air being supplied to the BLC system, and may allow the BLC valve to close. In the event of a BLC malfunction ("BLC Malfunction" light illuminated) proceed as follows:

1. Reduce engine power settings to 90% rpm.

If above procedure fails to allow valve to close and extinguish "BLC Malfunction" warning light:

1. Flap switch - DOWN
2. Land as soon as practicable.

EMERGENCY ENTRANCE

Procedures and precautions to be observed by rescue personnel when assisting the crew from the airplane following a crash landing are outlined in figure 3-7.

EMERGENCY ENTRANCE

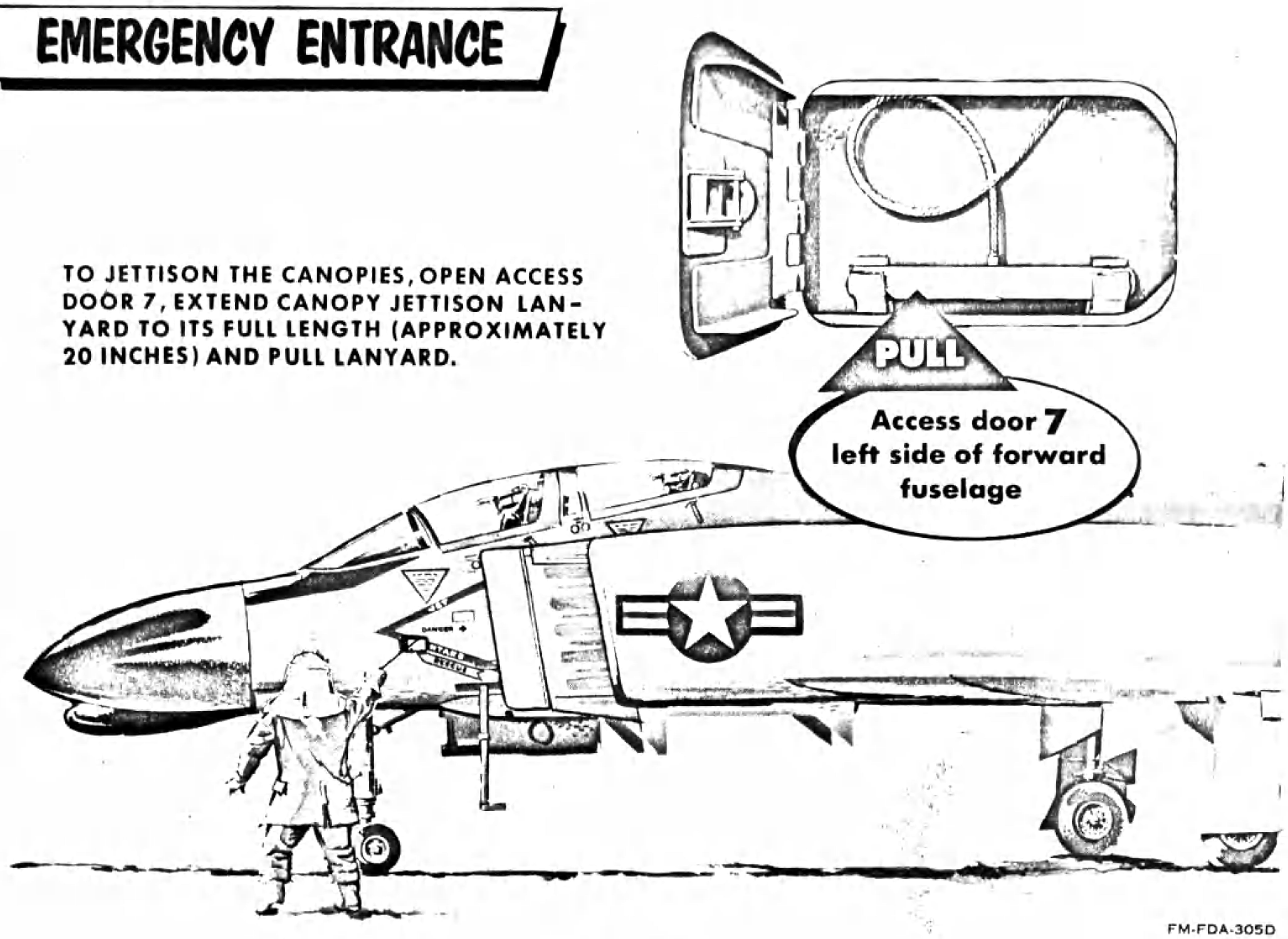

Figure 3-7

DITCHING

Ditching the airplane should be the pilot's last choice. All survival equipment is carried by the crewmember; thus ejection is advisable. However, if altitude and situation demand ditching, the procedures set forth on the ditching chart (figure 3-8) are believed to be the fastest and safest way of evacuating the cockpit, and should be observed. Two alternate methods of evacuating the cockpit are available, and procedures for each follow.

TO EVACUATE THE COCKPIT TAKING SURVIVAL KIT AND PARACHUTE ALONG, PROCEED AS FOLLOWS:

1. Emergency harness release handle - PULL
   After landing impact, pull up on the emergency harness release handle to free the shoulder harness, lap belt, and leg restraint lines.
2. Stand up to release survival kit sticker clips from the seat.

TO EVACUATE THE COCKPIT LEAVING BOTH THE PARACHUTE AND SURVIVAL KIT IN THE AIRPLANE, PROCEED AS FOLLOWS:

1. Parachute riser-shoulder harness release fittings - SQUEEZE
   After landing impact, squeeze the parachute riser-shoulder harness release fittings on the integrated harness to free the parachute.
2. Lap belt release fittings - RELEASE
   After landing impact, squeeze the lap belt release fittings on the integrated harness to free the survival kit.
3. Helmet face visor - OPEN
   Open helmet face visor or loosen oxygen mask.

WARNING

If the composite disconnect is manually pulled before the face visor is open, or the oxygen mask is loosened, the crewmember may suffer from lack of oxygen since oxygen cannot be supplied to the user once the composite disconnect has been pulled.

DITCHING CHART

CREW MEMBER	DUTIES BEFORE IMPACT	POSITION	DUTIES AFTER IMPACT	EQUIPMENT	EXIT
PILOT	1. Radar Observer–ALERT. 2. Make radio distress call. 3. IFF–EMERGENCY. 4. External stores–JETTISON. 5. Landing gear–UP. 6. Wing flaps–DOWN. 7. Arresting gear–DOWN. 8. Leg restraint release handle–PULL AFT. Release leg restraint lines before ditching to expedite egress from the cockpit. 9. Visor–DOWN. 10. Oxygen mask or face visor–TIGHTEN or SEAL. 11. Canopy – JETTISON (forward canopy first). 12. Lower seat, assume position for ditching. 13. Inertia reel handle–LOCKED. 14. Fly parallel to swell pattern. 15. Attempt touch down along wave crest. 16. When hook contacts water –SHUT DOWN ENGINES.	1. In Seat. 2. Feet on rudder pedals knees flexed.	1. Pull up on the Emergency harness release handle. 2. Squeeze parachute riser-shoulder harness release fittings. 3. Stand up to release survival kit sticker clips from the seat. NOTE • The Bail-Out Bottle will be actuated when the crewmember kicks free of the seat • In the event of ditching and sinking in water when immediate escape is impossible, it is possible to survive under water with oxygen equipment or full pressure suit until escape can be made. 4. Abandon aircraft. 5. Inflate life vest. 6. See that RO is clear. 7. Inflate life raft and secure Emergency equipment. 8. Proceed away from aircraft and tie rafts together.	1. One man raft and Emergency equipment. 2. Life vest 3. Flash light.	Over canopy sill.
RADAR OBSERVER	1. Acknowledge pilots ditching order. 2. Radar equipment–STOW. 3. Leg restraint release handle–PULL AFT. Release leg restraint lines before ditching to expedite egress from the cockpit. 4. Visor–DOWN. 5. Oxygen mask or face visor –TIGHTEN or SEAL. 6. Canopy – JETTISON (Aft canopy last). 7. Lower seat, assume position for ditching. 8. Inertia reel handle–LOCKED	1. In Seat. 2. Feet on foot ramp knees flexed.	Same as for Pilot Except: 6. See that Pilot is clear.	1. One man raft and Emergency equipment. 2. Life vest 3. Flash light.	Over canopy sill.

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Figure 3-8

4. Composite disconnect knob - PULL
   Pull up on the composite disconnect knob and kick free of the seat.

Note

In the event of ditching and sinking in water when immediate escape is impossible, it is possible for the crewmember to survive under water with oxygen equipment until escape can be made. The oxygen regulator is a suitable underwater breathing device since the regulator is always on 100% oxygen. If a pressure suit is not being worn it is essential that the mask be tightly strapped in place.

EXTERNAL STORES EMERGENCY RELEASE

The external stores emergency release button, when depressed, will jettison all external stores carried on the airplane except missiles. To jettison stores proceed as follows:

1. Landing gear handle - UP
   The landing gear handle must be in the UP position to energize the emergency jettison circuits.
2. External stores emergency release button - PUSH

WARNING

Fuselage mounted Sparrow III Missiles and wing missiles can be jettisoned from the missile status panel only.

EXTERNAL TANK EMERGENCY JETTISON

If external tanks must be jettisoned, proceed with one of the following procedures:

EXTERNAL WING TANKS ONLY

1. External tanks jettison switch - JETT
   Raise the guard and place the external tanks jettison switch on the fuel control panel to JETT.

CENTERLINE EXTERNAL TANK ONLY

1. Landing gear handle - UP
   The landing gear handle must be in the UP position to energize the bomb button circuit.
2. Bomb control switch - DIRECT
   Place the bomb control switch on the armament control panel to DIRECT.
3. Bomb release button - DEPRESS
   DEPRESS bomb release button on the control stick grip.

Note

For emergency landings retain external tanks if empty.

AFTERBURNER FAILURE

AFTERBURNER FAILURE DURING TAKE-OFF

If the afterburner(s) fails during take-off, the resulting loss of thrust is significant. Take-off need not be aborted if remaining runway is compatible with power available. After failure, the variable area exhaust nozzle will continue to function as directed by exhaust gas temperature. In this circumstance, the nozzle moves as a function of temperature limiting only. To insure closure of the nozzle to the military area so that military thrust will be immediately available:

1. Throttle of failed afterburner - MIL RANGE
2. If the exhaust nozzle is operating properly a relight may be initiated.

EXHAUST NOZZLE FAILURE IN AFTERBURNER

Upon initiating afterburner, failure of the variable exhaust nozzle to open is recognized by a rapid increase in exhaust temperature and a drop in rpm. If an overtemperature condition exists:

1. Immediately move throttle to MIL range.

WARNING

DO NOT attempt to relight afterburner. Damage to engine and airframe structure could result.

AFTERBURNER "BLOWOUT" DURING FLIGHT

In the event of afterburner "blowout" or loss of afterburning, the failed engine afterburner throttle must be moved inboard immediately to insure exhaust nozzle closure and Military power (thrust) available. If no obvious cause (overheat) is discernible, a relight may be attempted. If cockpit indications of resumed afterburner are normal, continue afterburner operation.

OIL SYSTEM FAILURE

An oil system failure of either engine is recognized by a drop in oil pressure or a complete loss of pressure.

1. If a minimum oil pressure of 55 psi cannot be maintained, throttle - IDLE
2. If a minimum of 20 psi cannot be maintained, engine - SHUTDOWN
3. In either of the above instances - LAND AS SOON AS POSSIBLE

Since the constant speed drive unit which drives the generator is supplied with oil under pressure by the engine oil system, a Gen Out Light, followed by sluggish exhaust nozzle action, are early indications of impending engine oil starvation. The engine oil pressure gage should be monitored closely subsequent to a generator failure. In general, it is advisable to shut the engine down as early as possible after a loss of oil supply is indicated, to minimize the possibility of damage to the engine and the constant speed drive unit. The engine will operate satisfactorily at military power for a period of one minute, with an interrupted oil supply. However, continuous operation, at any engine speed, with the oil supply interrupted will result in bearing failure and eventual engine seizure. The rate at which a bearing will fail, measured from the moment the oil supply is interrupted, cannot be accurately predicted. Such rate depends upon the condition of the bearing before oil starvation, temperature of the bearing and loads on the bearing. Malfunctions of the oil system are indicated by a shift (high or low) from normal operating pressure, sometimes followed by a rapid increase in vibration. A slow pressure increase may be caused by partial clogging of one or more oil jets; while a rapid increase may be caused by complete blockage of an oil line. Conversely, a slow pressure decrease may be caused by an oil leak; while a sudden decrease is probably caused by a ruptured oil line, or a sheared oil or scavenge pump shaft. Vibration may increase progressively until it is moderate to severe before the pilot notices it. At this time complete bearing failure and engine seizure is imminent. Limited experience has shown that the engine may operate for 4-5 minutes at 80 to 90 percent speed before a complete failure occurs. In view of the above, the following operating procedures are recommended:

1. Throttle - Reduce Thrust
   Shut the engine down when partial power is not required from the affected engine. Where mission or flight requirements demand partial power from the affected engine, set engine speed at 86-89%.
2. Avoid abrupt maneuvers causing high G forces.
3. Avoid unnecessary or large throttle bursts.

Note

To keep bearing temperatures and loads at a minimum, do not use high thrust settings.

Oil Pressure Change Accompanied by Vibrations:

1. Throttle - OFF
   The throttle should be moved to OFF, when partial power from the affected engine is not essential to mission or flight requirements.

WARNING

Increasing vibration is an indication of bearing failure. Severe vibration indicates that engine seizure will occur within a few seconds. Chop the throttle to OFF to prevent major engine, and possible aircraft, damage.

ENGINE FUEL SYSTEM FAILURE

ENGINE DRIVEN FUEL PUMP FAILURE

Since the engine driven fuel pump consists of two gear-type, equal capacity pumping elements, failure of either element will not cause a failure or a reduction in pumping capacity of the remaining element. The engine will continue to function normally except in low altitude high speed operation, where a loss in engine rpm may occur. There will be no indication to the pilot of a failure of a single element in the fuel pump.

AIRPLANE FUEL SYSTEM FAILURES

FUEL BOOST PUMP FAILURE

If fuel booster pumps fail, fuel will still be supplied to the engines by gravity feed. If booster pumps fail at altitudes above 20,000 feet, flameout of both engines may occur. During gravity feed, high fuel flow rates required by afterburner operation cannot be met. Boost pump pressure indication at 0 psi indicates that both boost pumps are inoperative. Proceed as follows:

If both engines have flamed out:

1. Reduce speed to 515 knots IAS or Mach 1.1 whichever is lower.
2. Wind driven turbine - EXTEND
   Extending the wind driven turbine will operate the left fuel boost pump at low speed. This will supply enough fuel to either engine to accomplish an airstart.
3. Attempt an airstart.

If an airstart has been accomplished or the engines have not flamed out:

1. Reduce power to minimum required.
2. Descend to below 20,000 feet using JP-5 or 10,000 feet using JP-4.
3. Using minimum power required, land as soon as possible.

Since the boost pumps feed into a common manifold before branching off to the engines and boost pump pressure transmitters, an inoperative pump will be noted on both boost pump indicators. Therefore, a boost pump pressure reading of from 0 to 5 psi will be a good indication that one of the boost pumps is inoperative. The power settings on each engine should be reduced as necessary until a boost pump pressure reading of 5 psi or greater is obtained.

INTERNAL TANKS TRANSFER SYSTEM FAILURE

Transfer system failure in this airplane can usually be attributed to failure of the fuel system to become pressurized or failure of the fuel indicator gage which automatically schedules wing fuel transferring. Failure of the fuselage cells transfer pumps is not likely, since two hydraulic and two electric transfer pumps operate simultaneously and either set of pumps (electric or hydraulic) is capable of transferring all fuselage fuel.

If fuel system fails to become pressurized:

1. Wing transfer pressure switch - EMERG

Note

Selecting the EMERG position performs the same functions as did the landing gear handle switch, all pressure regulators open and all pressure relief valves closed.

CAUTION

To prevent drop tank collapse, during high altitude descent with wheels down, place wing trans press switch to EMERG. Place wing trans press switch to NORMAL prior to landing.

In airplanes 145311b, 145312b and 145314b thru 145317b, it is possible for the fuel indicator gage switch to malfunction and prevent wing fuel from transferring. If wing fuel fails to transfer:

1. Internal wing transfer switch - MAN OVERRIDE

Note

Selecting the MAN OVERRIDE position overrides a failure in the fuselage fuel indicator gage switch and allows wing fuel transfer such as would normally occur at the 5700 pound level.

EXTERNAL TANKS TRANSFER SYSTEM FAILURE

Failure of external tank(s) to transfer may be caused by the collapsing of the external tank(s) transfer hose due to pressure differential. This pressure differential may occur due to the external tanks transfer hose passing through the pressurized internal wing tanks. If external tanks fail to transfer, proceed as follows:

1. Ext. transfer switch - OFF
2. Burn off internal wing fuel.
3. Int. wing dump switch - DUMP
   Select DUMP position in order to relieve internal wing tank pressure.
4. Ext. transfer switch - OUTBD

ELECTRICAL SYSTEM EMERGENCY OPERATION

SINGLE GENERATOR FAILURE

Failure of one generator will be noted by illumination of either the L.H. or R.H. Generator Out light. The light will determine which generator has failed. One generator in normal operation is sufficient to support the entire electrical demand or load. In the event of generator failure, the pilot should take the following action:

CAUTION

Upon fluctuation of the warning light, immediately check the corresponding oil gage. The generator failure could have been caused by oil starvation which will also affect the engine oil system.

1. Generator switch - CYCLE
   Cycle generator switch from ON to OFF and back to ON.
2. Check generator warning light - OUT
   If generator fault has been corrected, the generator will be reconnected to the system and the warning light will go out.

If generator warning light remains illuminated:

3. Generator switch - OFF
4. Land as soon as practicable.

DOUBLE GENERATOR FAILURE

Although a double generator failure is highly remote, the possibility of a double failure is still present. In most cases the generators will not fail at the same time, it is more likely that one generator will fail, followed by the failure of the other generator. As previously stated in the paragraph on single generator failures, one generator out light will be illuminated when one of the generators fail. When the other generator fails, both generator out lights will be illuminated. Although a double generator failure results in a complete electrical power failure due to the generator low speed switches dropping the generators off the line at about 53% rpm, the permanent magnet generators (provided for field excitation and generator warning lights) will continue to function while the generators are turning. Upon the loss of both generators, immediately perform the following:

1. Reduce speed to 515 knots IAS or Mach 1.1 whichever is lower.
2. Wind driven turbine - EXTEND
   Extend wind driven turbine. The Gen Out warning lights should illuminate indicating that the emergency generator is operating.

CAUTION

Prior to reaching 15 minutes of continuous operation, the emergency generator must be dropped off the line to allow for cooling. Operation of more than 15 minutes creates excessive temperatures within the generator which will drop the generator output 50%.

Note

- Extending the wind driven turbine will connect the emergency generator to the essential buses. Refer to figure 1-10, Electrical System Schematic, for equipment that is operated from the essential buses.
- Emergency wind driven generator operational time shall be logged on the yellow sheet (OPNAV FORM 3760-2).

3. All unessential electrical switches - OFF
   Turn off all electrical equipment not necessary to maintain flight.
4. Generator control switches - CYCLE
   Attempt to return generators to the line by cycling generator control switches.
5. Check generator warning lights - OUT
   If the fault has been corrected the generator warning lights will be extinguished.

If generator warning light(s) remain illuminated:

6. Generator control switches - OFF
7. Land as soon as possible.

With the loss of all electrical power in flight, the emergency pneumatic system should be utilized to extend the landing gear. If the gear has been lowered prior to loss of electrical power, the pneumatic system should still be utilized. This procedure assures that any pressure surge in the landing gear up line occurring with electrical power removal has not unlocked the gear. If total electrical power is lost while taxiing, the emergency pneumatic landing gear system should be actuated to assure that the gear remains locked.

Generator Failure (Bus Tie Open)

Due to the design of the airplanes electrical system, it is possible to lose the essential buses as a result of a short in the right generator system. A short on the right generator bus will be noted by the illumination of the BUS TIE OPEN warning light followed in 5 seconds by the illumination of the R.H. GEN OUT light, the BUS TIE OPEN warning light will be extinguished. If the BUS TIE warning light remains extinguished, the short will present no problem since the left generator will supply the power to the right buses. However, if the BUS TIE OPEN warning light illuminates again within 2 seconds of being extinguished, all the buses supplied by the right generator will be lost since the illumination of the BUS TIE warning light indicates that the bus tie relay has opened. Because the bus tie relay parallels the output of the two generators it can be seen that an open relay and an inoperative right generator will automatically deprive the airplane the use of the right main and essential airplanes buses. The essential buses may be regained but this necessitates the loss of the left generator system, since THE EMERGENCY GENERATOR WILL NOT COME ON THE LINES AS LONG AS EITHER MAIN GENERATOR IS IN SERVICE. Extension of the wind driven turbine must be followed by SWITCHING OFF THE LEFT GENERATOR in this circumstance. After the extension of the wind driven turbine and utilization of the emergency generator an attempt may be made to regain the right main generator. This is accomplished by recycling the right generator switch to ON. If the right generator comes on the line, the short is no longer present and the left generator may be placed in operation. The generators will parallel and normal operation will be resumed. If the right generator warning light does not go out when the right generator switch is placed on, the short remains and the emergency generator must still be used. A short in the left generator or in any of the left generator system buses is not of such a serious nature due to the fact that the right generator right main and essential will still be in operation. Upon illumination of the BUS TIE OPEN warning light, proceed as follows:

Both Generators Operating

1. Continue mission.

Left Generator Out

1. Left generator control switch - CYCLE (OFF wait 2 sec. ON)
   If the left generator warning light remains on with the bus tie open all equipment operated by the left main buses will be lost.
2. Turn off all electrical equipment not essential to flight.
3. Land as soon as possible.

Right Generator Out

1. Right generator control switch - CYCLE (OFF wait 2 sec. ON)
   If the right generator warning light remains on with the bus tie open all equipment operated by the right main and essential buses will be lost.
2. Turn off all electrical equipment not essential to flight.

To regain use of the essential bus:

3. Reduce speed to 515 knots IAS.
4. Wind driven turbine - EXTEND
5. Left generator control switch - OFF
   The left generator must be turned OFF in order to connect the emergency generator to the essential buses.
6. Land as soon as possible.

CAUTION

Prior to reaching 15 minutes of continuous operation, the emergency generator must be dropped off the line to allow for cooling. Operation of more than 15 minutes creates excessive temperatures within the generator which will drop the generator output 50%.

Note

In the event of engine failure or malfunctions, the generators will continue to operate at normal output until the engine rpm's drop to approximately 53%. At this time, underspeed switches in the generator circuits will drop the generators off the line to prevent any damage to equipment that may result from low frequency. The airplane will be without any electrical power and the emergency generator should be utilized.

HYDRAULIC SYSTEM EMERGENCY OPERATION

The loss of hydraulic pump in power control systems No. 1, No. 2 will be noted by the illumination of the Check Hyd. Gages Warning light. This single light serves both systems, and the pilot should check the hydraulic gages to assure which system has malfunctioned. In airplanes 146817c thru 148275e, the Check Hyd. Gages Warning light also monitors the Utility Hydraulic System.

SINGLE POWER CONTROL SYSTEM FAILURES

A hydraulic pump failure in power control system No. 1 presents no serious problem because of the following reasons:

1. An emergency hydraulic pump can feed the system if the need arises.
2. Power control system No. 2 is capable of assuming the full demand of the flight control dual cylinders.

If a failure should occur in power control system No. 2, power control system No. 1 will assume the full demand of the flight control dual cylinders. Systems No. 1 and No. 2 are independent of each other, but each system satisfactorily functions as an emergency system for the other.

Power Control System No. 1 and No. 2 Emergency Operation

If power control systems No. 1 or No. 2 should fail, proceed as follows:

1. Reduce airspeed to 515 knots CAS or Mach 1.1 whichever is lower.
2. Wind driven turbine - EXTEND
3. Land as soon as possible.

COMPLETE POWER CONTROL SYSTEMS FAILURE

The pilot should upon initial detection of hydraulic power loss note a trend of failure as to whether the gages show a definite steady drop, or gage fluctuations. With a steady drop indication, hydraulic power will probably not recover. In the event of complete power control hydraulic failure, the aircraft will become uncontrollable. Before this occurs, proceed as follows:

1. Wind driven turbine - EXTEND
2. Reduce airspeed to 515 knots CAS or Mach 1.1 whichever is lower.

If power control hydraulic pressure recovers:

3. Land as soon as possible.

If power control hydraulic pressure does not recover:

4. EJECT

UTILITY HYDRAULIC SYSTEM FAILURE

Failure of the utility system will prevent the hydraulic operation of the following essential items:

a. Rudder
b. Wing flaps
c. Landing gear
d. Speed brakes
e. Variable engine air duct ramps
f. Hydraulic fuel transfer pumps
g. Wheel brakes
h. Arresting gear (retraction)

Of the above items, emergency pneumatic operation is provided for the following:

a. Landing gear
b. Wheel brakes
c. Wing flaps

Speed brake retraction is available by placing the emergency speed brake switch in the RETRACT position.

The rudder can still be operated mechanically without hydraulic boost, however, pedal forces will be much higher than normal.

FLIGHT CONTROL SYSTEM EMERGENCY OPERATION

STABILATOR FEEL TRIM FAILURE

Partial Bellows Failure

Partial bellows failure is recognized by a mild nose down stick force proportional to the airspeed unless the failure occurs during maneuvering flight at which time it may not be noticeable. Reduction of stick centering and pitch stability will result. Should this failure occur:

1. Reduce airspeed - 250-300 knots CAS
2. Retrim airplane
3. Avoid abrupt fore and aft stick movements.
4. Land as soon as practicable.

Complete Bellows Failure

A complete bellows failure is recognized by a heavy nose down feel force at the control stick. The maximum amount that this stick force can attain is 30 pounds dependent on the trim position. This force can be reduced to 5 pounds by applying full NOSE UP trim. Should a complete bellows failure occur:

1. Reduce airspeed - 250-300 knots CAS.
2. Stab. Trim - FULL NOSE UP
3. Avoid abrupt fore and aft stick movements.
4. Land as soon as practicable.

Ice/Water Blockage of Ram Air Line

Ice or water blockage of the artificial feel bellows ram air line will result in conditions similar to a complete bellows failure. If ice or water blockage is suspected, longitudinal trim should not be applied to relieve control stick force. The intermittent nature of this condition and the suddenness of return to normal can cause violent pitch transients. When the ram air line is blocked, no stick force gradient will be felt by the pilot should a change in stock position be required. In the event of suspected ice or water blockage of the ram air line:

1. Reduce airspeed - 250-300 knots CAS
2. Maintain attitude by pilot effort.
3. If practical descend to air that is above freezing.

If this condition persists:

4. Land as soon as practicable.

Runaway Stabilator Trim

If stabilator trim appears to be running away it is possible under certain conditions to lessen the situation. Runaway stabilator trim can be alleviated by engaging the autopilot, providing:

a. The stab trim circuit breaker has been pulled IMMEDIATELY upon detection of runaway trim.
b. Runaway trim is in the nose up direction.
c. Nose down runaway trim has not exceeded 2-1/2 units.
d. Airspeed can be reduced to 300 knots CAS or less.

If the above conditions are met:

1. Reduce airspeed to 300 knots CAS or less.
2. Autopilot - ENGAGE

CAUTION

When the autopilot is used to alleviate a runaway trim condition, and excessive out of trim forces are present (full nose down runaway trim) the autopilot may alternately disengage and re-engage. If this occurs discontinue use of the autopilot.

Prior to landing:

3. Autopilot - DISENGAGE
   When in the landing configuration (gear and flaps extended) and at 180 to 190 knots CAS, grasp stick firmly and disengage the autopilot. Depending upon severity of the malfunction, the airplane may or may not be in trim; if out of trim the forces should not be too high and the airplane can be landed with the out of trim condition, or the autopilot can be re-engaged, and the landing made with control stick steering.

If the landing is made with autopilot engaged:

4. Immediately after touchdown - DISENGAGE AUTOPILOT
   Immediately after touchdown disengage autopilot to prevent damage to autopilot components.

AILERON RUDDER INTERCONNECT (ARI) SYSTEM DISENGAGEMENT

The ARI system can be temporarily disengaged by depressing the AFCS/ARI emergency disengage switch; this will disengage the ARI only as long as it is held depressed. To permanently disengage the ARI system, the circuit breaker on the left utility panel must be pulled and the Damper switch must be disengaged. Pulling the circuit breaker only and keeping the dampers engaged will still provide 5° of ARI rudder authority. In the event of an ARI system malfunction proceed as follows:

To temporarily disengage system:

1. AFCS/ARI emergency disengage switch - DEPRESS
   Depressing the emergency disengage switch will disengage the ARI system only as long as the switch is held depressed.

To permanently disengage system:

1. ARI circuit breaker - PULL
2. Damper switch - DISENGAGE

SPEED BRAKE SYSTEM EMERGENCY OPERATION

Should the selector valve or the utility hydraulic system fail, and the speed brakes are extended, they may be retracted by placing the emergency speed brake switch on the left console to the RETRACT position. This allows air loads to close the panels to a trailing position.

Note

When speed brakes are closed by use of the emergency switch, the air loads will not completely close the panels.

LANDING GEAR SYSTEM EMERGENCY OPERATION

There are no provisions for emergency retraction of the landing gear in flight.

LANDING GEAR EMERGENCY LOWERING

If normal gear operation fails, the gear can be lowered by utilizing the following procedures:

1. Airspeed - BELOW 250 KTS. CAS
2. Landing gear circuit breaker - PULL
3. Landing gear handle - DOWN

4. Landing gear handle - PULL AFT
   Pull handle full aft, full limit of travel, and hold in full aft position until gear indicates down and locked.

CAUTION

Hold handle in full aft position until gear indicates down and locked, and then leave the landing gear handle in the full aft position. Returning the handle to its normal position allows the compressed air from the gear down side of the actuating cylinder to be vented overboard, and may cause the flaps to extend. The above action is brought about because the vent cannot dump the compressed air pressure at the same rate at which it is being discharged from the down side of the gear actuating cylinder, and since the flaps and gear utilize a common vent, compressed air is forced into the flap vent line, forcing hydraulic fluid back through the flap selector.

5. Landing gear position indicators - CHECK

Note

Any pneumatic extension of the landing gear shall be logged on the yellow sheet (OPNAV FORM 3760-2).

CAUTION

To prevent drop tank collapse, during high altitude descent with wheels down, place wing trans press switch to EMERG. Place wing trans press switch to NORMAL prior to landing.

WING FLAPS SYSTEM EMERGENCY OPERATION

There are no provisions for emergency retraction of the flaps in flight.

WING FLAPS EMERGENCY LOWERING

If normal wing flap operation fails, the flaps can be lowered by utilizing the following procedures:

1. Airspeed - BELOW 200 KTS. CAS
2. Flap circuit breaker - PULL
3. Emergency wing flap extension handle - PULL AFT
   Pull emergency wing flap extension full AFT and down, full limit of travel.

Note

Trailing edge flaps will not indicate down with circuit breaker pulled.

4. Wing flap position indicators - CHECK

CAUTION

- Leave the emergency wing flap extension in the full aft position. Returning the handle to its normal position allows the compressed air from the flap down side of the actuating cylinder to be vented overboard, and may cause the landing gear to unlock and extend. The above action is brought about because the vent cannot dump the compressed air pressure at the same rate at which it is being discharged from the down side of the wing flap actuating cylinder, and since the flaps and gear utilize a common vent, compressed air is forced into the landing gear vent line, forcing hydraulic fluid back through the landing gear selector.

Note

- The wing flaps when extended by the emergency method are held down by pneumatic pressure only. If pneumatic pressure is lost, the wing flaps will be blown up by the airstream.
- Any pneumatic extension of the wing flaps shall be logged on the yellow sheet (OPNAV FORM 3760-2).

WHEEL BRAKE SYSTEM EMERGENCY OPERATION

In the event of a utility hydraulic system failure or loss of brake action, the airplane can be stopped by using the emergency brake system.

1. Allow aircraft to decelerate.
   Delay using brakes as long as safety will permit, allowing the airplane to decelerate as much as possible.
2. Hydraulic wheel brakes - APPLY
   Depress brakes and keep a constant increasing brake pressure. Do not pump brakes. There are approximately 6 to 8 brake applications available from the emergency hydraulic accumulators.

Note

With no utility hydraulic system pressure available the manual hydraulic brakes are still capable of furnishing flow and pressure to accomplish differential braking. This can be utilized with the emergency brake system to maintain directional control of the airplane. The number of such applications is unlimited, however, higher pedal travel and higher brake pedal forces will be necessary.

If unable to stop airplane using emergency hydraulic accumulator braking and manual brakes, perform the following:

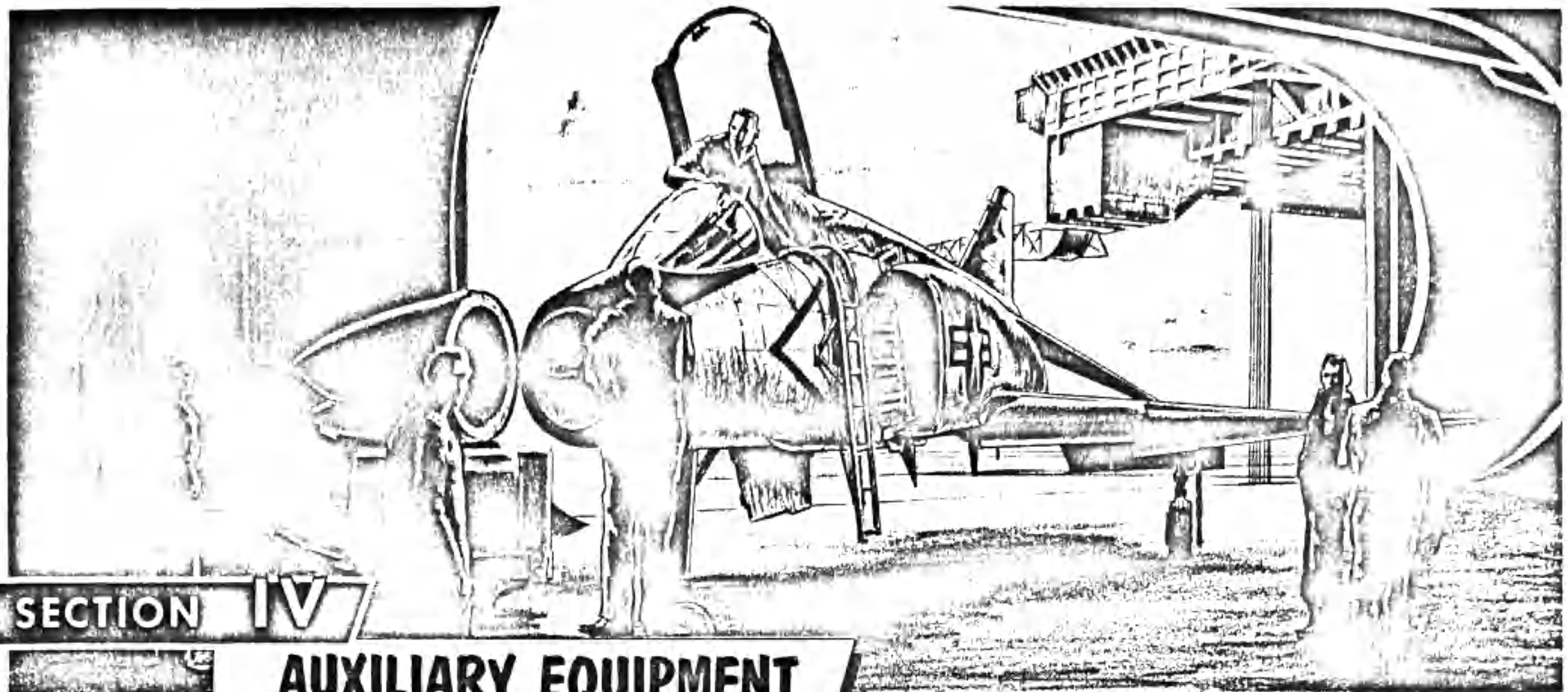
3. Emergency brake handle - PULL
   The emergency brake system meters air pressure in proportion to applied pilot effort, but does not provide differential braking.

CAUTION

Asymmetrical braking is prevalent during use of air emergency braking. This asymmetrical braking could be due to runway crown or crosswinds as well as unequal brake torque.

Note

- There is enough air stored in the emergency brake system to provide approximately 20 maximum deceleration applications of the emergency brakes.
- There will be a time lag between pulling the emergency brake handle and the application of pneumatic pressure to the wheel cylinders.



SECTION IV
AUXILIARY EQUIPMENT

FM-FDA-400

AIR CONDITIONING AND PRESSURIZATION SYSTEM

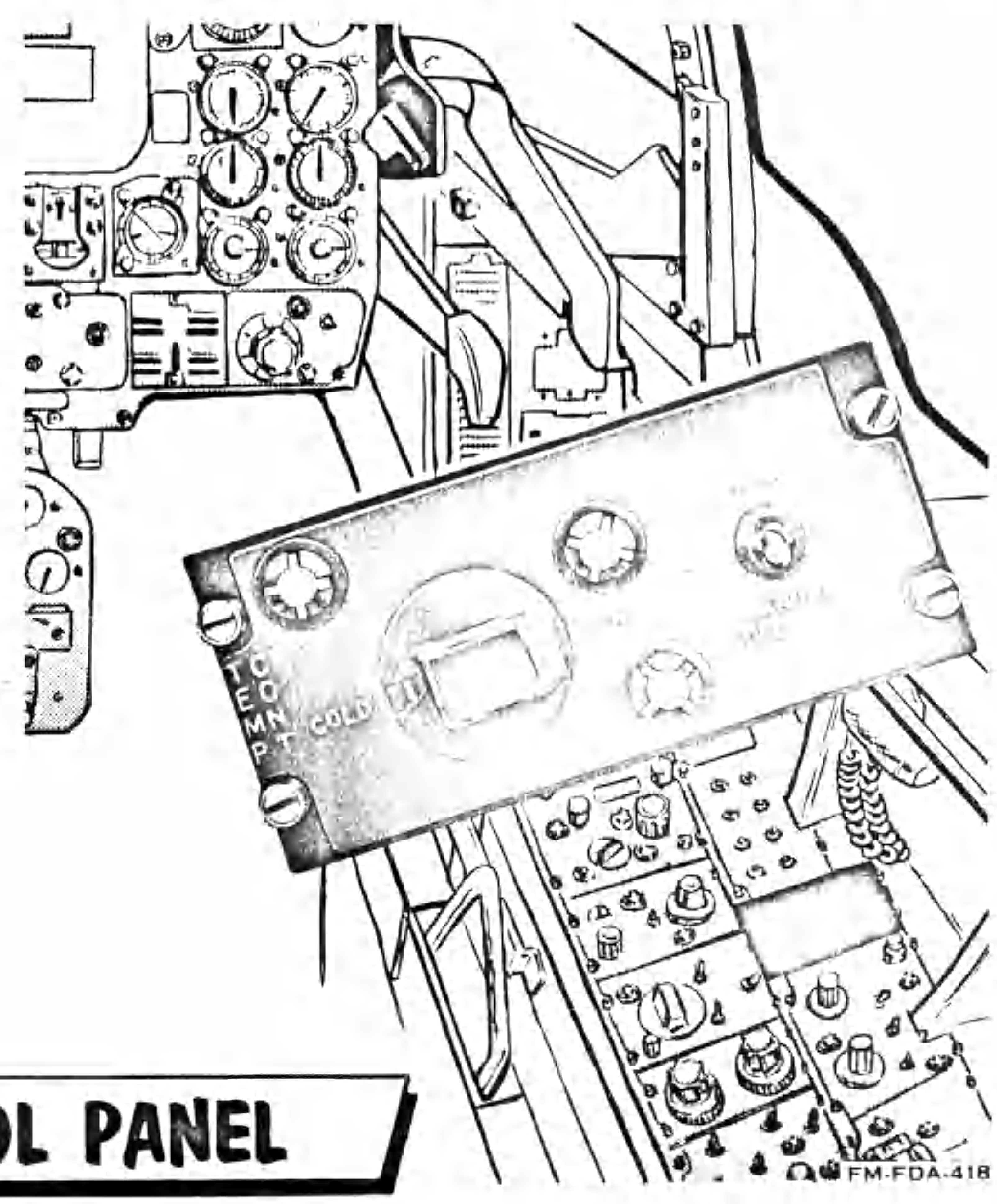
Air conditioning in the airplane is divided into two major systems, one for the cockpit areas and one for electronic equipment cooling. Both cockpits and the pressure suits for both crewmen are pressurized, and supplied with conditioned air from the cabin refrigeration unit. The same air that pressurizes and heats the cockpits also is used to keep the windshield free of fog, frost and rain. The equipment refrigeration unit provides cooling air for the main radar package and communication navigation identification equipment. Both systems utilize high temperature high pressure 17th stage engine compressor bleed air available from either or both engines.

CABIN/ PRESSURE SUIT AIR CONDITIONING

The cabin/pressure suit air conditioning system (figure 4-2) consists of two air-to-air heat exchanger, an expansion turbine, shutoff and pressure regulating mixing valves and temperature controls necessary to allow the pilot to select cabin conditioning temperatures, pressure suit temperatures, defogging, rain removal and ram air operations. Separate temperature ranges and control systems for the pressure suit and cabin are provided. High temperature high pressure, engine compressor bleed air passes through the primary and secondary heat exchanger and is expanded through the cooling turbine. After being mixed with hot compressor bleed air (as required by pilots temperature selection) it enters the cockpits through several manifolds, one near the RO's feet, one near the pilot's feet, one along the lower surface of each windshield side panel and one at the base of the flat optical panel of the windshield. Effective airplanes 143252d thru 148275e, two additional "eyeball" type air nozzles are located just below the canopy sill on the right and left side of the RO's cockpit.

