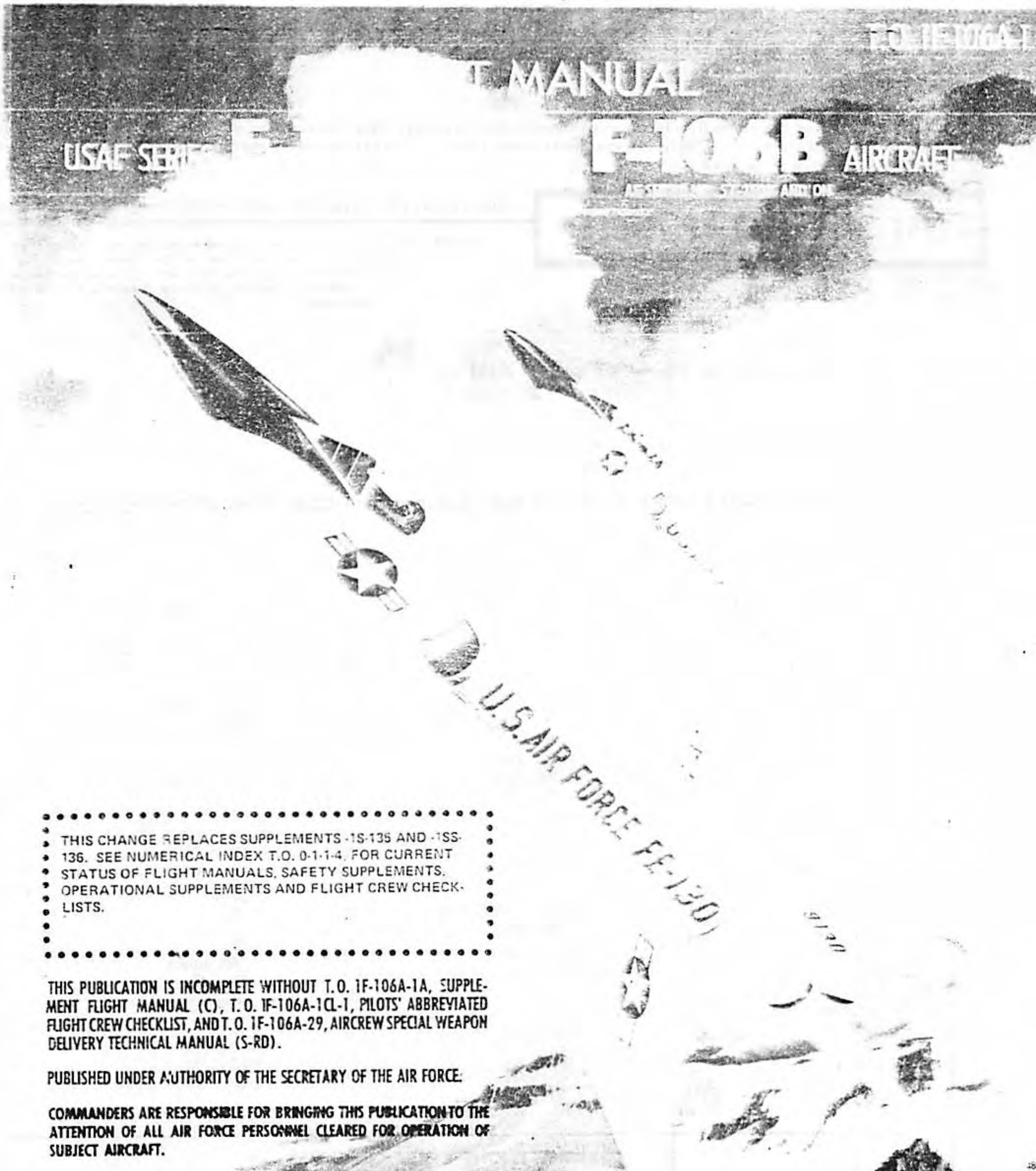
OTB



THIS CHANGE REPLACES SUPPLEMENTS -1S-135 AND -1SS-136. SEE NUMERICAL INDEX T.O. 0-1-1-4, FOR CURRENT STATUS OF FLIGHT MANUALS, SAFETY SUPPLEMENTS, OPERATIONAL SUPPLEMENTS AND FLIGHT CREW CHECKLISTS.

THIS PUBLICATION IS INCOMPLETE WITHOUT T.O. 1F-106A-1A, SUPPLEMENT FLIGHT MANUAL (C), T.O. 1F-106A-1CL-1, PILOTS' ABBREVIATED FLIGHT CREW CHECKLIST, AND T.O. 1F-106A-29, AIRCREW SPECIAL WEAPON DELIVERY TECHNICAL MANUAL (S-RD).

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE.

COMMANDERS ARE RESPONSIBLE FOR BRINGING THIS PUBLICATION TO THE ATTENTION OF ALL AIR FORCE PERSONNEL CLEARED FOR OPERATION OF SUBJECT AIRCRAFT.

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1 OCTOBER 1969

AIR FORCE 8 Mar 73—1800

CHANGE 5 1 DECEMBER 1972

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Change . . . . . . 2 . . . . . . 1 Dec 70
Change . . . . . . 3 . . . . . 15 Sep 71
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Change . . . . . . 5 . . . . . 1 Dec 72

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CURRENT FLIGHT CREW CHECKLIST

T.O. 1F-106A-1CL-1

1 October 1969 Changed 1 December 1972

Upon receipt of the second and subsequent changes to this technical order, librarians shall ascertain that all previous changes have been received and incorporated. Action should be taken promptly if the publication is incomplete.

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USAF

FLIGHT MANUAL, SAFETY SUPPLEMENT, AND OPERATIONAL SUPPLEMENT STATUS

This page will be published with each Safety Supplement, Operational Supplement, Flight Manual Change, and Flight Manual Revision. It provides a comprehensive listing of the current Flight Manuals, Flight Crew Checklist, Safety Supplements, and Operational Supplements. The supplements you receive should follow in sequence. If you are missing one listed on this page, see your Publications Distribution Officer and get your copy. Periodically check Numerical Index T.O. 0-1-1-4 to make sure you have the latest Supplements, Checklist, and Basic Manual.

CURRENT FLIGHT MANUALS	DATE	CHANGED
T.O. 1F-106A-1	1 October 1969	1 December 1972
T.O. 1F-106A-1A (Conf MA-1 System Operation)	1 April 1970	15 September 1971

CURRENT FLIGHT CREW CHECKLIST	DATE	CHANGED
T.O. 1F-106A-1CL-1	1 October 1969	1 December 1972

SUMMARY OF SUPPLEMENTS INCORPORATED IN THIS CHANGE

Number	Date	Short Title
-1S-135	26 Oct 72	Approach End Arrestments
-1SS-136	30 Oct 72	Revised Abort Procedure

CURRENT OPERATIONAL SUPPLEMENTS

-1S-134	28 Jun 72	Block S Modification

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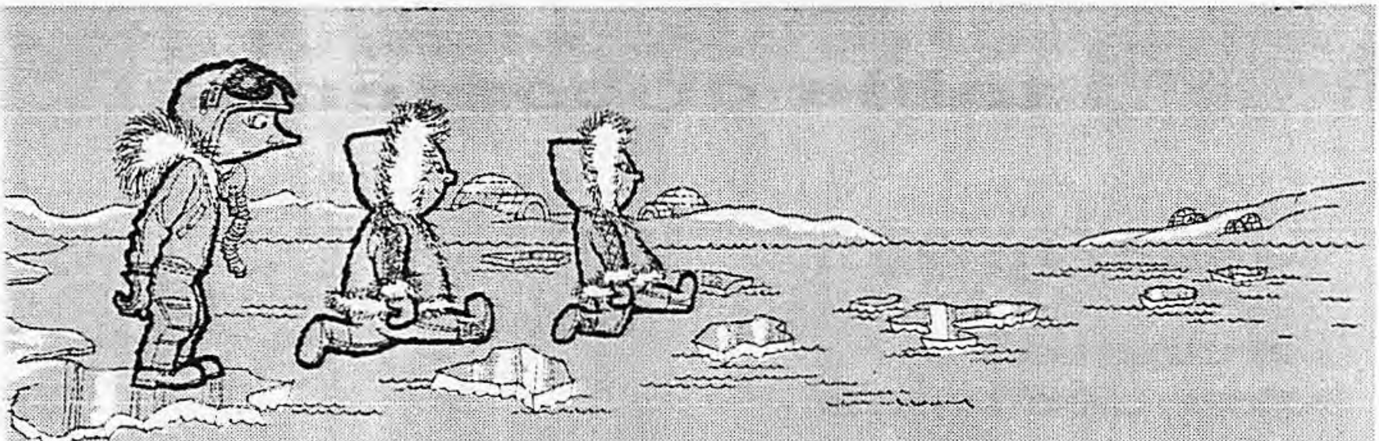

DON'T SLIP...!

...BY THESE PAGES...

SCOPE

This manual contains the necessary information for safe and efficient operation of the F-106A and the F-106B. These instructions provide you with a general knowledge of the airplane, its characteristics, and specific normal and emergency operating procedures. Your flying experience is recognized, and therefore, basic flight principles are avoided.

SOUND JUDGMENT

Instructions in this manual are for a pilot inexperienced in the operation of these airplanes. *This manual provides the best possible operating instructions under most circumstances, but it is a poor substitute for sound judgment.* Multiple emergencies, adverse weather, terrain, etc., may require modification of the procedures.

PERMISSIBLE OPERATIONS

The Flight Manual takes a "positive approach" and normally states only what you can do. Unusual operations or configurations (such as asymmetrical loading) are prohibited unless specifically covered herein. Clearance must be obtained from SAAMA (MMEAF) before any questionable operation is attempted which is not specifically permitted in this manual.

STANDARDIZATION AND ARRANGEMENT

Standardization assures that the scope and arrangement of all Flight Manuals are identical. The manual is divided into ten fairly independent sections to simplify reading it straight through or using it as a reference manual. The first three sections must be read thoroughly and fully understood before attempting to fly the airplane. The remaining sections provide important information for safe and efficient mission accomplishment.

CHANGES TO THE FLIGHT MANUAL AND CHECKLIST

Changes are periodically made to keep the Flight Manual and Checklist current. New or revised information in a Change will be indicated by a black vertical line in the margin of the page opposite the new or revised text, illustration, or table. When you receive a Change to the Flight Manual or Checklist, insert the changed pages into the basic publication and destroy the superseded pages.

CHECKLISTS

The Flight Manual contains only amplified checklists. Abbreviated checklists have been issued as separate technical orders—see the back of the title page for T.O. number and date of your latest checklist. The checklist consists of three parts, arranged in the following order: Emergency Procedures, Normal Procedures, and Performance Data. The numbered items (line items) in the emergency and normal procedure sections correspond to identically numbered items in the amplified procedures of Sections II and III of the Flight Manual. The checklist contains airplane performance data in tabular and chart forms for both Ⓐ and Ⓑ airplanes. Use of these data is explained in Part 9 of the Appendix in the Flight Manual.

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CHECKLIST CONCURRENCY

As changes are made to the amplified checklists in the Flight Manual, concurrent changes will be made to the abbreviated checklist so that both will agree. However, a change to the Flight Manual may not affect the amplified procedures, and thus the Flight Manual date may not be the same as the checklist date. To determine the checklist applicable to a given Flight Manual issue, refer to the bottom of the Flight Manual "A" page under "Current Flight Crew Checklist." For purposes of determining the concurrency between the Flight Manual and the checklist, the latest date of a Safety Supplement or an operational Supplement affecting the checklist will be considered to represent the latest change date of the Flight Manual.

SAFETY SUPPLEMENTS

Information involving safety will be promptly forwarded to you by Safety Supplements. Supplements covering loss of life will get to you in 48 hours by TWX and those concerning serious damage to equipment within ten days by mail. The current status of each Safety Supplement affecting your airplane can be determined by referring to the Numerical Index (T.O. 0-1-1-4) and supplements thereto. The Status Page of the Flight Manual and each Safety Supplement should also be checked to determine the status of existing supplements. You must remain constantly aware of the status of all supplements—current supplements must be complied with, but operations should not be restricted by complying with a replaced or rescinded supplement.

SAFETY SUPPLEMENT CHECKLIST CHANGES

Whenever you receive an Interim Safety Supplement affecting your checklist, write in the appropriate information. Printed replacement checklist pages will be made available to you in the formalized Safety Supplement. A notation on the bottom inside corner of these pages will indicate that they reflect certain Safety Supplements; however, discarding of a Safety Supplement is authorized only through the Status Page (Flight Manual or Supplement) or T.O. 0-1-1-4.

HOW TO GET PERSONAL COPIES

Each pilot is entitled to personal copies of the Flight Manual, Safety Supplements, and checklists. The required quantities should be ordered before you need them to assure their prompt receipt. Check with your supply personnel—it is their job to fulfill your Technical Order requests. Basically, you must order the required quantities on the Publication Requirements Table (T.O. 0-1-1-4). Technical Orders 00-5-1 and 00-5-2 give detailed information for properly ordering these publications. Loose leaf binders and sectionalized tabs are available for use with your manual. These are obtained through local purchase procedures and are listed in the Federal Supply Schedule (FSC Group 75, Office Supplies, Part-1).

WARNINGS, CAUTIONS, AND NOTES

The following definitions apply to "Warnings," "Cautions," and "Notes" found throughout the manual:

WARNING

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

CAUTION

Operating procedures, techniques, etc., which will result in damage to equipment if not carefully followed.

NOTE

An operating procedure, technique, etc., which is considered essential to emphasize.

YOUR RESPONSIBILITY— TO LET US KNOW

Every effort is made to keep the Flight Manual current. Review conferences with operating personnel and a constant review of accident and flight test reports assure inclusion of the latest data in the manual. *However, we cannot correct an error unless we know of its existence.* In this regard, it is essential that you do your part. Comments, corrections, and questions regarding this manual or any phase of the Flight Manual program are welcomed. These should be forwarded through your Command Headquarters for ADC (DOTW) to SAAMA (MMEAF), Kelly AFB, Texas.

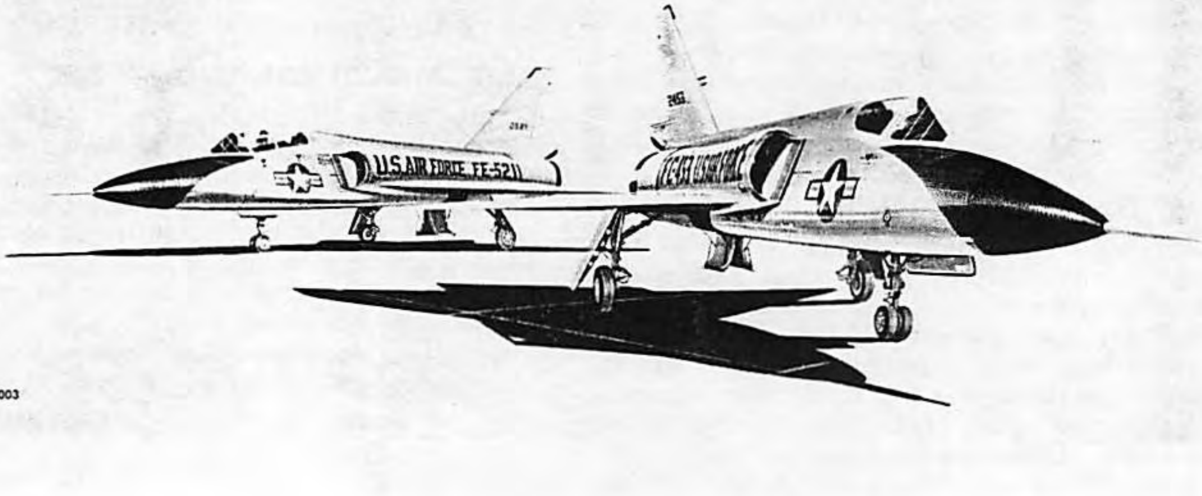
AIRPLANE DESIGNATION CODE

Where text or illustrations are not specifically identified for particular models, it may be assumed that such items are common to both the F-106A and F-106B airplanes. This is also true for items common to the forward and aft cockpits of the F-106B and the cockpit of the F-106A. Illustrations which do not show minor variations in the airplane models are labeled "Typical." Text and illustrations applicable to F-106A or F-106B airplanes are identified by the following code symbols:

CODE	AIRPLANE
Ⓐ	F-106A
Ⓑ	F-106B

48002-2

the F-106A and F-106B delta dart



48003

description

Section I

48054

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THE AIRPLANE

The F-106A is a single-place supersonic all-weather interceptor built by General Dynamics/Convair. The F-106B is a two-place trainer version of the F-106A and is designed for combat use, if conditions make such use necessary. The airplane is powered by a J75-P-17 axial-flow turbojet engine with afterburner and is equipped with an aircraft and weapon control system which includes an automatic flight control system, a fire control system, and communications and navigation equipment.

NOTE

This manual covers two aircraft and weapon control systems, the MA-1 incorporated in the F-106A airplane and the AN/ASQ-25 incorporated in the F-106B airplane. The text distinguishes between the two configurations only where necessary and will refer to all switches, controls, etc., in both the F-106A and F-106B aircraft weapon and control systems, as the "MA-1."

The airplane is typified by an area rule fuselage, a large delta wing, and the absence of a conventional empennage. A pressurized cockpit contains an ejection seat with a one-motion control for escape. On B airplanes the ejection seats (in a single pressurized cockpit) are in tandem with the aft seat higher than the forward seat to improve forward visibility in the aft seat. The fully powered flight control system uses "elevons" to provide both aileron and elevator action from conventional cockpit controls. All control surfaces are hydraulically actuated, and control feel is provided by an artificial feel system. An engine inlet duct airflow control system assures engine inlet airflow stability. Conventional tricycle landing gear is used for landing and takeoff. Nose wheel steering is electrically controlled and hydraulically powered. The integral wing tanks are serviced by a single-point pressure refueling system and fuel usage is sequenced automatically. The external tanks are also serviced by the single-point refueling system. Some airplanes are equipped with an air refueling system. On A airplanes an automatic fuel transfer system is included to control cg during supersonic flight.

airplane general

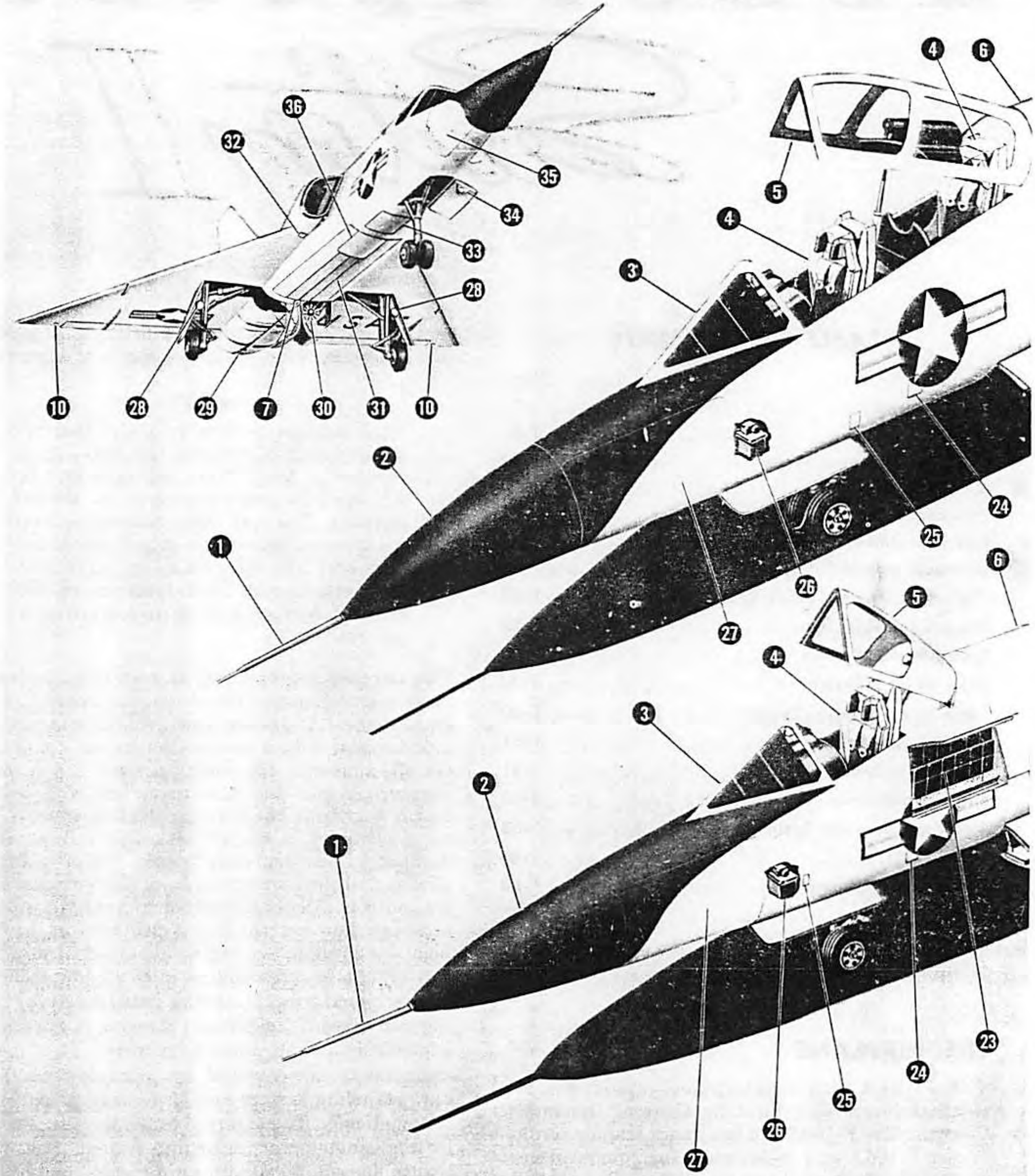
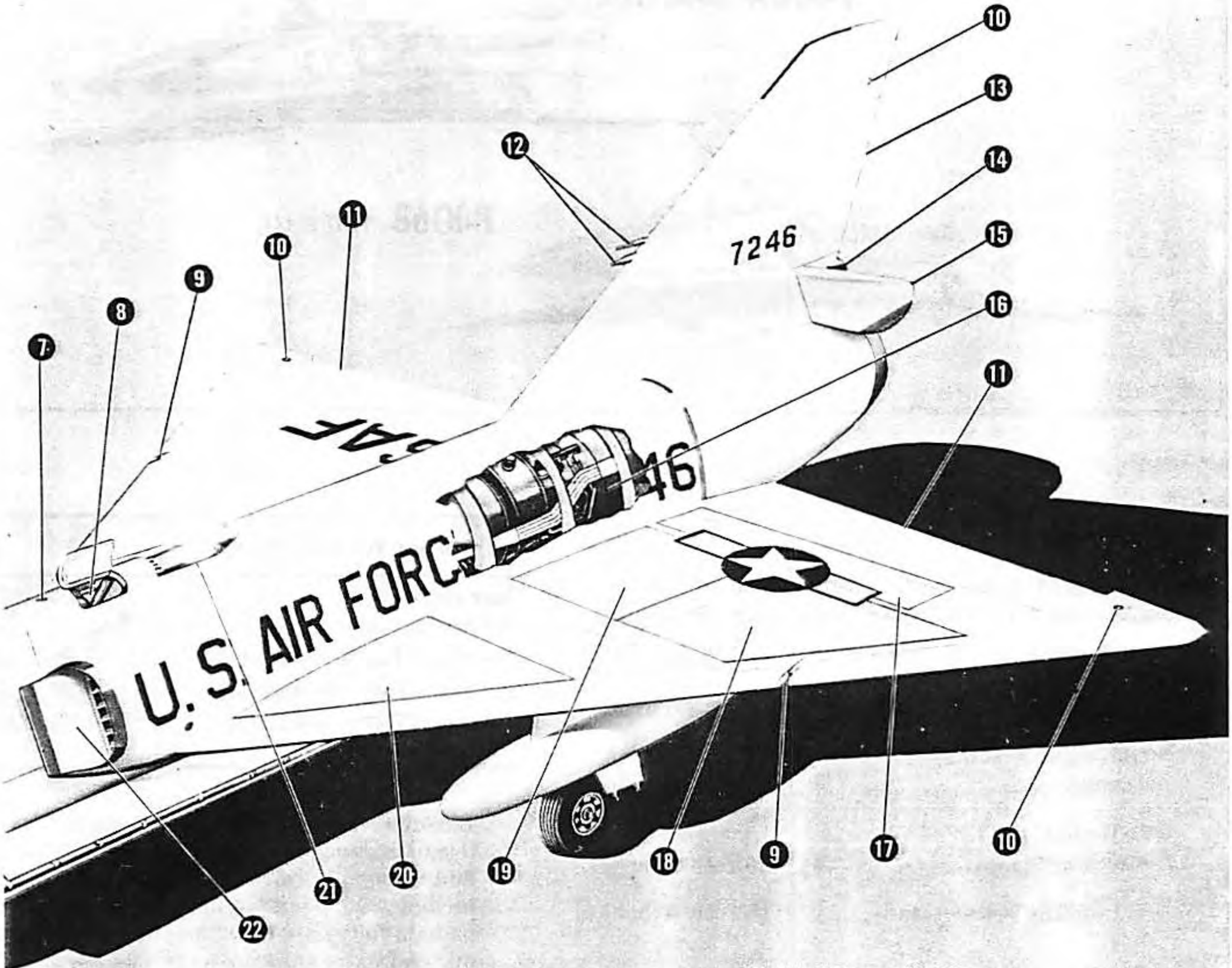


Figure 1-1

arrangement



1. Nose Boom
2. Radome
3. Windshield
4. Ejection Seats
5. Canopy
6. Fuselage Fuel Tank
7. Anticollision Lights (2)
8. Air Conditioning Compartment
9. Wing Slots (2)
10. Navigation Lights (3)
11. Elevons (2)
12. Ram Air (Q) Intake
13. Rudder
14. Drag Chute
15. Speed Brakes
16. J-75 Engine With Afterburner
17. Wing Transfer Fuel Tank
18. Fuel Tank No. 2
19. Fuel Tank No. 3
20. Fuel Tank No. 1
21. Air Refueling Slipway Door
22. Engine Intake Duct
23. Upper Aft Electronics Ⓐ
24. External Power Receptacle
25. Liquid Oxygen Filler Valve
26. Battery (Emergency DC Power Package) (Nose Wheel Well)
27. LH Forward Electronics Compartment
28. Landing Lights (2)
29. Tail Hook
30. Ram Air Turbine (Hydraulic Compartment)
31. Missile Bay
32. Single Point Refueling Receptacle (Under RH Intake Duct)
33. Lower Aft Electronics Compartment Ⓐ
    Lower Mid-Electronics Compartment Ⓑ
34. Taxi Light
35. RH Forward Electronics Compartment
36. Lower Aft Electronics Compartment Ⓑ

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main differences

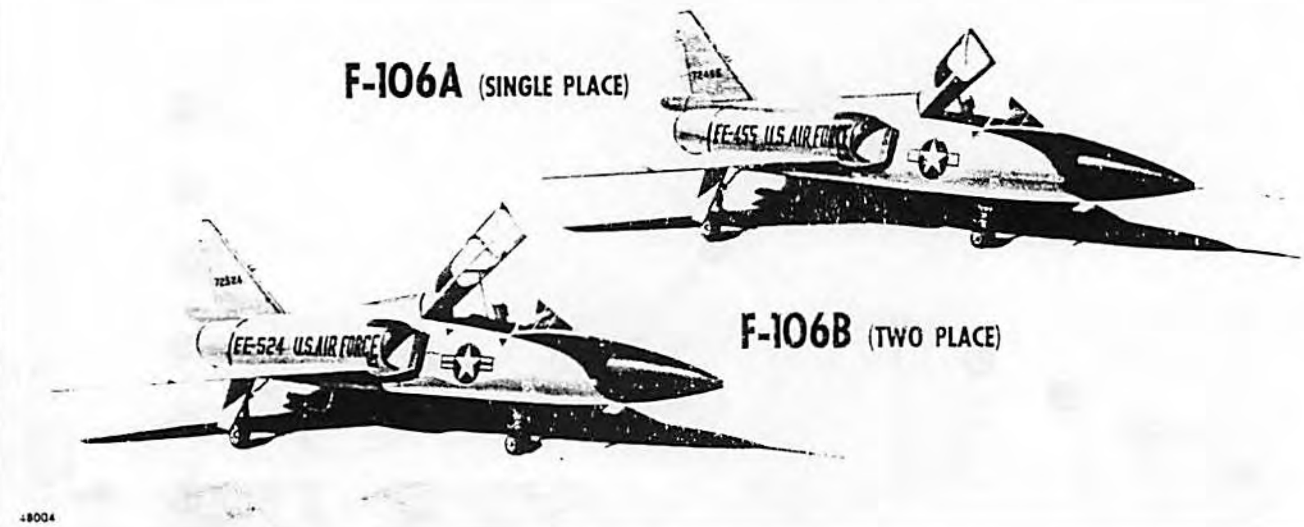


Figure 1-2

DIMENSIONS

The approximate overall dimensions of the airplane under normal conditions of gross weight, tire and gear strut inflation are as follows:

Wing span ....38 feet, 3.5 inches

Length (including pitot boom) ....70 feet, 8 inches

Height (to tip of vertical stabilizer) ....20 feet, 3 inches

Landing gear tread................15 feet, 5 inches

NOTE

Refer to Section II for Minimum Turning Radius and Ground Clearances.

GROSS WEIGHTS

The approximate gross weights for airplanes having 360-gallon external tank provisions and air refueling capability are as follows:

Gross Weight, Pounds	Ⓐ	Ⓑ
Basic Weight	25,130	26,200
Full Internal Fuel	35,240	36,129
Full Internal Fuel + Armament	36,663	37,552
Full Internal Fuel + External Fuel	40,408	41,297
Full Internal Fuel + External Fuel + Armament	41,831	42,720

Note

Basic weight includes the airplane, external tank pylons, unusable internal fuel, and unusable oil. Other gross weights include 235 pounds for pilot Ⓐ or 470 pounds for pilots Ⓑ, 34 pounds for usable oil, and external tanks (two 360-gallon) weight where an external tank configuration is specified.

NOTE

Gross weights with "Full Internal Fuel" refer to all internal fuel tanks having been filled to capacity. Gross weights with "Reduced Internal Fuel" refer to all internal fuel tanks having been filled to capacity with the exception of the T tanks on Ⓐ airplanes and the F tank on Ⓑ airplanes.

AIRPLANE CONFIGURATION

Block numbers are not used in this manual to identify airplane configuration. When changes are readily visible to the pilot, they are usually identified by "some airplanes" and "other airplanes." If the change is not readily visible, footnotes are used to identify airplanes by serial number. For instance, the illustration of the compass control panel uses "some airplanes" and "other airplanes" because the difference is easily recognized. The discussion of the nose wheel steering system uses serial numbers because different power sources are involved but the difference is not readily visible.

ENGINE

Thrust is supplied by a Pratt and Whitney J75-P-17 engine equipped with an afterburner (figure 1-3). This engine has a sea-level standard day static thrust rating of approximately 16,100 pounds at military thrust and 24,500 pounds at maximum thrust (afterburning). The engine consists of a split 15-stage axial-flow compressor, eight radially mounted combustion chambers, a split three-stage turbine, and an afterburner incorporating a two-position exhaust nozzle. The split multistage compressor consists of an eight-stage low-pressure rotor (N_1) and a seven-stage high-pressure rotor (N_2). The low-pressure rotor is connected by a drive shaft to the second and third stage turbine wheels. The high-pressure rotor is connected independently from the low-pressure rotor by a hollow drive shaft (rotating around the low-pressure drive shaft) to the first-stage turbine wheel. The rpm of the high-pressure rotor is governed by the throttle, and the rpm of the low-pressure unit is completely independent of pilot control. A compressor-air bleed system is used to direct part of the low-pressure rotor air overboard at low engine rpm. This bleed system is actuated automatically and is controlled by a governor driven by the low-pressure rotor, and serves to

J-75 engine

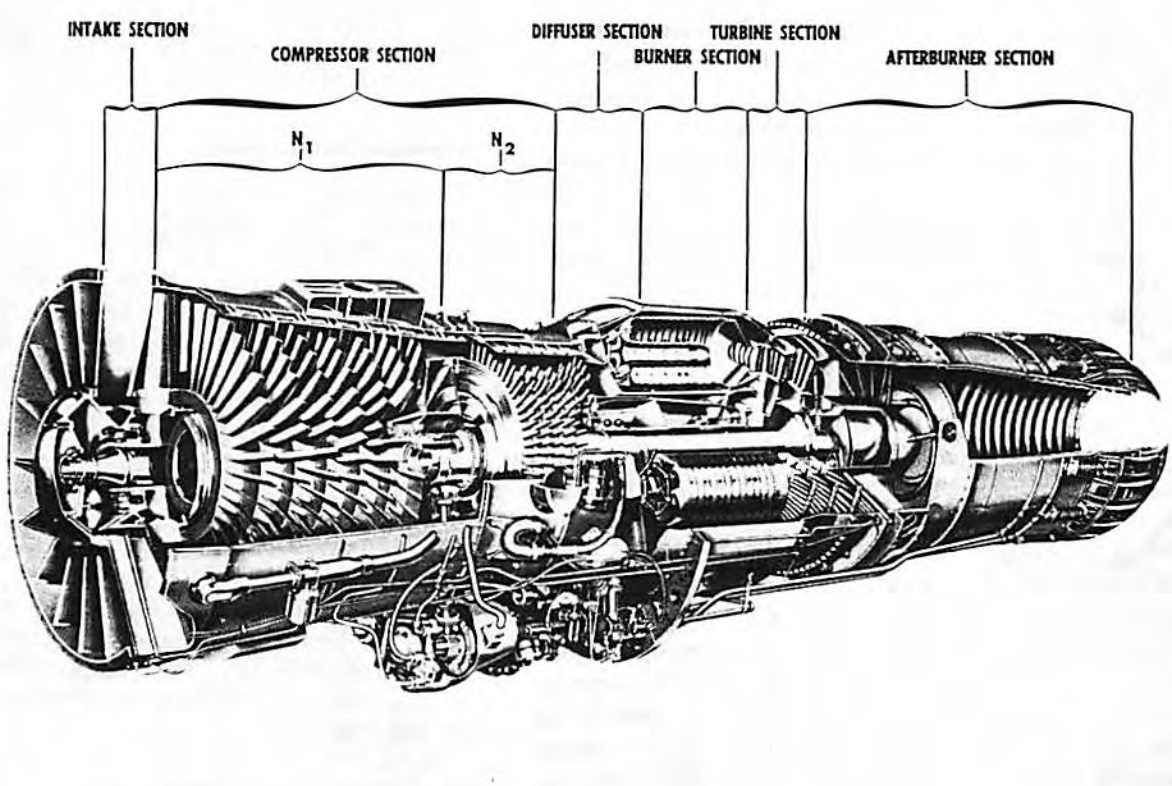


Figure 1-3

facilitate starting, to improve engine acceleration, and to prevent engine surging during low thrust operation. The main engine accessory section, driven by the high-pressure rotor, provides reduction gearing and mounting pads for all the engine-driven accessories. For additional information on this system, refer to T.O. 1F-106A-2-4.

VARIABLE RAMP SYSTEM

Variable ramps are provided to obtain optimum airplane performance and to maintain inlet stability over a wide range of speed and altitude by properly matching the inlet capabilities with the engine airflow demands. The variable ramp system (figure 1-4), provides optimum performance at full engine thrust and maintains inlet stability by adjusting the inlet duct area at higher Mach numbers. Duct design takes advantage of shock waves and airflow theories to efficiently reduce the local Mach number to subsonic speeds at the engine face. Engine-inlet matching is critical, especially at supersonic speeds. If the duct area is too small, pressure recovery will be low, resulting in decreased engine performance. If the duct area is too large, inlet stability problems arise which may result in stall-buzz. The variable ramps maintain an inlet throat static-to-total pressure ratio that varies with airplane Mach number. Static and total pressures are sensed in the throat of each inlet by a pitot-static probe. Whenever the sensed pressure ratio differs from the desired pressure ratio, the inlet control system drives the variable ramp to change the duct area, causing the duct throat pressures to vary and bring the pressure ratio within desired limits. To assist in maintaining the desired pressure ratio and to reduce the possibility of stall-buzz occurring, the static pressure lines from the two engine inlet duct pitot-static probes are manifolded to a shuttle valve. The shuttle valve operates to shift variable ramp control to the engine inlet duct having the highest static pressure. During yaw conditions when left and right engine inlet duct static pressure are not symmetric, the shuttle valve prevents variable ramp action which will aggravate the asymmetric condition. The first ramp on the intake ducts is a 9-degree wedge extending forward of the inlet openings. The first ramp is followed by a variable angle ramp assembly in each duct, which is composed of three hinged interlocked sections. These sections are automatically positioned to control duct area and the shock wave pattern, providing the required engine inlet matching. The forward edge of the forward ramp section is hinged to the inlet duct and may be varied from 8° to 30° with respect to the airplane centerline. Other movable sections of the ramps are hinged and sealed to the top and bottom of the intake ducts by an inflatable seal. The center sections of the variable ramps are slotted to bleed low-energy boundary-layer air out of the intake ducts. This air is ducted to the lower side of the fuselage where it is dumped overboard. The variable ramps in both left and right intake ducts are extended and retracted automatically by a hydraulic motor. Each ramp contains four screw jacks which are driven by a flexible drive system connected to the hydraulic drive unit. During normal operation the variable ramp system will start to maintain an inlet duct Mach schedule by partially extending the ramps at approximately Mach 1.4.

NOTE

> When the airplane accelerates to Mach 1.25 the variable ramp system is energized by the air data computer and remains energized until the airplane is decelerated to speeds less than Mach 1.20.

In the event of hydraulic pressure failure or loss of ac and dc nonessential power, an emergency source of high-pressure air (stored in a 100 cu. in. flask) can be selected to drive the variable ramps to the fully retracted position (engine inlets open) to facilitate flight at subsonic airspeeds. If the ramps are retracted by use of the emergency system, they should not be used during the remainder of the flight. After landing, the operation should be entered on Form 781 to direct ground servicing of the system.

Variable Ramp Switch

The guarded variable ramp switch (32, figure 1-10), placarded "Vari-Ramp," is located on the left-hand console. The switch has AUTO and EMER OPEN (retracted) positions. With the switch in AUTO position the system will operate automatically to maintain the optimum engine inlet airflow through the intake ducts. In the event of failure of normal operation of the variable ramps, placing the switch to EMER OPEN position will supply pneumatic pressure to the ramp drive motor to drive the ramps to the full retracted position. The emergency retraction system will only retract the ramps; therefore, flight at airspeeds above variable ramp inoperative limit speed should not be attempted during the remainder of the flight. After landing, use of the emergency system should be entered on Form 781 to direct ground servicing of the system. The switch receives power from the dc essential bus.

Variable Ramp Warning Light

An amber variable ramp warning light (figure 1-30), placarded "Variable Ramp Not Retracted,"

variable ramp system

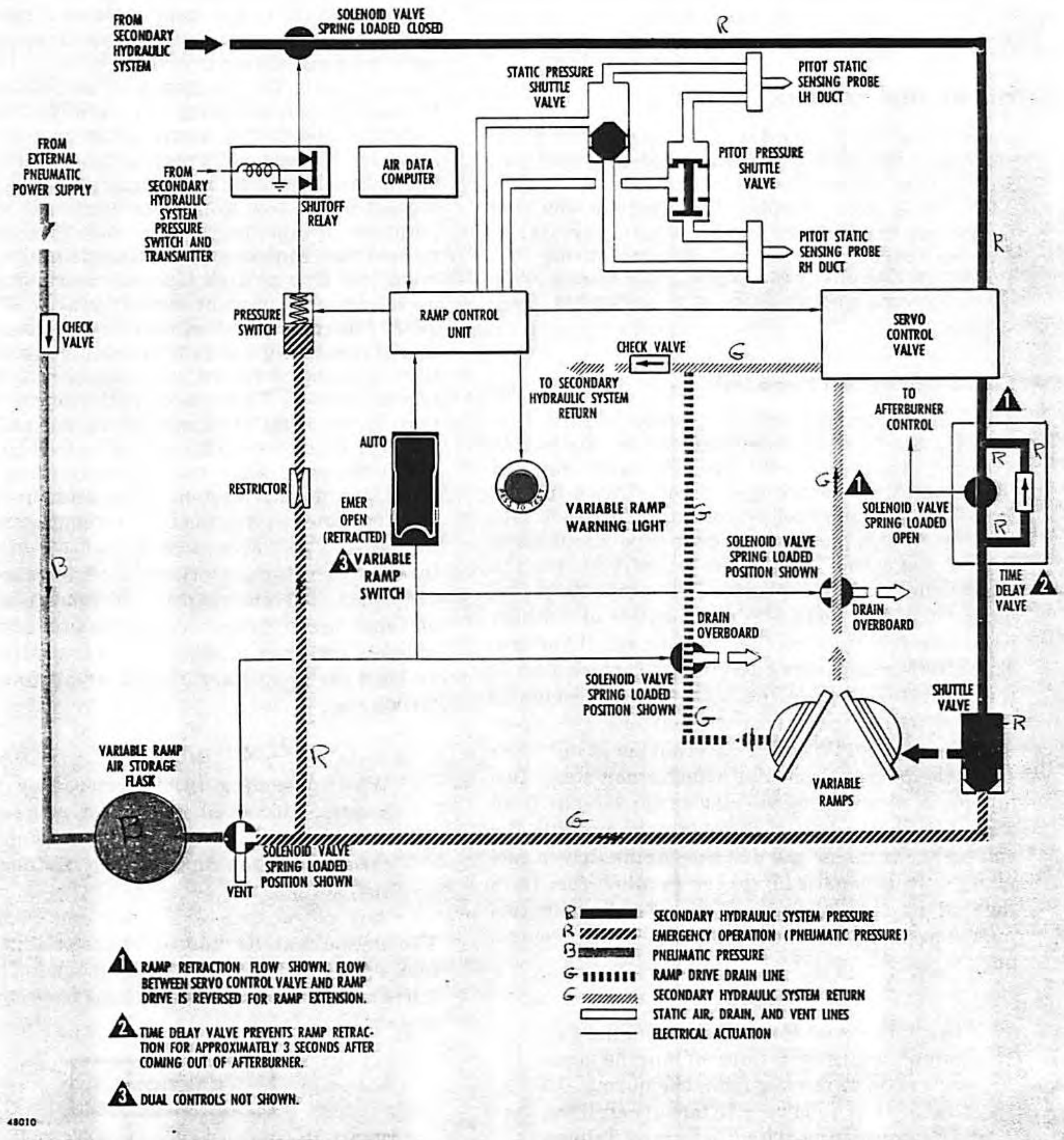


Figure 1-4

or "Vari-Ramp Not Retracted," is located on the instrument panel on Ⓐ airplanes and the forward instrument panel on Ⓑ airplanes. The light will illuminate to indicate failure of the ramps to fully retract at airspeeds below Mach 1.2. The light will extinguish when the ramps are fully retracted. The warning light receives power from the 28-volt dc essential bus.

NOTE

The variable ramp warning light is activated by a microswitch on the right ramp.

ENGINE FUEL CONTROL SYSTEM

Desired engine rpm is established by throttle movement, and fuel flow to the engine is delivered and regulated by the engine fuel control system (figure 1-5). The system includes the engine-driven fuel pump unit, the fuel control unit, the fuel pressurizing and dump valve unit, and the afterburner fuel system. For details of the afterburner system refer to ENGINE AFTERBURNER SYSTEM, this Section.

Engine-Driven Fuel Pump Unit

The engine-driven fuel pump unit (figure 1-5) supplies the fuel pressure required by the engine and the afterburner systems. The unit contains four pumps. A centrifugal pump draws in fuel from the airplane fuel system and forces it into three gear-type pumps. One gear-type pump is the engine stage fuel pump; the other two are the afterburner stage fuel pumps. The engine stage fuel pump furnishes fuel to the fuel control unit which regulates fuel flow to the combustion chambers. The afterburner stage fuel pumps furnish fuel to the afterburner metering valve which regulates fuel flow to the afterburner when afterburner operation is selected. When the afterburner is not operating the output from the afterburner stage fuel pumps is returned to the discharge stream from the centrifugal pump. If the engine pump fails, the emergency transfer valve in the engine-driven fuel pump unit automatically opens to allow fuel from the output side of one of the afterburner fuel pumps to flow to the hydromechanical fuel control unit.

NOTE

The only visual indication of engine fuel pump failure is failure of engine pressure ratio to recover from the normal 0.3 to 0.5 drop following afterburner light. When operating with fuel pump failure, fuel is supplied to both the engine and the afterburner systems, and if the throttle is in AFTERBURNER range a thrust loss of approximately 23% will occur at sea level on a standard day. No thrust loss will occur in the military thrust range.

Fuel Control Unit

The fuel control unit (figure 1-5) regulates fuel to the combustion chambers and incorporates normal and emergency fuel control systems. The normal fuel control system contains a mechanical computer, a governor, and temperature and pressure sensing elements which control the main throttle valve. The computer, in addition to throttle position, senses changes in flight conditions and regulates fuel flow to insure optimum engine operation for the selected thrust setting. During rapid engine accelerations the normal fuel control system regulates fuel flow to prevent overspeed, overtemperature, compressor stalls, and flameouts. The normal fuel control system also maintains a minimum fuel flow at high altitudes and during rapid decelerations to prevent engine flameout. The emergency fuel control system provides an alternate system of regulating fuel flow to the combustion chambers in event of failure of components within the normal system. To transfer to the emergency system, in the event of failure of the normal system, the fuel control switch must be placed to EMER. When the emergency fuel control system is energized, the normal system is rendered inoperative and fuel flow is controlled by the emergency throttle valve. This valve is connected directly to the throttle; therefore, emergency fuel flow is manually controlled. The emergency fuel system is capable of supplying sufficient fuel to obtain at least 95% military thrust on a 100°F day at low altitudes and at least 80% military thrust at altitudes up to 30,000 feet.

NOTE

When operating in the emergency fuel system, full obtainable thrust will result in lower rpm and exhaust gas temperature than when operating in the normal fuel system.

The engine may be started on the emergency system, either in flight or on the ground. The afterburner may be operated on the emergency fuel system.

CAUTION

To prevent possible heat damage to the turbine section, ground starts on the emergency system should be limited to emergency situations only.

engine fuel control system

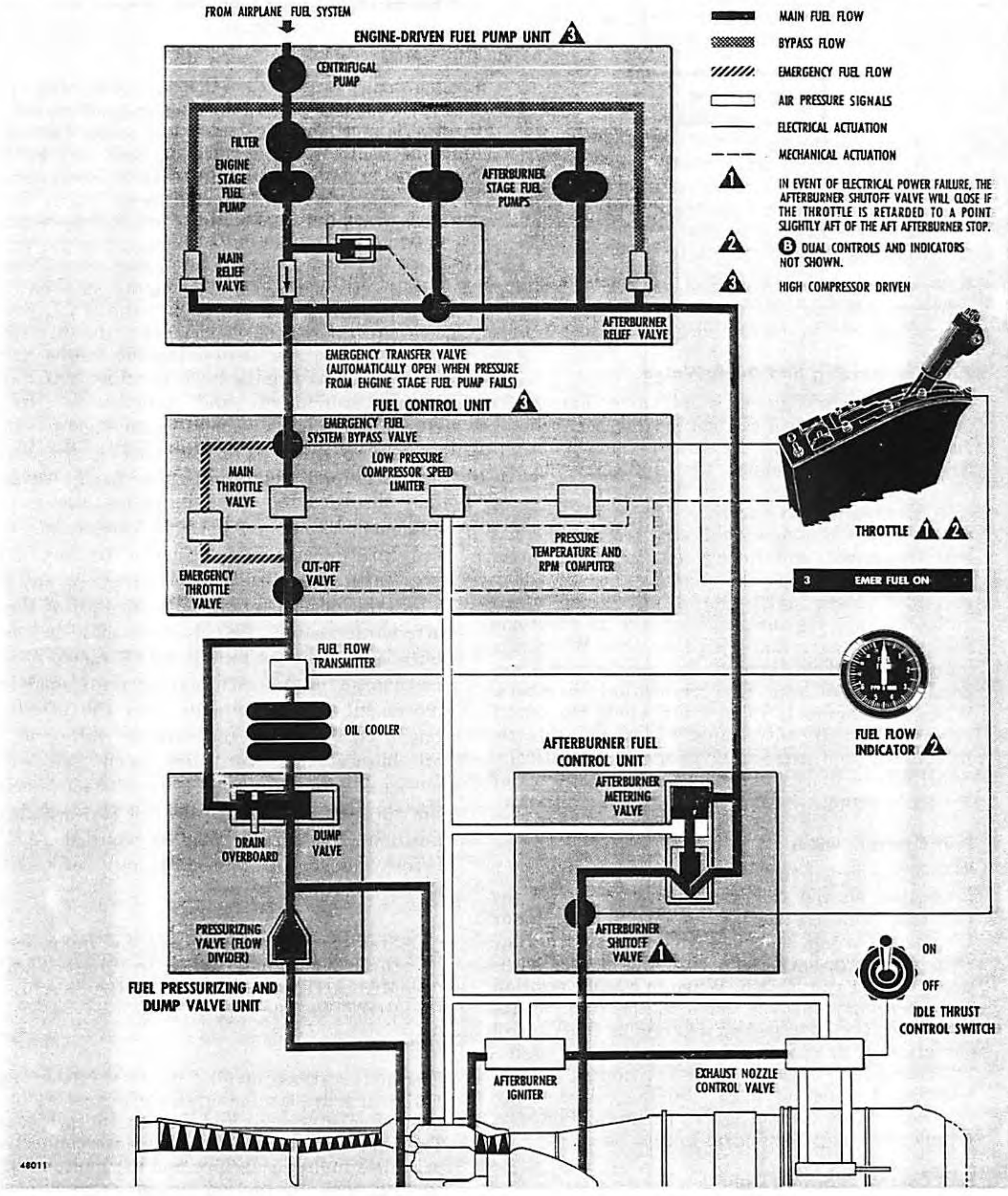


Figure 1-5

WARNING

Inflight use of the emergency fuel control system is permitted only during an actual emergency condition requiring its use.

CAUTION

When operating on the emergency fuel control system, rapid throttle movements must be avoided to prevent overspeed, overtemperature, compressor stalls, and flameouts since only the normal system contains these compensating features.

When the throttle is retarded to OFF a mechanically controlled cutoff valve in the fuel control unit cuts off all fuel to the combustion chambers.

Fuel Pressurizing and Dump Valve

The fuel pressurizing and dump valve (figure 1-5) is located in the fuel control system between the fuel cutoff valve and the combustion chambers. The unit controls fuel flow to the primary and secondary injector nozzles in the engine combustion chambers. To facilitate starting, fuel at relatively low pressure is directed through the primary manifold, and spring tension on the pressurizing valve keeps the port to the secondary manifold closed until increasing engine speed builds up fuel pressure high enough to overcome the spring tension and open the valve. When this happens, fuel flows through both primary and secondary manifolds to the combustion chambers. When the engine is to be shut down, the cutoff valve in the fuel control unit is closed by throttle movement, and the dump valve automatically opens to permit residual fuel in the manifolds of the main combustion system to drain overboard.

Fuel Control Switch

The side-guarded fuel control switch (figure 1-6) is located on the throttle quadrant, aft of the throttle. This switch is used to select either the normal or emergency fuel control system. The switch is placarded "Fuel Cont" and has two positions, NORM and EMER. When in EMER position the emergency throttle valve in the fuel control unit is positioned to permit emergency fuel control operation. If dc essential power is lost, the engine will continue to operate on the fuel control system selected. A warning light illuminates when the emergency fuel control is in operation. Power is supplied by the dc essential bus.

Fuel Control Warning Light

The fuel control warning light (3, figure 1-30), located on the warning light panel, illuminates and displays "EMER FUEL ON" when the fuel control switch is in the EMER position. Before engine start, the light is illuminated when the fuel control switch is in either position. The light receives power from the dc essential bus.

THROTTLE Ⓐ

Engine thrust is controlled by the throttle (figure 1-6) which is located on the left-hand console. The throttle is mechanically linked to the fuel control unit and controls the normal fuel system, emergency fuel system and afterburner operation. The throttle grip contains the speed brakes switch, engine ignition button, and the microphone button. An anticreep assembly is installed in the throttle quadrant to prevent throttle creep in the fore and aft directions, thus eliminating the need for a manual friction lock. A stop below the IDLE position prevents inadvertent movement of the throttle into the OFF position. Depressing the engine ignition button and moving the throttle outboard to START supplies air, fuel, and ignition to fire the combustion starter. While continuing to hold the ignition button depressed, moving the throttle inboard supplies ignition to the engine. Advancing the throttle to IDLE supplies fuel to the combustion chambers. After the engine has started the throttle may be advanced to any desired position in the normal thrust range. At any point along the normal thrust range forward of the rear afterburner stop, the throttle may be moved outboard to the AFTERBURNER range. A detent mechanism in the quadrant prevents inadvertent movement of the throttle into the afterburner range. It also holds the throttle outboard when afterburner operation has been selected. A mechanical pin or "gate" prevents movement of the throttle inboard at the full forward throttle position. The throttle must be retarded 2-1/2° aft before movement, inboard out of AFTERBURNER, can take place.

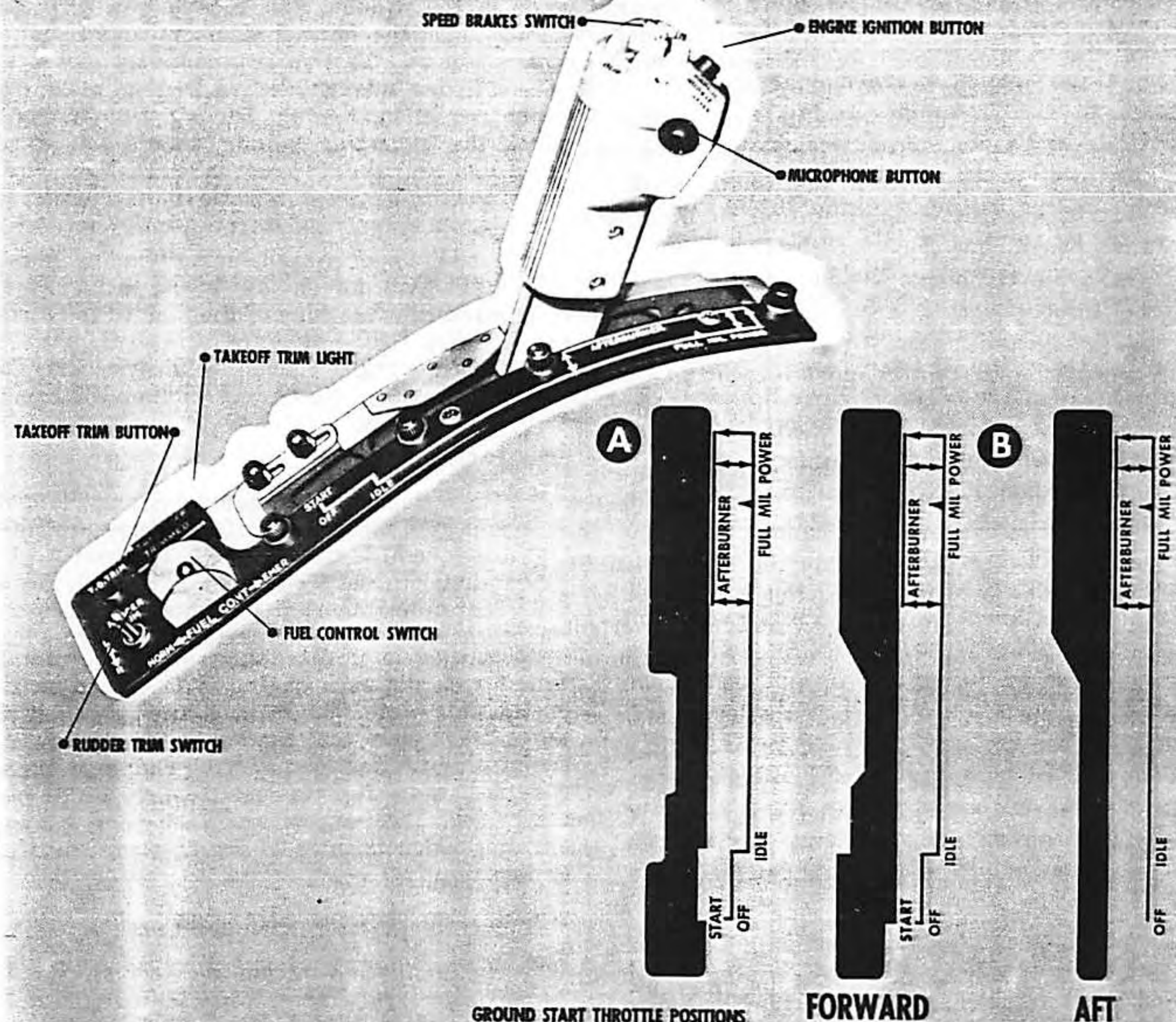
NOTE

Stall-buzz can be caused by improper throttle movement. Refer to THROTTLE MANAGEMENT TO AVOID STALL-BUZZ, Section VII.

THROTTLE Ⓑ

Engine thrust is controlled by the throttles (figure 1-6) which are located on the left consoles, one on the forward console and one on the aft console. The throttles are mechanically interconnected to provide simultaneous fore and aft motion but not simultaneous side motion (inboard and outboard). An anticreep assembly is installed on the aft throttle quadrant to prevent throttle creep fore or aft,

throttle quadrant (typical)



GROUND START THROTTLE POSITIONS
(WITH ENGINE IGNITION BUTTON DEPRESSED)

POSITION	ACTION
START	• GROUND STARTER UNIT OR AIRPLANE PNEUMATIC SYSTEM AIR-MOTORS STARTER INITIATING ENGINE ROTATION. • STARTER AND ENGINE IGNITION ARMED. • FUEL INJECTED TO STARTER AND STARTER IGNITES. (FULL COMBUSTION STARTS ONLY)
OFF	• ENGINE IGNITION ENERGIZED.
IDLE	• FUEL INJECTED TO ENGINE & IGNITES IN COMBUSTION CHAMBERS.