CABIN/PRESSURE SUIT AIR CONDITIONING OPERATION (WITHOUT PRESSURE SUIT)

The cockpit air conditioning system operation can best be explained and understood by referring to the cabin/



TEMPERATURE CONTROL PANEL

FM-FDA-418

Figure 4-1

CABIN AIR-CONDITIONING AND PRESSURIZING SYSTEM

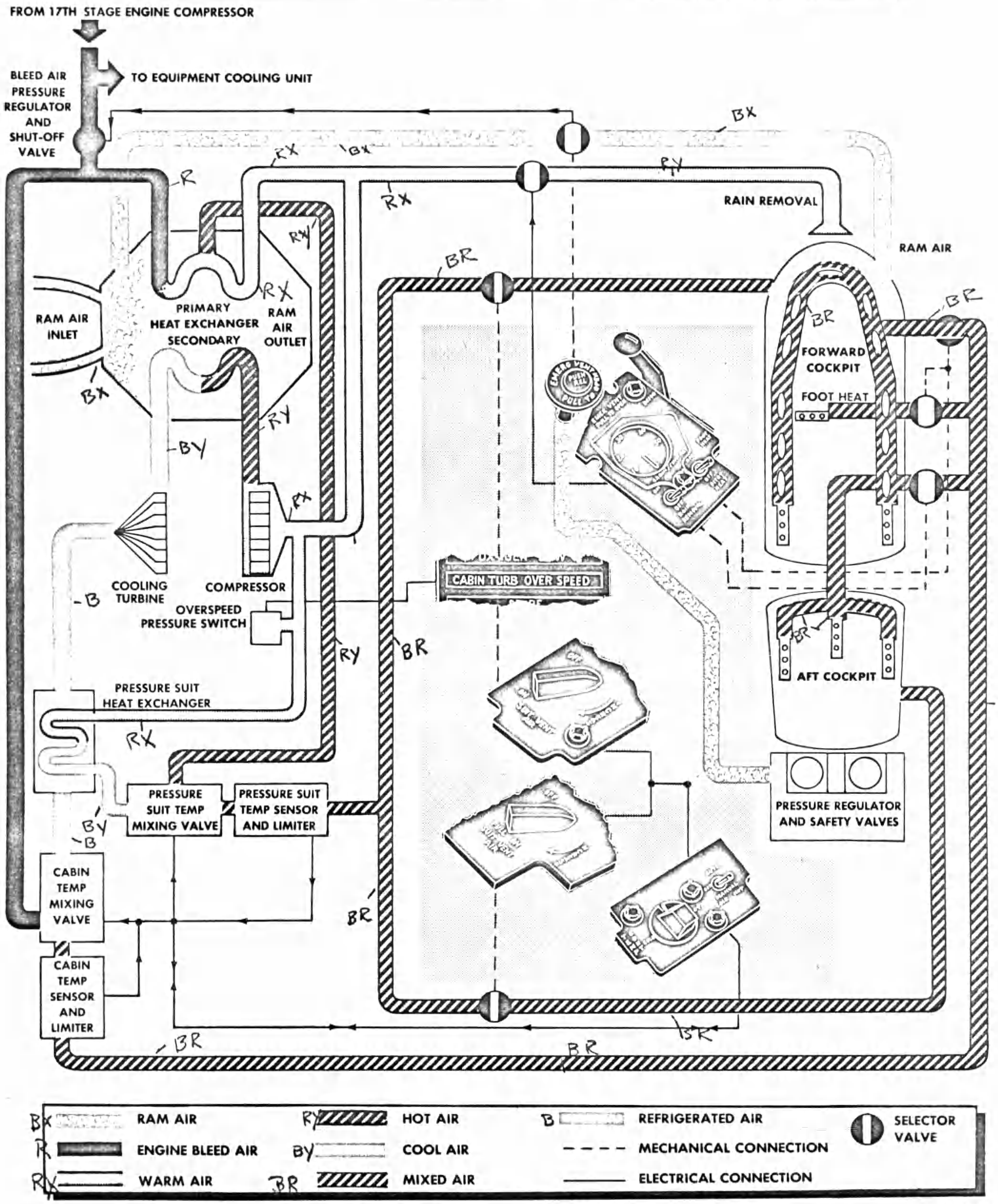


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Figure 4-2

CABIN PRESSURE SCHEDULE

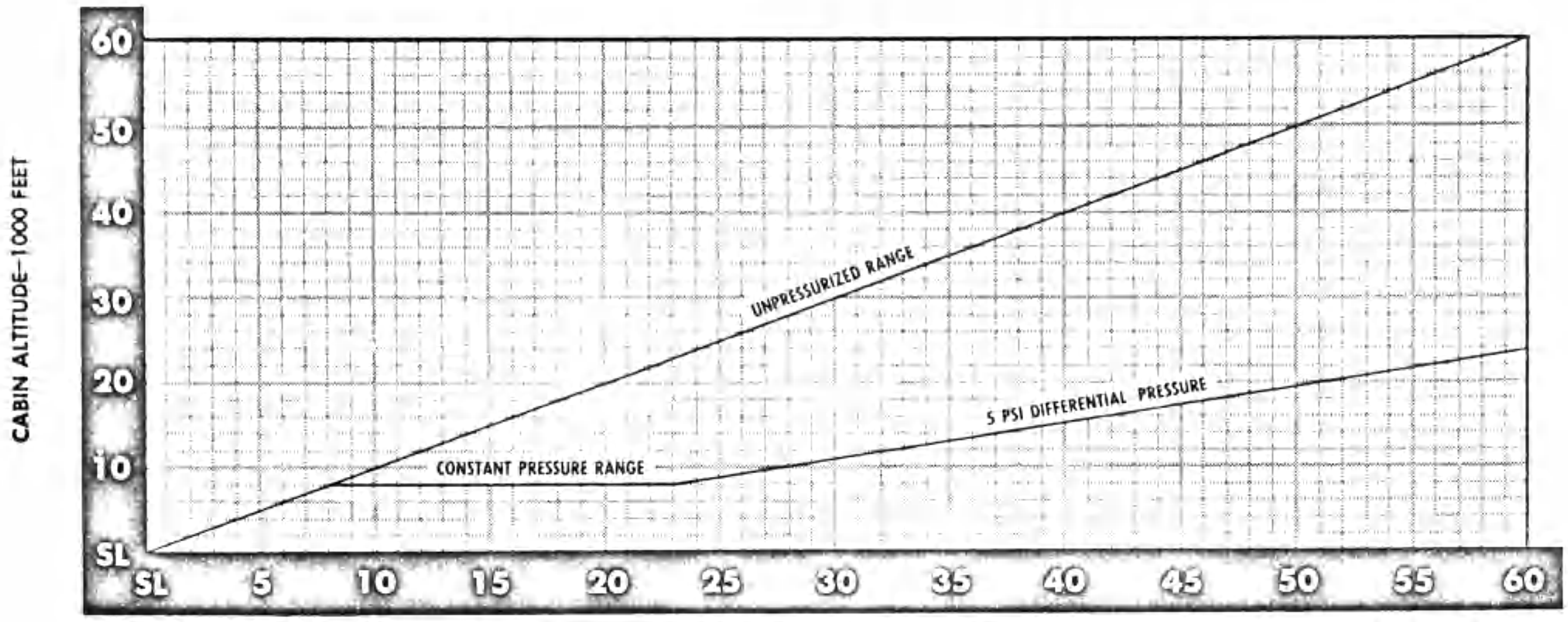

Figure 4-3

pressure suit temperature schedule (figure 4-5). The "normal" range, which refers to the curve labeled foot-heat produces temperatures from -20° F to 100° F. These temperatures refer to inlet air and not cabin temperature - so cabin temperature will be determined by a combination of inlet air and environmental conditions. The "normal" curve is the governing schedule for all air entering the cockpit while in automatic temperature control with the defog-foot heat lever aft of the halfway or 50% position. A little air is always entering through the defog port and this air increases (while foot-heat air decreases) as the lever is moved forward. But until a switch is made at about half travel position, both defog and foot-heat air enters on the "normal" schedule. Thus full range on the auto-rheostat (from 7 o'clock to 5 o'clock positions) will only produce -20° F to 100° F air - unless the defog lever is moved more than halfway forward. When the 50% switch is made, the temperature schedule of all entering air switches to the "defog" curve. Thus, if 250° were the knob position (about 3:00 o'clock) 87°F would be the temperature of incoming air in the "normal" range, but when the 50% switch was made, the temperature would change to 137°F. As the defog lever is moved forward through full travel, the footheat butterfly valves - for both front and rear cockpits - are closing down as the defog valve opens. Thus the defog air volume increases on a rather steep slope and when the lever is close to full defog position (full forward) the temperature of the air entering the cockpit is quite warm. If through some malfunction of the automatic temperature control system, it becomes necessary to use manual override, you should realize that "bumping" the switch to cold or hot can give you a full range of temperatures up to 230° F, but the 50% switch on the defog lever is bypassed. Thus the entire temperature range for both foot-heat and defog air is scheduled directly by the mixing valve position which in turn is moved only when the override switch is held to either hot or cold. The switch is spring loaded to off and in the off position the mixing valve is held stationary.

Note

The MAN position of the temperature control switch should not be used except as a back-up in the event of a failure in the automatic system. No advantage will be gained in the amount of heating time required by utilizing the MAN position over the AUTO position. Prolonged use of the full hot MAN position could result in excessive cockpit heat, and/or a circuit breaker popping phenomenon during the take-off phase, since airflow and temperature with the switch in the MAN position is directly proportional to engine rpm.

CABIN/PRESSURE SUIT AIR CONDITIONING OPERATION (WITH PRESSURE SUIT)

When the pressure suit mixing valve becomes active (vent air turned on), both pressure suit and cabin air schedules are followed while in automatic temperature control; that is, with the defog lever aft of half travel, vent air will be provided to the pressure suit at its schedule while simultaneously foot-heat and defog air will enter on the "normal" schedule. Moving the defog lever forward of halfway again switches foot-heat and defog air to the higher schedule, but the pressure suit schedule remains unchanged. Manual override operation complicates the picture a bit when the pressure suit is involved in that only the pressure suit mixing valve is actuated when the override switch is moved to hot or cold. The cabin air mixing valve remains in its last automatic selected position. The relative volumes of defog and foot-heat air can, of course, be changed by defog lever action but the temperature is fixed when manual override is selected. This characteristic sets the stage for an unhealthy situation: Suppose automatic temperature control becomes inoperative during the cruise portion of a flight. Cabin air temperature will not normally be at a high setting with the pressure suit on, so when manual override is selected, the cabin air mixing valve remains at a fixed, moderate temperature position. Now when higher temperature defog air is desired for letdown, it is not available since manual override controls only suit vent air. However, when the suit vent air lever is turned off, the suit mixing valve becomes stationary at the cold position and the cabin mixing valve again is operative. Since suit vent air would not be absolutely necessary during letdown into fog producing altitudes, this method to control cabin air temperatures, is plausible. However it must be remembered when operating in manual override, the suit vent air must be off. Suit vent air can be turned on again after increasing defog air temperature. It also must be remembered that the RIO has no control over pressure suit air temperature. He can control flow, but must accept the pilot selected temperature. **So if the pilot turns vent air off, driving the mixing valve to cold, the RIO will be receiving full cold air, unless he elects to turn it off.**

Note

If cockpit and suit temperature becomes too high at low altitudes, and cannot be lowered, open face plate, take-off gloves, and unzip pressure suit to afford some degree of comfort.

WINDSHIELD DEFOGGING

Fogging of the windshield is prevented by heating the inside surface with incoming cabin air that is diverted into the defogging manifolds located along the lower surfaces of the side and center panels. The defog lever (figure 4-4) located on the pilot's right console outboard of the right utility panel is provided to select windshield defogging. The lever proportions the cabin airflow between the footheaters and windshield defogging tubes such that in the full aft (FOOTHEAT) position approximately 90% of the total cabin airflow is delivered to the pilot's and RIO air distribution manifolds and 10% through the windshield defog manifold. At the full forward (DEFOG) position approximately 20% of the total airflow is delivered through the footheat manifolds and 80% through the windshield defog manifold. Actuation of the DEFOG (Hi Temp) range temperature schedule is achieved only after the lever travel has been moved approximately 50% toward DEFOG. The pilot should attempt to anticipate fogging conditions so that, through proper management of temperature and airflow, it will not become necessary to subject the windshield and the crew to high temperatures and defog airflows which are required in order to clear an already fogged windshield. **The following procedure is recommended for prevention of windshield fogging during letdown into hot humid atmospheres:**

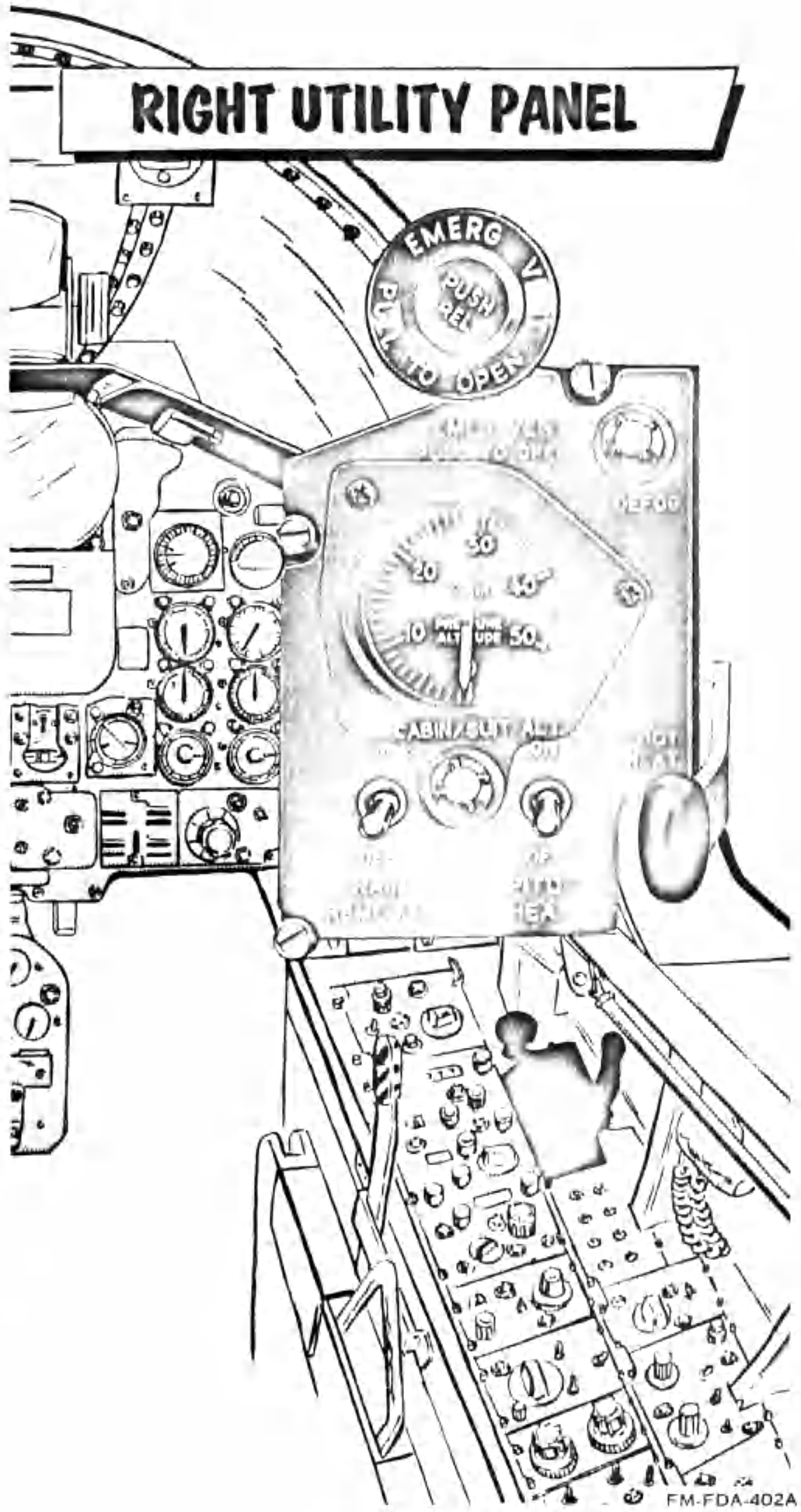

Figure 4-4

1. Place override selector switch (located on temperature control panel) in AUTO position.
2. During all cruise operations, position the defog lever not less than 5/8 of full travel toward the DEFOG position.
3. About 5 minutes prior to letdown, select the full DEFOG position of the defog lever and place the temperature control knob at the 2 o'clock (200 degree of clockwise rotation) position.
4. Retain these control settings throughout the letdown.

The same level of cabin temperature and windshield defogging should be maintained during both pressure suit and non-pressure suit operation.

Note

Retract flaps if fogging persist and will not clear up.

Windshield Rain Removal

During taxiing, take-off and landing, the pilot, by placing the rain removal switch (figure 4-4) located on the right utility panel, to the ON position will open a valve causing warm air to flow through nozzles directed up the outer surface of the windshield center and left side panels. This air breaks up the rain drops into smaller particles and diverts the majority of them up over the windshield. The system is adequate when flying through light rain, however, rain removal has been found to be marginal when flying through moderate to heavy precipitation with flaps down (BLC operating). Due to the fact that the rain removal system utilizes engine bleed air, rain removal is better or will be improved when the flaps are up or have been raised. A warning light on the caution lights panel will illuminate in the event the windshield material approaches a temperature which will cause optical deterioration. The pilot should then decide whether to risk optical deterioration or to turn the rain removal switch OFF. If the windshield temperature is known to result from aerodynamic heating only (occurs only near level flight maximum speed with maximum afterburning), the overheat signal should be disregarded.

Note

Prior to flight, a rain repellant should be applied to the windshield.

Cabin Turbine Overspeed Warning Light

The cabin turbine overspeed warning light (figure 1-10) is located on the pilot's caution lights panel. The warning light will illuminate when the cooling turbine in the cabin/pressure suit refrigeration unit is being subjected to pressures and temperatures in excess of normal operation, and is therefore subject to premature failure. If possible, the airplane speed and engine power should be reduced until the light goes out. If the light fails to go out, the pilot should then select ram air by pulling UP on the emergency vent knob, which will divert ram air into the cockpit and at the same time shut off bleed air to the refrigeration unit thereby stopping the cooling turbine.

CABIN PRESSURIZATION

With the canopy closed and the engine and cabin refrigeration system in operation, the cabin will automatically become pressurized at an altitude of 8,000 feet and above. The pressure in the cabin is maintained by the cabin pressure regulator (located on the cockpit floor aft of the RIO seat), which controls the outflow of air from the cockpit. Below 8,000 feet, the regulator relieves cockpit air at a rate to keep the cockpit unpressurized. Above 8,000 feet, the regulator relieves cabin air as necessary to follow a definite cabin pressure schedule (figure 4-3). Operation of the pressure regulator is completely automatic. The cabin safety (and dump) valve is used to prevent the cabin pressure differential from exceeding the limit positive or negative differential pressure in case of malfunctioning of the cabin pressure regulator, and to provide an emergency means of dumping the cockpit air. The dump feature of the safety valve is pneumatically connected to a dump feature on the cabin pressure regulator. Both valves which are operated pneumatically from a single control have sufficient capacity to permit the cabin differential pressure to be reduced from 5.5 psi to .05 psi within 5 seconds or less.

PRESSURE SUIT PRESSURIZATION SYSTEM

The cabin/pressure suit air conditioning is designed to deliver up to 10 cfm of air per suit at any temperature selected by the pilot. This air is provided at a pressure of 3 ± .2 psi above cabin pressure to insure proper flow of ventilating air through the suit. For operation when the cabin altitude exceeds 35,000 feet, the air is provided at a pressure of 6.5 ± .2 psi. In the event the pilot selects ram air for the cabin, thereby shutting off the flow of conditioned air, the flow of air from the refrigeration unit to the pressure suit will also be stopped. If the pilot should select ram air above 35,000 feet and the flow of conditioned air to the suit is stopped, suit pressurization will be automatically and instantly provided by the ship's oxygen system.

Note

The radar observer can control flow through his pressure suit by means of a manual flow control and shutoff valve located on the left side of his cockpit. The radar observer has no control over cockpit or pressure suit temperatures.

Cabin/Suit Pressure Altimeter

The pressure altitude of the cockpit and pressure suit is indicated on pressure altimeter. The pilot's cabin/suit altimeter (figure 4-4) is located on the right console. The radar observer's cabin/suit altimeter (figure 4-6) is located in a panel on the left side of the aft cockpit. The cabin/suit altimeters are vented to the pressure suit when worn or to cockpit pressure when pressure suit is not worn. Effective airplanes 148252d thru 148275e the cabin pressure altimeter indicates only cabin pressure.

MK IV MOD 1 PRESSURE SUIT

The full pressure suit consists of a two-layer garment. The inner garment is an air tight, ozone resistant, neoprene rubber layer, the only interruption being at the entrance which is sealed by a pressure sealing entrance slide fastener. The outer restraining garment is of nylon and is equipped with the necessary straps, tie-down, slide fasteners and adjustments to enable donning and fitting of the suit. The neck ring on the suit is the attachment point for the helmet and is also a pivot for the head and neck. On the left side of the suit there are two ports, one for ventilating air attachment, and one for the "g" suit attachment. On the right side of the suit is an exhaust port. The helmet incorporates the breathing regulator and communication equipment. Suit ventilating air is supplied from the aircraft's conditioned and regulated supply via the composite disconnect to the ventilating air hose and suit inlet port. From the suit inlet port air flows through ventilation tubes built into the pressure suit body and exists at the wrists, ankles, and the back of the neck. It then flows into the body of the suit, out through the exhaust port and through a hose to the suit controller which is located in the survival kit. In normal flight, air is allowed to pass through the controller and is expelled through its exhaust port, underneath the right forward corner of the survival kit. There is no need for the suit to be pressurized when safe cabin pressure prevails, and any slight pressurization that occurs is due to some suit resistance to ventilation airflow. However, the pressure suit system is devised so that the suit will be instantly pressurized if the ambient (cockpit) pressure falls below 35,000 ft. pressure altitude. The full pressure suit system is designed to respond to any function under several distinct conditions that might be encountered in flight, namely:

1. In normal flight with adequate cabin pressurization the suit is pressurized only to a very slight extent due to the suit resistance to air flow.
2. When ambient (cockpit) absolute pressure falls below 35,000 ft. pressure altitude the suit will become pressurized by the ventilating air.
3. If ventilation air supply is lost, the ship's oxygen supply will pressurize the suit as well as supply breathing oxygen.
4. Lastly, if ship's oxygen fails or in the event of bailout, the bailout oxygen will pressurize the suit as well as supply breathing oxygen.

HELMET

The helmet is equipped with an oxygen regulator (Model GR 70 or Model GR 90) face visor, visor seal, face seal, helmet liner adjusting mechanism, sun visor and a dynamic (AIC-10) lip mike and headset. The oxygen regulator is provided to control the flow of oxygen to the facial area. The regulator is divided into two compartments separated by a diaphragm. One compartment has an opening leading directly to the facial area, while the other compartment has an opening directly to the area behind the face seal. The second opening senses the suit pressure, and will keep the oxygen to the facial area at slightly higher pressure than that of the suit, so that the suit air will never flow into the facial area. When the demand valve is opened by the negative pressure of inhalation, oxygen is emitted to the facial area via the perforated visor defogging tube. During exhalation a positive pressure of .072 psi will force the exhaled gases to pass from the facial area of the pressure suit body via exhalation valves located in the area below the chin. These valves permit flow in one direction only and therefore will prevent any gases from the suit area into the facial area. The face visor is sealed when closed by means of a tube which is inflated by oxygen when the on-off switch on the regulator is switched to ON. To raise the face visor, the seal is deflated on early issue helmets equipped with GR 70 oxygen regulators, by first turning the on-off switch OFF and then by depressing the spring-loaded visor seal relief plunger and inhaling to open the tilt valve allowing the oxygen to escape from the seal. On current issue helmets equipped with GR 90 oxygen regulators, the face visor seal is deflated by merely turning the on-off switch OFF and inhaling. If the oxygen is turned on when the visor is in the up position, oxygen will flow freely to atmosphere from the visor defogging tube. Oxygen is prevented from flowing directly into the rear of the helmet and into the body section of the suit by a face seal. The face seal forms the closure that separates the 100% oxygen in the face compartment from the gases of the suit. This seal is shaped to form a continuous line over the occupant's forehead, down along his cheeks and across his chin. To gain this seal, contact with the subject's face should be as complete as possible. This requires that the seal be properly fitted and shaped when donning the helmet. A badly fitted face seal will be noted by a continuous flow of oxygen from the regulator. The helmet liner adjusting mechanism consists of an external knob and internal straps wrapped around a tightening mechanism. Turning the knob in either direction will tighten the helmet liner on the head and also force the head forward into the face seal. The face seal, when formed properly, and the helmet liner, which adjusted properly will provide the occupant with the best possible degree of comfort while wearing the helmet. The full pressure suit helmet provides equivalent crash protection to the standard Navy helmet, but the nature of its attachment to the suit body, provides superior helmet retaining qualities.

SUIT CONTROLLER

The suit controller, located in the survival kit, is the heart of the pressure suit system. It is completely automatic and requires no adjustment or control by the suit occupant. All exhaust air must pass through the suit controller prior to being exhausted. Through restriction of the exhaust flow of vent air, the controller prevents the suit pressure from dropping below a pressure equal to 35,000 ft. As the cockpit altitude rises above 35,000 feet, the controller will begin to restrict the exhaust flow, causing a pressure build-up in the suit. This pressure will be maintained at an absolute pressure equal to 35,000 feet pressure altitude. The differential suit pressure will be the difference between 35,000 feet equivalent and the cockpit pressure. The suit controller operates on a balance pressure being kept between the suit and the internal altitude reference chamber of the controller. The internal altitude reference chamber of the controller is continuously fed with a metered flow of the ship's oxygen at approximately 100 to 150 cc/min. The outlet flow of this oxygen from the reference chamber is controlled by an aneroid operated valve which senses cockpit pressure and regulates the outlet flow to maintain a pressure equal to 35,000 feet within the suit controller. In the event of ventilation air loss to the pressure suit, the pressure in the suit controller will drop, causing the suit controller exhaust valve to close. At the same time, a tilt valve will open allowing oxygen to flow through the suit controller and exhaust hose to the suit. The suit will then be pressurized by oxygen. The suit controller will continue to maintain suit pressures in the same way as it did when the suit was using ventilation air for pressurization. A check valve in the ventilation inlet line will prevent any oxygen in the suit from escaping. There will be no ventilation air flow when the suit is being pressurized by oxygen. Upon ejection above 35,000 feet, the suit controller will still maintain a no greater than 35,000 ft. pressure altitude in the suit. The only difference being that the oxygen supply for controller operation and suit pressurization will come from the occupant's bailout oxygen supply which will be triggered upon ejection.

COMPOSITE DISCONNECT

The composite disconnect is designed to be used in conjunction with the suit controller and the suit control system of which it is a part. The composite disconnect assembly consists of a lower block, an intermediate block, and an upper block. The intermediate block is fastened to the upper part of the survival kit and contains the tie-in between the crewman and the emergency oxygen supply. The disconnect is so designed that the aircrewman, during normal aircraft entrance or departure, is capable of quickly attaching or detaching all hoses and electrical lines leading from the aircraft to the survival kit and man. The lower block contains check valves in the ventilating air, anti-"g", and oxygen ports, that are open when the three sections of the disconnect are plugged in, and closed when either the upper or lower blocks are disconnected from the intermediate block. The check valves prevent gas leakage in the normal direction of flow when the valves are closed. The lower block is provided with a lanyard operated locking device, the free end of the lanyard being attached to airplane structure. As the seat is ejected, tension in the lanyard unlocks the device and separates the lower block from the intermediate block. The intermediate block serves as the connecting link between upper and lower block and, in addition, by means of a "Tee" in the oxygen line, connects the emergency oxygen to the system. The upper block provides the means for attaching all service lines corresponding to those leading to the lower block. It also contains a manual disconnect device that permits the aircrewman to free himself from all kit connections during normal aircraft departure by a single pull on the manual disconnect handle. This action simultaneously unlocks the upper block from the intermediate block and the vent exhaust hose from the kit.

WARNING

Replacement O ring seals for composite disconnect must be cleaned with solvent MIL-T-7003 prior to installation to preclude the possibility of an oxygen oil explosion occurring upon composite disconnect insertion or inflight.

Note

- Prior to each flight, lubricate the exposed seals of the composite disconnect upper block with lubricant MIL-C-21567.
- A portable pressure suit ventilating unit should be available for use from the ready room to the airplane.

PRESSURE SUIT OPERATION

After donning the suit, vent air should be applied if it is available, especially if any delay is anticipated before going out to the aircraft. It doesn't take long to become dehydrated in warm weather. To avoid visor fogging, coat the visor with the recommended anti-fogging compound. After getting to the airplane, vent air is available from the Auxiliary Power Unit (CP-105) either in the cockpit or standing outside the airplane. Before entering the cockpit the vent air hose, anti-g hose, pressure sensing hose and exhaust hose connections to the suit should be checked secure. The first two things to do after climbing in the cockpit is to connect the composite disconnect to the upper block and connect the exhaust hose. The exhaust hose cannot be connected until the composite disconnect is locked. After these two connections are made the vent air may be turned on before finishing the job of getting strapped in. (If the exhaust hose is not connected, above normal ballooning of suit will occur and ballooning will be slow to dissipate even after vent air has been turned off.) After the helmet and gloves have been donned and the assistant has plugged in the communications and oxygen line to the helmet, the oxygen system should be checked by closing the visor, turning on the oxygen and checking the visor seal. If the system checks out satisfactorily remain on oxygen. Normal flight procedures are then followed until take-off position is reached where the vent air should be turned down to prevent suit ballooning as power is advanced on the take-off roll. As soon as the airplane is cleaned up after take-off the vent air should be adjusted to a comfortable level. Vent air temperature is adjustable by use of the cockpit temperature control. There are two systems in use to monitor suit altitude. With the suit sensing line connected, the cockpit altimeter indicates altitude within the suit; or a suit altimeter may be installed on the left leg of the suit.

CABIN/PRESSURE SUIT TEMPERATURE SCHEDULE

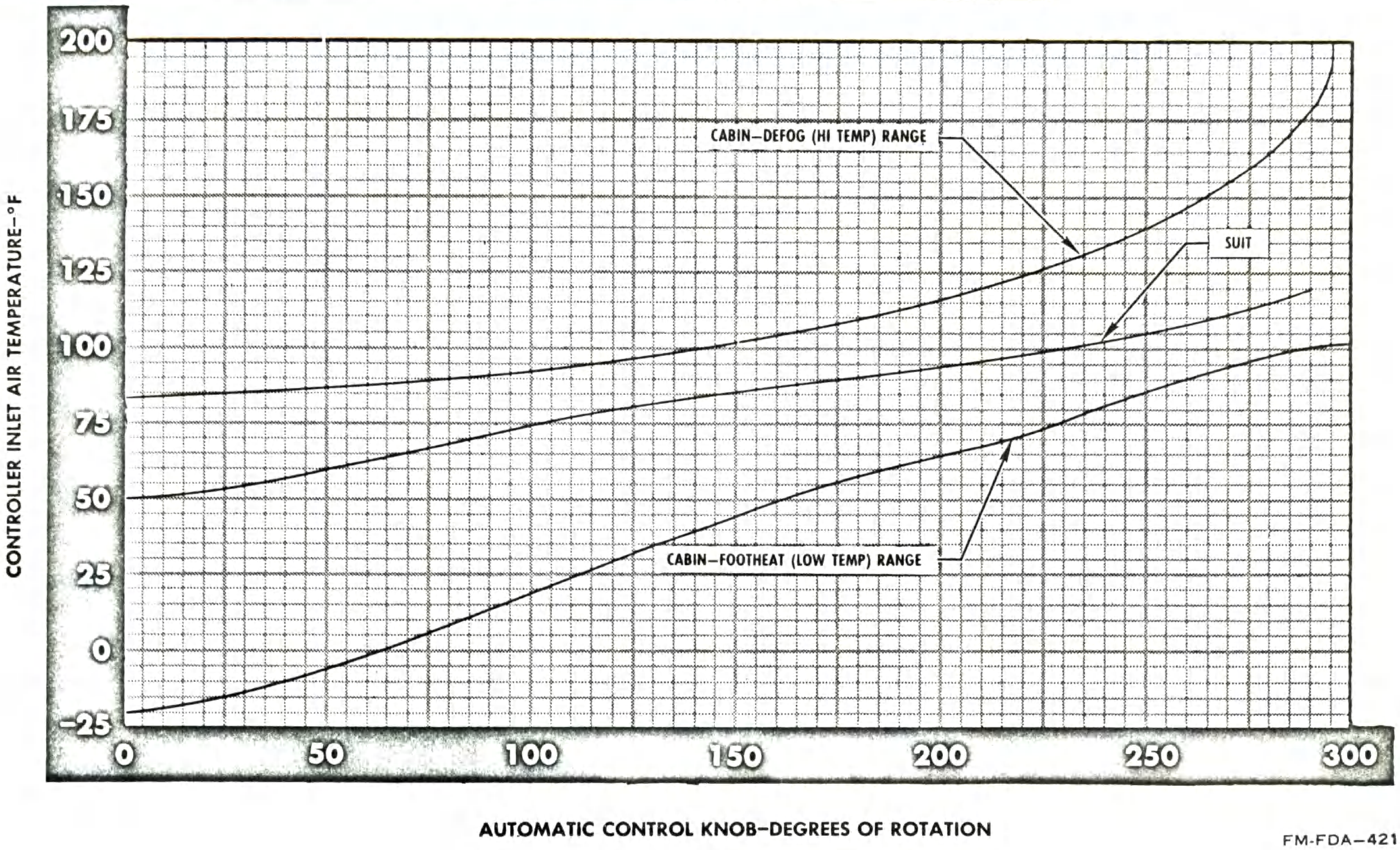

Figure 4-5

PRESSURE SUIT IN-FLIGHT CHECK

CAUTION

Prior to pulling pressure suit in-flight check turn radar equipment off. Depressuring cockpit above 32,000 feet with radar equipment on will result in damage to the radar indicators.

Pressure suit operation can be check in-flight by performing the following:

1. Between 12,000 and 20,000 feet.
   a. Emergency vent knob - PULL
2. Climb to 40,000 feet.
   a. Pressure suit operation - CHECK
3. After checking suit operation.
   a. Emergency vent knob - PUSH IN

EMERGENCY VENT HANDLE

The cockpits may be cleared of undesired smoke or fumes and the cabin/pressure suit refrigeration unit may be shut off by pulling up on the emergency ventilating handle (figure 4-4). Push button on top of knob and then pull up on the knob. The handle may be placed in an intermediate position to obtain desired amount of emergency ventilation. When pulled up, three actions occur simultaneously:

1. All air conditioning and pressurization air from the cabin/pressure suit refrigeration unit to the cockpits and pressure suits is shut off.
2. The cabin/pressure regulator and safety (dump) valve is opened and the cockpit becomes completely depressurized.
3. A ram air shutoff valve is opened and the atmospheric air is allowed to enter the cockpit through a port located just forward of the pilot's feet.

Note

Operation of the emergency ventilating control knob is the only method of shutting off the cabin/pressure suit refrigeration unit when the engine is operating.

COCKPIT HEATING PROCEDURE

1. Place the override selector switch in the AUTO position.
2. Adjust temperature control knob for desired cockpit temperature.
3. Position normal ventilating control lever for personal comfort and effective windshield defogging.
4. If the automatic temperature control system fails or if you desire a cooler or hotter temperature within the pressure suit when operating with the pressure suit a temporary adjustment may be obtained by "bumping" the override selector switch to the HOT or COLD position.

RIO'S UTILITY PANEL

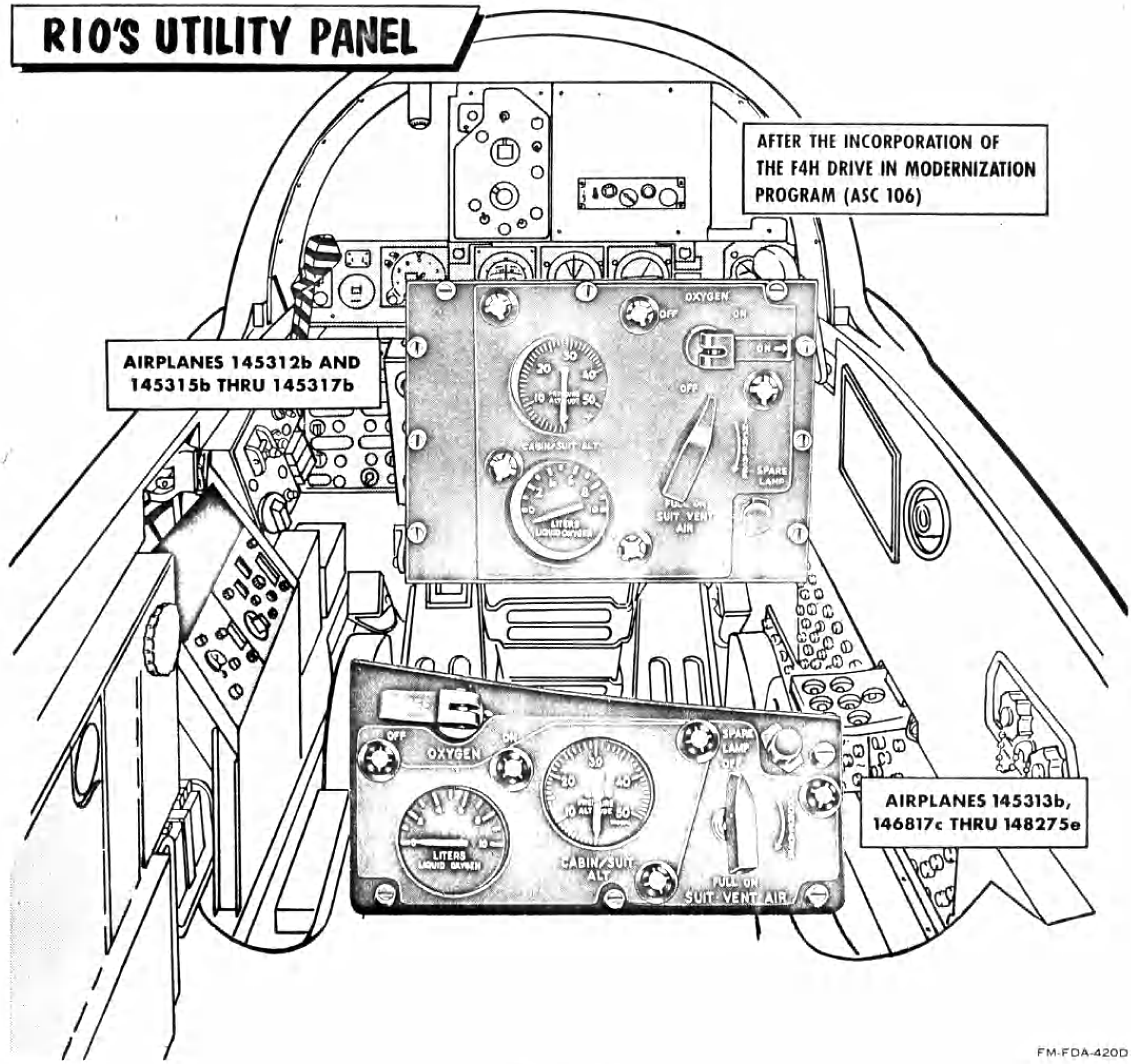

Figure 4-6

EQUIPMENT COOLING SYSTEM

The equipment air conditioning system consists of an air-to-air heat exchanger, pressure regulating and shutoff valve, expansion turbine, mixing valve and temperature controls. The unit provides refrigerated air for cooling the nose located radar equipment and the communication navigation identification equipment located under the RO's cockpit floor. It also provides high pressure, ram-cooled air (auxiliary air) for fuselage fuel tank pressurization, wing fuel tank pressurization and transfer, anti "g" suits, electronic equipment pressurization, high pressure pneumatic system air source, CADC, angle of attack transmitter, canopy seal pressure and turn and slip indicator. Operation of the system is entirely automatic with airflow being initiated on engine start. Engine bleed air, after flowing through the heat exchanger and pressure regulating valve, is expanded through the cooling turbine and then mixed with warm bleed air as necessary to provide a nominal 40°F delivery temperature. Effective airplanes 146817c thru 148275e, the nominal delivery temperature is 85°F from sea level to 25,000 feet, and 40°F above 25,000 feet. The air is then ducted directly to the electronic equipment. In the event of a system failure such that refrigerated air temperatures exceed 150 + 10°F, the refrigeration unit will be automatically shut off and emergency ram air cooling will be provided. A warning light labeled "Radar CNI Cool Off" will illuminate on the radar observers instrument panel whenever ram air is being utilized for cooling. A reset button labeled cooling

COMMUNICATION AND ASSOCIATED ELECTRONIC EQUIPMENT

TYPE DESIGNATION	FUNCTION	RANGE	LOCATION AND CONTROL
INTERCOM AN ASQ-19	Intercockpit communications.		Intercom panels on pilot's left console and RIO's instrument panel.
UHF RADIO AN ASQ-19	AM radio telephone communication between aircraft and ship, aircraft and shore, or between aircraft.	Up to line of sight, depending upon frequency and antenna coverage.	Com-Nav group control panels on pilot's left console and RIO's instrument panel.
DIRECTION FINDER AN ASQ-19	Indicates relative bearing of and homes on radio signal sources	Same as for UHF Radio	Com-Nav group control panels, pilot's course indicator on instrument panel.
TACAN AN ASQ-19	Indicates bearing and distance of suitably equipped ground stations. Determines identity of beacon and indicates dependability of beacon.	Line-of-sight distances up to 196miles depending upon altitude.	Com-Nav group control panels, Bearing-Distance Heading indicators on pilot's and RIO's instrument panels.
IFF RADAR AN APX-6B	Identifies aircraft as friend or foe.	0-200 miles or line-of-sight.	IFF control panel on pilot's right console.
SIF AN APA-89	Provides specific identification of separate airplanes or of single airplane within a group.	Same as IFF Radar.	SIF control panel on pilot's right console.
RADAR ALTIMETER AN APN-22	Indicates distance in feet from airplane to ground.	Land 0-10,000 feet Water 0-20,000 feet	Indicator on pilot's instrument panel, with self-contained warning light.
NAVIGATION COMPUTER	Provides great circle dead reckoning computations during flight.	Not applicable	Navigation Computer control panel aft cockpit.
RADAR SET AN APQ-72	Detects and tracks airborne targets; provides point-of-aim. Provides surface search for navigation purposes.	200 mile Map 100 mile Search 50 and 24 mile Track	Azimuth-Elevation range indicators on pilot's and RIO's instrument panels, and control panel below RIO's instrument panel.

FM-FDA-404B

Figure 4-7

reset is located on the RO's main instrument panel. In the event the "Radar CNI Cool Off" light illuminates, attempt to restart the refrigeration unit by reducing airspeed, waiting at least 15 seconds, and then depressing the cooling reset button. If the refrigeration unit fails to restart, no further restart attempt should be made. Effective airplanes 148252d thru 148275e, a "Radar CNI Cool Off" light and cooling reset button are located in the pilot's cockpit.

CAUTION

When operating with emergency ram air cooling, avoid high speed flight if possible. Maximum allowable cooling temperatures may be exceeded during high speed flight with the result that electronic equipment life and/or reliability may be effected.

Note

Illumination of the "Radar CNI Cool Off" light shall be logged on the yellow sheet (OPNAV FORM 3760-2).

NOSE RADAR COMPARTMENT PRESSURIZATION

The airplane nose section is automatically pressurized to a nominal 1.6 psi above ambient pressure by an outflow type pressure regulator mounted on the forward wall of the nose gear compartment. Pressurization is initiated when the landing gear is raised, and is relieved when the gear is lowered or equipment package shutoff occurs.

ENGINE ANTI-ICING SYSTEM

The engine anti-icing system is a compressor discharge air bleed type system, controlled by an on off pressure regulating valve. Air for anti-icing purposes is supplied from the 17th stage of the engine compressor at pressures up to 275 psig, and temperatures up to 593°C. A regulator incorporated in the anti-icing valve reduces the incoming air to a pressure of approximately 14-20 psig. Air from the anti-icing valve is distributed to the engine front frame struts, and to the variable guide vanes.

Engine Anti-Icing Switch

A two-position engine anti-icing switch (figure 1-4) is located on the outboard engine control panel. The switch is marked engine anti-icing and the switch positions are DE-ICE and NORMAL. Placing the switch in the DE-ICE position opens the regulator valve which starts anti-icing air flow. With the switch in the NORMAL position, no anti-icing operation is being performed. Electrical power for the anti-icing circuit is provided by the left main 115 volt a-c bus.

CAUTION

Do not place engine anti-icing switch in the DE-ICE position unless actual engine icing conditions are encountered.

COMMUNICATION-NAVIGATION-IDENTIFICATION (CNI) EQUIPMENT

The Integrated Electronic Control AN/ASQ-19 (XN-1) provides communication-navigation-identification (CNI) functions for the aircraft. Communication functions are provided by a UHF receiver-transmitter, an auxiliary receiver, and an intercom. Navigation functions are provided by TACAN (Tactical Air Navigation) and ADF (Automatic Direction Finding) equipment. An IFF (Identification Friend or Foe) Coder Unit provides identification of the aircraft when challenged. The CNI system is supplied with power from a central power supply.

CAUTION

Prior to zoom climbs, and/or flights above 60,000 feet turn all CNI equipment OFF. Flight above 60,000 feet with CNI equipment on will result in damage to equipment.

INTERCOM SYSTEM

An intercom panel (figure 4-8) in both the pilot's and radar observer's cockpits provides a means of intercockpit communication in the airplane. In the pilot's cockpit the panel is located on the aft inboard portion of the left console. The radar observer's intercom panel is located on the instrument panel below the true airspeed indicator. An external station, connected in parallel with the aft cockpit microphone and headset, is provided for use by ground personnel. An additional function of the intercom system is to amplify a-f signals received from various sources. Transmission over the UHF receiver transmitter may be accomplished in either cockpit in conjunction with the intercom system.

FUNCTION AND LOCATION OF CONTROLS

Volume Control

The intercom volume control knobs (figure 4-8) are located at the left side of the pilot's intercom panel, and at the right side of the radar observer's intercom panel. The input level of the intercom signals to the headsets is increased by rotating the respective volume control knobs in a clockwise direction. The signals received from the radio receivers are not affected by operation of these intercom volume controls.

Radio Override Switch (Pilot's)

A three-position toggle switch (figure 4-8) with positions of RADIO OVERRIDE, HOT MIC and COLD MIC is located on the right side of the pilot's intercom control panel. The RADIO OVERRIDE position of the switch is momentary, the HOT MIC and COLD MIC positions are fixed. The HOT MIC position is used when duplex operation of the intercom system is de-

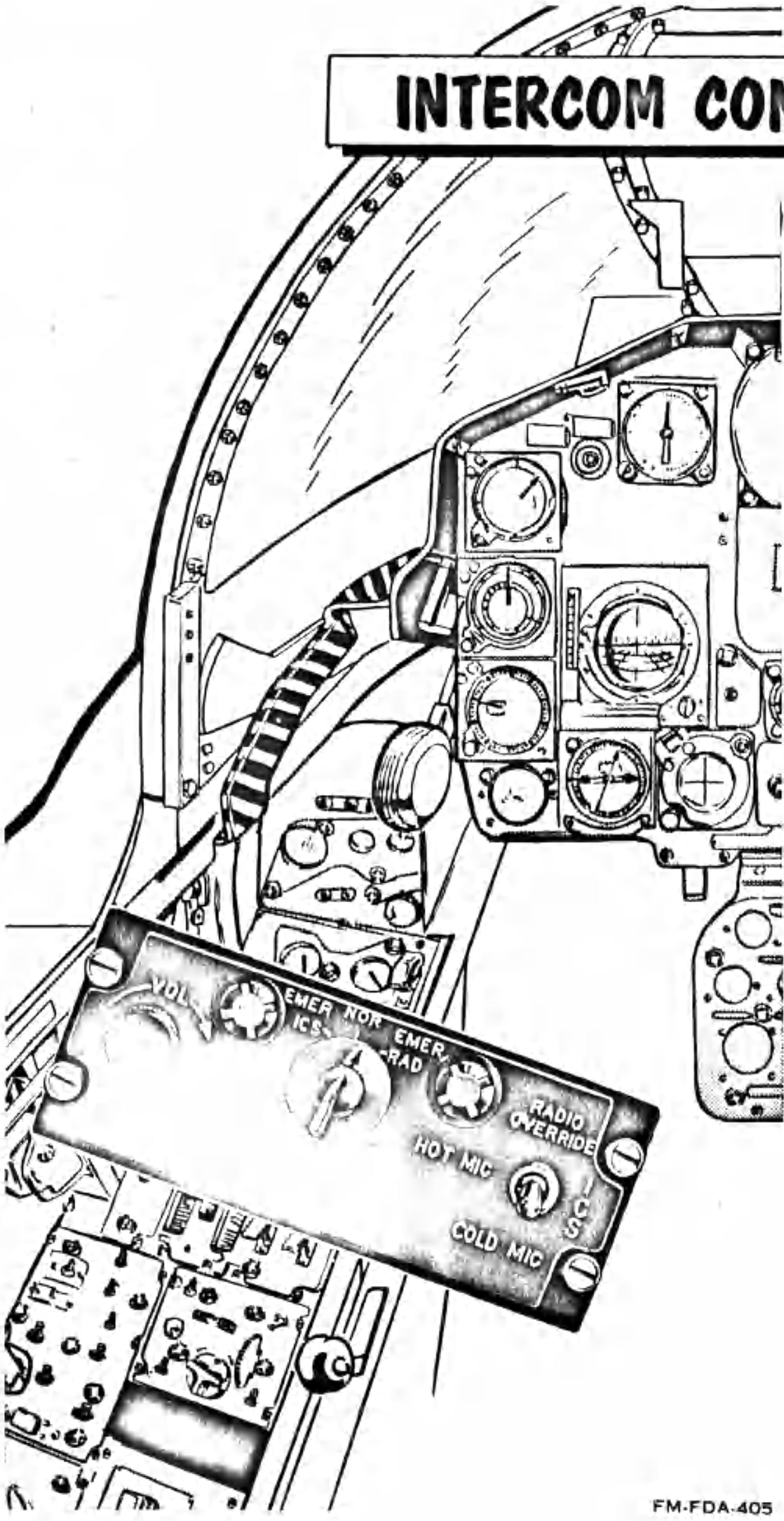

Figure 4-8

sired. The RADIO OVERRIDE position is identical with HOT MIC except that all radio gain is reduced, communication between cockpits then overrides radio reception. When the switch is set at the COLD MIC position, normal radio reception and transmission is still available, but the pilot can no longer communicate with the RO over the intercom system without going to the ICS switch on the throttle. However, the RO can still transmit to the pilot by placing the radio override switch in the RADIO OVERRIDE position.

Radio Override Switch (Radar Observer's)

Incorporated in all airplanes except 145313b, a two-position toggle switch with positions of RADIO OVERRIDE and NORMAL is located at the left side of the radar observer's intercom panel. The NORMAL position is used for normal duplex operation at the same time the pilot's switch is set at HOT MIC. Radio signals are received at normal volume when the radar observer's switch is set at NORMAL. The RADIO OVERRIDE position is used to reduce the reception of radio signals in both cockpits. This switch position may also be employed to accomplish intercockpit communication if the pilot's switch is set at COLD MIC. The RADIO OVERRIDE position is a momentary switch position.

Radio Override Foot Switch (Radar Observer's)

Applicable to airplanes 145313b, 146817c thru 148275e, a foot operated momentary switch is installed on the right foot ramp in the Radar Observer's cockpit. This switch is wired in parallel with the panel-mounted radio override switch. By depressing his foot switch the R.O. may override any of the positions selected on the Radio Override switch, allowing intercom transmission without the necessity of releasing other manual controls.

Emergency Switches

The emergency switches are three-position rotary type switches and are located in the center of both the pilot's and radar observer's intercom control panel. The operator uses these controls to bypass an amplifier if it should go dead. Both operators may have occasion to switch to one of the emergency settings at the same time. In certain instances of amplifier failure, this arrangement is necessary in order to maintain intercockpit communication. There are three possible settings for each control. The NOR position is used when both amplifying stages in the respective control boxes are functioning properly. The other two positions for each control are EMER RAD and EMER ICS, which are used when it is desired to bypass a faulty or dead amplifier.

Microphone Button

The microphone buttons are used to connect the microphone outputs to the UHF transmitter. The pilot's microphone button is located on the inboard throttle control, the radar observer's microphone button is foot operated by a toe switch on the left foot ramp on all airplanes except 145313b where it is located on the radar hand control. When the operator wishes to transmit, he depresses the microphone button, the transmitter portion of the UHF Radio Receiver-Transmitter is on the air and the output from the microphone is fed into the transmitter. The positioning of the other controls in no way affects the transmitting operation from either cockpit.

INTERCOM PREFLIGHT CHECK

The controls should be set in the following manner in order to check the equipment before take-off:

Pilot's Controls

Switch	Position
Function Selector Switch	HOT MIC
Amplifier Selector Knob	NOR
Volume.	Rotate Clockwise

Radar Observer's Controls

Switch	Position
Function Selector Switch	NORMAL
Amplifier Selector Knob	NOR
Volume.	Rotate Clockwise

With controls positioned as stated, check the duplex operation of the equipment by talking into the microphones. Rotate the VOL controls to insure that they are operating properly. Switch to the EMER ICS - NOR - EMER RAD positions to make sure that they are mechanically sound. The radio override functions of the interphone should be checked by each operator. In order to check the equipment properly, care should be taken that the operators do not switch to RADIO OVERRIDE at the same time, since reduction in the volume for the radio receivers is accomplished in both headsets when only one of the operators goes to RADIO OVERRIDE. Therefore, each control must be positioned at different times to check the radio override circuitry in each unit. The Radar Observer in airplane 145313b must check radio override functions by means of both his panel-mounted and foot operated Radio Override switches.

NORMAL OPERATION OF INTERCOM SYSTEM

The intercom system is operated with 27.5 V d-c power without additional switching as soon as the aircraft power supply is turned ON. The system will become inoperative when the aircraft power is switched OFF.

Operation Procedure

It is assumed here that all four of the amplifying stages are working and the intercom system is functioning normally.

1. The pilot positions his radio override toggle switch to the HOT MIC position to insure that both microphones are connected to the interphone system.
2. The radar observer's radio override toggle switch should be placed in the NORMAL position.
3. The emergency switches on both intercom panels should be set at the NOR positions.
4. The volume control may be rotated in each cockpit to set the input level to the respective headsets at a desired level.

No further switching is necessary to operate on duplex.

EMERGENCY OPERATION OF INTERCOM SYSTEM

Emergency operation of the intercom system is accomplished by manipulation of the various controls on each ICS panel. If, for example, the intercom system is operating in duplex, and one of the amplifying stages goes dead, the fault is corrected by the proper positioning of one or both of the amplifier selector knobs. If the dead amplifier is in the unit in the pilot's cockpit, the amplifier selector knob on the pilot's ICS panel must be placed in the EMER ICS position in order to bypass the faulty amplifier, and all other controls on both ICS panels remain in their normal position. However, if it is also necessary for the amplifier selector knob on the RO's ICS panel to be set at an emergency setting, duplex operation is no longer possible, and additional switching is necessary in order to maintain intercockpit communications. Assuming that the amplifier selector knob of the forward cockpit ICS panel is at any of the three possible positions, the other controls on the ICS panels should be set as follows to maintain operation of the intercom system.

Pilot's radio override switch - HOT MIC
Place the radio override switch in HOT MIC and utilize the RADIO OVERRIDE position of the switch as desired to reduce radio volume.

Radar observer's radio override switch - RADIO OVERRIDE
Place the radio override switch in RADIO OVERRIDE position to transmit to pilot and reduce radio volume. Place radio override switch in NORMAL position to receive from pilot.

All other controls are operated as if the system were operating on duplex.

Note

It should be pointed out that, even though the pilot's radio override switch is set at COLD MIC, the observer may talk and listen to the pilot if he switches to RADIO OVERRIDE. This is the only instance where duplex operation may be maintained when the pilot is not at HOT MIC. This switching arrangement is not normally used for intercockpit communication, since the pilot would usually be at HOT MIC position regardless of the setting of the other controls and the RADIO OVERRIDE settings are momentary switch positions.

RADIO RECEIVER TRANSMITTER

The radio receiver-transmitter is designed to broadcast and receive UHF frequencies in a range of 225.0 to 399.9 mc for interairplane or airplane-to-base communications. Complete control over the operation of the radio receiver-transmitter can be maintained by either the pilot or the radar observer through the Comm-Nav group control panel (figure 4-9) on panel located in each cockpit. The pilot's Comm-Nav group control panel is located on the right console, the radar observer's panel is located below the instrument panel,

COMM – NAV GROUP CONTROL PANEL

to the left of the radar scope. The Comm-Nav group control panel provides controls for operation of the radio receiver-transmitter on any one of the 1750 channels available, on 18 preset channels, a guard receiver frequency, or for ADF operation with associated direction finder equipment.

FUNCTION AND LOCATION OF CONTROLS

Communication Frequency Controls (Comm Freq MC)

The three control knobs at the top of the Comm-Nav group control panel are used to adjust manually the operating frequency of the radio receiver-transmitter when the comm channel is in the M position. When the comm channel is not in the M position, the manual control knobs do not affect the operating frequency.

Communication Channel Control (Comm Chan)

The comm channel control knob is located at the upper left on the Comm-Nav group control panel. This knob when rotated will select 18 preset channels of operation that will be shown in the comm channel window. There is also an M position, which permits the operator to select manually the operating frequency, and a G position, which permits operation on the guard frequency of 243.0mc.

UHF Remote Channel Indicator

The remote channel indicator is located on the left side of the instrument panel (figure 1-16). This enables the pilot to dial a channel with the communication channel control knob without shifting his vision from the instrument panel.

Function Switch

The function switch is located in the right center of the Comm-Nav group control panel. The switch is labeled UHF Comm and aux rec. Only the UHF Comm portion of the switch shall be discussed at this time. The switch consists of five positions described below:

STBY - The standby position indicates that only filament power is applied to the radio receiver-transmitter to warm-up the set.

T/R - In this position the radio receiver-transmitter is activated for transmitting and receiving operations. The set will normally be on receive until the microphone push-to-talk button is depressed.

T/R+G - In this position the radio receiver-transmitter will operate on transmit or receive within the 225.0 to 399.9 mc range; in addition, the guard receiver is turned on.

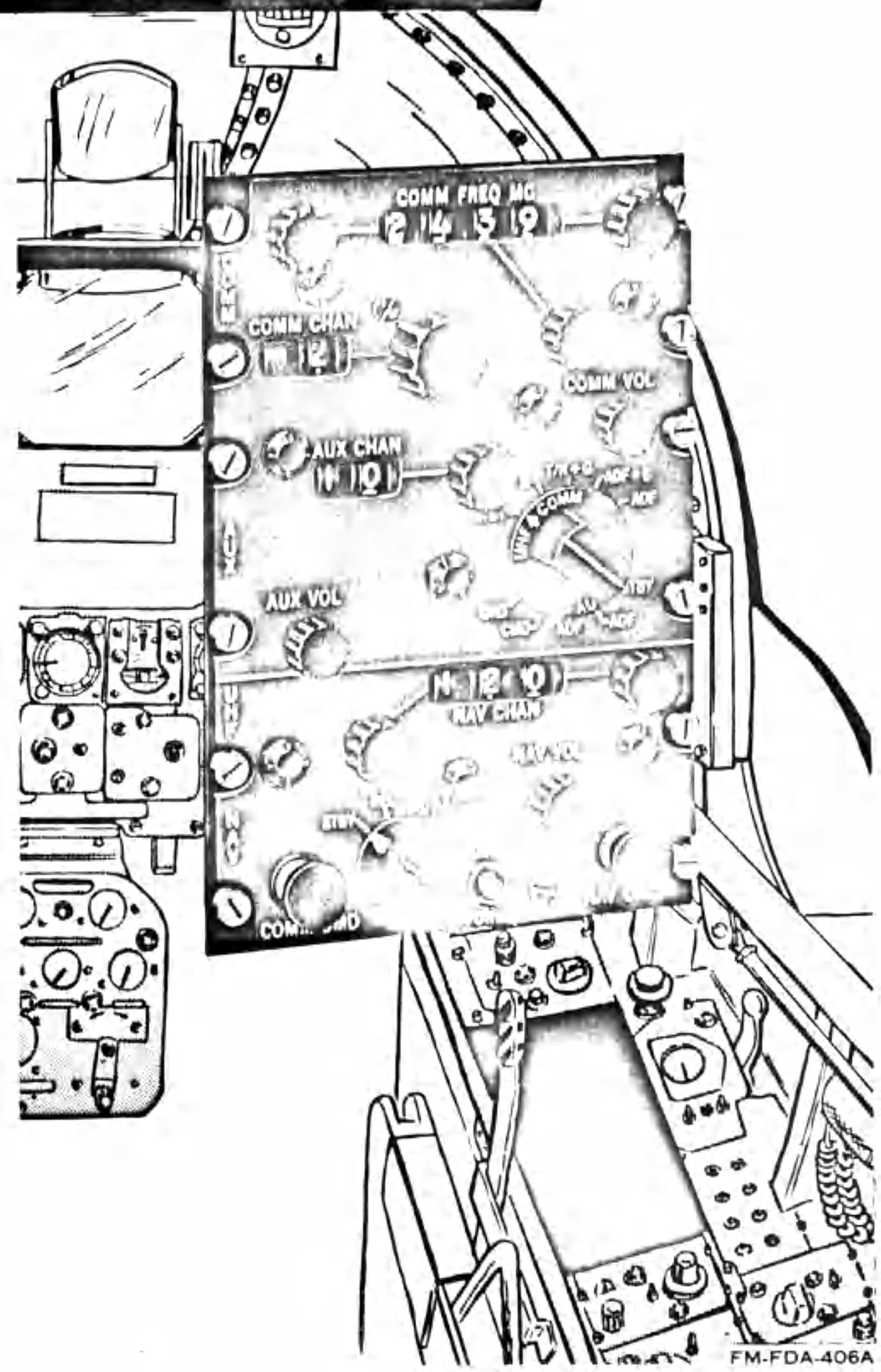

Figure 4-9

ADF+G - In this position the radio receiver-transmitter operates with the receiver-ADF-power supply unit and the antenna unit in an automatic direction finder system. This position will also furnish guard channel reception.

ADF - In this position the radio receiver-transmitter operates with the receiver-ADF-power supply unit and the antenna unit, in an automatic direction finder system.

RADIO RECEIVER-TRANSMITTER PREFLIGHT CHECK

A quick check to prove the equipment is operating at an adequate power level can be made by performing a receive-transmit check with the base control tower on several frequencies. If reception is poor at the airplane and/or at the control tower, the need for corrective maintenance is indicated.

OPERATING PROCEDURE

The power supply for the radio receiver-transmitter is energized when the airplane power source is activated.

Communication Channel Control Knob

1. Place the function switch in the T/R position.
2. Rotate the comm channel control knob to the desired channel of operation.
3. Adjust comm volume control knob to approximately one-half its complete clockwise rotation.
4. After the first received signal is heard in the headphones, the volume control may have to be adjusted.
5. Transmitting operation is accomplished on the same channel by depressing the push-to-talk microphone button.

Guard Receiver

1. Adjust function switch to the T/R+G position.
2. Adjust comm channel to any position other than G.
3. Check with air base control tower on guard channel frequency (243 mc).
4. Normal audio should be heard on headset.

Main Receiver Operation on Guard Frequency

1. Rotate comm channel control knob until G position is shown in comm channel window.
2. Place function switch in the T/R position.
3. Normal communication should result with the radio receiver-transmitter on guard frequency (243 mc).

Manual Adjustment of Frequencies

1. Rotate the comm channel control knob until the M position is shown in the comm channel window.
2. Place the function switch in the desired position, T/R or T/R+G.
3. Adjust each of the three manual frequency controls at the top of Nav-Comm group control panel to show the desired operating frequency in the comm frequency mc window.

ADF Operation

1. Place the function switch in the ADF+G position for ADF plus guard receiver operation, or place the switch in the ADF position for ADF operation only.

TAKE COMMAND CONTROL

A communication command push button control marked Comm. Cmd. is located in the lower left corner of each Comm-Nav group control panel, figure 4-9. Operation of this push button allows the operator to take or relinquish command of the aircraft communications system. A green light in the center of the push button lights up when the operator has control of the radio receiver-transmitter, the auxiliary receiver and the ADF functions. When one of the operators has command of the communication functions depressing the communication command button on the Comm-Nav group control panel will transfer the communication functions to the other operator. If one of the operators does not have control of the communication functions, depressing the communication command button will take away the communication functions from the other operator.

Note

Do not switch UHF channels when other operator is utilizing the UHF communications transmitter. The nature of the system is such that there is a possibility of an inadvertent change of channels on the users set.

EMERGENCY POWER OPERATION

Warning lights labeled "Emergency Power" are situated at the bottom of each Comm Nav group control panel. The light will illuminate when the power supply goes into emergency operation. Transistion from normal to emergency operation is completely automatic and is brought about by a malfunction in one of the major power supply sources. The portions of the system which operate in emergency power are radio receiver transmitter (reduced transmitter power) and the identification equipment. The TACAN functions are inoperative, and depending upon the type of failure, the IFF may also be inoperative.

AUXILIARY RECEIVER

An auxiliary receiver is utilized in the airplane communications system that works in conjunction with the main radio receiver-transmitter under normal conditions and operates as an emergency receiver in the event of power failure to the main radio receiver-transmitter. The auxiliary receiver can be used as a conventional radio receiver for reception of AM radio signals in the frequency range of 265.0 to 284.9 mc or as an ADF receiver for reception of radio signals in the same frequency. The auxiliary receiver can be placed in either function by operation of the controls on either of the Comm-Nav group control panels. Channel selection is also accomplished by operation of the aux chan control on one of the Comm-Nav group control panels. The function switch controls the functions of the auxiliary receiver equipment and provides for either the auxiliary receiver or the main radio receiver-transmitter to be operating as an ADF receiver while the other equipment is operating as a voice receiver. The direction finder group of the auxiliary receiver provides the pilot with continuous indication of the direction of arrival of r-f signals intercepted by either the radio receiver-transmitter or the auxiliary receiver which is used in conjunction with the ADF system. These receivers function to intercept amplitude modulated and unmodulated signals in the frequency range of 225 through 400 mc and 265 through 284.9 mc depending on which receiver is used. Continuous indication of the bearing of the intercepted signals relative to the airplane heading is presented by the No. 1 (narrow) pointer on the course indicator in airplanes 145311b, 145312b and 145314b thru 145317b. The same information is given by the bearing pointer on the horizontal situation indicator on airplane 145313b. Necessary primary power (115 V a-c) is applied to the auxiliary receiver when the aircraft electrical system is energized.

AUXILIARY RECEIVER CONTROLS

The controls for the operation of the auxiliary receiver are located in the middle of each Comm-Nav group control panel (figure 4-9). The controls consist of a channel selector switch, volume control, sensitivity control and function switch.

Channel Selector

The operating channels for the auxiliary receiver are selected by operation of the auxiliary channel control knob located on the left side of each Comm-Nav group control panel. The channel selected by rotating the control is shown in the window directly to the right of the auxiliary channel control knob.

Volume Control

The auxiliary receiver volume control knob is located directly below the auxiliary channel window on the group controls. Operating this control varies the audio input level to the intercom loop.

Function Switch

The function switch labeled Main Rec-Aux Rec is located at the right side of the group controls. This control provides facilities for selection of the various modes of operation of the auxiliary receiver and the main radio receiver-transmitter. As the control is rotated, different modes of operation are selected for the radio receivers. There are five possible control settings for each receiver. One control position for one receiver corresponds to a control position for the other. The auxiliary receiver functions are selected from the positions at the bottom of the dial. These positions are GRD, CMD, ADF, ADF and STBY.

GRD - When the function switch is positioned so that the auxiliary receiver setting is at the GRD position, the main receiver is in the ADF position. This allows the operator to monitor the guard channel frequency and maintain ADF operation at the same time.

CMD - When the auxiliary receiver is set for CMD position, a command antenna is connected to the auxiliary receiver. The main receiver is positioned in the ADF+G position.

ADF - The auxiliary receiver is placed in ADF operation at either of two positions. At one of these switch settings, the ADF antenna is connected to the auxiliary receiver and the radio receiver-transmitter is set at the T/R+G position for receiving transmitting and monitoring the guard channel frequency.

ADF - When the auxiliary receiver function switch is placed in the other ADF position the main receiver will be positioned at the T/R position. This setting will connect the ADF antenna to the auxiliary receiver and at the same time the main receiver will be set for receiving and transmitting operations.

STBY - When the auxiliary receiver is set for standby the main receiver will also be positioned at standby.

PREFLIGHT CHECKS

Auxiliary Receiver

The auxiliary receiver can be preflighted in the following manner. Position the function switch to the auxiliary receiver CMD position and rotate the auxiliary sensitivity control fully clockwise. A "live" sound should be heard on the headsets under these conditions. Make this check on several frequencies. A further check may be made by receiving from the control tower if deemed necessary.

ADF Loop

The ADF loop should be preflight checked with each of the two receivers utilized in the system, the main radio receiver-transmitter, and the auxiliary receiver. Place the function switch on either Comm-Nav group control panel to the main receiver ADF or the main receiver ADF+G position. Tune the main radio receiver transmitter to the frequency of a station of known geographical location by use of the main channel frequency selector control knobs, and adjust the main volume control knob to obtain a comfortable listening level in the headset. Observe the pilot's course indicator and note that it indicates the approximate direction of arrival of the known signal relative to the airplane heading. Place the function switch in the auxiliary receiver ADF position and tune the auxiliary receiver to a station of known geographical location by use of the auxiliary channel control knob. Note that the needle on the indicator indicates the approximate direction of arrival of the known signal relative to the airplane heading. Adjust the auxiliary volume control to obtain a comfortable listening level. (A 100 cycle buzz should be heard in the headset while the antenna is searching.) A preflight check may also be accomplished by utilizing the transmitting facilities in the control tower if the aircraft is taxied to a remote point of the airstrip.

Note

- When used on the ground, bearing error in the system will likely exceed 30° and therefore no accuracy tolerances are established for this condition.

- Due to antenna location the ADF is unreliable with the landing gear extended.

Due to ADF pattern distortion at the higher frequencies of the UHF band, sizable bearing inaccuracies can be expected at frequencies above 310 mcs. Precise navigational operation should be limited to assigned ADF frequencies (265 to 284.9 MCS) when using the auxiliary receiver, and to frequencies lower than 310 mcs when using the main receiver.

OPERATION PROCEDURE

The operation of the auxiliary and ADF parent loops are controlled completely by the operating controls on the Comm-Nav group control panels. A summary of the operation of these controls is listed in the following material.

Auxiliary Receiver Command Position

1. Place the function switch to the auxiliary receiver CMD position.
2. Select the desired channel by operating the auxiliary channel control knob.
3. The auxiliary receiver is now operating as an auxiliary command receiver, while the main radio receiver transmitter is operating in the ADF loop and also as a guard receiver.

Auxiliary Receiver Guard Position

1. Place the function switch to the auxiliary receiver GRD position.
2. Operation of the auxiliary channel selector control knob does not affect receiver function and may be positioned at any position.
3. The auxiliary receiver is now utilized in an auxiliary capacity and the main radio receiver-transmitter is operating in the ADF loop.

Auxiliary Receiver ADF Position

1. Place the function switch at one of the auxiliary receiver ADF positions.
2. Set the auxiliary channel control knob to the desired channel.
3. The auxiliary receiver is now operating as an ADF receiver. At one auxiliary receiver ADF position the main radio receiver-transmitter is operating on transmit-receiver operation. The guard receiver circuitry is also energized. On the other auxiliary receiver ADF position the main radio receiver-transmitter is on normal transmit-receiver operation.

Auxiliary Receiver Standby Position

1. Place the function switch at the auxiliary receiver STBY position.
2. Both the auxiliary receiver and the main radio receiver-transmitter are now on standby operation. The ADF loop is also inoperative.

ANTENNA SELECTION

A three-position antenna selection switch is located in each cockpit. The pilot's antenna switch is located on the left console outboard of the throttles, the antenna switch for the radar observer is located on the radar observer's instrument panel. The switch positions are upper, lower and security and they are used to select one of two communication antennas to be used with the command communication set. The security position is inoperative, as no security antenna is installed. A cutout metal shield has been placed over the switch, preventing the switch from being placed in the security position. Effective airplanes 146817c and subsequent, the antenna selection switch is a two position toggle switch with positions of upper and lower.

TACAN (TACTICAL AIR NAVIGATION) SYSTEM

The TACAN navigation system functions to give an aircraft precise geographical bearing and distance information at ranges up to 196 miles (depending on aircraft altitude) from an associated ground or shipboard radio beacon. It also determines the identity of the beacon and indicates the dependability of the beacon signal. It also will provide deviation indication from a selected course. The TACAN navigation system employs UHF radio frequencies, the propagation of which is virtually limited to line of sight distances. The maximum distances from the beacon at which reliable TACAN signals can be obtained depends on the altitude of the aircraft and the height of the beacon antenna.

FUNCTION OF CONTROLS

All airborne operating controls for the TACAN system are located on the lower third of the Comm-Nav group control panels (figure 4-9). The controls and their functions are as follows:

Function Switch

The function switch is a three-position rotary switch whose positions are marked STBY, REC, and T/R.

STBY - When the function switch is in the STBY position only filament and blower power is being supplied to the system's receiver and transmitter.

REC - The receiver position places only the receiver portion of the system in operation. When the switch is in this position, only bearing information is furnished to the bearing direction heading indicator.

T/R - When the switch is in the transmit-receive position, the airborne transmitter sends a signal to the beacon and receives a signal which is used to determine the aircraft distance from the beacon.

Navigation Channel Controls

Two control knobs, one to the right and one to the left of the navigation channel window permit channel selection. The left knob selects the tens and hundreds figures of the operating channel. The right knob selects the units figures of the operating channel. The

LINE-OF-SIGHT DISTANCES

LINE-OF-SIGHT DISTANCES-NAUTICAL MILES							
ALTITUDE	MILES	ALTITUDE	MILES	ALTITUDE	MILES	ALTITUDE	MILES
1,000	28	4,000	66	10,000	106	20,000	160
1,500	36	5,000	74	11,000	112	22,000	168
2,000	44	6,000	81	12,000	118	24,000	173
2,500	50	7,000	87	14,000	130	26,000	181
3,000	56	8,000	93	16,000	141	28,000	188
3,500	61	9,000	100	18,000	150	30,000	196

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Figure 4-10

dial system is numbered from 0 to 129, each number from 1 to 126 represents a specific pair (transmitting and receiving) of frequencies. Numbers 0, 127, 128 and 129 on the channel dial are not usable.

Volume Control

The volume control knob is used to adjust the volume of an audio identification signal received from the beacon. The identification signal, audible in the pilot's headphones when they are connected into the intercom system, consists of a two- or three-letter tone signal in International Morse Code. The identification signal is normally transmitted by the beacon every 30 seconds.

Nav CMD

The navigation command push button transfers control of UHF navigation (TACAN) functions from one cockpit to the other. A Comm-Nav group control panel (figure 4-9) is located in each cockpit. When the green light in the center of the navigation command button in one cockpit illuminates, command of the TACAN functions has been obtained in that cockpit. Specifically, the navigation channel control and function switch in that cockpit only are effective. The navigation volume control is effective in both cockpits regardless of the take-command situation.

OPERATIONAL PRECAUTIONS

To obtain maximum usefulness from the TACAN systems transmitter and receiver and the associated controls on the Comm-Nav group control panels, always observe the following operational precautions:

1. Be sure the navigation channel selector is set to the desired operating channel.
2. Be sure the function switch is set properly. When both bearing and distance indications are desired, the control should be set to T/R. When bearing information only is desired, the function switch should be set to REC.

Note

If radio silence is to be observed, the function switch must be set to either STBY or REC. If the control is set to REC, bearing information only may be obtained.

3. Be sure that the navigation volume control is set to provide adequate volume to hear the beacon identification signal.

Note

When a new operating channel is selected, check the identification code immediately to insure that the desired beacon has been obtained.

4. On the basis of figure 4-10 determine that the altitude of the aircraft is such that reliable signals may be expected.

FLIGHT CHECKS OF TACAN SYSTEM

In order to check operation of the TACAN receiver and transmitter while in flight, proceed as follows:

1. Set the function switch to REC. Allow a warm-up period of approximately 90 seconds.
2. Turn the navigation channel selector dial to the channel of a TACAN beacon known to be within operating range. Refer to figure 4-10. When the airplane is directly over a visual checkpoint, note the bearing on the indicator. This bearing should correspond to the bearing of the beacon as determined from an aeronautical chart.
3. Place the function switch in the T/R position. Allow about a 25-second search period. Note the reading of the bearing and distance indicators. Compare these with the bearing and distance to the beacon.
4. Adjust the navigation volume control until the beacon identification signal can be heard in the headphones. This signal should be the established identification letters of the selected radio beacon.

Note

When possible, make checks using other beacons within operating range.

5. Set the navigation channel to an unused beacon channel. After about 10 seconds, the flap on the range indicator should shutoff the distance display, the pointer of the indicator should resume searching, and the audio channel identification signal should cease.
6. Turn off the equipment by placing the function switch in the STBY position.

OPERATION PROCEDURE

Starting and stopping of the TACAN system is controlled by the function switch. When this switch is in the STBY position, only the equipment filament and blower power are on. When the switch is in either REC or T/R position, the equipment is ready to operate.

Channel Selection

Select the desired beacon channel by rotating the navigation channel control knobs on the Comm-Nav group control panel until the appropriate channel number is displayed on the navigation channel window.

Bearing and Distance Indication

(Airplanes 145311b, 145312b and 145314b thru 145317b)

One Bearing-Distance-Heading indicator is located in each cockpit. This indicator presents distance information in digital form, superimposed on the conventional RMI display consisting of a moving compass card to show heading and two bearing pointers. The No. 2 (wide) bearing pointer is the TACAN needle and indicates magnetic heading to the TACAN station. The No. 1 (narrow) pointer, provides magnetic heading to the low frequency. The BDH indicator is also capable of displaying distance information to a beacon station. When a usable signal is not being received, the red warning flag partly obscures the distance indicators from view. The word "OFF" in black letters will appear in the window. The units digit indicator dial is divided into 1/2 mile increments permitting distance information to be read to within 0.1 mile. If desired, input to the No. 2 pointer on the aft BDH indicator only may be switched from TACAN to navigation computer. This is accomplished by operation of a CNI-NAV comp switch on the instrument panel in the aft cockpit. One course indicator, ID-387/ARN is located in the forward cockpit only. Provision is made for course selection, indication of flight path deviation, and to-from beacon indication. A counter-type dial displays the course selected by the course set knob. The to-from flags indicate progress of the airplane with respect to the TACAN beacon. An OFF warning flag indicates when bearing information is unreliable. For bearing and distance indication information on airplanes 145313b, 146817c thru 148275e, refer to Flight Director Group, this section.

Beacon Identification

The beacon on which the aircraft is flying is identified by call letters spelled out in International Morse Code over the headphones. The volume of this tone signal is adjusted by the navigation volume control on the Comm-Nav group control panel.

RADAR ALTIMETER SYSTEM—AN/APN-22

The AN/APN-22 radar set is a microwave altimeter which measures the terrain clearance of the aircraft. 28 volt d-c and 115 volt, single-phase a-c power is required to operate the system. No antennas external to the surface of the aircraft are necessary. A frequency-modulated signal is radiated to the ground and a period of time elapses until a portion of this signal is reflected back to the aircraft. During this time lapse, the transmitter frequency changes, causing a frequency difference between the signal being transmitted and the signal arriving from the ground. This difference in frequency is proportional to time lapse which is proportional to height. The system is designed to provide reliable operation over the ranges of 0 to 10,000 feet over land and 0 to 20,000 feet over water. F4H ASC 96 will be incorporated in all airplanes having undergone the F4H Drive-in Modification program. ASC 96 will improve the reliability of the system, allowing higher altitude operation without drop-out. Accuracy of indication is ± 2 feet from 0 to 40 feet and ± 5 percent of indicated altitude from 40 to 20,000 feet. Accuracy is greatly impaired during steep angles of bank, dive or climb.

Radar Altimeter

The radar altimeter (figure 1-17), located on the instrument panel, provides indication of the airplane's altitude in a single turn type dial calibrated from 0 to 20,000 feet. "Drop-out" occurs when altitude limits of the system are exceeded. This disables the indicator and places the needle behind a mask. An adjustable "bug" pointer at the outside of the calibrated scale of the indicator can be preset to desired altitudes and used as a reference in flying at a fixed altitude. The indicator system provides a red limit light located just below and to the right of the indicator which is illuminated when the aircraft is flying at or below the preset altitude, and off when the aircraft is above the preset altitude.

Radar Altimeter Control Knob

A control knob (figure 1-17) marked "ON-LIMIT" is the only operating control in the radar altimeter system. This knob is located just below and to the left of the radar altimeter on the instrument panel. The radar altimeter control knob operates the system on-off switch and is also used to set the "bug" pointer on the desired preset altitude. The knob utilizes 28 volt d-c electrical power. Refer to Operation of Radar Altimeter, this section.

Low Altitude Warning Light

A red warning light (figure 1-17) located below and to the right of the altimeter, is provided to indicate upon illumination, that the aircraft is below the altitude preset by the radar altimeter control knob ("bug" pointer). The light will also illuminate when "drop-out" occurs. Intensity of the warning light is changed by turning lens housing. Power required is 28 volt d-c.

OPERATION OF RADAR ALTIMETER

With electrical power supplied, the radar altimeter system is set into operation by the initial turn (clockwise) of the radar altimeter control knob. Further clockwise rotation of the radar altimeter control knob positions the limit "bug". The system is turned off by rotating the control knob to its full counterclockwise position.

WARNING

Radar waves can penetrate the surface of snow and ice fields, therefore, when radar altimeter equipment is used for measuring terrain clearance, it may indicate greater terrain clearance than actually exists.

Note

Allow approximately 12 minutes warm-up time after starting the equipment to insure accuracy. If the temperature is below -40°C (-40°F), 25 minutes warm-up time should be allowed.

Drop-Out

The radar altimeter system will stop indicating altitude when the reflected signal is too weak to override the system noise. "Drop-out" should not occur at altitudes below 10,000 feet over land or 20,000 feet over water. F4H ASC 96 will further improve the system. However, a climb, bank or dive of 60° or more will somewhat reduce "drop-out" altitude. When "drop-out" occurs, circuits within the system automatically disconnect the indicator synchro from the servo system and apply a fixed signal to it, which causes the needle to assume a position behind the mask on the indicator dial between the 20,000 point and the 0 point. This fixed off scale position indicates to the pilot that the reading is meaningless.

Note

When "drop-out" occurs, it is necessary to reduce altitude to a point slightly below where "drop-out" occurred, or to level out from the maneuver which caused "drop-out". The return to normal operation following "drop-out" is indicated by the resumption of normal indicator operation.

IFF/SIF INTEGRATED RADAR IDENTIFICATION SYSTEM

Refer to Confidential Supplement, NAVWEPS 01-245FDA-1A.

LIGHTING EQUIPMENT

EXTERIOR LIGHTS MASTER SWITCH

The exterior lights master switch is located on the outboard side of the left engine throttle grip. The switch has three positions, ON, OFF and SIGNAL. Movement of the switch to the ON position will energize the exterior lights panel through which the exterior lights can then be controlled. Placing the switch in the SIGNAL position will cause the exterior lights to flash, providing the wing and tail light switches located on the exterior lights control panel are ON. This can be used as a "ready" signal to the catapult officer. The master switch and exterior lights panel receive power from the right main 28 volt d-c bus.

Exterior Lights and Control Panel

The exterior lights consist of the position lights, join-up lights, fuselage lights, anti-collision lights, angle-of-roll light and approach lights. The exterior lights control panel (figure 4-11) is located near the aft end of the pilot's right console and contains all the manual controls for the exterior lighting.