48015

Figure 1-6

thus eliminating the need for a manual friction lock. A stop below the IDLE position, on the forward throttle quadrant only, prevents inadvertent movement of the throttle into the OFF position, and because side motion of the throttles is not interconnected, prevents engine shutdown from the rear seat. Fore and aft movement of either throttle operates the fuel control unit (through a mechanical linkage) to control the normal fuel system and the emergency fuel system. Depressing the engine ignition button and moving the throttle outboard to START supplies air, fuel, and ignition to fire the combustion starter. While continuing to hold the ignition button depressed, moving the throttle inboard supplies ignition to the engine. Advancing the throttle to IDLE supplies fuel to the combustion chambers. After the engine has started, the throttles may be advanced to any desired position in the normal thrust range. At any point along the normal thrust range forward of the rear afterburner stop, afterburner operation may be initiated by moving either throttle outboard to the AFTERBURNER range. Afterburner operation is terminated when both throttles are moved inboard. The afterburner stop is canted to allow the opposite cockpit to terminate the afterburner by pulling the throttle below the aft stop. Each quadrant has a "fence" which prevents inadvertent movement of the throttle into the afterburner range and also holds the throttle outboard when afterburner operation has been selected. A mechanical pin or "gate" prevents movement of the throttle inboard, out of AFTERBURNER, unless the throttle is retarded to 2-1/2° aft. The throttle grip contains the speed brakes switch, the ignition button, and the microphone button.

NOTE

Stall-buzz can be caused by improper throttle movement. Refer to THROTTLE MANAGEMENT TO AVOID STALL-BUZZ, Section VII.

ENGINE PRESSURE RATIO GAGE

The engine pressure ratio gage (19, figure 1-8, and 21, figure 1-9) is located on the instrument panel and indicates engine thrust for use during preflight engine checks and for establishing inflight cruise thrust settings. During the preflight engine checks the gage is used to determine whether the engine thrust on the ground at full throttle is acceptable for takeoff. Engine pressure ratio (the ratio of turbine discharge pressure to pitot pressure from the pitot-static system) is indicated in increments from 1.2 to 3.4. Maximum and minimum takeoff thrust settings are obtained from the Takeoff Check Table, Section II. These limits vary with ambient temperature. Two windows are in the dial face: the lower window shows takeoff pressure ratio; the upper window shows cruise pressure ratio. The ratios that appear in the windows are set by use of a knob at the lower left corner of the instrument dial. To set the proper takeoff pressure ratio in the window, the knob is pushed in and turned until the desired numbers appear in the lower window. This adjustment also moves the takeoff thrust index marker, at the edge of the dial, to the correct dial reading. During preflight engine check the indicating pointer should fall within the arc of the takeoff thrust index marker. To set the proper cruise pressure ratio in the window, the knob is pulled out and turned until the desired numbers appear in the upper window. This adjustment also moves the triangular cruise index marker, at the edge of the dial, to the correct dial reading. Cruise pressure ratio settings are obtained from the cruise charts in the Appendix. Power to the pressure ratio gages is supplied from the ac essential bus.

TACHOMETER

The tachometer (22, figure 1-8, and 24, figure 1-9) indicates percentages of high-pressure rotor rpm, based on 8732 rpm as 100%. The rpm at which full military thrust is obtained varies with each engine. Therefore, the rpm percentage indicated by the tachometer is not necessarily a direct indication of the percentage of military thrust rpm. The military thrust rpm for each engine is placarded on the engine data plate. The main pointer is calibrated up to 100% rpm and the subpointer makes one complete revolution for each 10% change in engine rpm. By using the subpointer, up to 110% rpm can be read. The tachometer is self-energized and operates independently of the airplane electrical system.

ENGINE HOT-SECTION ANALYZER

The engine hot-section analyzer system provides exhaust gas temperature readings, exhaust thermocouple temperature spread check readings (forward cockpit only B), and an amber warning light indication of engine exhaust temperatures over 630°C. The analyzer system equipment consists of an exhaust gas temperature gage, a temperature spread check switch, a hot section factor computer, a temperature spread check computer, a data recorder, a 115-volt ac, 400 cycle emergency inverter, and six thermocouples. The analyzer system records accumulatively, on the data recorder digital counter in the left main wheel well, engine hot section factor units (temperature versus time) when the engine exhaust temperature is above 575°C. Other features of the recorder consist of two digital counters that record seconds of engine operation at temperatures above 635°C and above 700°C. The recorder also registers

cockpit general arrangement (typical)

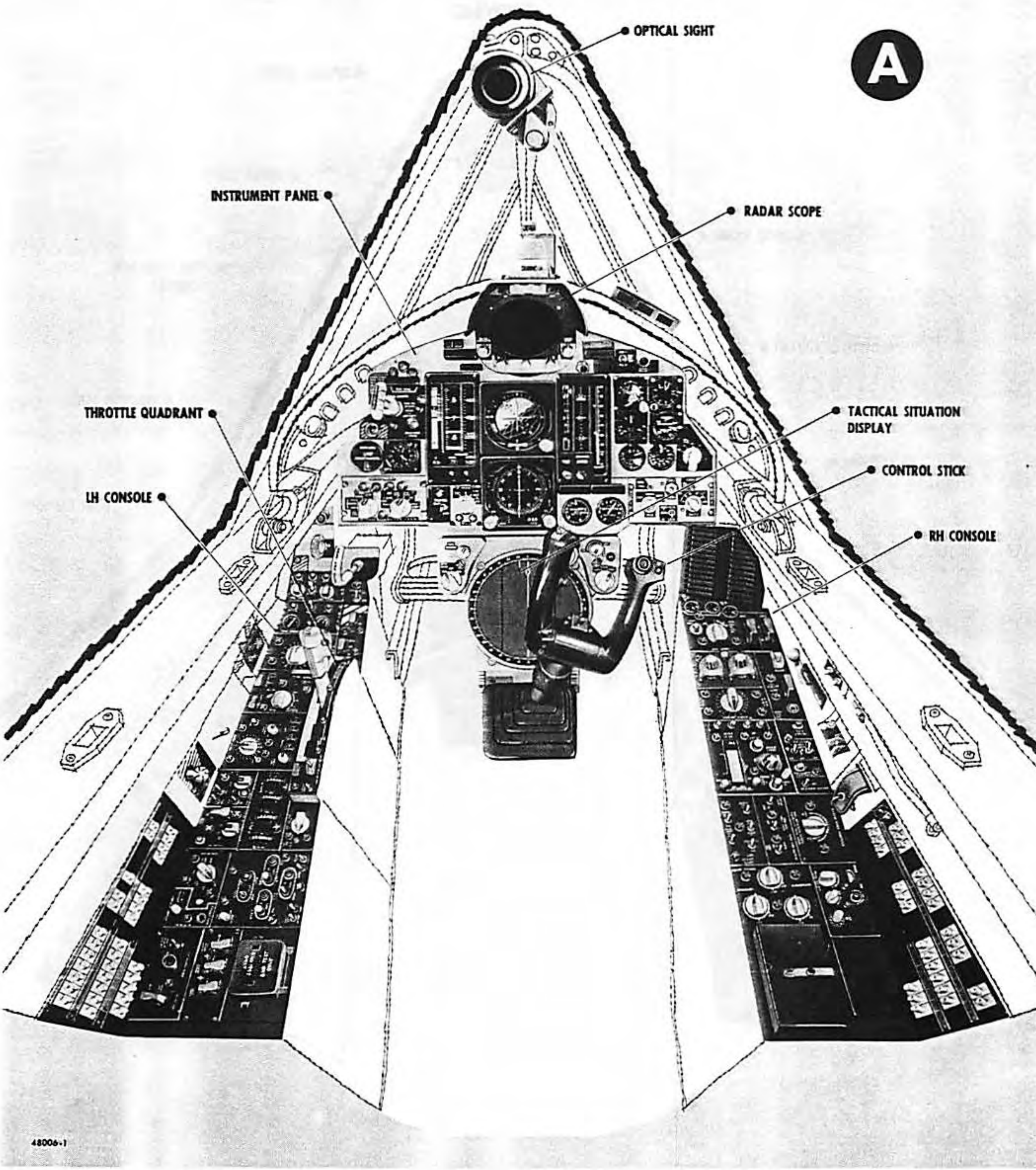


Figure 1-7 (Sheet 1 of 3)

cockpit general

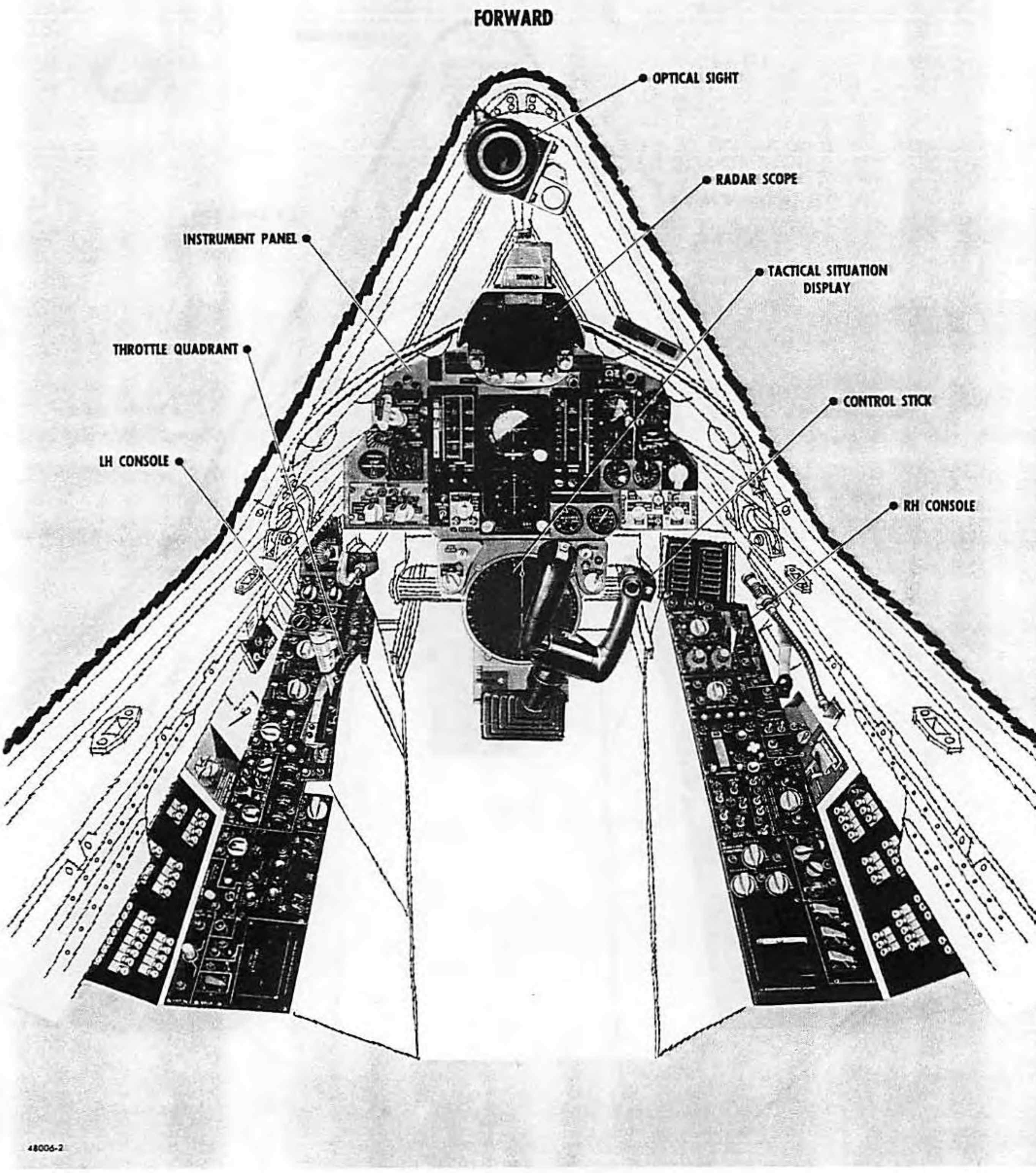


Figure 1-7 (Sheet 2 of 3)

arrangement (typical)

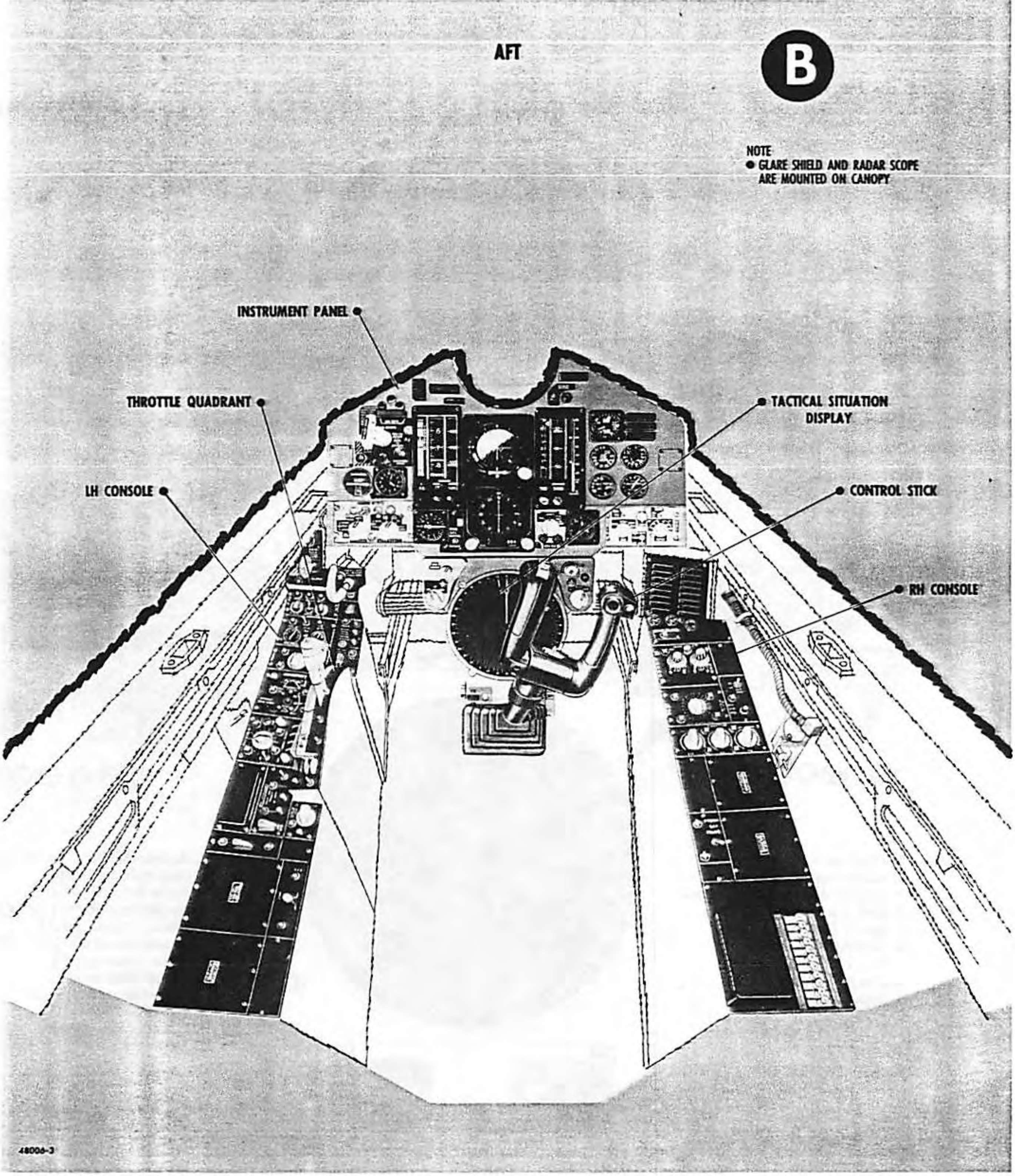


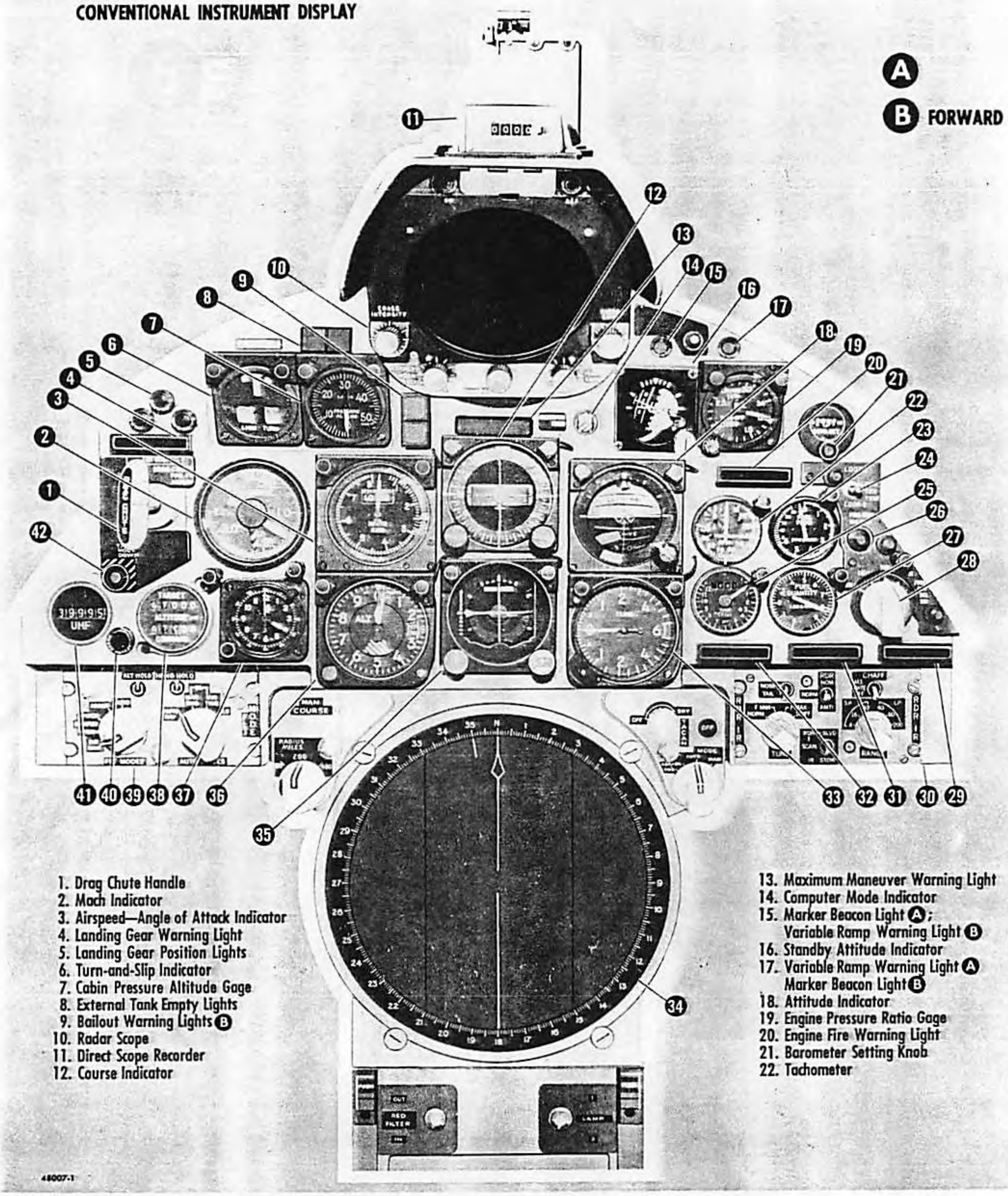
Figure 1-7 (Sheet 3 of 3)

instrument panel (typical)

CONVENTIONAL INSTRUMENT DISPLAY

A

B FORWARD



1. Drag Chute Handle
2. Mach Indicator
3. Airspeed—Angle of Attack Indicator
4. Landing Gear Warning Light
5. Landing Gear Position Lights
6. Turn-and-Slip Indicator
7. Cabin Pressure Altitude Gage
8. External Tank Empty Lights
9. Bailout Warning Lights B
10. Radar Scope
11. Direct Scope Recorder
12. Course Indicator
13. Maximum Maneuver Warning Light
14. Computer Mode Indicator
15. Marker Beacon Light A; Variable Ramp Warning Light B
16. Standby Attitude Indicator
17. Variable Ramp Warning Light A Marker Beacon Light B
18. Attitude Indicator
19. Engine Pressure Ratio Gage
20. Engine Fire Warning Light
21. Barometer Setting Knob
22. Tachometer

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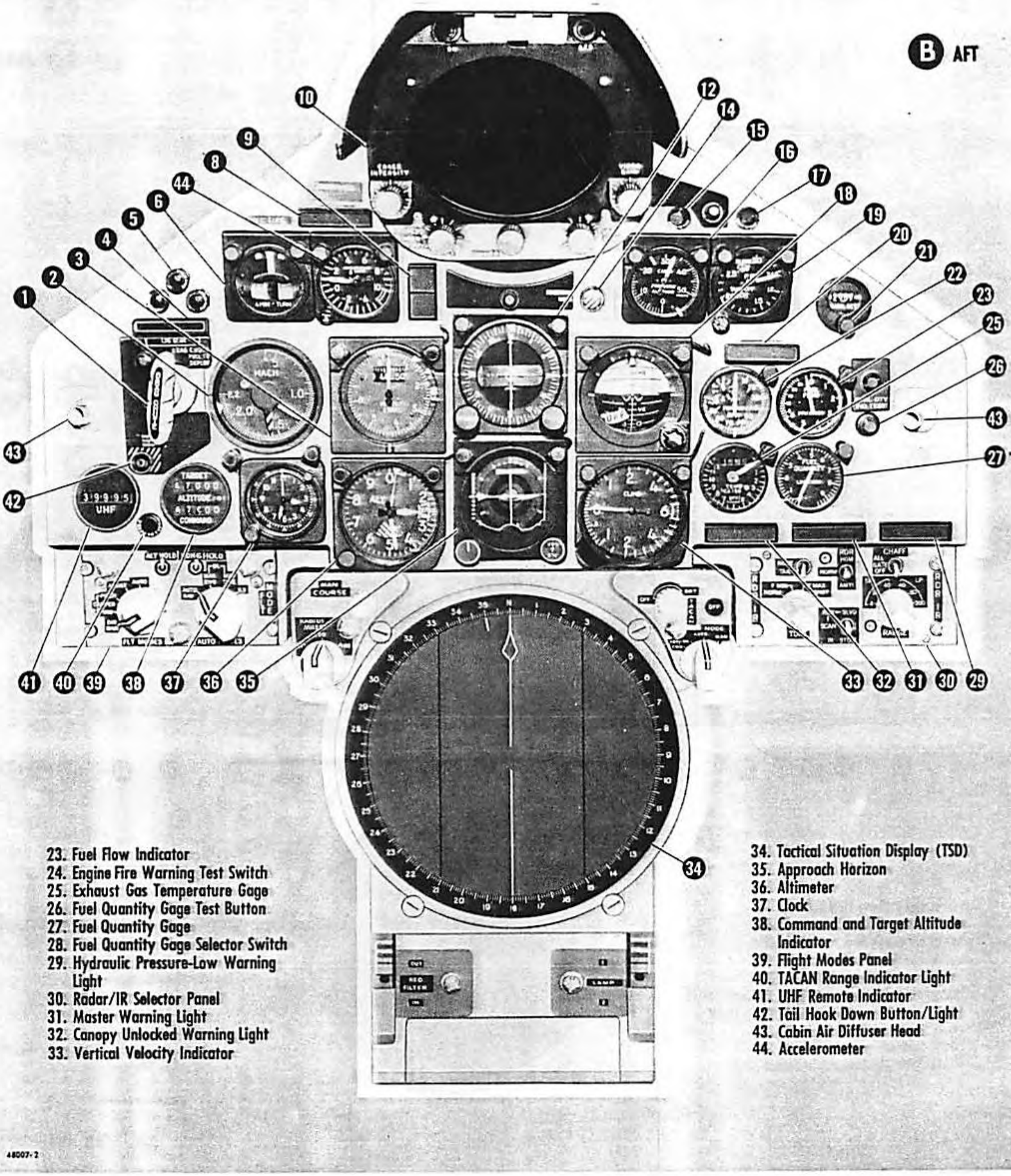
Figure 1-8

instrument panel (typical)

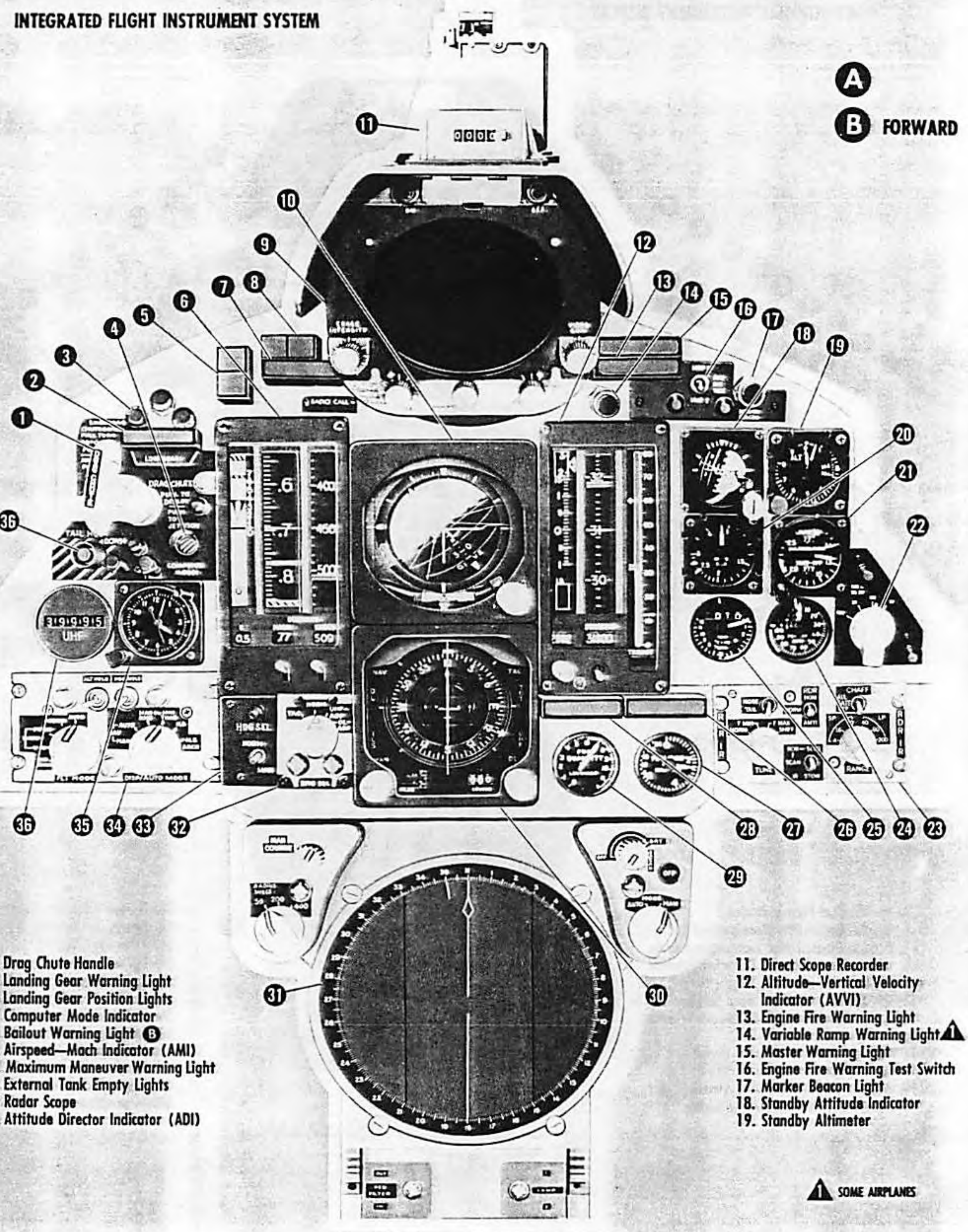
CONVENTIONAL INSTRUMENT DISPLAY



23. Fuel Flow Indicator
24. Engine Fire Warning Test Switch
25. Exhaust Gas Temperature Gage
26. Fuel Quantity Gage Test Button
27. Fuel Quantity Gage
28. Fuel Quantity Gage Selector Switch
29. Hydraulic Pressure-Low Warning Light
30. Radar/IR Selector Panel
31. Master Warning Light
32. Canopy Unlocked Warning Light
33. Vertical Velocity Indicator
34. Tactical Situation Display (TSD)
35. Approach Horizon
36. Altimeter
37. Clock
38. Command and Target Altitude Indicator
39. Flight Modes Panel
40. TACAN Range Indicator Light
41. UHF Remote Indicator
42. Tail Hook Down Button/Light
43. Cabin Air Diffuser Head
44. Accelerometer

48007-2

instrument panel (typical)



1. Drag Chute Handle
2. Landing Gear Warning Light
3. Landing Gear Position Lights
4. Computer Mode Indicator
5. Bailout Warning Light B
6. Airspeed—Mach Indicator (AMI)
7. Maximum Maneuver Warning Light
8. External Tank Empty Lights
9. Radar Scope
10. Attitude Director Indicator (ADI)
11. Direct Scope Recorder
12. Altitude—Vertical Velocity Indicator (AVVI)
13. Engine Fire Warning Light
14. Variable Ramp Warning Light A
15. Master Warning Light
16. Engine Fire Warning Test Switch
17. Marker Beacon Light
18. Standby Attitude Indicator
19. Standby Altimeter

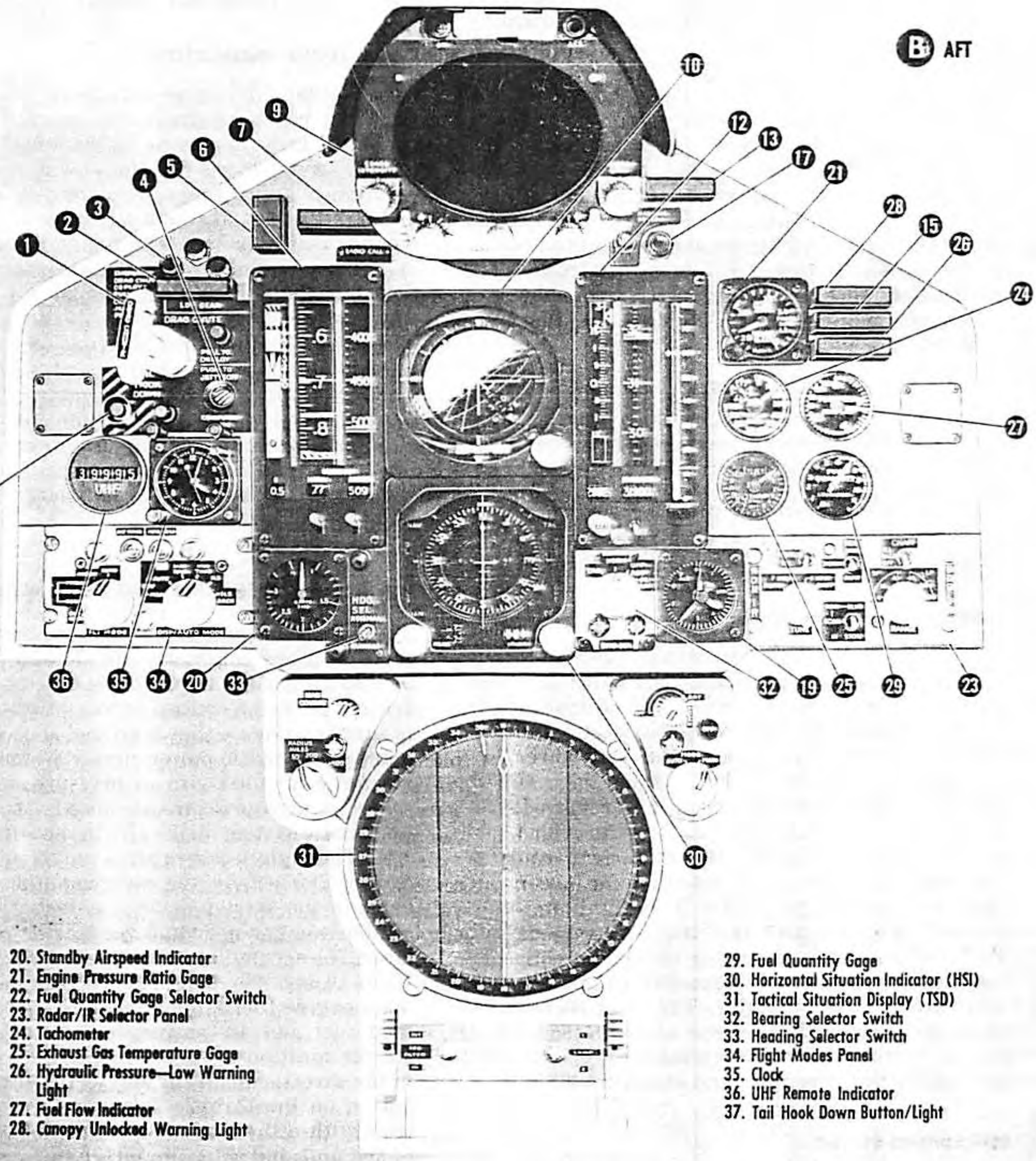
A SOME AIRPLANES

48039-1

Figure 1-9

instrument panel (typical)

INTEGRATED FLIGHT INSTRUMENT SYSTEM



20. Standby Airspeed Indicator
21. Engine Pressure Ratio Gage
22. Fuel Quantity Gage Selector Switch
23. Radar/IR Selector Panel
24. Tachometer
25. Exhaust Gas Temperature Gage
26. Hydraulic Pressure—Low Warning Light
27. Fuel Flow Indicator
28. Canopy Unlocked Warning Light
29. Fuel Quantity Gage
30. Horizontal Situation Indicator (HSI)
31. Tactical Situation Display (TSD)
32. Bearing Selector Switch
33. Heading Selector Switch
34. Flight Modes Panel
35. Clock
36. UHF Remote Indicator
37. Tail Hook Down Button/Light

48039-2

"OVER" for engine temperatures over 725°C; "OVER" for temperatures over 800°C; and "OVER" when the exhaust thermocouple temperature differential exceeds specified limits (temperature spread check). The temperature spread check system measures the temperature differential between the coldest and hottest of the exhaust thermocouples. On demand, the spread check system provides a cockpit reading of the number of degrees of temperature differential and a warning light indication if the differential exceeds 140°C (forward cockpit only Ⓑ). The temperature spread warning light feature is a secondary function of the exhaust overtemperature warning light. The analyzer system is in operation at any time there is electrical power on the airplane either from an external power source or from the airplane systems. Analyzer power for normal operation is supplied by the ac nonessential bus. In the event of ac power failure, power from the dc essential bus automatically operates the analyzer system emergency inverter, thereby providing the required 115 volt power.

NOTE

Temperature spread readings are not valid when the analyzer system is operating on emergency inverter power.

The analyzer circuitry is protected by two fuses on the right console. For additional information on the analyzer system, refer to T.O. 1F-106A-2-9.

Exhaust Gas Temperature Gage

The exhaust gas temperature gage (25, figure 1-8, 25, figure 1-9) on the instrument panel provides engine exhaust temperature by digital readings in 2°C increments and by conventional sweep hand readings in 50°C increments. An electrical power failure flag displays "OFF" when the gage is not operating. When electrical power is removed, the gage will retain the last reading provided by the analyzer system. An amber overtemperature warning light is incorporated in the lower right side of the gage face. This light is illuminated when exhaust gas temperature exceeds 630°C. The light is also used in conjunction with the engine thermocouple temperature spread check (forward cockpit only Ⓑ). The light illuminates when the temperature spread check is performed and the temperature differential between the coldest and hottest thermocouple exceeds 140°C.

EGT Spread Button

The EGT spread button, placarded "EGT SPREAD," is installed on the right console (9, figure 1-11) (forward cockpit only Ⓑ). When the button is depressed, the exhaust gas temperature gage reading will indicate the engine thermocouple temperature differential reading. Illumination of the exhaust gas temperature gage warning light at this time is an indication that the temperature differential between the coldest and hottest of the engine thermocouples has exceeded 140°C. Release of the EGT spread button will remove the spread reading, extinguish the warning light (if illuminated), and permit resumption of the exhaust gas temperature readings.

FUEL FLOW INDICATOR

The fuel flow indicator (23, figure 1-8, and 27, figure 1-9) registers the rate in pounds per hour of fuel flow into the engine as measured by a transmitter located in the fuel line downstream from the hydromechanical fuel control unit. It is possible to obtain readings through a range of 0 to 24,000 pounds per hour flow. The fuel flow indicator does not indicate fuel flow to the afterburner system. Power is supplied from the 26-volt instrument transformer, operating from the ac essential bus. Occasionally fuel flow fluctuations of approximately 400 pph, caused by opening and closing of the pressurizing and dump valve, will be experienced. The rpm fuel flow point at which these fluctuations occur will vary; however, experience indicates that the range of greatest probability is between 80 to 85% rpm with fuel flow of 3000 to 3500 pph.

ENGINE STARTER AND ENGINE IGNITION SYSTEM

The engine is equipped with a combustion starter which is a self-contained unit. The starter consists of a small gas turbine, a drive assembly, and electrically operated control valves. High-pressure air is supplied from either a ground source or the airplane's pneumatic power supply system. When the starter is supplied with air pressure from a ground source, a high-pressure air hose is attached to the pneumatic system filler valve in the left main wheel well. The high-pressure air is regulated to approximately 290 psi by the air regulator and shutoff valve in the starter prior to entering the starting combustion chamber and the starter fuel accumulator. The accumulator holds fuel for a start lasting 13 to 16 seconds. Fuel supply is tapped from the main engine fuel supply line. The electrically operated fuel and air valves in the starter, and the starter ignition system, are actuated by switches in the throttle quadrant and by the engine ignition button on the throttle lever grip. Automatic controls within the starter unit prevent overspeeding of the unit and shut off starter fuel and air valves if combustion stops or when the speed of the engine increases to approximately 35% engine rpm. Power for all electrical controls is taken from the dc essential bus. The engine ignition system consists of a pair of high energy capacitor discharge systems, two ignitor plugs located in combustion chambers Nos. 4 and 5, and the engine ignition button which is electrically connected with the starter control circuit. The ignition circuit is energized only during engine starting, as combustion is continuous once the engine starts. The afterburner is ignited by "hot-streak" ignition and requires no electrical ignition for operation.

NOTE

> The starter is inoperative during an air start. Engine rotation is dependent upon windmilling effect only.

Engine Ignition Button

The engine ignition button (figure 1-6) is located on the throttle. For ground starting, the ignition and starter circuits are electrically interconnected and it is necessary to depress the engine ignition button and move the forward throttle outboard to START to activate the ignition circuit. Ignition is then supplied to the engine by moving the throttle inboard to IDLE while continuing to depress the engine ignition button. On Ⓑ airplanes a ground start cannot be accomplished from the aft seat. When airborne, a start can be accomplished at any throttle position by depressing the engine ignition button. The engine starter is bypassed, and windmilling effect provides engine rotation. On Ⓑ airplanes airstarts can be made from the aft seat. During ground or flight operation, depressing the engine ignition button (when the surface and engine anti-icing switch is in AUTO) heats the ice detector probe to remove any accumulated ice. The engine ignition button receives power from the dc essential bus.

Engine Ignition Disconnect Switch

A two-position engine ignition disconnect switch, placarded "Engine Ignition," is located in the right main wheel well. The switch has positions ARM and DISARM and is used to interrupt power between the ignition power circuit breaker and the engine ignitors. With the switch in DISARM, the engine can be air-motored without ignition in the combustion chambers. The switch receives power from the 28-volt dc essential bus.

Starter Ignition Disarm Switch

A starter ignition disarm switch is located in the right main wheel well. The switch has two positions, ON and OFF, and is used to interrupt power to the starter ignition system. The OFF position is used to air-motor the starter without firing the combustion part of the starter system. The switch receives power from the dc essential bus.

ENGINE COOLING

The engine area is divided into two compartments: the burner and turbine compartment, and the engine accessory compartment. A fire resistant titanium shroud around the burner and turbine compartment separates it from the engine accessory compartment and from the airplane skin. Cooling is completely automatic and is accomplished by routing ventilating air through these compartments. In flight, the burner and turbine compartment is ventilated by air tapped from the engine air inlet duct, or if negative pressure develops between the shroud and engine, the burner and turbine compartment is ventilated by low-pressure compressor bleed air. For ground operation, the burner and turbine compartment is ventilated by low-pressure compressor bleed air. During flight the engine accessory compartment is ventilated by air taken from engine inlet air duct and routed through the constant-speed drive air-oil cooler and the engine air-oil cooler. In the accessory compartment, ventilation air flows aft through the area between the shroud and the airplane skin and exhausts between the tail cone and the shroud. For ground operation, flow is reversed through the constant-speed drive air-oil cooler by negative pressure in the engine air inlet ducts. To control excessive pressure in the fuselage skin, airflow into the accessory compartment is automatically controlled by positioning the constant-speed drive air-oil cooler shutoff valve and the engine air-oil cooler shutoff valve. When pressure differential between the engine accessory compartment and ambient reaches 1.5 psi, the constant-speed drive air-oil cooler shutoff valve goes to its minimum position, permitting only a small bypass airflow into the accessory compartment. If pressure differential increases to 2.0 psi, the air pressure is bled from the engine air-oil cooler valve actuator, preventing it from opening the valve. Should pressure differential rise to 3.0 psi the engine compartment overpressure warning light illuminates, automatic control is discontinued, and pneumatic pressure closes the constant-speed drive air-oil cooler shutoff valve while electrical power is removed from the temperature control. With electrical power off, the engine air-oil cooler valve remains in the spring-loaded closed position. The engine compartment overpressure warning light then remains illuminated for the duration of the flight; however, when differential pressure falls to 0.75 psi, the constant-speed drive and engine air-oil cooler valves return to automatic control.

Engine Compartment Overpressure Warning Light

The engine compartment overpressure warning light (17, figure 1-30) is located on the warning light panel and displays "ENG COMPT O PRESS" when pressure has reached 3.0 psi above ambient in the engine accessory compartment. Illumination of the light indicates that pressure must be reduced in the engine accessory compartment to prevent damage to the fuselage from high pressure. The light is also an indication that the high-pressure pneumatic system has been used to operate the constant-speed drive air-oil cooler air valve actuator and if pressure is not reduced, the actuator will partially deplete the high-pressure pneumatic system. Once illuminated, the light will remain on for the duration of the flight even though pressure differential may fall below 3.0 psi. The light receives power from the dc essential bus.

ENGINE AFTERBURNER SYSTEM

Operation of the afterburner is controlled by the throttle. When the throttle is placed outboard to the AFTERBURNER range, the afterburner shutoff valve is opened by power from the dc essential bus. Fuel flow to the afterburner spray bars is regulated by the afterburner metering valve which is controlled by compressor discharge pressure (which is governed by flight conditions and engine speed). The metered afterburner fuel flow is routed to the afterburner ignitor to supply "hot-streak" ignition from the number three combustion chamber to the afterburner. At the same time afterburner fuel pressure to the exhaust nozzle control pneumatically opens the exhaust nozzles during afterburner operation. When engine thrust is varied, afterburner thrust will also vary, since afterburner fuel flow is governed by compressor discharge pressure. For additional information on this system, refer to T.O. 1F-106A-2-4.

NOTE

The afterburner can be used while the engine is operating on the emergency fuel control system if the main system fails. However, rapid throttle movement should be avoided to prevent overspeed, overtemperature, compressor stalls, and flameouts.

AFTERBURNER EXHAUST NOZZLE

The two-position exhaust nozzle is closed or opened to provide the proper exhaust opening for either normal or afterburner engine operation. The nozzle flaps are moved to the full open position during afterburner operation and return to the minimum nozzle area when the afterburner is not in use. Positioning of the nozzle flaps is accomplished automatically by the exhaust nozzle control unit. Emergency override control of the exhaust nozzle is not possible.

Idle Thrust Control Switch

The idle thrust control switch (22, figure 1-10) is located on the left subconsole. On Ⓑ airplanes the switch is located in the forward cockpit only. The switch is used only when taxiing to reduce thrust. The switch is placarded "Idle Thrust Control" and has positions ON and OFF. When the switch is ON, the exhaust nozzle is held open by the action of N_2 compressor bleed air which is routed to the exhaust nozzle actuating cylinders. With the nozzle open, effective thrust at idle is reduced approximately 40%. Slower taxi speeds are then possible with a minimum of brake wear. When the switch is OFF, the exhaust nozzle is closed. Power is supplied by the dc nonessential bus.

NOTE

- The switch is operative only when the weight of the airplane is on the left main landing gear.
- Thrust in the afterburner range is not affected by the idle thrust control switch.

AFTERBURNER IGNITOR

When the afterburner system is actuated, fuel from the afterburner metering valve is directed to the ignitor unit (see figure 1-5). This metered fuel flow actuates the ignitor unit which momentarily injects fuel into the number three combustion chamber, thereby creating a local excessively rich fuel-air mixture. The excess fuel forms a long flame front that continues to burn past the turbines. The extended flame provides "hot-streak" ignition to ignite the fuel being discharged from the afterburner fuel spray bars. The ignitor is actuated only when full pressure is built up within the afterburner manifold, so that fuel is available at the spray bars when the ignitor introduces fuel to provide "hot-streak" ignition.

AFTERBURNER EXHAUST NOZZLE CONTROL UNIT

The afterburner exhaust nozzle control unit provides automatic control of the iris type exhaust nozzle. When the throttle is moved outboard to AFTERBURNER the electrically operated shutoff valve is opened allowing afterburner fuel pressure to open the engine nozzle control valve (by power from the dc essential bus), permitting engine bleed air to enter the exhaust nozzle control unit. This fuel pressure actuates a valve within the control unit which directs engine bleed air pressure to the nozzle actuating cylinders to open the nozzle segments. Moving the throttle inboard from AFTERBURNER range closes the shutoff valve so that air pressure is no longer supplied to the exhaust nozzle control unit. The valve in the control unit is then positioned to route engine bleed air pressure to close the nozzle segments for non-afterburning engine operation. An additional feature is incorporated into this system which aids afterburner ignition at high altitude by delaying the nozzle opening until the afterburner has ignited. At low altitude the afterburner nozzle opens as the throttle is moved outboard or slightly before afterburner ignition occurs.

AFTERBURNER EMERGENCY SHUTOFF

Should a failure of the afterburner electrically operated shutoff valve be encountered during afterburner operation, an afterburner shutdown may be accomplished by retarding the throttle to a point slightly aft of the afterburner stop. A hydraulically operated pilot valve in the afterburner fuel control unit is then returned to its spring-loaded closed position which will shut off all fuel to the afterburner. A mechanical lock then prevents the pilot valve from moving to the open position when the throttle is readvanced. During a normal afterburner start, the mechanical lock is momentarily moved out of position by a cam that is rotated by operation of the afterburner shutoff valve.

OIL SUPPLY SYSTEM

The engine oil system supplies the engine with oil for lubrication and cooling. The engine oil tank has a usable capacity of 18 quarts and a total capacity of 22 quarts. Oil from the oil tank is first taken into a gear-type boost pump, from which it is routed through an air-oil cooler, then through or around (depending upon the oil temperature) a fuel-oil cooler to the pressure stage of the main oil pump. The main oil pump then supplies oil under pressure through a filter to the engine gears and bearings. An oil pressure relief valve downstream of the filter is set to maintain the proper pressure differential across the oil metering jets in the engine. Four individual oil scavenge pumps return the oil to the engine oil tank. Pump pressure output is routed to an oil pressure transmitter and pressure switch which controls the engine oil pressure gage and warning light. An engine oil breather pressurizing valve is installed to regulate pressures in the bearing compartments and to maintain oil flow at all altitudes. This aneroid type valve is open at sea level and closes fully at approximately 34,000 (±4000) feet altitude. The breather vents overboard through the lower right-hand side of the fuselage. See figure 2-8, Servicing Diagram, for oil specifications. For additional information on this system, refer to T.O. 1F-106A-2-4.

OIL PRESSURE GAGE

An oil pressure gage (41, figure 1-11) is located on the right-hand console. This gage indicates output pressure of the engine oil pump in pounds per square inch. The oil pressure gage is powered by the ac essential bus.

NOTE

> The oil pressure gage needle may normally fluctuate as much as ±2.5 psi.

OIL PRESSURE-LOW WARNING LIGHT

The oil pressure-low warning light (15, figure 1-30) is located on the master warning light panel, and when illuminated displays "OIL PRESS." When oil pressure is increasing, the light should go out before pressure reaches 40 psi. When oil pressure is decreasing, the light will not illuminate until pressure drops to 37 ± 2 psi. The oil pressure-low warning light is controlled by a pressure switch in the oil line, and receives power from the dc essential bus.

NUCLEONIC OIL QUANTITY INDICATING SYSTEM.

The nucleonic oil quantity indicating system employs a low energy radiation source and a detector tube mounted externally on the engine oil tank. The radiation source is positioned to provide an even distribution of emitted radiation through the tank wall and oil, to the detector. The oil contained in the tank attenuates a portion of the emitted radiation proportional to oil quantity, and the remainder is received by the detector. From the amount of this attenuation and detection, the system can determine the quantity of oil in the tank.

NUCLEONIC OIL QUANTITY INDICATOR

The oil quantity indicator is installed above the master warning light panel. In **B** aircraft, it is installed in the front cockpit only. The gage is marked in 1/16 graduations from empty (E) to full (F). The E through F range measures the amount of usable oil in the oil tank. The operating range is color coded as follows: Empty to three-eighths - Red; Three-eighths to five-eighths - Amber; Five-eighths to full Green. The gage provides a continuous oil tank quantity indication to the pilot.

One characteristic of the nucleonic system gage is the "wandering" of needle. It is normal for the needle to fluctuate up to plus or minus two graduations even with wings level, unaccelerated flight. Movements of the needle in excess of this "wandering" are indications of actual oil quantity variations in the oil tank.

During turns, accelerations, and decelerations, the oil quantity in the tank varies slightly and this is reflected on the gage. This variation should not normally exceed plus or minus two graduations either side of normal needle movement. During maneuvers, positive g loads may cause the indicator to show a slight increase in oil quantity while negative g loads may show a decrease in oil quantity.

With power off or with gage or system failure, the gage should read overfull with the needle pointing steady at approximately one graduation beyond the F. Prior to take-off during the Engine Check, the needle movement should be in the Green range. During flight maneuvers, the needle may momentarily move into the Amber range. However, this is considered normal. Continuous needle movement in the Amber range or illumination of the Oil Low Warning Light is an indication of low oil quantity and the recommended Emergency Procedure should be followed. The oil quantity gage is powered by the A/C essential bus.

OIL QUANTITY LOW WARNING LIGHT

The oil quantity low warning light is actuated when the oil quantity in the tank drops to the one-half full indication plus or minus one-half of a graduation. The oil low warning light should not illuminate during flight unless the oil level has decreased to approximately one half of the usable oil in the tank.

On Ⓐ aircraft, the warning light is located on the master warning light panel and illuminates "OIL QUAN LOW." On Ⓑ aircraft, the amber "OIL LOW" indicator light is located on the mounting bracket that supports the oil quantity gage. Illumination of either oil quantity low warning lights will illuminate the master warning light. The oil low warning lights receive power from the dc essential bus.

FUEL SUPPLY SYSTEM Ⓐ

Fuel is supplied to the engine fuel control system by the airplane fuel system (figure 1-12) which consists of four tanks in each wing and an F (fuselage) tank located aft of the cockpit. To improve airplane performance at supersonic speeds, an automatic fuel transfer system is installed to shift the cg forward and aft under specified flight conditions. To facilitate this cg change as desired, fuel is transferred between the F tank and two T (transfer) tanks which are located along the aft wing sections. Provisions are also included for the installation of two 360-gallon external wing tanks. The airplane may be refueled to full internal fuel or full internal and external fuel configuration. The three main tanks in each wing function as an individual tank and the fuel systems in each wing operate independently of each other. (The pilot cannot transfer fuel from one wing to the other.) All internal fuel tanks are the integral type composed of airplane structure and are not self-sealing. For fuel tank capacities, refer to the Fuel Quantity Data Table, figure 1-15. Fuel specifications are shown in figure 2-8. For additional information on this system, refer to T.O. 1F-106A-2-5.