Position and Join Up Lights

The position lights include the wing tip position lights and the tail light. The join-up light consists of a red

EXTERIOR LIGHTS CONTROL PANEL

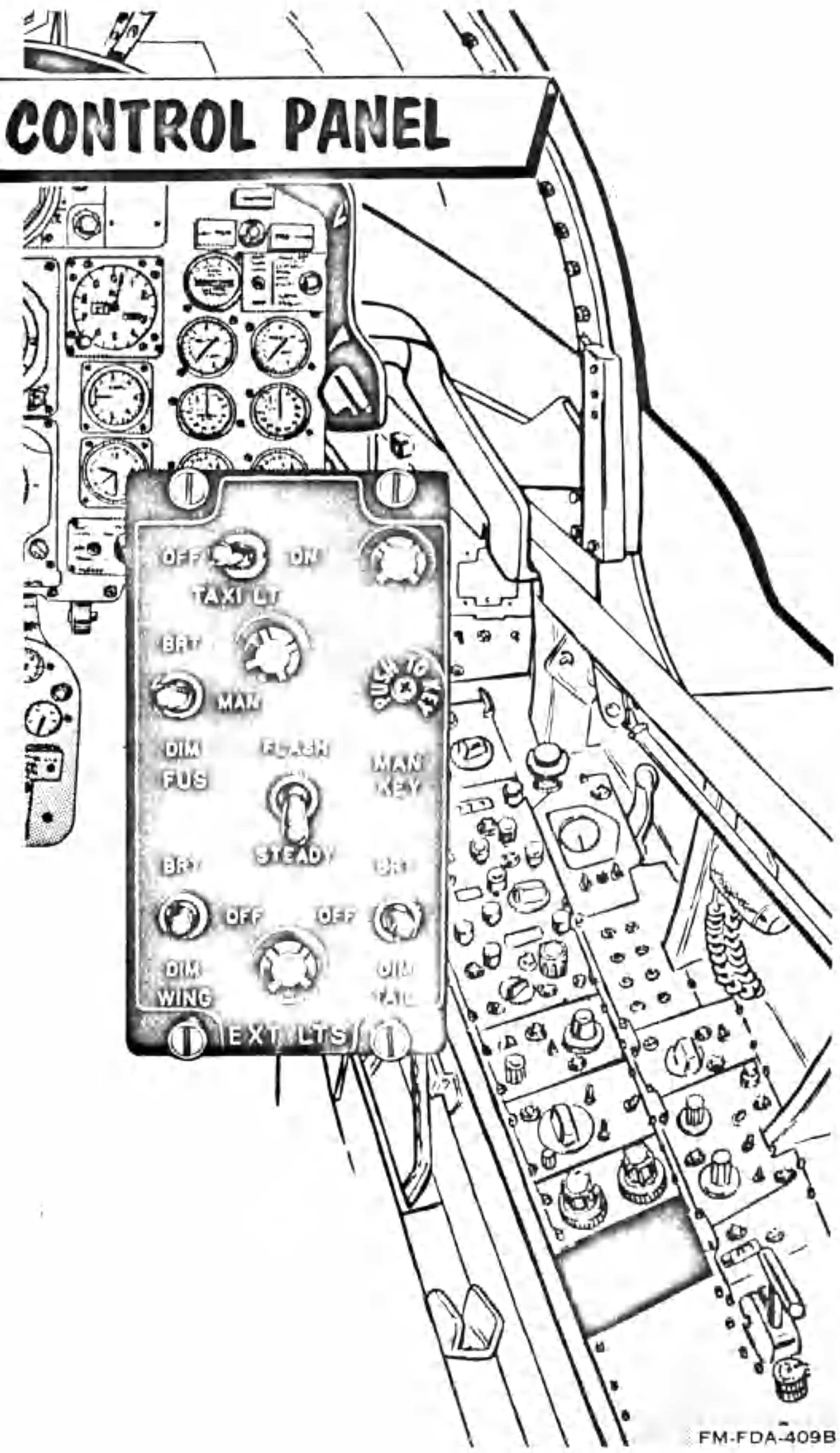

Figure 4-11

or green light on the trailing edge of the applicable wing tip. The wing lights and join-up lights are controlled by the same three position OFF, DIM, BRT wing toggle switch (figure 4-11). The tail light is controlled by a separate three position OFF, DIM, BRT tail toggle switch (figure 4-11). Both switches are located on the exterior lights control panel. The wing and tail switches are two separate switches but both function the same. With the switch in the OFF position the lights are de-energized and are not illuminated. When the switch is put in the BRT position the lights will illuminate at full brilliance and in the case of the wing lights switch the join-up lights will also illuminate. Place the switch in the DIM position and the lights will reduce in brilliance. The dim circuit of the position and join-up lights is powered by the left main 26/14 volt a-c bus. The bright lights circuit is powered by the right main 26 volt a-c bus.

Fuselage and Anti-Collision Lights

Three semi-flush white lights are installed on the fuselage, one above the number two fuselage fuel tank, and one light below each of the engine air inlet ducts. In addition to the fuselage lights, one red anti-collision light is installed in the leading edge of the vertical fin. The fuselage switch (figure 4-11) on the exterior lights control panel controls the three fuselage lights and the anti-collision light. The switch has three positions; DIM, BRT, and MAN. The anti-collision light will be on only when the fuselage switch is in BRT position and the flash steady switch is in the FLASH position. The MAN position of the switch allows the fuselage lights to be keyed by the manual key button. The fuselage lights are powered by the left main 26/14 volt a-c bus. One of the two lamps in the anti-collision light is powered by the R.H. main 26 volt a-c bus, and the other from the L.H. main 26 volt a-c bus.

Flash Steady Switch

The flash steady switch (figure 4-11) located on the exterior lights control panel will operate the flasher relay and flasher unit causing the fuselage lights and tail light to glow steadily or flash. The anti-collision light will operate only when the flash steady switch is in the FLASH position. The flasher unit and relay is powered by the right main 28 volt d-c bus.

Manual Key Button

The manual key button (figure 4-11) located on the exterior lights control panel is used to key the fuselage lights when the fuselage lights switch is in the MAN position and the FLASH STEADY switch is in the STEADY position. An indicator light on the exterior lights panel will glow when the manual key button is depressed. The manual key button receives its power from the left main 26/14 volt a-c bus.

Angle of Roll Light

The low, swept wing design of the airplane prevents the landing signal officer from seeing the starboard position light until the airplane is almost on final approach; a green angle of roll light is installed on the left side of the fuselage just above the trailing edge of the wing. This light is energized by the same circuit that operates the approach lights. This allows the landing signal officer to determine that angle of bank throughout most of the landing pattern.

Hook Bypass Switch

Airplanes 145311b, 145312b, 145315b, and 145317b have the hook bypass switch located in the left wheel well. Airplanes 145313b, 145314b, 145316b, and 146817c thru 148275e have the hook bypass switch located on the exterior lights control panel. Airplanes 145311b, 145312b, 145315b, and 145317b do not have a flasher unit. In these airplanes the approach lights and angle of roll lights will not illuminate unless the landing gear is down and either the hook is down or the hook bypass switch is in the BYPASS position. For operation on airplanes 145313b, 145314b, 145316b, 145317b thru 148275e, see figure 4-12.

INTERIOR LIGHTING

Interior lighting in the airplane is powered by the a-c electrical system, either from the engine-driven generators or by the emergency generator. The instrument panel lights are the only lights that are illuminated by utilizing the emergency generator system.

Pilot's Cockpit Lighting

Most of the lighting controls in the pilot's cockpit are located on the cockpit lights control panel (figure 4-13) which is located on the pilot's right console. The utility light is mounted under the right canopy sill and has its own on-off and intensity control. The instrument panel emergency red floodlight switch (figure 1-19) is mounted above the cockpit lights control panel on the right console.

Instrument Lights

The pilot's instrument panel on some airplanes is lighted by post type light fixtures adajcent to the individual instruments. On other airplanes the instruments are illuminated by integral instruments lights. Variations in instrument lighting intensity on all airplanes is controlled. Variation in instrument lighting intensity is controlled by the instrument panel lights control knob (figure 4-13) located on the forward inboard corner of the cockpit lights control panel. The control knob varies the brilliance of the instrument lights from OFF to BRT. Also as the control is rotated from OFF to BRT, a switch within the control knob will energize the warning lights dimming relay reducing the brilliance of the warning lights. Secondary instrument lighting is provided by red floodlights on the instrument panel glare shield. The control for the floodlights is mounted above the forward portion of the right console and is labeled instrument panel emergency flood. The three-position switch is labeled OFF, DIM and BRT and provides only bright or dim positions. The instrument lights are powered from the essential 115 volt a-c bus. The emergency floods are powered by the essential 26 volt a-c bus in the bright position and by the left main 26/14 volt a-c bus in the dim position.

APPROACH-ANGLE OF ROLL LIGHT INDICATIONS

AIRPLANES 145313b, 145314b, 145316b, 145317b THRU 148275e

ANGLE OF ROLL AND APPROACH LIGHT INDICATION	GEAR-FLAPS-HOOK-BYPASS SWITCH POSITION
FLASHING	GEAR-DOWN FLAPS-DOWN HOOK-UP BY-PASS SWITCH-NORMAL
STEADY	GEAR-DOWN FLAPS-UP OR DOWN HOOK-DOWN (BY-PASS SWITCH-NORMAL) HOOK-UP (BY-PASS SWITCH-BY-PASS)
NO LIGHT	GEAR-UP OR UNLOCKED OR GEAR-DOWN FLAPS-UP HOOK-UP BY-PASS SWITCH-NORMAL

Note

THE ANGLE OF ROLL AND APPROACH LIGHTS ARE AUTOMATICALLY DIMMED WHEN THE EXTERIOR LIGHTS MASTER SWITCH IS ON. WHEN THE AIRPLANE HAS WEIGHT ON THE MAIN GEAR THE ANGLE OF ROLL AND APPROACH LIGHTS ARE OFF.

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Figure 4-12

Console Lights

Console lighting is designed for combination edge and floodlighting. Variation in edge lighting intensity is controlled by the console lights control knob (figure 4-13) located on the aft inboard corner of the cockpit lights control panel. This knob controls all edge lighting on the left and right console, the pedestal panel and the armament switch panel on the pilot's instrument panel. The console control knob varies the brilliance of the console edge lights from OFF to BRT. Also, as the control is rotated from OFF to BRT, a switch within the control knob will energize the console floods switch thus providing console floodlight illumination along with the edgelighting. The console floods switch located above the console control knob selects BRT, DIM or MED brilliance for the red console floodlights. The console floodlights will be off only when the console floods switch is in the DIM position and the console control knob is rotated to the OFF position. The console edge lights receive power from the left main 115 volt a-c bus. The left main 26/14 volt a-c bus delivers power to the dim circuit of the console floodlights. The essential 26/14 volt a-c bus supplies the bright circuit.

COCKPIT LIGHTS CONTROL PANEL

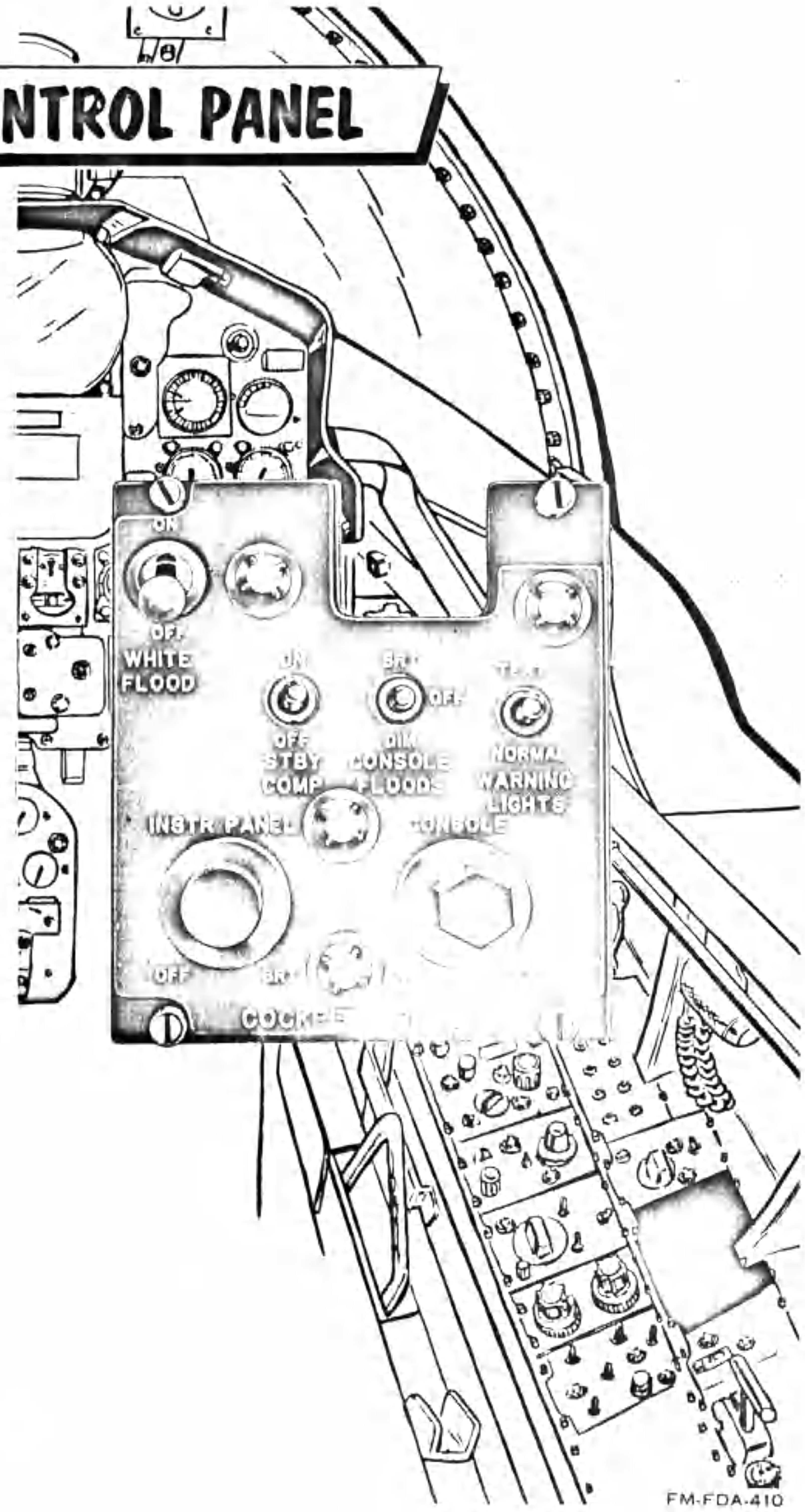

Figure 4-13

White Floods Switch

One white floodlight is provided above each console under the canopy sill. Control is by the white flood switch (figure 4-13) located in the forward outboard corner of the cockpit lights control panel. This switch is of the lever-lock type to prevent inadvertent operation. No intensity variation is provided on these lights.

Standby Compass Switch

The standby compass switch (figure 4-13) located on the cockpit lights control panel is used to turn the standby compass light on and off.

Warning Lights Switch

The warning lights test switch (figure 4-13) is a two-position switch located on the cockpit lights control panel. The switch positions are NORMAL and TEST and the switch is spring-loaded to the NORMAL position. When placed in the TEST position, the warning lights will be illuminated. The warning lights test circuit receives power from the right main 28 volt d-c bus.

Utility Light

A utility light is provided and is mounted under the right canopy sill. The light has its own on-off and intensity control. This light may be changed from red to white by depressing the latch button and rotating the lens housing. An additional plug-in socket for the utility light is provided on the right windshield sill aft of the instrument panel. The utility light is powered from the right main 26 volt a-c bus.

Spare Edge Lamps

Spare edge lamps are located in a spring-loaded cylindrical container on the wing fold control panel on the right console.

RIO'S COCKPIT LIGHTING

The light controls in the radar observer's cockpit are of a simplified nature. All primary lighting both of instrument and control panels is controlled by the three-position "INSTR" switch located on the left side of the instrument panel. The switch positions are OFF, DIM and BRT and provides only two intensities of lighting for the control and instrument panel edge lights. Secondary red floodlighting is provided for the instrument panel and all control panels and is controlled by a three-position cockpit floods switch located inboard of the "INSTR" switch on the instrument panel. The switch positions are OFF, DIM and BRT and they also only give fixed intensities of DIM and BRT to the five red floodlights located throughout the radar observer's cockpit. When the "INSTR" switch is positioned from OFF to DIM or from OFF to BRT, the drogue in, drogue out and fuel flow warning lights on the Buddy Tank control panel will be automatically dimmed. The instrument and cockpit floods switches in the DIM position receive power from left main 26/14 volt a-c bus. The "INSTR" switch in the BRT position is powered off the right main 26 volt a-c bus. The BRT position of the cockpit floods switch receives power from the essential 26 volt a-c bus.

Warning Lights Switch

The warning lights switch is a two-position switch located on the instrument panel inboard of the cockpit floods switch. The switch positions are OFF and TEST and the switch is spring-loaded to the OFF position. When placed in the TEST position the canopy unlock, left wing pin unlock, right wing pin unlock and radar CNI cool off warning lights will illuminate. The warning lights test circuit receives power from the right main 28 volt d-c bus.

Utility Light

The utility light is mounted above and to the left of the radar observer's instrument panel. An additional plug-in socket for the light is provided on the upper structure aft of the instrument panel to provide an alternate location to illuminate the chartboard when it is being used. This light may be changed from red to white by rotating the lens housing. The light has an integral on-off and intensity control. The utility light is powered from the right main 26 volt a-c bus.

Spare Edge Lamps

Spare edge lamps are located in a spring-loaded cylindrical container on the utility panel which is located in the left forward section of the cockpit.

OXYGEN SYSTEM

A liquid oxygen (LOX) system consisting of a ten liter capacity vacuum insulated container, build-up coils, check valves, vent valves and quantity gages is provided on the airplane. The system is designed to manufacture and deliver gaseous oxygen to the crew at a continuous rate of up to 120 liters per minute at 60 to 90 psi. The liquid oxygen is stored in a double-walled vacuum container. From the container the liquid oxygen flows to the build-up coils which are predetermined lengths of tubing wrapped around the outside bottom of the container. When the liquid flows through the build-up coil it absorbs heat from the surrounding area and gasifies. The gasified oxygen is now increased in pressure to assure a 40 to 90 psi pressure to the regulators if they should demand it. From the build-up coils the oxygen now flows into the warm-up plate located in the rear cockpit aft of the ejection seat. The warm-up coils will further gasify the oxygen and warm it up to a temperature no colder than 20°F under the cockpit temperature. From the warm-up plate the oxygen is now ready for crewmember consumption. A system relief valve set at 110 psi vents excessive pressures that may occur due to the boil-off of the LOX when the system is not being used. A blow out patch in the oxygen container provides added safety if a relief valve should fail. An electrical capacitance type indicator provides the pilot and radar observer with an accurate means of determining the amount of LOX remaining at any time. An oxygen low warning light operated by the indicator circuit and located on the telelight panel alerts the pilot when the liquid oxygen supply is reduced to one liter.

OXYGEN SUPPLY LEVERS

A two-position ON-OFF oxygen supply lever is located on a panel in each cockpit. The pilot's supply lever (figure 4-14) is located on the aft end of the left console. The radar observer's supply lever (figure 4-6) is mounted on the utility panel which is located in the left forward section of the cockpit.

OXYGEN QUANTITY GAGE

An oxygen quantity gage is located in each cockpit which reads 10 liters maximum. The pilot's oxygen gage (figure 4-14) is located on the forward end of the left console. The radar observer's oxygen gage (figure 4-6) is mounted on the utility panel which is located in the left forward section of the cockpit.

OXYGEN REGULATOR

The oxygen breathing regulator is personnel mounted and is used both in the inflight and bailout or emergency conditions. The regulator is so designed that with an inlet pressure of 40 to 90 psig it will deliver 100% oxygen automatically to the user, between the altitudes of 0 to 50,000 feet. In addition, the regulator incorporates automatic safety pressure build-up to a maximum of 2 inches of water below 35,000 feet and automatic pressure breathing for altitudes above 35,000 feet and is designed to integrate with the A13'A oxygen breathing mask.

AN/ASA-32 AUTOMATIC FLIGHT CONTROL SYSTEM

The AN/ASA-32 flight control system is an electro-hydraulic autopilot system designed to provide stable, accurate coordinated flight maneuvers without interferring with effective manual control. The flight control system is capable of performing two modes of operation, damper and autopilot. Damper operation improves airplane stability in pitch, roll and yaw; it opposes any changes of attitude but will not return the airplane to a given attitude or heading. This mode of operation may be used while the airplane is under manual control. The autopilot mode of operation will maintain any airplane heading and/or attitude selected within the autopilot limits and will correct for any deviation from the selected heading or attitude of the airplane within the autopilot limits. Two additional operating features are available while in the autopilot mode of operation, they are altitude hold and Mach hold. The altitude hold mode of operation will hold any altitude selected by the pilot while in autopilot operation. The Mach hold mode of operation will hold any indicated Mach number selected while in autopilot operation.

STABILITY AUGMENTATION MODE (DAMPER)

In the Stab. Aug. (damper) mode of operation the system senses motion about the horizontal, vertical and lateral axes, by means of rate gyro sensors. All attitude, heading and bank angle changes will cause

PILOT'S OXYGEN PANELS

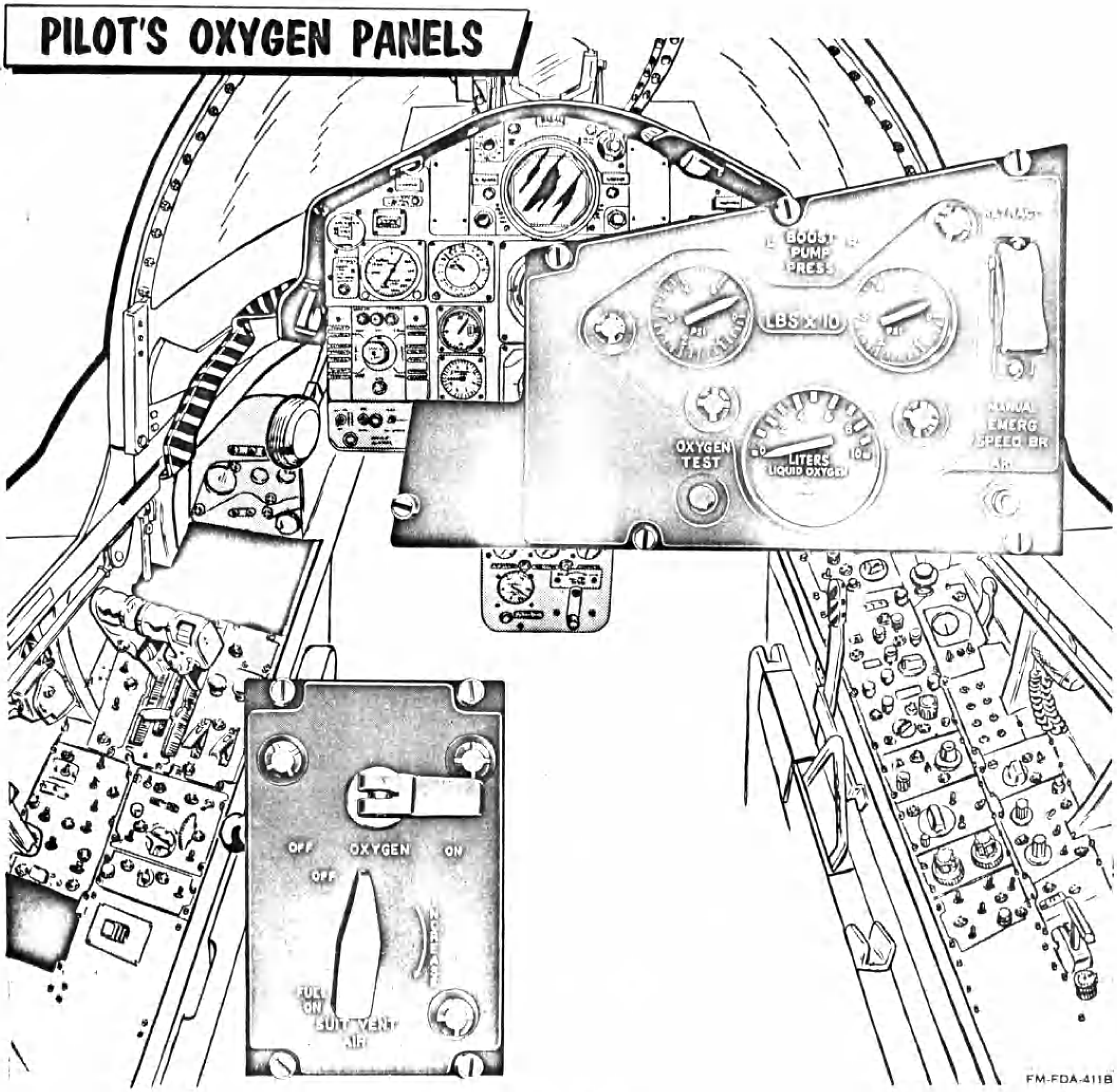

Figure 4-14

these sensing devices to transmit signals representing the changing motion about their respective axes. These signals are sent to servo valves in the control surface actuators, therefore any output signals from the rate gyro sensors indicating yawing or pitching motion causes the flight control system to position the appropriate control surfaces to oppose that motion. This action decreases any tendency of the airplane to oscillate in roll, yaw or pitch. In the damper mode the rate gyro sensors will send signals to the surface controls to oppose any deviations from selected flight attitude but will not return the airplane back to its original heading or attitude. The damper mode is in operation whenever the damper switch is in the DAMPER position.

AUTOPILOT MODE

In the autopilot mode of operation, vertical gyro, directional gyro and accelerometer signals are used in addition to the rate gyro sensor signals to maintain the airplane in a desired attitude with maximum pitch, roll and yaw stability. The autopilot system can be engaged and hold maneuvers and attitudes within a range of ± 70° in pitch, ±70° in bank and 360° in azimuth, providing the G limits are not being exceeded. The autopilot utilizes various components to make the system function properly. These components are a control amplifier, the AN/AJB-3 bombing system, the central air data computer, force transducer, accelerometers, lateral series servos and rate gyro sensors.

OXYGEN DURATION CHART

OXYGEN DURATION-HOURS

Cabin Pressure Altitude Feet	Gage Quantity-Liters						
	10	8	6	4	2	1	Below 1
40,000 and Above	60.6	48.5	36.4	24.2	12.0	6.0	EMERGENCY DESCENT TO ALTITUDE NOT REQUIRING OXYGEN
35,000	37.0	29.6	22.2	14.8	7.4	3.6	
30,000	27.2	21.8	16.4	10.8	5.4	2.8	
25,000	20.4	16.4	12.4	8.2	4.0	2.0	
20,000	16.0	12.8	9.6	6.4	3.2	1.6	
15,000	12.8	10.2	7.6	5.2	2.6	1.2	
10,000	10.0	8.0	6.0	4.0	2.0	1.0	
5,000	8.4	6.6	5.0	3.2	1.6	0.8	
SEA LEVEL	7.0	5.6	4.2	2.8	1.4	0.6	

- DURATION TIME IS HALVED WHEN TWO CREW MEMBERS ARE USING OXYGEN

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Figure 4-15

Control Amplifier

The control amplifier comprises the control center for the entire flight control system. It receives the signals from the various sensing elements in the system and supplies power to the flight control components.

AN/AJB-3 Bombing System

The AN/AJB-3 bombing system provides the vertical and directional references for the autopilot.

Central Air Data Computer

The central air data computer performs three functions for the autopilot. First, it provides all required gain changes, this is necessary to maintain constant maneuvering rates regardless of changes in airspeed and altitude. Second, the CADC contains a clutched synchro which supplies the autopilot with a signal proportional to the deviation from the barometric altitude which existed when the altitude switch was placed in the ALTITUDE position. This signal is used by the autopilot to move the stabilator as necessary to maintain constant barometric altitude. The third function of the CADC is to provide clutched synchro output proportional to the deviation from the Mach number that existed at the time that the Mach switch was placed in the MACH position. This signal is used by the autopilot to move the stabilator as necessary to decrease or increase the airplane's airspeed in order to hold a constant Mach number.

Force Transducer

The force transducer is a unit which senses the force applied by the pilot to the control stick. This unit actually comprises the visible portion of the control stick with the stick grip mounted on top of it. The force transducer contains pressure sensitive switches which react to longitudinal and lateral stick forces. A lateral stick force of three pounds or more closes a switch that will, in effect, disengage the autopilot in roll. The autopilot will stay disengaged as long as the three pound force remains on the stick. When the stick forces are released the autopilot will be re-engaged and hold the new roll attitude. Fore and aft stick forces close a switch to operate certain autopilot components and also cause a force sensing device to send a signal proportional to the applied stick force to the servo amplifier and stabilator actuator thus causing the desired pitch attitude change. During this operation the pilot controls stabilator position through the autopilot. During this operation the aircraft is being manually controlled by the pilot. When the stick forces are released the autopilot will hold the new pitch altitude. In the event the pitch or roll limits of the autopilot are exceeded (±70°), the autopilot in effect will be disengaged, although the autopilot switch will remain ON. Therefore, when the airplane returns to within the limits of the autopilot, the autopilot will once again take over.

AUTOMATIC FLIGHT CONTROL SYSTEM ENGAGING CONTROLLER

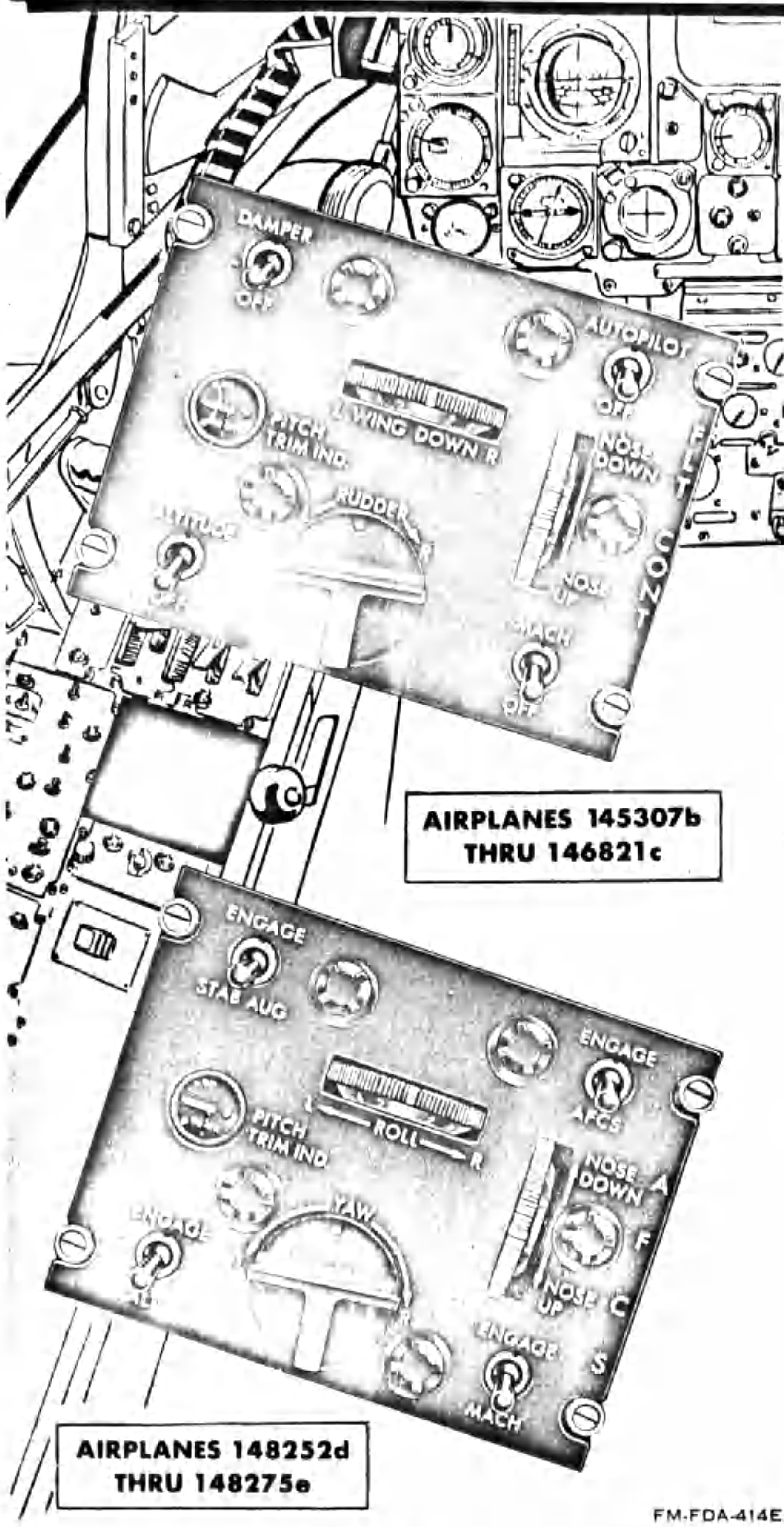


Figure 4-16

Accelerometers

During autopilot operation three accelerometers are being utilized to insure proper functioning of the autopilot system. Two of the accelerometers are of the G limiting type which prevent excess G loads from occurring as a result of autopilot operation. The other accelerometer is a lateral accelerometer which is used to perform coordinated maneuvers while in autopilot operation.

G-LIMIT ACCELEROMETER. The normal load factor interlock (G - disengage) feature the AFCS is designed to inhibit that system from commanding excessive load factor on the airplane. The system will revert automatically from whatever mode is engaged to STAB AUG in the event that plus 4 or minus 1 G absolute is sensed by the G - disengage accelerometer switch. This switch is mounted forward on the radar bulkhead so that if the airplane is rotated rapidly into a maneuver, disengagement will occur at lower values of normal load factor due to the "Anticipation" resulting from the forward location sensing a component of pitching acceleration. If, in addition to the G switch being operated, the STAB AUG servo is hardover in a direction that would tend to increase the magnitude of the existing load factor then the STAB AUG mode will also disengage. It is possible to check out the proper functioning of this feature in flight by engaging STAB AUG and AFCS modes and gradually increasing load factor to approximately plus 4 or minus 1. AFCS modes should disengage, STAB AUG switch should not. The G disengage feature is inoperative outside the ± 70° limits of the autopilot.

TRIM-CUTOUT ACCELEROMETER. This unit cuts off power to the automatic pitch trim circuit when acceleration forces along the vertical axis of the airplane reach or exceed the limits of -0G or +3G's. This circuit prevents the airplane from being trimmed automatically into a high "G" maneuver.

LATERAL ACCELEROMETER. This accelerometer detects airplane skids or slips and produces error signals proportional to the lateral forces developed. These error signals cause the autopilot to take corrective action with the rudder to coordinate the maneuver being performed.

Lateral Series Servos

The lateral series servos function to operate the spoilers and ailerons during autopilot operation.

Rate Gyro Sensors

Refer to Damper Mode, previously discussed in this section.

AUTOMATIC FLIGHT CONTROL SYSTEM CONTROLS

The automatic flight control system engaging controller (hereafter referred to as the AFCS panel) (figure 4-16) is located on the pilot's left console. This panel contains all the controls for the normal operation of the flight control system.

Stab-Aug Switch (Damper Switch)

On airplanes 145307b thru 146821c the damper switch is a two-position toggle switch located on the AFCS panel. The switch positions are OFF and DAMPER. On airplanes 148252d thru 148275e (figure 4-16), the damper switch has been changed to a stab aug switch, and the DAMPER position on the switch has been changed to ENGAGE. Placing the switch in the DAMPER or ENGAGE position will establish the damper mode of operation.

Autopilot Switch

On airplanes 145307b thru 146821c the autopilot switch is a two-position toggle switch located on the AFCS panel. The switch positions are OFF and AUTOPILOT. On airplanes 148252d thru 148275e, (figure 4-16) the autopilot switch has been changed to an AFCS switch, and the AUTOPILOT position on the switch has been changed to ENGAGE. This switch can become energized only if the damper switch is energized. Placing the switch in the AUTOPILOT or ENGAGE position will energize the autopilot system and the system will become fully effective, holding any attitude and heading selected.

Altitude Hold Switch

The altitude switch is a two-position toggle switch located on the AFCS panel. The switch positions are OFF and ALTITUDE. On airplanes 148252d thru 148275e, (figure 4-16) the ALTITUDE position has been changed to ENGAGE. This switch will function only if the autopilot is engaged and the Mach hold switch is disengaged. Placing the switch in the ALTITUDE or ENGAGE position energizes an altitude sensor in the CADC which is controlled by barometer altitude. As the airplane's altitude varies, an error signal is produced and fed to the pitch servo amplifier. The amplifier will then send a signal to the stabilator actuator which will deflect the stabilator as necessary to return the airplane to its "hold" altitude. The altitude sensor will hold the airplane within an altitude ± 50 feet or ± 0.3 percent of the reference altitude at speed up to .9 Mach, and at speeds greater than 1.0 Mach. Altimeter fluctuations while accelerating through the transonic range (.9 to 1.0 Mach) will produce transient fluctuations which, although not violent, may cause the reference altitude to slip. When altimeter fluctuations have ceased, normal altitude hold operation should again be apparent. Engaging the altitude hold mode in climbs of greater than 1,000 feet per minute may result in a reference attitude other than the engage altitude. Opening the altitude switch, or moving the control stick fore or aft will disconnect the altitude hold operation.

Mach Hold Switch

On airplanes 145307b thru 146821c the Mach hold switch is a two-position toggle switch located on the AFCS panel. The switch positions are OFF and MACH. On airplanes 148252d thru 148275e (figure 4-16), the MACH position has been changed to ENGAGE. The Mach switch will cause the autopilot to maintain a constant Mach number within ± .03 mach when it is closed. This switch will function only if the autopilot mode is engaged and the altitude hold switch is disengaged. Opening the Mach switch, or moving the control stick fore or aft will disconnect the Mach hold operation. Placing the switch in the MACH or ENGAGE position energizes a Mach sensor in the CADC. As the airplane's Mach number varies, an error signal is produced and fed to the pitch servo amplifier. The amplifier will then send a signal to the stabilator actuator which will deflect the stabilator down to put the airplane in a shallow dive to pick up Mach number or it will deflect the stabilator up to put the airplane in a gradual climb to decrease Mach number. In Mach hold autopilot operation, the airplane will hold a selected heading, roll and yaw attitude but its pitch attitude (altitude) may vary. Due to the fact that the stabilator is the primary control factor in both altitude and Mach hold it can readily be seen why the two modes of operation cannot be used together.

Pitch Trim Wheel

The pitch trim wheel located in the right center of the AFCS panel is inoperative.

Wings Level Trim Wheel

The wings level trim wheel located in the upper center of the AFCS panel is inoperative.

Heading Trim Knob

The heading trim knob located in the center of the AFCS panel is inoperative.

Pitch Trim Indicator

The pitch trim indicator (figure 4-16) is located in the left corner of the AFCS panel. It indicates the direction in which the automatic pitch trim circuits are trimming the airplane control system.

AFCS/ARI Emergency Disengage Switch

The spring-loaded disengage switch (figure 1-14) is located on the control stick. This switch when depressed will automatically cause all the switches on the flight control panel to return to OFF, and will also disengage the aileron rudder interconnected system as long as the switch is held depressed. When the switch is released the ARI system will again be in operation, but the autopilot will no longer be engaged. To permanently disengage the ARI system, the ARI circuit breaker on the left utility panel must be pulled.

OPERATIONAL PRECAUTION

Roll Reversal

There is a possibility of a condition called "roll reversal" occurring when operating the automatic flight control system in the autopilot mode. This condition will occur infrequently and is apparent only when attempting small changes in bank angle. Roll reversal is associated with a small out-of-trim condition in the lateral channel, and is apparent to the pilot as a slow rolling of the aircraft in the opposite direction the low lateral stick force. If, for instance, the airplane is out of trim laterally to the left when the autopilot mode is engaged, roll reversal may occur when low right lateral stick forces are applied. A roll reversal situation may also be caused by operating the manual lateral trim button while in the autopilot mode, followed by low lateral stick forces being applied opposite to the direction of the trim, There is also a possibility of roll reversal occurring even if the airplane has been trimmed prior to engaging the autopilot mode, and the manual trim button had not been touched. This condition is brought about by changes in aircraft trim accompanying changed flight conditions. In view of the above, the following instructions should be observed:

1. Trim-up airplane in damper mode before engaging autopilot mode.
2. Do not operate manual trim button while autopilot mode is engaged unless roll reversal is encountered, at which time, judiciously use a small amount of manual trim to counteract the encountered roll reversal.

Pitch Oscillations

On occasion the aircraft may experience pitch oscillations in the transonic region and below, using the Altitude Hold mode. These oscillations may vary from "stick pumping" to divergent pitch oscillations. In the event this phenomenon is encountered, use the following procedures:

1. AFCS - DISENGAGE
2. Static Pressure Compensator - OFF
3. AFCS - ENGAGE

If oscillations persist after the above action:

4. Altitude hold - OFF

WARNING

Divergent pitch oscillations should not be allowed to develop. If any divergent pitch activity is noted, corrective action should be taken immediately.

Autopilot Operation With Static Pressure Compensator Inoperative

Effective airplanes 145311b thru 145317b, a Malfunction of the Static Pressure Compensator (indicated by the illumination of the STATIC CORR OFF light) will automatically disconnect the autopilot system including the DAMPER mode. Effective airplanes 146817c thru 148275e, a malfunction of the Static Pressure Compensator will not disconnect the autopilot. The autopilot will operate satisfactorily with the Static Pressure compensator out but the MACH or ALT hold modes should be disengaged to avoid erratic operation.

Compass Mode Engagement

It is possible while operating in the autopilot mode to set up a condition which will cause uncontrollable aircraft oscillations that will roll the airplane in either direction. This condition is brought about by selecting the COMPASS mode of the AN/AJB-3A system while the autopilot mode of the AFCS is engaged. The compass mode of the AN/AJB-3A is an emergency mode, for use only when the AN/AJB-3A displacement gyro has failed. If the compass mode is selected when the autopilot mode is engaged, the autopilot is no longer getting signals from the displacement gyro, but is picking up undamped transient signals from the compass transmitter. These undamped transient signals are sent through the autopilot circuits to the actuators, causing erratic displacement of the ailerons and spoilers resulting in uncontrollable aircraft oscillations. In view of the above, the following instructions should be observed:

WARNING

To prevent uncontrollable oscillations of the airplane DO NOT engage autopilot mode of the AFCS or have autopilot mode engaged when the compass mode of the AN/AJB-3A system is selected.

NORMAL OPERATION

Note

Effective airplanes 145311b thru 145317b, the "Static Corr Off" warning light must be extinguished prior to engaging the damper mode of the automatic flight control system. If the static pressure compensator has malfunctioned ("Static Corr Off" light illuminated), the automatic flight control system will be inoperative.

1. Engage damper mode, place damper (stab. aug.) switch to DAMPER or ENGAGE.

Note

The autopilot mode of operation cannot be engaged until the damper mode is engaged.

2. Trim-up airplane in damper mode before engaging autopilot mode.
3. Engage autopilot, establish an airplane attitude within the autopilot's limits. Place autopilot switch to AUTOPILOT or ENGAGE.

WARNING

- To prevent uncontrollable oscillations of the airplane DO NOT engage autopilot mode of the AFCS or have autopilot mode engaged when the compass mode of the AN/AJB-3A system is selected.
- Do not depress bomb button with bomb control switch energized, autopilot engaged and airplane in a bank attitude. The airplane will roll in the direction of the bank. This is caused by an erratic signal being sent to the autopilot from the gyro. The gyros outer gimbals are caged when the bomb control switch is energized and the bomb button is depressed causing the erratic signal to the autopilot.