NOTE

Fuel quantities quoted in this manual are exact. However, they may vary because of temperature, probe tolerances, and gage tolerances.

cockpit left side (typical)

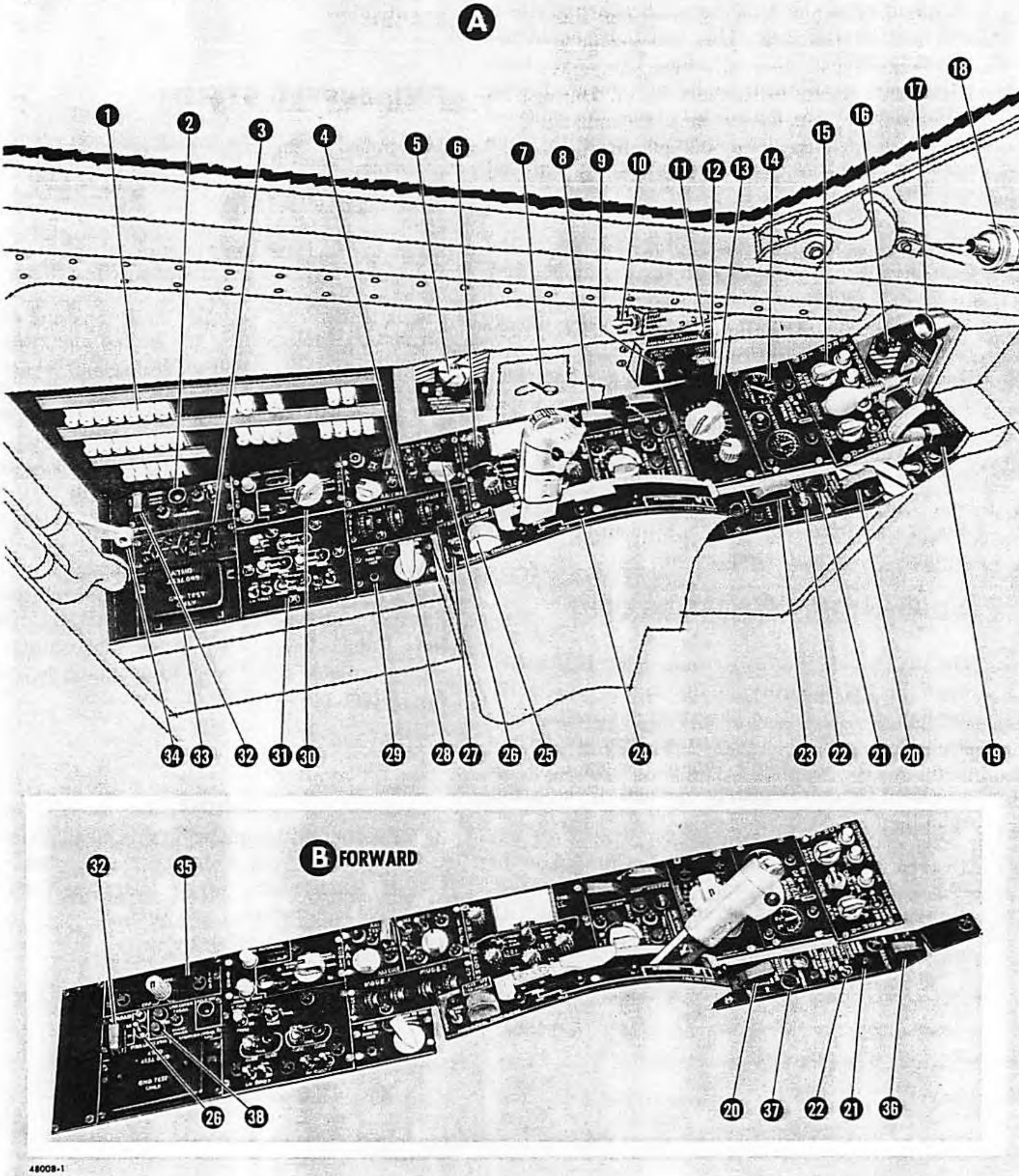
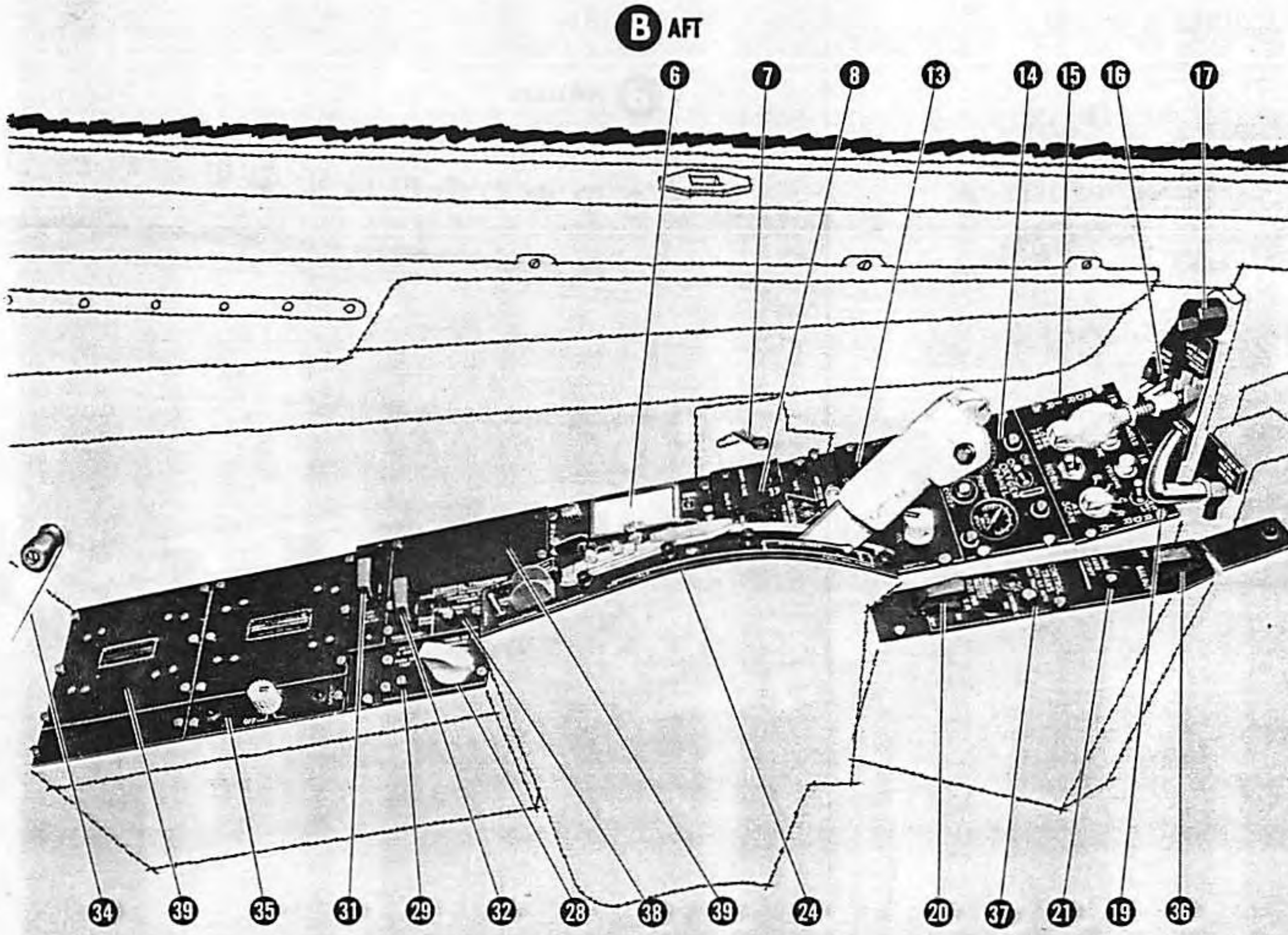


Figure 1-10

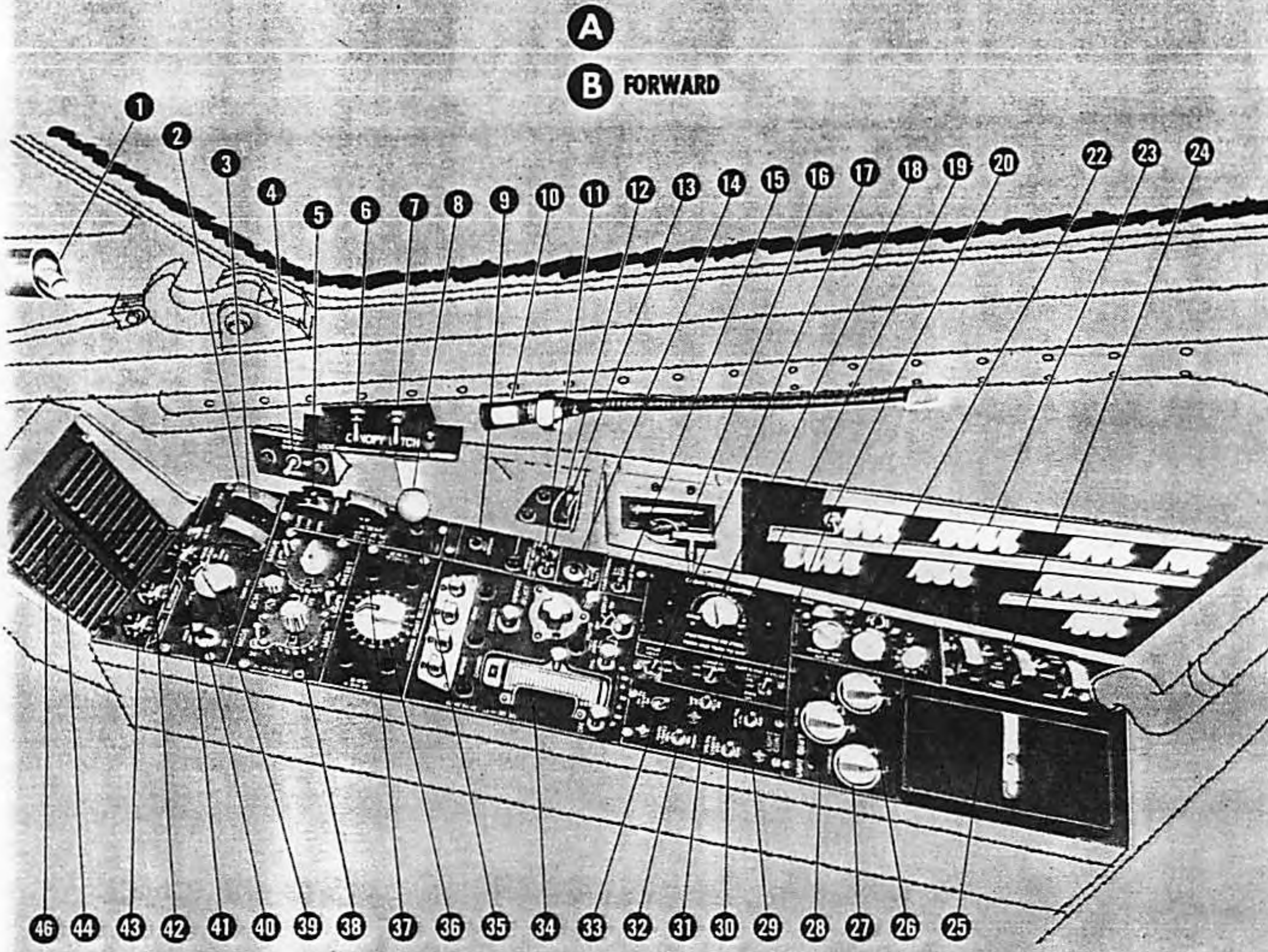
cockpit left side (typical)



1. LH Cockpit Fuse Panel
2. Armament Recycle Button
3. Air Refueling Panel A
4. IFF/SIF Control Panels
5. Ram Air Turbine (RAT) Handle
6. Communications Frequency Selector Panel (UHF)
7. Pressure Suit Handle
8. Armament Control Panel (A and B Forward) Armament Control Monitor Panel (B Aft)
9. Cockpit No-Fog and Ventilated Suit Switch
10. Reset/MBL Switch
11. AIR-2A Arm/Safe/Monitor Power Circuit Breaker
12. Landing and Taxi Light Switch
13. ILS Channel Selector Panel
14. Oxygen Control Panel
15. Radar/IR Control Panel
16. Landing Gear Control Panel
17. External Wing Tanks Release Button
18. Cabin Air Diffuser Head
19. Landing Gear Emergency Extension Handle
20. Master Electrical Power Switch
21. Landing Gear Audio Warning Cutoff Button
22. Idle Thrust Control Switch
23. CG Control Switch A
24. Throttle Quadrant
25. Fuel Control Switch
26. Pitch 'G" Limit Test Switch
27. Rudder Trim Switch
28. Mask De-Fog Rheostat
29. Anti-G Suit Button
30. MA-1 Power Control Panel
31. Fuel Control Panel (A and B Forward), Fuel Shutoff Switch (B Aft)
32. Variable Ramp Switch
33. MA-1 Test Panel
34. Cabin Air Selector Handle
35. Intercom Volume Control Panel
36. Bailout Switch
37. Automatic Flight Control System Transfer Panel B
38. Control Transfer Panel B
39. Transfer Relay Panels (B Aft)

48008-2

cockpit right side (typical)

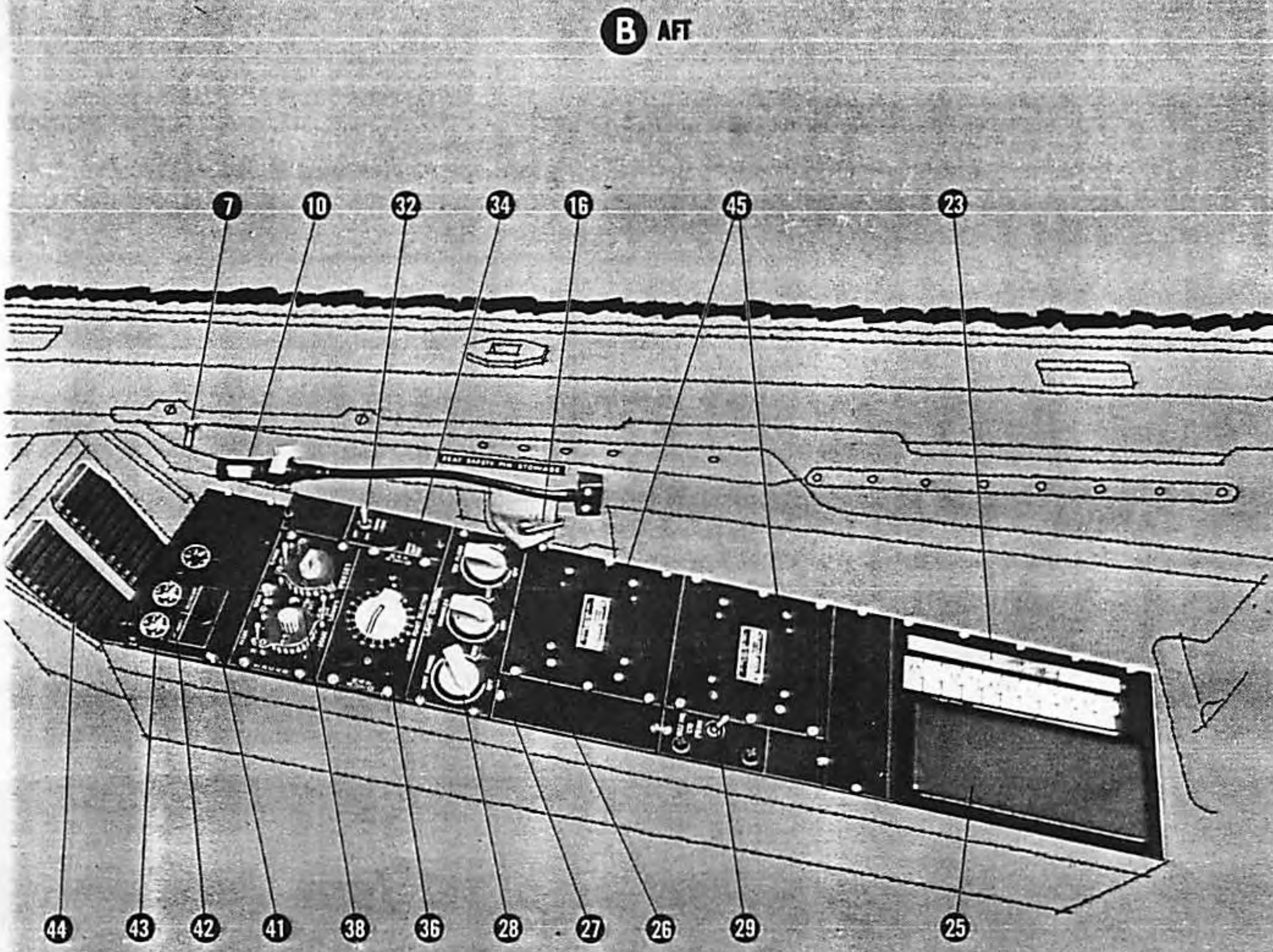


1. Cabin Air Diffuser Head
2. DC Generator Switch
3. AC Generator Switch
4. #3 Fuel Tank Switch
5. Canopy Switch
6. ATG Switch (A Location Only)
7. Map Reading Light Switch
8. Canopy Latch Handle
9. EGT Spread Button
10. Map Reading Light
11. Windshield Anti-Icing, Anti Fog Switch—RH
12. Windshield Anti-Icing, Anti Fog Switch—LH
13. Emergency AC Generator Switch (Some Airplanes)
14. Engine Anti-Ice Warning Test Button
15. Rain Removal Switch
16. Ejection Seat Ground Safety Pin (Stowed)
17. Pitot Heat Switch
18. Cabin Temperature Control Knob
19. Canopy Anti Fog Switch
20. Surface and Engine Anti-Icing Switch
21. DELETED
22. Compass Control Panel
23. RH Cockpit Fuse Panel
24. Air Refueling Panel B

48009-1

Figure 1-11 (Sheet 1 of 2)

cockpit right side (typical)



25. Map and Data Case
26. Cockpit Floodlights Powerstat
27. Console Lights Powerstat
28. Instrument Panel Lights Powerstat
29. Augmentation Lights Switch
30. Formation-Navigation Lights Dimmer Switch
31. Formation-Navigation Lights Switch
32. Warning Lights Dimmer Switch
33. Thunderstorm Lights Switch
34. Data Link Control Panel
35. Warning Lights Test Button
36. Auto-Navigation Homing Point Selector Switch
37. Directional Data Link Antenna Switch
38. TACAN Control Panel
39. Cabin Air-Selector Switch
40. Refrigeration Unit Switch
41. Oil Pressure Gage
42. Secondary Hydraulic System Pressure Gage
43. Primary Hydraulic System Pressure Gage
44. Master Warning Light Panel
45. Transfer Relay Panels (B Aft)
46. IFF Caution Light

48009-2

Figure 1-11 (Sheet 2 of 2)

airplane fuel system

(typical)

A

1. VALVES OPEN ONLY DURING TRANSFER OF FUEL FROM THE TRANSFER TANKS AND 390 POUNDS FROM THE FUSELAGE TANK.
2. THE FUEL LOW WARNING SWITCHES ALSO ACTUATE AN AIR SCAVENGE SYSTEM WHICH RECOVERS FUEL NORMALLY UNAVAILABLE FROM THE TRANSFER LINES, FUSELAGE AND TRANSFER TANKS.
3. FUEL PRESSURE ACTUATED SHUTOFF VALVES WILL SHUTOFF GROUND REFUELING WHEN TANKS ARE FILLED. NO. 3 TANK VALVE WILL NOT CLOSE UNTIL TANKS 1, 2 AND 3 ARE FILLED DURING FLIGHT, THE VALVES WILL OPEN WHEN THE FUEL LEVEL IN EACH NO. 3 TANK DECREASES TO 1200 POUNDS TO ALLOW THE TRANSFER FUEL TO REPLENISH NO. 3 TANKS.
4. WHEN A SHUTOFF VALVE IS IN OTHER THAN THE OPEN POSITION, ITS RESPECTIVE WARNING LIGHT ON THE FUEL CONTROL PANEL AND THE "FUEL VALVE CLOSED" WARNING LIGHT ON MASTER WARNING LIGHT PANEL WILL ILLUMINATE.
5. SOME AIRPLANES
6. AIRPLANES WITHOUT F TANK EMERGENCY PRESSURE

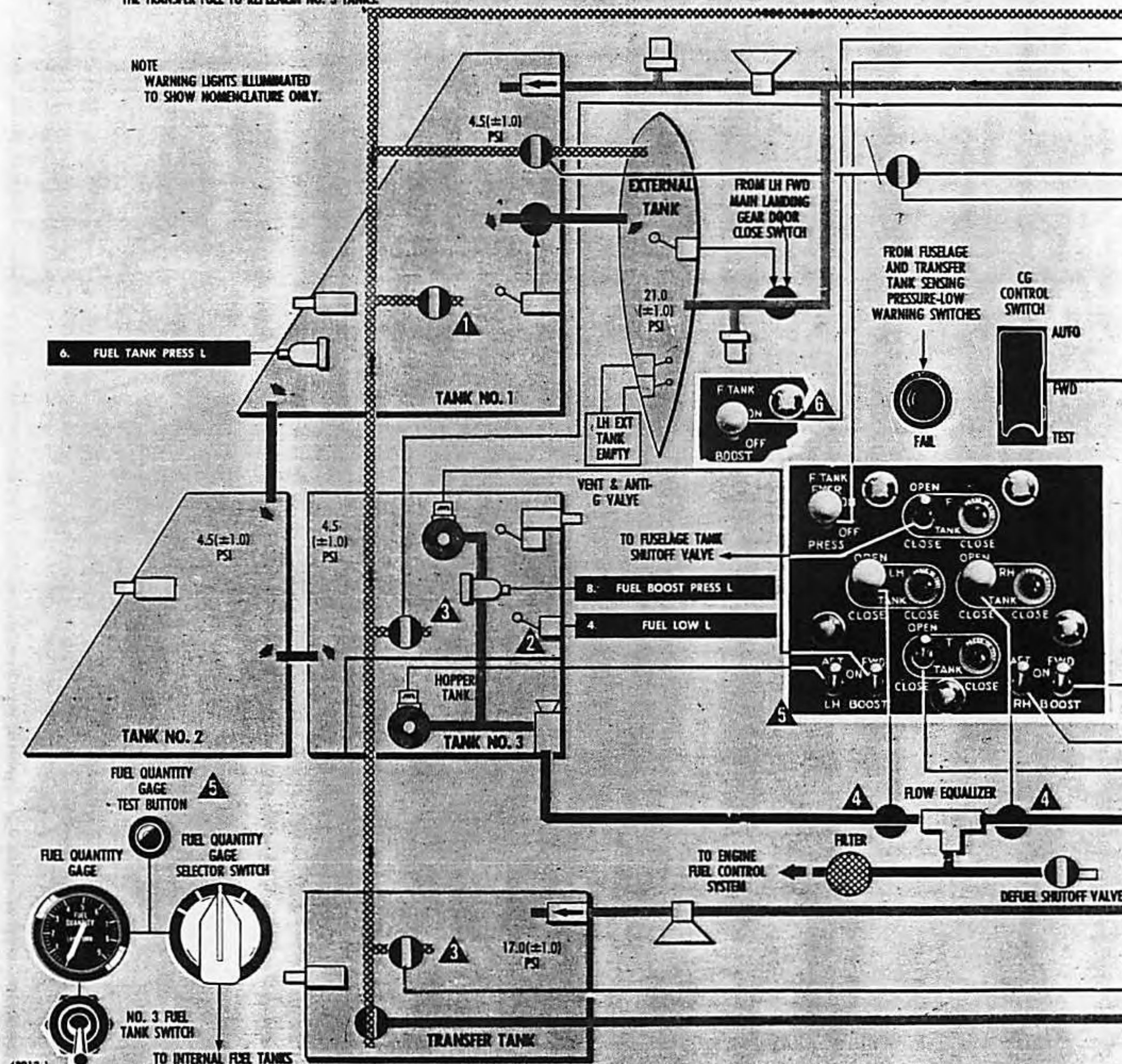


Figure 1-12 (Sheet 1 of 4)

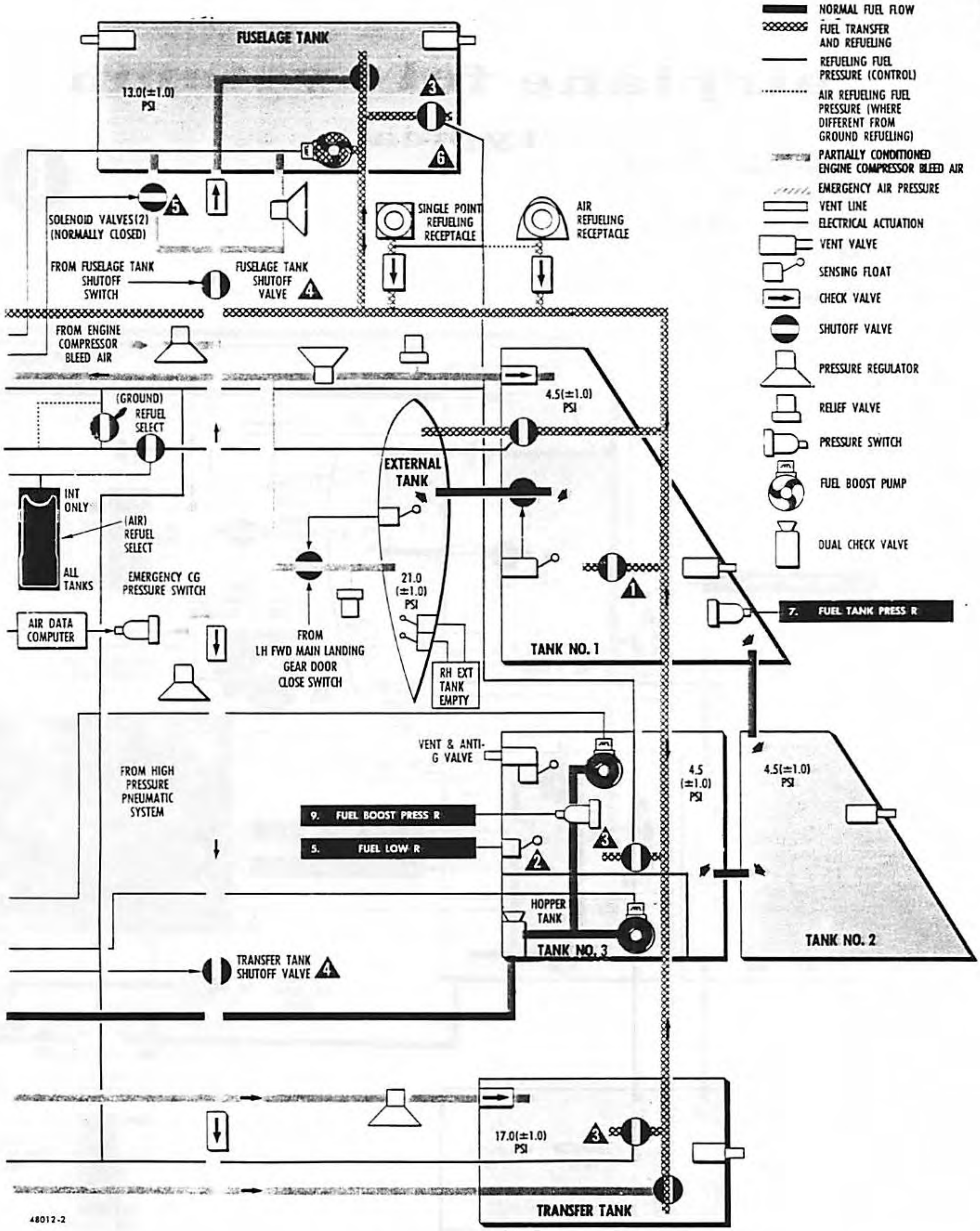


Figure 1-12 (Sheet 2 of 4)

airplane fuel system
(typical)

B

1 OPENS AFTER EXTERNAL TANK DEPLETION AND WHEN 227 POUNDS USED FROM TANK NO. 1
2 CLOSED EXCEPT DURING SCAVENGE CYCLE
3 CLOSED EXCEPT WHEN REFUELING
4 DELETED
5 SOME AIRPLANES
6 AIRPLANES WITHOUT F TANK EMERGENCY PRESSURE

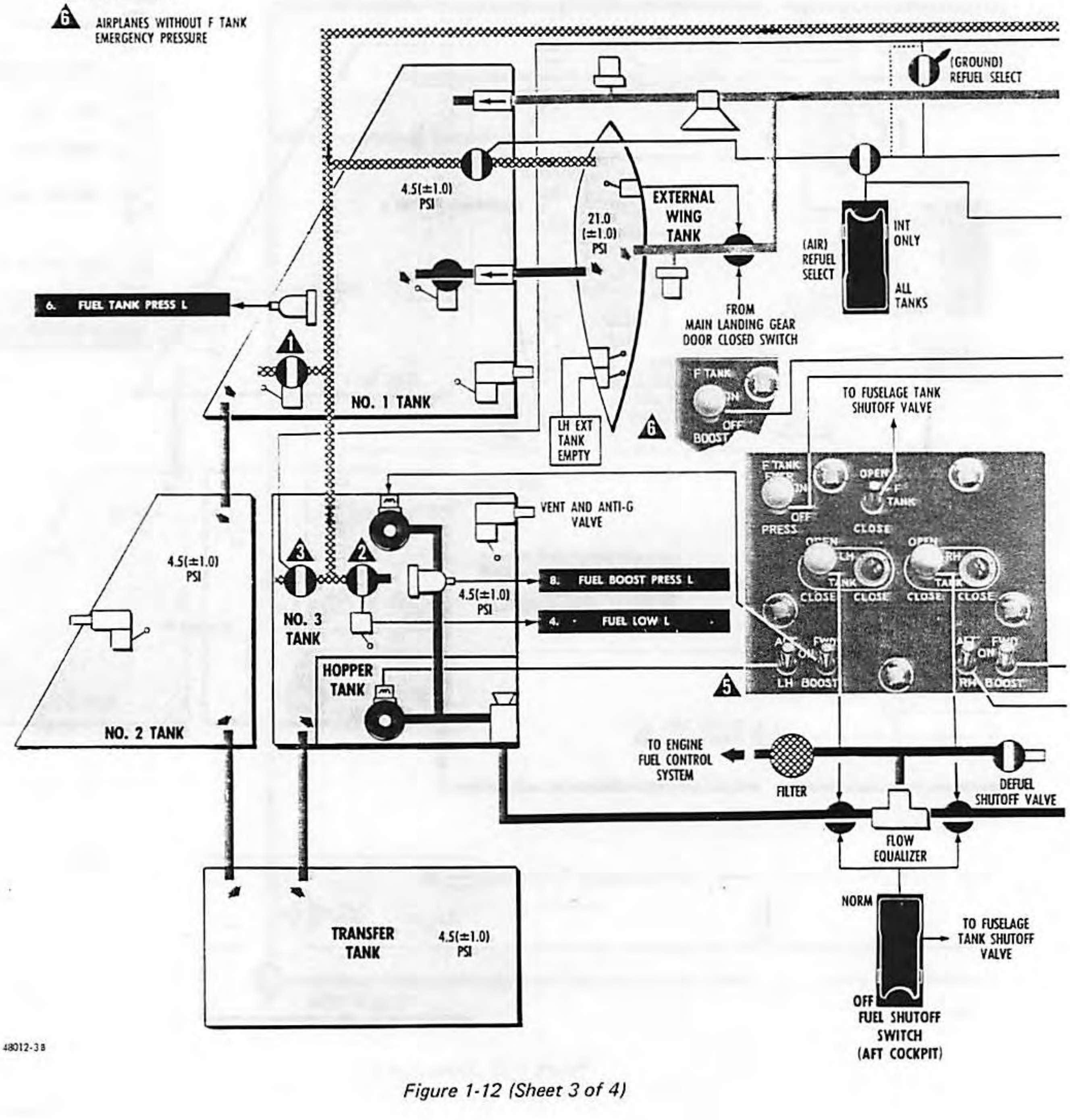


Figure 1-12 (Sheet 3 of 4)

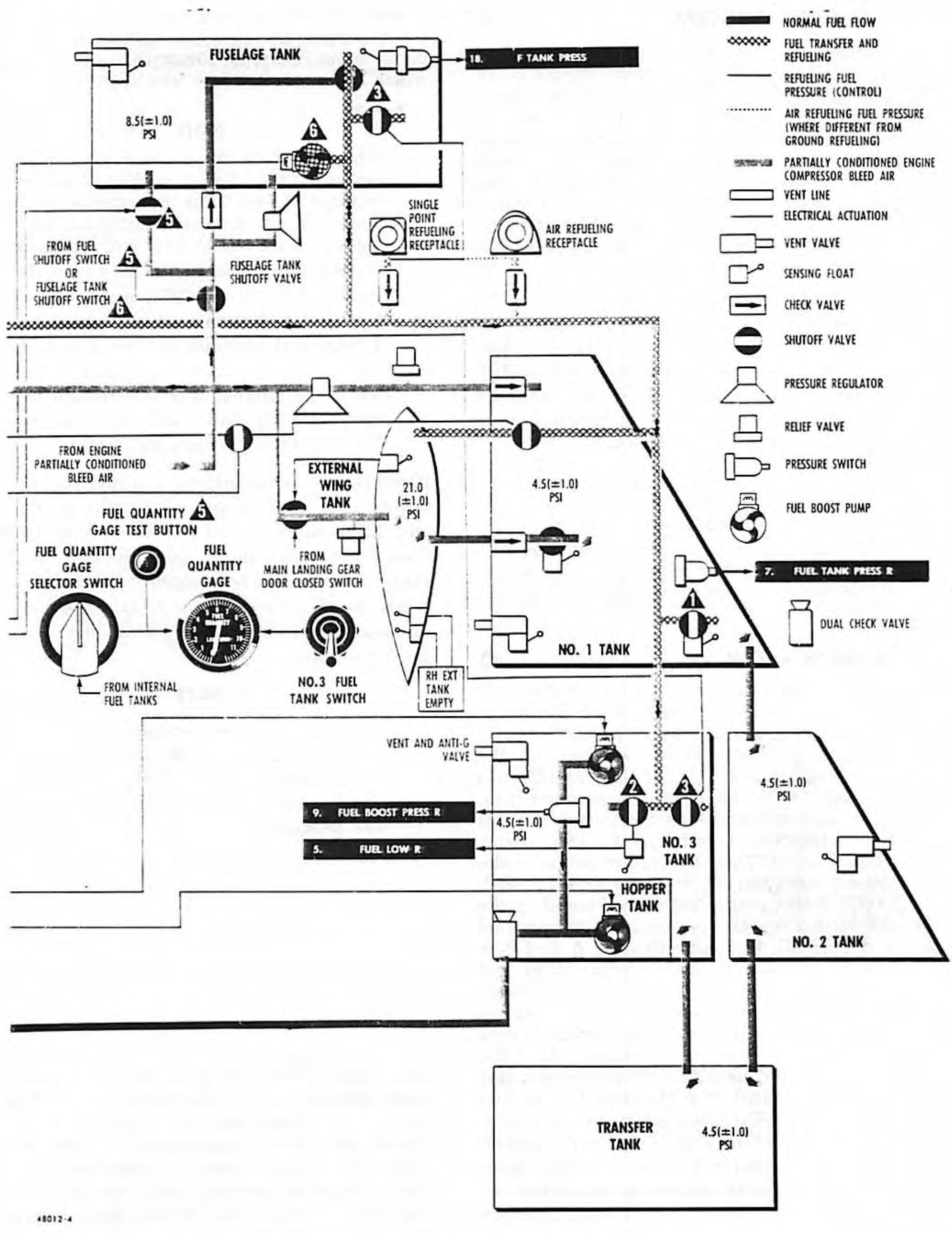


Figure 1-12 (Sheet 4 of 4)

FUEL SUPPLY SYSTEM

Fuel is supplied to the engine fuel control system by the airplane fuel supply system (figure 1-12) which consists of two wing tank systems (three main tanks and one T tank in each wing) and an F tank. The three main tanks, and the T tank in each wing, operate as an individual tank system and the fuel systems in each wing operate independently of each other. (The pilot cannot transfer fuel from one wing to the other.) The airplane may be refueled to the full internal fuel or the full internal and external fuel configuration. Provisions are included for two 360-gallon external wing tanks. All internal fuel tanks are the integral type composed of airplane structure and are not self-sealing. For fuel tank capacities, refer to the Fuel Quantity Data Table, figure 1-14. Fuel specifications are shown in figure 2-8. For additional information on this system, refer to T.O. 1F-106A-2-5.

NOTE

Fuel quantities quoted in this manual are exact. However, they may vary because of temperature, probe tolerances, and gage tolerances.

FUEL TRANSFER SYSTEM

The main fuel tanks in each wing are numbered in the order in which they are emptied. Tanks No. 1, 2, and 3 are located respectively in the forward, aft-outboard, and aft-inboard sections of the wings. A T tank is located aft of No. 2 and 3 tanks in each wing. Fuel is supplied under pressure from each No. 3 tank by the two electrically driven boost pumps through fuel shutoff valves to the engine fuel control system. These pumps are located in the forward-outboard and aft-inboard sections of each No. 3 tank. Boost pump inlets are located in the top and bottom of the fuel tanks to ensure fuel supply during all airplane attitudes. A fuel flow equalizer is provided to insure symmetrical fuel usage from the left and right-hand fuel systems. If the boost pumps are inoperative, the engine-driven fuel pumps will draw fuel through a dual check valve inlet located on the output side of the boost pumps. Fuel tank pressurization is provided to facilitate fuel transfer, to provide adequate fuel pressure to the engine fuel control system, and to prevent excessive high-altitude fuel vaporization. The fuel tank pressurization air is engine compressor bleed air which is pressure regulated and has passed through the air-conditioning primary heat exchanger. For tank pressurization and to provide fuel flow through each wing tank system, air pressure enters No. 1 tank in each wing fuel system. As fuel is drawn from No. 3 tank by the boost pumps, the pressure differential between tanks forces fuel from No. 1 tank into No. 2 tank, which in turn, forces fuel into No. 3 tank.

NOTE

When installed, the external wing tanks replenish the No. 1 tanks. However, transfer of fuel from the external wing tanks will not commence until the landing gear is up and locked. With the landing gear down, fuel will be used from the No. 1 tanks. After 227 pounds of fuel has been used from each No. 1 tank, the T tanks will replenish the No. 1 tanks at approximately engine consumption rate. When the landing gear is retracted the external wing tanks will replenish the No. 1 tanks to full capacity.

If refueled for a full internal and external fuel condition, after the external wing tanks are emptied and 227 pounds of fuel have been removed from each No. 1 tank, the T tanks replenish the No. 1 tanks at approximately engine consumption rate. When the T tanks are empty, approximately 390 pounds of fuel is transferred from the F tank to the No. 1 tanks.

NOTE

While each T tank normally replenishes its own wing No. 1 tank, it will replenish the opposite wing No. 1 tank through the transfer line should its own wing fail to feed properly.

The remainder of fuel in the No. 1 tanks is then used, followed by all of the fuel in the No. 2 tanks. Fuel is used from each No. 3 tank down to the 1200-pound level (approximately 3500 pounds total fuel) at which time the remainder of F tank fuel is transferred to the No. 3 tanks. When the fuel level in either No. 3 tank drops below the 570-pound level, the fuel scavenge system forces the fuel from the bottom of the F and T tanks and 110 pounds of fuel from the transfer line into the No. 3 tank. During a high-angle dive, the fuel line from tank No. 2 to tank No. 3 may become uncovered, disrupting normal fuel flow from No. 2 tank and permitting air to enter No. 3 tank as fuel is removed by the boost pumps. After return to level flight, the air in No. 3 tank is vented overboard by a float controlled vent and anti-g valve system, and normal fuel flow is resumed.

fuel control panels
(typical)

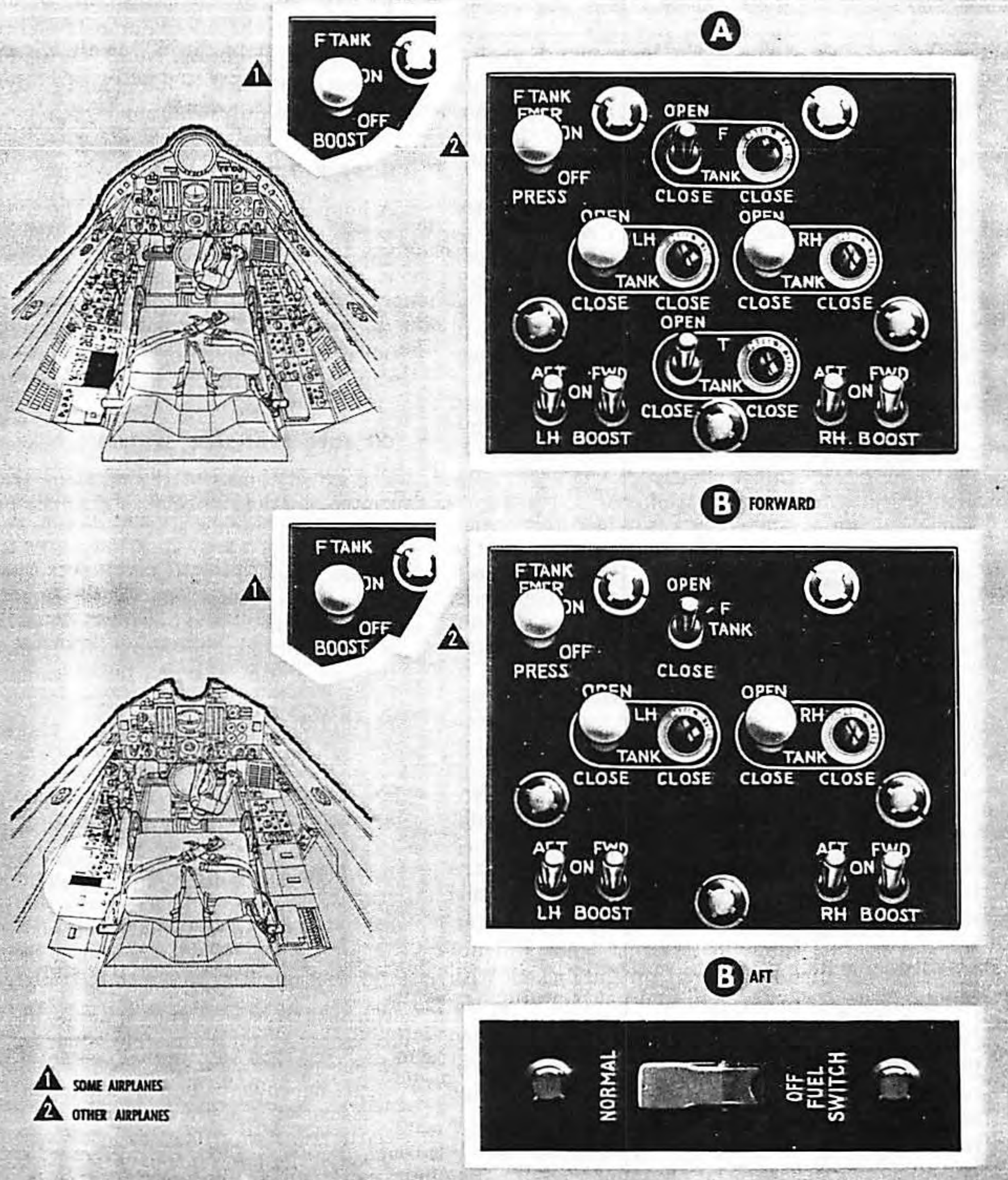


Figure 1-13

FUEL TRANSFER SYSTEM B

The main fuel tanks in each wing, tanks No. 1, 2, and 3, are located respectively in the forward, aft-outboard and aft-inboard sections of the wings. The T tanks, one in each wing, are located aft of tanks No. 2 and 3. Two electrically driven boost pumps supply fuel from each No. 3 tank to the engine fuel control system. These pumps are located in the forward-outboard and aft-inboard sections of each No. 3 fuel tank. Boost pump inlets are located in the top and bottom of the fuel tanks to ensure fuel supply during all airplane attitudes. A fuel flow equalizer is provided to ensure symmetrical fuel usage from the left and right fuel systems. Fuel shutoff valves located in the outlet of each No. 3 tank can be operated independently (from the forward seat only) to shut off one wing tank at a time, or simultaneously (from both seats) to shut off the entire fuel supply. If the boost pumps are inoperative, they will be bypassed and the engine-driven fuel pumps will draw fuel through a dual check valve inlet (one in each No. 3 tank) located downstream of the boost pumps. Fuel tank pressurization is provided to facilitate normal fuel flow through each wing tank system, to assure adequate fuel pressure to the engine fuel control, and to prevent excessive high-altitude fuel vaporization. Air pressure is supplied from engine compressor bleed air which is pressure regulated and has passed through the air-conditioning heat exchanger. Relief and check valves are provided to protect the fuel tanks against excessive pressures and to prevent vacuum or incorrect fuel or air flow within the system. Without external tanks installed and refueled to a full internal capacity, fuel flows from the F tank after 227 pounds of fuel have been used from each No. 1 tank. All F tank fuel is then transferred to No. 1 tanks at approximately engine consumption rate. Both right and left wing tanks are replenished independently. Fuel then flows from tank No. 1 to tank No. 2, to the T tank and into the No. 3 tank. System design ensures that No. 3 tanks are the last to empty. When installed, the external wing tanks replenish the No. 1 tanks. However, transfer of fuel from the external tanks will not begin until the landing gear is up and locked. With the landing gear down, fuel will be used from the No. 1 tanks. After 227 pounds of fuel have been used from each No. 1 tank, the F tank will replenish the No. 1 tanks at approximately engine consumption rate. When the landing gear is retracted, the external tanks replenish the No. 1 tanks to full capacity. When the fuel level of either No. 3 tank drops below 570 pounds the fuel quantity-low warning light illuminates and the scavenge system in the F tank is automatically actuated. Air pressure is directed to open the F tank pressure regulator and the residual fuel in the F tank and F tank line is forced into the No. 3 tanks. During a high-angle dive, the fuel line from the T tank to tank No. 3 may become uncovered, disrupting normal flow from the T tank. This permits air to enter the No. 3 tank as fuel is removed by the boost pumps. After return to level flight, the air in tank No. 3 is vented overboard by a float controlled vent and anti-g valve system, and normal fuel flow is resumed.

HOPPER TANK

A hopper tank is installed in the inboard aft corner of each No. 3 wing tank. Fuel flows into the hoppers through one-way flapper valves and traps fuel in the area around the aft bellmouth. The hopper will supply fuel for approximately 30 seconds in the event of complete boost pump failure when the airplane is nosed down, in extreme banks, and/or during sudden decelerations.

CG FUEL TRANSFER SYSTEM A

The effectiveness of the airplane is improved by controlling the airplane weight distribution in such a way that during specific speeds and at specific altitudes the cg is as far aft as permissible to reduce elevator trim drag. At lower speeds and altitudes, it is necessary to shift the cg forward to facilitate airplane stability. This cg control inflight is accomplished by automatic transfer of fuel between the F tank and the T tanks. This results in moving the average cg aft during the supersonic portion of the flight.

NOTE

- Fuel transfer aft to forward requires approximately 20 to 25 seconds.
- CG fuel transfer can be noted by placing the fuel quantity gage switch in the FWD position and monitoring the fuel quantity gage for an increase in fuel quantity during forward transfer, or a decrease in fuel quantity during aft transfer.

The fuel transfer signal is applied by the air data computer at approximately Mach 1.2. After application of the transfer signal, some delay may occur, dependent upon which tanks (F or T) are pressurized, and the available air pressure. Forward transfer will also occur when descending through 13,000 ($\pm$500) feet. Regardless of the flight condition, if the fuel in either No. 3 tank drops to 1200 pounds, fuel will be transferred from the F tank or the wing T tanks to maintain about 1200 pounds in the No. 3 tank until all transfer fuel is used. Engine compressor bleed air is utilized to provide the motive power for all normal fuel transfer operations. After completion of each fuel transfer operation, the empty tanks are automatically depressurized.

NOTE

- In the event of loss of engine compressor bleed air while fuel is aft in the T tanks, high-pressure air from the pneumatic system will automatically transfer the fuel forward to ensure airplane stability at subsonic speeds.
- In event of loss of ac essential power while fuel is aft in the T tanks, fuel will still automatically transfer forward at Mach 1.2 or when descending through 13,000 (±500) feet.
- In event of dc nonessential power failure, while fuel is aft in T tanks, fuel will immediately transfer forward.
- Refer to FLIGHT CHARACTERISTICS WITH AFT CG, Section VI, for additional information.

CG Control Switch Ⓐ

A guarded three-position cg control switch is provided to manually control the automatic fuel control transfer system. The cg control switch (23, figure 1-10) is located forward of the throttle quadrant on the left-hand subconsole. The switch is placarded "CG Control" or "CG Cont" with AUTO, FWD and TEST positions. When the switch is in AUTO position, fuel transfer aft and forward will automatically occur at a preset altitude and Mach as determined by the air data computer. When the switch is placed in FWD position, fuel transfer aft is prevented at all flight conditions or if the fuel is in the T tanks it will transfer forward to the F tank. When the switch is placed in the TEST position, fuel will be transferred aft and the cg transfer test failure light will illuminate on the fuel control panel if F tank pressure is low. During ground operation with the switch in the FWD or AUTO position, the cg transfer test failure light will illuminate if transfer tank pressure is low. Power is supplied by the dc nonessential bus.

CG Transfer Test Failure Light Ⓐ

A cg transfer test failure light placarded "Transfer Test Fail Lt," is located on the left console and indicates low pressurization in the transfer and fuselage tanks. With the engine running at a minimum of 75% rpm and with anti-icing equipment off, the light will illuminate when:

1. Transfer tank pressurization is low, with the cg control switch in either the FWD or AUTO positions, landing gear extended, the T tanks contain more than scavenge fuel, and T tank pressure is below 10 psi pressure.
2. The cg control switch is actuated to the TEST position and fuselage tank pressurization below 10 psi.

NOTE

- The light will illuminate momentarily after placing the switch to TEST until fuselage tank pressure builds up.
- At thrust settings below 75% rpm or with anti-icing equipment on, illumination of the light may be due to insufficient engine compressor bleed air.

EXTERNAL WING TANKS

Two jettisonable external wing tanks can be installed to augment the internal fuel supply. Installation provisions are made on the lower surface of each wing for the tanks which can be jettisoned by a ballistic charge. The 360-gallon tanks are mounted on ejection racks under each wing. The fuel in the external tanks is transferred into No. 1 tanks by engine bleed air pressure which is regulated at a higher pressure than that of the normal fuel system.

NOTE

Transfer of fuel from the external tanks to the No. 1 tanks will not commence until the landing gear is up and locked.

As the fuel level in No. 1 tanks lowers, the high-level floats within the No. 1 tanks open the external tanks shutoff valves and allow air pressure to force fuel from the external tanks into the No. 1 tanks. A combination valve dumps external pressure when the fuel is exhausted and also provides pressure relief and dumping for the external tanks during climbs and descents. Since external fuel is not indicated on the fuel quantity gage, fuel transfer from the external tanks can be noted by the fact that fuel quantity indication will not decrease until the external tanks have emptied.

NOTE

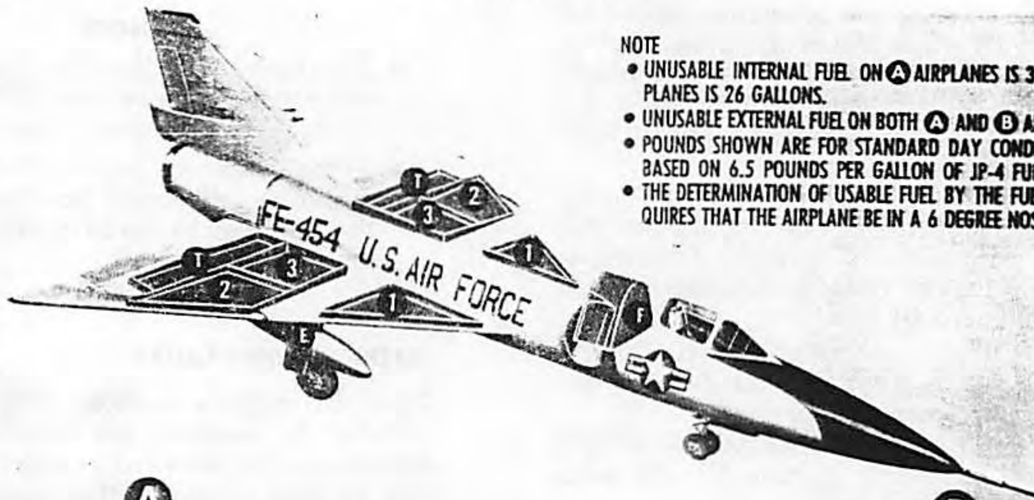
Loss of dc nonessential power to the external wing tank fuel shutoff valve will cause fuel transfer to cease.

The external tanks are jettisoned by depressing the wing tank release button. The ejection racks are fixed and do not jettison when the external tanks are ballistically jettisoned. Refer to Section V for operating limitations with external tanks installed.

fuel quantity data table

DATE: 21 FEBRUARY 1967
DATA BASIS: ACTUAL

U.S. GALLONS AND POUNDS



NOTE

- UNUSABLE INTERNAL FUEL ON Ⓐ AIRPLANES IS 37 GALLONS ON Ⓑ AIRPLANES IS 26 GALLONS.
- UNUSABLE EXTERNAL FUEL ON BOTH Ⓐ AND Ⓑ AIRPLANES IS 4 GALLONS.
- POUNDS SHOWN ARE FOR STANDARD DAY CONDITIONS ONLY AND ARE BASED ON 6.5 POUNDS PER GALLON OF JP-4 FUEL.
- THE DETERMINATION OF USABLE FUEL BY THE FUEL QUANTITY GAGE REQUIRES THAT THE AIRPLANE BE IN A 6 DEGREE NOSE-HIGH ATTITUDE.

A: USABLE FUEL			B: USABLE FUEL	
POUNDS	GALLONS		POUNDS	GALLONS
1944	299	NO. 1 WING TANKS	1944	299
2021	311	NO. 2 WING TANKS	2067	318
2756	424	NO. 3 WING TANKS	2782	428
1365	210	TRANSFER TANKS	1443	222
1560	240	FUSELAGE TANK	1144	176
195	30	LINES	45	7
4654	716	EXTERNAL TANKS	4654	716
9841	1514	TOTAL USABLE FUEL FULL INTERNAL FUEL CONFIGURATION	9425	1450
14,495	2230	TOTAL USABLE FUEL WITH EXTERNAL TANKS	14,079	2166

48014-1

Figure 1-14

FUEL QUANTITY CHECKS

GROUND CONDITION

NOTE
THE FUEL QUANTITY GAGE READINGS IN THIS TABLE ARE BASED ON THE AIRPLANE BEING IN NORMAL GROUND POSITION.

A POUNDS		B POUNDS
9650 ±300		9380 ±300
4040 ±190		4120 ±190
1560 ±140		1160 ±135
1100 TO 1500		1100 TO 1500

48014-2

External Wing Tanks Release Button

The external tanks release button (figure 1-21) is provided to jettison the external tanks. The button is located on the landing gear control panel and is placarded "Wing Tank Release." When the button is depressed, electrically fired ballistic charges unlock and separate the tanks from the wings. The jettison circuit receives power from the 28-volt dc essential bus.

CAUTION

If the tanks are jettisoned while the landing gear is extended, the landing gear fairing doors will be damaged as the external tanks are released.

External Tank Ground Safety Pin

A ground safety pin may be inserted into either side of the tank ejection racks. The safety pin is a ball-locking type pin. It safeties the ballistic jettison system and provides a mechanical lock to prevent inadvertent separation of the tank from the rack. The pin is equipped with a remove-before-flight streamer.

SINGLE-POINT REFUELING A

The internal and external fuel tanks are serviced through a single-point refueling adapter located in the lower right engine inlet duct fairing. The tanks are refueled in a reverse manner to normal tank transfer sequence, and selective refueling of internal tanks is not possible. If the refuel selector valve is in the OPEN position all internal and external tanks will be refueled; however, if the refuel selector valve is in the CLOSED position internal tanks only will be refueled. During refueling operations fuel is routed to the F tank, T tanks and to each No. 3 tank through a pressure regulator check valve and refueling shutoff valves located in each tank. When each No. 3 tank fills, fuel then passes into and fills each No. 2 tank and then into the No. 1 tanks. As each No. 1 tank fills, a high-level float valve will automatically close the respective refueling shutoff valve by hydraulic action and stop fuel flow input. When the external tanks are full, float valves close the external tank refueling shutoff valves. The external tanks may also be refueled through a filler cap on each tank. External electrical power is not required during refueling operations.

SINGLE-POINT REFUELING ⓑ

The internal and external fuel tanks are serviced through a single-point refueling adapter located below the lower right engine inlet duct fairing. Selective refueling is possible only to the extent that the external tanks can be either refueled or not refueled. If the refuel selector valve, located in the right main wheel well, is in the CLOSED position, the external tanks will not be refueled and the internal tanks will fill in a reverse manner to normal tank transfer sequence. During refueling, fuel enters each No. 3 tank through a refueling shutoff valve. When the No. 3 tanks are filled, fuel is routed through transfer tubes to fill, in order, T tanks, No. 2 tanks, and No. 1 tanks and F tank. When the tanks are full, the refueling shutoff valve closes and fuel flow into the wing tanks is stopped. When the refuel selector valve is in the OPEN position, the wing tanks are filled in the same sequence (No. 3, T, No. 2 and No. 1) and the external tanks are refueled simultaneously with the wing tanks. When the external tanks are full, float valves close the external tank refueling shutoff valves. The external tanks may also be refueled through a filler cap on each tank. External electrical power is not required during refueling operations.

NOTE

The refueling system permits use of any refueling equipment capable of pumping 200 to 600 gpm of fuel at 30 to 60 psi.

Refuel Selector Valve

A manually controlled refuel selector valve is located in the right main wheel well and has two marked positions, OPEN and CLOSED. When the handle is in the OPEN position the internal and external tanks are refueled during the refueling process. However, when the handle is in the CLOSED position, the external tanks cannot be refueled.

FUEL SHUTOFF VALVES

Two two-position, electric motor-driven, sliding-gate valves are used to shut off the fuel supply to the engine from the fuel tanks. The valves, one in each wing tank system outlet fuel line, can be controlled individually or simultaneously by the fuel shutoff switches.

NOTE

Normally the valves require one second or less to rotate to the fully closed position. However, if the master electrical power switch is turned OFF during valve rotation, the sliding gate fuel shutoff valves will remain partially open.

The boost pumps will continue operating when the fuel shutoff valve on the appropriate side is closed. On all ⓑ airplanes the fuel shutoff valves are closed simultaneously by actuation of the aft cockpit fuel shutoff switch. The fuel shutoff valves are normally open during flight and are powered by the dc essential bus.

FUEL FLOW EQUALIZER

A fuel flow equalizer is used to regulate symmetrical fuel usage from each wing tank system. The flow equalizer is located in the fuel line between No. 3 tanks and the engine fuel control unit. A bypass condition is automatically established to ensure fuel supply in the event of boost pump failure (within one or both wings) or malfunction of the flow equalizer. Power is received from the dc essential bus.

FUEL SHUTOFF SWITCHES

WARNING

If a fuel shutoff switch is inadvertently moved to the CLOSED position by the pilot or is found in the CLOSED position during the Preflight Interior Inspection, have maintenance personnel verify that the fuel shutoff valve is fully OPENED prior to flight.

NOTE

- The fuel shutoff switches should not be placed OFF until the engine ceases windmilling. This will prevent cavitation and possible failure of the engine-driven fuel pump and subsequent starvation at the engine fuel control.
- The fuel shutoff valves are normally open throughout a flight. When a malfunction occurs where the thrust setting cannot be changed (such as throttle hang-up), the only controlled fuel shutoff is with the fuel shutoff switches. Ground tests have demonstrated the following lag times before thrust loss after closing the fuel shutoff valves:

Thrust Setting	Time from Fuel Shutoff to Thrust Loss
Full Military	4 Seconds
90% rpm	6 Seconds
80% rpm	10 Seconds

On Ⓐ airplanes, four two-position switches (figure 1-13), located on the fuel control panel, control the fuel and air shutoff valves. The two switches placarded "LH Tanks" and "RH Tanks" control the respective fuel shutoff valves; and the two switches placarded "F Tank" and "T Tank" control the respective air shutoff valves. Each switch has an OPEN and CLOSE position. When either the F tank or T tanks valves are closed, the respective tank vents to the atmosphere and fuel flow from that tank will cease. When any shutoff valve is in other than the fully open position, warning lights on both the fuel control panel and the master warning light panel will illuminate. The switch receives power from the dc essential bus.

On Ⓑ airplanes, three two-position shutoff switches are located on the forward fuel control panel and a two-position fuel shutoff switch is located on the aft fuel control panel. Two of the forward switches are placarded "LH Tanks" and "RH Tanks" and have OPEN and CLOSE positions. The two switches control the respective fuel shutoff valves. The third forward switch is placarded "F Tank" and has OPEN and CLOSE positions. It controls the F tank air shutoff valve which controls fuel flow from the F tank.

NOTE

On airplanes with the F tank emergency boost pump switch, the F tank shutoff switch should be in the CLOSED position during operation of the F tank emergency boost pump.

The aft switch has NORMAL and OFF positions. With the aft switch in the NORMAL position, the fuel shutoff valves are controlled by the forward fuel shutoff switches. When the aft fuel shutoff switch is in the OFF position, all fuel shutoff valves are closed regardless of the position of the forward fuel shutoff switches. Power is supplied from the dc essential bus.

BOOST PUMP SWITCHES

Four two-position fuel boost pump switches (figure 1-13) are located on the fuel control panel to provide individual control of the fuel boost pumps. The switches are placarded "LH Fwd Booster," "LH Aft Booster," "RH Fwd Booster," and "RH Aft Booster" with ON and OFF positions which control the boost pumps accordingly. The boost pumps will remain in operation even though the fuel shutoff valves are closed.

WARNING

Before either right or left fuel shutoff switch is placed in the CLOSE position, thereby closing a fuel shutoff valve in either tank, the respective fuel boost pump switches must be placed in the OFF position. If the boost pumps are not turned OFF, failure of the fuel equalizer absolute pressure switch may result in engine fuel starvation.

The boost pumps are powered from the ac nonessential bus or the ATG bus and the control circuit receives power from the dc essential bus.

NOTE

When all boost pumps are turned OFF, a normal increase or decrease of as much as two percent engine rpm may occur.

TRANSFER TANK SHUTOFF SWITCH Ⓐ

On Ⓐ airplanes the T tank shutoff switch (figure 1-13), placarded "T Tank," is located on the fuel control panel. The switch has OPEN and CLOSE positions and controls the aft motor-operated air shutoff valve. Selecting the CLOSE position shuts off air to the low-level float valves to allow for depressurization of the T tanks in the event of a fire, during an emergency landing, or in the event of a malfunction in the T tank pressurization system, to allow the F tank to be pressurized for fuel transfer. The switch receives power from the 28-volt dc essential bus.