Note

Do not operate manual trim button while autopilot mode is engaged unless roll reversal is encountered, at which time, judiciously use a small amount of manual trim to counteract encountered roll reversal.

Note

- The altitude and Mach hold are effective only in the autopilot mode of operation.
- The pitch rate gyro, as utilized by the AFCS modes, inadvertently senses structural vibrations of the airframe and tends to amplify these vibrations by generating commands through the autopilot to the stabilator. These vibrations can become quite pronounced while flying at very low indicated airspeeds. There should be no cause for alarm if this phenomenon occurs and the vibration, or "chatter", can be eliminated by reverting to STAB AUG mode or by increasing airspeed above approximately 190 to 200 knots IAS.

4. Engage altitude hold, place altitude switch to ALTITUDE or ENGAGE.

Note

To change altitude when operating in altitude hold, use control stick. This will disengage the altitude hold circuits and the altitude hold switch will move to OFF. Re-engage altitude hold switch at new altitude if altitude hold is desired.

5. Engage Mach hold, place Mach switch to MACH or ENGAGE. If altitude hold is operating when Mach hold operation is desired, disengage altitude hold and then place Mach hold switch to MACH or ENGAGE.

Note

To change Mach number when operating in the Mach hold mode, use the control stick. This will disengage the Mach hold circuits and the Mach hold switch will move to OFF. Re-engage the Mach switch at new Mach number if Mach hold is desired.

6. Disengage autopilot, place autopilot switch to OFF. Airplane will not be operating in damper mode.
7. Disengage damper, place damper (stab. aug.) switch to OFF.

Note

To disengage both the damper and autopilot, the emergency disconnect lever on the control stick can be used.

GENERATOR SWITCHING

Power to the autopilot, CADC, and AN/AJB-3 may be momentarily interrupted during the starting and stopping of airplane engines or generators. When the right engine or generator is started with the left generator already on the line, the connection between the right and left main busses is momentarily opened to allow the right generator to come on the line. This momentary interruption will allow the solenoid held switches or the engaging controller to disengage. This will necessitate re-engaging the autopilot to bring back autopilot operation. The autopilot, CADC, and AN/AJB-3 will not be effected by starting or stopping the left engine or generator with the right generator on the line.

AUTOPILOT PITCH TRIM WARNING LIGHT

An "A/P Pitch Trim" warning light (figure 1-10) is located on the warning lights panel. This light when illuminated for twenty seconds or more will inform the pilot that the autopilot pitch trim circuit is not functioning. A momentary illumination of the light will be of no concern since a time delay relay in the circuit causes this condition and it does not indicate a malfunction. If the airplane is in autopilot operation when the light is illuminated, the pilot should make sure the airplane is in a neutral trim and that the control stick is gripped firmly before disengaging the autopilot. If this precaution is not observed the airplane may perform a severe nose up or nose down maneuver upon disengagement of the autopilot.

AILERON RUDDER INTERCONNECT SYSTEM

The aileron rudder interconnect (ARI) system causes rudder displacement proportional to aileron displacement which provides coordinated turns at low airspeeds. The limits of the system are 15° of rudder displacement when automatic flight control system is in the damper or autopilot mode, and 10° rudder displacement when the damper switch is OFF. The airspeed pressure switch automatically operates to activate the system at airspeeds below 225 knots. Components of the ARI system include the ARI control amplifier, the 10° servo actuator, the airspeed pressure switch and the aileron transducer. The yaw servo amplifier and the force transducer of the AFCS are also used. The rudder integrated actuator is utilized for part of the rudder movement.

Operation

When airspeed drops below 225 knots, the airspeed pressure switch closes, applying 28 volts d-c to the engage relay solenoids. The contacts of the engage relays close to energize the 10° servo actuator shut off valve. This allows the hydraulic 10° servo actuator to move the control linkage (if aileron displacement is present) and actuate the integrated rudder actuator for rudder displacement. As previously stated, the ARI system is automatically engaged below 225 knots. The system can be disengaged by depressing the AFCS/ARI emergency disengage switch; this will disengage the ARI only as long as it is held depressed. To permanently disengage the ARI system, the circuit breaker on the left utility panel must be pulled and the damper switch disengaged. If the circuit breaker is pulled and the damper is engaged 5° of ARI authority will be provided.

COMPASS SYSTEM CONTROLLER

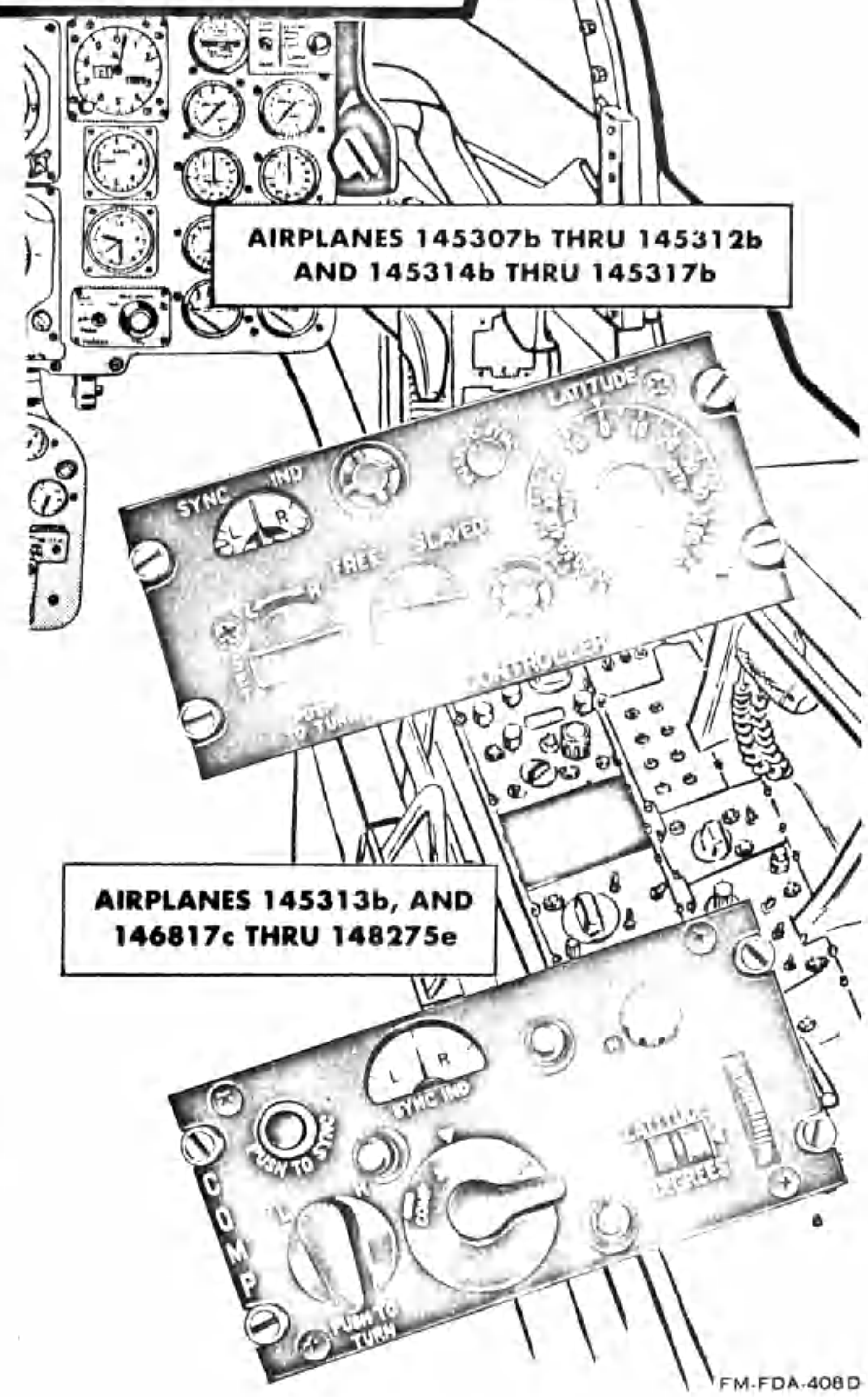


Figure 4-17

AN/AJB-3 ALL ATTITUDE INDICATING SYSTEM

The AN/AJB-3 All Attitude Indicating system consists of the following components: (1) a three-axis all-attitude indicator on the pilot's instrument panel, (2) a G-tube and programmer which attaches to the front of the attitude indicator, (3) bomb control switches on the pilot's instrument panel, (4) an interval timer on the right-hand side of the radar operator's cockpit, (5) an amplifier-power supply unit, (6) a two gyro reference unit, (7) a switching rate gyro, (8) a distribution box, (9) a bomb release angle computer, (10) a compass adapter unit, (11) a compass controller unit on the pilot's right-hand console and (12) a magnetic flux valve. The attitude indicator has three axes of freedom; pitch, roll and yaw. The sphere is two-tone gray. The light upper half represents sky and the dark lower half represents earth. A roll dial between the indicator case and the sphere rotates with the sphere to display roll attitudes. On airplanes 145307b thru 145312b and 145314b thru 145317b, minor roll indices on the indicator case are mechanically obliterated at pitch angles of 75° or greater. A window on the face of the instrument indicates power on or off and phase reversal. The OFF letters are covered by a flag when power is on and the phasing is correct. The single control on the indicator is a pitch trim knob which may be used to give normal indications for various angles of attack caused by different aircraft fuel or armament loading and speed. The limits of this adjustment are 6° nose-down to 20° nose-up.

COMPASS CONTROLLER

A compass system controller (figure 4-17) located on the pilot's right hand console provides the necessary controls and indication to operate the gyro-magnetic compass system. This unit is used to select the mode of directional gyro control, indicate compass vs. heading data synchronization, control manual fast synchronization, and to set up latitude compensation slaving of the directional gyro. The controls include the following:

1. A two-position rotary mode switch labeled FREE, SLAVED.
2. A latitude compensator knob and dial calibrated 90°N - 0° - 90°S in 5° increments.
3. A push-to-turn heading set knob which controls the speed and direction of rotation of the output data shaft in the compass adapter. The speed may be controlled from 1/2 to 10 rpm.
4. A push-to-sync button which is used to synchronize the output data shaft with the magnetic heading.

When operating in the free mode, the directional gyro is free from external control except for latitude compensation. In the slaved mode, the directional gyro is slaved to the magnetic compass heading at a rate of from 1° to 2° per minute. Automatic fast synchronization between the angle data shaft and the magnetic compass occurs when switching from FREE to SLAVED or at initial turn-on when in slaved mode. Effective airplanes 145313b, 146817c thru 148275e, a slightly different compass system controller has been incorporated on the aircraft. A three-position rotary mode switch labeled FREE, SLAVED and COMPASS is utilized. The free and slaved modes operate the same as previously described. In the compass mode, the signals from the magnetic compass bypass the gyro, and are sent on to other systems to be used directly for azimuth information. Aside from the compass position on the mode switch, a separate hemisphere knob and separate latitude select knob and latitude readout window are incorporated on the panel. The hemisphere knob is used to select either a north (N) or south (S) hemisphere. Selecting the correct hemisphere keeps the gyro parallel to the earths surface, thereby, minimizing procession errors. The latitude select knob is used to select any latitude between 0° and 90° viewed through the readout window.

CAUTION

In the compass mode, the attitude director indicator (ADI) is still coupled to the displacement gyro and is not receiving magnetic compass correction. Therefore, the azimuth reading of the ADI may be in error and the HSI should be used as the heading reference.

NAVIGATION EQUIPMENT

FLIGHT DIRECTOR GROUP

(Airplanes 145313b, 146817c thru 148275e)

The purpose of the flight director group is to provide an integrated display of the navigation situation of the airplane. The flight director group consists of a flight director computer, the horizontal situation indicator (HSI), and a selector panel. Although the attitude director indicator (ADI) is not a component of the flight director group, it does receive some signals from the flight director computer and shall be discussed along with the flight director group.

FLIGHT DIRECTOR COMPUTER

(Airplanes 145313b, 146817c thru 148275e)

The flight director computer is used in the instrumentation system, and provides navigation information to the HSI, and steering information to the ADI. Except for the bearing and distance display on the HSI, all signals for the HSI, and signals for portions of the ADI pass through or originate in the computer. The flight director computer has no control over the 3 axis sphere portion of the ADI. Steering signals are computed to provide the pilot with flight direction information when flying either manually or remotely set headings and manually selected TACAN radials. These computed signals, together with the required flag signals and off scale signals, are supplied by the computer to the ADI. The steering signals are limited to insure safe operation without affecting the inherent performance capabilities of the airplane. The computer also includes two servo computing mechanisms. The first adds compass heading to the ground track signal from the Nav Computer such that it can be displayed on the course pointer of the HSI. The second mechanism, in conjunction with the first, subtracts ground track from bearing to destination and adds magnetic heading to the difference in order that the command heading marker on the HSI can be positioned. Heading error is then obtained from the control transformer in the HSI and used together with bank angle to compute a steering signal for display on the vertical director pointer of the ADI.

HORIZONTAL SITUATION INDICATOR

(Airplanes 145313b, 146817c thru 148275e)

The HSI (figure 4-19) provides the horizontal or plan view of the aircraft with respect to the navigation situation. It provides an integrated display of navigation data from various sources and presents this data to the pilot in a symbolic-pictorial display for quick and easy assimilation. The central portion of the display contains a compass card which displays aircraft heading when read against the lubber line and against which a command heading marker, a bearing pointer, and a course pointer may be read for numerical values of magnetic bearing, course and heading. The bearing pointer provides pictorial bearing information to the radio facility, destination, target, or computed waypoint. The course pointer indicates the selected TACAN: or computed course, while the course bar (center segment of the pointer) indicates the aircraft's deviation from this course, as shown pictorially with respect to the stationary miniature airplane symbol at the center of the display. A "to-from" pointer shows the direction along the selected course leading to the radio facility or computed waypoint. A heading marker just outside the compass card indicates the selected heading and, by its angular displacement from the lubber line, the heading error angle. Both the course pointer and the heading marker may be set manually by means of the course set and heading set knobs, or remotely by means of the indicator's course command and heading command servos. The selected course is also shown by the digital counter at the lower right of the display, while a digital counter at the lower left indicates the distance in nautical miles to the radio facility or computed waypoint. Four mode-of-operation word messages are shown around the display. These are illuminated internally to indicate the selected operating modes.

ATTITUDE DIRECTOR INDICATOR

(Airplanes 145313b, 146817c thru 148275e)

The attitude director indicator (figure 4-18), as used with the flight director group, provides attitude and heading information in all three axes simultaneously with flight director indication. The vertical pointer is the only pointer used in conjunction with the flight director group, and is used in the following manner. When the HDG, TACAN or NAV COMP mode is selected, a steering signal for the selected heading, TACAN radial or destination, is sent from the flight director computer to the ADI. The vertical director pointer on the ADI is deflected and the pilot merely banks the airplane to center the vertical director pointer. This enables the pilot to make an asymptotic approach to the selected heading, TACAN radial or destination. The vertical director pointer does not indicate direction or displacement from the desired heading, TACAN radial or destination, but rather the corrective action required to intercept the heading, TACAN radial or destination.

Note

In the event the ADI is inoperative the artificial horizon on the radar scope display is suitable for use as a standby attitude indicator.

MM-3 ALL-ATTITUDE INDICATOR

The F4H Drive-In Modification Program will install the MM-3 all-attitude indicator as a back-up indication to the pilot. This instrument (figure 1-17), is controlled by a remote gyro which establishes the vertical reference line from which pitch and roll deviation is measured. Changes in airplane attitude are electrically relayed from the control unit to the indicator causing displacement of the indicator sphere in relation to the fixed miniature airplane. The amount of displacement is directly proportional to actual airplane attitude deviation from level flight. The indicator is unlimited in roll but will indicate pitch only up to 80 degrees in dive or climb; beyond 80 degrees, a 180 degree controlled precession "flips" the sphere 180 degrees in roll. Horizontal marking with 5 degrees of separation on the face of the sphere show accurate airplane attitudes during climb or dive. This system is automatically put into operation after 115 volt a-c and 28 volt d-c power is supplied. With power applied after approximately 2-1/2 minutes, the "Off" flag in the upper portion of the instrument will retract. This automatic operation eliminates manual caging. Complete failure of either d-c or a-c power causes the "Off" flag to appear.

WARNING

- The "Off" flag does not indicate instrument accuracy but only indicates that the indicator is not receiving power. The absence of the flag does not necessarily mean that the attitude indicator is displaying correct information since the flag does not indicate mechanical malfunctions. Therefore, periodically in flight, attitude indications given by the MM-3 should be checked against other flight instruments, such as the directional indicator, turn-and-slip and vertical velocity indicators.
- A slight amount of pitch error in the indication will result from accelerations or decelerations. It will appear as a slight climb indication after a forward acceleration and as a slight dive indication after deceleration. This error will be most noticeable at the time the airplane breaks ground during the take-off run. At this time, a climb indication error of about 1/2 bar width will normally be noticed, however, the exact amount of error will depend upon the acceleration and elapsed time of each take-off run. The erection system will automatically remove the error after the acceleration ceases.

A pitch trim knob is provided on the indicator for the pilot to center the horizon bar in relation to the fixed miniature airplane.

MODE-BEARING/DISTANCE SELECTOR PANEL

(Airplanes 145313b and 146817c thru 148275e)

The mode-bearing/selector panel (figure 4-21) is located on the pilot's main instrument panel. The panel contains a mode selector switch and a bearing distance selector switch. Refer to figure 4-19 for pointers that are used on the HSI and ADI in conjunction with the mode and bearing distance selector switch positions.

Mode Selector Switch

The mode selector switch (figure 4-21) is a rotary type switch with switch positions of ATT, HDG, TACAN, and NAV COMP; The switch is used to select the source of information to be displayed on the HSI and ADI. The function of each switch position is described below. The NAV COMP position is inoperative at this time.

ATT - Selection of the ATT position essentially eliminates navigational assistance from all indicators but the bearing pointer and distance window on the HSI. All pointers on the ADI are deflected out of view as is the to-from pointer on the HSI. The heading marker and course pointer on the HSI will be slaved to the lubber line, and the course bar will be centered. Except for the function selected on the BRG/DIST switch all that is left is attitude display and mag. heading indication.

HDG - The HDG position activates circuitry to assist the pilot in steering "asymptotically" to a new heading and holding that heading. The desired heading should be selected by moving the heading marker with the HDG. SET knob on the HSI. The vertical director pointer of the ADI will then deflect to indicate in which direction the airplane should be banked. When the proper bank is established the pointer will center and remain centered until bank angle should be decreased, at which time it will so indicate. Thus, when the aircraft has been turned to the desired heading the director pointer will call for zero bank. When HDG position is active, the word MAN will be illuminated on the HSI face.

TACAN - The TACAN position provides asymptotic steering to a preselected TACAN radial provided both heading marker and course pointer are set at the selected radial on the HSI. Under these conditions the vertical director pointer on the ADI will give a steering signal such that the pointer is centered when the proper bank is established to intercept the radial (similar to steering under HDG above). Simultaneously, the course bar of the HSI will deflect to indicate the position of the selected radial relative to the miniature airplane on the face of the instrument. This bar will deflect at the rate of one dot per 5° of the selected radial with maximum deflection being 2 dots. Thus, the bar will provide an estimate of aircraft displacement from selected radial up to 10°. If the course pointer is set to the desired radial, the course should appear in the course readout window. Also, in TACAN position, a to-from arrow will be present and will indicate whether the aircraft will be flying to or from the station on the radial selected. The mode word TAC will be illuminated on the HSI face when TACAN is selected.

WARNING

When operating in the TACAN mode, the course pointer and command heading marker on the HSI must be aligned or an erroneous signal will be sent to the vertical director pointer of the ADI. This erroneous signal will center the vertical director pointer indicating that a proper bank angle has been established to intercept the selected radial, and in reality the airplane will actually be off course. When the aircraft is on the selected TACAN radial and heading with the heading marker and course pointer set at the selected course, both the vertical director pointer (ADI) and the course bar (HSI) will be centered. If it then becomes necessary to establish a crab angle (aircraft heading different from the selected radial) the vertical director pointer on the ADI will indicate a heading error. To eliminate this apparent heading error, the heading marker should be manually set to correspond to the new aircraft heading. Do not expect the ADI to automatically correct for wind drift.

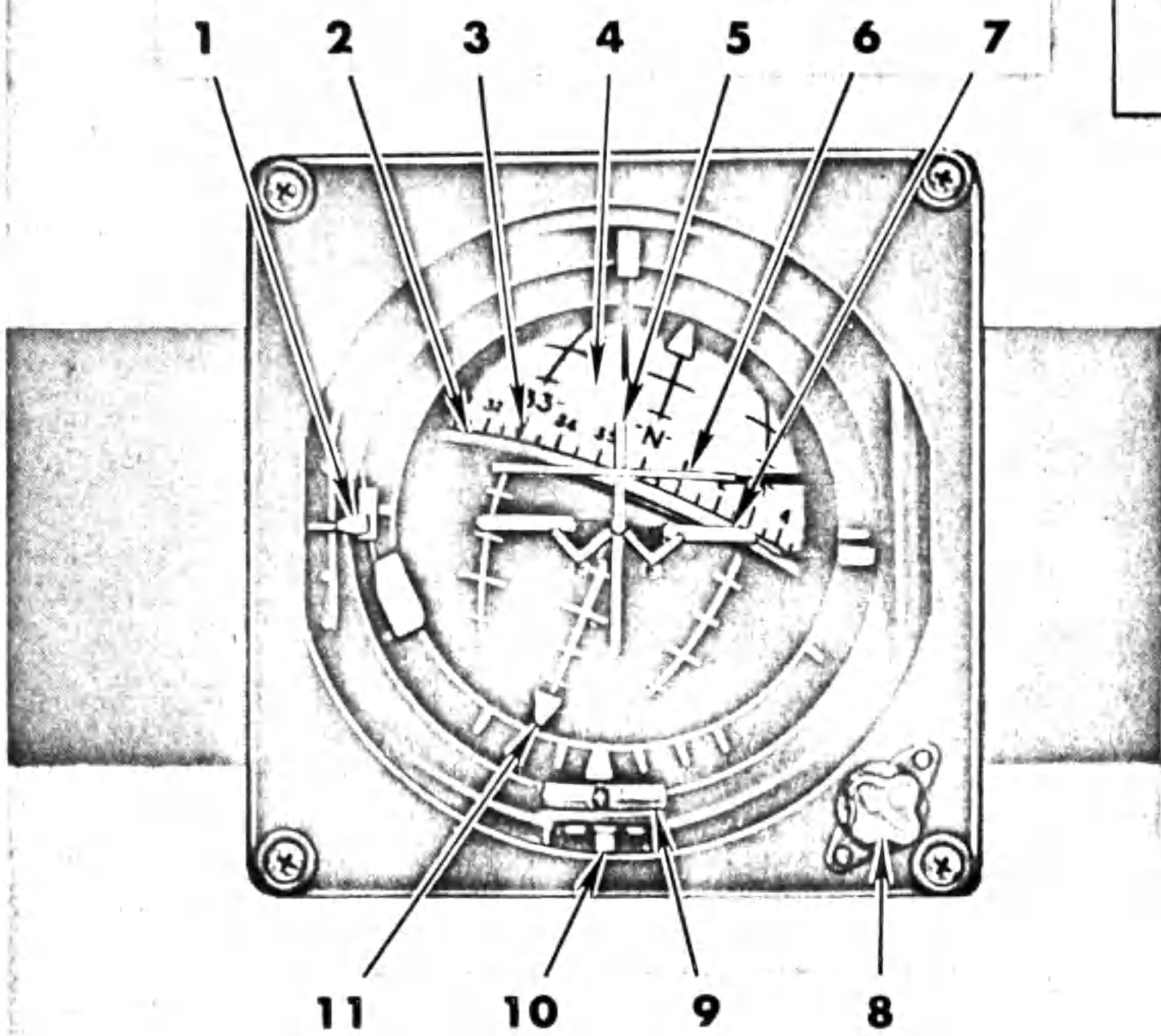
Bearing/Distance Selector Switch

The bearing/distance selector switch (figure 4-21) is a three-position toggle switch with switch positions of NAV COMP, ADF and TACAN. The NAV COMP position of the switch is inoperative and will not be discussed. The ADF and TACAN positions function as described below.

ADF - With the bearing/distance switch in ADF position, the bearing pointer of the HSI will point at the magnetic bearing of the UHF signal being received. Simultaneously, the UHF mode word will be lit on the HSI instrument if the instrument lights are turned up. The distance window will be covered.

ATTITUDE DIRECTOR INDICATOR

AIRPLANES 145313b, AND 146817c THRU 148275e



1. VERTICAL DISPLACEMENT POINTER
2. HORIZON
3. AZIMUTH SCALE
4. 3-AXIS SPHERE
5. VERTICAL DIRECTOR POINTER
6. HORIZONTAL DIRECTOR POINTER
7. WINGS SYMBOL
8. PITCH TRIM KNOB
9. BANK INCLINOMETER
10. RATE OF TURN NEEDLE
11. ROLL INDEX

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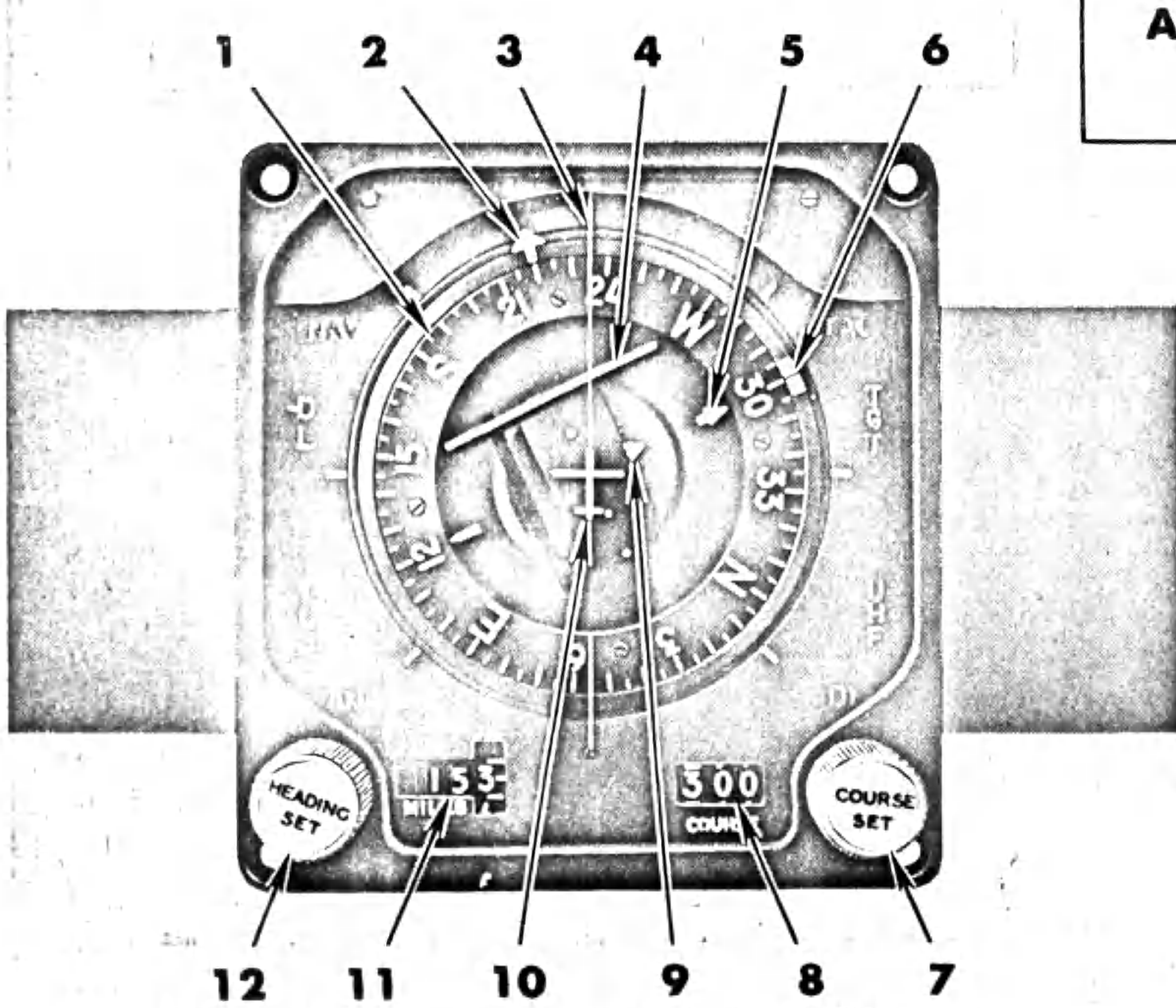
Figure 4-18

HORIZONTAL SITUATION INDICATOR

AIRPLANES 145313b, AND 146817c THRU 148275e



1. COMPASS CARD
2. BEARING POINTER
3. LUBBER LINE
4. COURSE BAR
5. COURSE POINTER
6. COMMAND HEADING MARKER
7. COURSE SELECTOR KNOB
8. COURSE READOUT WINDOW
9. TO-FROM POINTER
10. AIRPLANE SYMBOL
11. DISTANCE READOUT WINDOW
12. HEADING SELECTOR KNOB

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Figure 4-19

MODE CHART

AIRPLANES 145313b, AND 146817c THRU 148275e

INDICATOR	PARAMETER	MODE SELECTOR SWITCH				BOMB MASTER SWITCH
		ATT	HDG	TACAN	NAV COMP	LABS
ATTITUDE DIRECTOR INDICATOR	Vertical Director Pointer Off Flag	Off Scale From Flight Director Computer	Off Scale From Flight Director Computer	From Tacan Set	Off Scale From Flight Director Computer	Off Scale From Flight Director Computer
	Vertical Director Pointer	Off Scale From Flight Director Computer	Steering Signal For Selected Heading	Steering Signal For Selected Tacan Radial	Steering Signal For Selected Destination (BRG To Dest Minus Ground Track)	Steering Signal From AN/AJB-3
	Horizontal Director Pointer Off Flag	Off Scale From Flight Director Computer	Off Scale From Flight Director Computer	Off Scale From Flight Director Computer	Off Scale From Flight Director Computer	Off Scale From Flight Director Computer
	Horizontal Director Pointer	Off Scale From Flight Director Computer	Off Scale From Flight Director Computer	Off Scale From Flight Director Computer	Off Scale From Flight Director Computer	Steering Signal From AN/AJB-3
	Vertical Displacement Pointer	Off Scale From Flight Director Computer	Off Scale From Flight Director Computer	Off Scale From Flight Director Computer	Off Scale From Flight Director Computer	Off Scale From Flight Director Computer
HORIZONTAL SITUATION INDICATOR	Command Heading Marker	MAG Hdg Of A/C	Set By Pilot To Selected Hdg	Set By Pilot To Tacan Radial (Rotates With Compass Card)	Cmd Hdg From Flight Director Computer	Determined By Mode Selector
	Course Pointer	Mag Hdg Of A/C	Mag Hdg Of A/C	Set By Pilot To Selected Tacan Radial	Ground Track From Nav Comp	Determined By Mode Selector
	Course Bar Deviation	Zero	Zero	Deviation From Selected Tacan Radial	Zero	Determined By Mode Selector
	To-From Pointer	Out Of View	Out Of View	From Tacan	Out Of View	Determined By Mode Selector
	Mode Words		MAN	TAC	NAV	Determined By Mode Selector

INDICATOR	PARAMETER	BEARING DISTANCE SELECTOR SWITCH			
		NAV COMP	ADF	TACAN	
HORIZONTAL SITUATION INDICATOR	Bearing Pointer	Bearing To Destination	Bearing To ADF	Bearing to TACAN	Determined By BRG/Dist Selector
	Distance Window	Distance To Destination	Off	Distance to TACAN	Determined By BRG/Dist Selector
	Mode Words	NAV	UHF	TAC	Determined By BRG/Dist Selector

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Figure 4-20

MODE-BEARING/DISTANCE SELECTOR PANEL

TACAN - Selecting the TACAN position of the bearing/distance switch will light the mode word TAC on the HSI. It will also cause the bearing pointer to indicate the mag. bearing to the selected TACAN station, and will activate the distance measuring capability such that the nautical mileage to the TACAN station appears in the distance readout window.

It should be noted that simultaneous ADF and TACAN steering can be achieved by setting the bearing-distance switch to ADF and the mode switch to TACAN. ADF bearing will then be available on the bearing pointer and TACAN steering will be reflected by the vertical director pointer and the course bar. The mode words TAC, UHF, and MAN will not be illuminated on the face of the HSI unless the instrument lights are turned on.

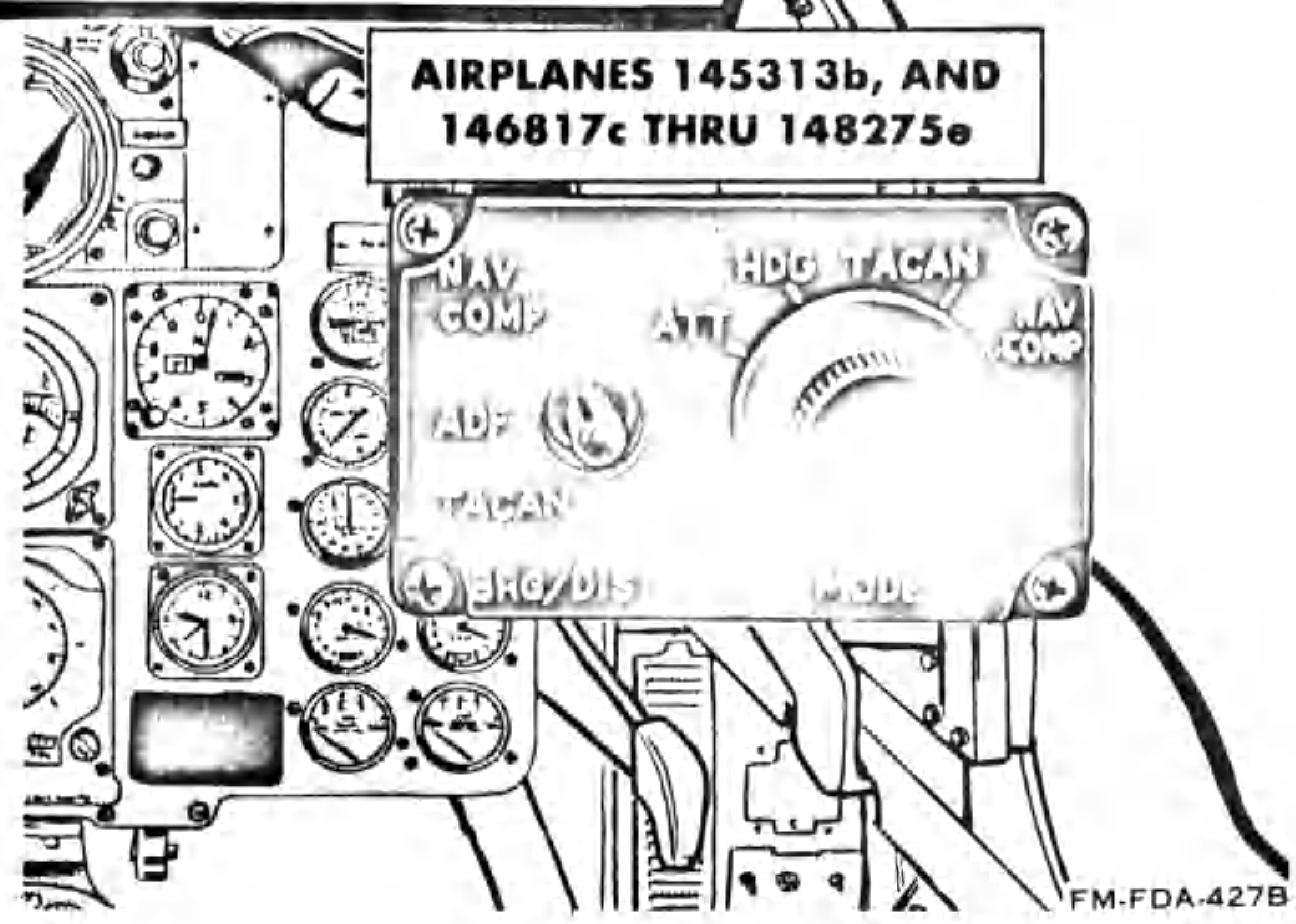


Figure 4-21

NAVIGATION-COMPUTER SYSTEM

F4H ASC 76 will be accomplished in all aircraft having undergone the F4H Drive-In Modification program. This ASC installs the AN/ASN-39 navigation computer. The AN/ASN-39 is a great circle computer and consists of a control panel and amplifier. The computer system makes the following dead reckoning computations during flight:

a. The latitude and longitude of the present position of the aircraft, which is displayed on the computer-control.

b. The aircraft ground track angle, relative to true heading.

c. The great circle distance from the preset base, as selected.

d. The great circle bearing to the preset target or preset base, as selected, relative to true heading.

FUNCTION AND LOCATION OF CONTROLS

All airborne operating controls for the navigation computer are located on the computer control panel in the aft cockpit. By means of this unit the crewmember sets in wind, variation and target and base position inputs, and reads out his present position. The controls and their functions are as follows:

Function Switch

The function switch is a five position rotary switch whose positions are marked OFF, TARGET, BASE, STBY, and RESET.

OFF - Remove all power from navigation computer.

TARGET - When the switch is in the TARGET position, the range to target, ground track relative to true heading, and course angle (bearing) relative to true heading will be displayed on the BDHI, HSI, and ADI.

BASE - When the switch is in the BASE position, the range and bearing output will be oriented to base, and will be displayed on the BDHI, HSI, and ADI.

STBY - In this position only filament power is supplied to the computer amplifier, and the latitude and longitude integrator channels of the system are inoperative.

RESET - In this position the latitude and longitude of the base, alternate target, or return point can be set into the system. Placing the switch in the RESET mode will cause the original set in coordinates to be lost. A restriction on the selector switch prevents accidental switching to the RESET position.

Wind Velocity Control Knob

The wind velocity control knob is used to manually insert the wind velocity affecting flight into the system and is displayed on the wind velocity counter.

Wind From Control Knob

The wind from control knob is used to manually insert the true wind direction. The true wind direction is expressed as an angle measured clockwise from true north, and is presented in degrees on the wind from counter.

Magnetic Variation Control Knob

The magnetic variation control knob is used to manually insert the magnetic variation angle into the system. The magnetic variation is the angular difference between true north and magnetic north, and is not a constant but varies over the face of the earth. The magnetic variation is displayed in degrees on the variation counter.

NAVIGATION COMPUTER CONTROL PANEL

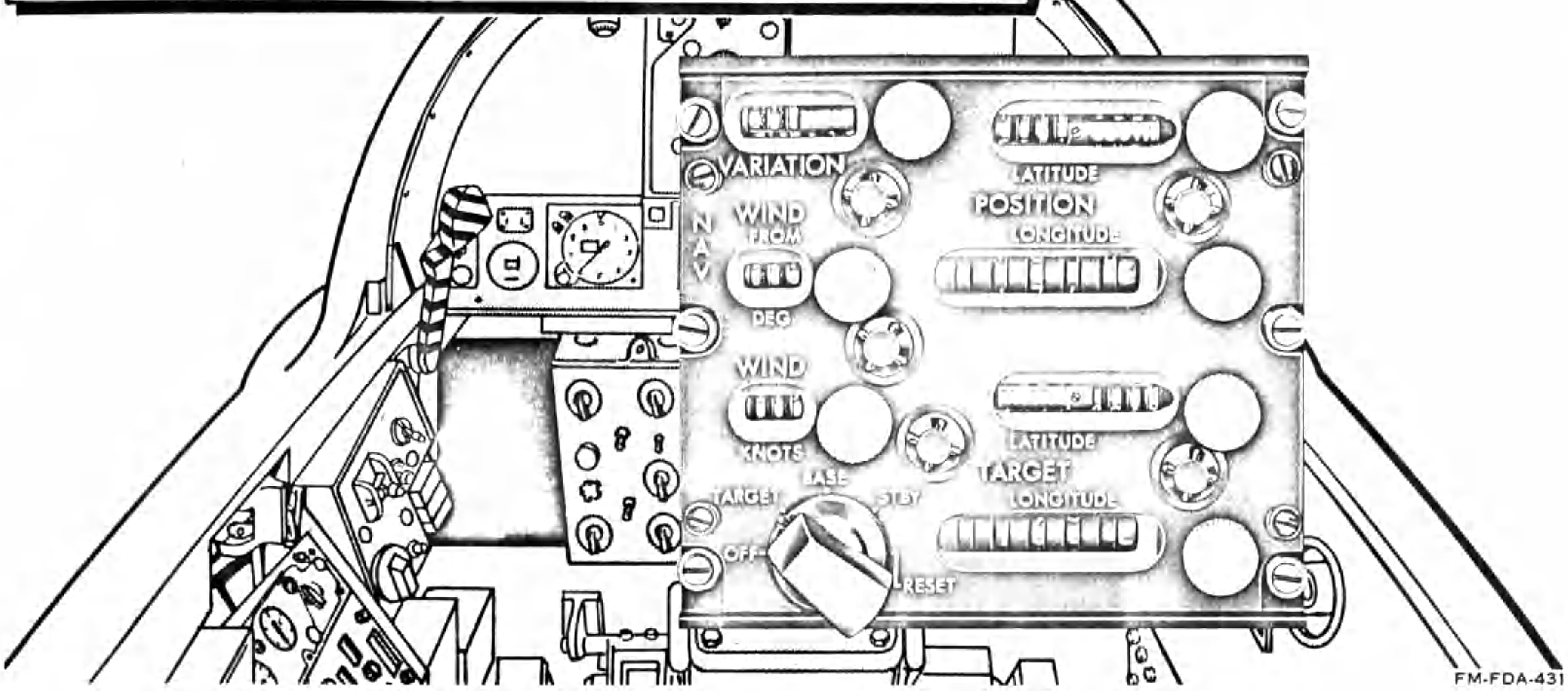


Figure 4-21A

Position Latitude Control Knob

This knob is initially used to manually insert the base latitude into the system, thereafter, it is used to insert the present position latitude as required. The base latitude is not displayed anywhere during flight, a memory of this ordinate is retained by the system so long as the function switch is not turned to RESET. The position latitude counter continuously indicates the aircraft latitude in degrees and minutes during flight.

Position Longitude Control Knob

The position longitude control knob functions the same as the latitude control knob described above.