FUSELAGE TANK EMERGENCY PRESSURE SWITCH (SOME AIRPLANES) Ⓐ

On airplanes not equipped with an F tank emergency boost pump switch, an F tank emergency pressure switch is provided on the fuel control panel (figure 1-13). The switch is placarded "F" Tank Emer Press" and has EMER ON and OFF positions. In the EMER ON position, two solenoid operated air shutoff valves are opened, permitting air to flow directly into the F tank. Two pressure relief valves are installed to prevent F-tank overpressurization. After initial transfer of 390 pounds of fuel from the F tank to the No. 1 tank, the remaining fuel in the F tank will not transfer until the fuel in each No. 3 tank decreases to 1200 pounds. Then the fuel remaining in the F tank is forced through a transfer line into the No. 3 tanks (L and R). After all fuel has been transferred, the switch should be returned to the OFF position to prevent continued loss of air through the pressure relief valve. The switch receives power from the dc essential bus.

FUSELAGE TANK EMERGENCY PRESSURE SWITCH (SOME AIRPLANES) Ⓑ

On airplanes not equipped with an F tank emergency boost pump switch, an F tank emergency pressure switch is provided on the fuel control panel (figure 1-13). The switch is placarded "F Tank Emer Press" and has EMER ON and OFF positions. If tank pressure falls below six psi, the F tank low pressure warning light will illuminate. Placing the switch in EMER ON position will open a solenoid valve and permit air to bypass the F tank pressure regulator. The bypassed air will pressurize the F tank and extinguish the F tank pressure-low warning light. Two pressure relief valves are installed to prevent F-tank overpressurization. When the F tank is empty the F tank emergency pressure switch should be placed in the NORMAL (OFF) position. The F tank emergency pressure switch receives power from the dc essential bus.

FUSELAGE TANK EMERGENCY BOOST PUMP SWITCH (OTHER AIRPLANES)

Ⓐ* and Ⓑ** airplanes not equipped with an F tank emergency pressure switch have an F tank emergency boost pump switch located on the fuel control panel (figure 1-13). The switch is placarded "F Tank Boost" and has ON and OFF positions. With the switch in the ON position (and the F tank shutoff switch in the closed position) a boost pump in the F tank will supply fuel to the No. 3 tanks in the event F tank pressure fails. After all fuel has been transferred, the switch should be returned to the OFF position. The switch receives power from the dc nonessential bus.

Fuselage Tank Pressure-Low Warning Light Ⓑ

On Ⓑ airplanes the F tank pressure-low warning light (18, figure 1-30), is located on both the forward and aft warning light panels. When pressure in the F tank drops below 6.0 (±0.5) psi, the lights illuminate and display "F TANK PRESS," warning the forward pilot to place the F tank pressurization switch in the EMER position. After the F tank has been repressurized to more than 7.0 (±0.5) psi, the lights will extinguish. The lights automatically extinguish when all fuel is used from the F tank. The warning lights receive power from the dc essential bus.

FUEL QUANTITY GAGE SELECTOR SWITCH

On Ⓐ airplanes, the four-position fuel quantity gage selector switch (28, figure 1-8, and 22, figure 1-9), is located on the instrument panel. On Ⓑ airplanes the switch is located on the instrument panel of the forward cockpit only. The switch permits the reading of total or individual fuel tank quantities on the fuel quantity gage. If LH is selected, the fuel quantity in the left wing (No. 1, No. 2, No. 3, and T tank) will be indicated on the fuel quantity gage. If RH is selected, the fuel quantity in the right wing system will be indicated. If FWD (or F on some Ⓑ airplanes) is selected, the fuel quantity in the F tank will be indicated; and if TOT is selected, the gage will indicate total internal fuel. However, since the individual fuel quantity indicating systems are calibrated separately from the total indicating system, and as a result of the inherent calibration tolerances in each system, an addition of the individual (LH, RH, and FWD) indicated quantities may or may not equal the total (TOT) indicated quantity. Therefore, total usable fuel aboard should be obtained in the TOT position. Power is supplied from the ac essential bus.

*Ⓐ *54-453, -454, 56-456 thru 57-245, 57-2465, 59-087 & on.*

**Ⓑ *57-2508 thru 57-2515, 57-2523, 59-160 & on.*

NUMBER 3 FUEL TANK SWITCH

On some airplanes a No. 3 fuel tank switch (5, figure 1-11) is located above the right console. On Ⓑ airplanes the switch is located in the forward cockpit only. The switch is placarded "Fuel Qty" and has spring-loaded RH #3 TANK and LH #3 TANK positions. With the fuel quantity gage selector switch in LH, RH, or TOT positions, the No. 3 fuel tank switch permits reading the No. 3 fuel tank quantities on the fuel quantity gage.

NOTE

- Do not attempt to obtain No. 3 fuel tank readings with the fuel quantity gage selector switch in the FWD position, as an erroneous indication will result.
- No. 3 fuel tank readings can be as much as 210 pounds low and still be within gage tolerances.

The switch receives power from the ac essential bus.

FUEL QUANTITY GAGE AND GAGE TEST BUTTON

The fuel quantity gage (27, figure 1-8, and 29, figure 1-9), located on the instrument panel, indicates internal fuel quantity. The fuel quantity indicating system is a capacitance-type and is capable of indicating total of fuel quantity in left and right sides or fuel quantity in the F tank or total internal fuel. The gage indicates the quantity of fuel in pounds and the system compensates for changes in fuel density.

NOTE

The fuel quantity gage readings may fluctuate momentarily when switches which utilize ac power are activated. The readings are also affected by airplane attitude and state of acceleration or deceleration; therefore, the individual system readings are more valuable as indications of proper fuel feeding than as accurate quantity readings.

On airplanes having a conventional instrument display, gage operation can be checked by a test button (26, figure 1-8) placarded "Fuel Qty Ind Test" on the instrument panel. When the test button is held depressed, the gage pointer should move toward zero, and when the button is released, the pointer should return to its original position. Failure of the pointer to move indicates a faulty system. The indicating system receives power from the ac essential bus.

FUEL QUANTITY-LOW WARNING LIGHTS

Two fuel quantity-low warning lights (4 and 5, figure 1-30), located on the warning light panel, illuminate and display "FUEL LOW—L" and "FUEL LOW—R" when the usable quantity of fuel in each No. 3 tank reaches approximately 570 pounds. These lights are a more accurate indication of low fuel level than the LH or RH or No. 3 tank fuel quantity readings. Fuel quantity-low warning light operation is predicated on a 6° nose high flight attitude. On the ground, the lights may illuminate with more than 570 pounds of fuel in the No. 3 tank. The fuel quantity-low warning circuits receive power from the dc essential bus.

NOTE

The fuel quantity-low warning lights will illuminate during maneuvers of less than 1 g.

EXTERNAL TANK EMPTY LIGHTS

Two external tank empty lights (8, figure 1-8, and 8, figure 1-9) are installed on the instrument panel. The lights are in the Ⓑ forward cockpit only. The lights illuminate amber, one displaying "LH EXT TANK EMPTY" and the other displaying "RH EXT TANK EMPTY." The lights illuminate when the external tanks are empty as indicated by the low level float switches. The lights remain on until the high-level float switches indicate that the tanks are full. The tanks aboard switch prevents the lights from illuminating when the tanks are off the airplane. The lights receive power from the dc essential bus.

NOTE

Momentary negative G or uncoordinated maneuvers during flight may cause the lights to illuminate before the tanks are empty; however, the tanks will continue to transfer fuel.

FUEL SHUTOFF VALVE WARNING LIGHTS

On Ⓐ airplanes, four shutoff valve warning lights (figure 1-13), located on the fuel control panel, illuminate when their respective shutoff valves are in any position other than fully open. On Ⓑ airplanes, two fuel shutoff valve warning lights (figure 1-13), located on the forward fuel control panel, illuminate when their respective fuel shutoff valves are in any position other than fully open. When any one of the fuel shutoff valves is in any position other than fully open, the "FUEL VALVE CLOSED" warning light (20, figure 1-30), located on the master warning light panel, will also be illuminated, and will remain on until all fuel shutoff valves are in the open position. Power is supplied from the dc essential bus.

NOTE

On Ⓑ airplanes the "FUEL VALVE CLOSED" warning light will not illuminate when the F tank shutoff switch is in the CLOSE position.

FUEL BOOST PRESSURE-LOW WARNING LIGHTS

Two fuel boost pressure-low warning lights (8 and 9, figure 1-30), located on the warning light panel, illuminate and display "FUEL BOOST PRESS —L" or "FUEL BOOST PRESS—R" if the left or right tank outlet pressure drops below 10.5 psi. The appropriate light will remain illuminated until the boost pump pressure exceeds 12 psi. The fuel boost pressure-low warning circuits receive power from the dc essential bus.

FUEL TANK PRESSURE-LOW WARNING LIGHTS

Two fuel tank pressure-low warning lights (6 and 7, figure 1-30), located on the warning light panel, illuminate and display "FUEL TANK PRESS—L" or "FUEL TANK PRESS—R" if the left or right No. 1 tank pressurization falls below 1.0 psi. The appropriate light will remain illuminated until tank pressurization exceeds about 1.5 psi. The fuel tank pressure-low warning circuits receive power from the dc essential bus.

ELECTRICAL POWER SUPPLY SYSTEM

Two generators, a dc and an ac, supply electrical power to airplane systems (figure 1-15). Two other generators supply power to the MA-1 system.

NOTE

- This manual covers two aircraft and weapon control systems, the MA-1 incorporated in the F-106A airplane and the AN/ASQ-25 incorporated in the F-106B airplane. The text distinguishes between the two configurations only where necessary and will refer to all switches, controls, etc., in both the F-106A and F-106B aircraft weapon and control systems as the "MA-1."
- Refer to MA-1 ELECTRICAL POWER SUPPLY SYSTEM, Section IV, and to the Confidential Supplement, T.O. 1F-106A-1A, for additional information.

electrical power

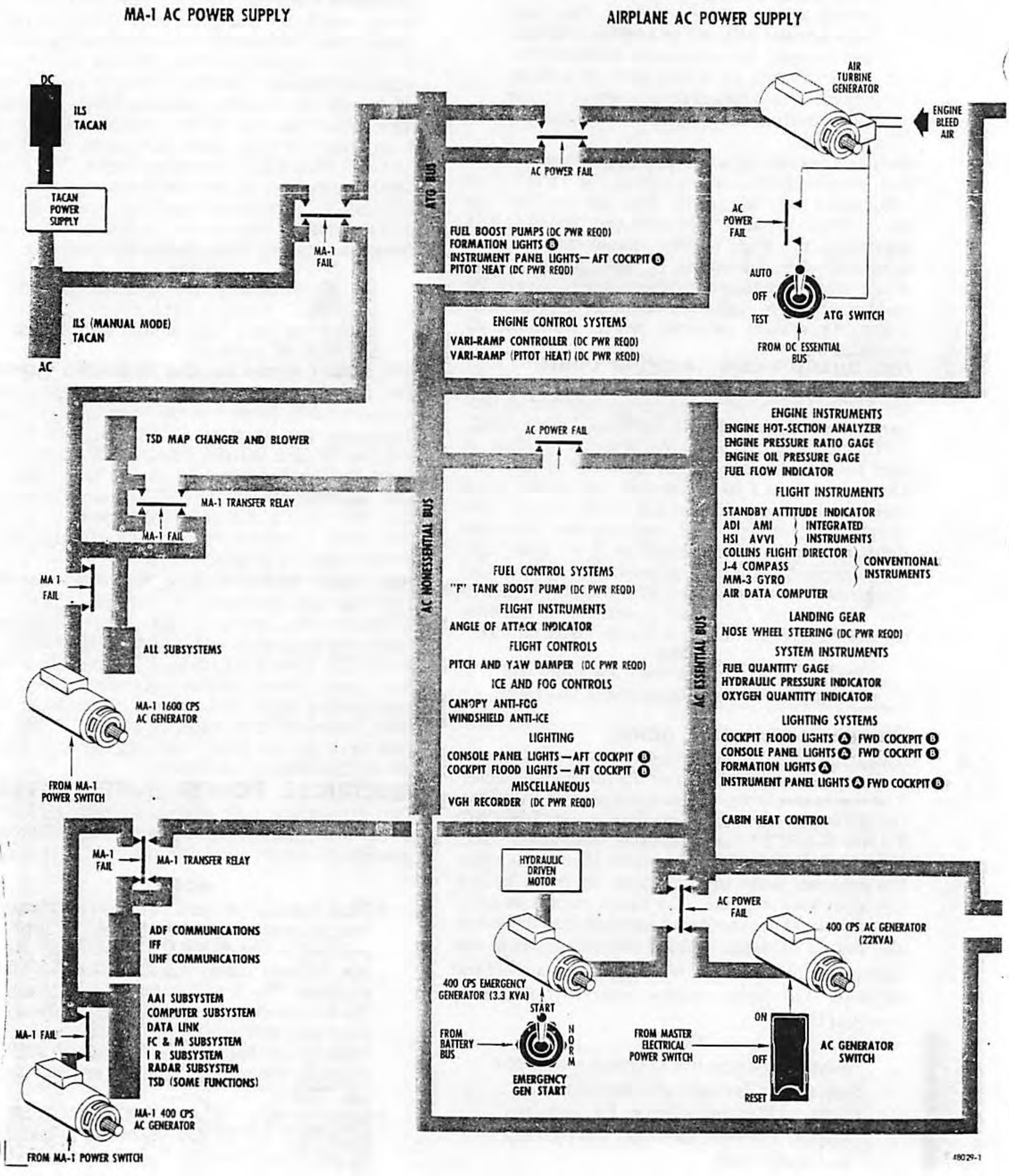
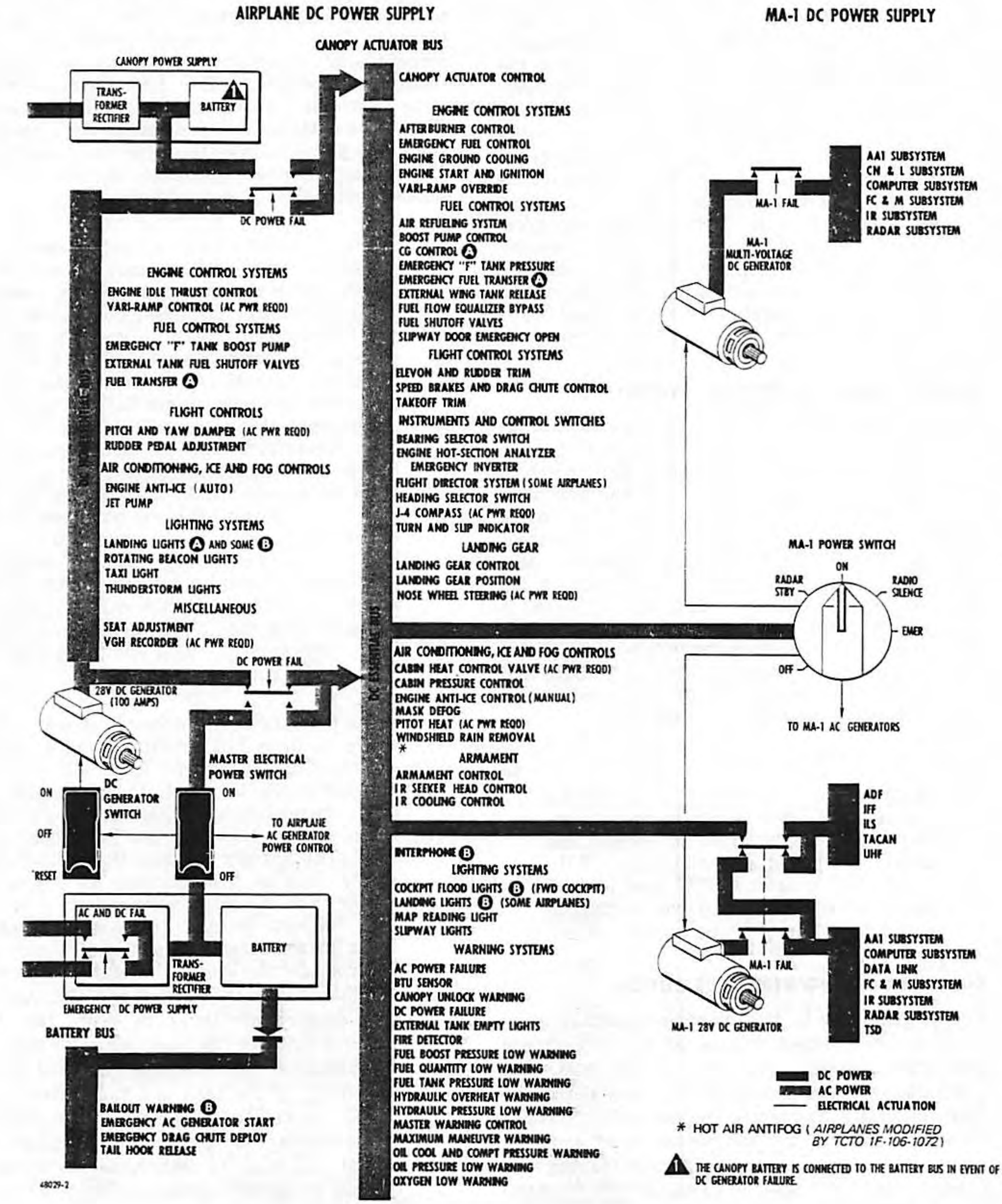


Figure 1-15

supply system



The four generators are driven at constant speed by an engine-driven, constant-speed drive. An emergency dc power package supplies power if the dc generator fails. A hydraulically powered emergency ac generator and an air turbine motor-driven ac generator (ATG), supply power if the ac generator fails. Emergency systems energize automatically. External power, ac, dc and MA-1, is connected through a single receptacle on the left side of the airplane. The master electrical power switch controls all airplane generators including the MA-1 generators, but it does not control external power. External power takes precedence over airplane power. While external power is connected, airplane generators cannot be energized. Fuses, installed in panels, protect electrical circuits in the airplane. For additional information on this system, refer to T.O. 1F-106A-2-10.

MASTER ELECTRICAL POWER SWITCH

A guarded master electrical power switch (20, figure 1-10) is located on the left console and controls electrical power to airplane buses. The switch is placarded "Master Elec Pwr" and has two positions, OFF and ON. Actuating the switch to the ON position connects the battery to the dc essential bus. With the switch in the ON position, generators can be energized. With the switch in the OFF position, all generator and battery power is disconnected from airplane buses; however, the canopy still operates from its independent power package, and in emergencies the drag chute and tailhook can be deployed. On Ⓑ airplanes, bailout warning light power will be available.

NOTE Ⓑ

> On Ⓑ airplanes the switches in both the forward and aft cockpits must be actuated to the ON position to connect the battery to the dc essential buses. With either switch in the OFF position, all generator and battery power is disconnected from the airplane buses.

DC ELECTRICAL POWER DISTRIBUTION

The dc generator is driven by the constant-speed drive system. Output voltage of the 100-ampere generator is closely regulated to 28 volts. The generator supplies power to two bus networks, essential and nonessential (figure 1-15). Systems necessary for flight draw power from essential buses. Red placards on fuse panels identify the fuses of systems connected to the 28-volt essential bus. Systems not necessary for flight draw power from nonessential buses. Green placards identify the fuses of systems connected to 28-volt nonessential buses.

Emergency DC Power Package

The emergency dc power package supplies power to essential buses only; that is, to fuses with red placards. The power package consists of a 15-ampere-hour silver-zinc storage battery and a transformer-rectifier. The transformer-rectifier steps down and rectifies 115-volt ac power to charge the battery, and also assists the battery in supplying power to essential buses in emergencies. Emergency dc power package operation is described under the following conditions:

a. During normal operation with primary dc generator supplying dc buses, no output is derived from the emergency power package. With the master electrical power switch in the ON position (both cockpits on Ⓑ airplanes) the battery relay connects the battery to the dc essential bus. Energizing the dc nonessential bus energizes the dc relay which disconnects the battery from the essential bus. Energizing the ac nonessential bus causes the T-R to draw three-phase ac power from the nonessential bus to charge the battery. These power leads are not fused. When dc power energizes the nonessential bus, the bus relay in the canopy package energizes and connects nonessential bus power to the canopy actuator bus; thus canopy power is supplied from the dc nonessential bus. When the ac nonessential bus is energized, the T-R charges the canopy battery.

b. If the primary dc generator fails, emergency power package T-R output current increases and the T-R and battery, in parallel, assume the load on the dc essential bus. Initially, the fully charged battery assumes about two-thirds of the load and the T-R about one-third. The battery assumes the greater part of the load at first because its voltage is higher than the fixed voltage output of the T-R. Battery voltage drops so that after about 15 minutes of operation it equals the T-R voltage; the load is therefore distributed equally between the battery and T-R. As battery voltage continues to drop, the T-R assumes more of the load. After 30 minutes of operation, the T-R has assumed about two-thirds of the load and the battery one-third, a reversal of the original ratio. Failure of the primary dc generator also causes the canopy bus relay to deenergize. As a result, the canopy battery alone supplies power during canopy operation.

c. If the primary ac generator fails, the ac emergency control relay energizes and starts the emergency generator. When the emergency generator reaches operating speed, the frequency-sensitive ac emergency power disconnect relay energizes and connects the emergency generator to the ac essential bus. The emergency generator does not energize the ac nonessential bus. T-R input remains connected to the ac nonessential bus; therefore, the battery and the canopy battery cease to be charged.

d. If the primary ac generator fails, the ac emergency generator powers ac essential buses. If in addition, the primary dc generator fails, the emergency ac generator will supply the T-R output current and the T-R and battery, in parallel, power the dc essential bus regardless of airspeed. In either configuration, the ac emergency generator continues to power ac essential buses.

e. If the airplane engine fails in flight and the engine windmills, sufficient hydraulic power is available to drive the ac emergency generator at rated speed until airspeed falls below 280 knots. Below this airspeed, hydraulic power decreases as airspeed decreases. The emergency generator will continue to power the T-R unit as long as sufficient hydraulic power is available. The frequency-sensitive relay does not disconnect the emergency generator from the ac essential buses even though frequency and output voltage of the generator fall far below normal operating range. If the airplane engine freezes in flight, the emergency generator does not operate and the battery alone supplies dc power.

Battery

The battery is a part of the emergency dc power package. It consists of an 18-cell silver-zinc unit which is sealed in a fiberglas case. A fully charged battery measures 33.5 volts and has a life expectancy of approximately 10 charge-discharge cycles or nine months wet life, whichever occurs first. The limited number of charge-discharge cycles of the silver-zinc battery requires that it be used only for its intended purpose of emergency operation. The battery must be kept fully charged and available for emergencies. The electrolyte in the battery is a 44% solution of potassium hydroxide, a strong alkaline and caustic liquid that is more destructive to metal, wood, and flesh than the sulphuric acid used in lead-acid storage batteries. The battery performs best at temperatures between 32°F and 140°F and its capacity decreases as the temperature approaches 32°F. In the event of battery failure, the canopy battery will automatically provide power to the battery bus. If normal ac and dc power failure occurs while flying with a dead or disconnected airplane battery, the emergency ac generator and the ATG will not energize automatically. To energize the ac generator, position the emergency ac generator switch to START for approximately 10 seconds. When the generator comes on the line, the ATG will start automatically.

Canopy Battery

The canopy battery is similar to the battery described above except it is a 17-cell battery and supplies power for the canopy only when the dc generator is off. The canopy battery automatically provides power to the battery bus in the event of airplane battery failure. This provides power for tail hook release, emergency drag chute deploy, emergency ac generator start, and bailout warning Ⓑ. The only means available to check the canopy battery is to operate the canopy with all external and battery power off.

DC Generator Switch

On Ⓐ airplanes a guarded dc generator switch (2, figure 1-11) is located on the right console. On Ⓑ airplanes the switch is located on the right console in the forward cockpit only. The switch controls the dc generator if the master power switch is ON. The switch is placarded "DC Gen Cont" and has three positions, OFF, ON, and RESET. The RESET position is inoperative. Actuating the switch from OFF to ON position with the master power switch ON energizes the dc generator. The generator then powers dc essential and nonessential buses. When dc essential buses are energized by the generators the battery is disconnected. The generator is reset by actuating the generator switch to OFF, then ON.

DC Power Failure Warning Light

The dc power failure warning light (2, figure 1-30) located on the warning light panel, illuminates and displays "DC POWER FAIL" if the dc generator fails. The dc power failure warning circuit receives power from the dc essential bus.

AC ELECTRICAL POWER DISTRIBUTION

The ac generator is driven by the constant-speed drive system. Output voltage of the 22-KVA three-phase generator is closely regulated to 115 volts on each phase. The generator supplies power to two bus networks, essential and nonessential (figure 1-15). Systems necessary for flight draw power from essential buses. Blue placards on fuse panels identify the fuses of systems connected to 115-volt ac essential buses. An instrument transformer steps down 115-volt ac to 26 volts ac to operate three engine instruments. The transformer is connected to the ac essential bus. Systems not necessary for flight draw power from nonessential buses. Yellow placards identify the fuses of systems connected to 115-volt ac nonessential buses. The ac emergency generator supplies power to essential buses only; that is, to fuses with blue placards. A pair of hydraulic motors that operate from the secondary hydraulic system drive the emergency generator. If the ac generator fails, the 3.3 KVA emergency generator starts automatically, and when it reaches operating speed, a frequency sensitive relay connects the emergency generator to ac essential buses. An air turbine motor (ATG) generator is installed to provide ac power (in the event of ac generator failure) to the fuel boost pumps, AMI, AVVI, and air data computer (integrated instrument airplanes). If ac and MA-1 generators fail, the ATG will provide power to TACAN and ILS. It will operate simultaneously with, but independently of, the existing hydraulic powered emergency ac generator. The ATG is driven by the low-pressure pneumatic system. An automatic changeover feature is incorporated to energize the ATG in the event of main ac power failure.

AC Generator Switch

On Ⓐ airplanes a guarded ac generator switch (3, figure 1-11) is located on the right console. On Ⓑ airplanes the switch is located on the right console in the forward cockpit only. The switch controls the ac generator if the master electrical power switch is ON. The switch is placarded "AC Gen Cont" or "AC Gen" and has three positions, OFF, ON, and RESET. The RESET position is inoperative. Actuating the switch from OFF to ON position with the master power switch ON energizes the ac generator. The generator then powers ac essential and nonessential buses. The generator is reset by actuating the generator switch to OFF, then ON.

Emergency AC Generator Switch

On Ⓐ airplanes the emergency ac generator switch is located above the right console. On Ⓑ airplanes the switch is located beside the right console in the forward cockpit only. The switch has two positions, NORMAL and START, and is springloaded to the NORMAL position. With the ac generator switch OFF, placing the switch to START energizes the emergency ac generator. The switch receives power from the battery bus. The dc generator switch must be OFF prior to checking operation of the emergency ac generator.

ATG Switch

The ATG switch is located above the right console. On Ⓑ airplanes the switch is located in the forward cockpit only. The switch is placarded "EMER AC GEN ATM" on Ⓐ airplanes and "ATM GEN" on Ⓑ airplanes and has TEST, OFF, and AUTO positions. The switch is spring-loaded to OFF from the TEST position. Holding the switch to TEST permits ground test of the ATG. The switch should be in OFF during engine start and shutdown to avoid unnecessary operation of the ATG. The switch should be in AUTO during flight to energize the ATG in the event of main ac power failure. The switch receives power from the dc essential bus.

AC Power Failure Warning Light

The ac power failure warning light (1, figure 1-30), located on the warning light panel, illuminates and displays "AC POWER FAIL" if the ac generator fails. The light receives power from the dc essential bus.

ELECTRICALLY OPERATED EQUIPMENT

See figure 1-15 for complete reference to electrically operated equipment.

EXTERNAL POWER

External dc, ac, and MA-1 power is connected to the airplane through a single external power receptacle on the left side of the airplane. External dc and ac power take precedence over airplane power.

MA-1 ELECTRICAL POWER SUPPLY SYSTEM

The MA-1 electrical power supply system (figure 1-15) consists of two multivoltage generators driven by an engine-driven constant-speed drive unit, and voltage regulators which maintain proper voltages for the airplane and weapon control system components. One generator is a dc multivoltage generator; the other is a dc and ac multivoltage generator. Transfer relays and switches are provided to transfer essential MA-1 components to the airplane ac and dc essential buses in the event of MA-1 electrical power supply failure. The power supply is controlled by the MA-1 power switch.

MA-1 Power Switch

The MA-1 power switch is located on the MA-1 power control panel (figure 1-15A)on the left console. On Ⓑ airplanes the MA-1 power control panel and the MA-1 power switch are located in the forward cockpit only. The switch is placarded "Power" and has OFF, WARM, RADAR STBY, ON, RADIO SILENCE, and EMER positions. Switch functions of the MA-1 power switch are as follows:

NOTE

Refer to figure 1-15B for a detailed description of the delay times associated with the positions of the power switch.

1. OFF — No power is applied to the MA-1 system.
2. WARM — Components requiring more than two minutes warmup time are energized with the switch in this position. The stable platform will not be energized in this position when ground power is used. The UHF command radio will be operative after approximately two minutes. This is the normal position of the MA-1 power switch when the airplane is on the ground and on a two-minute alert status with ground power applied.
3. RADAR STBY — After initial delay period, all system functions are operative except the radar transmitter and antenna servo.
4. ON — Power is supplied to all system components after delay times have elapsed.
5. RADIO SILENCE — All transmitting equipment is suppressed except the UHF transmitter which can be operated in the normal manner. The radar and IFF will not be available. TACAN distance information is not available, but TACAN bearing information will be available if a valid TACAN signal is being received.
6. EMER — All MA-1 powered equipment except UHF command radio, command radio ADF, TACAN, ILS, and ground-to-air IFF is shut off.

The MA-1 power switch receives power from the airplane dc essential bus.

Power Annunciator

The power annunciator is located on the power control panel adjacent to the MA-1 power switch. On B airplanes the power annunciator is located in the forward cockpit only. The Annunciator displays "TEST" if test switches in the system have been left in the test position. TEST may not be displayed until STBY power is received. If the test circuits are not in "TEST" condition, the annunciator displays "WAIT" indicating that the gyro has not erected. When the gyro is erected, the annunciator displays "OK," indicating that the gyro is ready for operation. The annunciator receives power from the MA-1 power supply system. When ground power is used, the annunciator will be operative only when the MA-1 power switch is in RADAR STBY or above.

Gyro Erect Button

The gyro erect button is located on the MA-1 power control panel (figure 1-15A) and is placarded "Press to Erect." On B airplanes the gyro erect button is located in the forward cockpit only. The button is used to erect the gyros of the stable platform prior to takeoff or during flight. Momentary pressure on the switch causes the gyros to return to the erection cycle for 20 seconds, during which time the gyro references itself to the heading shown in the gyro reference indicator window. If the flight modes switch is at AUTO or ASSIST when the gyro erect button is depressed, the switch will automatically disengage and the flight mode selector switch will rotate to the PITCH DAMPER position. Rotation of the button controls the vertical position of the artificial horizon on the radar scope. The gyro erect button receives power from the MA-1 system.

Gyro Grid Reference Knob And Indicator Window

The gyro grid reference knob and indicator window (figure 1-15A) are located on the MA-1 power control panel. On B airplanes the gyro grid reference knob and indicator window are located in the forward cockpit only. They are marked with a common placard which reads "Grid Reference." The grid reference control is used to set the computer grid reference at the time of gyro erection. The grid reference heading is read in the indicator window. When the indicator is set at airplane heading, the gyro erects and aligns to the same heading as the airplane. A two-minute delay from time of power application is incorporated into the gyro erection circuit to allow the gyro to obtain operating speed. Power is supplied to the gyro at any time the power switch is not in OFF, WARM, or EMER and external power is on the airplane. With airplane power on, power is supplied to the gyro any time the power switch is not in OFF or

MA-1 power control panel (typical)

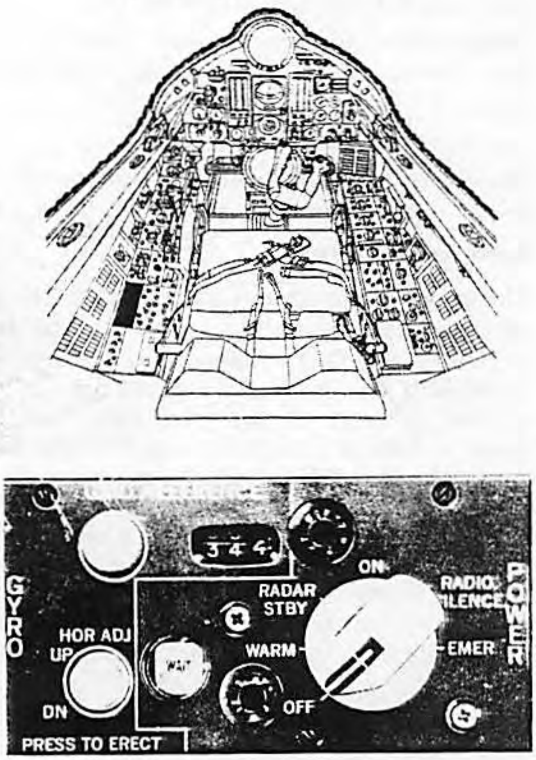


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Figure 1-15A

EMER. If the heading in the indicator window is the same as the computer grid reference at completion of the two-minute delay period, the gyro will automatically align correctly. If the heading set into the window is different from computer grid reference at completion of the two-minute delay period, computer grid reference must be set into the window, the gyro erect button must be depressed, and airplane heading must be held constant for 20 seconds. While the gyro is in the erection cycle, the annunciator will read "WAIT." After 20 seconds, the annunciator will read "OK." The artificial horizon on the scope should be checked for alignment. If the artificial horizon is not aligned, repeat the gyro erection procedures before takeoff.

ELECTRICAL POWER SWITCHING CHARTS

The Electrical Power Switching Charts (figure 1-15C) show the effect of MA-1 power switch positions upon various electrically dependent components.

NOTE

The following are either independent of electrical power, or require battery power only. Therefore, these items do not appear on figure 1-15C.

1. All airplanes:
   - a. Armament salvo.
   - b. Cabin pressure altitude gage.
   - c. Canopy jettison.
   - d. Drag chute.
   - e. Emergency fuel transfer.
   - f. Exhaust gas temperature gage.
   - g. Landing gear warning light.
   - h. Manual engine anti-icing.
   - i. Speed brakes.
   - j. Standby compass.
   - k. Surface trim.
   - l. Tachometer
   - m. Turn-and-slip indicator.
   - n. Variable ramp emergency retraction.
2. Conventional instrument display only:
   - a. Accelerometer.
   - b. Airspeed indicator.
   - c. Altimeter.
   - d. Vertical velocity indicator.
3. Integrated flight instrument system only:
   - a. Standby airspeed indicator.
   - b. Standby altimeter.

HYDRAULIC POWER SUPPLY SYSTEM

The hydraulic power supply system (figure 1-16) consists of two separate constant-pressure type systems, the primary and secondary, which supply power to actuate most of the airplane's major operating components. Normal operation of both hydraulic systems is automatic whenever the engine is running. An emergency system is also provided to supplement the primary system in an emergency. Two gages, one for each system, provide an indication of hydraulic system pressure. A hydraulic pressure-low warning light flashes to indicate single system failure or illuminates steadily with failure of both systems. Hydraulic pressure line tubing, routed through the No. 3 fuel tanks, cools the hydraulic fluid. See figure 2-8 for hydraulic fluid specifications. For additional information on this system, refer to T.O. 1F-106A-2-3.

MA-1 system time-delay block diagram

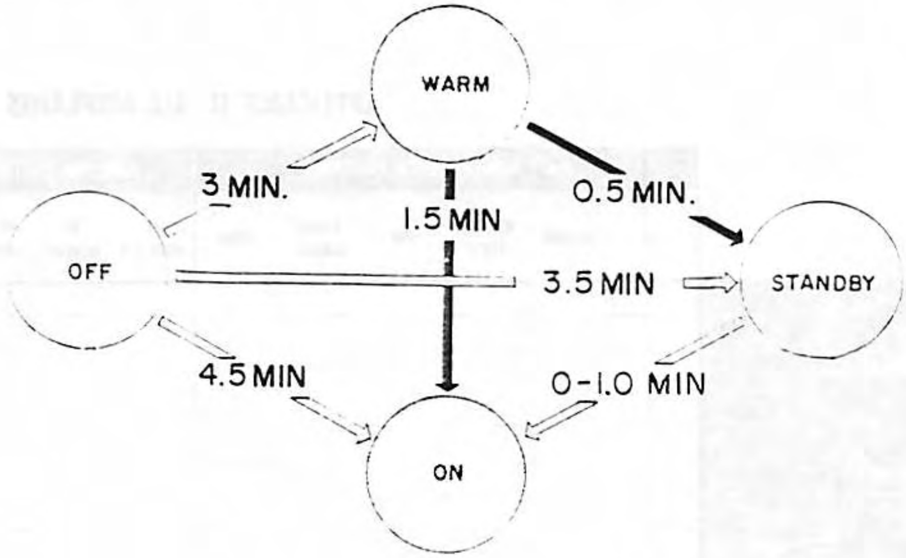


Figure 1-15B

electrical power

APPLICABLE TO ALL AIRPLANES

ELECTRICAL COMPONENT	MA-1 POWER SWITCH POSITIONS						AIRPLANE ELECTRICAL POWER FAILURE				
	OFF	WARM	RADAR STBY	ON	RADIO SILENCE	EMER	DC NON-ESS	AC NON-ESS	AC & DC NON-ESS	ALL AC	ALL AC & DC NON-ESS
RADAR											
AFCS (ASSIST)											
OXYGEN QUANTITY GAGE											
IFF/SIF											
DATA LINK											
MARKER BEACON											
UHF RADIO											
ARMAMENT CAPABILITY USING OPTICAL SIGHT — "IR" MISSILES											
ARMAMENT CAPABILITY USING OPTICAL SIGHT — SPECIAL WEAPON											
ARMAMENT CAPABILITY USING OPTICAL SIGHT — RETICLE LIGHT											
PITCH AND YAW DAMPER											
VARIABLE RAMPS (AUTO)											
NOSE WHEEL STEERING											
BOOST PUMPS											
WINDSHIELD ANTI-ICING											
AUTO ENGINE ANTI-ICING											
COCKPIT FLOODLIGHTS											
STORM LIGHTS											
ROTATING BEACON LIGHT											
FORMATION LIGHTS											
INSTRUMENT & CONSOLE LIGHTS											
MAXIMUM MANEUVER LIGHT											
GEAR AUDIO WARNING											
CABIN HEAT											
PITOT HEAT											
AIR DATA COMPUTER											
NORMAL FUEL TRANSFER											
FUEL QUANTITY GAGE											
FUEL FLOW INDICATOR											
ENGINE PRESSURE RATIO GAGE											
ENGINE OIL PRESSURE GAGE											
HYDRAULIC PRESSURE GAGE											
EGT SPREAD											
STANDBY ATTITUDE INDICATOR											

48207-1

Figure 1-15C

switching chart

AIRPLANES WITH CONVENTIONAL INSTRUMENT DISPLAY

ELECTRICAL COMPONENT		MA-1 POWER SWITCH POSITIONS: OFF	WARM	RADAR STBY	ON	RADIO SILENCE	EMER	AIRPLANE ELECTRICAL POWER FAILURE: DC NON-ESS	AC NON-ESS	AC & DC NON-ESS	ALL AC	ALL AC & DC NON-ESS
MACH INDICATOR	MACH											
	COMMAND INPUTS											
COURSE INDICATOR	COMPASS											
	ILS & TACAN											
	UHF-ADF											
	DL-ADF											
APPROACH HORIZON	ATTITUDE											
	ILS											
COMMAND & TARGET ALT. INDICATOR						1						
ANGLE OF ATTACK												
ATTITUDE INDICATOR												

AIRPLANES WITH INTEGRATED FLIGHT INSTRUMENT SYSTEM

ELECTRICAL COMPONENT		MA-1 POWER SWITCH POSITIONS: OFF	WARM	RADAR STBY	ON	RADIO SILENCE	EMER	AIRPLANE ELECTRICAL POWER FAILURE: DC NON-ESS	AC NON-ESS	AC & DC NON-ESS	ALL AC	ALL AC & DC NON-ESS
AIRSPEED-MACH INDICATOR	MACH & AIRSPEED											
	COMMAND INPUTS											
ATTITUDE-DIRECTOR INDICATOR	ATTITUDE											
	ILS & TACAN											
ALT-VERTICAL VELOCITY IND	ALTITUDE											
	COMMAND INPUTS											
HORIZONTAL SITUATION INDICATOR	COMPASS											
	ILS & TACAN					1						
	COMMAND INPUTS											
	UHF-ADF											
	DL-ADF											

WILL FUNCTION

1 ONLY TACAN DME LOST

48207-26

Figure 1-15C

PRIMARY HYDRAULIC SYSTEM

The primary hydraulic system supplies power for operation of the flight controls and yaw damper only. The system is completely independent of the secondary system and consists primarily of a reservoir, a 3000 psi variable volume engine-driven pump, an accumulator, supply lines, and a thermal-pressure relief valve for system protection. The system contains conventional filters with bypass features. The reservoir has a capacity of approximately 1.72 U.S. gallons and is pressurized by pressure-regulated engine bleed air and the high-pressure pneumatic system.

NOTE

> Primary hydraulic reservoir pressurization provides additional hydraulic inlet pump pressure, reducing possible hydraulic pump cavitation during engine start. Primary hydraulic reservoir pressure can be checked with the primary hydraulic reservoir pressure gage in the hydraulic compartment.

A pressure gage is located adjacent to the piston-type accumulator to permit ground checking of the preload pressure. The ram air turbine will also supply pressure to the primary hydraulic system as an emergency source of pressure.

SECONDARY HYDRAULIC SYSTEM

The secondary hydraulic system supplies power for operation of the flight controls in response to control stick movement (parallels the primary system action) and pitch damper control action in response to electronic pitch signals. The system also supplies power for operation of the landing gear and doors, nose wheel steering, speed brakes, engine variable inlet ramp, air refueling slipway door and boom latches (if installed), and the emergency ac generator. The secondary hydraulic system consists primarily of a reservoir, a 3000 psi variable volume engine-driven pump, an accumulator, supply lines, and a thermal-pressure relief valve for system protection. The reservoir has a capacity of approximately 1.9 U.S. gallons and is pressurized by pressure-regulated engine bleed air and the high-pressure pneumatic system. A pressure gage is located adjacent to the piston-type accumulator to permit ground checking of the preload pressure.

NOTE

> Pressurization of the secondary hydraulic reservoir provides additional hydraulic inlet pump pressure to reduce the possibility of hydraulic pump cavitation during engine start. Secondary hydraulic reservoir pressure can be checked with the secondary hydraulic reservoir pressure gage located in the hydraulic compartment.

MAGNETRON HYDRAULIC SYSTEM

The magnetron hydraulic system operates independently of the airplane hydraulic power supply system. The system provides power for tuning the MA-1 radar subsystem magnetron. System components are an engine driven pump, an electric motor driven pump (for operation on external power), a reservoir, an accumulator, and an oil cooler. Reservoir capacity is approximately 0.7 gallon. A filler valve and quantity gage are located inside the right engine access door. There are no separate controls for the magnetron hydraulic system.

NOTE

> The engine must be at 70% rpm or above to supply sufficient hydraulic pressure for proper magnetron tuning.

EMERGENCY HYDRAULIC SYSTEM

The emergency hydraulic system supplies power for operation of the flight controls in event of failure of the primary and secondary hydraulic systems or when the engine fails and is "frozen." The emergency hydraulic pump is driven by a variable pitch ram air turbine (RAT), which is pneumatically extended into the airstream by actuating the hydraulic emergency power handle. Since the emergency hydraulic system utilizes the same hydraulic lines as the primary hydraulic system, it is inadvisable to extend the RAT when only the secondary hydraulic system has failed, as this could cause damage and possible loss of the primary hydraulic system. Once extended the RAT

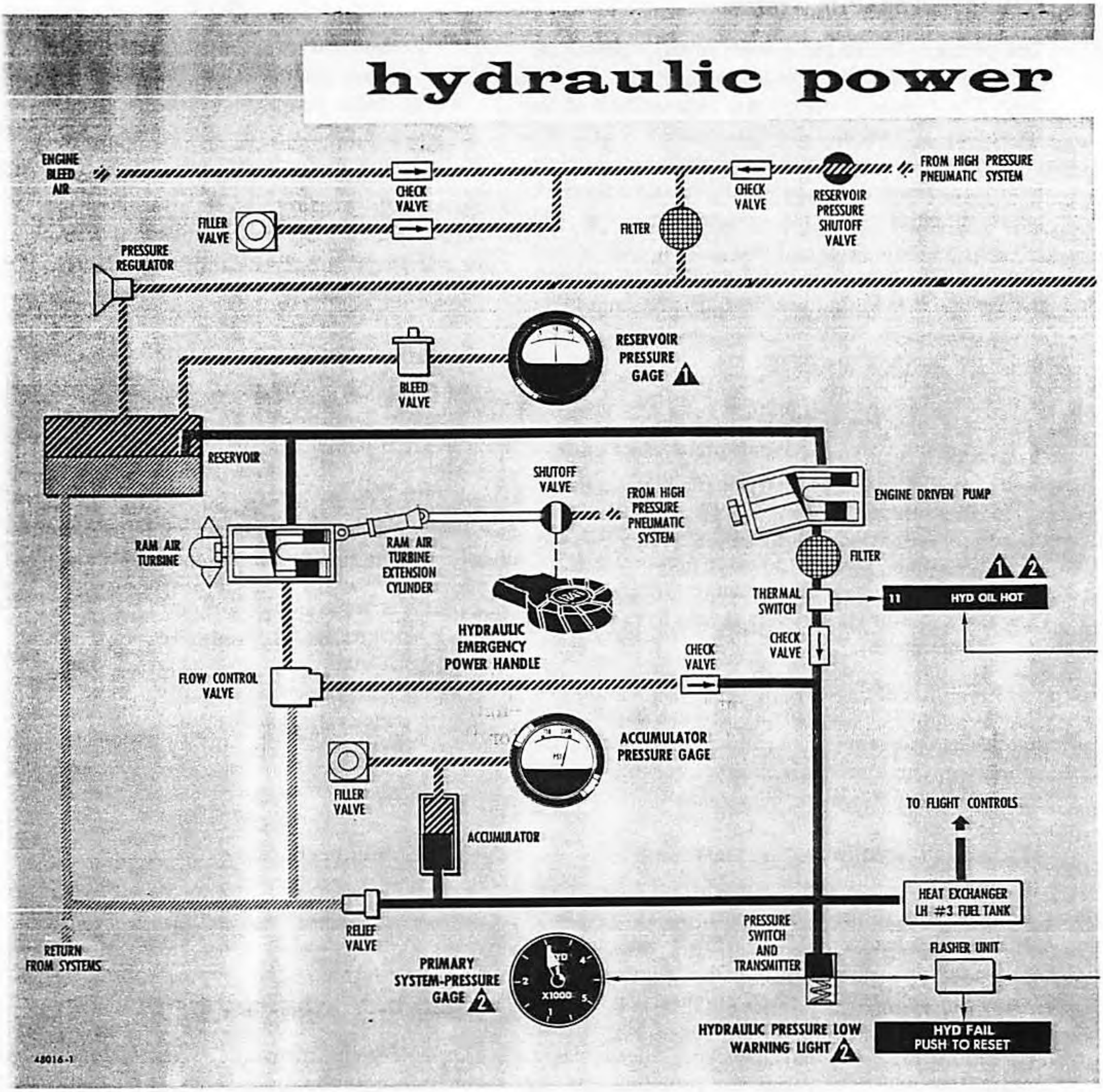


Figure 1-16

cannot be retracted in flight. This emergency system will supply sufficient power for limited maneuvering, approach and landing. A RAT door test hook is installed on the RAT door eyebolt, to insure that the RAT door is locked. The hook is equipped with a warning streamer and must be removed prior to flight.

Ram Air Turbine (RAT) Handle

The ram air turbine (RAT) handle (5, figure 1-10), located on the left side of the cockpit, is a round grooved handle formed to resemble a turbine. The handle is used to energize the emergency hydraulic power system which supplies hydraulic pressure to the primary hydraulic system. Moving the handle down mechanically selects pneumatic pressure to extend the ram air turbine into the airstream. When the handle is moved fully down, it locks in this position. Normally, after the handle is moved down during an inflight emergency, it should remain in this position.

supply system

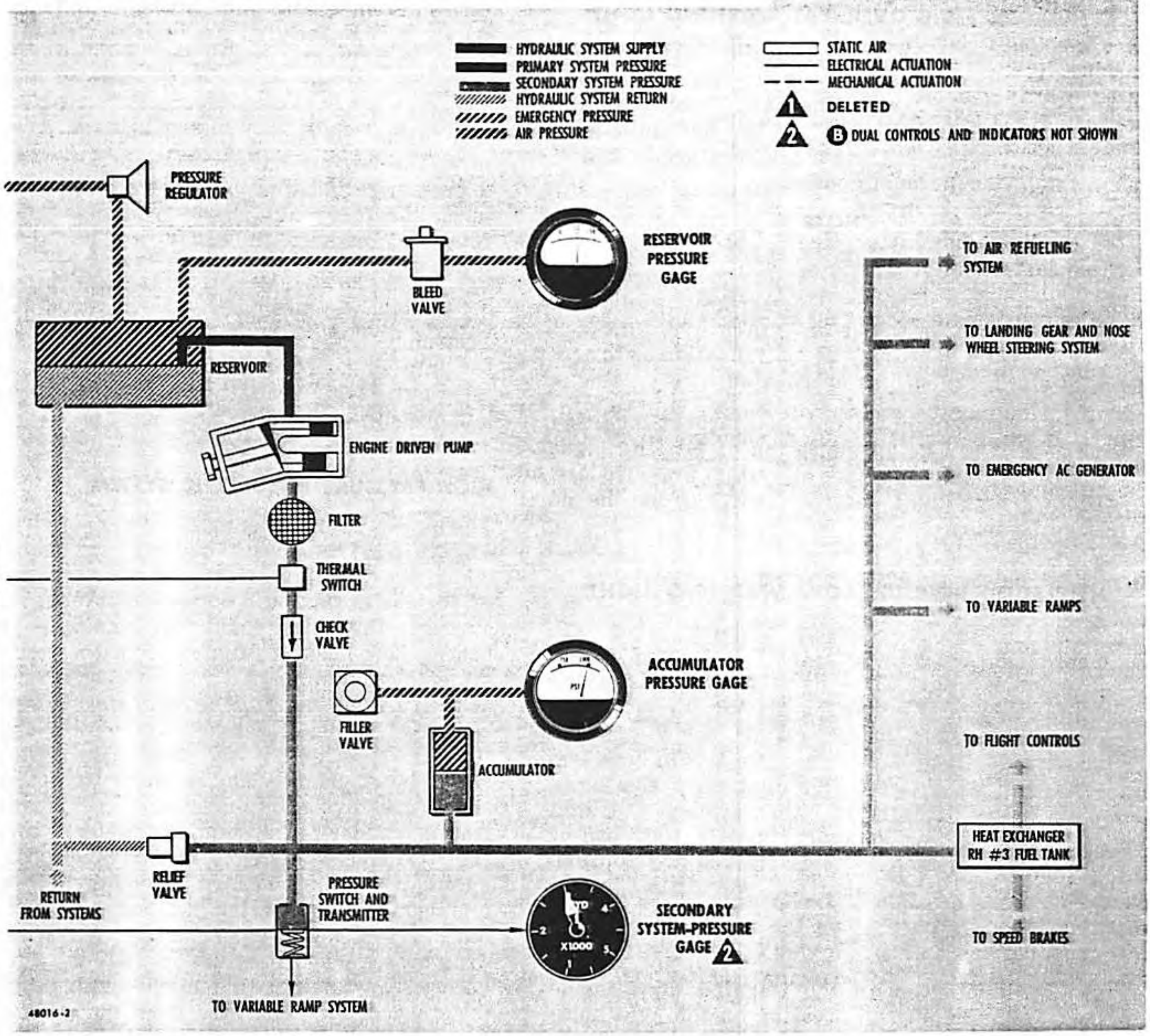


Figure 1-16

WARNING

During certain circumstances (e.g., preflight checks) the handle must be pulled up to prevent injury.

NOTE B

On some B airplanes* the RAT handle must be pushed aft and down to extend the ram air turbine. The handle can be returned to the up position by pushing aft, then raising the handle.

**AF 57-2516 thru 57-2519.*

HYDRA'JLIC SYSTEM PRESSURE GAGES

Two hydraulic system pressure gages (42 and 43, figure 1-11), located on the right-hand console, indicate the hydraulic systems pressure. One gage is placarded "Primary" and indicates primary system pressure in psi. The other gage is placarded

"Secondary" and indicates secondary system pressure in psi. The hydraulic pressure gages receive power from the ac essential bus.

HYDRAULIC FLUID OVERHEAT WARNING LIGHT

Some airplanes have a hydraulic fluid overheat warning light (11, figure 1-30) located on the master warning panel. The light illuminates and displays "HYD OIL HOT" if the hydraulic fluid in either system reaches a temperature of 275°F at the hydraulic pump outlet.

NOTE

If the light illuminates, there is no positive method of determining which system has overheated. Excessive hydraulic pressure, as noted on either hydraulic pressure gage, may be an indication as to which system has overheated.

If the light illuminates and the overheat condition persists, it can be an indication of pending flight control system oscillations or system pressure loss. The warning light receives power from the dc essential bus.

HYDRAULIC PRESSURE-LOW WARNING LIGHT

The hydraulic pressure-low warning light (29, figure 1-8, and 26, figure 1-9) is located on the instrument panel. The light flashes, displaying "HYD FAIL," if either the primary or secondary hydraulic system pressure falls below approximately 900 (±100) psi. Pressure loss in both systems will cause steady illumination of this light. With rising pressure in either system, the light will start flashing at approximately 1000 psi, or if pressure in both systems rises above approximately 1000 psi, the light will go out. If only one hydraulic system has failed, the flashing light can be extinguished by pushing on the light housing. Extinguishing the flashing light will not prevent steady illumination of the warning light should both systems fail, and pressing on the light housing will not turn the light off if both hydraulic systems have failed. The hydraulic pressure-low warning light is tested together with the other lights in the master warning system, and will flash when the master warning light test button is depressed, during engine operation. The light receives power from the dc essential bus.

PNEUMATIC POWER SUPPLY SYSTEM

The pneumatic power supply system consists of two separate systems, low- and high-pressure, which are used for pressurizing and actuation of system components. For additional information on this system, refer to T.O. 1F-106A-2-3.

LOW-PRESSURE PNEUMATIC SYSTEM

The low-pressure pneumatic system (figure 4-1) obtains bleed air from the last compression stage of the high-pressure compressor. The bleed air varies in temperature and pressure up to approximately 875°F and 225 psi under extreme operational and climatic conditions. A portion of this air is passed through a refrigeration unit that includes air-to-air heat exchangers and an expansion turbine to reduce the temperature and pressure. The low-pressure pneumatic system is used to pressurize and air-condition the cockpit; to pressurize the fuel tanks and hydraulic system reservoirs; to pressurize the canopy seal and the anti-g suits; to pressurize the CSD; to supply operating pressure to the variable air-pressure regulator in the elevator feel system; to cool and pressurize the electronic compartment; and to supply warm air for anti-icing, rain-clearing, defogging, and ATG. See figure 4-1 for additional information on the low-pressure pneumatic system.

HIGH-PRESSURE PNEUMATIC SYSTEM

The high-pressure pneumatic power supply system (figure 1-17) consists of several component parts which provide high-pressure air for the operation of various normal and emergency systems. Air is stored in three flasks and four landing gear drag braces. One flask is isolated from the rest of the system, and supplies air for emergency operation of the variable ramps. The main storage system is composed of the other two flasks, which supply air for the operation of the armament system, IR seeker head, primary and secondary hydraulic reservoir pressurization; emergency cockpit pressurization, rudder feel system, emergency forward cg fuel transfer system on Ⓐ airplanes; constant-speed drive air-cooler valve emergency system, and combustion starter. In addition, the main storage system provides air for recharging the drag braces when pressure in these storage areas becomes lower than that in the main flasks. The drag braces are separated from the main system by check valves which allow the braces to maintain their pressure in the event of failure of the main system. The aft drag braces supply air for the operation of the wheel brakes only; the forward drag braces supply air for emergency landing gear extension, ram air turbine extension, air refueling slipway door emergency operation, emergency speed brake operation, and drag chute operation. The pneumatic system is charged on the ground through a filler valve located in the left-hand main wheel well. No provision is made for inflight recharging. Main system pressure is indicated on a pressure gage located near the ground filler valve. When fully serviced, the system pressure is 3000 psi. A pressure regulator

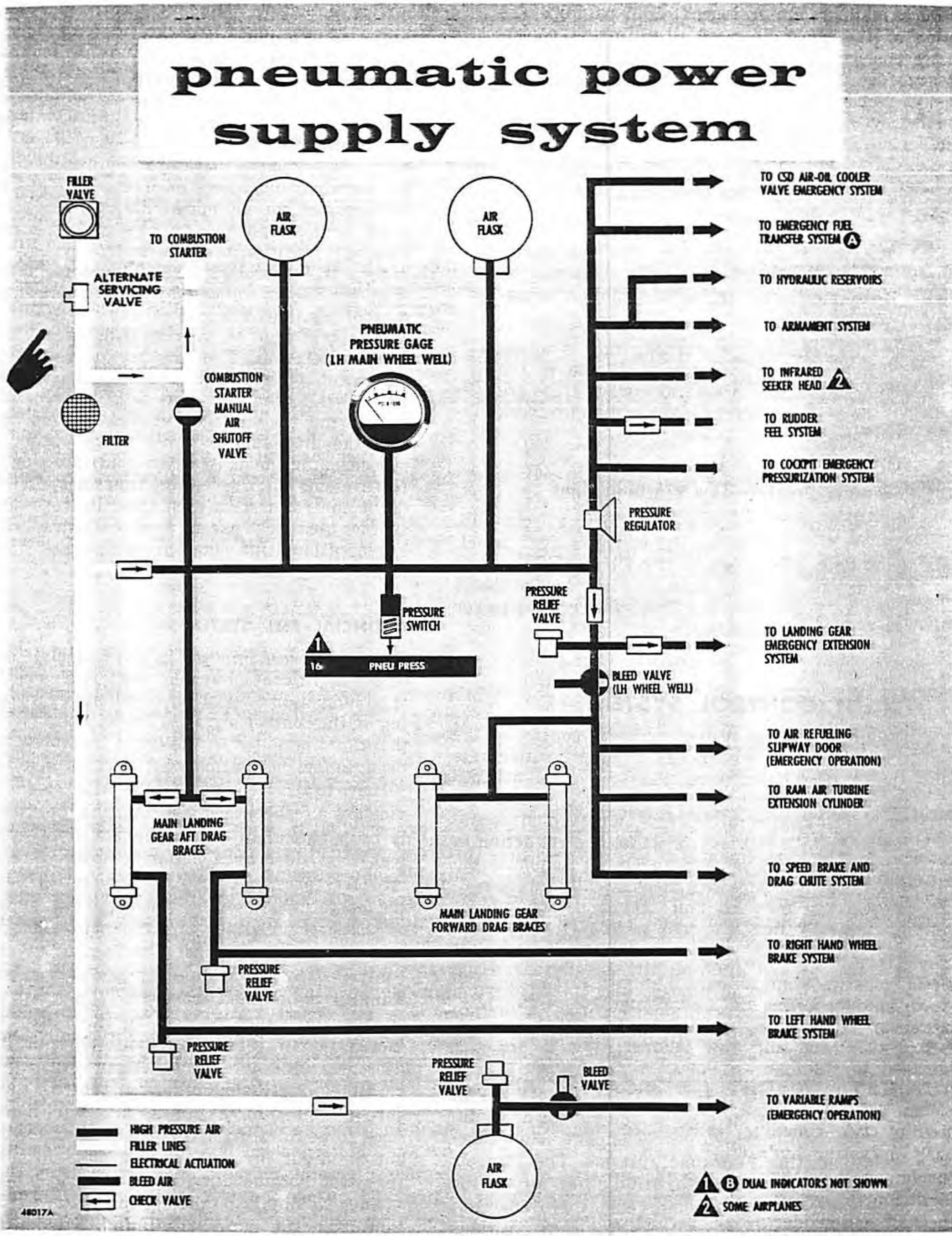

pneumatic power supply system
FILLER VALVE
TO COMBUSTION STARTER
ALTERNATE SERVICING VALVE
AIR FLASK
AIR FLASK
PNEUMATIC PRESSURE GAGE (LH MAIN WHEEL WELL)
COMBUSTION STARTER MANUAL AIR SHUTOFF VALVE
FILTER
TO CSD AIR-OIL COOLER VALVE EMERGENCY SYSTEM
TO EMERGENCY FUEL TRANSFER SYSTEM A
TO HYDRAULIC RESERVOIRS
TO ARMAMENT SYSTEM
TO INFRARED SEEKER HEAD 2
TO RUDDER FEEL SYSTEM
TO COCKPIT EMERGENCY PRESSURIZATION SYSTEM
PRESSURE REGULATOR
PRESSURE RELIEF VALVE
PRESSURE SWITCH
1
16 PNEU PRESS
TO LANDING GEAR EMERGENCY EXTENSION SYSTEM
BLEED VALVE (LH WHEEL WELL)
TO AIR REFUELING SLIPWAY DOOR (EMERGENCY OPERATION)
TO RAM AIR TURBINE EXTENSION CYLINDER
MAIN LANDING GEAR AFT DRAG BRACES
TO SPEED BRAKE AND DRAG CHUTE SYSTEM
MAIN LANDING GEAR FORWARD DRAG BRACES
TO RIGHT HAND WHEEL BRAKE SYSTEM
PRESSURE RELIEF VALVE
TO LEFT HAND WHEEL BRAKE SYSTEM
PRESSURE RELIEF VALVE
PRESSURE RELIEF VALVE
BLEED VALVE
TO VARIABLE RAMPS (EMERGENCY OPERATION)
HIGH PRESSURE AIR
FILLER LINES
ELECTRICAL ACTUATION
BLEED AIR
CHECK VALVE
AIR FLASK
1 B DUAL INDICATORS NOT SHOWN
2 SOME AIRPLANES
48017A

Figure 1-17

reduces main system pressure to 1500 psi for operation of the constant-speed drive air-oil cooler valve emergency system, IR seeker head, emergency fuel transfer system, armament system, rudder feel system and cockpit emergency pressurization system. A pneumatic pressure-low warning light illuminates when pressure in the main system is too low to adequately supply air for the systems that rely upon the main flasks.

NOTE

- After illumination of the pneumatic pressure-low warning light, there may be insufficient air to initiate another armament cycle.
- The pressure-low warning light is not an indication of pressure available in the drag braces, since these storage areas are separated from the main system by check valves.

Pneumatic Pressure-Low Warning Light

The pneumatic pressure-low warning light (16, figure 1-30), located on the warning light panel, illuminates and displays "PNEU PRESS" when pressure in the two main system storage flasks drops to 1700 (±50) psi. The light receives power from the dc essential bus.

FLIGHT CONTROL SYSTEM

The flight control system provides control of the airplane from low speeds through supersonic speeds. The delta wing configuration utilizes elevons instead of aileron and elevator control surfaces. The system incorporates a control stick and rudder pedals with conventional control action in response to stick movement. On Ⓑ airplanes the system incorporates two mechanically linked control sticks (forward and aft) and two sets of mechanically linked rudder pedals (forward and aft). The elevons, when moved coincidentally, act as elevators and differentially as ailerons. Each elevon consists of an inboard and outboard panel which function as one unit, permitting free surface movement unimpaired by normal inflight wing deflections. Pitch and yaw damper systems are installed to stabilize high-speed flight. Refer to AUTOMATIC FLIGHT CONTROL SYSTEM, Section IV. Both the elevons and rudder are actuated by two complete, independent, simultaneously operating hydraulic systems. The ram air turbine can also supply hydraulic pressure to the flight control system in emergencies. Movement of the stick and rudder pedals mechanically positions hydraulic control valves, which direct primary and secondary hydraulic pressure to the respective control surface actuating cylinders. Control surface deflection is proportioned to cockpit control movement by followup linkages which shut off hydraulic pressure to the control surface actuators. Since aerodynamic force against the control surfaces is restricted by the hydraulic action, the forces are not transmitted to the cockpit controls. An artificial feel system is therefore required to simulate the aerodynamic forces encountered. The control surfaces are not equipped with trim tabs as trimming is accomplished by changing the neutral (no-load) position of each control system which deflects the control surfaces. Elevator and rudder trim position is indicated by the no-load cockpit control position. The control stick does not move with application of aileron trim except during AFCS operation. The absence of control stick movement with application of aileron trim is attributed to the fact that trim is accomplished by varying the linkage between the feel system and the hydraulic control valve rather than repositioning the neutral (no-load) position of the feel system. The irreversible characteristics of the flight control system eliminate the necessity for surface gust locks except during storm conditions. For additional information on this system, refer to T.O. 1F-106A-2-7.

ARTIFICIAL FEEL SYSTEM

Since the forces imposed by the hydraulic portion of the flight control system are irreversible, there is no indication of existing aerodynamic forces acting on the control surfaces. An artificial feel force, therefore, is added to the flight control system to produce feel on the control stick and rudder pedals relative to airspeed and altitude. Aileron feel is provided by a feel-centering spring, and the stick force is proportional to stick deflection only. Elevator feel is provided by a centering spring and a variable feel force cylinder. A large piston in the elevator feel force cylinder is attached so that movement of the stick in either direction moves the piston against ram air pressure. This ram air pressure is controlled by a variable air pressure regulator. Ram air is obtained from the ¾-inch "q" intake on the fin. Since control surface effectiveness increases with increasing speed subsonically and decreases with increasing speed supersonically, the variable air pressure regulator controls the pressure in the feel force cylinder to provide a uniform stick force for a given airplane response throughout the flight envelope. The variable air pressure regulator obtains airspeed intelligence from the ¼-inch "q" intake on the fin, and altitude information from the main wheel well. Rudder feel is also provided by a centering spring and feel force cylinder. A small piston in the rudder feel force cylinder is attached so that movement of either rudder pedal moves the piston against pneumatic system pressure. This pressure is obtained from high-pressure pneumatic system pressure which has been regulated by the rudder air pressure regulator. The rudder air pressure regulator derives airspeed intelligence from the 1/4-inch "q" intake on the fin.

TRIM SYSTEM

Flight control trim is accomplished by deflecting the control surfaces through the use of electrical trim actuators. Elevator and rudder trim actuators are installed in the flight control system to reposition the neutral (no-load) position of the cockpit controls and control surfaces; therefore, cockpit controls and control surfaces will move in response to trim changes. Elevon (aileron and elevator action) trim is controlled by a switch on the control stick, and rudder trim is controlled by a switch on the throttle quadrant. Takeoff trim can be attained automatically by depressing a button on the throttle quadrant, which will reposition aileron, elevator, and rudder trim to a preset position. A green indicator light illuminates when the proper takeoff trim positions are obtained. Manual trim changes are required to maintain level flight from minimum to maximum speed. The trim system is powered from the dc essential bus.

Elevon Trim Switch

Lateral and longitudinal trim is controlled by a five-position elevon trim switch (figure 1-18) located on the control stick grip. The switch has NOSE UP, NOSE DOWN, LWD, and RWD positions, and is spring-loaded to the center (off) position. Holding the switch in the direction of desired trim powers the aileron or elevator trim actuator to reposition the neutral (no-load) position of the respective flight control system. When

control stick

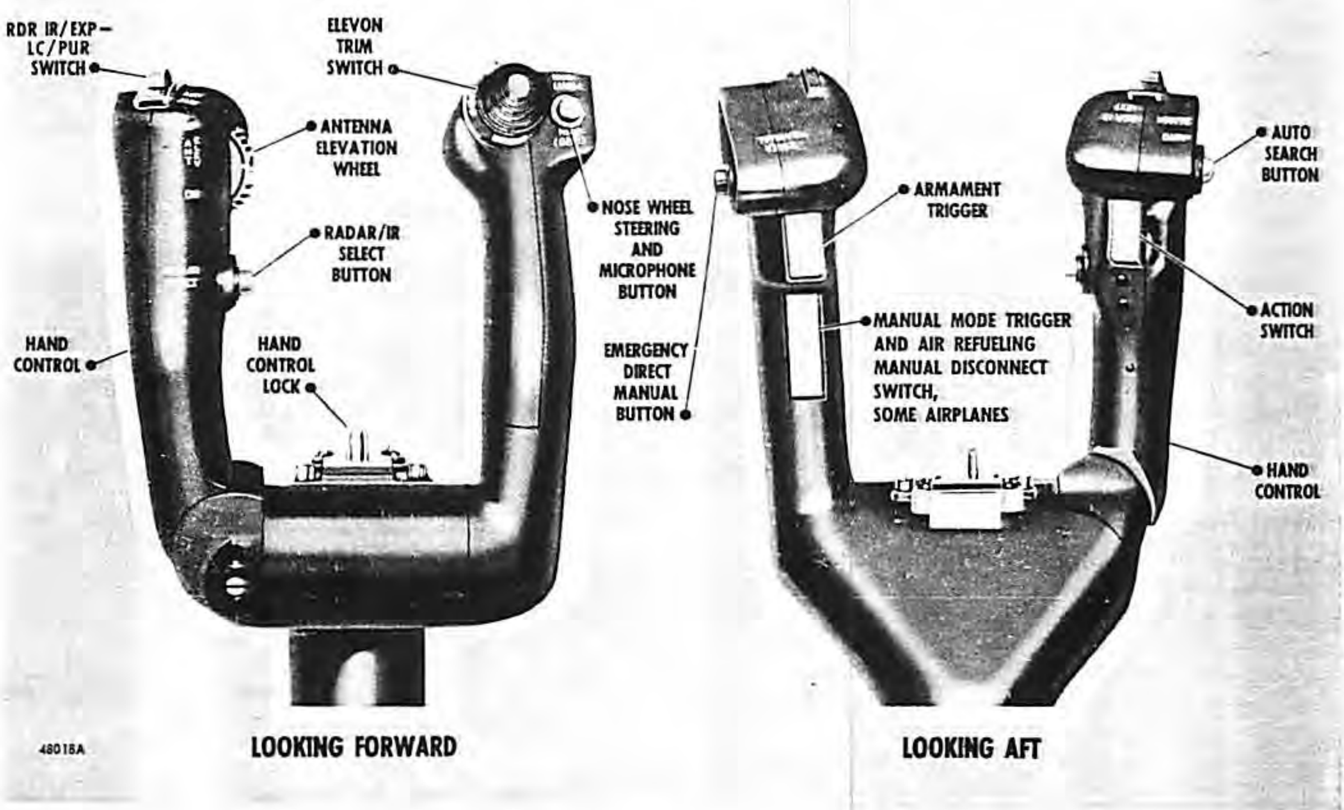


Figure 1-18

released the switch automatically returns to its spring-loaded center (off) position. When the automatic flight control system is in assist mode, the elevon trim switch can be used to make small changes in bank or pitch attitude. The trim switch receives power from the dc essential bus.

CAUTION

Should the trim switch stick in an actuated position, an extreme application of trim will result. The trim system can be overridden by control stick movement. If this condition exists on preflight inspection, the airplane should not be flown until the condition is corrected. If the switch sticks in flight it will be necessary to return to the center (off) position manually after the desired trim change is made. A sticking trim switch should be noted on Form 781.

Rudder Trim Switch

The three-position rudder trim switch (figure 1-6), located on the throttle quadrant, controls the position of the rudder trim actuator to reposition the neutral (no-load) position of the rudder control system. The switch, placarded "Rudder Trim," has positions L and R and is spring-loaded to the center (off) position. Holding the switch to L or R applies trim in the respective direction.

CAUTION Ⓑ

On Ⓑ airplanes, when the forward and aft elevon and rudder switches are used to simultaneously apply opposite trim, electrical power will be supplied to both sides of the trim actuator motor and may burn out the motor.

When released the switch automatically returns to its spring-loaded center (off) position. The trim switch receives power from the dc essential bus.

Takeoff Trim Button And Indicator Light

The takeoff trim button (figure 1-6), located on the throttle quadrant, provides for automatic trimming of the control surfaces to the proper position for takeoff. When the takeoff trim button is depressed, rudder and aileron are trimmed to the neutral position and elevator is trimmed to the 3½° up position. While the button is held depressed, a green indicator light will illuminate "TAKEOFF TRIMMED" when the trim system reaches the proper position for takeoff. The takeoff trim circuit receives power from the dc essential bus.

CAUTION

Do not actuate the takeoff trim system unless hydraulic pressure is available, to prevent damage to the actuator motors.

CONTROL STICK

The control stick (figure 1-18) controls the position of the elevon hydraulic control valves which direct primary and secondary system hydraulic pressure to the actuating cylinders to displace the elevon control surfaces. On Ⓑ airplanes the forward and aft control sticks are mechanically interconnected. Followup linkages then reposition the control valves spool and enable control surface deflection to be in proportion to stick movement. Fore and aft control stick movements are used to obtain elevator action of the elevons. The control stick is moved right and left to obtain aileron action. The control stick has two grips mounted on a common base. The right-hand grip is the primary grip and incorporates a nose wheel steering and microphone button, elevon trim switch, manual mode trigger (momentary interrupt trigger), armament trigger, and an emergency direct manual button (emergency damper disconnect button). On airplanes with air refueling capability, the manual mode trigger serves as a manual disconnect switch when the air refuel switch is ON (Ⓑ front cockpit only). The left-hand grip, when unlocked, serves as the radar antenna hand control and incorporates controls for the fire control system.

RUDDER PEDALS

The rudder pedals control the position of the rudder hydraulic control valve which directs primary and secondary system hydraulic pressure to the rudder actuating cylinder. On Ⓑ airplanes the forward and aft rudder pedals are mechanically interconnected. Followup linkage repositions the control valve which maintain rudder deflection in proportion to pedal movement. The rudder pedals can be simultaneously adjusted fore and aft by using the rudder pedal adjustment switch. The wheel brakes are applied conventionally by toe action on the rudder pedals. Rudder pedal movement also controls nose wheel steering when the nose wheel steering system is engaged. Refer to WHEEL BRAKE SYSTEM and NOSE WHEEL STEERING SYSTEM, this Section.

Rudder Pedal Adjustment Switch

A two-position rudder pedal adjustment switch (figure 1-33), placarded "Rudder Pedals Adjust," is located on the right side of the ejection seat. The switch is used to unlock the rudder pedals, permitting independent fore and aft adjustment. The switch has an ADJ position and a spring-loaded neutral (off) position. Holding the switch in the ADJ position will actuate the rudder pedal locking solenoid to unlock the rudder pedals. With the rudder pedals unlocked, spring-loaded adjustment links push the rudder pedals toward the seat. Releasing the adjustment switch when the pedals are in the desired position deenergizes the locking solenoid, causing the pedals to lock in position. The switch receives power from the dc nonessential bus.

WARNING

To prevent the pedals from springing aft and striking the shins, both feet should be placed on the pedals before the rudder pedal adjustment switch is actuated.

SPEED BRAKES SYSTEM

The speed brakes (15, figure 1-1) are located above the tail cone. They can be used to slow the airplane at all speeds. The speed brakes are hydraulically operated and electrically controlled (figure 1-19). A relief valve in the speed brakes hydraulic system allows the speed brakes to retract, as necessary, to prevent structural damage under excessive aerodynamic loads. Secondary hydraulic system pressure is supplied through an electrically (dc) operated selector valve to actuate a hydraulic cylinder for each brake. A switch, located on the throttle, electrically operates the speed brakes selector valve to port hydraulic pressure to either the open or close side of the speed brakes actuation cylinders. The speed brakes are synchronized by hydraulic flow to give equal angular operation, and require approximately two seconds to open or close. The speed brakes also serve as compartment doors for the drag chute, and when the drag chute is deployed the brakes cannot be closed until the chute is jettisoned. The speed brakes are controlled by a switch located on the throttle and are opened automatically when the drag chute handle is pulled. A speed brakes emergency opening system is installed to be used in the event of secondary hydraulic failure, to facilitate drag chute deployment. The emergency system is energized by the emergency dc power package, and bypasses the speed brakes switch on the throttle. Pneumatic system air is directed to the speed brakes actuating cylinders when the drag chute handle is pulled out (about 1½ inches) then rotated clockwise 90 degrees and pulled again (about ½ inch). For

speed brakes and drag chute system (typical)

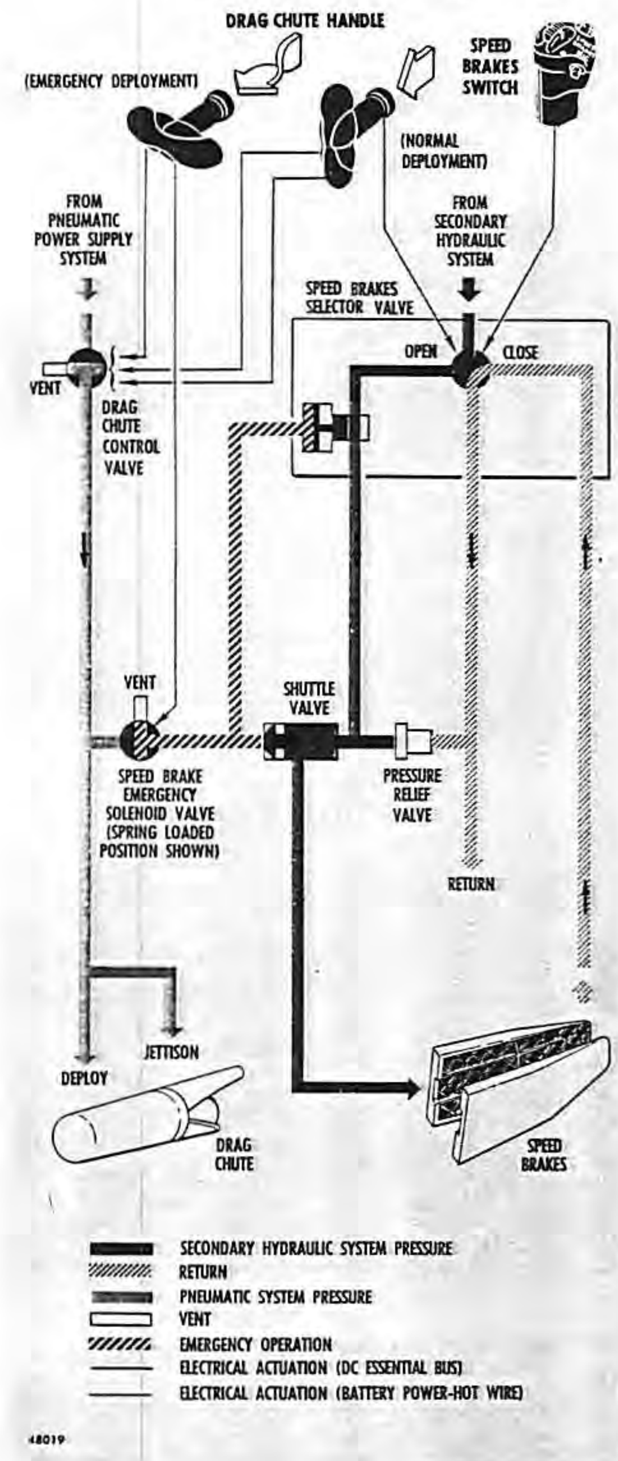


Figure 1-19

additional information on this system, refer to T.O. 1F-106A-2-7.

CAUTION

If normal speed brakes extension is not possible and drag chute deployment is desired, emergency deployment should be delayed until on the landing roll. Inflight use of this system will result in loss of the drag chute or excessive loss of airspeed. An emergency drag chute extension must be noted on Form 781 to insure bleeding of the speed brakes system.

SPEED BRAKES SWITCH

The three-position speed brakes switch (figure 1-6), located on the throttle grip, is used to control speed brakes operation except when drag chute is deployed. The switch is placarded "Speed Brakes" and has fixed positions of IN, OUT, and a center (off) position which controls the selector valve accordingly. On Ⓑ airplanes the switch in the aft cockpit is spring-loaded to the center (off) position. The neutral (off) position is indicated by a black alignment mark on the switch guide. When the switch is in the center position, the control valve is closed and the speed brakes are held in the selected position. Partial speed brake operation is possible by rapidly moving the switch from IN to OUT to (off). The speed brakes switch receives power from the dc essential bus.

WARNING

Return the speed brakes switch to neutral after opening or closing of the speed brakes. This is done to prevent fluid loss in the event of line or speed brake failure.

NOTE Ⓑ

On Ⓑ airplanes, if the switches are placed in opposing positions (one switch at the IN position and the other switch at the OUT position), the speed brakes will not move because electrical power will be supplied to both sides of the selector valve and the valve will remain in the closed position.

SPEED BRAKES GROUND SAFETY LOCKS

Ground maintenance safety locks (figure 1-23) may be installed on the speed brakes actuators when the brakes are extended, primarily during repacking of the drag chute, and must be removed before flight.

LANDING GEAR SYSTEM

The tricycle landing gear and wheel well doors (figure 1-20) are electrically controlled and sequenced, and hydraulically actuated. The main gear retracts inboard into the lower surface of the wing and fuselage, and the nose gear retracts forward into the fuselage. The wheel well doors remain open when the gear is extended and fair the gear flush with the airplane contour when the gear is retracted. The main landing gear fairings are mechanically tied to the strut assembly and are actuated with gear movement. The nose gear drag brace contains a combination up-and-down lock and is unlocked by initial travel of the nose gear actuating cylinder. The main gear is locked up by the wheel well doors, and the down-lock is unlocked by initial travel of the gear actuating cylinder. Safety switches preclude normal gear retraction while the airplane is on the ground. However, an override control bypasses the safety switches to permit emergency gear retraction while on the ground or while airborne. Normal gear extension, retraction, and emergency retraction is actuated by secondary hydraulic system pressure. In the event of electrical or secondary hydraulic system failure, the gear can be extended by pneumatic system pressure which is routed to the normal hydraulic actuating cylinders.

During normal operation, gear extension takes approximately six to eight seconds and gear retraction takes approximately four to six seconds. An electro-hydraulic servo steering unit is built into the nose gear assembly to provide nose wheel steering, and also serves as a conventional shimmy damper. The main wheels are equipped with pneumatically operated multiple-disc brakes. The main gear aft drag braces serve as pneumatic pressure reservoirs for the wheel brake system. For additional information on this system, refer to T.O. 1F-106A-2-8.

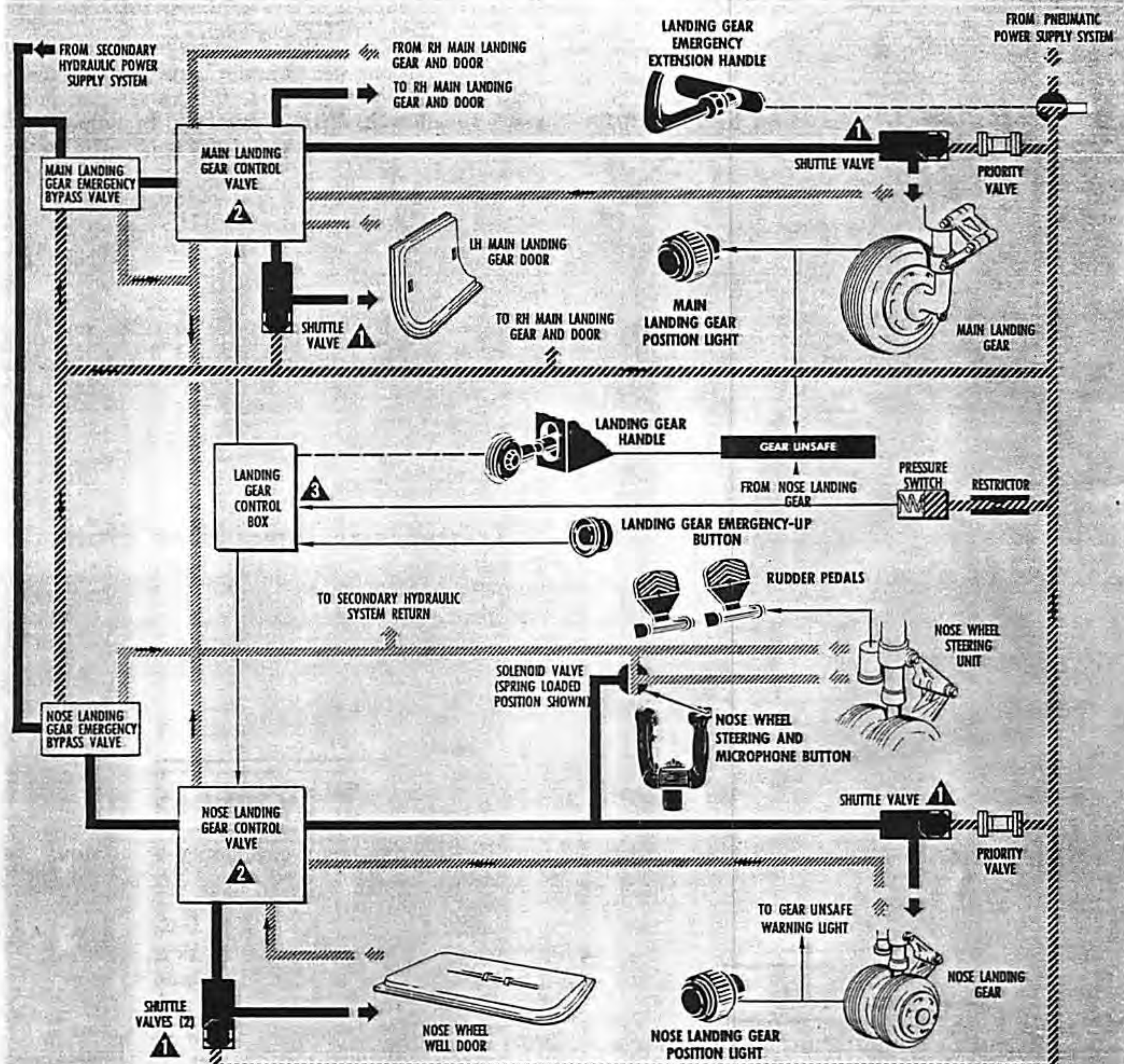
LANDING GEAR GROUND SAFETY LOCKS

Removable ground safety locks (figure 1-23) may be installed in the landing gear assemblies to prevent collapsing of the gear while the airplane is on the ground. The locks are equipped with warning streamers and must be removed before flight.

LANDING GEAR HANDLE

The wheel-shaped landing gear handle (figure 1-21), located on the landing gear control panel, electrically controls normal operation of the gear and wheel well door hydraulic selector valves.

landing gear and nose wheel steering system



NOTE

B DUAL CONTROLS NOT SHOWN.

1 VALVE SHOWN IN NORMAL POSITION. FOR EMERGENCY GEAR EXTENSION, PNEUMATIC PRESSURE MOVES SHUTTLE TO SHUT OFF HYDRAULIC SUPPLY AND ADMIT PNEUMATIC PRESSURE TO THE EXTEND SIDE OF GEAR AND GEAR DOOR ACTUATING CYLINDERS.

2 SOLENOID VALVES, SPRING LOADED TO ROUTE GEAR AND GEAR DOOR ACTUATION LINES TO SECONDARY HYDRAULIC SYSTEM RETURN.

3 WHEN THE LANDING GEAR EMERGENCY EXTENSION HANDLE IS PULLED, THE ELECTRICAL PORTION OF THE NORMAL LANDING GEAR CONTROL SYSTEM IS DEENERGIZED.

SECONDARY HYDRAULIC SYSTEM PRESSURE
VENT
AIR PRESSURE
SECONDARY HYDRAULIC SYSTEM RETURN
EMERGENCY OPERATION
ELECTRICAL ACTUATION
MECHANICAL ACTUATION

Figure 1-20

landing gear control panel (typical)

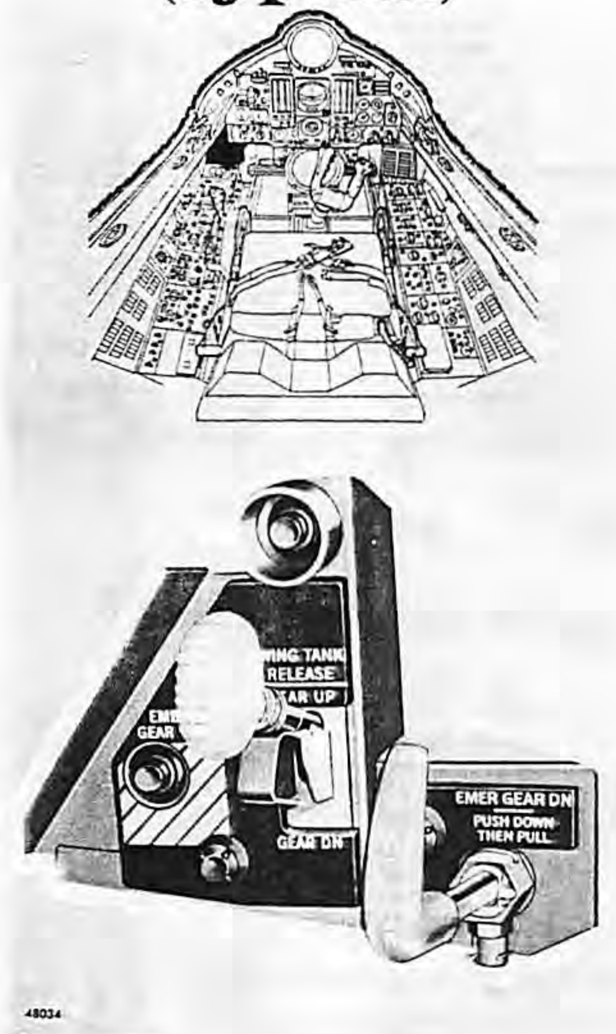


Figure 1-21

CAUTION

Because of the proximity of the landing gear handle to the external tank release button, use care to avoid inadvertent tank jettison when operating the landing gear handle.

On Ⓑ airplanes the forward and aft handles are mechanically interconnected and move in unison. When the airplane is airborne, moving the handle to UP position energizes the hydraulic selector valves to apply secondary hydraulic system pressure to retract the gear. When the main and nose gear are fully up, the door selector valves are positioned to close the doors. When the doors are closed and locked, the gear actuating system is automatically depressurized.

NOTE

When the weight of the airplane is on the gear, ground safety switches prevent gear retraction if the handle is inadvertently moved to UP position.

When the landing gear handle is moved to DOWN position, the hydraulic selector valves are energized to allow hydraulic pressure to unlock and open the doors, then extend the gear. Hydraulic pressure is maintained on the gear and doors when they are extended. The landing gear circuit is powered from the dc essential bus.

NOTE

A trigger on the landing gear handle locks the handle in the UP position and must be pulled prior to placing the handle in the DOWN position. The trigger does not have to be pulled to place the handle in the UP position. On Ⓑ airplanes the trigger is on the forward landing gear handle only.

LANDING GEAR EMERGENCY-UP BUTTON

The ring-guarded landing gear emergency-up button (figure 1-21), located on the landing gear control panel, when depressed momentarily causes the landing gear control circuit to bypass the ground safety switches, and will retract the gear if the normal landing gear handle is in the UP position when the airplane is on the ground.

WARNING

The landing gear emergency-up button should not be used to retract the landing gear during landing emergencies (forced landing, engine failure on takeoff, etc.) or aborted takeoffs when insufficient runway remains to stop. The landing gear should remain extended during these emergencies to absorb impact shock, minimizing airplane damage and personal injury.

The button is placarded "Emer Gear Up." In the event of ground-safety switch malfunction, the emergency-up button may be used when the landing gear handle is UP to retract the gear when airborne. Once energized, the emergency retract system cannot be deenergized except by placing the handle landing gear handle to DOWN position. The emergency-up button receives power from the dc essential bus.

wheel brake system

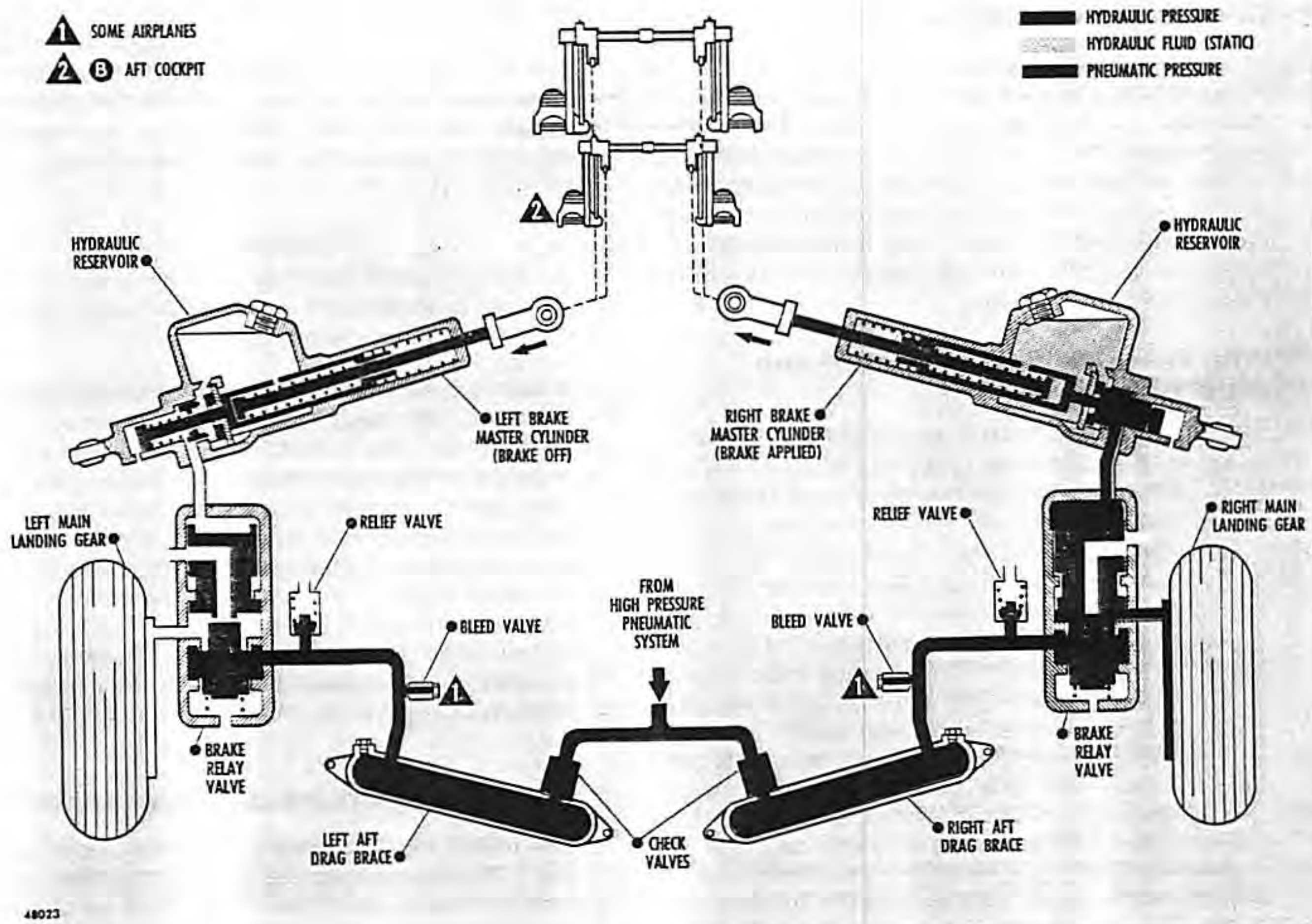


Figure 1-22

LANDING GEAR EMERGENCY EXTENSION HANDLE

The landing gear emergency extension handle (figure 1-21), suspended below the left side of the instrument panel, is used to pneumatically extend the landing gear in the event of failure of the normal landing gear extension system. The handle is placarded "Emer Ldg. Gear" on Ⓐ airplanes and "Emer Gear Dn" on Ⓑ airplanes. Pulling the handle out fully (approximately four inches) mechanically opens a valve allowing pneumatic system pressure to enter the wheel well door and landing gear actuating cylinders, which will open the doors and extend the gear. Simultaneously, pneumatic system pressure actuates the bypass valves. These valves block hydraulic pressure to the control valves, and hydraulic fluid downstream of the bypass valves is vented to the return line. Spring-loaded detents are provided on the shaft portions of the handle to lock the handle in the full retracted or extended position. To make an emergency landing gear extension, the handle must first be pressed down to release the retract detent and then pulled fully out until the extend detent is felt. The emergency extension handle will extend the landing gear regardless of the position of the normal landing gear handle because the electrical portion of the normal control system is deenergized when the extension handle is pulled.

NOTE

The normal landing gear handle should be in the DOWN position prior to landing. If the handle is in the UP position, the landing gear warning light will illuminate and the audio warning will sound.

Emergency extension of the landing gear takes approximately 4-6 seconds. There are no provisions for retracting the gear pneumatically. An emergency landing gear extension should be noted on Form 781, to insure bleeding of the secondary hydraulic system.

LANDING GEAR POSITION LIGHTS

The landing gear position lights (5, figure 1-8, and 3, figure 1-9) are provided to indicate landing gear position and prelanding warning. Three green lights (one for each corresponding gear) are located on the left side of the instrument panel and illuminate when the corresponding landing gear is fully down and locked. The green position lights are the push-to-test type. Power is supplied from the dc essential bus.

LANDING GEAR WARNING LIGHT AND AUDIBLE WARNING

A red light on the instrument panel (4, figure 1-8, and 2, figure 1-9) illuminates and displays "GEAR UNSAFE" when the landing gear or gear doors are not in the position selected by the landing gear handle. This light will also illuminate if the landing gear is not extended during the landing approach (when the airplane is below 10,000 feet and below 220 (± 10) KCAS with the throttle aft of FULL MIL POWER position). The warning light has a "push-to-dim" feature and is tested together with the lights in the master warning system. The light will illuminate when the master warning light test switch is depressed. An audible warning signal through the radio receiver is provided to give the second means of a gear-unsafe condition. The audible warning signal operates simultaneously with the red warning light during the landing approach conditions outlined above. The audible warning can be shut off by means of a pushbutton (21, figure 1-10), located on the forward left subconsole. This button is placarded "Audio Warn Cutoff" and when depressed will cut out the audible signal but will not affect operation of the warning light. After the button has been depressed, the audible warning system is not reactivated until the throttle is advanced to the FULL MIL POWER position, 220 (± 10) KCAS is exceeded, or until 10,000 feet is exceeded. There is no means for checking the audible warning signal from the cockpit. On some Ⓐ* and Ⓑ** airplanes, the audible warning signal volume cannot be controlled from the cockpit. On other Ⓐ† and Ⓑ‡ airplanes the volume can be controlled through the command radio volume control knob. Power is supplied from the dc essential bus.

*AF 57-246 thru 57-2499.
**AF 57-2516 thru -2522 & -2524 thru -2528.
†AF 56-453, -454, -456 thru 57-245, -2500 & on.
‡AF 57-2508 thru -2515, -2523, -2529 & on.

NOSE WHEEL STEERING SYSTEM

The nose wheel steering system is provided for directional control during taxiing and for portions of the takeoff and landing roll, as desired. The system is electrically engaged, controlled by the rudder pedals, and powered by secondary hydraulic system pressure. Steering is engaged by momentarily depressing a nose wheel steering button on the control stick grip. Engagement will occur regardless of the relative positions of the rudder pedals and the nose wheel. Nose wheel steering is capable of turning the nose wheel through a range of 146° (73° left and 73° right of center).

NOTE

> Nose wheel steering is inoperative when the landing gear is extended by the emergency extension system.

A mechanically operated switch and centering cam automatically depressurizes the steering unit and centers the nose wheel as the gear retracts. The nose wheel steering system is irreversible which prevents forces applied to the nose wheel from being transmitted to the rudder pedals. When the system is not engaged (scissors linkage connected) the nose wheel is free to swivel through a range of approximately 150° (75° left and 75° right of center). The steering unit also serves as a conventional shimmy damper. For additional information on this system, refer to T.O. 1F-106A-2-8.

NOSE WHEEL STEERING UNIT GROUND LOCK PIN

The nose wheel steering unit ground lock pin (figure 1-23) may be installed in the steering unit to facilitate jacking of the nose wheel or to provide stability of the nose unit when mooring the airplane. This lock pin must be removed before flight.

NOSE GEAR TORQUE LINK DISCONNECT

The torque link or scissors assembly on the nose wheel strut is equipped with a quick-release pin, which allows the torque links to be disconnected for towing operations. When the torque links are disconnected, the nose wheel can swivel freely during towing operations. The torque links are disconnected by pulling out the spring-loaded release pin and pulling the links apart. If the torque links are not connected prior to flight, nose wheel steering and shimmy damping will not be available.

NOSE WHEEL STEERING AND MICROPHONE BUTTON

The nose wheel steering and microphone button (figure 1-18), is located on the right-hand control stick grip and is marked "NWS." Momentarily depressing the button engages the steering system.

Whenever the system is engaged, the nose wheels will align to the existing position of the rudder pedals. Because of holding relays in the system, it is not necessary to hold the button depressed to maintain steering; however, while the button is held depressed, nose wheel steering will be engaged. To disengage the system, the button should be depressed again. The system will also disengage whenever the nose wheel strut is fully extended or reaches the maximum steering limit of 73°. In flight the button keys the UHF transmitter. The nose wheel steering button receives power from the dc essential bus.

WHEEL BRAKE SYSTEM

The airplane is equipped with pneumatically operated multiple-disc type brakes (figure 1-22) which are installed on the inboard side of the main landing gear wheels. The left and right wheel brakes are individually operated by depressing the upper portion of the respective rudder pedal. On Ⓑ airplanes the brake actuating function of the forward and aft rudder pedals is mechanically interconnected. Either the forward or aft rudder pedals may be used to apply the brakes; however, braking action on both the forward and aft pedals must be released before the brakes will release. This action applies independent hydraulic pressure (closed system) to control position of spring-loaded brake relay valves which, in turn, meter pneumatic pressure to actuate the brakes. The aft main landing gear drag braces serve as pneumatic pressure reservoirs for the brake system and are connected to the main portion of the pneumatic system through check valves which maintain braking pressure if pneumatic system pressure is depleted.

NOTE

There is no method of checking brake system pneumatic pressure in flight. As the brakes are hydraulically controlled and pneumatically actuated, the feel of pressure in the rudder pedals is not a definite indication that pneumatic pressure is available to the brakes.

Relief valves are installed to protect the brake system against excessive pressure. Emergency, anti-skid, and parking brake systems are not provided. For additional information on this system, refer to T.O. 1F-106A-2-8.

DRAG CHUTE SYSTEM

A drag chute system (figure 1-19) is provided to reduce landing roll distance and is to be used after touchdown. The ringslot type parachute, packed in a deployment bag, is stowed in a compartment below the rudder. The speed brakes serve as compartment doors for the drag chute. The drag chute is deployed and jettisoned by a drag chute control handle which electrically controls pneumatic pressure to operate the chute deployment mechanism. If the handle is pulled and the speed brakes are closed, either secondary hydraulic system pressure or pneumatic system pressure will open the brakes for drag chute deployment. The chute pack is secured to the airplane to prevent deployment by the slipstream in flight when the speed brakes are opened. Should this feature fail and the chute accidentally deploy in flight (without pulling the drag chute control handle) the deployment mechanism will release the entire chute assembly from the airplane. The chute mechanism incorporates a shear pin to prevent structural damage if the chute is deployed at speeds in excess of its structural limit. For additional information on this system, refer to T.O. 1F-106A-2-7.

DRAG CHUTE HANDLE

The drag chute handle (1, figure 1-8 and 1, figure 1-9), located on the left side of the instrument panel, is formed to resemble a parachute and is used to deploy and jettison the drag chute. Pulling the handle fully straight out (approximately 1½ inches) actuates a switch that electrically controls the speed brake selector valve which applies secondary hydraulic system pressure to open the speed brakes. When pulled fully out, the drag chute handle also supplies electrical power direct from the dc essential bus to operate a selector valve which directs high-pressure pneumatic air pressure to the chute deployment mechanism. When pneumatic pressure is supplied to the chute deployment mechanism, the chute risers are secured and the ripcord pin is pulled to deploy the drag chute. Mechanical sequencing prevents release of the chute until the speed brakes have opened sufficiently to clear the chute as it deploys. Pushing the drag chute handle fully in shuts off electric power to selector valve which then releases pneumatic pressure to the deployment mechanism which jettisons the chute. On Ⓑ airplanes if both handles are pulled fully out, pushing either handle fully in will jettison the drag chute. In the event of secondary hydraulic system failure, pulling the drag chute handle fully out, rotating 90° to the right, then out again (approximately ½ inch) supplies electrical power direct from the airplane battery to power a solenoid-operated pneumatic selector valve which supplies high-pressure pneumatic air to the speed brake cylinders.

ground safety locks

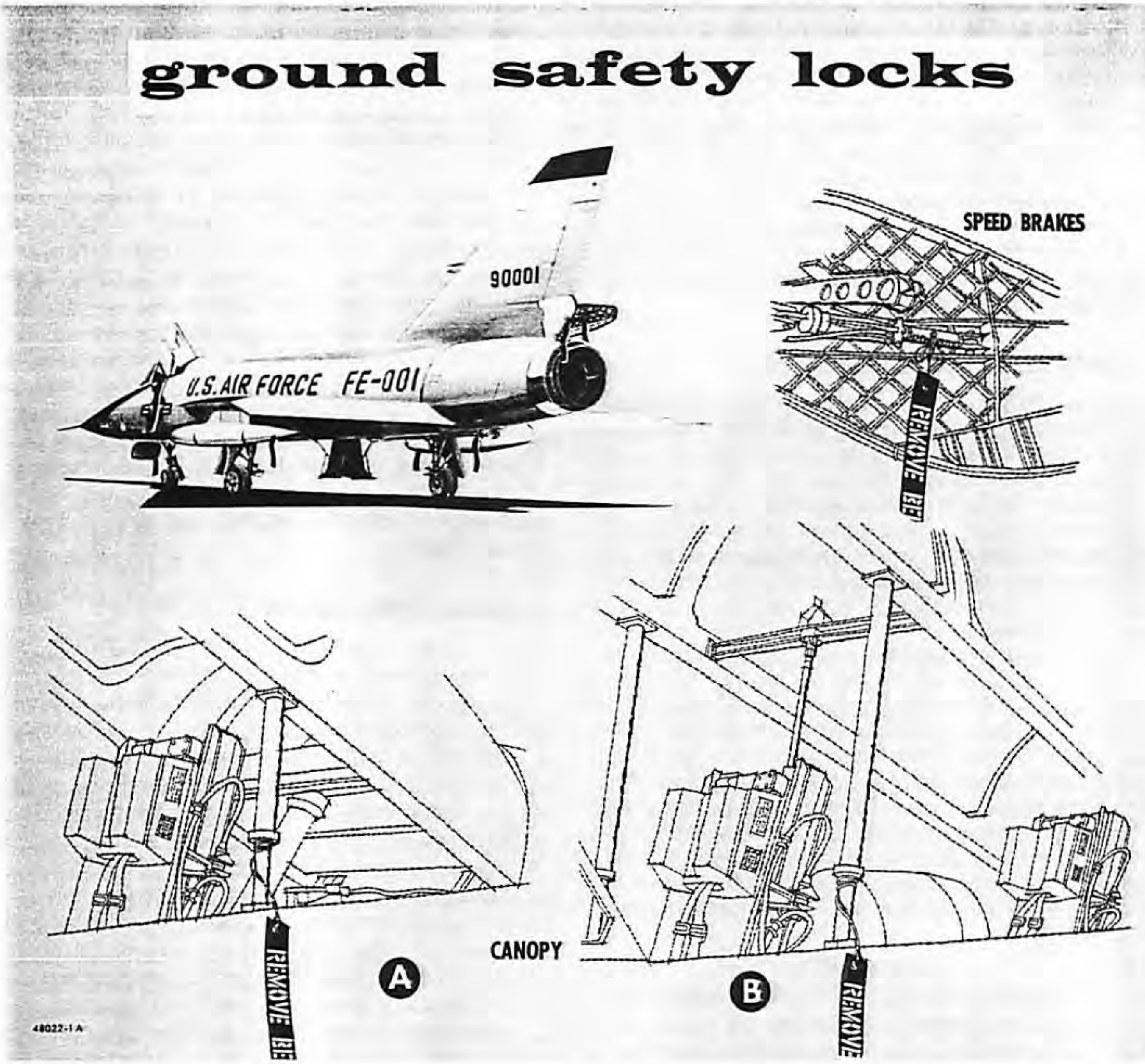


Figure 1-23

CAUTION

When the drag chute is jettisoned after emergency deployment, the drag chute handle must be rotated 90° counterclockwise to the vertical position. The rotation will disconnect power to the speed brakes emergency solenoid valve and prevent a continued load on the battery.

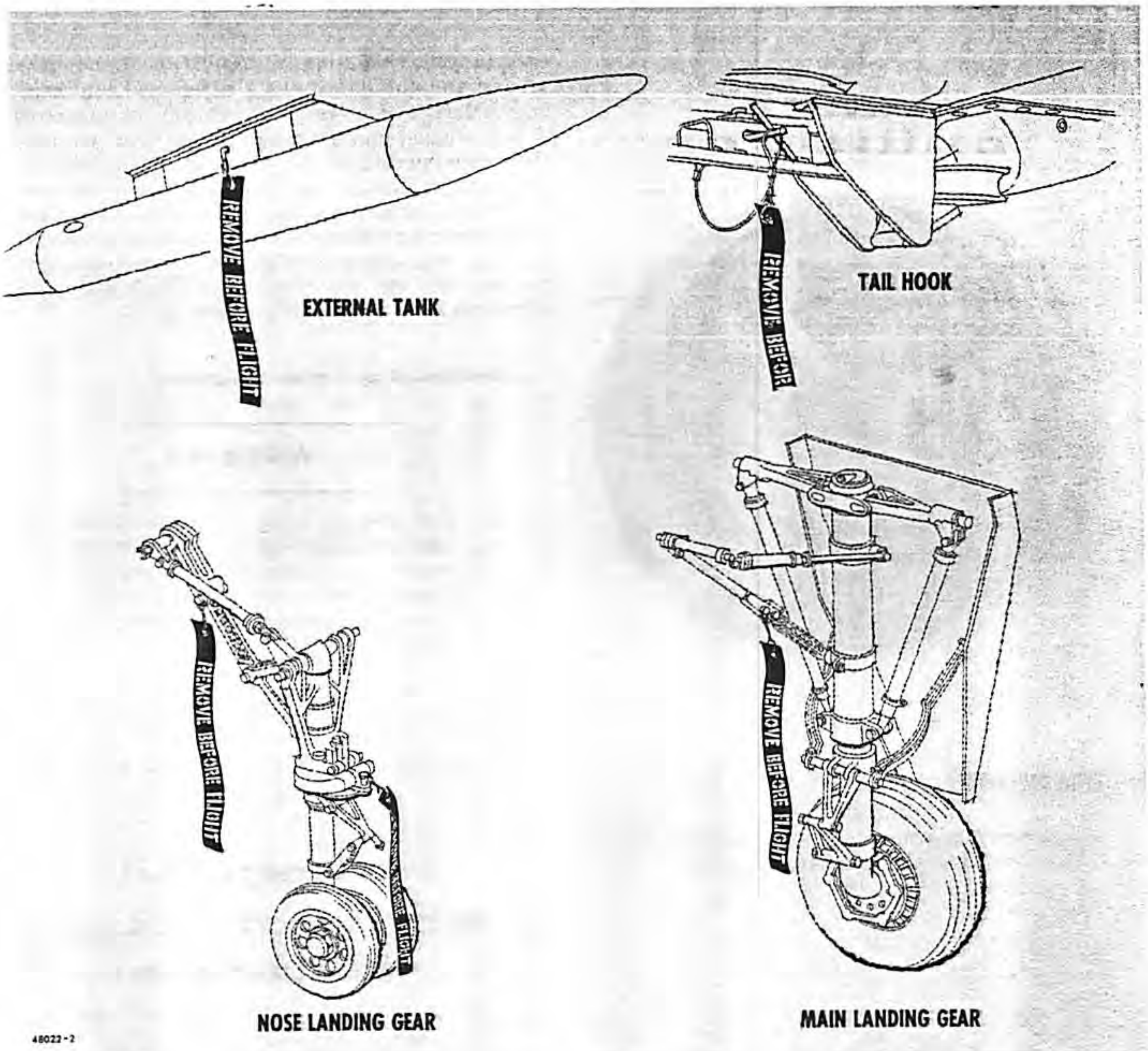
Speed brakes cannot close until the chute is jettisoned. If the speed brakes switch is IN when the chute is jettisoned the speed brakes will close, otherwise they will remain in the extended position.

CAUTION

To prevent loss of drag chute or to prevent chute from slowing airplane to excessively slow speeds, do not deploy drag chute in flight.

NOTE

After emergency drag chute deployment, a notation must be made on Form 781 so that secondary hydraulic system bleeding may be accomplished.



PITOT-STATIC SYSTEM

The pitot-static system supplies pitot pressure to the airspeed indicator, the air data computer system, fuel transfer system on Ⓐ airplanes, the engine pressure ratio gage, and the velocity gravity height recorder on some airplanes*. Static pressure is applied to the airspeed indicator, the vertical velocity indicator, the air data computer system and the altimeter. The pitot-static tube is mounted on the end of the nose boom. Ram air pressure ("q" pressure) is supplied from two tubes, located on the leading edge of the vertical fin, to control the elevator and rudder artificial feel systems.

**AF 56-453, -454, -456 thru 57-245, 59-087 & on.*

AIR DATA COMPUTER SYSTEM

The air data computer is an electronic analog computer which converts airstream data into electrical switching, variable capacitance values, variable resistance values, and synchro-electrical signals. Input to the air data computer consists of total pressure, static pressure, stagnation temperature, and angle of attack. Computer output is fed to the pitch and yaw damper systems, the Mach indicator, the variable ramp control system, the landing gear warning system, fuel transfer system on Ⓐ airplanes, and the MA-1 aircraft and weapon control system. The air data computer receives power from the ac essential bus.

mach indicator

48031

Figure 1-24

INSTRUMENTS

NOTE

Airspeeds in this manual are shown in knots calibrated airspeed (KCAS).

CONVENTIONAL INSTRUMENT DISPLAY

Mach Indicator

The Mach indicator (figure 1-24) is located on the instrument panel. A pointer indicates airplane true Mach number on a circular scale having a range of 0.4 to 2.2 Mach. A triangular command Mach marker moves around the outside of the scale to indicate command Mach as received from the digital computer. A striped maximum allowable Mach pointer presents maximum safe Mach received from the air data computer. A Mach warning flag marked "OFF" appears in the upper left quadrant of the indicator to warn the pilot of power supply failure. The flag will automatically reset when power is restored to the indicator. The Mach indicator receives power from the ac essential bus.

Airspeed-Angle of Attack Indicator

WARNING

The movable outer ring portion of the angle of attack indicator is deactivated and must not be used for reference in flight; however, the correct airspeed indication can still be read from the pointer and fixed face dial.

The airspeed-angle of attack indicator (figure 1-25), is located on the instrument panel. A pointer on the face of the instrument indicates airspeed on the fixed airspeed scale which is calibrated in

airspeed-angle of attack indicator

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Figure 1-25

Figure 1-26 Deleted

increments of 10 knots from 80 to 800 knots. Displayed in a cutout on the face of the instrument is a rotating airspeed scale marked at 10-knot intervals from 0 to 100 knots and calibrated in increments of 2 knots. The airspeed indicator phase of the instrument operates directly from pitot-static pressure.