Target Latitude Control Knob

The target latitude control knob is used to manually insert the target or destination latitude into the system. This information is displayed target latitude counter in degrees and minutes.

Target Longitude Control Knob

The target longitude control knob functions the same as the target latitude control knob described above.

OPERATION PROCEDURE

All control knobs required to operate the navigational computer are located on the computer control panel. To simplfy the procedure, it will be understood that where a counter setting is specified the control knob associated with the particular counter will be rotated to set the counter. In the case of the position latitude and position longitude counters, the associated control knobs must be pressed in to engage them with their respective counters before they can be rotated effectively.

Note

The flight plan may be set into the navigational computer set prior to starting the aircraft engine if desired. Steps 1 through 10 may be accomplished without power.

1. Magnetic variation counter - SET
2. Wind velocity counter - SET
3. Wind from direction counter - SET
4. Target latitude counter - SET
5. Target longitude counter - SET
6. Function switch - RESET
7. Position latitude counter - SET TO POINT OF RETURN
8. Position longitude counter - SET TO POINT OF RETURN
9. If the point of return differs from present position:
   a. Function switch - STBY
   b. Position latitude counter - SET TO PRESENT POSITION
   c. Position longitude counter - SET TO PRESENT POSITION
10. If engines have not been started
   a. Function switch - OFF
   b. Have engines started
   c. Function switch - STBY

After airspeed reaches 160 knots CAS.

1. Function switch - TARGET
   Placing the switch in the target position will result in system outputs of range to target, ground track relative to true heading, and course angle (bearing) relative to true heading.

Upon completion of mission over target:

1. Function switch - BASE

   Placing the switch in the base position will orient the system range and bearing output to the base.

INDICATOR DISPLAYS

The navigation computer information is displayed on the HSI, and ADI in the forward cockpit, and on the BDHI in the aft cockpit.

FWD Cockpit Displays

To display navigation computer information on the HSI and ADI select the NAV COMP position on the mode switch and the NAV COMP position on the bearing/distance selector switch of the mode-bearing/distance selector panel. The displayed information is as follows:

1. Bearing to target or base displayed on the bearing pointer of the HSI.
2. Ground track displayed of the course pointer of the HSI.
3. Distance to target or base displayed on the distance counter on the HSI.
4. Command heading (the heading the pilot must fly to align the bearing and course pointers) displayed on the heading marker of the HSI.
5. Steering information to fly an asymptotic approach to the command heading displayed on the vertical pointer of the ADI.

To fly the aircraft directly to the target or base turn the aircraft until the vertical pointer on the ADI is centered. Keep the vertical pointer centered by changing the heading and bank of the aircraft. The heading marker will rotate to the lubber line and the vertical pointer on the ADI will be deflected from center. As long as the mode selector switch on the mode-bearing/distance selector panel is in the NAV COMP position the ADI will present steering information to the NAV COMPUTER course, the course pointer will display ground track, and the heading marker will display command heading to the nav computer course. The bearing/distance switch on the mode-bearing/distance selector panel can be switched to any other function without interrupting the NAV computer display on the ADI. It only removes the NAV computer bearing and distance displays from the HSI.

AFT Cockpit Displays

To display navigation computer information on the BDHI, select the NAV COMP position on the CNI-NAV comp switch. The displayed information is as follows:

1. Bearing to target or base displayed on the #1 needle of the BDHI.
2. Ground track displayed on the #2 needle of the BDHI.
3. Distance to target or base displayed on the range counter.

BOMBING EQUIPMENT

Wings Level Bombing System

(Airplanes 145311b, 145312b and 145314b thru 145317b)

Bombing controls are located on the pilot's instrument panel (figure 1-17) and consist of two switches labeled LABS-OFF-DIRECT and INST o/s, TIMED o/s and LOFT. The second switch is effective only when the first is set to the LABS position. With the bomb control switches set to LABS and LOFT or TIMED o/s positions, and the pickle button is depressed, power is applied to the timing circuit and the G-programmer is energized upon completion of the time set in the Interval Timer. Switching to INST o/s bypasses the timing circuit. The pickle button is used to energize the G-programmer in this mode. The interval timer located in the aft cockpit is used to set the time interval between the identification point and the target. The timing motor is energized when the TIMED o/s or LOFT mode is selected. The interval timer, which is normally set prior to take-off is connected to the motor through a clutch when the pickle button is depressed. This action occurs when the aircraft passes over the I.P. The interval timer then runs for the preset time, at the end of which the pull-up signals (aural tone and light) are given and the G-programmer is energized. The interval timer also causes a 1/4 second long warning signal one second before the pull-up signal is given. The timing motor is de-energized at the end of the preset interval. The interval timer automatically recycles when the pickle button is released or at bomb release.

Wings Level Bombing System

(Airplanes 145313b, 146817c thru 148275e)

The bombing function of the system is identical to that described in the previous paragraph except for the method of attitude display. On the attitude director indicator, the "g" error magnitude and direction is displayed by the horizontal director pointer. The roll/yaw signal actuates the vertical director pointer so that when the pointer is flown back to the center position, compensating roll is introduced. Thus, if the two pointers (vertical and horizontal) are maintained at zero throughout the maneuver until the point of bomb release, proper profile and flight in the proper vertical plane will be assured. In the bombing mode, the pitch angle is indicated by the sphere. Although the attitude director indicator (ADI) is used as an indicating device for other systems, the LABS position of the bomb master switch has the final authority over the ADI displays. Refer to figure 4-20 for pointers that are used on the ADI and HSI when bomb master switch is in the LABS position.

AIRBORNE MISSILE CONTROL SYSTEM

Refer to Confidential Supplement, NAVWEPS 01-245FDA-1A.

PILOTS SIGHT UNIT

Refer to Confidential Supplement, NAVWEPS 01-245FDA-1A.

REFUELING SYSTEM

IN-FLIGHT REFUELING SYSTEM

The inflight refueling probe is located on the starboard side of the fuselage above the engine air inlet duct. The probe is equipped with an MA-2 refueling nozzle which is capable of receiving fuel from any drogue type refueling system. The refueling operation is actuated by the REFUEL PROBE switch which is located on the fuel control panel. The REFUEL PROBE switch has three positions; REFUEL, EXTEND, and RETRACT. The REFUEL position conditions the airplane fuel system for inflight refueling and extends the inflight refueling probe. The EXTEND position retains the probe in the extended position, but returns the airplane fuel system to normal operation. This position is used when normal fuel transfer is desired with the probe extended or in the event the probe is damaged and cannot be retracted. The RETRACT position returns the fuel system to normal transfer operation and retracts the probe. The refuel selection switch is located on the fuel control panel. This is a two-position guarded switch with ALL TANKS and INT. ONLY position. The ALL TANKS position opens the external tank fuel shutoff valves when refueling. The INT. ONLY position closes the external tank fuel shutoff valves and allows only the internal tanks to be refueled during inflight refueling. The airplane will refuel from Navy tankers at a rate of approximately 250 gallons per minute, and from Air Force tankers at a rate of approximately 610 gallons per minute.

INFLIGHT REFUELING PROCEDURES

Prior to the rendezvous with the tanker airplane, check the following equipment:

1. Radar Master switch - STBY
2. Radar Altimeter Power switch - OFF
3. Missile Power switch - OFF
4. Arming switch - OFF
5. UHF NAV function switch - REC.
6. IFF - STDBY
7. All unnecessary electrical equipment - OFF
8. Int Wing Dump switch - NORMAL

After the aircraft position is stabilized slightly below and aft the drogue:

9. Refuel probe switch - REFUEL
10. Refuel selection switch - ALL TANKS (if refueling of the external tanks is desired)

Visually check that the refueling probe is fully extended. An amber light on the tanker will indicate that the refueling tanker is ready for transfer. Move forward to engage the tanker drogue. After engagement of the drogue is accomplished, move forward about ten feet to initiate refueling a green light on the tanker will indicate that fuel is being transferred. An increase in power will be required to compensate for the increase in gross weight while taking fuel aboard. Upon completion of refueling reduce speed slightly to disengage from the drogue.

GROUND REFUELING SYSTEM

The airplane is capable of either single point or two point pressure refueling. The single point refueling receptacle is located on the right underside of the fuselage in the area below the aft cockpit. Single point pressure fueling at the rate of approximately 250 gallons per minute may be accomplished. Two point pressure fueling at the rate of approximately 480 gallons per minute may be accomplished by utilizing the inflight refueling probe with a special fitting attached. Effective airplanes 146817c thru 148275e, a wiring change has been made which allows a controlled partial refueling capability. If desired, fuel is locked out of the left and right wing tanks, and number 5 and 6 fuselage cells. This allows the airplane to be partially refueled to an amount of 902 gallons (approximately 5,683 lbs.) without creating an undesirable c.g. condition.

Cockpit Switch Positions

The switches on the fuel control panel located on the left console in the pilot's cockpit should be in the following position before single point pressure fueling. External transfer switch OFF, wing transfer pressure switch NORMAL, refuel selection switch ALL TANKS, buddy fill switch STOP FILL, refuel probe switch RETRACT. Refueling of the internal tanks only, with any or all external tanks installed may be accomplished by selecting INT ONLY on the refuel selection switch. The Buddy Tank is interchangeable with the centerline external tank and is refueled in the same manner as the centerline tank. The landing gear control handle must be in the gear down position and master switches and throttles should be in their OFF positions. If two point pressure refueling is desired the refuel probe switch should be placed in the REFUEL position.

REFUELING PROCEDURE (ENGINES OFF WITH ELECTRICAL POWER)

Apply auxiliary a-c electrical power to the airplane. Open filler door and attach fueling nozzle to service inlet valve. Set ground fueling switch located in the right wheel well to REFUEL position. (The ground fueling switch is only effective with the engine master switches OFF and ground electrical power applied.) With the REFUEL position selected on the ground fueling switch all valves in the fuel system will be closed with the following exceptions. The fuselage air pressure regulators will be open, all internal tank vent valves will be open, all external tank vent valves will be open if their respective tanks are installed and the refuel selection switch is on ALL TANKS. All fuel level control valves are open to receive fuel until their respective tanks are filled at which time floats rise in the valves to shut off fuel. Outboard and centerline external tanks motor operating shutoff valves will be open allowing fuel to fill the external tanks installed. A fuel level control valve, incorporated in each external tank, shuts off fuel entering the tank when the tank is full. A fuel flow transmitter in each refueling line to any external tank energizes a caution indicator light in the cockpit corresponding to the tank not accepting fuel. Partial refueling is accomplished by actuating and holding the left and right wing tank and No. 5 fuselage tank fuel level control valve switches.

PREPARATION FOR REFUELING WITHOUT ELECTRICAL POWER

1. Refuel probe circuit breaker - PULL (if probe extension is not desired)
   The RIO may pull refuel probe circuit breaker (if desired) to prevent the extension of the refuel probe.
2. Refuel probe switch - REFUEL
   Placing the refuel probe switch in the REFUEL position will condition the fuel system for refueling.
3. Throttles - OFF
4. When generators drop off the line (approximately 53% rpm):
   a. Engine master switches - OFF
      Placing the engine master switches off after the generators drop off the line will prevent the motor operated fuel level shutoff valve from closing.
5. Refuel probe switch - RETRACT
6. Refuel probe circuit breaker - IN (if previously pulled)

PREPARATION FOR REFUELING (ENGINES RUNNING)

1. Refuel probe circuit breaker - PULL (if accessible)
   If RO is being carried, pull refuel probe circuit breaker (if desired) to prevent the extension of the refuel probe.
2. Refuel probe switch - REFUEL
   Placing the refuel probe switch in the REFUEL position will condition the fuel system for refueling.

Functional Precheck of Electric Transfer Pumps

Individual momentary type check switches for each electrically operated transfer pump and a pressure indicator light are located on a panel in the left wheel well to provide a functional check for each electric transfer pump. When either switch is placed in the CHECK position, the primary circuit shuts off both fuselage fuel level control valves in fuselage cells No. 1 and No. 2, energizes the pressure transmitter switch and operates the transfer pumps in tanks No. 4 and No. 6. The pressure transmitter switch energizes the green indicator light if the discharge pressure of the selected pump is normal. No light while the pump is being checked indicates a malfunction in the pump.

Functional Precheck of Hydraulic Transfer Pumps

A momentary type check switch is provided on a panel in the left wheel well to check the operation of the hydraulic transfer pumps. The switch works in conjunction with two indicator lights. The check switch when placed in the CHECK position closes transfer pump level control valves in fuselage tank No. 1 and No. 2, opens the hydraulic shutoff valve to allow both pumps to operate and energize each pressure transmitter switch. Each pressure transmitter switch illuminates the green indicator light for each pump if their discharge pressure is normal. No light with the switch in the CHECK position indicates a malfunction of that pump. Hydraulic and electrical ground power must be connected to the airplane to conduct the above check.

Note

During the precheck of the electric and hydraulic transfer pumps, if the indicator light does not illuminate immediately, continue holding check switches for at least one minute, since the No. 1 fuel tank must be full before the No. 1 tanks fuel level control valve closes.

Functional Precheck of Refueling Level Control Valves

A double throw momentary type master check switch and seven individual momentary type check switches are located in a panel in the right wheel well. The master switch has positions of CHECK NO. 1 and CHECK NO. 2. With fuel flow started from the fueling source, hold the master check switch to the CHECK NO. 1 position. This position closes the motor operated shutoff valves of any external tank installed and energizes a solenoid in the primary float unit of the refueling level control valves, causing the primary floats to rise and shut off fuel flow to all internal tanks. Placing the master check switch in the CHECK NO. 2 position, closes the motor operated shutoff valves of any external tank installed and energizes a solenoid in the secondary float unit of the refueling level control valves which causes the secondary floats to rise and shut off fuel flow to all internal tanks. Continuation of fuel flow with the master switch in the CHECK NO. 1 or CHECK NO. 2 position indicates a malfunction of one or more of the refueling level control valves and/or motor operated shutoff valves. When a malfunction occurs in the primary or secondary system that respective position on the master switch shall be held. Malfunction of any refueling level control valve can then be isolated by operating the individual momentary type check switches one at a time to their respective position until fuel flow is stopped. The respective position of the individual switches energizes the solenoid in the circuit of each valve opposite to the circuit checked on the master switch. The individual switch that stops fuel flow indicates a malfunction of that valve in the primary or secondary unit respective to the circuit checked. In the event the refueling level control valve check does not stop fuel flow with an external tank installed, this indicates a malfunction of a motor operated shutoff valve corresponding to the tank or tanks installed.

Fuel Boost Pump Check

Two momentary type check switches are located on the fuel control panel to check the operation of the fuel boost pumps. This check can be conducted only with external power applied and the master switches OFF since either master switch in the ON position will operate both boost pumps. Placing either switch in the CHECK position operates the corresponding boost pump and opens the corresponding left or right engine shutoff valve allowing a pressure transmitter to pick up engine inlet pressure. This pressure is indicated on the applicable pressure gage for that pump. The pressure gages are located on the forward end of the pilot's left console. A below normal discharge pressure (30 ± 2 psi) reading while a pump is being checked indicates a malfunction. During flight or on the deck when the fuselage tanks are pressurized, the tank pressure will be additive to the pump pressure and indicated on the gages. The fuselage tank pressure will normally be as high as 4 psi, therefore, the gages could indicate as high as 36 psi when the fuselage tanks are pressurized. Pressures indicated in excess of 38 psi after an acceptable boost pump preflight pressure check is indicative of a malfunction in the fuselage tank pressurization system. The boost pump pressure check limits are 28 to 32 psi for preflight check with ground electrical power on or engines at idle and fuel tanks unpressurized.

DUAL CONTROLS CONFIGURATION

Dual flight control kits are available for installation in airplanes 148262e thru 148275e. This kit enables the airplane to be converted into a dual controlled airplane for pilot training purposes. The kit is installed in the RO's cockpit, and can be easily removed to return the airplane to its pilot, RO configuration. The dual flight controls kit provides the pilot in the converted RO's cockpit with a control stick, rudder pedals, throttle controls for each engine, and an instrument panel which contains the additional instruments needed by the aft cockpit pilot. The following controls are not available in the aft cockpit; flaps, speed brakes, landing gear, arresting gear, wheel brakes, and nose wheel steering.

FLIGHT CONTROLS

The conventional type control stick and grip controls the stabilator, ailerons, and spoilers. The grip contains a trim button for aileron and stabilator trim. The rudder pedals are not linked to the brakes, nor are they adjustable. A rudder trim switch is located on the instrument panel.

THROTTLES

Throttle controls for each engine are mounted on the CNI control panel support structure on the left side of the aft cockpit. The throttles have full throw between the range of IDLE and MIL, but they cannot be placed in the OFF or MAX positions. A microphone button with positions of ICS and UHF is the only electrical connections to the rear cockpit throttles. The dual throttle configuration incorporates a load limiter device which enables the pilot in the forward cockpit to overpower the throttle controls in the aft cockpit.

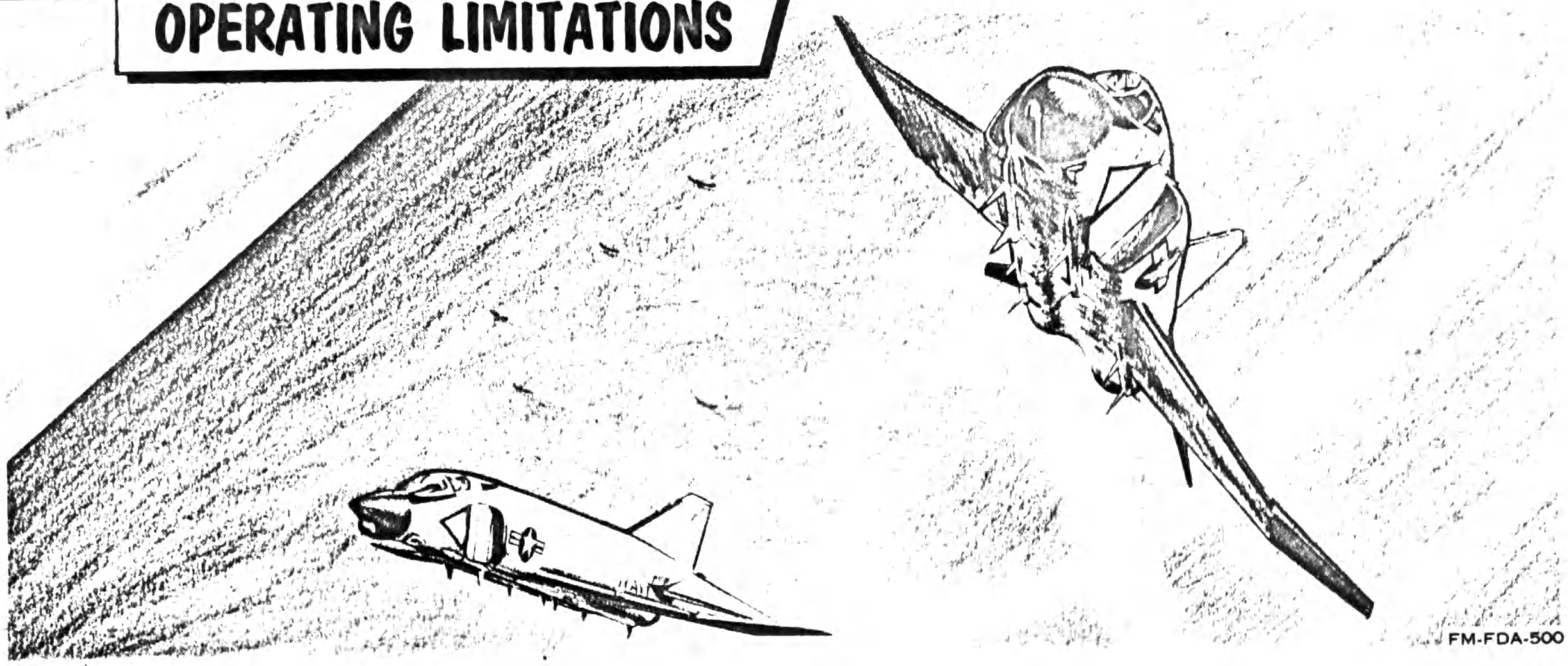
INSTRUMENTS

The dual control instrument panel contains the following instruments in addition to the ones normally installed in the RO's cockpit: vertical velocity indicator, turn and slip indicator, accelerometer, and engine tachometers. The aft cockpit instruments are powered from the same sources as those in the forward cockpit.

MISCELLANEOUS EQUIPMENT

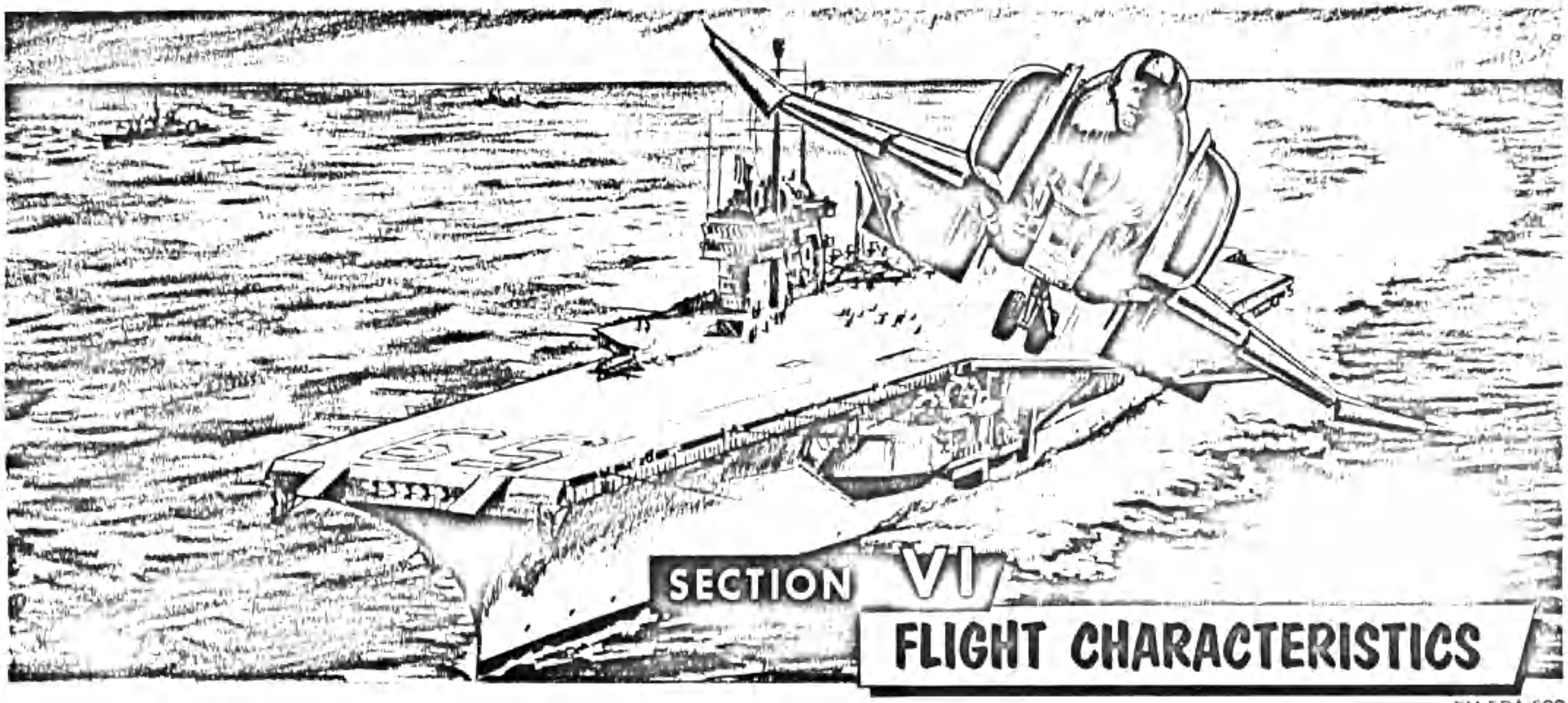
This information will be supplied when available.

SECTION V
OPERATING LIMITATIONS



FM-FDA-500

SEE PUBLICATION
NAVWEPS 01-245FDA-1A
OPERATING LIMITATIONS
FOR NAVY MODEL
F4H-1F AIRCRAFT



FM-FDA-600

INTRODUCTION

All information in this section is based on flight test data. Revised or additional data will be supplied in subsequent revisions as it becomes available. Refer to Section **VI NAVWEPS 01-245FDA-1A for Dive** Recovery Charts.

STALLS

ONE G STALLS

Normal stalls in the clean configuration are preceded by a wide band of buffet warning. Onset buffet occurs 40 knots above the stall and increases to moderate to heavy buffet immediately preceding the stall. Mild wing rocking will occur 5-10 knots prior to the stall. The stall is usually characterized by a right yaw and right roll, however, the aircraft will occasionally yaw and roll to the left. **Recovery is easily effected by positioning stabilator forward of neutral while maintaining neutral ailerons and rudder.**

WARNING

If the aircraft is close to or at the stall, use of ailerons to raise a wing will aggravate the yawing motion. Left aileron produces right yaw, etc.

STALLS IN LANDING CONFIGURATION

Stalls in the landing configuration are safe and comfortable with satisfactory control about all axes to within 8 knots of the stall. Longitudinal trim is available down to the actual stall. The three characteristics preceding each stall are: (1) nose rise, (2) roll oscillations, and (3) moderate buffet. The usual sequence of events during an approach to stall are:

1. Pedal Shaker - This is activated by the angle-of-attack system and is a stall warning device set at minimum approach airspeed. It actuates approximately 19 knots above the stall.
2. Nose Rise - A mild rate 2° to 3° nose rise will occur along with stick force lightening at 10 knots above stall, and occasionally just prior to the stall pitch down.
3. Wing Rock - A gentle roll oscillation will occur at 8 knots above stall and usually does not exceed 10° bank angles until the stall is reached. However, during some stall approaches the wing rock increases to as much as 60° to 80° left and right and the stall approach should be broken off at this point by gentle forward stick movement until recovery attitude is reached.
4. Buffet - A moderate buffet will usually be detected 1-2 knots above stall.
5. Stall and Recovery - If wing rock does not exceed 20°, the stall can be carried to the point of the second nose rise followed immediately by a nose down pitch. In all cases recovery is initiated or assisted by easing forward on the stick. Recovery attitude is usually about 30° nose down. External stores including wing pylon tanks have no effect on stall speeds or characteristics. Stalls with one half flaps are very similar to full flap stalls. Stall airspeeds for various gross weights and bank angles in the clean and landing configuration are shown on the Stall Speed Chart. A flat final approach can be flown at minimum airspeeds, (pedal shaker, angle-of-attack) if necessary. Minimum airspeeds are dictated by flare capability since extremely rapid engine response offers excellent altitude and speed control.

INVERTED STALLS

Inverted stall entries have been made up to negative pitch angles of 60°. A negative angle of attack stall can be instantaneously obtained only with abrupt full forward stick deflection at a negative (inverted) pitch angle of 20° or more. Light to moderate buffet will occur at the stall, and there are no distinct roll or yaw tendencies.

CAUTION

Loss of engine oil pressure will result from sustained negative accelerations.

ACCELERATED STALLS

Subsonic

Accelerated subsonic stalls are preceded by buffet increasing progressively to a moderate level immediately prior to the stall. Lateral instability in the form of wing rocking occurs concurrently with buffet and progresses to a fairly high frequency, large amplitude roll oscillation. The airplane usually yaws and rolls to the right at the stall; occasionally a yaw and roll to the left may occur. Increasing the rate of aft stick displacement will increase the magnitude and rate of yaw and roll at the stall. Abruptly applying and holding full aft stick can result in a rapid spin entry. Prompt neutralization of ailerons and rudder, and positioning the stabilator forward of neutral will usually effect recovery from accelerated stall entries. A right roll may result during recovery if the stall entry was extremely abrupt. Below 35,000 feet, if the stick is abruptly pulled full aft and held, a spin may develop, and it is also possible that the spin cannot be broken within a one spin turn. Rapid stall entry with abrupt control applications will result in a stall without noticeable occurrence of the above-mentioned buffet and wing rocking as warning prior to stall. Moderate buffet will be evident at the stall. As altitude is increased above 35,000 ft. the aircraft becomes more spin resistant to abrupt full stick stalls.

WARNING

Stalls performed in the landing configuration should not be carried beyond uncontrollable wing rock. Stalls performed in the clean configuration should not be carried beyond uncontrollable wing rock and/or the onset of airplane nose-right yawing (nose "slicing"). Forward stick and rudder as required will effect an immediate recovery from any stall that does not exceed the above conditions.

CAUTION

Do not practice landing configuration stalls above 10,000 ft. The effectiveness of the BLC system, and engine bleed pressures decrease with altitude. Use of BLC at extreme altitudes may cause systems using engine bleed air to become inoperative.

Supersonic

Supersonic accelerated stalls exhibit less buffet and wing rocking than subsonic stalls.

Note

Stall speed charts are included in Section VI of the Confidential Supplement to Flight Manual NAVWEPS 01-245FDA-1A.

SPINS

The F4H is considered to be highly spin resistant. Because of excellent longitudinal stability characteristics, the aircraft can usually be flown well past stalled angle-of-attack with positive control recovery without a spin ensuing. A wide variety of spin entry conditions have been investigated and included spins from:

1. Level flight.
2. Accelerated turns below and above 50,000 ft.
3. Accelerated turn reversals.
4. Vertical climbs.
5. Supersonic (1.2 Mach) will speed brakes extended (accelerated turns).
6. Inverted climbs up to vertical.

All developed upright spins are stable, and none approached a flat spin mode. Control recoveries are possible from all maneuvers.

UPRIGHT SPINS

The upright spin has an oscillatory behavior in pitch, roll and yaw which is predominant on the first two or three turns but diminishes as the spin is increased. Nose position varies from level to slightly above the horizon at the start of each turn to 50°-70° nose down at the 1/2 turn position. The roll and yaw oscillations that are present do not produce uncomfortable accelerations on the pilot. Altitude loss per turn in a steady state spin is approximately 2,500 feet. Altitude to accomplish a control recovery described below is approximately 15,000 ft. It can take 5,000 ft. to break a steady state spin and regain flying speed and an additional 10,000 ft. to level out while holding onset buffet. This altitude can be shortened by pulling moderate buffet on level out or by having a rapid recovery to flying speed. High energy effects on spin mode are noted during spin entries from supersonic accelerated turn conditions with speed brakes extended. High rotational rates occur during the incipient stage of the spin, but return to the normal spin rate by the conclusion of the third turn. Influence of extended speed brakes on spin mode and recovery is negligible. Forward stick motion during the incipient stages of a spin will cause recovery, however, application of forward stabilator during steady state spins produces a significant increase in rotational rate and a slight increase in aircraft angle of attack. No divergence toward a flat spin mode will be evidenced. Right spin direction evidences higher rotation rates and angle of attack than spins to the left. The aircraft is especially spin resistant from spin entries out of vertical climbs, however, if an incipient spin is entered at a high pitch angle and low airspeed, positioning stabilator forward of neutral while holding neutral ailerons and rudder is the recommended procedure. The aircraft will pitch forward at the top of the trajectory and regain flying speed.

UPRIGHT SPIN RECOVERY

If the aircraft enters an out-of-control flight condition, the following recommended steps should be taken for recovery from upright stalls/spins.

1. Position stabilator forward of neutral, neutralize ailerons and rudder, and hold in neutral position for 1 1/2 spin turns. The aircraft should fly out of nearly all out-of-control conditions with these control motions.
2. If the aircraft is not flying after taking the action prescribed above and altitude is over 30,000 ft. with daytime visual conditions prevailing and a control recovery is desired to be attempted, apply:
   a. Aileron with spin (with apparent roll), rudder against spin and stabilator aft. Hold these controls until yaw rate has definitely stopped and then neutralize rudder, ailerons and position stabilator forward of neutral.
   b. If the aircraft then exhibits mild reversals, counter with rudder and aileron while holding stabilator forward of neutral. If reversals are of a magnitude such that it is believed that another spin is being entered, repeat spin recovery technique outlined in 2.a.
3. If under 30,000 ft., at night, or in instrument conditions, and airplane is not flying after the 1 1/2 turn waiting period described in 1 above, deploy drag chute and position stabilator forward of neutral while holding neutral ailerons and rudder for recovery from the spin. The drag chute can then be jettisoned when the aircraft is solidly under control. The drag chute will produce a recovery within one turn from pilot actuation. It takes 1/3 to 1/2 turn to fully deploy and produce load. Drag chute recovery is extremely positive and no reversal tendencies will be encountered.

Control recovery can be more rapid if application of recovery aileron is timed to re-inforce the aircraft's roll oscillation.

INVERTED SPINS

No inadvertent inverted spins should be encountered since the aircraft possesses a high degree of directional stability at negative angles of attack. Without introduction and persistence of pro-spin controls, an inverted spin cannot be maintained in a stalled attitude at any pitch angle.

INVERTED SPIN RECOVERY

Recovery from an inverted spin is as follows:

1. Apply rudder opposite to visual spin direction. Keep stabilator forward of neutral and ailerons neutral.
2. When yaw rate stops, neutralize all controls.

ENGINE EFFECTS

Spin tests have shown that the engines will probably flame-out during spins and spin entries. Normal flight control operation for recovery will deteriorate after three turns and may not be possible after five turns. Continuous ignition has proved to be effective in keeping the engines running, and will be retrofit to all airplanes.

HIGH SPEED FLIGHT AT LOW ALTITUDE

The stability of the F4H is good with or without stability augmentation operative, but the stabilator sensitivity is very pronounced in the transonic region. The amount of sensitivity will vary with the center of gravity through an approximate range of 1.7 "g" per degree of stabilator to 2.5 "g" per degree. For example, when flying in the transonic region at low altitude with 4 Sparrows and the TX-28 store installed, three degrees of stabilator motion will create 6.6 "g's" of load factor. This is only 1.09 inches of stick travel, measured at the point of the trim button. Therefore, it is strongly recommended to fly the aircraft smoothly while in this regime, steering clear of unnecessary rapid stick inputs. The sensitivity is still present with the stability augmentation operative; however, the dampers will decrease the stabilator response to rapid stick inputs.

PILOT-INDUCED OSCILLATION

Transonic flight at low altitude presents low stick force gradients which in turn results in an area of sensitivity or possible over-control. Even though the inherent dynamic stability of the airplane is positive, it may be possible to create a short period longitudinal oscillation if the pilot's response lag and the control system lag become coupled with the airplane motion, thereby inducing negative damping. Such a condition is commonly known as "PIO" or pilot-induced oscillation. Since the dampers will decrease the stabilator response to rapid stick inputs, the possibility of inducing an oscillation are minimized because of the change in the effective time constant in a favorable direction. Therefore, it is recommended that the stability augmentation be used when flying at high speed and low altitude. The standard, and most effective, recovery technique from a pilot induced oscillation is to release the controls. Therefore, it becomes advisable to trim out longitudinal stick forces while accelerating in this flight regime. Releasing controls while stick forces are present because of an out-of-trim condition could be additive and simply amplify the oscillation. If the altitude of a mission is such that it would not be desirable to release the flight controls, the forcing function must be eliminated by making the arm and body as rigid as possible, and possibly bracing the left hand against the canopy. While flying in this highly sensitive region and a sharp stick input is necessary, a nose up correction is much more desirable than a sharp nose down correction, since the lower lift coefficient is more conducive to PIO. In this regard, caution should be exercised during a rapid afterburner acceleration at low altitude while maneuvering such as steeply banked flight. In this case, an out-of-trim condition will be present when returning to one g level flight necessitating a pushover to remain level. This type of correction should be performed smoothly. Always have the shoulder straps locked while flying at low altitude and high speed. The pilot's body from the lap belt up could become the forcing function during an inadvertent pitch input if the shoulder straps are unlocked.

SUPERSONIC FLIGHT

Supersonic flight characteristics are essentially the same as exhibited during subsonic flight. There are no unusual trim changes above Mach 1.0. At Mach numbers above 1.3 dihedral effect becomes neutral or slightly negative. This is of little concern to the pilot and usually will escape notice unless a deliberate attempt is made to yaw the airplane at high Mach numbers.

FLIGHT CONTROL EFFECTIVENESS

STABILATOR

Airplane response to longitudinal stick displacement is good in all flight conditions, however, stabilator effectiveness varies considerably throughout the flight envelope. Longitudinal sensitivity, and consequently stabilator effectiveness is greatest at high "q" i.e., at high subsonic speeds and low altitudes. Effectiveness is greatly reduced in the BLC landing configuration and lessened still further when in ground effect. Enough effectiveness is retained however, that only on rare occasions will it be necessary to use full aft stick for landing. Stabilator effectiveness also deteriorates at high supersonic speeds; here it becomes apparent that more aft stick is required to increase load factor than is necessary at subsonic speeds. It is quite possible to use full aft stick at supersonic speeds at high altitudes without exceeding the airplanes' limit load factor.

LATERAL CONTROL

Lateral control is adequate in all flight conditions except in the single-engine landing configuration, where lateral control becomes marginal below 130-135 knots IAS at landing gross weight. Roll rates at any given altitude remain fairly constant throughout the subsonic speed range, and then start to decrease as speed is increased supersonically. However, the rate of decrease is such that roll rate is still adequate at the maximum airplane Mach number. Caution should be used in lateral control until the pilot has become familiar with their use, due to the extremely high roll rates obtainable under certain conditions. Incorporation of the ARI (Aileron-Rudder Interconnect) effectively prevents adverse yaw with lateral control application at low speeds (below 225 knots). At speeds above 225 knots the ARI system is automatically cut out since adverse yaw is negligible.

RUDDER

Rudder control becomes effective at approximately 70 knots and remains so throughout the high speed range. As in most jet aircraft, very little pilot initiated rudder control is required. The yaw damper is very effective in damping short period oscillations and the ARI system mentioned previously takes care of the adverse yaw which occurs below 225 knots.

SPEED BRAKES

The speed brakes are very effective decelerating devices under most conditions. Extension and retraction occurs with very minor pitch transients. In the extended condition the speed brakes create a mild buffet. For this reason and also because speed brake effectiveness is low at landing approach speed, it is recommended that landings be made with speed brakes in. Only in the case of a no-flaps landing is it recommended that speed brakes be extended. A mild "nip-up" is usually experienced in the transonic speed range while decelerating from supersonic to subsonic speeds with the speed brakes out and while holding "g's" on the airplane.

MANEUVERING FLIGHTS

Stick force per "g" is fairly constant throughout the speed range, in contrast to some aircraft in which large differences exist between subsonic and supersonic speeds. Stick force per "g" does increase slightly at high supersonic speeds but only on the order of approximately 2 pounds per "g". This is due to a combination of an increase in static stability and a decrease in stabilator effectiveness.