Servoed Altimeter

The AAU-19A altimeter (figure 1-26A) located on the instrument panel is driven by the air data computer. It is a servo/pneumatic type consisting of a precision pressure mechanism combined with a servo repeater which is controlled by the air data computer. Altitude indications are shown as a counter-drum-pointer display. The 10,000 feet counter, 1,000 feet counter, and 100 feet counter drums provide for a direct digital read out. The pointer repeats the 100 feet indications of the drum and gives a quick indication of the rate of altitude changes. The 10,000 feet counter drum is blanked out by barber pole markings until the 10,000 feet level is reached. In servo mode of operation, the altimeter displays altitude correct for installation error as transmitted from the computer. Electrical operation is the primary mode for this altimeter. With the air data computer operating, servoed mode is achieved by moving the Reset-Stby lever, located on the lower right corner of the altimeter case, to the RESET position. In the event of AC power interruption or an out-of-tolerance condition, the altimeter reverts to pneumatic operation and the STBY flag will appear on the face of the instrument. Altitude indications are then accurate excluding installation error. Altitude corrections can be made by consulting the altimeter correction chart. If no problems exist, the altimeter can be reset to servo-mode by actuating the Reset-Stby lever to RESET. The STBY flag will disappear unless a fault remains. Pneumatic operation can be selected at any time by holding the Reset-Stby lever in the STBY position until the STBY flag appears (normally 1 to 3 seconds). Servo and STBY altimeter readings will not normally be identical. During STBY operation, an internal vibrator powered by the 28V DC essential bus will operate continuously enabling the instrument to provide a smooth altitude change display. If the vibrator fails, the instrument will continue to operate pneumatically but a less smooth display will be evident. A Kollsman dial (range from 28.1 to 31.0 in hg) for setting the altimeter is located on the lower right side of the instrument face. Setting knob is on the lower left case. Field evaluation checks should be made in both servo and STBY modes using the standard ± 75 feet as the maximum allowable error in either mode. During this check, the maximum allowable difference between modes is 75 feet.

Figure 1-26A.

Barometer Setting Control

The barometer setting control located on the instrument panel (figure 1-26B) applies a signal proportional to the barometer pressure at sea level to the digital computer subsystem. The barometer setting compensates for variations of barometric pressure at the local airport from true pressure altitude.

Attitude Indicator

The airplane is equipped with an MM-3 or ARU-13/A attitude indicator (figure 1-27) which displays information received from an MD-1 remote gyro control assembly. The attitude indicator displays precise attitude information through 360° bank and ± 82° of pitch.

NOTE

> At approximately 82° of pitch, the attitude sphere will rotate 180°. This momentary rotation is known as controlled precession and should not be confused with gyro tumbling. After the rotation is complete, the pitch and roll indications will be accurate except for a small amount of precession. After straight and level flight is resumed, this precession will be corrected at the rate of 0.8 to 1.8° per minute.

baro. set control

Figure 1-26B.

attitude indicator

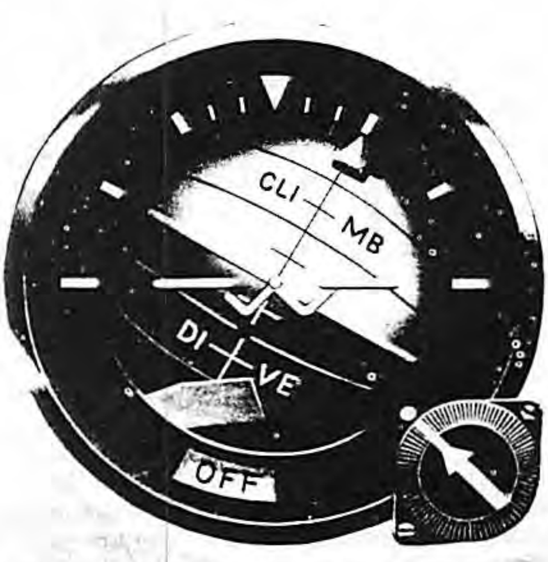

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Figure 1-27

The attitude sphere is divided by a white horizon bar which provides a sensitive pitch reference near a level flight attitude. The top or sky half of the sphere is colored light gray and the lower or ground half is black. A pitch reference scale indicates pitch angle through 90° climb or dive. These pitch lines are graduated at 5° intervals with numerical indications at 30° and 60° of pitch. The words CLIMB and DIVE are depicted on the sphere at 15°. Bank attitude is indicated by the position of the bank pointer in relation to the fixed bank scale which is marked at 0, 10, 20, 30, 60, and 90°. An attitude warning symbol placarded OFF will be visible for one minute (± 10 seconds) after power is applied to the instrument or any time the instrument is not receiving proper electrical power. In either case, the instrument is unreliable until the warning symbol is not visible.

WARNING

The attitude warning symbol will not be visible with a slight electrical power reduction or failure of other components within the system. This can result in erroneous or complete loss of pitch and bank presentations without the warning symbol appearing. Therefore, other flight instruments, the artificial horizon on the radar scope, and the approach horizon attitude indicator should be cross-checked to insure accuracy of the attitude indicator.

The pitch trim knob electrically positions the attitude sphere to provide desired pitch presentations relative to the fixed miniature airplane. Turn, acceleration, and deceleration errors have been virtually eliminated in the attitude indicator. Acceleration error during takeoff will be the most noticeable error and will appear as a climb indication error of approximately 1 1/2° just prior to breaking ground. The exact amount of error will depend upon the duration of acceleration. The system is powered by the ac essential bus. The system is inoperative on ground power unless the MA-1 power switch is in "WARM" or higher.

Vertical Velocity Indicator

A vertical velocity indicator (33, figure 1-8), located on the instrument panel, shows the rate of change of altitude in feet per minute. Changes in pressure due to changes in altitude are sensed by the pitot-static tube and transmitted to the indicator which is capable of indicating vertical speeds from 0 to plus or minus 6000 fpm. An overpressure diaphragm and valve prevent excessive rate of climb or descent from damaging the instrument.

Accelerometer

The accelerometer is located in the apex of the windscreen. The instrument is graduated in increments of 0.5 g's and indicates flight loads from +10 to −5 g. One pointer continuously indicates the g-loading on the airplane. Two other ratchet-mounted pointers indicate the maximum positive and negative g's indicated by the continuous reading pointer. Depressing the knob at the lower left of the accelerometer releases these maximum reading pointers which, when released, will indicate the current operating position in flight or will point to +1 g on the ground. The accelerometer is a self-contained instrument and operates independently of all other airplane instruments and systems.

Turn-and-Slip Indicator

The turn-and-slip indicator (6, figure 1-8) is located on the instrument panel. The indicator is rated for a four-minute turn (1½° per second). The indicator is powered by the dc essential bus.

Course Indicator

NOTE

On airplanes with the conventional instrument display, the course indicator and the approach horizon instruments comprise the airplane's flight director system.

The course indicator (figure 1-27A) presents a plan view of the airplane's position derived from the localizer or TACAN receiver and magnetic compass. The instrument provides heading course deviation, and "TO-FROM" information. The indicator consists of compass card, a course arrow, course deviation indicator, "TO-FROM" indicator, a heading marker, lubber line reference arrow, heading selector knob, and a course selector knob. The compass card is connected to the magnetic compass repeater and can be adjusted by the set switch on the magnetic compass control panel. The course arrow may be set to the desired TACAN course or ILS localizer course by use of the course selector knob. The course arrow displays angular difference between the selected course and the airplane heading. The course deviation indicator, which is the center section of the course arrow, represents the selected course and displays lateral deviation from course. The heading selector knob on the lower left side of the instrument turns the heading marker to the magnetic heading desired. A miniature airplane and lubber line are etched on the instrument's glass to indicate airplane heading. The aircraft symbol is comparable to a shadow of the airplane on the ground. When the aircraft symbol is pointed to intercept the course deviation indicator, the airplane is approaching the selected course. A "TO-FROM" TACAN indication is provided by a large arrow which appears beneath the course deviation indicator on the appropriate side of center. The instrument receives power from the ac and dc essential buses and from the MA-1 electrical power supply system. The compass card will be inoperative when the airplane is receiving ground power unless the MA-1 power switch is in "WARM" or higher.

Approach Horizon

NOTE

On airplanes with the conventional instrument display the approach horizon and the course indicator instruments comprise the airplane's flight director system.

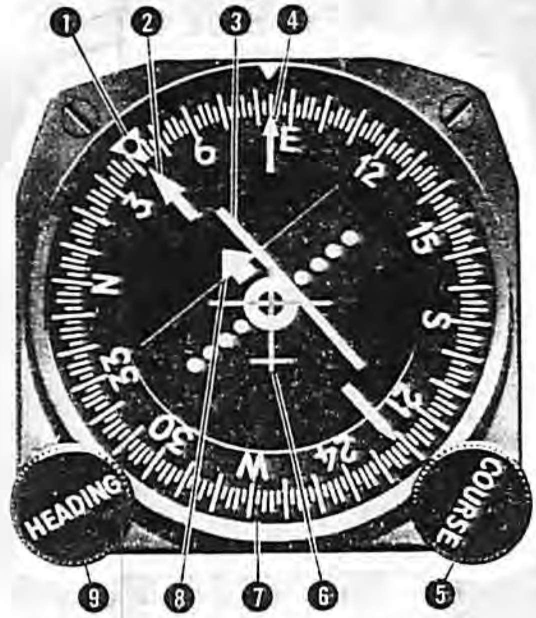
The approach horizon (figure 1-27B) is a forward view instrument indicating attitude and position, with glide position and computed steering information superimposed on a pitch and roll reference. The instrument consists of a bank steering bar, glide-slope indicator, bank pointer and horizon bar, pitch steering bar, pitch trim knob, and a function selector knob. Steering information for an ILS approach is given by the bank steering bar. The function selector knob on the lower right side of the instrument selects the desired function. In HDG position, the bank steering bar is fed heading change and bank information. In ILS position, the bank steering bar is fed heading error, localizer deviation and bank information. The glide-slope indicator on the left side of the instrument shows displacement of the airplane above or below the glide-slope. The bank pointer and horizon bar operate together. The bank pointer gives amount of right or left bank and the horizon bar indicates lateral attitude of the airplane. The pitch steering bar moves in a vertical plane above and below the center of the instrument to show nose-up or nose-down attitude. The pitch trim knob at left centers the bar at desired flight attitude in HDG function. In ILS function, the pitch bar indicates desired pitch correction to maintain or reintercept the glide slope. In this mode, the pitch trim knob has no effect on the pitch steering bar. Warning flags are located in the top center of the instrument. When either the GS (glide-slope) or LOC (localizer) warning flag is visible, the corresponding receiver is not responding or receiving correct signals.

NOTE

This instrument can be used as a backup attitude indicator.

The instrument receives power from the ac and dc essential buses and from the MA-1 electrical supply system.

course indicator



1. Heading Marker
2. Course Arrow
3. Course Deviation Indicator
4. Lubber Line
5. Course Selector Knob
6. Miniature Airplane
7. Compass Card
8. To-From Arrow
9. Heading Selector Knob

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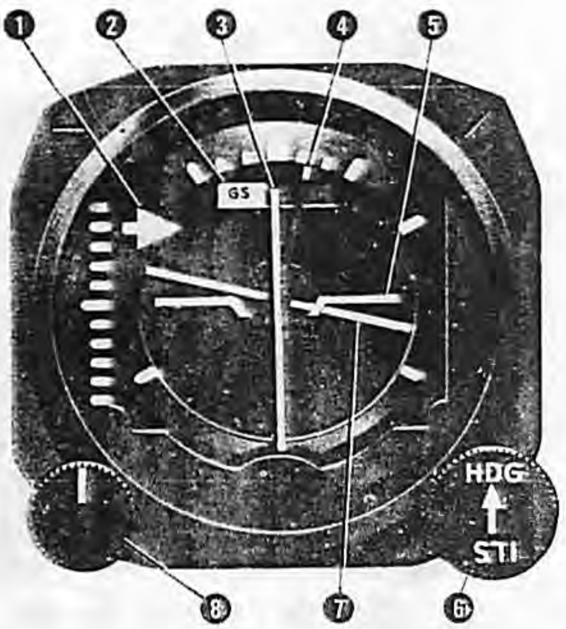
Figure 1-27A

approach
horizon



1. Glide Slope Indicator
2. Warning Flags
3. Bank Steering Bar
4. Bank Pointer
5. Pitch Steering Bar
6. Function Selector Knob
7. Horizon Bar
8. Pitch Trim Knob

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Figure 1-27B

INTEGRATED FLIGHT INSTRUMENT DISPLAY

NOTE

Refer to Confidential Supplement, T.O. 1F-106A-1A, for additional indoctrination information regarding this system.

The integrated flight instrument display is designed to display flight and navigation information from one interrelated source rather than from individual, unrelated instruments. Information is displayed on four instruments: airspeed-Mach indicator (AMI), attitude director indicator (ADI), altitude-vertical velocity indicator (AVVI), and horizontal situation indicator (HSI). Arrangement of these instruments on the panel is in the form of a "T."(See figure 1-27C)The "T" arrangement establishes two lines of reference—one horizontal and one vertical. Across the horizontal reference line are the AMI, ADI, and the AVVI. Display elements of these instruments (angle of attack, acceleration, Mach airspeed, pitch, rate of climb, altitude) are controlled by fore and aft movement of the control stick and by thrust. Aft movement of the stick establishes a related display showing nose-up attitude, increasing altitude and load factor, and decreasing speed across one line of reference. These indications are reversed with forward stick movement. Down the vertical reference line are the ADI and HSI. Display elements of these instruments (heading, bank, turn rate, navigational and tactical information in the horizontal plane) are controlled by side-to-side movement of the control stick. The integrated instrument display is referred to as a "fly-to" display The "fly-to" concept means that necessary corrections (as presented by command indicators and displacement needles and pointers) are made by flying toward an indication. For example, if the command altitude is higher than existing altitude, the command altitude marker will be positioned above the existing altitude on both the sensitive altitude scale and the gross altitude scale.

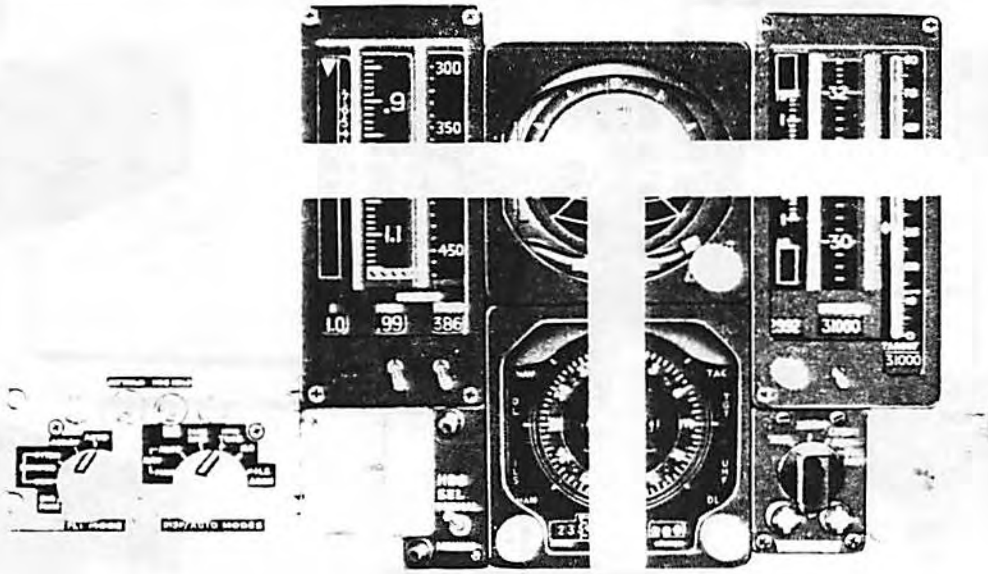
COMPONENTS OF THE INTEGRATED FLIGHT INSTRUMENT DISPLAY

In addition to the instruments discussed in the previous paragraph, the integrated flight instrument system consists of sensing units, coupler, and a flight director system. The air data computer system also has inputs. The general flow of signals between these components is shown in figure 1-27D. The sensing units consist of pitot-static system, accelerometer, temperature probe, angle of attack sensor, and gyro reference unit. The air data computer system is composed of an air data computer, air data converter, and a compensator. Changes of pressure, temperature, angle of attack, and acceleration are converted to electrical signals to supply the following inputs to the instrument: calibrated airspeed, maximum safe speed, Mach (true), maximum safe Mach, load factor, altitude and vertical speed. The coupler unit converts information from the airplane navigation equipment to signals for the instruments. Normal inputs to the coupler are from TACAN, data link and ILS. The flight director unit combines inputs from the heading and course settings on the HSI and from the navigation equipment to supply signals to the steering bars and glide-slope indicator on the ADI.

Airspeed-Mach Indicator (AMI)

An airspeed-Mach indicator (figure 1-27E) located on the instrument panel, gives a vertical presentation of speed and load factor information. The

integrated flight instrument display concept



1 INSTRUMENT DISPLAYS ACROSS THE HORIZONTAL REFERENCE LINE ARE CONTROLLED BY FORE AND AFT MOVEMENT OF THE CONTROL STICK.

2 INSTRUMENT DISPLAYS OF THE VERTICAL REFERENCE LINE ARE CONTROLLED BY LATERAL MOVEMENT OF THE CONTROL STICK.

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Figure 1-27C

instrument contains four display columns. From left to right, the columns are: the angle of attack indicator, the accelerometer, the Mach indicator, and the airspeed indicator.

NOTE

Mach and airspeed information are electronically converted to true Mach and calibrated airspeed (CAS) before being transmitted to the display scales.

In addition, the AMI has an acceleration readout window, a maximum allowable Mach marker, a command Mach marker and readout window with a slewing switch, and a command airspeed marker and readout window with a slewing switch. In the event of ac essential power failure to the drive motor of the AMI, an "OFF" flag will appear in the center of the airspeed scale.

data flow of integrated flight instrument system

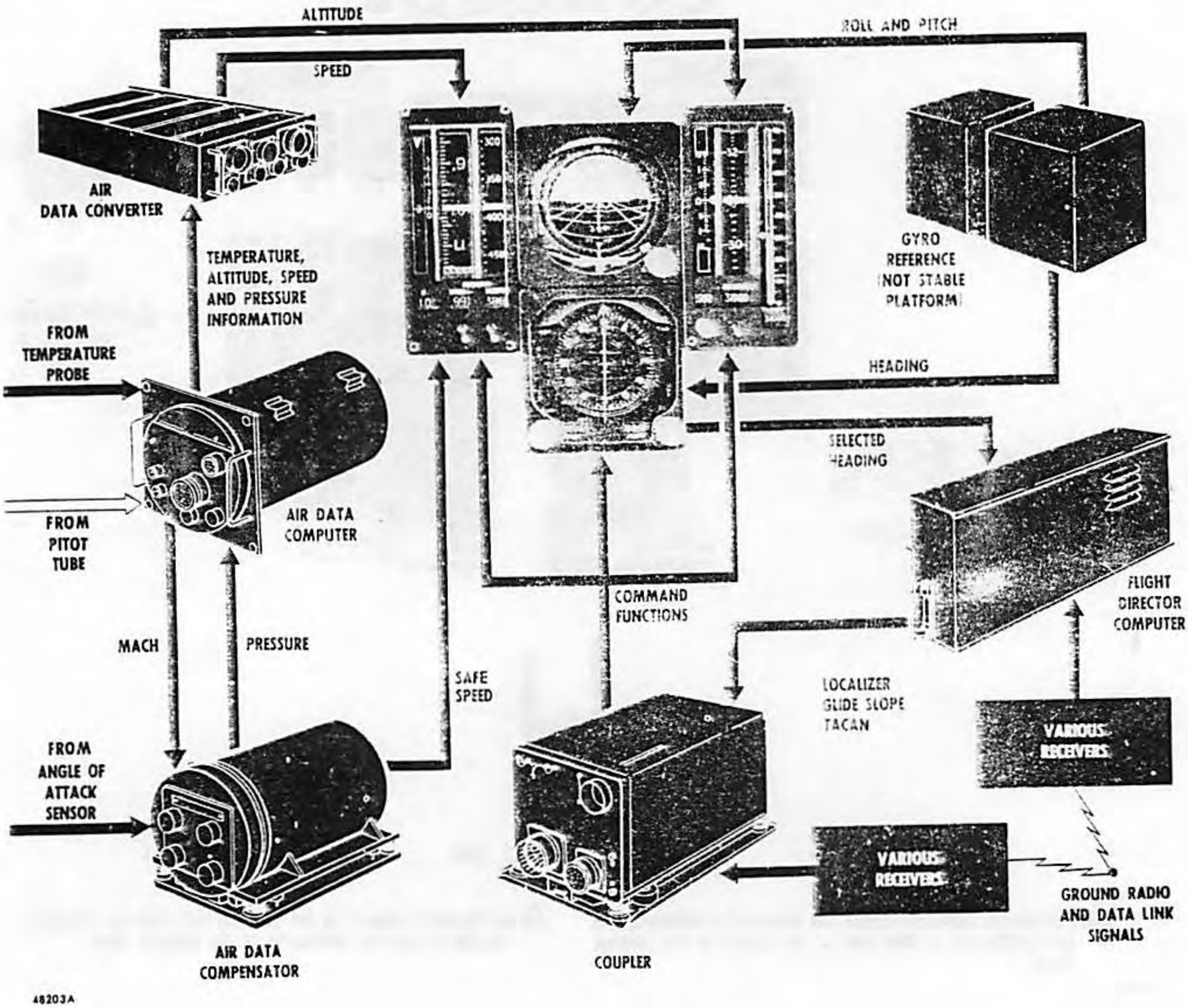


Figure 1-27D

NOTE

Although the "OFF" flag appears only on the airspeed scale, it indicates that all the functions of the AMI are inoperative.

The electrical driving power to the AMI is from the ac essential bus. The command Mach input to the indicator, however, is dependent upon MA-1 electrical power.

Angle of Attack indicator. The angle of attack indicator, located on the AMI (figure 1-27E), provides minimum safe speed and positive g information. The angle of attack scale contains a triangular-shaped final approach symbol marked "Final," and a minimum safe speed symbol marked "Min Safe Spd." Final approach airspeed is based on a given constant angle of attack. This approach angle of attack is a compromise between maneuverability and slowest possible flying speed. Regardless of airplane weight, the best angle of

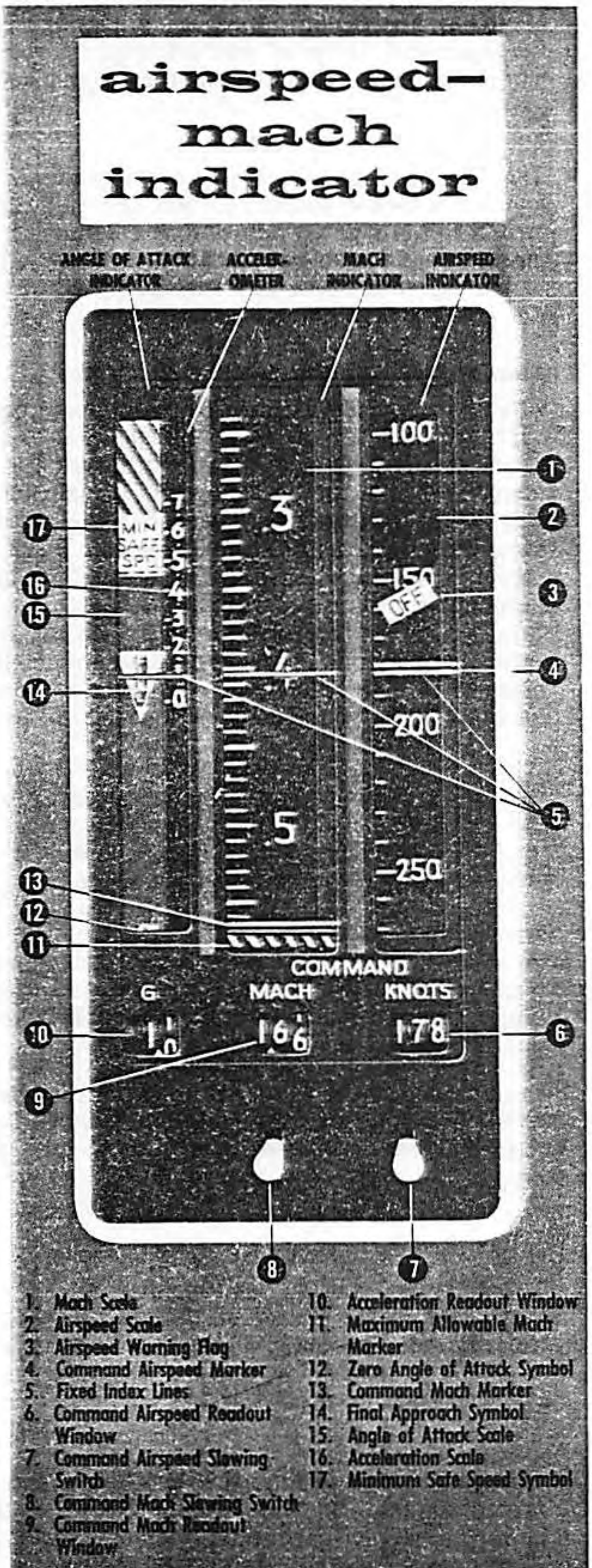


Figure 1-27E

attack for approach and landing remains the same. If final approach airspeed is held as shown in figure 7-4 (F-106B, 2000 pounds of fuel, no armament, 185 KCAS) the lubber line on the AMI will be located halfway down the triangle on the angle of attack indicator or through the letter "N" in the word "FINAL." At flare and touchdown, the base of the triangle (top edge) will be approximately under the lubber line. To fly a final approach speed using the angle of attack indicator, position the lubber line on the AMI halfway down the triangle on the angle of attack indicator or through the letter "N" in the word "FINAL." By using the angle of attack indicator, the airplane can be landed safely with complete pitot system failure. This instrument is calibrated primarily to indicate angle of attack at landing speeds, but it can be used to give pilots a general indication of maximum or optimum performance at high subsonic mach numbers. Exact performance at high subsonic mach numbers cannot be determined because variations exist between true angle of attack and tape positions. Pilots may, however, correlate tape positions with light, moderate, and heavy buffet and optimum or maximum turn. With any buffet present, the indicator tape will oscillate in time with the buffet. However, when attempting to obtain maximum performance at high subsonic speeds, positioning the lubber line between the tip of the triangle and the base of the triangle(figure 1-27F)will indicate the general area of optimum turn which can be defined as the best rate for the least energy loss. Light to moderate buffet will be present. A lubber line position from the base of the triangle to half way up the black will indicate the general area to be used for a defensive break or maximum turn. In this area, maximum turn can be generated, but airspeed is rapidly decreasing. Sustained lubber line position beyond half way up the black will produce lateral and directional instability from which recovery procedures must be initiated to preclude entry into post stall gyration or uncontrolled flight.

Accelerometer. The accelerometer (figure 1-27E) consists of a vertical, moving scale and a fixed index line adjacent to the angle of attack indicator. The index line monitors the moving scale to indicate g load from 0 to +7 g's. This reading is repeated in the acceleration readout window below the scale. There is no provision for indicating negative g's, or to record maximum g's during flight.

Mach Indicator. The Mach indicator in the center of the AMI(figure 1-27E)indicates true Mach number. A fixed index line monitors a vertical, moving Mach scale which is calibrated in hundredths, and calls out each tenth of a Mach from .2 through

3.0. At speeds below .4 Mach, the Mach scale will continue to indicate .4. A striped maximum allowable Mach marker that normally rests at the bottom of the display column will climb toward the index line when maximum safe Mach is approached. A double-line command Mach marker and a command Mach readout window below the scale indicate selected command Mach.

NOTE

> The command Mach marker will remain at the top or bottom of the Mach indicator until the selected command Mach reading comes into view on the moving Mach scale. At this time the marker will move with the scale toward the fixed index line. When indicated Mach equals command Mach, the command Mach marker and the index line will coincide.

Command Mach is controlled manually by the slewing switch under the Mach readout window, or automatically by the digital computer program. The command system is capable of selecting speeds from .4 through Mach 2.2.

Airspeed Indicator. The airspeed indicator located on the AMI (figure 1-27E) gives calibrated airspeed.

NOTE

> Airspeed information is electronically converted to calibrated airspeed (CAS) before being transmitted to the airspeed scale. Calibrated airspeed instrument system tolerances are a maximum of ±9 knots. This error could consist of ±2 knots instrument error, ±2 knots pitot static system error, and ±5 knots air data computer error.

A fixed index line monitors a vertical moving airspeed scale which is graduated in 10-knot markers and calls out each 50-knot level from 50 through 1000 knots. At airspeeds below 50 knots, the airspeed scale will continue to indicate 50. A double-line command airspeed marker and a command airspeed readout window below the indicator show manually selected command airspeed in knots.

NOTE

> The command airspeed marker will remain at the top or bottom of the airspeed indicator until the manually selected airspeed reading comes into view on the moving airspeed scale. At this time the marker will move with the scale toward the fixed index line. When indicated airspeed equals selected command airspeed, the command marker and the index line will coincide.

Command airspeed is always controlled manually by the command airspeed slewing switch under the command airspeed readout window and will not reflect data link or computer program information.

NOTE

> The command airspeed slewing switch has a center detent position. Moving the switch to the right, when centered, locks the switch and enables calibrated airspeed to be read from the command airspeed readout window.

Altitude-Vertical Velocity Indicator (AVVI)

The altitude-vertical velocity indicator (figure 1-27G) located on the instrument panel gives a vertical presentation of altitude information. The instrument contains three display columns. From left to right the columns are: the vertical velocity indicator, the altimeter, and the gross, cabin, and target altimeter. In addition, the AVVI has a barometric pressure readout window and set knob, a target altitude marker and readout window, a cabin altitude marker, two command altitude markers with a single command altitude readout window, and a command altitude slewing switch.

NOTE

> When local barometric reading is set into the AVVI by use of the barometric pressure set knob, the reading is also automatically set into the digital computer.

An error, noted when local pressure is set, may be corrected by use of the adjustment screw located adjacent to the barometric pressure set knob. This is a screwdriver type adjustment and is accessible by lifting the swing away tab.

In the event of ac power failure to the drive motor of the AVVI, an altitude warning flag will appear in the center of the pressure altitude display column.

NOTE

> Although the altitude warning flag appears only in the center of the altitude scale, it indicates that all the functions of the AVVI are inoperative (except cabin altitude marker).

The electrical driving power to the AVVI is from the ac essential bus. The command inputs to the indicator, however, are dependent upon MA-1 electrical power.

angle of attack indications

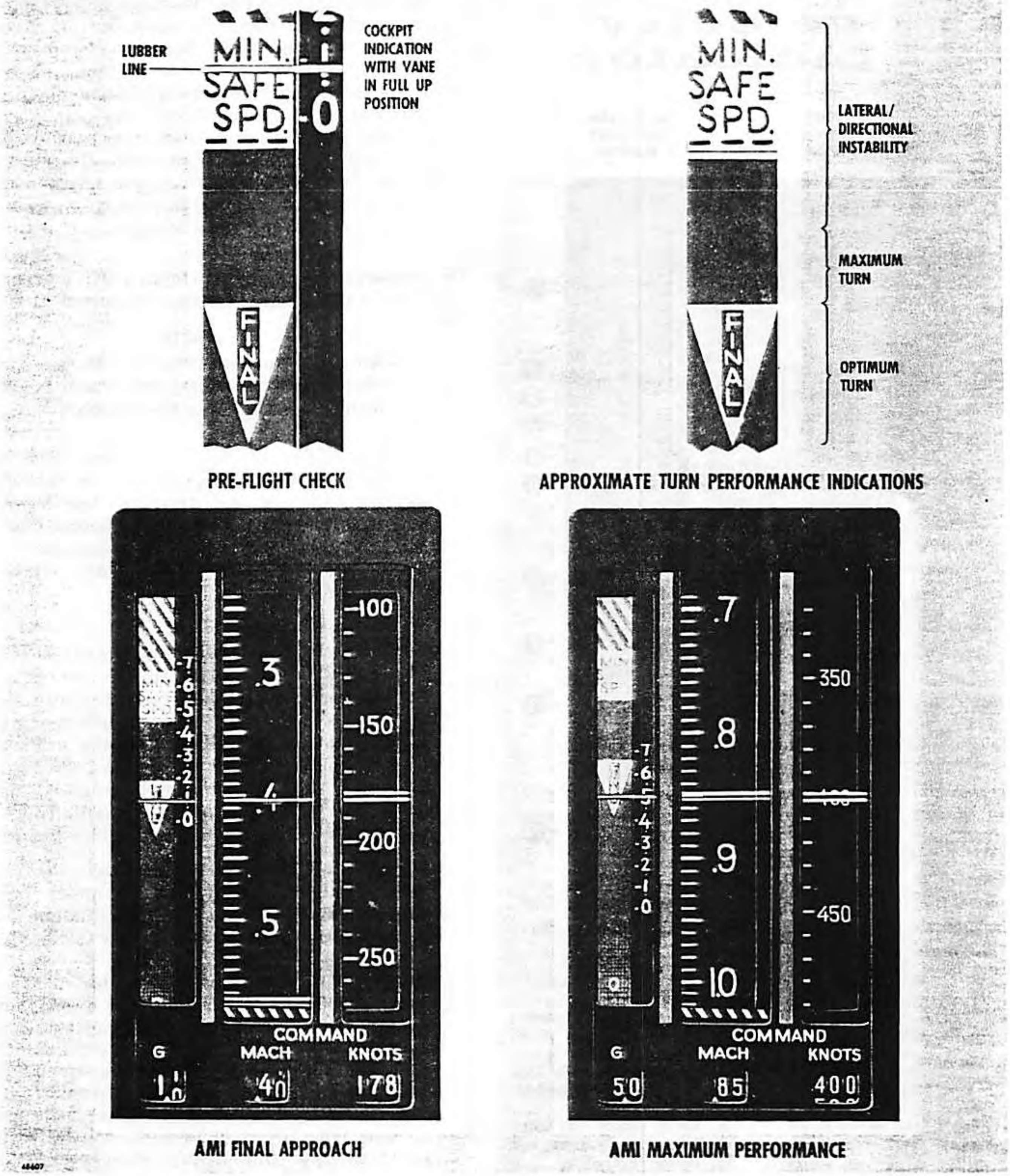


Figure 1-27F

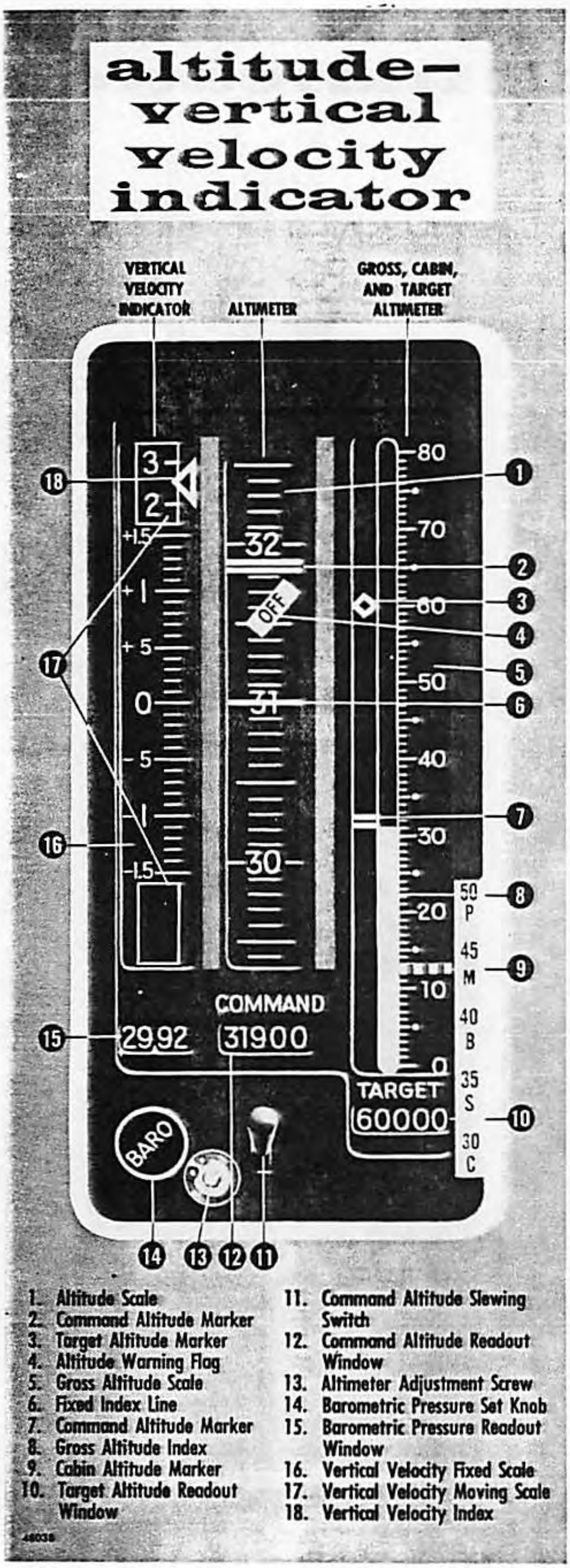


Figure 1-27G

Vertical Velocity Indicator. The vertical velocity indicator(figure 1-27G),located on the AVVI, indicates vertical velocity on a vertical velocity fixed scale calibrated from 0 to ±1500 feet per minute. All vertical velocities in excess of ±1500 fpm, up to ±40,000 fpm, are displayed on a vertical velocity moving scale and are viewed through readout windows located at the top and bottom of the indicator. The vertical velocity moving scale calls out only thousand-foot levels up to 10,000 feet, five-thousand-foot levels from 10,000 to 30,000 feet, and ten-thousand-foot levels from 30,000 to 40,000 feet. A vertical velocity index monitors vertical velocity and will move to the appropriate window as ±1500 fpm is exceeded.

Altimeter. The altimeter (figure 1-27G), in the center of the AVVI, indicates true pressure altitude.

NOTE

> Altitude information is electronically converted to true pressure altitude before being transmitted to the display scale.

A fixed index line monitors a vertical, moving altitude scale which is graduated in hundreds of feet and calls out each thousand-foot level from −1000 through 80,000 feet. A double-line command altitude marker and the command altitude readout window below the altimeter indicate selected command altitude.

NOTE

> The command altitude marker will remain at the top or bottom of the altimeter until the selected command altitude reading comes into view on the moving altitude scale. At this time the marker will move with the scale toward the fixed index line. When indicated altitude equals command altitude, the command altitude marker and the index line will coincide.

Command altitude is controlled manually by the command altitude slewing switch under the command altitude readout window, or automatically by the digital computer programs.

Gross, Cabin, and Target Altimeter. The gross, cabin, and target altimeter (figure 1-27G) located on the AVVI accumulates and displays all available altitude information. A moving thermometer-type gross altitude index indicates airplane true pressure altitude by referring to a fixed gross altitude scale. The scale is graduated in thousands of feet and calls out 10,000-foot levels up to 80,000 feet. A moving cabin altitude marker shows cabin altitude. Selected command altitude is shown by a double-line command altitude marker and is controlled manually by the command altitude slewing switch under the command altitude readout window or automatically by the digital computer program. Target altitude is shown by a small diamond-shaped target altitude marker and by the target altitude readout window located below the scale.

NOTE

Target altitude indication is dependent upon remote data link and homing point information: the target altitude marker cannot be adjusted manually.

Attitude-Director Indicator (ADI)

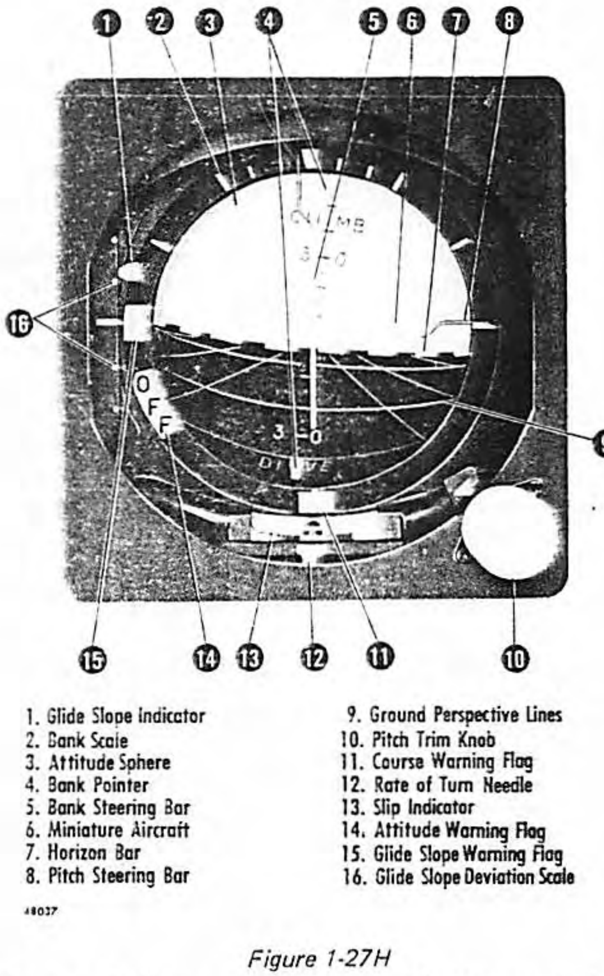
The attitude-director indicator (figure 1-27H) is located on the instrument panel. The ADI combines an attitude indicator, a turn-and-slip indicator, and a flight director and ILS glide-slope presentation. Unrestricted motion of the attitude sphere allows presentation of pitch and roll through 360°. The sphere moves behind a miniature aircraft fixed at the center of the instrument. A pitch trim knob allows manual positioning of the sphere with relation to the miniature aircraft. A pitch reference scale is located on the sphere and marked by small dots at 5°, small lines at 10°, callouts every 30°, and large dots at both poles. A bank scale at the top circumference of the instrument shows bank angles with 10° graduations up to 30°, and 30° graduations up to 90° of bank. Some ADI's have the bank index on the bottom of the instrument only. The turn-and-slip indicator is mounted at the bottom of the ADI and is centered with the vertical axis. A deflection of one needle width indicates a four-minute 360° turn. The turn indicator receives power from the dc essential bus. A bank steering bar and a pitch steering bar are superimposed on the attitude indicator to provide steering information. The bank steering bar shows amount and direction of bank required to position the airplane on a desired heading or course. The bar will center when the airplane is at the required bank angle.

NOTE

The bank steering bar indicates command bank angles dependent upon the selection of the display/automatic mode selector switch. If either DL-MAX RNG or DL-MIN TIME is selected, command bank angles can be up to 45° (45 degrees of bank required to center needle if substantial turn required); in AUTO NAV, 30°; in ILS, 30°; in ILS APCH, 15°.

The pitch steering bar indicates amount and direction of pitch angle required to position the airplane on the ILS glide-slope. The bar centers (1) when on glide-slope with proper pitch angle, (2) when pitch angle is correct for return to glide-slope, and (3) when pitch angle is correct for leveling out on glide-slope. The steering bars do not indicate direction or displacement from desired course or glide-slope, but rather, the corrective action required. The two bars are automatically stowed when not required, and warning flags appear to show that the glide-slope or localizer signals are invalid. The glide-slope indicator at the left center of the instrument shows actual position with reference to the glide-slope. An attitude warning flag appears in the lower left-hand portion of the ADI when ac power to the instrument is interrupted. The electrical driving power to the ADI is from the ac essential bus. The navigation and ILS functions, however, are dependent upon MA-1 electrical power. The ADI is inoperative when the airplane is receiving ground power unless the MA-1 power switch is in "WARM" or higher.

attitude-director indicator (typical)



1. Glide Slope Indicator
2. Bank Scale
3. Attitude Sphere
4. Bank Pointer
5. Bank Steering Bar
6. Miniature Aircraft
7. Horizon Bar
8. Pitch Steering Bar
9. Ground Perspective Lines
10. Pitch Trim Knob
11. Course Warning Flag
12. Rate of Turn Needle
13. Slip Indicator
14. Attitude Warning Flag
15. Glide Slope Warning Flag
16. Glide Slope Deviation Scale

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Figure 1-27H

WARNING

Malfunction of the ADI attitude indicators may be due to a failure other than ac electrical failure. In such an event, the malfunction will not be indicated by the attitude warning flag, but may be determined by cross-reference of other instrument indications, and the artificial horizon displayed on the radar scope.

Horizontal Situation Indicator (HSI)

The horizontal situation indicator (figure 1-27J) is centrally located on the instrument panel on airplanes with the integrated flight instrument system. The HSI integrates and visually presents information received from TACAN, UHF, and data link stations, the compass sytem, the computer, and certain pilot-operated controls. The main electrical power source to the instrument is from the ac essential bus. The compass card will be inoperative when the airplane is receiving ground power unless the MA-1 power switch is in "WARM" or higher. A description of the individual components of the HSI follows:

Compass Card. The compass card (4, figure 1-27J) consists of a compass rose against which a lubber line, located at the 12-o'clock position, indicates airplane magnetic heading. An extension of the lubber line at the 6-o'clock position indicates reciprocal heading.

Heading Marker. The heading marker (2, figure 1-27J is a rectangular marker located just outside the azimuth ring. It indicates the selected or command heading and, by its angular displacement from the lubber line, the heading error angle. The marker may be set manually by means of the heading set knob, or automatically by computer or data link information.

NOTE

The only reference for the command heading marker is the stable platform, consequently, the command heading marker is independent of any inaccuracies of the J-4 compass.

Bearing Pointer. The bearing pointer (3, figure 1-27J ia a heavy arrowhead located on the outside of the compass card. Depending upon the position of the bearing selector switch and display automatic mode selector switch, the bearing pointer will show bearing to a TACAN station, a UHF transmitter, a data link station, or a data link target.

Course Arrow and Course Deviation Indicator. The course arrow (6, figure 1-27J) and the course deviation indicator (8, figure 1-27J) are located within the compass card. In the NAV and ILS display/automatic modes the course arrow must be set manually with the course set knob to the desired course, and the course deviation indicator then indicates angular deviation from the selected course. In the DL modes, the course selector knob is inoperative, the course arrow is automatically positioned to target course, and the course deviation indicator indicates lateral distance from the interceptor to the target track.

NOTE

- In an ILS mode, maximum deflection of the CDI from center represents 2½°, or greater, angular deviation from either side of the selected course.
- In the NAV modes, each dot on the course deviation scale represents a 5° angular deviation from the selected course.
- In the DL modes, each dot on the course deviation scale adjacent to the miniature airplane represents 15 nautical miles of lateral deviation.

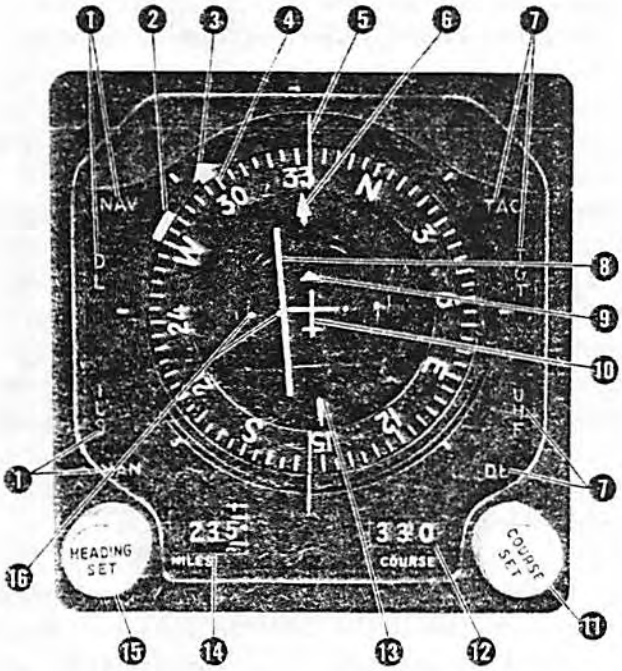
A "TO-FROM" indicator above or below the aircraft symbol indicates whether the selected course, if flown, will take the airplane to or from the TACAN station. The TO-FROM indicator is in relation to the head or tail of the course arrow and not the aircraft symbol.

NOTE

When either ILS mode is selected, the "TO-FROM" indicator is not operative, and is out of view.

Mode Indicator Lights. Eight mode indicator lights (abbreviated words) are located lateral to the compass card on the HSI. The four bearing pointer indicator lights on the right side indicate the information being given by the bearing indicator by illuminating "TAC," "TGT," "UHF," or "DL." If no right-hand bearing pointer indicator light is illuminated, the bearing indicator is slaved to airplane heading. Three of the four mode indicator lights on the left side indicate the operative mode of the display as selected by the display/automatic mode switch (NAV, DL, ILS). The lower left mode indicator light (MAN) is illuminated when the heading selector switch is in the MANUAL position or the display/automatic mode switch is in the MAN NAV position.

horizontal situation indicator



1. Mode Indicator Lights
2. Heading Marker
3. Bearing Pointer
4. Compass Card
5. Lubber Line (Upper)
6. Course Arrow (Head)
7. Bearing Pointer Indicator Lights
8. Course Deviation Indicator
9. To-From Indicator
10. Aircraft Symbol
11. Course Set Knob
12. Course Selector Window
13. Course Arrow (Tail)
14. Range Indicator and Range Warning Flag (Not Shown)
15. Heading Set Knob
16. Course Deviation Scale

J6086A

Figure 1-27J

Course Selector and Range Indicator. The course selector window (12, figure 1-27J) and the range indicator (14, figure 1-27J) are located below the compass card on the HSI. The course indicated by the course arrow is repeated in the course selector window. Depending upon the mode of operation, the range indicator indicates (in nautical miles) either the range to a target, range to responding aircraft, or the distance to a selected TACAN station.

NOTE

- If valid TACAN information has been received (range lock-on) and then lost, the computer will continue to present a range indication during subsequent operation. However, the accuracy of the indication will be compromised.
- The range indicator will not be shuttered and range will normally be erroneous if the range selector switch is in the 70 position and TACAN range is greater than 70 nautical miles.

Bearing Selector Switch

A four-position bearing selector switch (32, figure 1-9), placarded "Brg Sel," is installed on the instrument panel. The switch controls the bearing pointer indicator lights on the HSI. With the switch in the TAC position, the TAC indicator light will be on and the bearing indicator (3, figure 7-7) will indicate the bearing to a selected TACAN station. In the NORM position, the TAC indicator will be on and the bearing indicator will indicate the bearing to a selected TACAN station when the automatic mode switch (figure 7-1) is in the AUTO NAV or MAN NAV position, or bearing to a target when the automatic mode switch is in either DL position. The bearing indicator will servo to airplane heading in the NORM position when the automatic mode switch is in the ILS or ILS APCH position. In the UHF, or CMD ADF (some airplanes) position, the UHF indicator light will be on and the bearing indicator will indicate bearing to a selected UHF station. In the DL ADF position, the DL indicator light will be on and the bearing indicator will indicate bearing to the GCI transmitter to which the data link receiving equipment is tuned.

NOTE

If ac essential power is lost or if a change is made in the basic display mode (NAV, DL, ILS) with the automatic mode selector switch, the bearing selector switch will return to the NORM position.

Heading Selector Switch

A heading selector switch (33, figure 1-9), placarded "Hdg Sel," is installed on the instrument panel. The two-position switch selects the control input to the heading marker on the HSI and the bank steering bar on the ADI. With the switch in the MANUAL position, the heading marker and the bank steering bar may be set manually by the heading set knob. With the switch in the NORMAL position the heading marker will be controlled automatically.

NOTE

If ac essential power is lost, or if a change is made in the basic display mode (NAV, DL, ILS) with the automatic mode selector switch, the heading selector switch will return to the NORMAL position.

INSTRUMENT FAILURE

Instrument failure may be caused by (1) loss of electrical power, (2) individual instrument malfunction (other than electrical), or (3) failure of system components. See figure 1-15C for an analysis of instrument failure due to the loss of various electrical power sources. Individual instrument malfunction will be noted in some cases (see figure 1-15C) by an "OFF" flag which indicates loss of electrical power to the instrument. Failure of any other nature in these instruments, or in other instruments, except as noted can be detected only by observance of erroneous indications or faulty operation. The AMI and the AVVI may "freeze" at the last registered indication. If airplane speed and altitude changes are relatively small (as in final approach), a failure of this nature is difficult to detect. Therefore, these instruments, as well as all others, must be continually monitored and cross-checked with regard to known conditions.

STANDBY INSTRUMENTS

Standby Attitude Indicator

The standby attitude indicator (figure 1-28), located on the upper right side of the instrument panel, operates independently of the MM-3 or ARU-13A attitude indicator or the attitude-director indicator (ADI). The indicator functions only as an alternate cockpit display and may be used as a check against the primary indicator, or as a backup indicator in the event of primary indicator malfunction. The indicator provides indications of airplane attitude through 360° of bank and up to ± 82° of pitch. The indicator is remotely operated from an MD-1 vertical gyro in the vertical stabilizer. Climb, dive, and roll attitudes are shown by the circular motion of a universally mounted sphere displayed as the background for a miniature reference airplane. The miniature reference airplane is always in proper physical relationship to the simulated earth, horizon, and sky areas of the background sphere. The horizon is represented by a white horizontal bar, sky by a light grey area, and the earth by a black area. The pitch trim knob rotates the sphere to the proper position in relation to the miniature reference airplane to correct for variations of airplane level flight attitude. During initial gyro erection, and when power is off or is insufficient to keep the gyro stabilized, a flag placarded OFF appears in the bottom left side of the indicator. The indicator system is powered by the ac essential bus.

standby attitude indicator

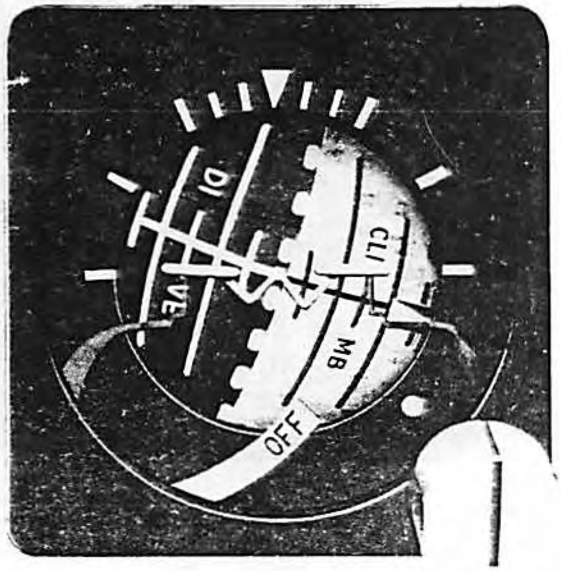


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Figure 1-28

Standby Airspeed Indicator

On airplanes with the integrated flight instrument system, the standby airspeed indicator (20, figure 1-9) is a pressure-sensitive airspeed instrument. The indicator is located above the exhaust gas temperature gage on the instrument panel and is a direct-reading pressure instrument not requiring electrical power for operation. The standby airspeed indicator gives essentially calibrated airspeed. Instrument tolerance of approximately ± 5 knots (at landing speeds) may cause error and should be corrected for if known.

Standby Altimeter (AAU-2/A)

On airplanes with the integrated flight instrument system, the pressure-sensitive standby altimeter (figure 1-29) is located above the engine pressure ratio gage on the instrument panel and is a direct-reading pressure instrument not requiring electrical power for operation. The standby altimeter should be corrected for instrument and installation error. Some standby altimeters can be adjusted by use of the set knob to set in the proper barometric setting.

CAUTION

On some type standby altimeters, barometric set knob must be pushed in to set barometric pressure. On other types, set knob must be pulled out to set barometric pressure. Assure that set knob returns to center position after setting barometric pressure on all types.

Standby Magnetic Compass

A standby magnetic compass is suspended from the canopy above the instrument panel and is furnished for navigational purposes in the event of electrical failure or the failure of navigation equipment. The compass is mounted on a track to facilitate manual positioning fore or aft. The compass will normally be in the aft position to avoid interference with the optical sight in the stowed position. However, when the sight is in use, the compass can be moved forward. This will provide the helmet clearance needed for proper use of the sight.

WARNING

- The standby magnetic compass is calibrated in the aft position and is unreliable in the forward position because of residual magnetism.
- The standby magnetic compass is calibrated with the optical sight in the stowed position and is unreliable with the optical sight in the unstowed position. With the optical sight in the unstowed position, the compass can be as much as 60° in error.

The compass is illuminated when the instrument panel lights are on. A standby magnetic compass correction card is located next to the compass.

COMPASS SYSTEM

Refer to COMPASS SYSTEM, Section IV.

CLOCK

A clock (37, figure 1-8 and 35, figure 1-9), located on the instrument panel, is an 8-day springwinding type. It contains an elapsed-time mechanism which uses a sweep-second hand. The elapsed-time mechanism is started, stopped, and reset by pushing in on the elapsed time button.

EMERGENCY EQUIPMENT

MASTER WARNING SYSTEM

A warning light panel (figure 1-30), located at the forward end of the right-hand console, has 24 individual amber warning lights which indicate malfunctions or failures of supporting equipment. Illumination of any individual light also illuminates an amber master warning light (31, figure 1-8, and 15, figure 1-9) on the main instrument panel, within the normal line of vision. Once illuminated, the master warning light can be extinguished (reset) by depressing the light. However, the individual warning light will remain illuminated until the malfunction is cleared. Subsequent malfunction will again illuminate the master warning light. Warning lights are automatically dimmed when the instrument panel lights are on if the dimmer switch has been momentarily placed in the DIM position and if the thunderstorm and augmentation lights are off. The master warning system does not include the landing gear unsafe warning light, canopy unlocked warning light, hydraulic pressure-low warning light, fire warning light, and variable ramp warning light (if located on the instrument panel).

NOTE

The landing gear unsafe, and fire warning lights can be pushed and the intensity of the lights will be dimmed. The master warning and the hydraulic pressure-low warning lights can be pushed and the lights will extinguish.

Power is supplied from the dc essential bus. For additional information on this system, refer to T.O. 1F-106A-2-10.

Warning Lights Test Button

A warning lights test button, (37, figure 1-11), placarded "Warn Light Test," is located on the right-hand console. Depressing the button tests the system as follows: The canopy unlocked warning light, the maximum maneuver warning light, the landing gear unsafe warning light, the tailhook light, and all the lights in the master warning system illuminate, and the hydraulic pressure-low warning light flashes (with hydraulic pressure).

Figure 1-29

On Ⓑ airplanes the control transfer lights and the takeoff trim indicator lights illuminate.

NOTE

Depressing the button is a functional check of lights only, not of complete warning circuits, nor of the landing gear audible warning system.

Warning Lights Dimmer Switch

A warning lights dimmer switch (34, figure 1-11) is located on the right-hand console. The switch has positions BRT and DIM. The warning lights may be dimmed by momentarily operating the switch to DIM, providing the flight instrument lights are on, the augmentation lights are off, and the thunderstorm lights are off. The light can be restored to bright by momentarily operating the switch to BRT. They will automatically return to bright if the flight instrument lights are turned off, the augmentation lights are turned on, or the thunderstorm lights are turned on. The warning lights will also return to bright if there is any power interruption to insure that the warning lights are never left on dim when the general lighting intensity is increased. The switch receives power from the dc essential bus.

ENGINE FIRE WARNING SYSTEM

The fire detection system (figure 1-31) consists of two independently routed detector loops which surround the entire engine compartment. Either loop may warn of a local hot spot (flashing warning light), but when both loops are energized a definite fire is indicated by a steady warning light. For additional information on this system, refer to T.O. 1F-106A-2-10.

Engine Fire Warning Light and Test Switch

An abnormally high temperature in the engine compartment is indicated by the red fire warning light (figure 1-31) located on the instrument panel. When illuminated, the light displays "FIRE." On airplanes with a test switch marked "Fire Det Test," the zone of the fire or local hot spot is not identified. When illuminated and steady, the warning light indicates a fire. When flashing, the light indicates that only one detector loop has been affected, representing a local hot spot (a fire that is too small to affect the other loop). During ground testing, when the test switch is placed to either LOOP 1 or LOOP 2 position, the warning light should flash, indicating proper operation of the loop, detector flasher, and warning light.

NOTE

> The warning light will illuminate steadily only when both circuits are closed, thus giving indication of a possible fire. No provision is made for testing the circuit for a steady illumination.

Warning light indication of fire can be confirmed by trailing smoke, high exhaust gas temperature, or visual sighting by a wingman. On B airplanes the fire warning test switch is located in the forward cockpit only and tests both the forward and aft warning lights. The warning system receives power from the dc essential bus.

TAILHOOK SYSTEM

The airplane is equipped with a tailhook system for use with "chain," "water squeezer," or "rotary friction unit" type arresting barriers. The system consists of a tailhook (29, figure 1-1) installed on the bottom aft portion of the fuselage, a tailhook uplatch and uplatch-release solenoid, and a combination release button/indicator light. The tailhook arm is a steel bar which provides spring tension to the down position. The spring tension acts as a snubbing force to minimize bounce of the hook when extended for barrier engagement. A ground safety pin with a red streamer (figure 1-23) is installed on the hook arm to prevent inadvertent extension when the airplane is on the ground. For additional information on this system, refer to T.O. 1F-106A-2-8.

Tailhook-Down Button

A tailhook-down button (42, figure 1-8, and 37, figure 1-9) is located on the instrument panel. The ring-guarded button is placarded "Tail Hook Down." The button electrically controls the uplatch release solenoid which actuates the uplatch mechanism that locks the tailhook in the retracted position. When the button is depressed, the uplatch cam rotates and releases the tailhook, and the spring tension in the hook arm places the hook in the extended position. A tailhook light is installed as an integral part of the tailhook-down button and when illuminated provides an indication that the tailhook has been extended. If the tailhook is inadvertently extended in flight, it will not introduce any problems while airborne. A safe landing can be made with the hook arm extended. There are no provisions for retracting the tailhook in flight. The tailhook-down button receives power directly from the battery.

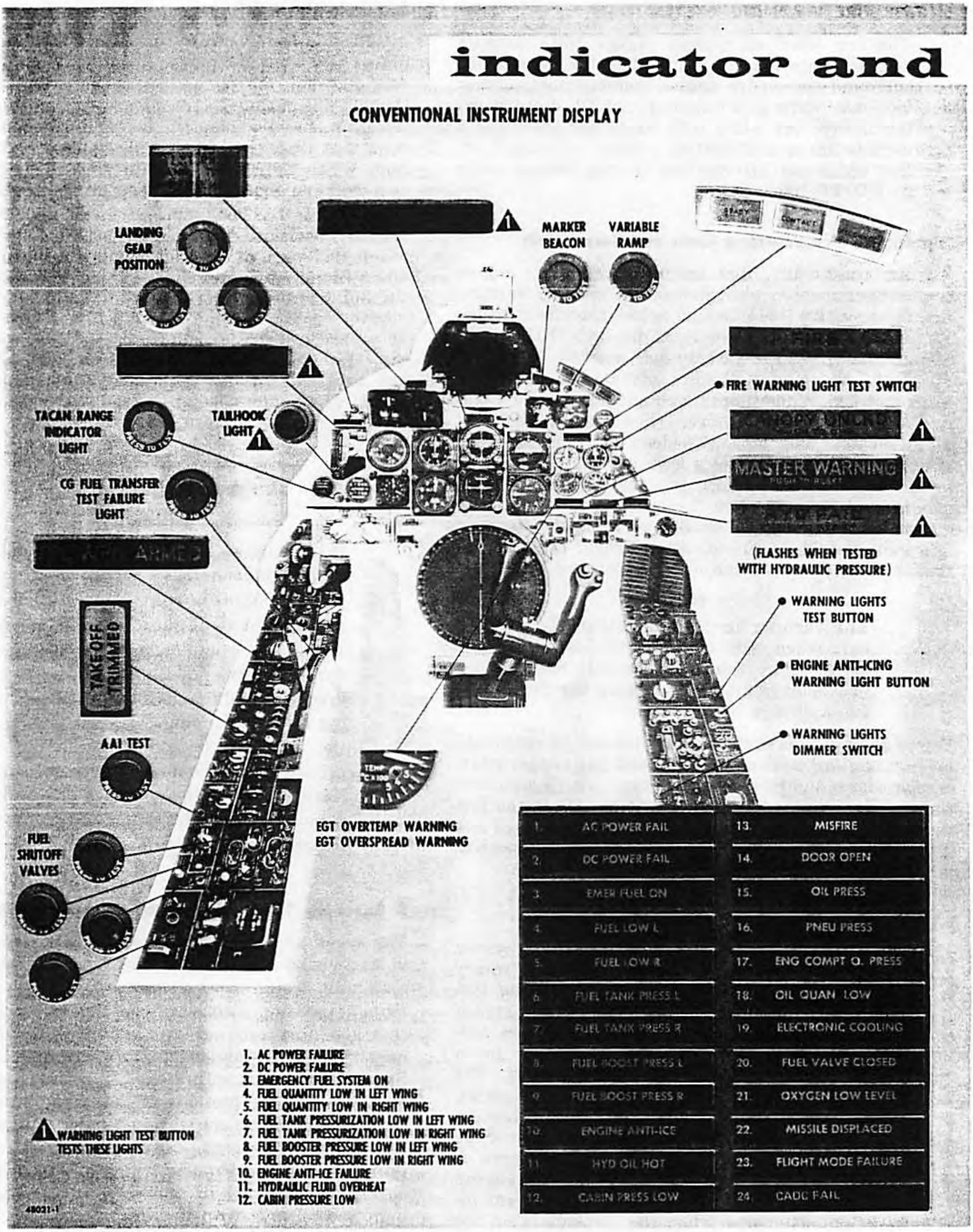
WARNING

> If the tailhook is inadvertently extended in flight, and if there is a barrier on the approach end of the runway, the airplane touchdown point must be compensated to avoid inadvertant engagement with the barrier. When extended, the tailhook may drag for 1000 feet before touchdown during a normal landing. When landing with the tailhook extended, fly a high final approach and touch down well past the arresting cable on the approach end of the runway or the tailhook may engage the barrier and severly damage the airplane.

SPEED AND TEMPERATURE WARNING SYSTEM

The speed and temperature warning system provides a visual indication, on the Mach indicator, of maximum design speed or the speed at which maximum stagnation temperature is reached. The system also provides an indication of maximum maneuver limit stagnation temperature of 174°F. Stagnation temperature (the temperature of the air heated by compression at the impact areas on the airplane) is measured by a temperature probe. The temperature probe feeds an electrical signal into the air data computer which in turn provides electrical switching of the maximum maneuver light and servo control of the maximum safe Mach pointer on the Mach indicator. If stagnation temperature is 174°F or above, the amber maximum

indicator and



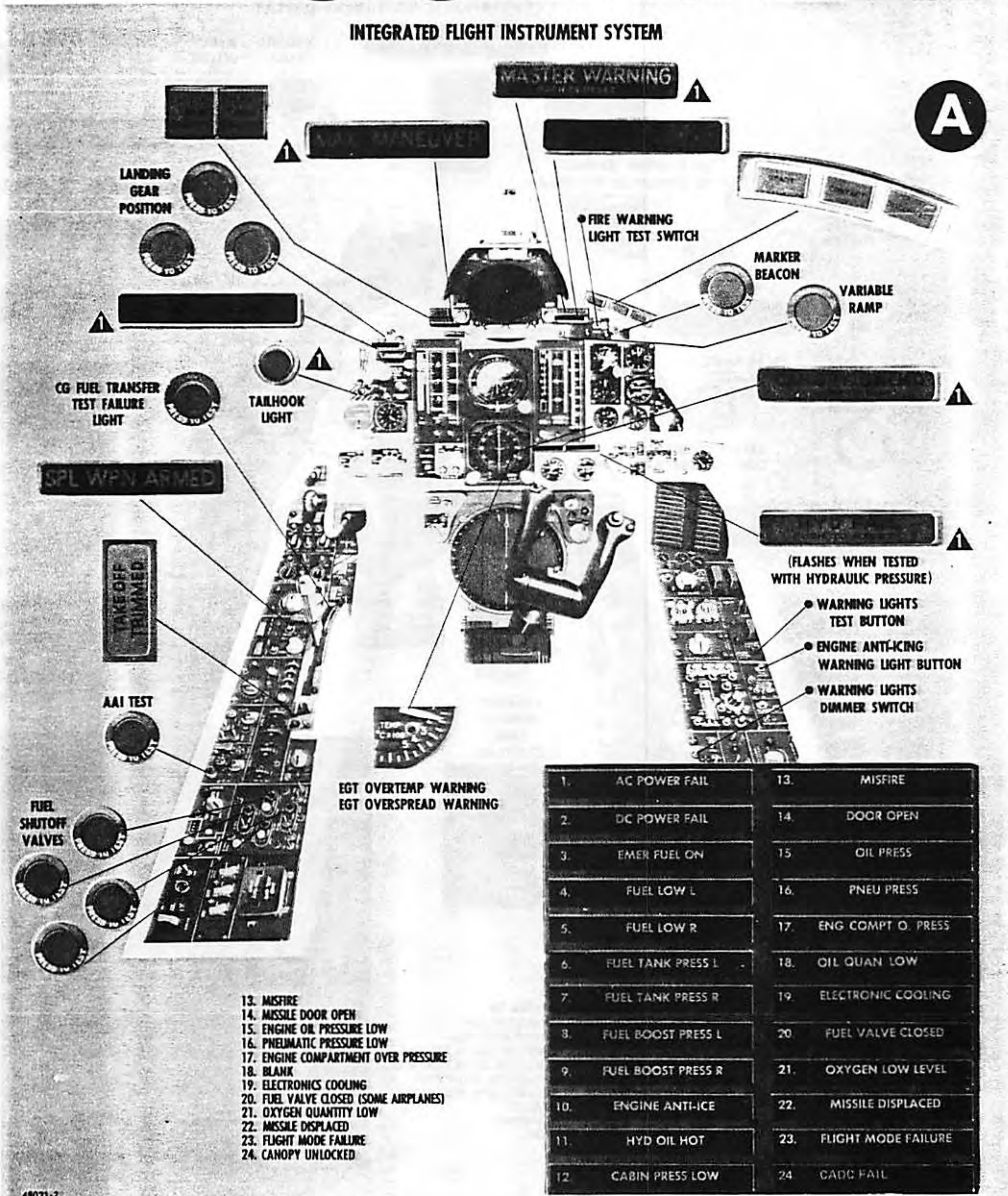
1.	AC POWER FAIL	13.	MISFIRE
2.	DC POWER FAIL	14.	DOOR OPEN
3.	EMER FUEL ON	15.	OIL PRESS
4.	FUEL LOW L	16.	PNEU PRESS
5.	FUEL LOW R	17.	ENG COMPT O. PRESS
6.	FUEL TANK PRESS L	18.	OIL QUAN LOW
7.	FUEL TANK PRESS R	19.	ELECTRONIC COOLING
8.	FUEL BOOST PRESS L	20.	FUEL VALVE CLOSED
9.	FUEL BOOST PRESS R	21.	OXYGEN LOW LEVEL
10.	ENGINE ANTI-ICE	22.	MISSILE DISPLACED
11.	HYD OIL HOT	23.	FLIGHT MODE FAILURE
12.	CABIN PRESS LOW	24.	CADC FAIL

1. AC POWER FAILURE
2. DC POWER FAILURE
3. EMERGENCY FUEL SYSTEM ON
4. FUEL QUANTITY LOW IN LEFT WING
5. FUEL QUANTITY LOW IN RIGHT WING
6. FUEL TANK PRESSURIZATION LOW IN LEFT WING
7. FUEL TANK PRESSURIZATION LOW IN RIGHT WING
8. FUEL BOOSTER PRESSURE LOW IN LEFT WING
9. FUEL BOOSTER PRESSURE LOW IN RIGHT WING
10. ENGINE ANTI-ICE FAILURE
11. HYDRAULIC FLUID OVERHEAT
12. CABIN PRESSURE LOW

Figure 1-30 (Sheet 1 of 4)

warning lights (typical)

INTEGRATED FLIGHT INSTRUMENT SYSTEM



1.	AC POWER FAIL	13.	MISFIRE
2.	DC POWER FAIL	14.	DOOR OPEN
3.	EMER FUEL ON	15.	OIL PRESS
4.	FUEL LOW L	16.	PNEU PRESS
5.	FUEL LOW R	17.	ENG COMPT O. PRESS
6.	FUEL TANK PRESS L	18.	OIL QUAN LOW
7.	FUEL TANK PRESS R	19.	ELECTRONIC COOLING
8.	FUEL BOOST PRESS L	20.	FUEL VALVE CLOSED
9.	FUEL BOOST PRESS R	21.	OXYGEN LOW LEVEL
10.	ENGINE ANTI-ICE	22.	MISSILE DISPLACED
11.	HYD OIL HOT	23.	FLIGHT MODE FAILURE
12.	CABIN PRESS LOW	24.	CADC FAIL

Figure 1-30 (Sheet 2 of 4)

indicator and

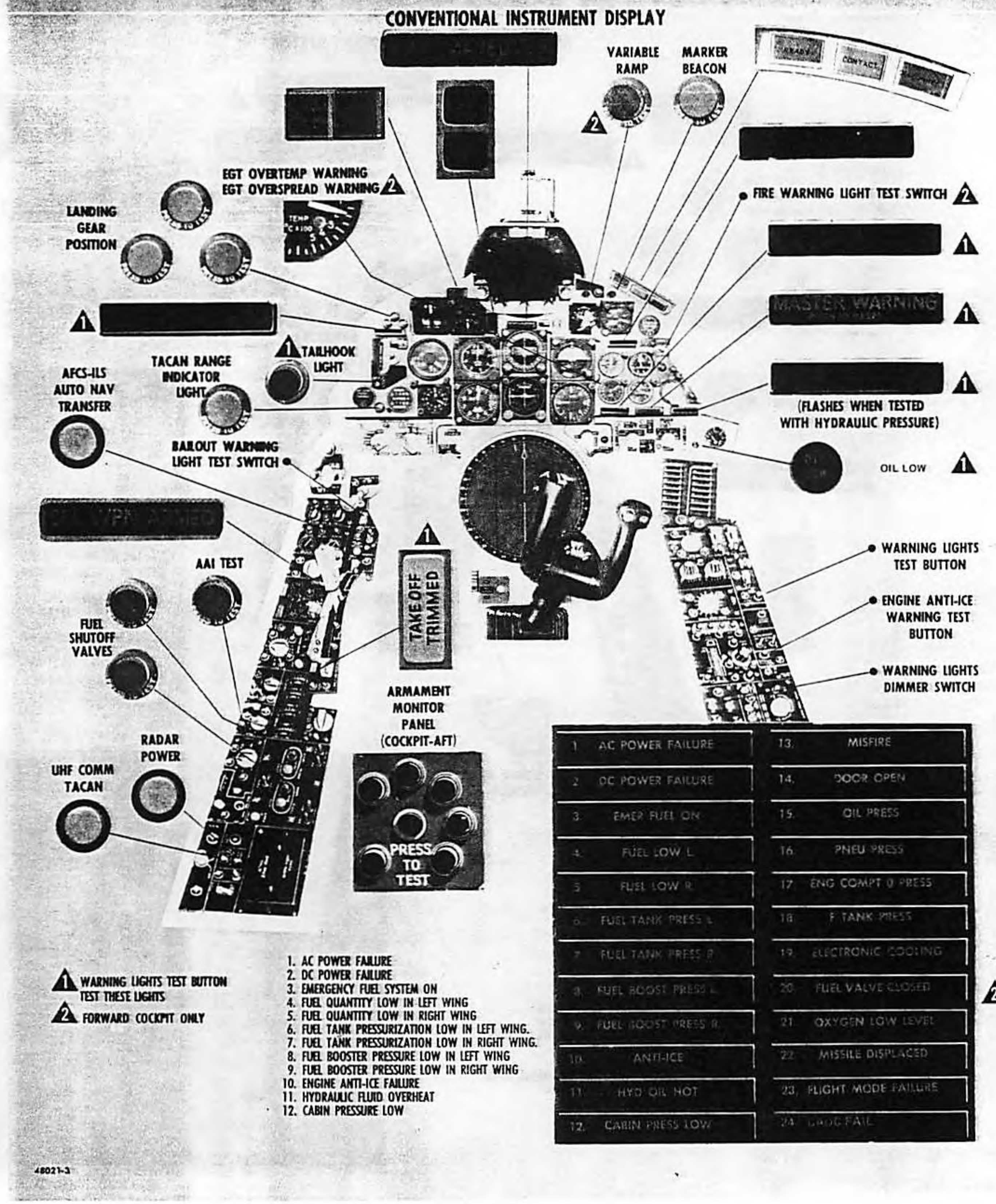
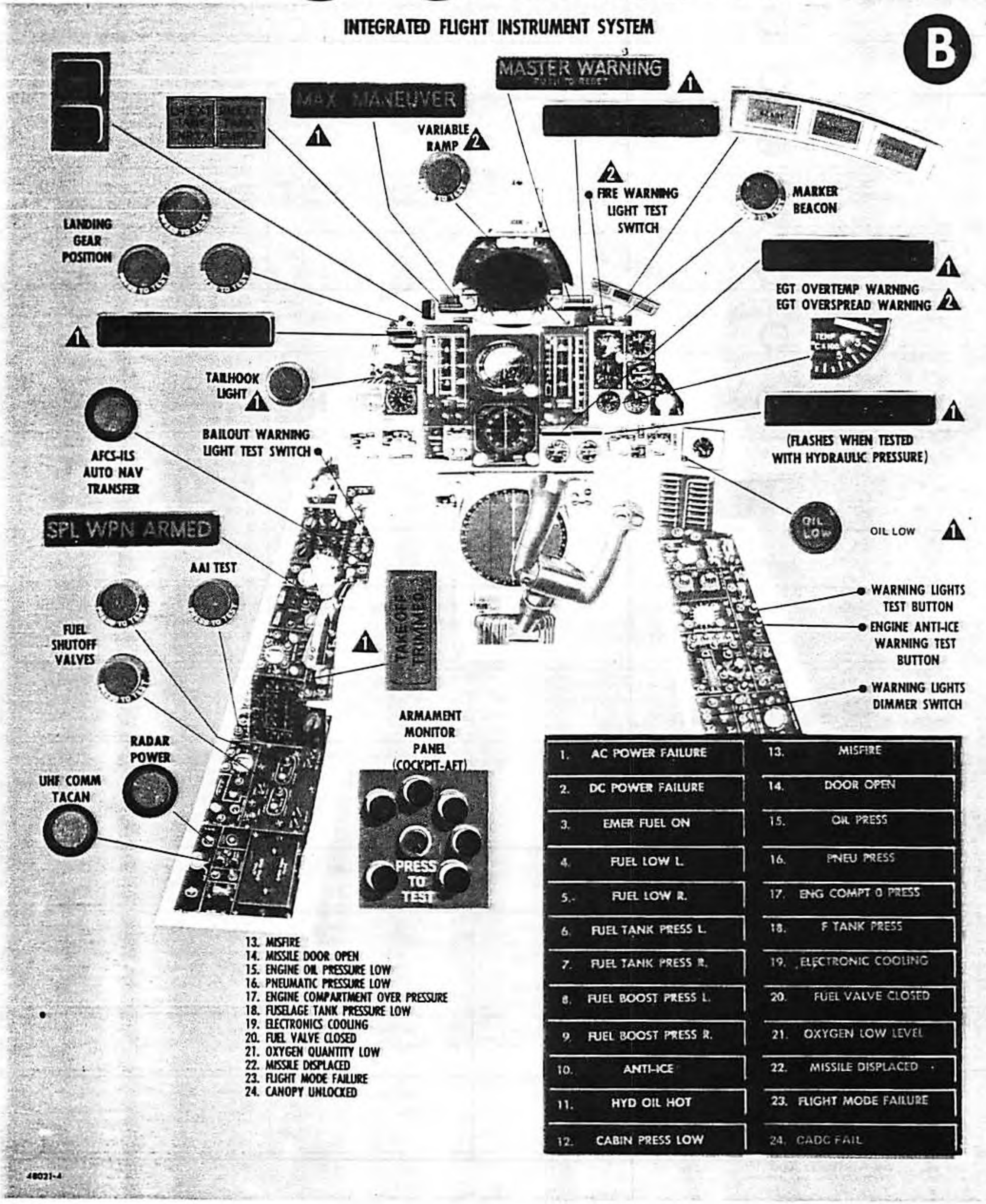


Figure 1-30 (Sheet 3 of 4)

warning lights (typical)



1.	AC POWER FAILURE	13.	MISFIRE
2.	DC POWER FAILURE	14.	DOOR OPEN
3.	EMER FUEL ON	15.	OIL PRESS
4.	FUEL LOW L.	16.	PNEU PRESS
5.	FUEL LOW R.	17.	ENG COMPT 0 PRESS
6.	FUEL TANK PRESS L.	18.	F TANK PRESS
7.	FUEL TANK PRESS R.	19.	ELECTRONIC COOLING
8.	FUEL BOOST PRESS L.	20.	FUEL VALVE CLOSED
9.	FUEL BOOST PRESS R.	21.	OXYGEN LOW LEVEL
10.	ANTI-ICE	22.	MISSILE DISPLACED
11.	HYD OIL HOT	23.	FLIGHT MODE FAILURE
12.	CABIN PRESS LOW	24.	CADC FAIL

Figure 1-30 (Sheet 4 of 4)

INDICATOR AND WARNING LIGHTS DESCRIPTION

LIGHT	CAUSE	CORRECTIVE ACTION	REMARKS
(B) UHF - Comm Tacan	UHF - Comm Tacan controls are transferred to the cockpit in which the light is illuminated	None	Control automatically transfers to forward cockpit when MA-1 power switch is in EMERG.
(B) Radar Power	Radar controls are transferred to the cockpit in which the light is illuminated.	None	Control automatically transfers to forward cockpit when MA-1 Power Switch is in EMERG.
Fuel shutoff	Respective shutoff valve in some position other than fully open.	Follow prescribed emergency procedure.	On (B) airplanes the Fuel Valve fail light will not illuminate when the F tank shutoff switch is in CLOSED.
(A) CG Fuel Transfer Test Failure	Low pressurization in the transfer and fuselage tanks.	Follow emergency procedures for Fuselage Tank Pressurization Failure	None
(B) AFCS - ILS Auto Nav Transfer	AFCS and ILS are transferred to the cockpit in which the light is illuminated.	None	Before control can be transferred both cockpit flight and auto mode switch settings must be the same.
TACAN Range Indicator (Conventional inst display)	TACAN/command altitude switch is in the TACAN RANGE position.	Info only.	This shows that the command altitude indicator is functioning as a TACAN distance indicator.
Tailhook Hook Down	Tailhook uplatch in extended position releasing the tailhook	Info only.	Illumination inflight indicates that the tailhook is extended and precautions should be observed to avoid inadvertant barrier engagement during landing.
External Tank Empty (LH and RH) LH EXT TANK EMPTY / RH EXT TANK EMPTY	External tanks are empty as indicated by the low-level float switches.	Info only.	The light will remain on until the high-level float switches indicate the tanks are full. It may illuminate during negative G maneuvers.
Max Maneuver MAX MANEUVER	Stagnation temperature is above 174°F.	Observe reduced maximum allowable load factors.	None
Variable Ramp Not Retracted	Failure of the ramps to fully retract at speeds below MACH 1.2	Follow prescribed emergency procedure.	The warning light is activated by a switch in the right ramp. The light will extinguish when the ramps are fully retracted.
Fire	The fire detection system has sensed a local hot spot (flashing light) or fire (steady light when both loops are energized).	Follow prescribed emergency procedure.	This system is deactivated when the master electrical power switch is OFF.
Canopy Unlocked	Both canopy latches are not fully engaged.	Follow prescribed emergency procedure if airborne.	The light dimming feature has been deactivated. If the IFF selector switch is in NORM or LOW emergency IFF will be activated while the light is illuminated.
Master Warning MASTER WARNING PUSH TO RESET	Illuminates when any of the warning light panel lights illuminate.	Extinguish by depressing the light.	The individual warning light will remain illuminated until the cause for illumination is corrected.
Hydraulic Fail	Either primary or secondary hydraulic system pressure has fallen below 900 (± 100) psi causes the light to flash. Pressure loss in both systems will cause steady illumination.	Follow prescribed emergency procedure.	The light may be extinguished by a pressure rise above 1,000 psi or by depressing the light if it is flashing.
Oil Quantity Low OIL QUAN LOW	Usable oil quantity in the tank has decreased to approximately one half.	Follow prescribed emergency procedures.	None
AC Power Failure 1. AC POWER FAIL	Failure of AC Generator.	Follow prescribed emergency procedure.	ATG and emergency AC generator should provide AC power automatically to ATG and AC essential buses.

Figure 1-30A (Sheet 1 of 2)

INDICATOR AND WARNING LIGHTS DESCRIPTION

LIGHT	CAUSE	CORRECTIVE ACTION	REMARKS
DC Power Fail 2. DC POWER FAIL	Failure of DC generator.	Follow prescribed emergency procedure	DC non-essential bus power will not be available. The Battery and TR will power the essential bus.
Emerg Fuel ON 3. EMER FUEL ON	Fuel control switch in EMERG position.	Info only.	Before engine start the light is illuminated with switch in either position.
Fuel Low - L Fuel Low - R 4. FUEL LOW L 5. FUEL LOW R	Illuminates when unusable fuel in each number 3 tank reaches approximately 570 pounds.	Follow prescribed emergency procedure and observe flight restrictions when operating with with low fuel quantity.	None.
Fuel Tank Press - L 7. FUEL TANK PRESS R R	Left or right number 1 tank pressurization below 1.0 psi.	Follow prescribed emergency procedure.	Normal fuel transfer into the number 3 tanks cannot be expected.
Engine Anti-Ice 10. ENGINE ANTI-ICE	Automatic engine anti-icing system is inoperative.	When flying in heavy icing conditions, select MAN ON otherwise select AUTO ON.	Light may illuminate during reduced airflow in intake duct or if in AUTO ON and probe heater is on for more than 25 secs.
Hyd Oil Hot 11. HYD OIL HOT	Hydraulic fluid in either system reaches a temp of 275°F at pump outlet. It can be an indication of pending flight control oscillations or pressure loss.	Follow prescribed emergency procedure.	There is no positive means for determining which system has overheated.
Cabin Press Low 12. CABIN PRESS LOW	Cabin pressure altitude has risen to 44,000 (±2,000) ft. and emergency cabin pressurization system is operating.	Follow prescribed emergency procedure.	Maximum recommended cabin altitude with MBU-3/P or MBU-5/P oxygen mask is 25,000 feet.
Misfire 13. MISFIRE	A firing signal does not result in launching selected armament.	Follow prescribed emergency procedure.	Light will illuminate briefly during any firing.
Door Open 14. DOOR OPEN	Illuminates approx 12 secs after missle bay doors are opened if armament firing cycle is not completed and the doors do not close.	Follow prescribed emergency procedure. Monitor fuel for range considerations.	May also illuminate if armament trigger is released after firing but before doors close.
Oil Press 15. OIL PRESS	Oil press has dropped to 37 ± 2 psi or below.	Follow prescribed emergency procedure.	None.
Pneu Press 16. PNEU PRESS	Pressure in the two main air storage flasks has dropped below 1700 (± 50) psi.	Follow prescribed emergency procedure.	None.
Eng Compt 0 Press 17. ENG COMPT O. PRESS	Engine accessory compartment pressure has reached 3.0 psi above ambient pressure.	Follow prescribed emergency procedure.	Once illuminated, the light will remain on for the rest of the flight.
F-tank Press 18. F TANK PRESS	Fuselage tank pressure has dropped below 6.0 (± 0.5) psi.	Follow prescribed emergency procedure for either F-tank emergency boost pump or F-tank emergency pressure system airplanes.	None.
Electronic Cooling 19. ELECTRONIC COOLING	Cooling air in the electronic compartment is insufficient for sustained equipment operation.	Follow procedure prescribed for illumination of electronic cooling light, ground or air operation.	Electronic cooling light is inoperative with MA-1 switch in EMERG.
Fuel Valve Closed 20. FUEL VALVE CLOSED	Any one of fuel shutoff valves in a position other than fully open.	Follow prescribed emergency procedure.	None.
Oxygen Low Level 21. OXYGEN LOW LEVEL	Liquid oxygen quantity below 1 liter.	Follow prescribed emergency procedure.	None.
Missile Displaced 22. MISSILE DISPLACED	Missile is not properly seated on the launcher.	Follow prescribed emergency procedure.	May be accompanied by the Misfire and Door-Open Warning Lights.
Flight Mode Failure 23. FLIGHT MODE FAILURE	Selected flight mode is inoperative.	None. Selector switch automatically steps back to an operative mode or direct manual.	Light will illuminate when power first applied to the airplane and is extinguished by depressing MMT.
Fuel Boost Press L Fuel Boost Press R 8. FUEL BOOST PRESS L	Left or right tank outlet pressure has dropped below 10.5 psi.	Follow prescribed emergency procedure.	None.

Figure 1-30A (Sheet 2 of 2)

Figure 1-31

maneuver warning light. is illuminated. With the amber light illuminated, load factor limits are more restrictive (refer to ACCELERATION LIMITATIONS, Section V). Maximum stagnation temperature is reflected by the maximum safe Mach pointer on the Mach indicator. The pointer indicates the speed at which maximum stagnation temperature is reached or the maximum design speed of the airplane, whichever occurs first. The speed and temperature warning system receives power from the ac essential bus. For additional information on this system, refer to T.O. 1F-106A-2-9.

Maximum Maneuver Warning Light

A maximum maneuver warning light (figure 1-30) is located on the instrument panel. The amber light illuminates and displays "MAX MANEUVER" whenever stagnation temperature is above 174°F. When the maximum maneuver warning light is illuminated, maximum allowable load factors must be reduced. For maximum load factor limits, refer to ACCELERATION LIMITATIONS, Section V. The light receives power from the dc essential bus.

SURVIVAL KIT B

A global survival kit (figure 1-33) is installed in the seat pan of the ejection seat. The kit is of fiberglass construction containing a rubber seat cushion and is to be used with a back-type parachute. The survival kit attaches to the parachute harness by means of an adjustable strap on each side which contains quick release features.

WARNING

The straps should be adjusted to secure the kit firmly in order to prevent the kit from "riding" up the back and interfering with chute deployment during ejection.

The kit is automatically actuated upon parachute deployment and contains an oxygen regulator, two emergency oxygen bottles, personal equipment leads, a one-man life raft (if required), a provision kit, and a reflector located on the back side of the survival kit lid. The oxygen regulator in the kit supplies pressure oxygen to be used with the pressure helmet and pressure suit during normal flight, ground operation, or in the event of ejection.

NOTE

This regulator will also perform satisfactorily with the MBU-3/P or MBU-5/P mask, if a reducer-adapter is inserted in the personal leads.

In the event of ejection, the oxygen regulator regulates breathing oxygen and pressure suit pressure for a period of 10 to 15 minutes. Oxygen pressure contained in the emergency oxygen bottles is indicated by a pressure gage which is visible through a small window located in the rear portion of the kit. Oxygen pressure in the emergency oxygen bottles should be 1800 psi and should be checked prior to each flight. A pressure reducer lowers the pressure from the emergency oxygen bottles to 40 to 60 psi prior to entering the oxygen regulator. The emergency bottles are actuated automatically as the ejection seat leaves the airplane during ejection or may be manually actuated any time the ship's oxygen supply is depleted or not supplying oxygen for breathing or pressure suit operation. Manual actuation of the emergency oxygen bottles is accomplished by pulling the emergency oxygen manual release (round green knob) attached to a cable which is in the personal equipment lead bundle. In the event of actuation of the emergency oxygen bottles while ship's oxygen is being supplied, oxygen from the emergency oxygen bottles will not be used until the ship's supply pressure furnished to the regulator falls below emergency oxygen pressure or the seat is ejected. Manual actuation of the emergency oxygen bottles also provides a minimum of 10 minutes breathing and pressure suit oxygen. The bundle of personal equipment leads is inserted into a receptacle in the right-hand corner of the kit. The receptacle contains connections for oxygen, partial pressure suit, mask defog, communications, and the green knob for manual control of the emergency oxygen bottles. The receptacle should be checked prior to each flight to determine that all leads are properly connected before the pilot connects the leads on his equipment to the opposite end of the bundle. The life raft is inflated by a carbon dioxide cylinder that is gravity actuated upon kit deployment. The provision kit is a waterproof packet strapped to the botttom of the survival kit and should contain items for survival in the area of operation. For additional information on this system, refer to T.O. 1F-106A-2-2.

Survival Kit Operation

There are three conditions for operation of the automatic survival kit container assembly: flight emergency egress, ground emergency egress, and ground normal egress from aircraft.

Flight Emergency Egress. Following ejection from the aircraft a sequence of events occurs consisting of free fall, separation from ejection seat, and parachute and survival kit deployment. During parachute deployment, tension is applied to a cable between the crewman's parachute harness and the automatic actuator located in the survival kit. This tension starts a timer running for a pre-determined period of 5 seconds. When the timer has run to zero, and if the crewman is at or below the predetermined operating altitude of 10,000 ± 1000 feet for kit deployment, a pressure sensing aneroid permits the firing of a gas cartridge. The pressure from the cartridge forces the release mechanism to operate and the following events occur:

1. Lid unlocks.
2. Personal leads separate from kit.
3. Kit harness release assemblies separate from kit.
4. Kit falls away (attached to crew member on a 25-foot dropline).
5. Life raft inflates.
6. Multi-stage release handle remains in place on survival kit.

NOTE

Incorporation of the automatic actuator in the survival kit does not in any way prevent manual deployment of the kit via the multi-stage release handle located on the right side of the kit. Should the manual system be used, all events in the preceding paragraphs, steps 1 through 6, will occur.

WARNING

- In the event of automatic actuator failure do not raise survival kit emergency multi-stage release handle during descent until the parachute is fully inflated to prevent the survival kit or the attaching lanyard from fouling the parachute.
- In the event of automatic actuator failure do not raise the survival kit emergency release handle until after descent to an altitude not requiring oxygen. The oxygen supply will be cut off due to personal leads release.

Ground Emergency Egress. Emergency egress from the aircraft, after landing, is accomplished by stripping the multi-stage release handle from the kit. This releases the left and right harness release wedges and the personal leads thus permitting the crew member to exit from the aircraft.

WARNING

Extreme caution must be exercised in the use of the emergency multi-stage release handle because of its similarity and close proximity to the right ejection handgrip.

Ground Normal Egress. Normal egress from the aircraft is accomplished by releasing quick-disconnect fittings of the left and right harness releases from the accessory rings on crew member's parachute.

CANOPY

The metal-reinforced plexiglass canopy (some airplanes) or stretched acrylic clear top canopy (other airplanes) (5, figure 1-1) is an electrically operated clamshell type which is hinged at the rear and swings up to provide access to the cockpit. Manually operated latches secure the canopy in the closed position, and a warning light illuminates whenever the latches are not in the closed position. On Ⓐ airplanes operation of the canopy is controlled through an actuator-remover installation by toggle switches, which are held for the desired direction of canopy travel and may be released at any time for intermediate positions. The actuator-remover also functions to jettison the canopy in emergencies as part of the escape system. Upon closing the canopy, an inflatable canopy seal around the base of the canopy is actuated automatically by movement of the canopy latch handle to the LOCKED position, permitting pressurization of the cockpit. The canopy may also be raised manually. With the canopy latches released, the actuator-remover clutch is disengaged, allowing the canopy to be raised manually to the full open position. On Ⓑ airplanes the canopy is normally opened and closed by an electric motor. Toggle switches control power to the canopy motor. The canopy motor has a brake which holds the canopy in any desired position. When the canopy latches are locked, a cable linkage to the motor brake releases the brake, allowing canopy seal expansion during cockpit pressurization. From outside the airplane, the canopy may be raised manually after the canopy latches are unlocked, and then the brake released through operation of the canopy motor brake release handle. Power for the canopy system is provided by either the airplane dc nonessential bus or a canopy power package containing a 31-volt battery. The canopy power package is normally charged by ac nonessential bus power. If the airplane electrical power is off and no external power is available, the canopy power package has power stored for a limited number of opening and closing cycles. A ballistic charge is used to jettison the canopy in emergencies. Emergency jettisoning of the canopy only is accomplished by pulling the canopy jettison handle or by pulling the canopy external jettison handle. When the ejection seat handgrips are raised to actuate the seat ejection system, the canopy is automatically jettisoned before the seat is ejected. For additional information refer to T.O. 1F-106A-2-2.

CAUTION

To avoid depletion of the canopy battery, the canopy should be operated, whenever possible, with either external power or airplane dc generator power on the line.

CANOPY SEAL

An inflatable rubber seal is installed around the base of the canopy to provide sealing of the canopy to the fuselage and windshield. Engine compressor bleed air is used to inflate the seal to permit cockpit pressurization during flight. A valve operated by the canopy latch mechanism automatically admits air pressure to inflate the seal when the canopy latch handle is moved fully forward to the LOCK position. The initial movement of the canopy latch handle to the aft position relieves pressure to the canopy seal. Actuation of the emergency cabin pressurization system also inflates the canopy seal.

CANOPY HOLD-OPEN SUPPORT

A removable canopy hold-open support(s) (figure 1-23) is provided for use during ground operations.

The hold-open support(s) fits between the canopy and the canopy sill on the left side only on Ⓐ airplanes and on both sides of Ⓑ airplanes.

WARNING

The canopy hold-open support(s) should be in place before entering or leaving the cockpit to prevent the possibility of serious personal injury should inadvertent closing of the canopy occur. When the hold-open support(s) is not in position, special care should be taken to keep clear of the area between the canopy and the canopy sill.

CANOPY SWAY BRACES Ⓑ

On Ⓑ airplanes canopy sway braces consist of a bracket mounted on the right forward ejection seat track, and an antisway guide mounted on the canopy cross member. During taxi operations, canopy side sway is eliminated by the antisway guide. The airplane can be taxied with the canopy open up to 12 inches, when the bracket is engaged by the antisway guide.

CANOPY SWITCHES

Two toggle switches, one located on the right console (5, figure 1-11) and the other in the external canopy control compartment below the left windshield (figure 1-32), permit electrical raising and

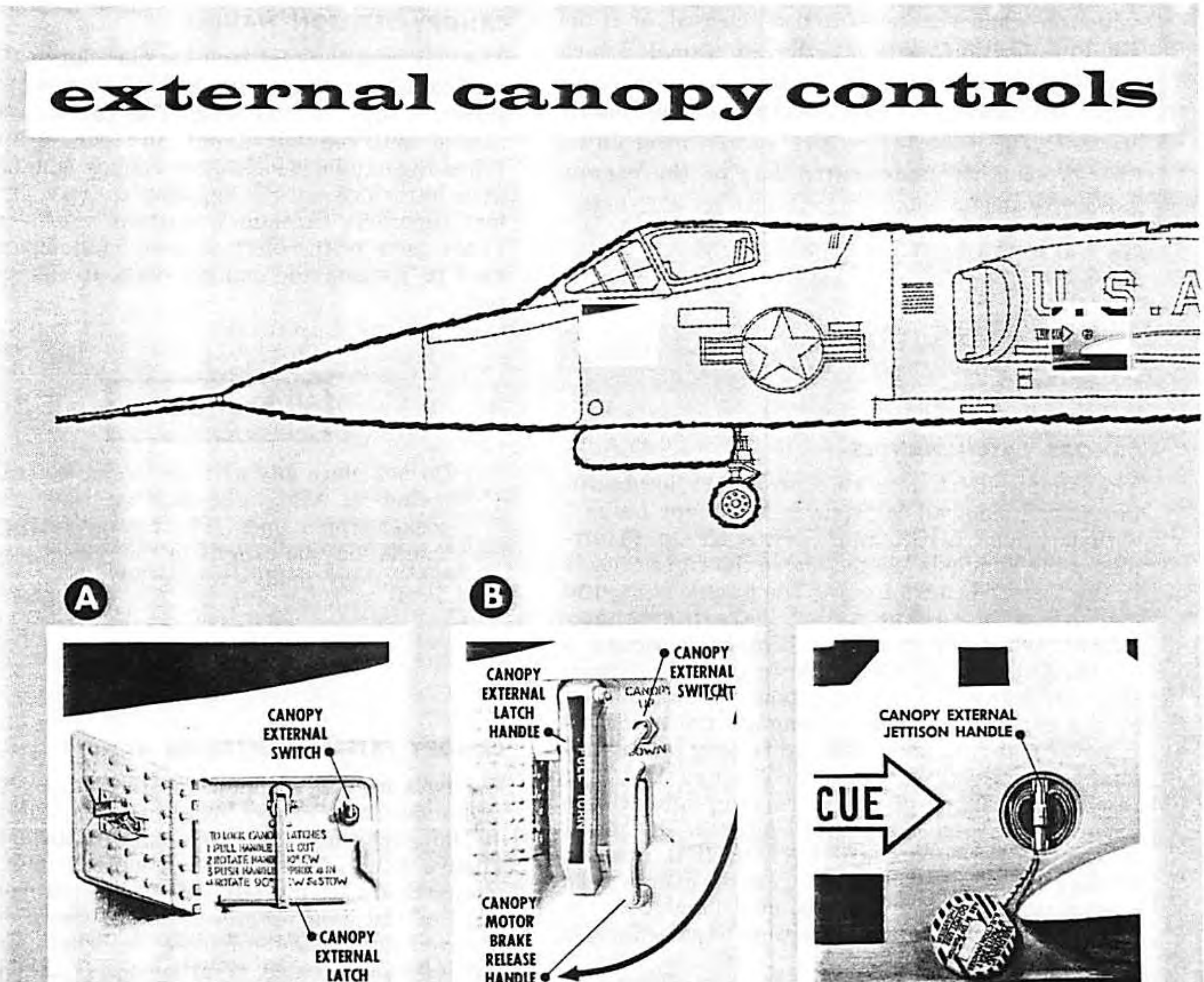


Figure 1-32

lowering of the canopy. On Ⓑ airplanes the canopy switch located on the right console is in the forward cockpit only. The switches have three positions: CLOSE, OPEN, and a spring-loaded STOP position. The exterior and interior switches are electrically interconnected. For canopy operation, either switch must be held for the desired direction of canopy travel and may be released at any time for intermediate positions. When the CLOSE position of the canopy switch is selected, a time delay feature is activated and the canopy will raise approximately 5 inches if it is not locked within three seconds after releasing the canopy switch. The canopy may be stopped at any intermediate position on the down cycle; however, it will raise approximately 5 inches three seconds after releasing the switch. On the up cycle the canopy will stop whenever the switch is released, or at the full up position, without the additional 5 inch travel. The automatic up feature is inoperative when the canopy is being operated on the canopy battery. The switches receive power from either the airplane dc nonessential bus or the canopy power package.

CANOPY LATCH HANDLE

The canopy latch handle (8, figure 1-11), located on the cockpit sidewall, is placarded "Canopy Latch," with positions LOCK and UNLOCK. On Ⓑ airplanes the handle is located above the right console in the forward cockpit only. The handle is used to mechanically engage and release the canopy latches. When the canopy is completely closed, a forward motion of the handle to the LOCK position will engage the canopy latches, allow inflation of the canopy seal, and disengage the actuator-remover clutch to allow manual raising of the canopy (if necessary) when the latches are later released. A canopy-unlocked warning light will go out when the latches are fully engaged. Movement of the handle to the aft UNLOCKED position mechanically dumps pressure from the canopy seal, disengages the canopy latches, and illuminates the warning light. The canopy can then be raised either electrically or manually.

NOTE

The canopy latch handle should not be moved to the LOCK position with the canopy open and the engine running, as the canopy seal will be damaged.

WARNING

- Do not move the canopy latch handle to the LOCK position with the canopy open. If the canopy-closed switch malfunctions or is not properly adjusted, the canopy will fall when the canopy latch handle is moved to the LOCK position.
- When in flight, if it is necessary to manually jettison the canopy by unlocking the canopy latch handle, ensure that the ejection seat handgrips are in the full down (in detent) position. Otherwise the canopy initiator units will be mechanically fired, which will in turn fire the seat initiators and eject the seat.

CANOPY JETTISON HANDLE

The yellow canopy jettison handle (figure 1-33) is used to jettison the canopy and is located on the left forward edge of the seat. The canopy jettison handle must be pulled out and raised upward. When the handle is raised the canopy is unlatched, then jettisoned with a ballistic charge. This system functions through a portion of the mechanisms used by the ejection seat handgrips and is used to jettison the canopy without ejecting the seat.

CAUTION

Do not place any articles in the left calf pocket of the flight suit or leave the pocket zipper open. The canopy jettison handle may be caught in the pocket, and the canopy inadvertently jettisoned.

CANOPY EXTERNAL JETTISON HANDLE

During emergency conditions the canopy can be jettisoned (by a ballistic charge) from outside the airplane by the canopy external jettison handle (figure 1-32). The handle is located in a small compartment on the left side of the fuselage forward of the wing intersection. An access door is removed to expose the handle. Approximately six feet of excess cable is attached to the handle, allowing the operator to stand a safe distance from the airplane before jettisoning the canopy. The canopy should travel up and aft sufficiently to clear the cockpit; however, it will probably strike the airplane midsection.

WARNING

The canopy external jettison handle should not be pulled except for emergency reasons. Prior to using the canopy external jettison handle the position of the ejection seat handgrips must be checked. If the handgrips are raised, do not pull the handle or the pilot and seat will be ejected. The ground safety pin, installed in the ejection seat right handgrip linkage, prevents jettisoning of the canopy from the cockpit only.

CANOPY EXTERNAL LATCH HANDLE A

The canopy external latch handle (figure 1-32), located below the left windshield, can be used to either release or close the canopy latches. Access to the handle is obtained by opening an access door. The latches are released by pulling the handle directly outward approximately five inches. Pushing the handle in returns the handle to the stowed position. To close the latches, the handle must be pulled out its full travel, rotated 90°

clockwise, and pushed in approximately four inches. After the latches are closed, the handle can be returned to the stowed position by turning the handle 90° counterclockwise and then pushing it in its full travel.

CANOPY EXTERNAL LATCH HANDLE (B)

The canopy external latch handle (figure 1-32), located below the left windshield, can be used either to open or close the canopy latches. Access to the handle is obtained by opening an access door placarded "Canopy Control." The handle is disengaged when stowed. If latches are open, the handle is engaged by pulling straight out as far as possible (approximately three inches). If latches are closed, the handle is engaged by pulling the handle straight out as far as possible (approximately 2½ inches), rotating the handle aft and up as far as possible (approximately 100°), and then pulling the handle out as far as possible (approximately ½ inch). With the handle engaged the latches are opened by rotating the handle forward and down to the vertical position and are closed by rotating the handle up and aft to a position approximately 10° above horizontal. With the latches open, the handle is disengaged and stowed by pushing the handle straight in as far as possible. With the latches closed, the handle is disengaged and stowed by pushing the handle in about ½ inch, rotating the handle forward and down to the vertical position, and then pushing the handle straight in as far as possible.

CANOPY MOTOR BRAKE RELEASE HANDLE (B)

A canopy motor brake release handle (figure 1-32) is located adjacent to the exterior canopy latch handle. The handle is used to disengage the canopy motor brake so that the canopy can be manually opened without use of electrical power. After the canopy latches have been unlocked, the canopy motor brake release handle can be pulled down through an arc of 60°, and a cable attached to the motor brake releases the brake, thus allowing manual operation of the canopy.

CANOPY UNLOCKED WARNING LIGHT

The red canopy unlocked warning light (figure 1-30), located on the instrument panel, illuminates and displays "CANOPY UNLKD" when both canopy latches are not fully engaged. The canopy unlocked warning light is tested together with the lights in the master warning system and will illuminate when the master warning light test switch is depressed. The warning light receives power from the dc essential bus.

EJECTION SEAT

EMERGENCY EGRESS SYSTEM

On (A) airplanes, emergency egress is accomplished by a one-motion ejection seat equipped with a high impulse rocket, automatic opening safety belt, seat-man-separator, global survival kit, pressure actuator assembly, parachute disconnect assembly, pilot air and seat electrical disconnect, ditching control handle assembly, and a ballistically deployed BA-18 parachute. Gripping and raising either ejector seat handgrip will jettison the canopy and fire the two-stage rocket catapult. The booster phase boosts the seat up the rails and the sustainer phase ignites just prior to seat/airplane separation. The nozzle angle of the rocket sustainer phase provides an upward and forward thrust to the seat allowing tail clearance and reducing the deceleration forces experienced by the pilot. The correct ejection procedure is to grasp both handgrips and pull up simultaneously. The shoulder harness inertia reel has a manual lock-unlock lever on the left side of the seat and locks automatically during ejection and when a rapid pull (equivalent to 2- to 3-g deceleration force) is exerted on the shoulder harness straps. The pressure suit ventilation hose is attached to a disconnect unit on the left side of the seat. The airplane-to-survival kit disconnect connects through the right corner of the seat to the middle block of the personal leads blocks and completes the connection for faceplate heating, communications, and oxygen between the pilot and the airplane. Both of these connections disconnect automatically during egress sequence. A limit switch is installed to disengage the AFCS allowing the stick to return to neutral so that interference between the seat, pilot and control stick will be eliminated. The arm guards are retracted until raised mechanically by the actuation of the ejection seat handgrips.

The seats installed in (B) airplanes look and function identically to those of (A) airplanes but are interlocked to ensure safe and orderly egress of both pilots should the rear pilot be unconscious or incapacitated. When the front seat ejection handgrip is raised, the canopy is jettisoned, both seats are armed, the aft shoulder-harness inertia reel is locked ballistically, and the rear seat ejects, followed in one second by the front seat. When the rear ejection seat handgrips are raised, the canopy is jettisoned, both seats are armed, and the rear seat is ejected. Unless circumstances dictate otherwise, the rear seat occupant should always initiate his own ejection. This ensures proper preparation and body position.

WARNING

- On Ⓑ airplanes, unless circumstances dictate otherwise, the rear seat occupant should initiate his own ejection. If the front pilot initiates ejection, the aft seat handgrips will not be raised and the front seat will eject after a one-second-delay, regardless of aft seat position.
- Ground maintenance safety pins are inserted in the canopy initiator, the seat arming initiator, and the initiator for the safety-belt, seat-man separator and pressure actuator during maintenance operations. The canopy jettison and seat ejection system is safetied by a ground safety-lock pin inserted through the right handgrip linkage. On Ⓑ airplanes, the seats are interlocked but safetied independently; therefore, the safety pin safeties only the seat in which it is inserted. The ground safety-lock pin does not safety the canopy jettison system if the external canopy jettison handle is pulled. This pin must be removed before flight and replaced after flight by the pilot. Stowage for the red warning streamer and the ground safety pin is provided on the bulkhead above the right console. If any of these pins are not removed, canopy jettisoning, seat ejection and/or automatic opening of the safety belt will not take place.
- The handgrips should be raised to the fully up and locked position.
- Do not squeeze the handgrip release trigger at any time other than for an actual ejection. If the handgrip release trigger is squeezed, the handgrip downlock cable will be released and the ejection seat handgrips will be released and the ejection seat handgrips will not be properly stowed. During the pre-flight, the pilot must check that the handgrip downlock cable is in place and the ball in the race.
- The seat ejection system is dependent upon the canopy jettison system. If the canopy is not jettisoned, the seat cannot be ejected. If raising the ejection seat handgrips does not jettison the canopy, the ejection seat handgrips must be returned to the stowed position by moving the uplock release lever forward and pushing the handgrips down into the detent (fully stowed) position. Then the canopy must be alternately jettisoned, either by the canopy jettison handle, or by unlocking the canopy and raising it into the airstream with the electrical switch on the right console. After the canopy has been removed, the handgrips can be raised to eject the seat.

EGRESS SEQUENCING

On Ⓐ airplanes, when either ejection seat handgrip is raised to the full up position, mechanical linkage will lock the shoulder harness, raise both arm guards, disengage the seat firing pin pawl, and fire an initiator unit. (See figure 1-34.) If the canopy has not been jettisoned, the expanding gases will fire a thruster unit which disengages the canopy latches and fires an additional initiator unit. The resulting expanding gases are then routed to remove a pin which fires the seat catapult, ejecting the seat. If the canopy has been alternately jettisoned, raising either ejection seat handgrip will mechanically disengage the seat firing control pawl firing the seat catapult initiator. The seat catapult initiator then fires the seat catapult, ejecting the seat.

On Ⓑ airplanes, the aft seat should be ejected first, followed by the forward seat. Raising either handgrip on the forward seat to the full up position, mechanically locks the front shoulder harness, raises both front arm guards, ballistically locks the aft seat inertia reel, disengages the seat firing control pawls, and fires an initiator unit. (See figure 1-34.) The expanding gases are then routed to energize the canopy remover which will jettison the canopy. When the canopy leaves the airplane, a lanyard fires two initiator units. The expanding gases produced are routed to remove a seat firing control pin in the aft seat. When the firing control pin is removed, the seat firing initiator is fired. This in turn fires the rear seat rocket catapult, ejecting the seat. After one second delay (regardless of rear seat position) the seat firing control pin is removed from the forward seat, and the seat firing initiator fires the forward seat rocket catapult. When either aft seat handgrip is raised, the sequence is similar. However, the forward seat is armed but can be ejected only when either forward handgrip is raised. If the canopy is jettisoned, either manually or by use of the canopy jettison handle, the seat firing control pins in both seats are removed. Both seats are then armed so that when the jettison handgrips on either seat are raised, the seat firing control pawl is disengaged firing the seat catapult initiator which in turn fires the seat rocket catapult, ejecting the seat. In this sequence, the ejection seats are interlocked so that lifting the forward seat handgrip will eject the aft seat first, then the forward seat. When the aft handgrips are raised, only the aft seat is ejected.

Low-Altitude, Low-Speed/Egress Sequencing

During zero-altitude, zero-airspeed and low-altitude, low-airspeed ejections, as the catapult moves the seat up the rails, the pilot air vent tubes and seat electrical cables are separated from the airplane, the airplane oxygen supply is terminated, the survival kit oxygen supply is activated, and the rocket motor is ignited. The rocket motor burns for approximately one-half second providing sufficient thrust to propel the man-seat combination to approximately 200 to 300 feet above ejection altitude (assuming level flight ejection). Approximately one second after the seat starts up the rails, the ejection seat ballistic system initiator fires, automatically opening the safety belt, actuating the seat-man separator, arming the deployment gun, and releasing the firing lanyard from the parachute disconnect. Two seconds after seat-man separation (three seconds from the time the seat starts up the rails) the deployment gun fires a 13 ounce slug off the right shoulder of the pilot that pulls the pack pins and extracts the pilot chute and main canopy from the pack and forces them into the airstream. Full canopy is realized approximately three seconds later at an altitude of approximately 150 feet above ejection altitude and 400 feet downrange (assuming level flight ejection).

WARNING

Do not attempt to beat the system by manually pulling the D-ring prior to automatic chute deployment by the drogue gun. At airspeeds below 150 knots, the automatic system must be depended upon for successful ground level ejection.

High-Altitude, High-Speed Egress Sequencing

Sequencing for a high-altitude, high-speed ejection is the same as the low-altitude, low-speed ejection except that the pilot free falls to approximately 15,500 feet before the barometrically controlled deployment gun fires. Full canopy is realized approximately 1 1/2 seconds later.

Low-Altitude, High-Speed Egress Sequencing

Sequencing for a low-altitude, high-speed ejection is the same as the low-speed ejection except that the man-seat ballistic trajectory is not as high as the low-speed ejection. However, a full canopy is realized approximately 1 1/2 seconds after drogue gun firing.