STALL SPEEDS

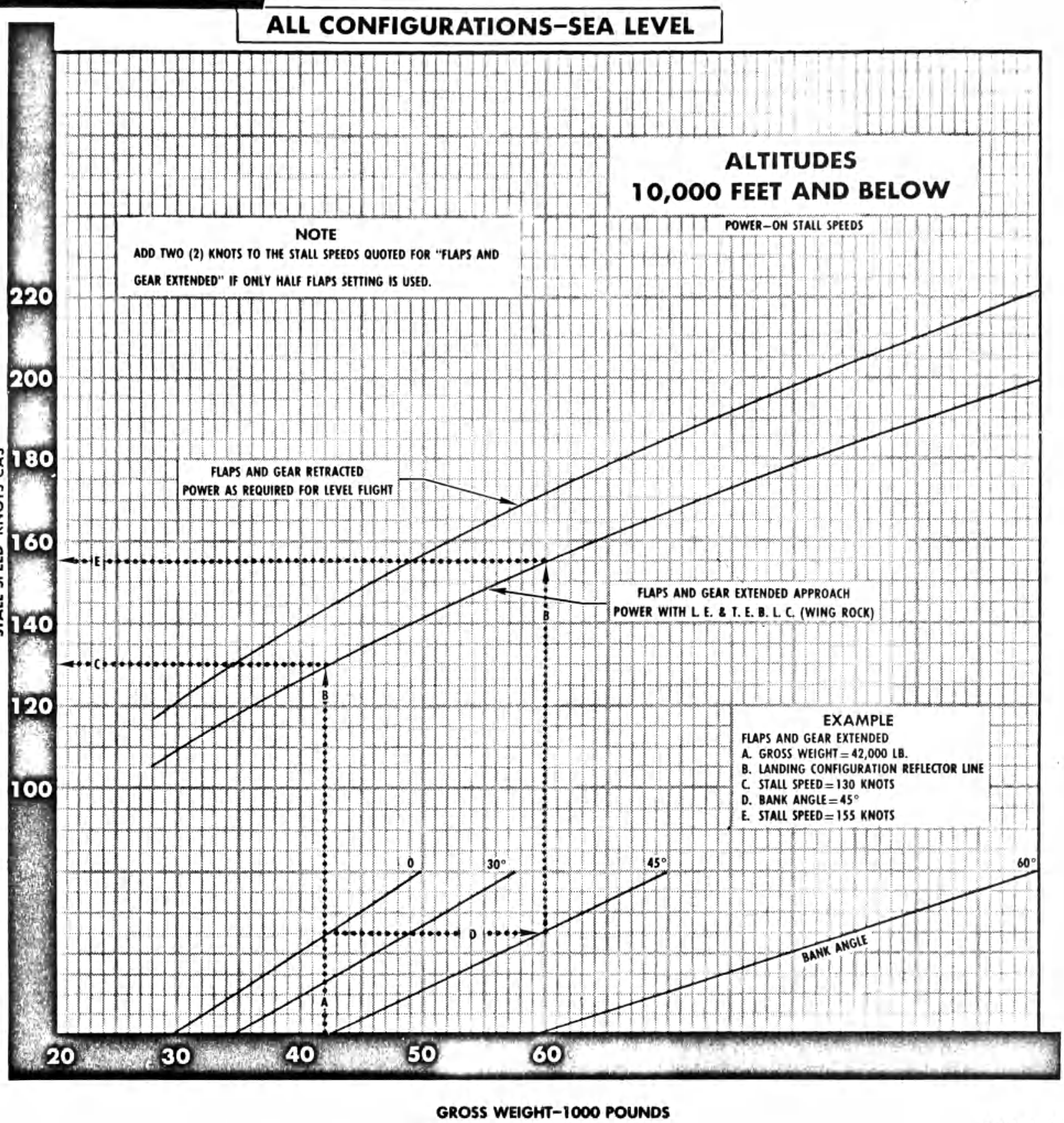


Figure 6-1

ANGLE-OF-ATTACK RELATIONSHIP

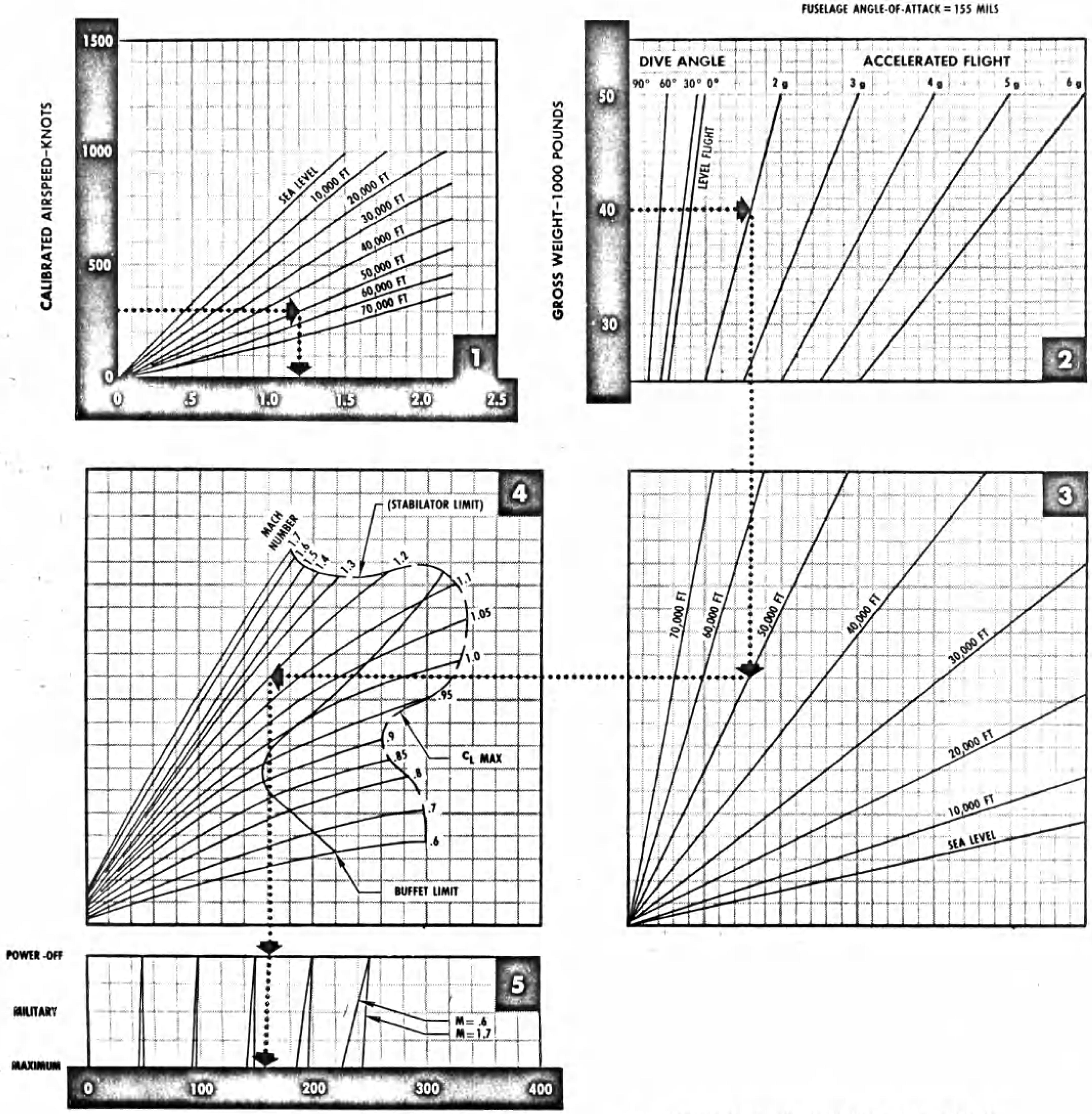
ALL CONFIGURATIONS

EXAMPLE

STEP 1: CALIBRATED AIRSPEED = 310 KNOTS
ALTITUDE = 50,000 FEET
MACH NUMBER = 1.2

EXAMPLE

STEP 2: GROSS WEIGHT = 40,000 POUNDS
LOAD FACTOR = 2g
ALTITUDE = 50,000 FEET
MACH NUMBER = 1.2
FUSELAGE ANGLE-OF-ATTACK = 155 MILS



FUSELAGE ANGLE-OF-ATTACK-MILS

- ANGLE-OF-ATTACK OF WING = ANGLE-OF-ATTACK OF FUSELAGE +1°
- 1° = 17.45 MILS

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Figure 6-2

ANGLE OF ATTACK CONVERSION

AIRPLANES 145314b THRU 145317b

ALL CONFIGURATIONS–FLAPS AS NOTED, GEAR DOWN

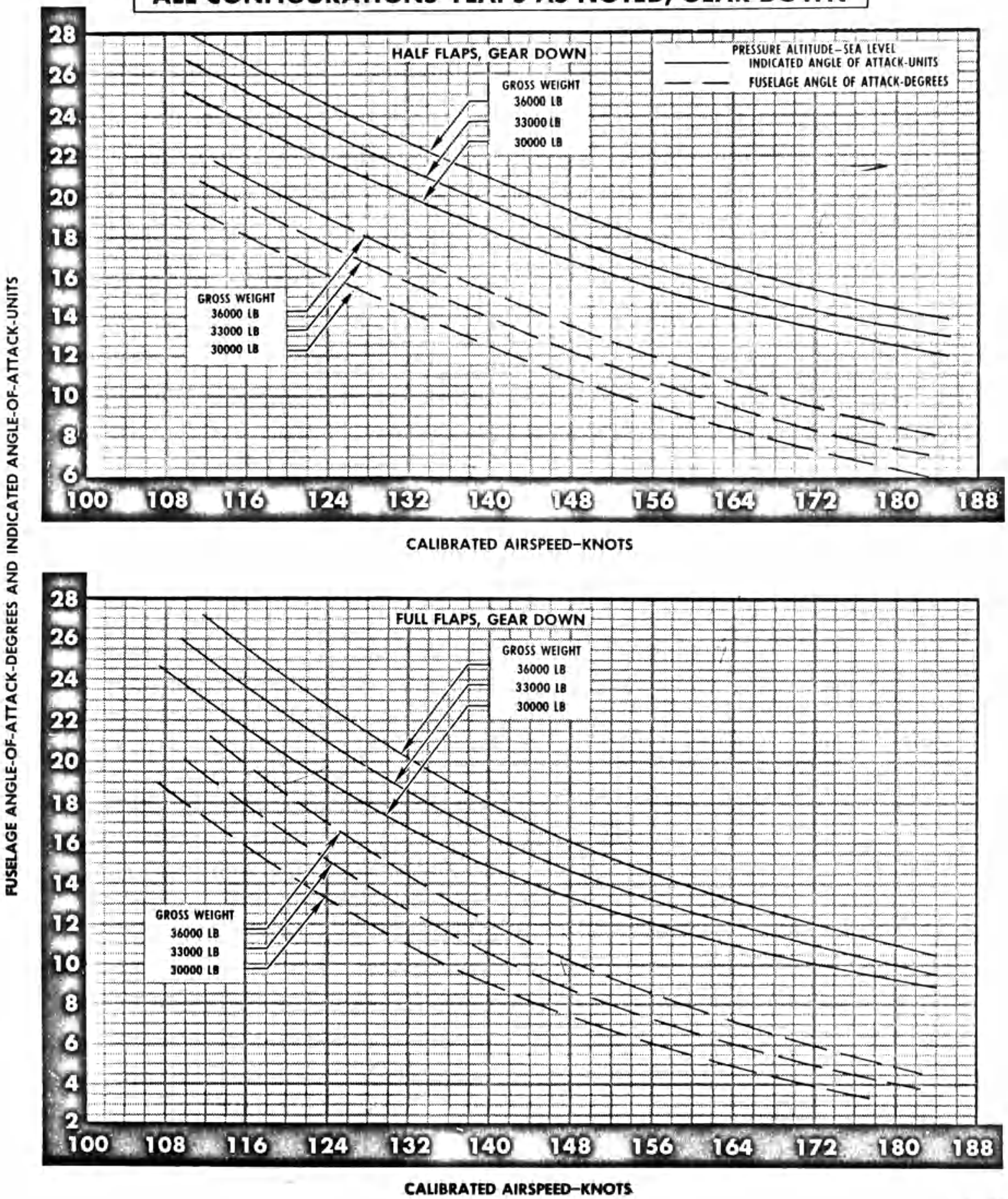


Figure 6-3

ANGLE OF ATTACK CONVERSION

AIRPLANES 145314b THRU 145317b

ALL CONFIGURATIONS–FLAPS RETRACTED, GEAR UP

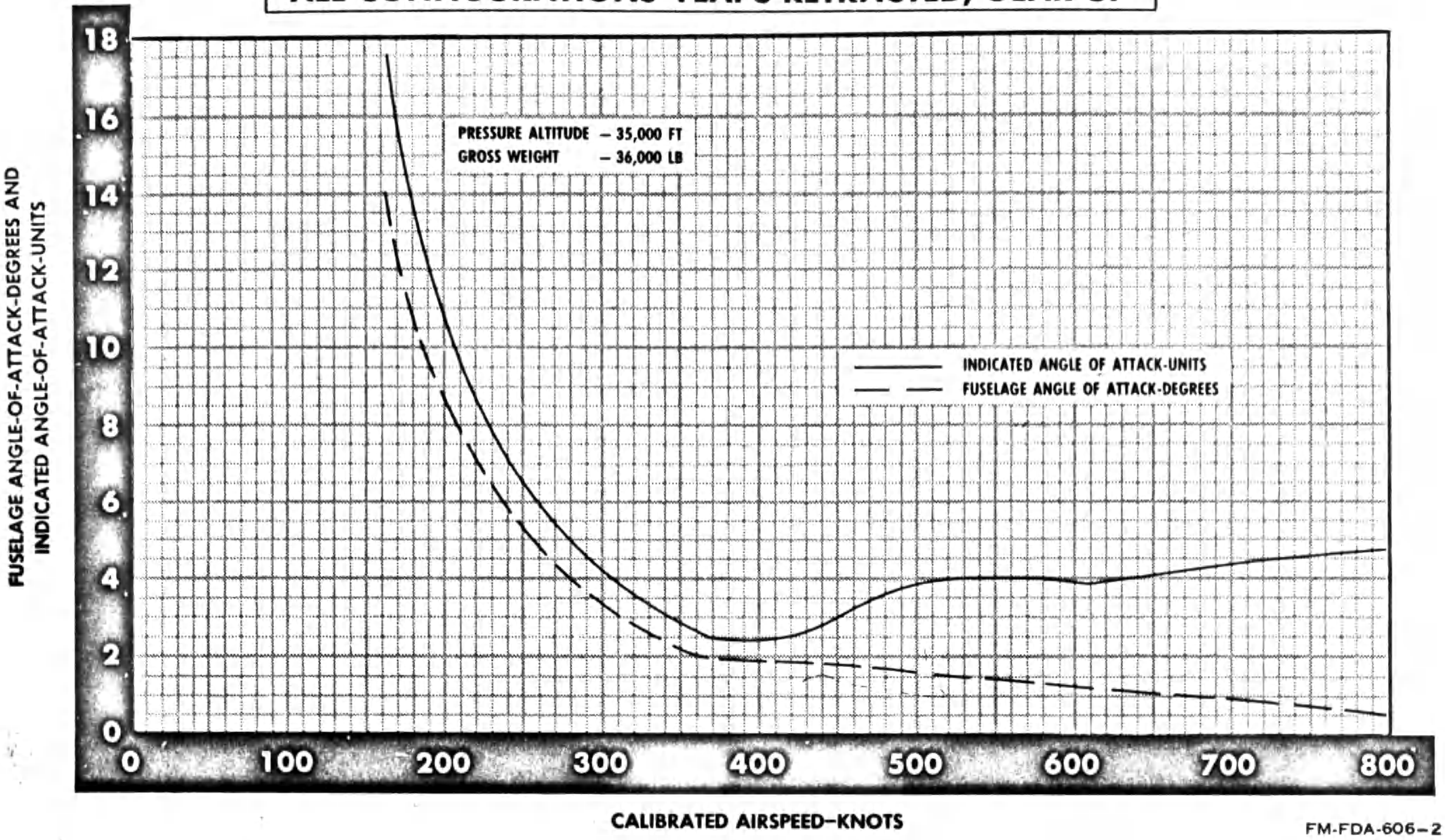
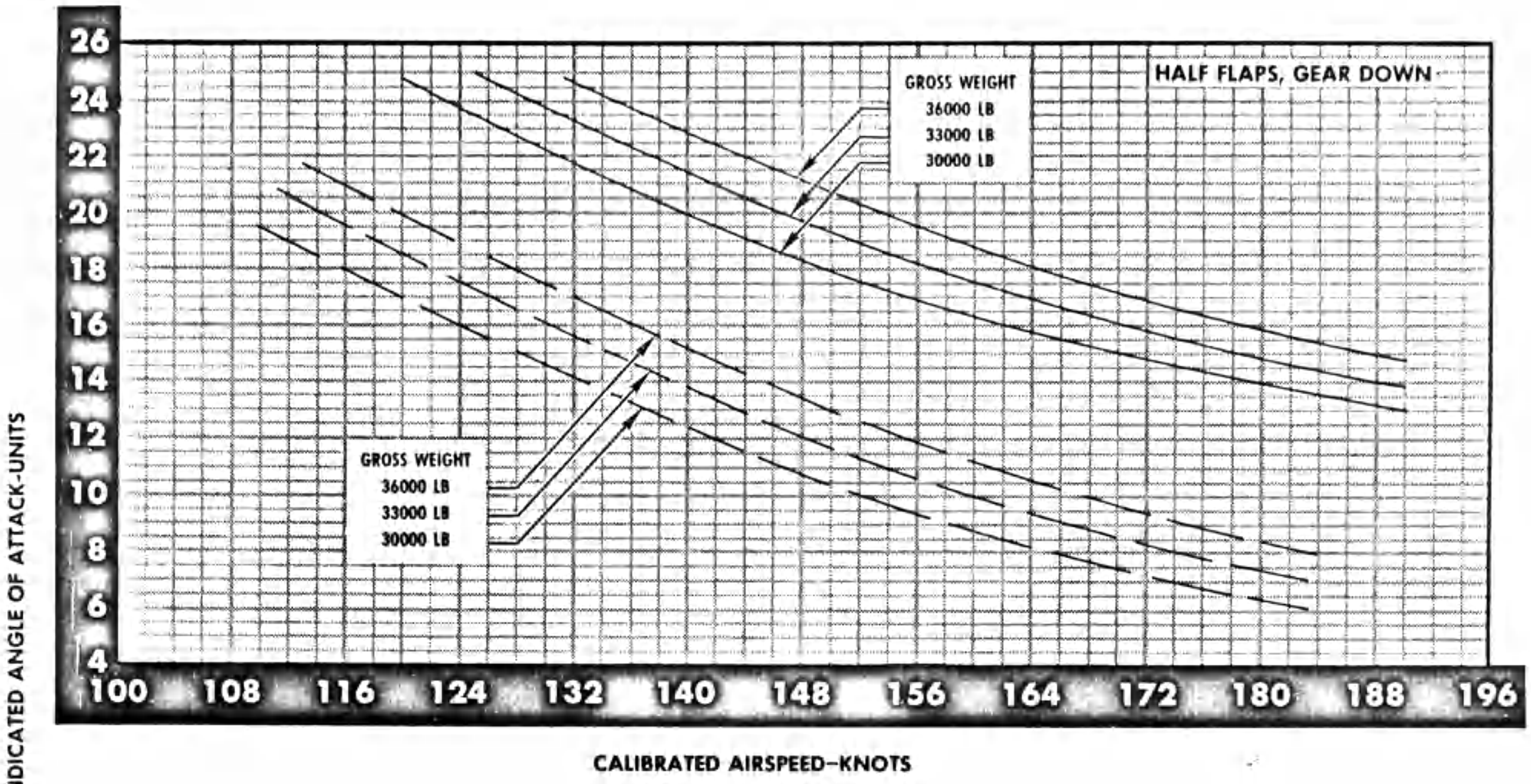


Figure 6-4

ANGLE OF ATTACK CONVERSION

AIRPLANES 146817c THRU 148275e

ALL CONFIGURATIONS—FLAPS AS NOTED, GEAR DOWN



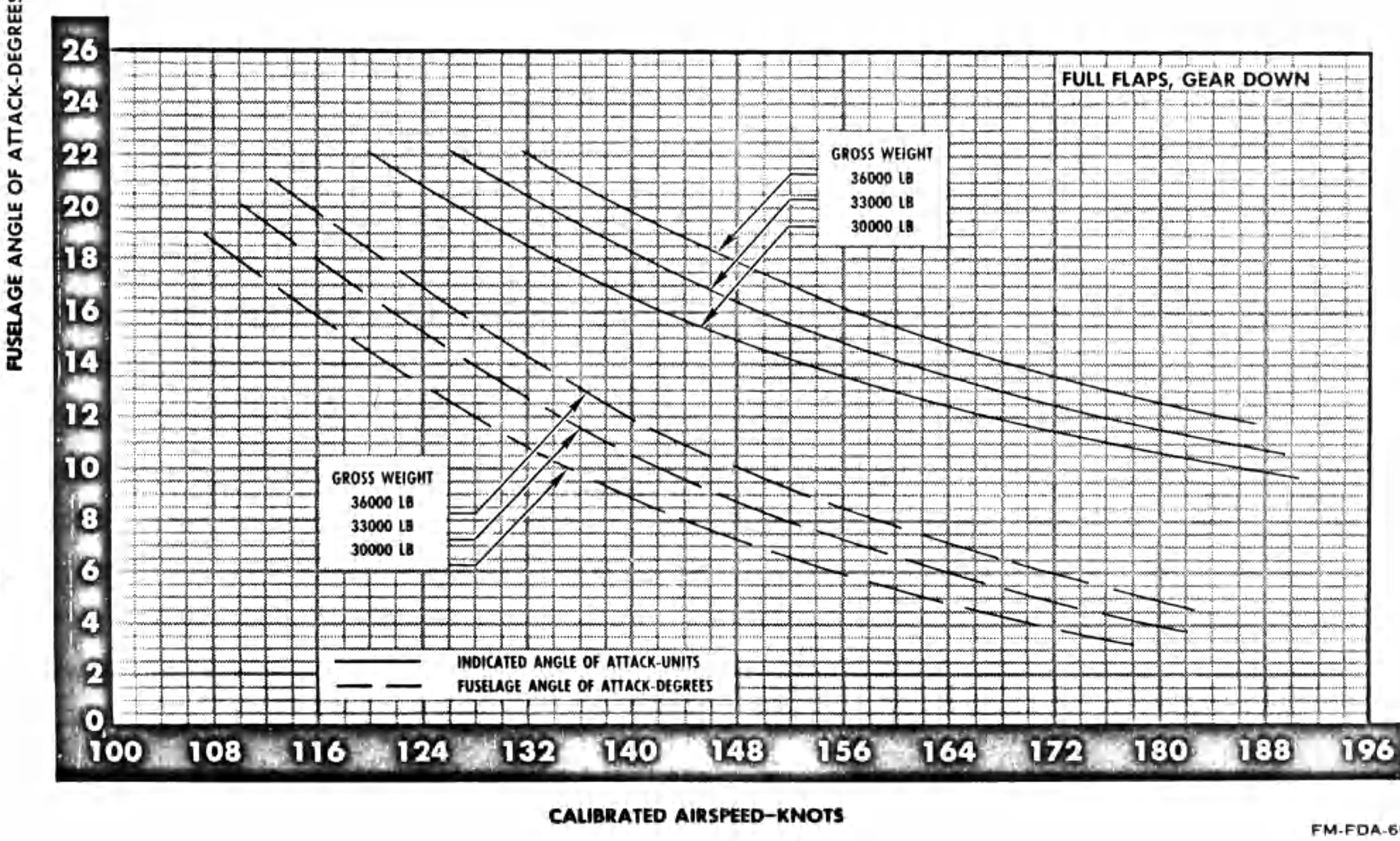


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Figure 6-5

ANGLE-OF-ATTACK CONVERSION

AIRPLANES 146817c THRU 148275e

ALL CONFIGURATIONS – FLAPS RETRACTED, GEAR UP

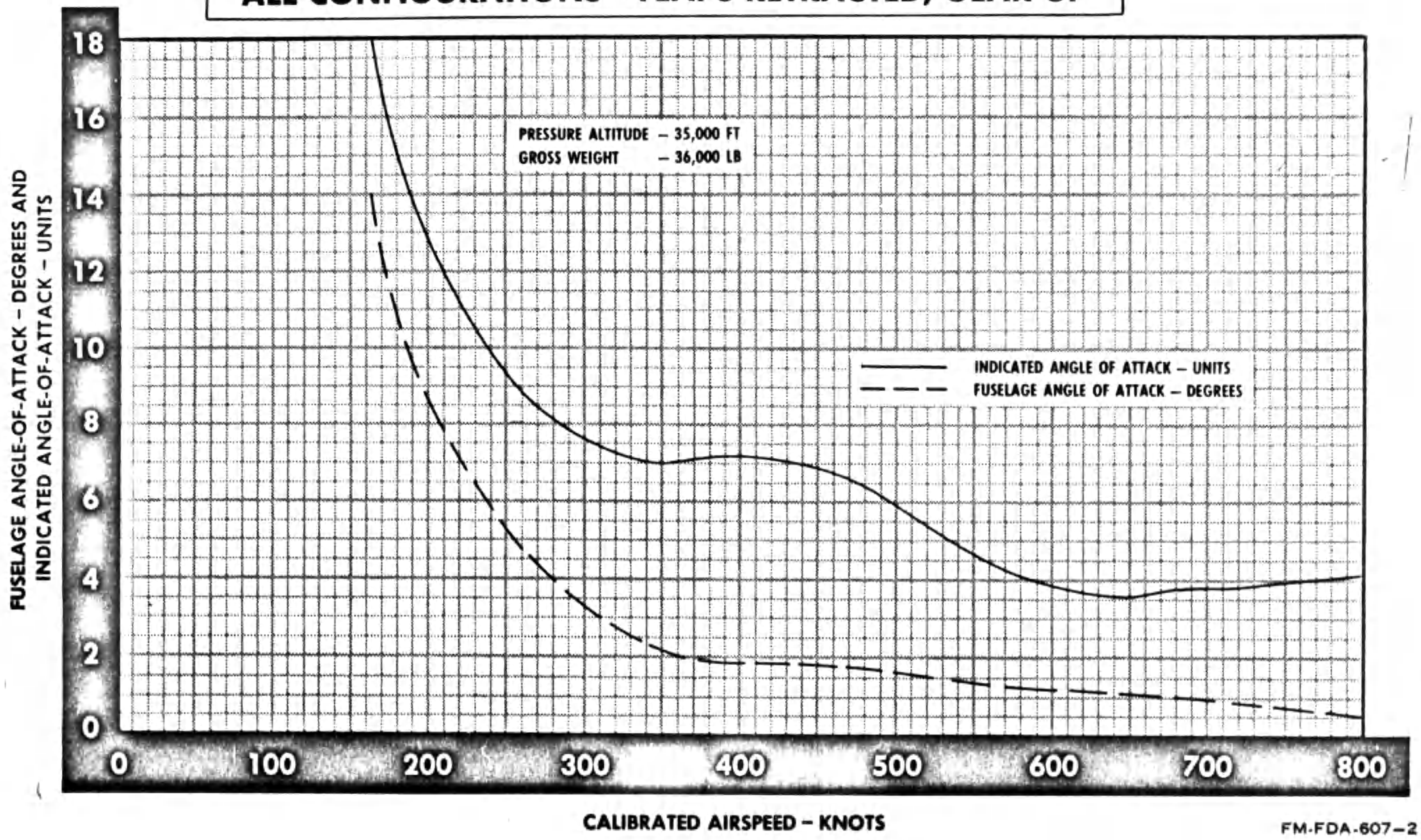
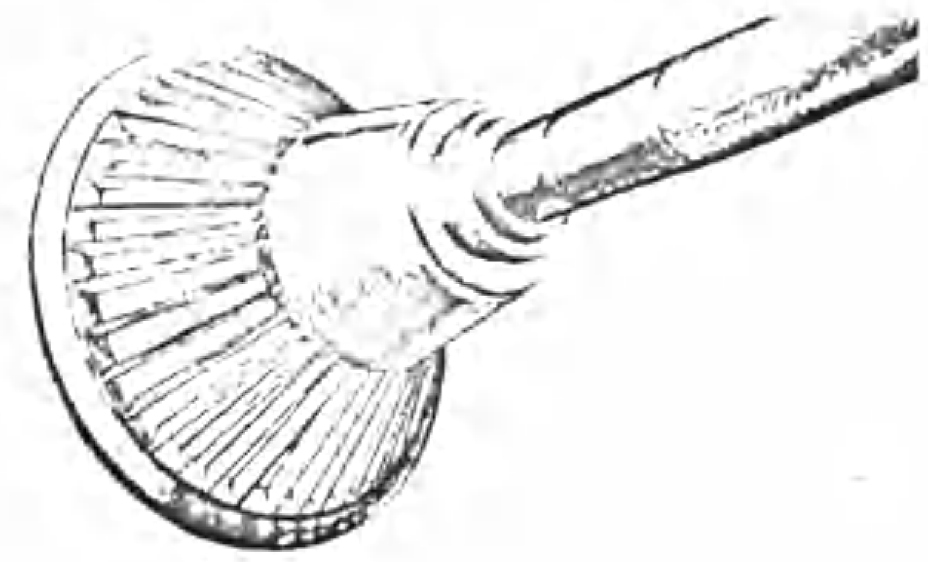
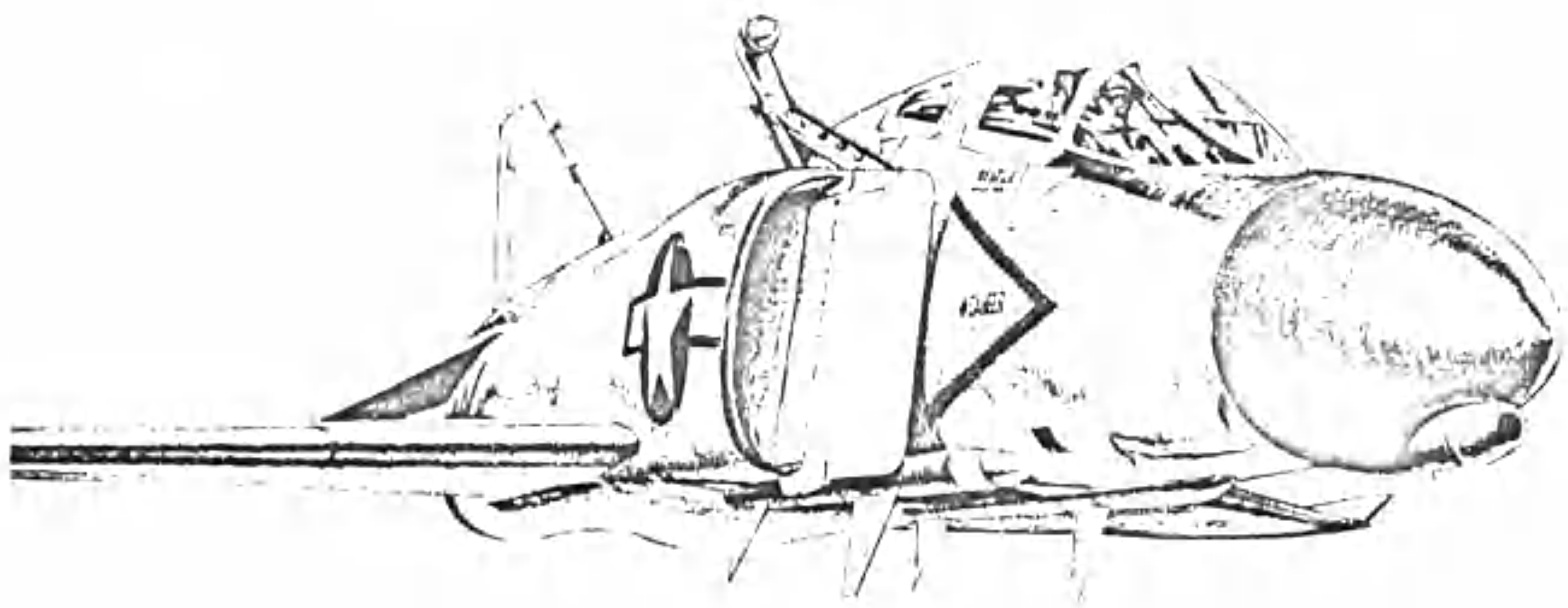


Figure 6-6

SECTION VII SYSTEMS OPERATION





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WHEEL BRAKE SYSTEM

The normal brake system utilizes utility hydraulic system pressure to assist toe action in stopping the airplane. Refer to Section I of this handbook for a description of the brake system. The system has three emergency features; these are:

1. In the event of loss of the utility hydraulic system pressure, a 25 cubic inch hydraulic accumulator provides sufficient pressure for approximately 10 maximum effort applications of the normal brake system.
2. If the entire brake hydraulic system is lost, an independent pneumatic actuation system is provided. Fed by a 100 cubic inch air bottle, up to twenty maximum effort retardations may be made by means of the handle just inboard of the right console.
3. Loss of all hydraulic pressure need not mean loss of hydraulic braking. The brake control valves will act as master cylinders in a conventional non-power system as long as integrity has been maintained between the control valves and the wheel brakes. Pilot effort in this manual operation will be capable of securing the airplane in deck rolls of up to 8°. Although it has not been determined by test flight, pilot assumption is that the manual brake system should probably be capable of successfully stopping the airplane on a typical jet runway provided the drag chute is also used. The manual braking feature is selective and may be used for differential brake steering, while the emergency pneumatic brake system is deployed in stopping the airplane.

WHEEL BRAKE OPERATION

The brakes are conventionally operated by toe action on the rudder pedals. This action meters utility hydraulic pressure to force the brake disks together. Pedal pressure felt by the pilot is proportionate to braking force applied, and braking effort is fully definable from the cockpit. The pilot is capable of locking the brakes by both normal and emergency braking systems. Caution must be exercised in overbraking, since a fully locked wheel usually offers LESS retardation than very light normal braking. If one wheel is locked during application of the brakes, there is a very definite tendency for the airplane to turn away from that wheel and further application of brake pressure will offer no corrective action. This produces a rapidly decreasing coefficient of friction between the skidding tire and the runway, while the coefficient of friction between the other tire and the runway remains near optimum for braking effectiveness. It is, therefore, apparent that a wheel once locked will never free itself until brake pressure to that wheel is reduced sufficiently to permit the wheel to rotate. It has been found that optimum braking occurs when the wheel is in a slight skid. The wheel continues to rotate, but at a speed of approximately 80 to 85 percent of its normal free rolling rotational speed. Increasing the rolling skid above approximately 15 to 20 percent will only decrease the braking effectiveness. Since no anti-skid system is fitted, recognition of maximum braking force is strictly a matter of pilot sensitivity. As with other examples of operating "on the limit", the only sure way of determining maximum braking effort is to exceed it. Since this is seldom a desirable technique, the pilot should attempt to mentally catalogue his body response to normal braking, in order to more readily recognize the maximum, if an emergency should require it. For all conditions, normal and emergency, the most desirable braking technique is a single, smooth application of the brakes with a constantly increasing pedal pressure (to just below the skid point) as the airplane decelerates. In the event of a reduction in retardation being felt while exercising maximum braking, pedal force must be fully released in order to

EXTERNAL POWER REQUIREMENTS

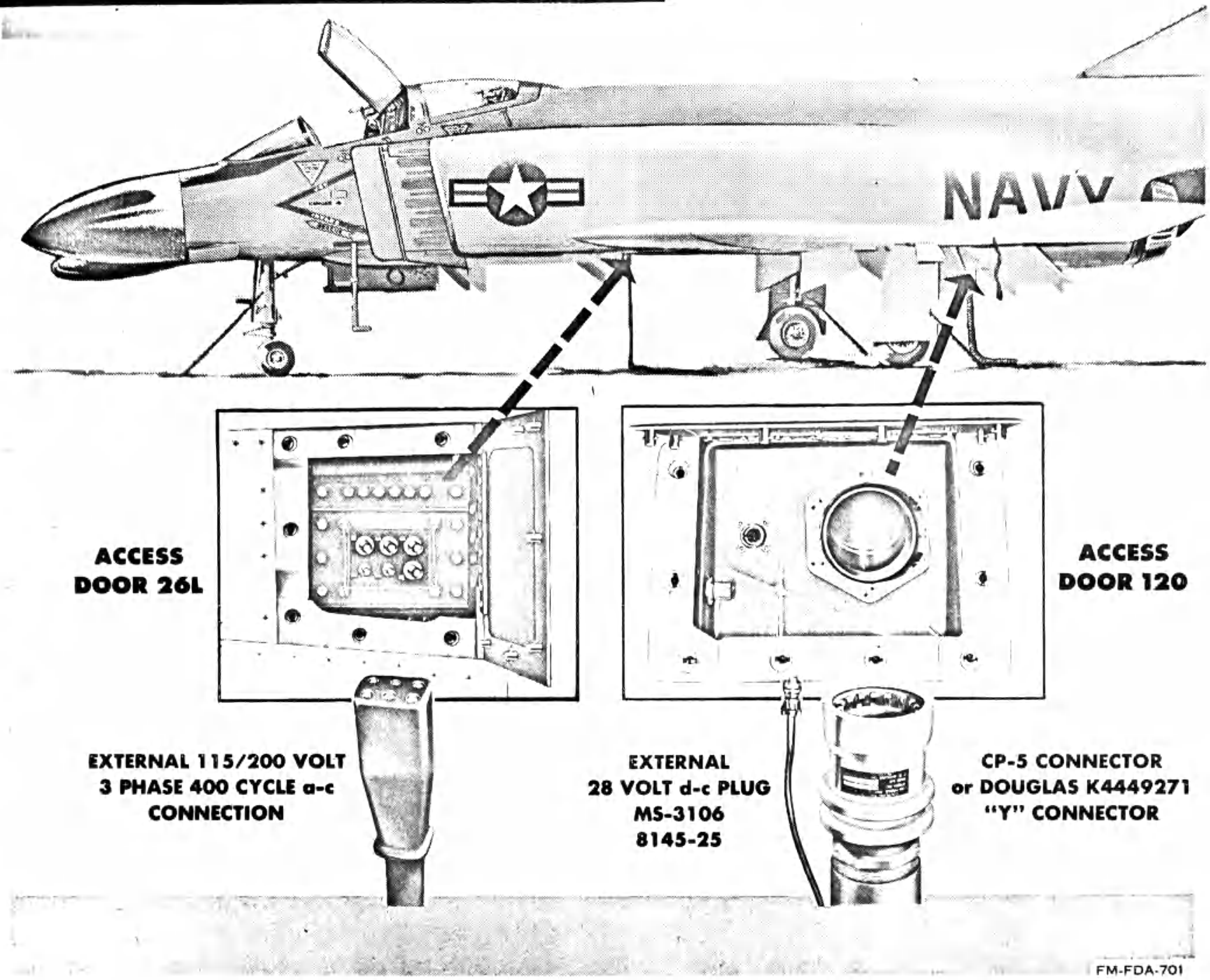


Figure 7-1

allow the skidding wheels to regain full rolling speed before further application of brakes.

CAUTION

Do not pump the brakes at any time, since this action will only reduce the amount of constant pressure available at the brakes and will tend to store heat energy within the brake assembly.

Note

Rough runways will tend to emphasize the skip or bounce characteristics of the airplane which are caused by relatively stiff struts. In order to preclude the possibility of locking a wheel while momentarily off the ground, use light braking until the airplane is solidly on the ground and all skipping has ceased.

If it is suspected that the brakes have been used excessively, and are in a heated condition, the airplane should not be taxied into a crowded parking area. Peak temperatures occur in the wheel brake assembly from 5 to 15 minutes after maximum braking and is further transferred to the tire. To prevent brake fire and possible tire explosion, the specified procedures for cooling brakes should be followed. It is recommended that a minimum of 15 minutes elapse between landings where the landing gear remains extended in the slip stream, and a minimum of 30 minutes between landings where the landing gear has been retracted to allow sufficient time for cooling between brake applications. Additional time should be allowed for cooling if brakes are used for steering, crosswind taxiing operation, or a series of landings.

EXTERNAL POWER REQUIREMENTS

The turbine impingement starting for the F4H-1 airplane requires external power for 28 volt d-c, 115/200 volt 3 phase, 400 cycle a-c and a compressed air source which will supply a minimum of 110 lb./min. of air at 50 pounds pressure at an ambient temperature of 60°F. The external a-c power receptacle is located through access door No. 26L. The compressed air and d-c power receptacles are located through access door No. 120. The auxiliary power units that are most frequently used are the BUWEPS CP-5 (RCPP-105) (1 unit) or the Douglas GTC 85 (2 units).

BUWEPS CP-5 UNIT

The BUWEPS CP-5 (RCPP-105) unit supplies 115/200 volt a-c, 28 volt d-c, and sufficient compressed air to meet all external power requirements from a single unit.

DOUGLAS GTC 85 UNITS

The Douglas GTC 85 units must be connected in parallel by a "y" connector (Douglas K4449271 or similar fitting) to supply sufficient air for turbine impingement starting. 115/200 volt a-c power must be supplied by a separate source. 28 volt d-c power must be applied in order to operate the air selector valves. An MS3106-A14S-2S plug must be used for the d-c power receptacle. 28 volt d-c power can be supplied by an external power unit or by a 24 volt battery.

CAUTION

The use of two GTC-85 units in conjunction with a Douglas "Y" fitting is considered an emergency starting method. This is due to the possibility of higher EGT's longer starts or hang starts occuring with the use of the two GTC 85's.

EMERGENCY SYSTEMS CHECK PROCEDURES

EMERGENCY LANDING GEAR CHECK

1. Pull landing gear circuit breaker.
2. Pull landing gear handle DOWN and then AFT to the emergency position.
3. If landing gear is down and locked, land with the control in the emergency position.
4. If landing gear does not lock down by emergency extension, return the control to the normal DOWN position.
5. Allow four minutes for the high pressure air to bleed off then reset the landing gear circuit breaker.
6. When landing gear extends and locks normally, land as soon as practicable. DO NOT cycle the landing gear.

Note

Any pneumatic extension of the landing gear shall be logged on the yellow sheet (OPNAV FORM 3760-2).

EMERGENCY WING FLAP EXTENSION CHECK

1. Pull wing flaps circuit breaker.
2. Pull emergency wing flaps handle to the DOWN position.
3. If the wing flaps extend, land with the emergency wing flaps handle in the down position.
4. If the wing flaps do not extend by emergency extension, return the emergency extension lever to the up position.
5. Allow four minutes for the high pressure air to bleed off.
6. Place the normal wing flap switch to the DOWN position and reset wing flap circuit breaker.
7. When wing flaps extend normally, land as soon as practicable. DO NOT cycle the wing flaps.

Note

Any pneumatic extension of the wing flaps shall be logged on the yellow sheet (OPNAV FORM 3760-2).

EMERGENCY WIND DRIVEN GENERATOR

1. Generator control switches - OFF.
2. Emergency hydraulic pump lever - PUSH DOWN

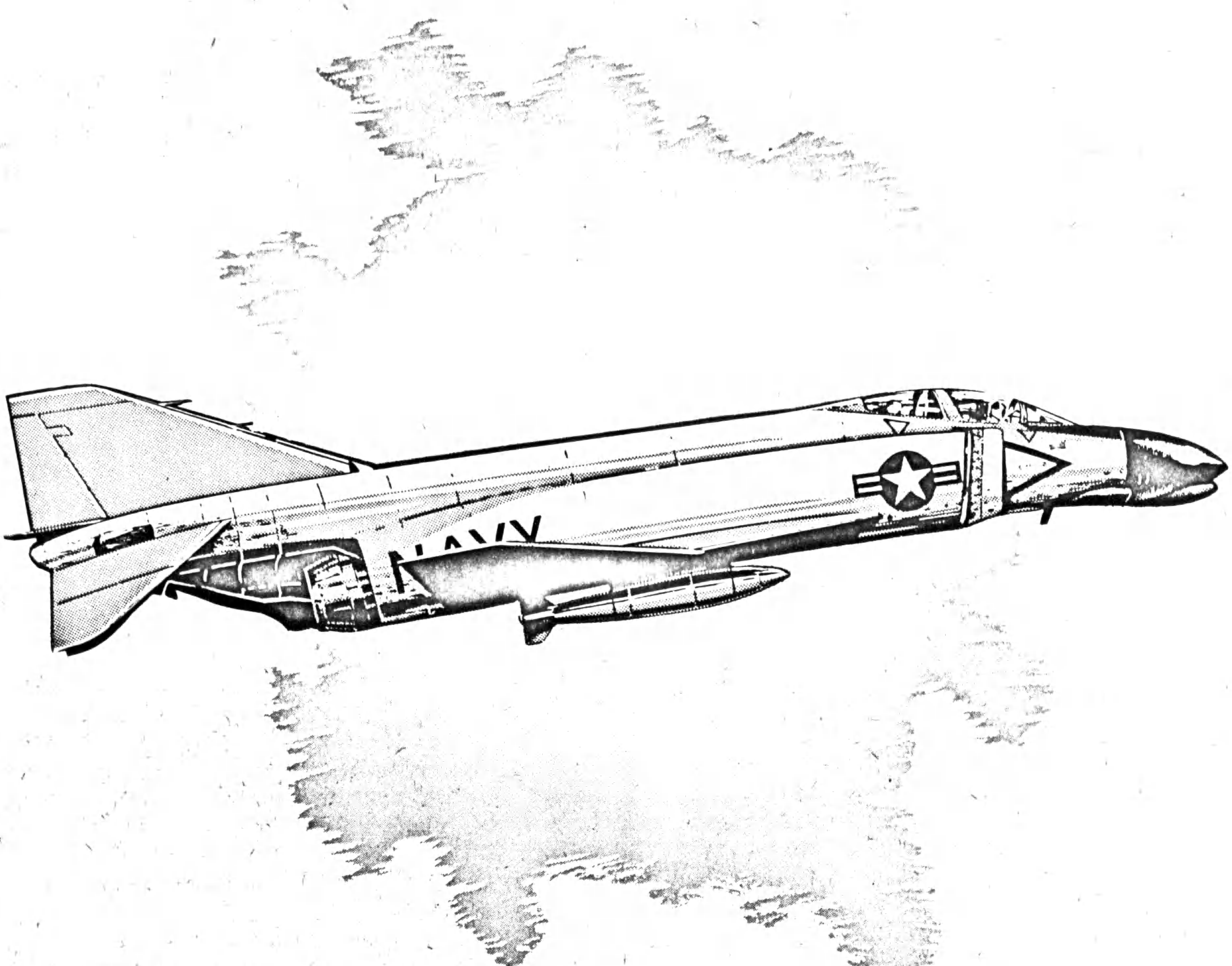
CAUTION

Do not retract wind-driven turbine until pneumatic system pressure gage reads 2,000 psi. Attempted retraction at pressures lower than 2,000 psi may cause wind-driven turbine to close on turbine doors, preventing retraction.

3. Check items on essential a-c 115/200 volt bus (ADI OFF flag to operating position).
4. Generator control switches - ON
5. When electrical system is again operating normally, emergency hydraulic pump lever - PULL UP to retract.

Note

- Emergency power control hydraulic pressure cannot be checked when the power control systems are operating normally.
- Emergency wind driven generator operational time shall be logged on the yellow sheet (OPNAV FORM 3760-2).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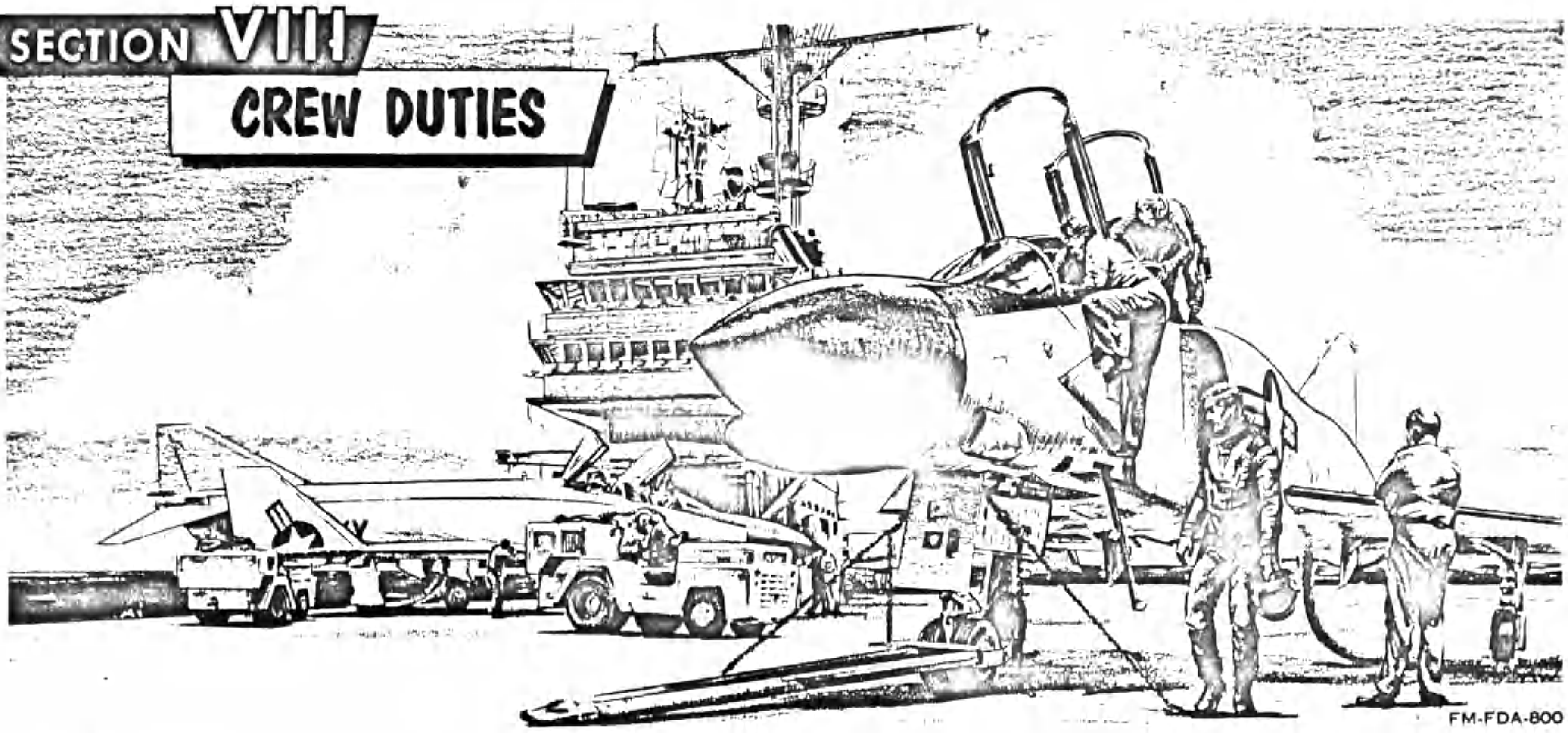
SECTION VIII CREW DUTIES



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PILOT'S DUTIES

The primary duties of the pilot are to operate the flight and landing controls, together with all normal and emergency controls related to aircraft operation; he also has command of all direct armament firing controls. The procedures pertaining to the operation of the applicable equipment necessary for the pilot to carry out his assigned duties is discussed in Sections II, III and IV of this publication.

RADAR OBSERVER'S DUTIES

The radar observer's primary function is to operate the airborne missile control system, the counter-electronic countermeasure equipment and the navigation equipment. The duties connected with the operation of the above equipment are as follows:

1. Operation of the airborne missile control system.
   a. Checkout and scope adjustments.
   b. Monitor scope displays (search).
   c. Radar and/or infrared mode selection.
   d. Interpret scope displays.
   e. Antenna slewing and radar lock-on.
2. Operation of the counter-electronic countermeasure equipment.
   a. Reduce clutter (FTC and gain).
   b. Re-establish lock-on.
   c. Employ proper AI radar and/or infrared mode.
   d. Discriminate against IR or radar decoys.
3. Operation of the navigation equipment.
   a. Provide dead-reckoning range and bearing to target or base.
   b. Provide present position.
   c. Radar ground mapping.
   d. Ground track.
   e. Provide the pilot with target azimuth, elevation and range for visual sighting and identification.
   f. Assist in communications and maintain radar surveillance during instrument letdowns and GCA approaches.
   g. Assist in avoiding turbulent air masses.

Secondarily, the radar observer should assist the pilot in any way possible so that lengthy procedures may be quickly and accurately accomplished, resulting in a well executed mission.

1. Operation of the counter-electronic countermeasure equipment.
   a. Employ ADF when the communication selection of CNI is under his command.
2. Operation of the communications equipment.
   a. Monitor UHF vectoring information.
   b. Relieve pilot of some communication functions.
   c. Give verbal identifications.
3. Operation of buddy tank control panel.

RADAR OBSERVER'S EXTERIOR INSPECTION

At the discretion of the pilot, the radar observer will assist in making the exterior inspection (figure 2-2).

BEFORE ENTERING COCKPIT

1. Harness assembly - CHECK
   Check emergency harness release handle down. Check if pin is installed at reel assembly. Check pins that secure lap belt and survival kit to seat are in place.
2. Composite disconnect lower block - CHECK
   Check lower block locking indicator (yellow metal tab) on bottom of lower block is tight.
3. Canopy interlock block - CHECK
   Check canopy interlock block in place, and interlock line secured to canopy.
4. Drogue withdrawal line - CHECK
   Check drogue withdrawal line (in wire braid sleeve) passes over and lays on top of all other lines.
5. Scissor shackle tie-down thread - CHECK
   Check that scissor shackle tie-down thread passes below the flap securing pin, and does NOT pass through the loop of the drogue flap.
6. Banana links - CHECK
   Check links attached to sear pin.
7. Seat latch nut - CHECK
   Check seat latch nut finger tight and safety wire.
8. Parachute withdrawal line quick disconnect - CHECK
   Check connection of parachute withdrawal line quick disconnect.
9. Seat wedge pack (headrest) - CHECK
   Check seat wedge pack (headrest) has approximately one (1) inch side play.
10. Drogue gun trip rod - CHECK
    Check drogue gun trip rod on left side of seat secure to airplane structure.
11. Safety banner - REMOVED
    Check safety banner and all five safety pins - face curtain, ejection gun, canopy initiator, (seat mounted) drogue gun, canopy initiator (bulkhead mounted) - and both dust covers (composite disconnect, exhaust port), are removed.

WARNING

- When removing or checking for the removal of the face curtain safety pin, make sure that the safety pin shank has been removed from the hole. The safety pin collar has been known to separate from the pin shank upon attempted safety pin removal, leaving the pin shank in the hole and the face curtain safetied.
- Do NOT pull down on the face curtain ejection handle. Seat and canopy ejection systems are fully armed when safety pins are removed.

12. Alternate ejection handle safety guard - CHECK
    Check alternate ejection handle safety guard in front of the seat bucket is in the vertical (up) position.
13. Survival kit oxygen gage - CHECK
    Check survival kit oxygen gage within limits.

RADAR OBSERVER'S INTERIOR INSPECTION

1. Time delay mechanism trip rod - CHECK
   Check time delay mechanism trip rod on right side of seat secure to airplane structure.
2. Composite disconnect intermediate block - CHECK
   Check yellow tubing projects from oxygen connector on top of intermediate block.
3. Composite disconnect - INSERTED and LOCKED
   Check upper block properly inserted and locked into intermediate block by exerting an upward pull on block assembly after composite disconnect release knob is locked to cable housing.

Note

The composite disconnect should be carefully inserted with a downward force parallel to the seat ejection plane.

CONTINUED ON NEXT PAGE

RADAR OBSERVER'S INTERIOR INSPECTION CONTINUED

4. Pressure suit lines - CONNECTED
   Connect pressure suit exhaust hose to receptacle on survival kit.

WARNING

When the pressure suit is being worn without an anti-G garment, the anti-G hose must not be connected and the corresponding port on the pressure suit must be capped. Explosive decompression will result upon ejection if the anti-G hose is connected. In the event of ejection over water or ditching, the water tight integrity of the pressure suit will be nullified.

5. Pressure suit vent air valve - ON
   Adjust air flow as desired.
6. Oxygen supply lever - ON
   Check for normal flow.
7. Leg restraint lines - CONNECTED
   Pass leg line through garters and plug into seat pan.

WARNING

It is imperative that leg restraint system be hooked up at all times during flight to ensure legs are pulled aft upon ejection. This will prevent leg injury and enhance seat stability by preventing legs from flailing following ejection. An unhooked leg restraint system necessitates pulling legs aft against seat to preclude hitting instrument panel. This action will cause spine to flex and will increase the possibility of spinal injury during ejection.

8. Harnessing - FASTENED
   Check parachute riser-shoulder harness release fittings fastened to main sling buckles. Check lap belt release fittings fastened to lap belt buckles. Check locking reel operation.

CAUTION

Make sure that the automatic harness release assembly is securely to the seat. The pins must be in their proper receptacles, one pin on each side of the bucket seat and one pin on the harness locking reel assembly. The manual harness release handle must be down.

9. Liquid oxygen gage - CHECK
   Check oxygen quantity is sufficient for flight.
10. Anti-G suit pressure regulating knob - AS DESIRED
    Adjust G-suit pressure on HIGH or LO as desired.
11. Buddy tank control panel - CHECK
    a. Fuel dump switch - NORMAL
    b. Hose jettison switch - NORMAL
    c. Drogue control switch - OFF
    d. Tank lights switch - NORMAL
12. CNI panel - CHECK (Airplanes 145313b, 146817c thru 148275e)
    a. Communication function selector switch - STANDBY
    b. TACAN function selector switch - STANDBY
13. Navigation computer power switch - OFF

CONTINUED ON NEXT PAGE

RADAR OBSERVER'S INTERIOR INSPECTION CONTINUED

14. Radar set control panel - CHECK (Airplanes 145313b, 146817c thru 148275e)
    a. Radar power switch - OFF
    b. Mode switch - RDR
    c. Clutter switch - NOR
    d. Display switch - WIDE
    e. Gyro switch - NOR
    f. Polar switch - LIN
    g. Test switch - 0
    h. Range switch - AI
15. CNI panel - CHECK (Airplanes 145312b and 145315b thru 145317b)
    a. Communication function selector switch - STANDBY
    b. TACAN function selector switch - STANDBY
16. Intercom panel - CHECK
    a. Function selector switch - NOR
    b. Volume control knob - AS DESIRED
17. Communication antenna selector switch - AS REQUIRED
18. Equipment lights - OFF
19. Instrument lights - OFF
20. Warning lights test switch - OFF
21. Cockpit floodlights switch - OFF
22. Altimeter - SET
23. Clock - SET
24. Gyro cutout switch - NORMAL (Airplanes 146817c thru 148275e upon incorporation of ASC 46A)
25. Radar set control panel - CHECK (Airplanes 145312b and 145315b thru 145317b)
    a. Radar power switch - OFF
    b. Pulse switch - WIDE
    c. Receiver tune control - AFC
    d. Manual V_c control - 0
    e. Gyro erect switch - NOR
    f. Missile switch - NOR
    g. Range switch - 100
    h. Dot balance switch - NOR
    i. FTC switch - OUT
    j. Scan switch - WIDE
    k. Display switch - RDR (Airplanes 145312b and 145317b only)
    l. Mode switch - AI
26. Radar test switch - OFF

WITH EXTERNAL POWER CONNECTED-

27. Seat - ADJUST
28. Cockpit lights - CHECK
    If conditions warrant, check operation of all cockpit lights and adjust desired brilliance.
29. Essential d-c bus test switch - DEPRESS
    Depress test switch and note illumination of essential d-c bus test light indicating the right transformer-rectifier is inoperative. Hold switch depressed and proceed with next step.
30. Warning lights - CHECK
    Depress warning lights test switch and check that all warning lights are illuminated. This indicates that the warning light circuits are completed and also indicates that the left transformer-rectifier and bus tie current limiter are operative.
31. Interphone system - CHECK
    Check the interphone system between pilot and ground crew.
32. Navigation computer power switch - STBY
33. Radar power switch - OFF

BEFORE TAXIING

1. Taxi area - CLEAR
   Check with pilot that taxi area and jet blast area is clear of personnel and ground equipment.
2. Wing spread - CHECK
   After wings have spread, check that wing pin unlocked warning lights are extinguished.
3. Alternate ejection handle safety guard - CHECK
   Check safety guard for freedom of movement.
4. Radios - CHECK
   Perform radio and ADF preflights. Set radios on proper channels.

TAXIING

1. Canopy - CLOSE
   Before closing canopy, check that canopy sill is clear. Check that canopy unlocked warning light is extinguished after closing.

BEFORE TAKE-OFF

1. Integrated harness - CHECK
   Check that integrated harness is properly secured to the parachute.
2. Inertia reel handle - LOCKED
3. Personal equipment leads - CHECK
   Check composite disconnect in place and secure. Check all equipment leads properly attached and that they create no obstruction to movement.
4. Canopies - CLOSED
   Check canopies closed and canopy unlocked warning light extinguished.
5. Warning lights - CHECK
   Check all warning lights extinguished.
6. Alternate ejection handle safety guard - ROTATE CLEAR
   Rotate safety guard to the horizontal position.

AFTER TAKE-OFF

1. Radar power switch - ON
2. Altimeter - CHECK
   Check altimeter against pilot's indication after level-off.

BEFORE LANDING

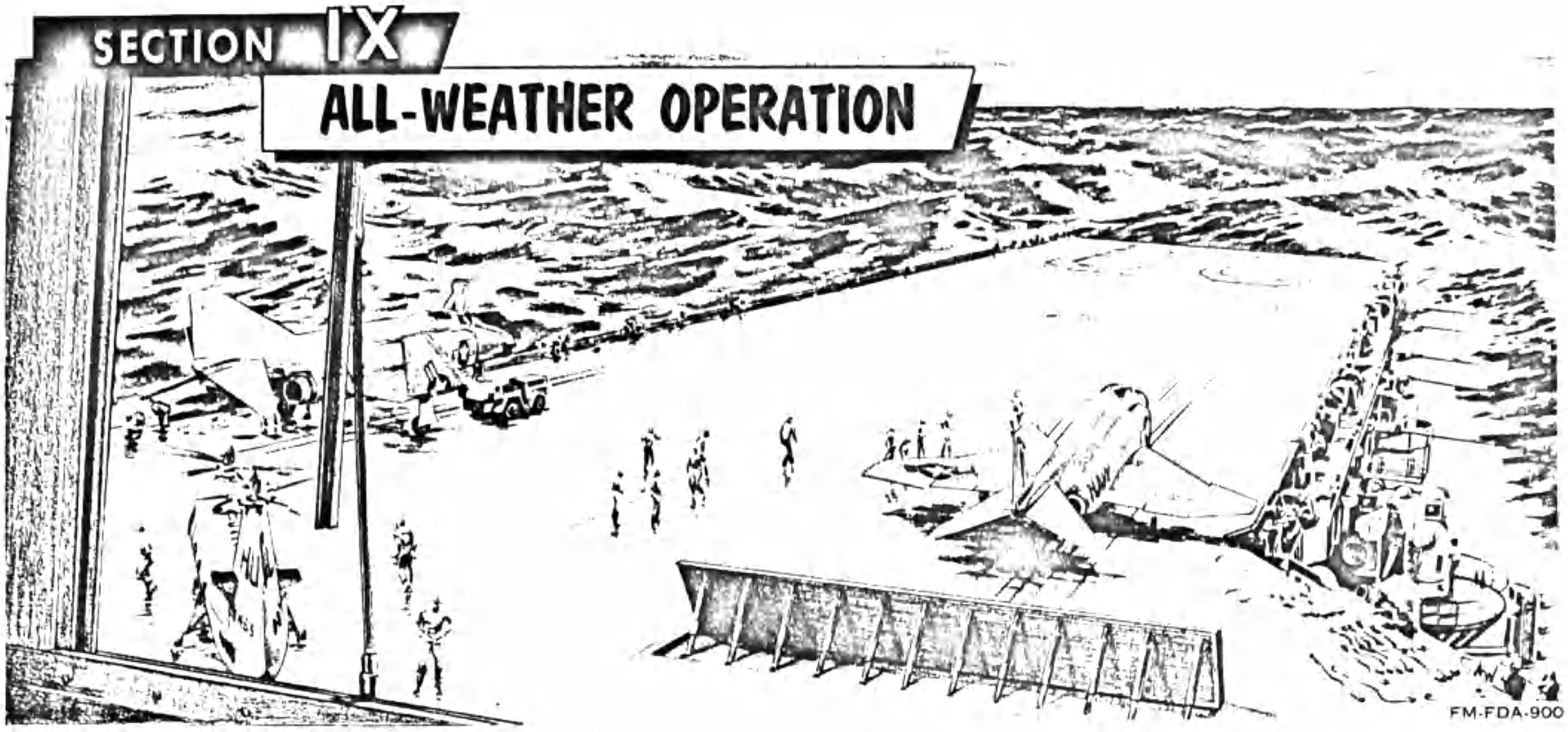
1. Radar power switch - OFF
2. Integrated harness - CHECK
   Check integrated harness tightened.
3. Inertia reel handle - LOCKED

BEFORE ENGINE SHUTDOWN

1. Radar power switch - OFF
   Turn radar power switch OFF prior to engine shutdown to prevent damage to radar equipment from low voltage during engine run down.

BEFORE LEAVING AIRPLANE

1. All switches, levers, and personal equipment - OFF or DISCONNECTED
2. Install plastic guard on composite disconnect.
3. Alternate ejection handle safety guard - UP
   Rotate safety guard to the vertical position.
4. Safety banner - INSTALLED
   Install the canopy and ejection seat safety pins and dust covers.
5. Flight forms - COMPLETED



FM-FDA-900

This section provides information for operation during conditions of instrument flight, flight in turbulent air, various penetration/approach procedures, and extreme temperature conditions. These are procedures that differ from, or are in addition to, those contained in the normal operating procedures covered in Section II.

INSTRUMENT FLIGHT PROCEDURES

The F4H-1 is an all-weather airplane that is designed to perform operational missions in all extremes of weather. Rapid acceleration rates and high pitch angles during climb, of necessity dictate some modification of standard instrument procedures.

BEFORE ENTERING AIRPLANE

On instrument flights, delays in departure and descent and low climb rates to altitude are often required in high density control areas. Those factors make fuel consumption and flight endurance critical and demand that all instrument flights be carefully planned and consideration given to the additional time and fuel which may be required.

BEFORE TAXI

1. Determine that UHF COMM. channels are preset for tower control, departure control, FAA Frequencies, etc.
2. Check UHF NAV for proper channel operation.

BEFORE INSTRUMENT TAKE-OFF

If a climb-out through precipitation or clouds is anticipated perform steps 1 and 2.

1. Engine anti-ice switch - DE-ICE
   While at IDLE rpm, position switch to DE-ICE and check for a rise in EGT of approximately 10°C and a slight increase in fuel flow.
2. Pitot heat switch - ON
3. ADI wings symbol - 1° wings above horizon.
4. Mode-Bearing Distance selector switch - AS REQUIRED
5. HSI heading set and course set - AS REQUIRED
6. Stab Aug - ENGAGE

INSTRUMENT TAKE-OFF

Instrument take-off should be made with Military rather than Maximum thrust unless the urgency of the mission makes this a necessity. Maximum thrust take-offs at normal gross weight require rapid initial changes in attitude and extremely high pitch angles, and also increase the possibility of accelerating past the gear and flaps down limit airspeed.

TAKE-OFF AND TRANSITION

1. Align airplane with runway and check the ADI and HSI headings.
2. Use nose wheel steering until rudders become effective (approximately 70 knots CAS).
3. At 125-130 knots CAS rotate airplane to 5° nose up (nose strut extend).
4. At 140-145 knots CAS rotate to 10°-12° nose up (lift-off).
5. Flaps up at 300 feet or 200 knots CAS (maintaining 10° to 12° nose up attitude.

INSTRUMENT CLIMB (MIL POWER)

A simplified climb schedule as outlined below may be used with minimum sacrifices in fuel consumption and climb rates. Turns should be kept to a minimum during climb-out due to the difficulty in determining bank angles and rates of turn while at high pitch angles. After becoming safely airborne with gear and flaps retracted climb out as follows:

1. Maintain 10° to 12° nose-up attitude until reaching 400 knots CAS.
2. Vary pitch attitudes as necessary to maintain 400 knots CAS until reaching Mach .9.
3. Vary pitch attitude as necessary to maintain Mach .9 to cruise altitude.

Upon reaching clear air:

4. Engine anti-icing switch - NORMAL
5. Pitot heat switch - OFF

INSTRUMENT CLIMB (MAX POWER)

1. Maintain 10° to 12° nose up attitude until reaching 250 knots CAS.
2. At 250 knots CAS smoothly rotate to 20°-25° nose up attitude until reaching Mach .92
3. Vary pitch attitude as necessary to maintain Mach .92 until reaching cruise altitude.

Upon reaching clear air:

4. Engine anti-icing switch - NORMAL
5. Pitot heat switch - OFF

INSTRUMENT CRUISE FLIGHT

After leveling off from climb:

1. Establish cruise Mach number.
2. Adjust power.
3. Align ADI wings symbols for level flight.
4. Antenna selector switch - AS REQUIRED

HOLDING/LOITER

Holding patterns or loitering flight may be flown at most altitudes at 265 knots CAS. Single needle width turns (approximately 30° angle of bank) are recommended.

INSTRUMENT DESCENT

Three to five minutes prior to making a descent:

1. Defog/Foot heat lever - DEFOG
2. Cabin Temperature control - Maximum heat compatible with comfort.

If descent through precipitation or clouds is anticipated:

3. Engine anti-ice switch - DE-ICE
4. Pitot heat switch - ON
5. Speed brakes - OUT
6. Penetration airspeed - 250 knots CAS at 78% rpm.

GCA (PAR) APPROACHES

1. Descend to GCA pick up altitude and transition to landing configuration when directed.
2. Maintain 150-160 knots CAS (approximately 88-90% rpm).

When directed to commence descent:

3. Retard power to approximately 82-84% rpm.
4. Allow airplane to decrease speed to approximately 145 knots CAS.
5. Adjust power as necessary to maintain 600-800 feet per minute rate of descent, or, as directed.

A straight-in TACAN penetration followed by a GCA final will require approximately 500-800 pounds of fuel. A missed approach followed by a second GCA requires approximately 1000 pounds of fuel.

NIGHT FLYING

The following checks and information are in addition to those given for normal instrument flight.

ON ENTERING AIRPLANE

1. Interior lighting - ADJUST

Note

For night catapult launches, place instrument panel emergency floodlights on DIM to provide instrument panel lighting if normal instrument lights fail.

2. Emergency floodlights - CHECK
3. Navigation lights and exterior lights - CHECK
4. Flashlight - CHECK

DURING FLIGHT

Adjust interior and instrument lights to desired brilliance.

VFR conditions:

1. Exterior lights - FLASH
   When in VFR conditions, exterior lights should be on FLASH except during formation flight.

IFR conditions:

1. Exterior lights - STEADY
   During night or instrument conditions, exterior lights should be on STEADY due to the vertigo inducing effect on flashing light reflections from surrounding clouds.

CAUTION

Due to the unreliability of exterior wing and tail lights, the anti-collision light should be on during all night VFR conditions except formation flight (flash steady switch - FLASH).

COLD WEATHER PROCEDURES

Normal operating procedures as outlined in Section II should be adhered to with the following additions and exceptions:

1. Check entire airplane for freedom from frost, snow and ice; see that all light snow, ice or frost is removed.

Note

Do not chip or scrape away ice as damage to airplane may result.

WARNING

Snow, ice and frost collections on the airplane surface are a major flight hazard. The result of this condition is loss of lift and increased stall speeds.

2. Shock struts and actuating cylinders free of ice and dirt.
3. Fuel drain cocks free of ice and drain condensate.
4. Pitot tube, fuel vents, all ice or dirt removed.
5. All exterior covers removed.

WARM-UP AND GROUND CHECK

Turn cockpit air-conditioning system ON as soon as possible after starting.

WARNING

- Be sure all instruments are allowed adequate warm-up period and are operating normally before take-off.
- If any abnormal sounds or noises are present during starting, discontinue starting and apply intake duct preheating for 10 to 15 minutes.

TAXIING

1. Avoid taxiing in deep or rutted snow since frozen brakes will likely result.
2. Increase space between airplanes while taxiing at sub-freezing temperatures, to insure safe stopping distance and to prevent icing of airplane surfaces by melted snow and ice on the jet blast of a preceding airplane.

BEFORE TAKE-OFF

1. Make normal power run-up.

Note

The thrust developed by the engine in low temperature is noticeably greater and brake demands will be greater to hold position.

CAUTION

If engine icing conditions are encountered or expected, place the engine anti-ice switch in the DE-ICE position.

2. Pitot heat - ON

LANDING

If snow and ice tires are installed, use brakes intermittently and carefully to keep tread from filling and glazing.

Note

Hard braking on ice or wet runways will result in dangerous skidding or fishtailing.

AFTER LANDING

1. Pitot heat - OFF

BEFORE LEAVING AIRPLANE

1. Leave canopy partly open, unless weather prevents, to permit circulation. This helps prevent canopy cracking from differential cooling and decreases windshield and canopy frosting.
2. Check all protective covers installed.

HOT WEATHER PROCEDURES

Normal operating procedures as outlined in Section II should be adhered to with the following additions and exceptions:

CAUTION

Do not attempt take-off or engine operation in a sand storm or dust storm if avoidable. Park aircraft crosswind and shut down engine to prevent sand or dirt from damaging engine.

TAKE-OFF

The required take-off distances are increased by temperature increase. Check required take-off distances charts, figures A2-1 and A2-2.

BEFORE LEAVING AIRPLANE

1. Leave canopy open to permit circulation.
2. Check all protective covers installed.

TURBULENCE AND THUNDERSTORMS

Intentional flight through thunderstorms should be avoided, unless the urgency of the mission precludes a deviation from course, due to the high probability of damage to the airframe and components by impact ice, hail, and lightning. The radar provides an excellent means of navigating between or around storm cells and the airplane is capable of climbing over the top of small and moderately developed thunderstorms.

PENETRATION

If necessary to penetrate, the basic structure of the airplane is capable of withstanding the accelerations and gust loadings associated with the largest thunderstorms at subsonic airspeed. Supersonic thunderstorm penetrations have not been investigated to date. The airplane is exceptionally stable and comparatively easy to control in the severe turbulence; however, the effects of turbulence becomes noticeably more abrupt and uncomfortable at airspeeds above Mach .9. The airplane is not displaced significantly from the intended flight path and desired heading. Altitude, airspeed, and altitude can be maintained with reasonable accuracy.

PENETRATION AIRSPEED

The optimum thunderstorm penetration speeds, based on pilot comfort, control, and engine considerations are as follows:

1. Low and medium altitudes - 300-350 knots CAS OR Mach .9, whichever is less.
2. High altitudes (above 35,000 feet) - .9 Mach or 270 knots CAS, whichever is more.

Note

Optimum thunderstorm penetration airspeeds are a compromise between pilot comfort, controllability, structural stress (due to gust loads and impact precipitation), and engine inlet air distortion. At high airspeeds pilot discomfort and structural stress are greater. At slow speeds controllability is somewhat sacrificed and inlet airflow distortion due to turbulence may induce compressor stalls and/or engine flameout.

APPROACHING THE STORM

If storm cannot be seen, it may be located by use of radar, radio compass swing, and crash static. Establish the recommended penetration airspeed and perform or check the following:

1. Adjust throttle to maintain desired penetration speed.
2. Pitot heat switch - ON
3. Engine anti-icing switch - DE-ICE
4. Autopilot - OFF
5. Lower seat

If night penetration:

6. Daylight floodlights - ON
7. Instrument lights - FULL BRIGHT
8. Console lights - FULL BRIGHT

CAUTION

Do not become overly intent on "topping" the storm or allow airspeed to decrease below the minimum recommended for safe penetration. Flight through a thunderstorm at the proper airspeed is much more advantageous than "floundering" into the storm at a dangerously slow airspeed while attempting to reach the top.

IN THE STORM

1. Maintain a normal instrument scan with added emphasis on the attitude gyro (ADI). Attempt to maintain constant pitch attitude, and accept small altitude and airspeed fluctuations.
2. Adjust the throttle as necessary to maintain airspeed within the recommended range.

Note

In heavy precipitation a reduction in engine RPM will be necessary due to the increased thrust resulting from water ingestion.

CAUTION

- If compressor stall or engine stagnation develop, retard throttle to idle momentarily, then attempt to regain normal engine operation by advancing the throttle. If the stall persists, shut down the engine and attempt a relight. If engine remains stagnated at reduced power, and EGT is within limits, maintain reduced power until clear of the thunderstorm.
- The angle-of-attack probe may become distorted by impact ice and/or hail, which may result in actuation of the rudder shaking and stall warning device.

ICE AND RAIN

The possibility of engine and/or airframe icing is always present when the airplane is operating under instrument conditions. Icing is most likely to occur when take-offs must be made into low clouds with temperature at or near freezing. Normal flight operations are carried on above the serious icing levels, and the airplane's high performance capabilities will usually enable the pilot to move out of the dangerous areas quickly. When an icing condition is encountered immediate action should be taken to avoid further accumulation by changing altitude and/or course and increasing the rate of climb or airspeed.

BEFORE TAXIING

Before taxiing prior to severe weather flight, the engine anti-ice system should be checked as follows:

1. Engine rpm - IDLE
2. Anti-ice switch - DE-ICE
3. EGT - CHECK
   Check for rise in EGT of approximately 10°C.
4. Fuel flow - CHECK
   Check for slight increase in fuel flow.

IN-FLIGHT

Flight through ice and/or rain requires no special pilot technique, however, certain aircraft systems do require particular attention when ice and/or rain is encountered. These systems are engine anti-ice, windshield rain removal, longitudinal feel, and CADC.

Engine Anti-Ice System

There is no indication during flight of engine icing such as decreased power or reduction in fuel flow. Therefore, the engine anti-ice system should be utilized whenever visible precipitation is encountered and inlet duct temperatures are suspected to be near or below freezing. Anti-ice system operation can be noted by an increase in EGT and fuel flow when the system is actuated. After clearing all precipitation and clouds, it must be remembered to turn the anti-ice switch off. Unnecessary use of engine anti-ice air produces the following adverse effects:

a. Slightly decreased range.
b. Slightly decreased performance.
c. Probable compressor front end damage at high supersonic speeds.
d. Probable decrease in compressor stall margin.

When icing conditions are encountered:

1. Engine anti-ice switch - DE-ICE
2. Pitot heat switch - ON

After clearing all precipitation and clouds:

1. Engine anti-ice switch - OFF
2. Pitot heat switch - OFF

Windshield Rain Removal

Windshield rain removal has been found to be marginal when flying through any precipitation regardless of flap positions or power settings. Rain removal during ground taxiing operations is satisfactory. The windshield rain removal system is somewhat effective in de-icing the windshield.

Longitudinal Feel Trim

When flying through areas of precipitation, partial or complete failure of the longitudinal control artificial feel system may result due to ice and/or water blockage of the bellows ram air line. If this condition occurs, excessive stick force will be required to maintain the desired airplane attitude. Since sudden longitudinal trim changes may occur several minutes after flying through freezing precipitation especially during descent to altitudes below the freezing level, the application of corrective longitudinal trim when a blocked bellows inlet is suspected is not recommended.

CAUTION

In the event of ice and/or water blockage of the artificial feel bellows ram air line is suspected, longitudinal trim should not be applied to relieve control stick force, due to intermittent nature of the failure and suddenness of return to normal. Instead, use extra pilot effort on stick to maintain desired airplane attitude.

Central Air Data Computer

The central air data computer may malfunction during flight through ice and/or rain due to impact forces imposed by water and ice on the CADC total temperature sensor. A momentarily flashing "Duct Temp Hi" warning light usually indicates that the sensor probe has been blocked or shorted by ice accumulation. A sensor probe that has completely failed will be evidenced by a continuously flashing "Duct Temp Hi" warning light. These total temperature malfunctions result in erroneous true airspeed signals to the AMCS computer, and cyclic operation of the intake duct ramps at all airspeeds. A malfunction in the CADC may also be caused by rain and/or ice impact damage to the angle-of-attack probe. An erroneous and erratic angle-of-attack system will also result in erroneous airspeed and altimeter indications since static pressure is corrected for angle-of-attack in the CADC system. These errors will become significantly larger at supersonic speeds. If the above malfunctions occur, proceed as follows:

If "Duct Temp Hi" light flashes intermittently or continuously:

1. Decelerate as rapidly as practical to subsonic flight.
2. Maintain normal subsonic cruise airspeeds.

If erratic or erroneous airspeed, vertical speed or altimeter indications are suspected:

1. Static pressure correction switch - OFF

CAUTION

Pending the outcome of F4H test programs, penetration of heavy precipitation above the freezing level should be avoided unless dictated by operational necessity.

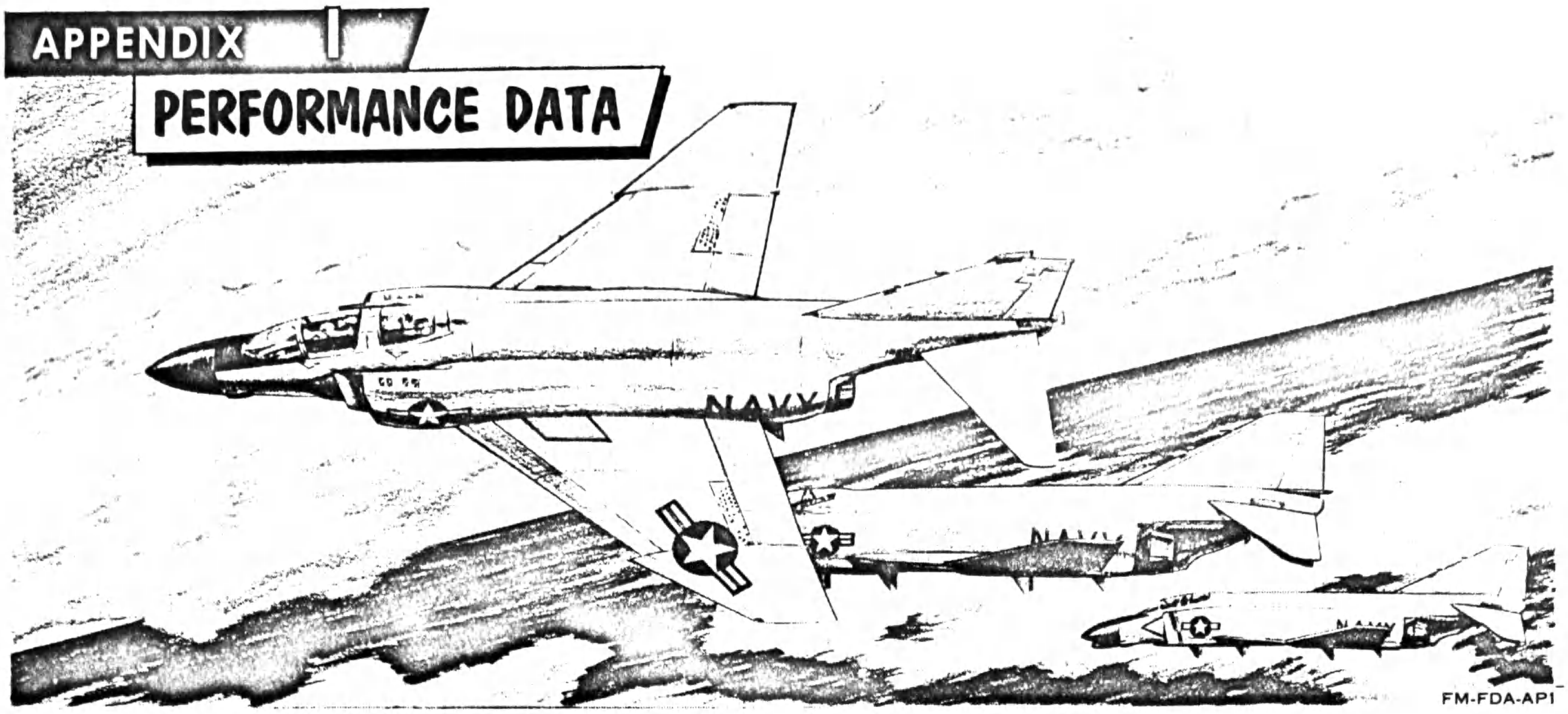
PRECIPITATION PENETRATION DESCENTS

When making a descent in precipitation, the minimum power setting should be approximately 78% rpm. This will result in an average rate of descent at 250 knots CAS of 4,000-4,500 fpm from 20,000 feet. Power reductions below the above values, to provide an increased rate of descent are not advisable due to:

a. Marginal windshield defogging.
b. Marginal engine anti-icing effectiveness.
c. Severe drop in RPM which is likely to occur if power is retarded to minimum fuel flow (IDLE) in heavy precipitation.

LANDING

Landing on wet or snow and ice covered runways presents no directional control or stopping problem as long as the drag chute is utilized.

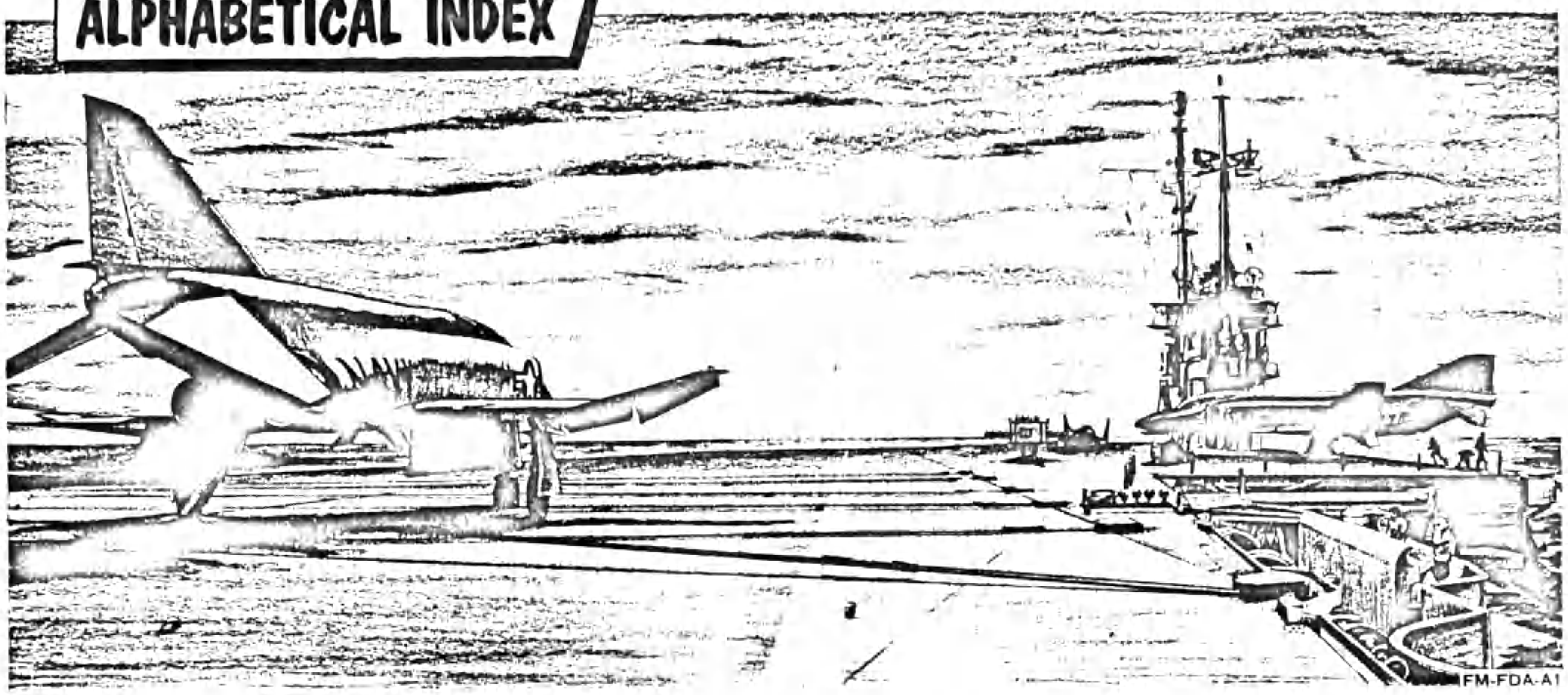


FM-FDA-AP1

SEE PUBLICATION
NAVWEPS 01-245FDA-1A
PERFORMANCE DATA
FOR NAVY MODEL

F4H-1F AIRCRAFT

ALPHABETICAL INDEX



FM-FDA-A1

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