Automatic-Opening Safety Belt

Thorough testing of the automatic-opening belt has determined that the system is completely reliable and allows faster separation from the seat than does manual operation. It has also been determined that under no circumstances should the belt be manually opened prior to ejection.

EJECTION SEAT HANDGRIP (LEG GUARD)

The ejection seat handgrips (figure 1-33) operate the one-motion ejection system. The handgrips are locked in the down (stowed) position by a detent and a handgrip downlock cable. When raised, the handgrips serve as leg guards during the ejection. The right and left ejection seat handgrips are interconnected so that unlocking and raising either handgrip will raise the other. The arm guards are mechanically positioned by raising either handgrip.

EJECTION SEAT HANDGRIP (LEG GUARD) UPLOCK RELEASE LEVER

The ejection seat handgrip uplock release lever is located adjacent to the right ejection seat handgrip and is placarded "Leg Guard Uplock Release". In the event the ejection seat handgrips have been raised and the canopy and seat do not eject; the lever must be moved forward and the ejection handgrips pushed to the fully down position (in detent) in order to deactivate the canopy and seat ejection system.

DITCHING CONTROL HANDLE

The ditching control handle (figure 1-33) is located on the right arm rest of ejection seat. The yellow handle is placarded "Pull Ditch" and is used for normal or emergency egress from the seat when wearing the ballistically deployed parachute. Raising the handle releases the parachute firing lanyard from the parachute disconnect assembly. This allows separation from the seat without firing or arming the parachute deployment gun.

SEAT-MAN SEPARATOR

A seat-man separator is installed on the ejection seat. The seat-man separator will separate the pilot from the seat approximately one second after the seat is ejected from the airplane. The automatic opening safety belt, the seat-man separator, and ballistically deployed parachute are sequenced to provide the most desirable operation and will be more reliable than manual operation.

CHAFF DISPENSER

A chaff dispenser assembly is installed on the upper right side of the seats. It is secured by a lanyard pin assembly which is attached to the airplane structure. Upon ejection, the lanyard pin assembly opens the folded flaps of the chaff dispenser and exposes the chaff to the airstream. This radar reflective chaff aids in the location and rescue of the ejected pilot.

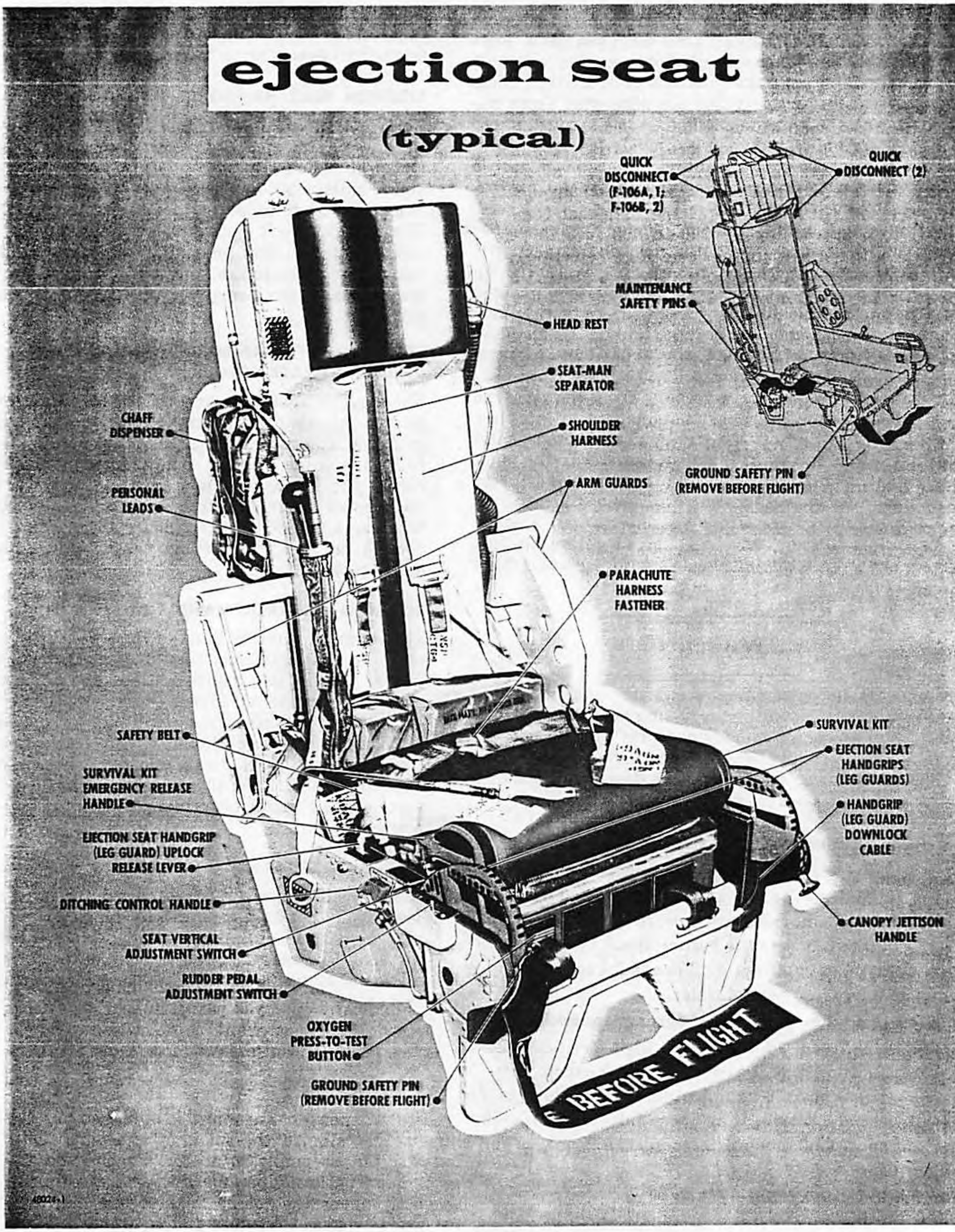
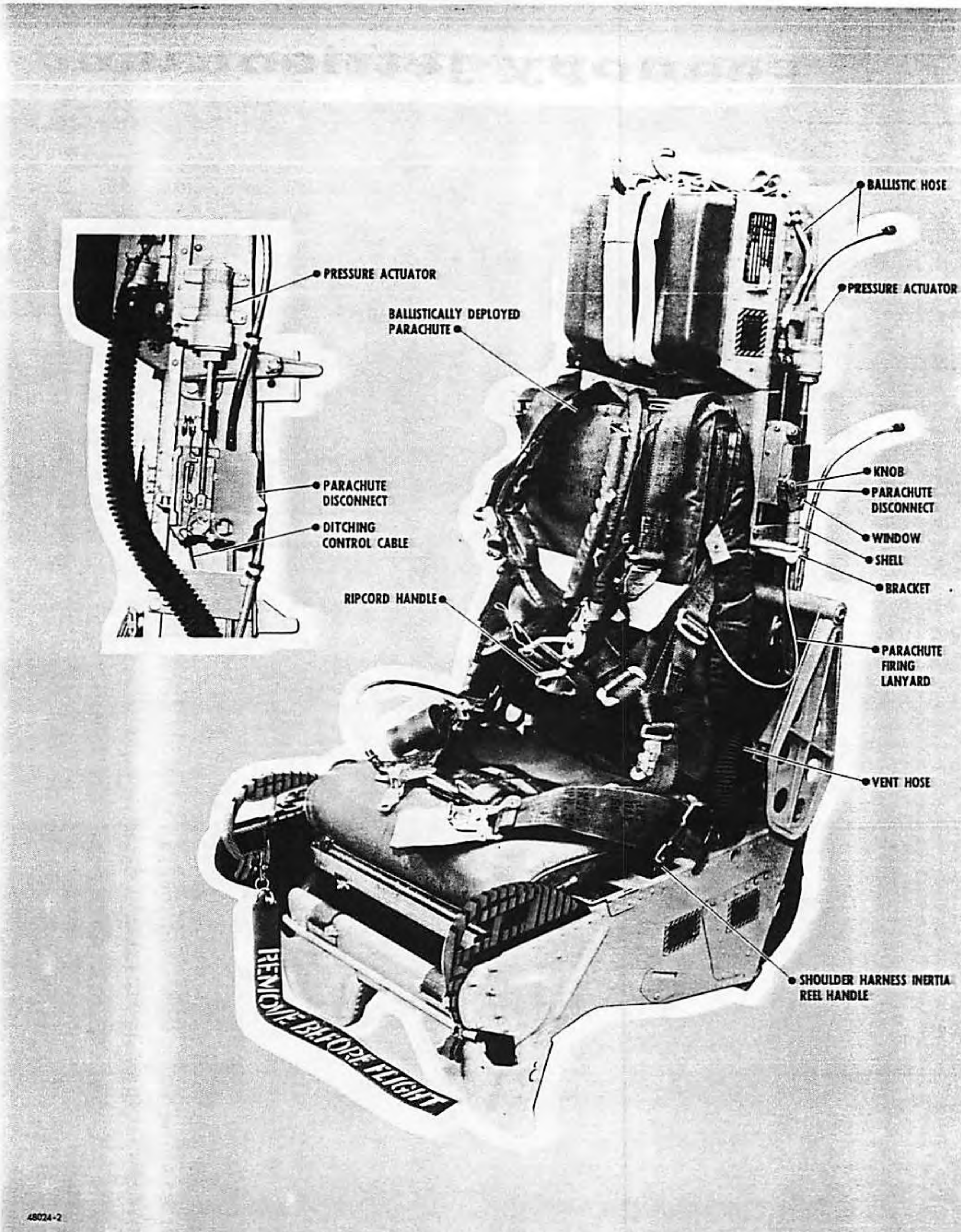

ejection seat
(typical)
QUICK DISCONNECT (F-106A, 1; F-106B, 2)
QUICK DISCONNECT (2)
MAINTENANCE SAFETY PINS
HEAD REST
SEAT-MAN SEPARATOR
SHOULDER HARNESS
CHAFF DISPENSER
ARM GUARDS
GROUND SAFETY PIN (REMOVE BEFORE FLIGHT)
PERSONAL LEADS
PARACHUTE HARNESS FASTENER
SAFETY BELT
SURVIVAL KIT
EJECTION SEAT HANDGRIPS (LEG GUARDS)
SURVIVAL KIT EMERGENCY RELEASE HANDLE
HANDGRIP (LEG GUARD) DOWNLOCK CABLE
EJECTION SEAT HANDGRIP (LEG GUARD) UPLOCK RELEASE LEVER
DITCHING CONTROL HANDLE
CANOPY JETTISON HANDLE
SEAT VERTICAL ADJUSTMENT SWITCH
RUDDER PEDAL ADJUSTMENT SWITCH
OXYGEN PRESS-TO-TEST BUTTON
GROUND SAFETY PIN (REMOVE BEFORE FLIGHT)
BEFORE FLIGHT

Figure 1-33


PRESSURE ACTUATOR
BALLISTICALLY DEPLOYED PARACHUTE
PARACHUTE DISCONNECT
DITCHING CONTROL CABLE
RIPCORD HANDLE
BALLISTIC HOSE
PRESSURE ACTUATOR
KNOB
PARACHUTE DISCONNECT
WINDOW
SHELL
BRACKET
PARACHUTE FIRING LANYARD
VENT HOSE
SHOULDER HARNESS INERTIA REEL HANDLE
REMOVE BEFORE FLIGHT
48024-2

canopy jettison and

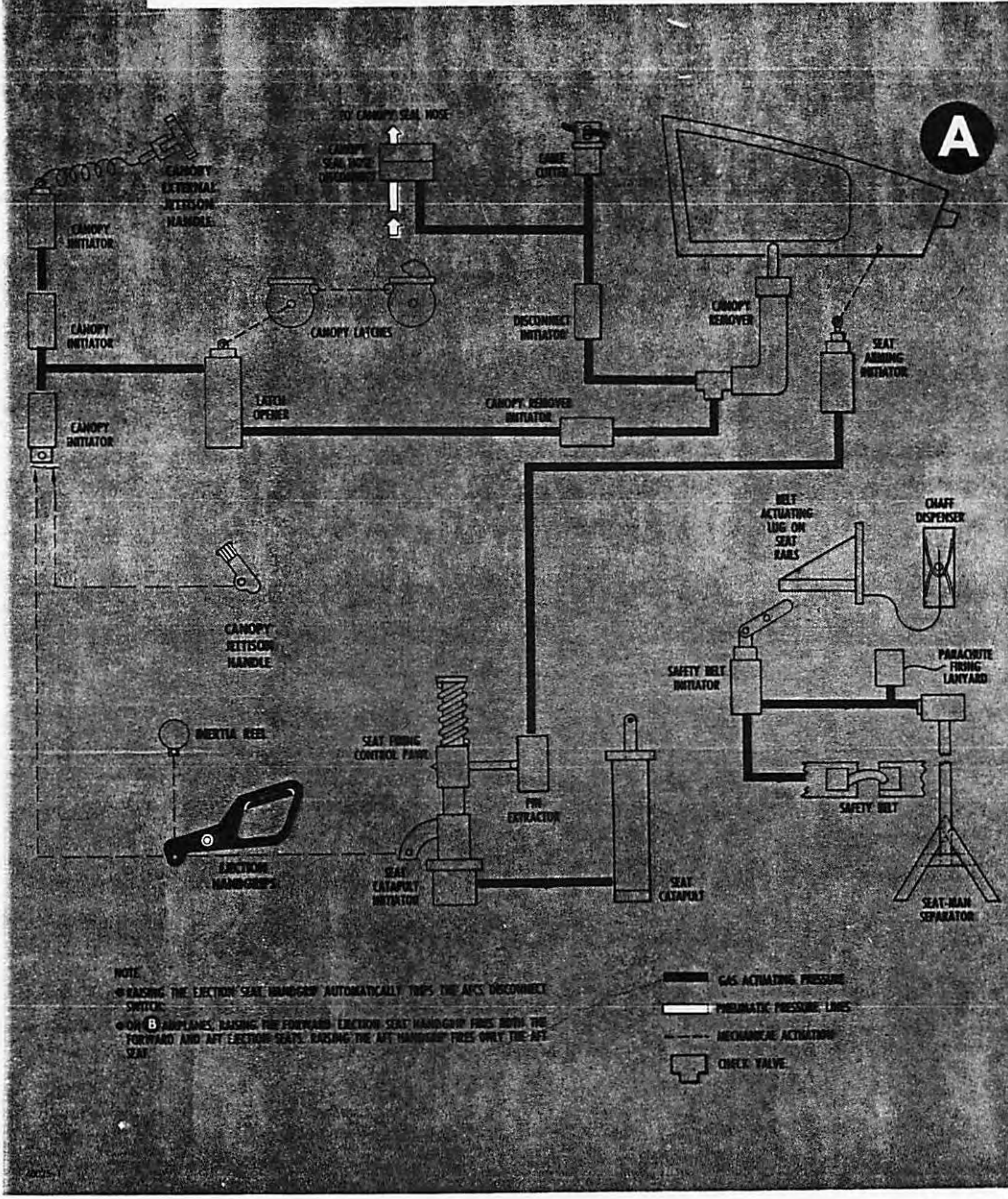


Figure 1-34

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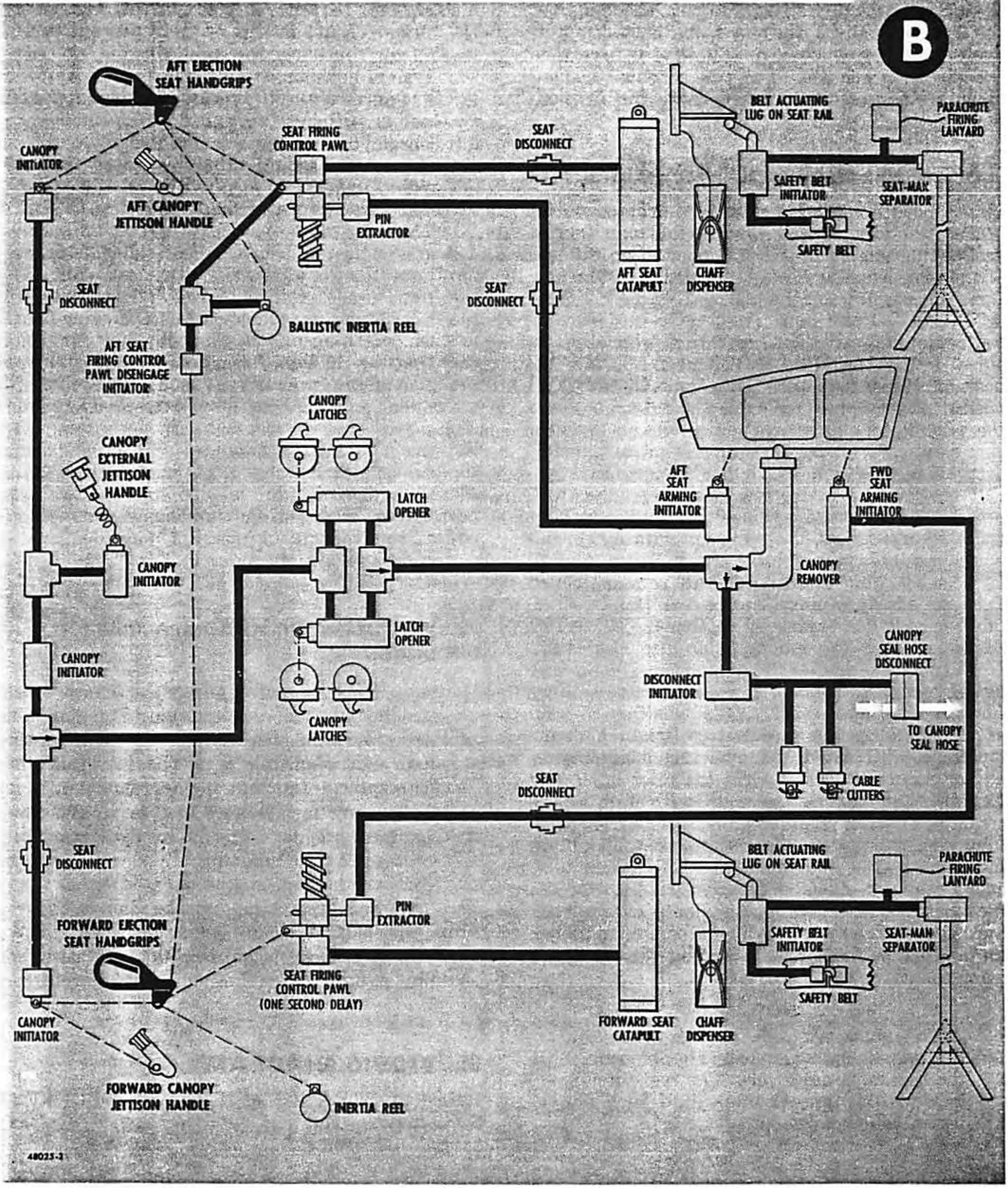
seat ejection system



SEAT VERTICAL ADJUSTMENT SWITCH

The three-position seat vertical adjustment switch (figure 1-33), located on the right side of the seat, has positions UP, DOWN, and is spring-loaded to the center (OFF) position. Holding the switch to the desired position energizes an electric motor-driven actuator which moves the seat vertically in the direction selected. The seat vertical adjustment system receives power from the dc non-essential bus.

SHOULDER-HARNESS INERTIA REEL HANDLE

Manual control of the shoulder-harness inertia reel is provided by the shoulder-harness inertia reel handle (figure 1-33), located on the left side of the seat. The handle is placarded "Inertia Reel Control Handle" and has MANUAL LOCK and AUTOMATIC positions. The handle incorporates overcenter detents which restrain it from slipping out of either MANUAL LOCK or AUTOMATIC position. When the handle is in the AUTOMATIC position, the reel harness cable will extend to allow the pilot to lean forward in the cockpit. Sudden forces applied by crash landing impact, turbulence, or rapid maneuvers, which tend to rapidly separate the pilot from the seat (including forward, upward, or sideward motion), will automatically lock the harness reel. The reel locks within approximately one-half inch of cable travel. When the reel is locked in this manner, it will remain locked until the handle is moved to the MANUAL LOCK position and then returned to the AUTOMATIC position. When the handle is in the MANUAL LOCK position, the reel harness cable is manually locked so that the pilot is prevented from bending forward. The MANUAL LOCK position is used as an added safety precaution when a crash landing is anticipated. If the harness is automatically or manually locked while the pilot is leaning forward, the harness retracts with him as he straightens up, moving into successive locked positions as he moves back against the seat. To unlock the harness, the pilot must be able to lean back enough to relieve tension on the lock. Therefore, if the harness is locked while the pilot is leaning back hard against the seat, he may not be able to unlock the harness without first loosening it slightly by using the adjustment buckles.

NOTE

A preflight check of the shoulder harness inertia reel can be made by simply giving the harness a quick jerk. The reel should lock within approximately half inch or less of cable movement.

EJECTION WARNING SYSTEM (B)

On (B) airplanes, an ejection warning system provides a visual reference to aid in assuring that the aft seat is ejected before the forward seat is ejected. This sequence of ejection (aft seat before the forward seat) is necessary to prevent rocket blast from injuring the pilot in the aft seat. The system consists of two red bailout warning lights (figure 1-30), one on each instrument panel; two bailout switches (36, figure 1-10), one on each left console; and one switch actuated when the aft seat is ejected. Placing either pilot-operated bailout switch in the ON position causes the bailout warning lights to flash, displaying "BAILOUT." When the aft seat is ejected, it automatically actuates a switch which causes the bailout warning lights to illuminate steadily. To signal for ejection, either pilot can move his bailout switch to the ON position, causing the bailout warning lights to flash. Flashing bailout warning lights indicate that the pilot in the aft seat should or will eject. The pilot in the forward seat should not eject until the bailout warning light illuminates steadily, indicating that the aft seat has left the airplane. If the pilot in the aft seat ejects without actuating the bailout switch, the bailout warning lights will illuminate steadily as soon as the aft seat has been ejected. The ejection warning system receives power from the battery bus. For additional information on this system, refer to T.O. 1F-106A-2-9.

PRESSURE ACTUATOR AND PARACHUTE DISCONNECT

A pressure actuator and a parachute disconnect are installed on the ejection seat. The pressure actuator transmits ballistic pressure from the seat-man separator into a mechanical function. This mechanical function is the retraction of a rod that in turn is connected to the link and claw in the parachute disconnect. The parachute disconnect contains the link and claw assembly which pulls the ball end of the parachute firing lanyard. This resultant retraction of the firing lanyard arms the barometrically controlled deployment gun. It then releases permitting the firing lanyard assembly to fall free of the parachute disconnect.

SERVICING DIAGRAM

Refer to STRANGE FIELD PROCEDURES, Section II.

Figure 1-35

normal procedures

Section II

48055

TABLE OF CONTENTS

PREPARATION FOR FLIGHT

FLIGHT RESTRICTIONS

Refer to Section V for operating restrictions and limitations.

FLIGHT PLANNING

Refer to the Appendix for information on required fuel, airspeed, thrust settings, etc., necessary to complete the mission.

TAKEOFF AND LANDING DATA CARD

Refer to T.O. 1F-106A-1-1 for takeoff and landing data card instructions, and complete cards in the Pilot's Abbreviated Checklist, T.O. 1F-106A-1CL-1.

WEIGHT AND BALANCE

Refer to Section V for weight limitations. For detailed loading information, refer to Handbook of Weight and Balance Data, T.O. 1-1B-40. Before each flight, check takeoff and anticipated landing gross weights and weight and balance clearance (Form 365F).

CHECKLISTS

The Flight Manual contains only amplified checklists; the abbreviated checklists have been issued as a separate technical order, T.O. 1F-106A-1CL-1. Refer to CHECKLISTS in the Introduction.

NOTE

The term "climatic," as used in the checklists, indicates equipment operation or settings which may be necessary for other than daylight VFR conditions. This includes IFR, night, cold weather, tropic, and desert conditions. The equipment operation or setting will vary depending on the prevailing conditions. In practice, the response to climatic items will be the required switch or control position.

PREFLIGHT CHECKS

It shall be the responsibility of the pilot to accomplish an interior and exterior visual inspection as outlined in the Flight Manual.

NOTE

- The visual inspection procedures in this Section are predicated on the assumption that maintenance personnel have completed all the requirements of the Manual of Scheduled Inspection and Maintenance Requirements, T.O. 1F-106A-6. Therefore, duplicate inspections and operational checks of systems have been eliminated, except for certain items required in the interest of flying safety.
- Checklist items coded (FP-RP) may be applicable to both Ⓐ and Ⓑ models but are coded to show the requirement for checking by the front pilot and the rear pilot in the F-106B. Items coded (RP) are the responsibility of the rear pilot.
- Actuation of all switch guards should be with a firm positive motion to assure that complete switch actuation will result.

ENTRANCE

The canopy is opened by the canopy external latch handle and canopy external switch which are reached through an access door on the left side of the fuselage below the windshield. After the canopy is opened, the cockpit is entered from the left side of the airplane by use of a ladder hooked over the left canopy sill. See figure 2-1.

BEFORE EXTERIOR INSPECTION

1. Form 781—Check.
   Check for engineering, servicing, and armament status. For detailed servicing requirements, see figure 2-8.
2. External power—OFF.
3. Canopy hold-open support(s) — In place.
4. Windshield and canopy—Check.
   Check that windshield delamination (marred windshield glass) does not extend more than two inches from the outer edge of the glass panels nor more than one inch from sectional tension straps. If delamination exceeds these limits, the windshield should be considered unsatisfactory for flight.
5. Ejection seat ground safety pin—Installed. (FP-RP)
6. Handgrip downlock cable—Check ball secure.
7. Canopy jettison handle—Down. (FP-RP)
8. Seat arming lanyard—Secured. (FP-RP)

WARNING

If seat arming lanyard is not properly attached to the canopy remover, jettisoning the canopy will not complete the cycle to eject the seat.

8A. Personal Leads bundle—Secure. (FP-RP)

9. Ejection seat quick-disconnects and ballistic hoses—Check. (FP-RP)
   Check quick-disconnects on the left and right ejection seat rails to see that the red circle around the disconnects is not exposed. Check ejection seat ballistic hoses for a smooth continuous curve from the quick-disconnects to the clamp on the seat, and see that the hoses are not twisted, flattened, or kinked.
10. Canopy support — Check for security (if installed).
    If canopy support is not installed, insure that the tie-down straps are secured to the brackets to prevent interference with ejection sequencing.
11. Canopy, survival kit and seat ground maintenance safety pins — Removed. (FP-RP)

WARNING

If any ejection system ground maintenance safety pins are installed, do not remove them until the status of the ejection system has been checked with maintenance personnel.

12. Canopy breaker tool—Check for security. On Ⓑ airplanes front cockpit, the tool is attached to the bulkhead behind the ejection seat on the right side. On Ⓑ airplanes rear cockpit and all Ⓐ airplanes, tool is located above left console.
13. Inertia reel cable and survival kit lanyard —Secure.
14. Lap belt connectors—Secure.
15. Oxygen bailout bottle gage—1800 psi minimum. (FP-RP).
16. Parachute—Install in airplane. (FP-RP)
    a. Parachute—Place in seat.
    b. Ditching control handle—Stowed (forward).

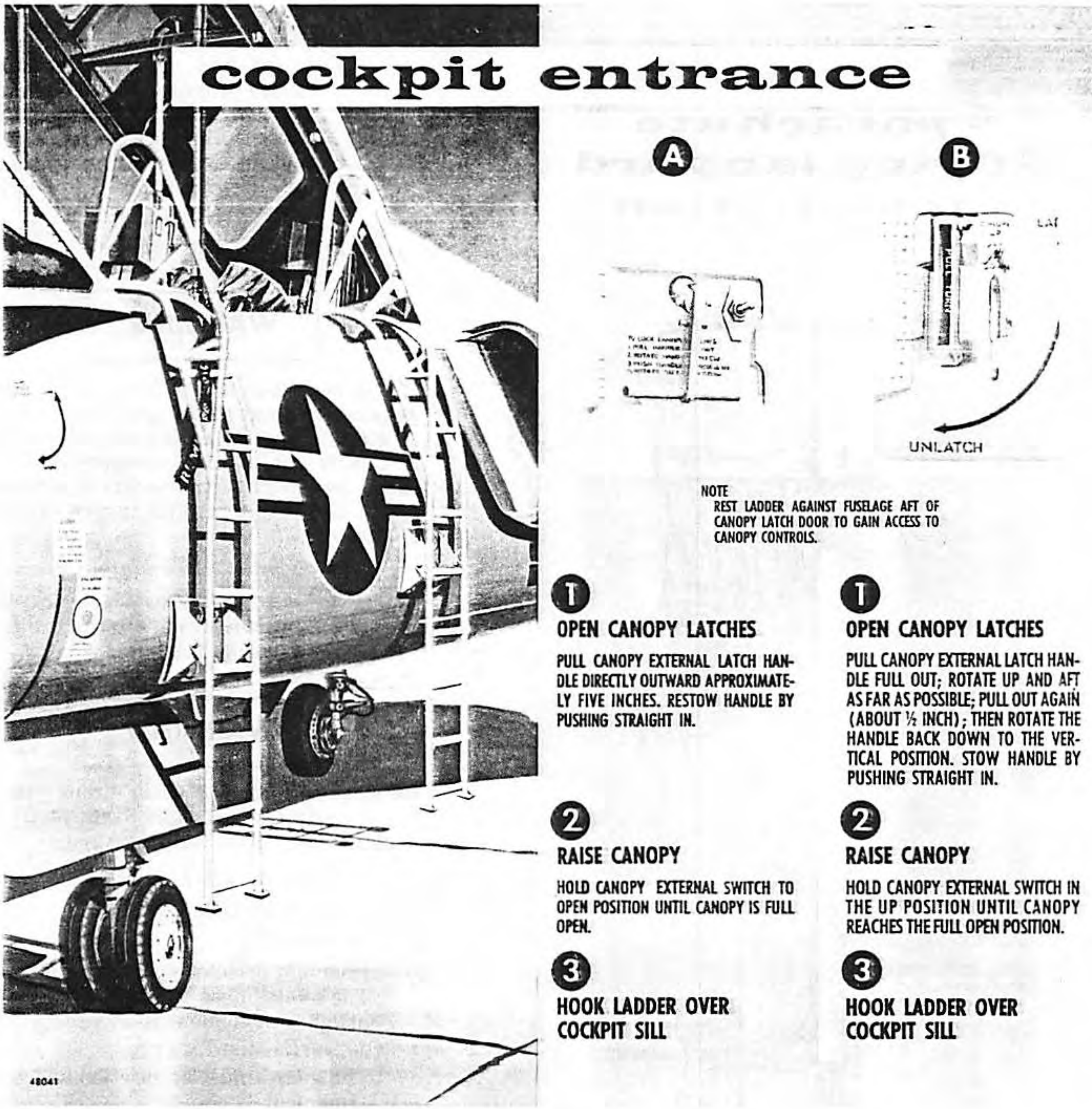


Figure 2-1

c. Parachute firing lanyard—Install as follows:

(1). Knob — Unscrew and open parachute disconnect door (ensure that jaw is reset).

(2). Dust cap—Remove from firing lanyard housing.

(3). Safety pin — Remove from firing lanyard.

(4). Firing lanyard shell—Pull down to expose the cone end of the lanyard.

WARNING

The firing lanyard must be handled with care at all times to prevent the ball from being pulled away from the shell causing an inadvertent firing. Excessive movement of the ball will fire the deployment gun and deploy the parachute. Firing of the deployment gun in the cockpit can cause injury or loss of life to personnel working around the airplane. It can also damage the cockpit and canopy.

parachute firing lanyard installation

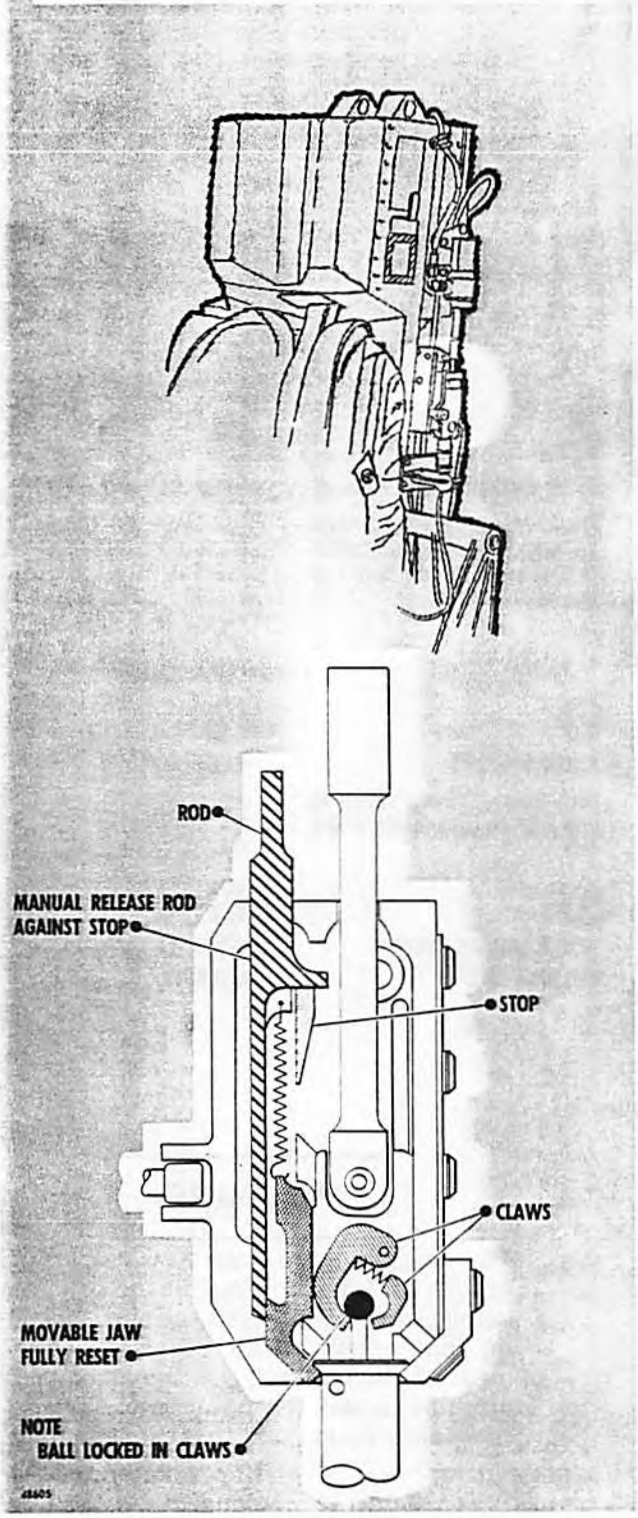


Figure 2-2

(5). Cone end of lanyard housing—Place in position in parachute disconnect (figure 2-2) and ensure that ball is between jaws of claw, manual release rod is against stop, and that movable jaw is locked in reset position.

(6). Shell—Release.

WARNING

It is imperative that the firing lanyard be installed in front of the arm guard pivot tube and aft of the cable guide assembly. If this is not installed properly, it can cause an abortive man-seat separation which can result in pilot injury or loss of life.

(7). Door—Close and tighten knob.

(8). Ball and claw—Visually inspect through window to assure that installation is correct.

WARNING

Be extremely careful when either entering or exiting the cockpit to prevent engaging the ejection seat handgrip.

17. MA-1 power switch—OFF.
18. RAT handle—UP.
19. Throttle—OFF.
20. Armament selector switch — VIS IDENT. The guard must be safetied and sealed if primary armament is aboard.
21. Arm-safe switch—SAFE. The guard must be safetied and sealed if armament is aboard.
22. Special weapon release lock switch—LOCK. The guard must be safetied and sealed if primary armament is aboard.
23. Special weapon release lock indicator—Striped.
24. Master switch—ON; Battery—Check. Check warning lights for illumination.
25. AIR-2A Arm/Safe/Monitor power circuit breaker—Open.
26. Master electrical power switch—OFF.
27. Emergency ac generator switch—START, momentarily, and check illumination of ac power fail warning light; then release.
28. Landing gear handle—Down.

29. Emergency landing gear extension handle —In and secured.
30. Optical sight—Stowed.
31. All fuses—In. (FP-RP)

    Check that all fuses are installed in the left console, the right console, and the right rudder well.
32. Slipway door—Closed (if installed).

Aft Cockpit (Solo Flights) B

If the flight is to be solo, the following inspection of the aft seat, consoles, and instrument panel must be made before entering the forward seat:

B 1. Ejection seat ground safety pin — Installed.

NOTE

With the ground safety pin installed, the ejection seat will sequence normally from the forward cockpit.

B 2. Handgrip down lock cable — Check ball secure.

B 3. Canopy jettison handle — Down.

B 4. Ejection seat disconnect and hoses—Check.

Check quick-disconnects on the left and right ejection seat rails to see that the red circle around the disconnects is not exposed. Check ejection seat ballistic hoses for a smooth continuous curve from the quick-disconnects to the clamp on the seat, and see that the hoses are not twisted, flattened, or kinked.

B 5. Canopy and seat maintenance pins— Removed.

WARNING

If any ejection system ground maintenance safety pins are installed, do not remove them until the status of the ejection system has been checked with maintenance personnel.

B 6. Survival kit—Remove or secure.

B 7. Personal equipment leads and all loose items —Stow.

B 8. Safety belt and shoulder harness—Secure.

B 9. Fuel shutoff switch — NORMAL.

B 10. Variable ramp switch — AUTO.

B 11. Throttle quadrant dust cover—Remove and stow.

B 12. Fuel control switch—NORM.

B 13. Throttle — OFF.

Ensure throttle is in the fully inboard OFF position.

B 14. Oxygen supply switch—OFF.

B 15. Master electrical power switch — ON.

B 16. Landing gear handle — DOWN.

B 17. Landing gear emergency extension handle — In and secure.

B 18. Flight mode selector switch — DIR MAN.

B 19. Drag chute handle—In.

B 20. TSD light intensity rheostat—OFF.

B 21. TSD mode selector switch—MAN.

B 22. Cockpit lights—Off.

B 23. Fuse panel—Check.

B 24. UHF—off.

EXTERIOR INSPECTION

Perform the following checks in accordance with figure 2-3.

NOTE

While checking the items listed below, check that doors and inspection plates are closed. Check for fluid leaks, indications of defective or insecure installations, and the general over-all appearance of the airplane.

A. Forward Left Side

1. Canopy external handle access door— Secured.
2. Aircraft placarding—Check for armament status.
3. Static ports—Clear.
4. Oxygen filler valve access door—Secured.
5. Nose wheel well door hinges—Check.
6. Forward electronics bay door—Secured.
7. Angle of attack transducer vane—Condition, guard removed, free, and gently placed in full up position. Leave the vane in the full up position in order to check instrument during interior inspection.

NOTE

- The angle of attack transducer vane must be handled with care to prevent internal damage to the transducer.
- The angle of attack transducer vane may be hot and can burn the hand if touched.

B. Nose

1. Radome — Condition and secure.
2. Mast and pitot tube — Condition and cover removed.

C. Forward Right Side

1. Forward electronics bay door — Secured.

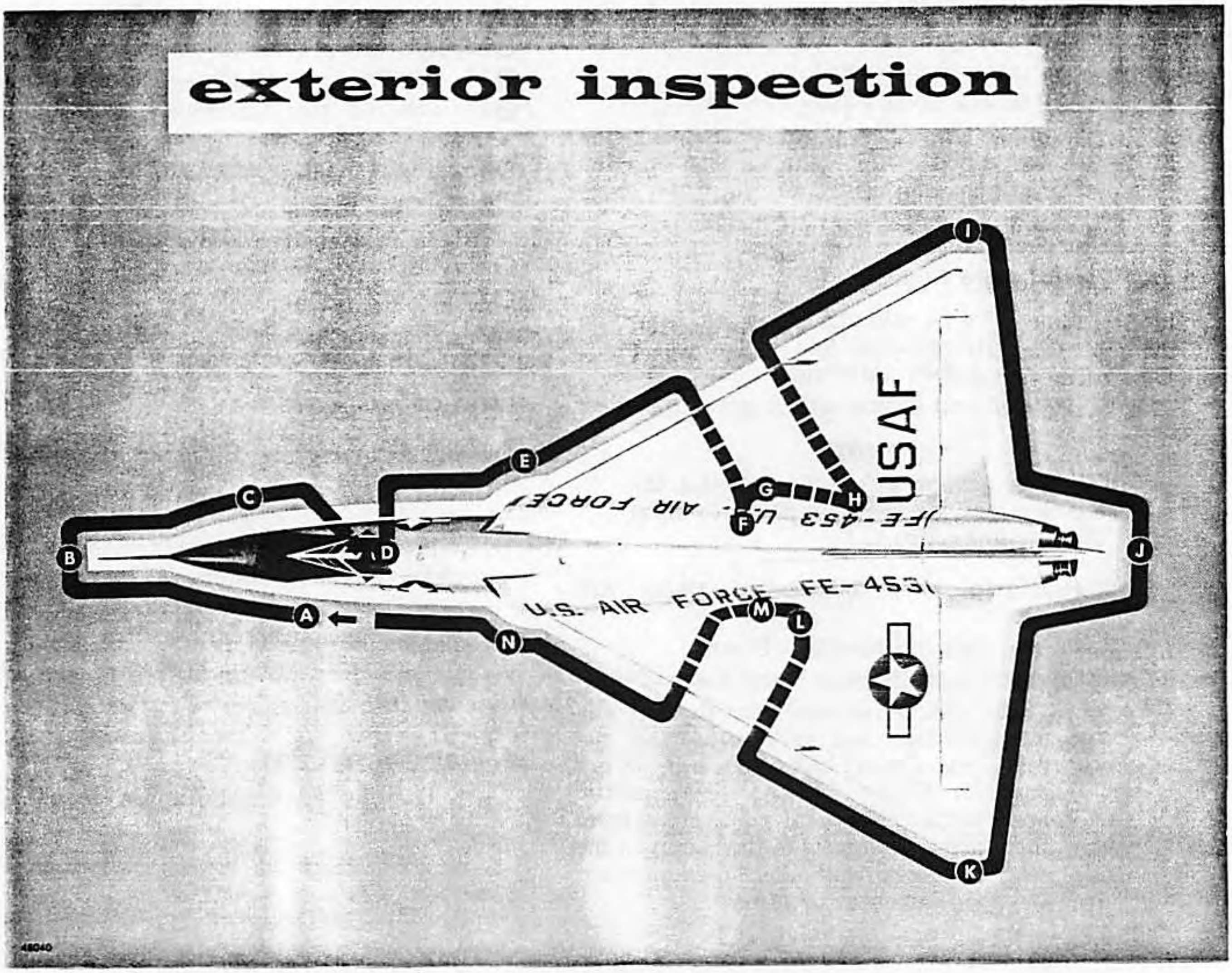


Figure 2-3

2. Static ports — Clear.
3. Direction finder antenna — Secured.

D. Nose Wheel Well

1. Taxi light — Condition and security.
2. Nose gear door seal — Condition.
3. Control outflow valve (cabin pressure regulator) — FLIGHT position and safetied.
4. Wheel brake reservoirs — Checked (some airplanes).
5. Batteries — Secured.
6. IR circuit breaker — In.
7. Fuses and circuit breaker — Check.
8. Nose gear ground safety pin — Remove.
9. Nose wheel steering unit ground pin — Removed.
10. Scissors linkage — Connected.
11. Strut extension — 4 to 5 inches.
12. Tires — Check.
    Check for wear, cuts, inflation, and valve cap installed.

E. Right Side

1. Temperature probe — Condition and clear.
2. Lower aft electronics bay door — Secured.
3. B Lower mid-electronics bay door — Secured.
4. Missile bay door — Secured.
5. A Upper aft electronics bay door — Secured.
6. Air-conditioning compartment access door — Secured.
7. Boundary layer duct — Clear.
8. Variable ramp — Condition and retracted.
9. Intake duct — Condition and loose articles.
10. Fuel filler cap and access door — Secured.
    Check that fuel credit card is installed inside access door.

F. Hydraulic Compartment

1. RAT door—Check that pressure is relieved.
2. RAT rotor and blades—Condition and turns freely with slight drag.
3. Primary and secondary hydraulic accumulator pressure—750 psi.
4. Primary and secondary hydraulic fluid levels—Not more than 3/4-inch below the full mark corresponding to temperature on reservoir temperature gage.
5. Reservoir pressure gages—Check approximately 55 psi.
6. Reservoir pressure shutoff valve—Open and pin installed.

CAUTION

The reservoir pressure shutoff valve shall be locked in the OPEN position at all times except when hydraulic system maintenance is being performed.

7. Lower anticollision light—Condition.
8. Hydraulic access compartment door—Secured.

   Have the ground crew secure the hydraulic access compartment door.

G. Right Main Wheel Well

1. Gear ground safety pin—Remove.
2. Refuel selector valve—Horizontal.
3. Armament control panel—Check.
4. Armament lock valve—FLIGHT position.
5. Hydraulic and fuel lines—Check.
6. Fuses—Check.
7. Starter ignition disarm switch—ON.
8. AC generator exciter—Check.

   Check data plate showing on outboard edge of forward side.
9. Engine ignition disconnect switch—ARM.
10. Brake and hydraulic lines—Check.
11. Strut extension—5 to 6 inches.
12. Deleted.
13. Landing gear fairing, door, and light—Condition and security.
14. Tire and chocks—Check.

    Check for wear, cuts, inflation, valve caps installed, and chocks in place.

H. Right Engine Access Compartment

1. Magnetron hydraulic system accumulator—1200 to 1500 psi.
2. Magnetron hydraulic system quantity gage—Full for specified level.

I. Right Wing

1. External wing tank—Check (if installed).

   Check external wing tank for security, fuel quantity, cap and pylon access doors secure. Ground safety pin - remove.
2. Fuel ambient sense and vent ports—Clear.
3. Wing, condition and position lights—Check.
4. Elevon actuator fairing—No hydraulic leaks.
5. Trailing edge and elevon—Condition.
6. CSD oil filler cap—Secured.

J. Tail Section

1. Ram air "q" intake covers—Removed.
2. Rudder and position light—Condition.
3. Speed brakes and drag chute—Check speed brakes condition and ground safety locks removed. Check drag chute stowage and release pin in place; upper drag chute retaining strap attached last.
4. Tailpipe and exhaust nozzle—Check for fuel and condition.

CAUTION

Check for fuel puddles at drain lines and in the tailpipe, as unburned fuel creates a fire hazard.

5. Tailhook—Check and remove ground safety pin.

   Check tailhook for security.
6. Data link antenna—Check for condition.

K. Left Wing

1. Trailing edge and elevon—Condition.
2. Elevon actuator fairing—No hydraulic leaks.
3. Wing condition and position lights—Check.
4. Oil cap access door—Secured.
5. Fuel ambient sense and vent ports—Clear.
6. External wing tank—Check (if installed).

   Check external wing tank for security, fuel quantity, cap and pylon access doors secure. Ground safety pin remove.

L. Left Engine Access Compartment

1. Hydraulic lines, fuel lines, and throttle linkage—Check general condition and safetywired.

2. Engine access compartment door—Open.
   Alert airplanes may have door secured. Ground crew will normally secure engine access compartment door after start.

M. Left Main Wheel Well

1. Gear ground safety pin — Remove.
2. External air — Connected.
3. Emergency ac generator — Condition and leaks.
4. Fuses—Check.
5. Hydraulic and fuel lines—Check.
6. Combustion starter manual air shutoff valve—CLOSED and saftied (if external air source is available).
7. Pneumatic system pressure gage—2000 to 3000 psi.
8. Missile bay doors—Manually opened (slow).
9. Missile bay—Check for armament.

CAUTION

- Extreme caution should be exercised when checking equipment in the missile bay area.
- Avoid movement of any of the door control valve indicator pins, as damage to the missile bay doors will result.

10. General condition of rocket—Check.

NOTE

If the rocket motor safety pin is installed or other discrepancies exist, the discrepancies should be reported to the ground crew.

11. General condition of missiles—Check.
12. Missile bay doors—Closed.
13. Pneumatic system pressure gage—3000 psi.
14. Engine hot-section analyzer data recorder —Reset to preflight condition.
15. Brake and hydraulic lines—Check.
16. Strut extension—5 to 6 inches.
17. Deleted.
18. Landing gear fairing, door, and light — Check condition and security.
19. Tire and chocks—Check.
    Check for wear, cuts, inflation, valve cap installed, and chocks in place.
20. RAT door and test hook—Unlock and close by gradual increase in force. Pull test hook to check door security.

N. Left Side

1. Missile bay door — Condition and secured.
2. Emergency canopy jettison access lanyard — Check.
   Check that lanyard is stowed and access door secure.
3. Boundary layer duct — Clear.
4. Variable ramp — Secure and retracted.
5. Intake duct — Condition and loose articles.
6. Air-conditioning compartment access door — Secured.
7. Upper aft electronics bay door—Secured.

INTERIOR INSPECTION

NOTE

Items marked with the symbol ▲ preceding the step cannot be performed if making the interior inspection with battery power prior to battery start. These items (▲) should be checked after battery start. (Refer to INTERIOR INSPECTION AFTER BATTERY START, this Section.)

General

1. Personal equipment, safety belt, and shoulder harness — Attach and adjust (as required). (FP-RP)
   The oxygen hose and communications lead should be routed under the right shoulder harness strap.

WARNING

- The force required to open the safety belt should not be less than five pounds pull nor exceed 20 pounds pull to insure proper operation.
- The survival kit attaching straps must be snugged up closely after the kit is hooked to the parachute harness. During ejection, if the kit is allowed to hang too far below the harness, it may swing up behind the parachute after seat separation and prevent or slow deployment of the parachute. Check routing of kit straps to assure that they are not between safety belt and ballistic hose.

▲ 2. External power—Connected (if available).
   If external electrical power is not available, turn the master electrical power switch ON.

▲ 3. Seat and rudder pedals—Adjust. (FP-RP)

Left-Hand Console

1. Cabin air selector handle — Adjust for vertical outlet. (FP-RP)
2. MA-1 test panel cover — Closed.

Ⓑ 3. Intercom volume control knob—As desired. (FP-RP)

Ⓐ 4. Emergency slipway door open switch (if installed)—NORM (guard closed).

Ⓐ 5. Refuel select switch (if installed) — ALL TANKS (guard closed).

Ⓐ 6. Air refuel switch (if installed) — OFF (guard closed).

▲ 7. Armament recycle button — Depress for 5 seconds.

If the missile intervalometer is not reset by the ground crew or the pilot, the armament bay doors will not open on a missile attack. The armament selector switch must be in VI before reset will occur.

8. Variable ramp switch — AUTO (guard closed). (FP-RP)
9. Fuses (LH panel)—Check.
10. Fuel shutoff switches—OPEN.

Check that the fuel shutoff valve warning lights (on the fuel control panel) and fuel valve closed warning light (on the master warning light panel) are out. Press to test fuel valve indicator lights for illumination.

Ⓑ 11. Fuel shutoff switch—NORM. (RP)

▲ 12. Boost pump switches — Check, then ON.

Turn fuel boost pump switches ON, one at a time, then OFF. Check that respective fuel boost pump warning light goes out when pump is operating, then turn all fuel boost pump switches ON.

CAUTION

If the warning light does not extinguish and/or the warning light extinguishes on the opposite wing, the cause should be established and corrected prior to flight.

▲ 13. Fuselage tank emergency pressure or boost pump switch—OFF.

14. MA-1 power switch—Recheck OFF.
15. Gyro grid reference knob — Set

NOTE

The grid reference heading is obtained by adding the correction angle, obtained on the local 50 mile radius map, to the aircraft magnetic heading.

16. SIF code selector wheels—Set.
17. IFF control panel—Set (as required).
18. RAT handle—UP.
19. Throttle quadrant dust cover—Remove and stow.
20. Fuel control switch—NORMAL. (FP-RP)
21. Throttle—OFF. (FP-RP)
22. Speed brakes switch—Center (off). In for cocking.
23. UHF function selector switch—BOTH.
24. UHF radio frequency—Set. (FP-RP)
25. Special weapon armed light—OFF, press-to-test.

WARNING

If the special weapon armed warning light is illuminated, immediately remove electrical power and reject the airplane (if primary armaments aboard).

26. Armament selector switch — VIS IDENT.

The guard must be safetied and sealed if primary armament is aboard.

27. Arm-safe switch—SAFE.

The guard must be safetied and sealed if armament is aboard.

28. Special weapon release lock switch—LOCK.

The guard must be safetied and sealed if primary armament is aboard.

▲ 29. Special weapon release lock indicator—"LOCK," (if primary armament, ATR, or MSR is aboard).

30. Armament selection indicator—"NO."
31. AIR-2A Arm/Safe/Monitor power circuit breaker—Open.
32. ILS channel selector switch—Set as desired, volume at minimum.
33. Cockpit no-fog and ventilated suit switch—As desired.

▲ 34. Landing and taxi light switch—Check.

35. Reset/MBL switch (if installed)—NORM.

Ⓑ 36. Bailout light switch—Check. (FP-RP)

Ⓐ 37. CG control switch — AUTO (guard closed).

▲ Ⓐ 38. CG transfer test failure light—On (with full internal fuel). Press-to-test (with reduced internal fuel).

39. Idle thrust control switch — OFF (if installed.

40. Master electrical power switch — Recheck OFF (if starting with external power).

41. Oxygen system — Check as outlined in Section IV. (FP-RP)

42. RDR/IR control panel - Set.
    a. AZ SCAN Switch - Broad.
    b. LOBE FREQ Switch - As desired.
    c. IR Video knob - Fully ccw.
    d. IR Tone knob - Fully ccw.
    e. IR Volume knob - Fully ccw.
    f. EL SCAN Switch - Norm.

43. Landing gear handle — Recheck DOWN. (FP-RP)

44. Landing gear emergency extension handle —In and secure. (FP-RP)

Instrument Panel

1. Flight mode selector switch — DIR MAN. (FP-RP)
2. Heading hold switch—OFF. (FP-RP)
3. Altitude hold switch—OFF. (FP-RP)
4. DISP/AUTO MODE switch - ILS.

   Selection of ILS will clear the computer of Data Link. The MCC target velocity will be erased and cannot affect the firing bar position. The radar antenna will slave dead ahead and full down. The computer will insert initial position if TACAN is not locked on in range and bearing.

5. Clock — Set. (FP-RP)
6. TACAN range indicator light—Push-to-test (if installed).
7. Drag chute handle — In. (FP-RP)
8. Landing gear position lights — On. (FP-RP)
9. Landing gear warning light — OUT. (FP-RP)
10. External tank empty lights (if installed) —Check.
    Check lights out if external tanks are full; press-to-test.
11. Computer mode indicator — Striped. (FP-RP)
12. Deleted.

12A. Radar scope controls - Set.
   a. Erase intensity knob - Set at 10 o'clock position (90° from full cw).
   b. Attack intensity knob - Fully cw.
   c. Dimmer knob - As required.
   d. IF gain knob - Fully cw.
   e. Video gain knob - Fully cw.

13. Engine fire warning loops—Test.

▲ 14. Marker beacon light—Press-to-test. (FP-RP)

15. Variable ramp warning light—Press-to-test (if installed).

16. Warning lights — On (Canopy unlocked, master warning, hydraulic pressure-low). (FP-RP)

17. Engine instruments — Check. (FP-RP)
   Check tachometer, fuel flow indicator and engine pressure ratio gage for zero indication, and the pressure ratio set. The exhaust gas temperature gage should read ambient temperature if engine is cold.

18. EGT power flag — Not displayed.

▲ 19. Fuel quantity — Check.
   Select FWD, LH, RH, #3 and TOT positions with the fuel quantity gage switch(es) and check the fuel quantity gage for proper fuel quantities. Leave switch in TOT position.

NOTE

No. 3 tank fuel quantity cannot be obtained with the fuel quantity gage selector switch in the FWD position.

20. RDR/IR selector panel - Set.
   a. Range selector switch - 4 miles.
   b. IR stow switch - RDR SCAN.
   c. Tune switch - SNIFF.
   This insures that radar will be working outside the Paramp frequency band. Scope video checks can be evaluated with less external noise in this position.
   d. Nose/Tail switch - Nose.
   e. Chaff switch - OFF.

21. Oil Quantity Gage - Indicating normally.
   a. Press to test warning light on
   Ⓑ aircraft.

22. Bearing selector switch — NORM (if installed). (FP-RP)

23. Heading selector switch — NORMAL (if installed). (FP-RP)

24. TSD controls — As desired. (FP-RP)
   Place OFF BRT/CONTROL
   to first detent.

Right-Hand Console

1. Hydraulic pressure gages — Check.
2. Oil pressure gage — Check.
3. Refrigeration unit switch — ON.
4. Cabin air selector switch — OFF.
5. AC generator switch — OFF.
6. DC generator switch — OFF.
7. TACAN function selector knob (if installed)—As desired.
8. TACAN mode selector switch (if installed) — As desired.
9. TACAN range selector switch (if installed) — As desired.
10. TACAN channel—Select.
11. TACAN volume control knob — Minimum volume.
12. TACAN ECM indicator light—Out.
13. Homing point selector - Not "A", "T", o. "U".
   This step insures that data from the computer will not be called up and the test pattern can be evaluated proper

14. Data link antenna selector switch—NORM.
15. Master warning lights — Check. (FP-RP)
    Depress warning lights test button and check that all lights in the master warning system illuminate.

NOTE

The following warning lights will normally be illuminated prior to depressing the master warning lights test button: ac power failure, dc power failure, emergency fuel on, left and right fuel tank pressure-low (left and right fuel boost pressure-low, if fuel boost pumps are off), engine anti-ice, oil pressure-low, and flight mode failure.

16. ATG switch — OFF.
17. Canopy latch handle — Unlock (fully aft).
18. Map reading light — Check (if required).
19. Windshield anti-icing, antifog switches — ON.
20. TACAN-command altitude switch — As desired (if installed).
21. Emergency ac generator test switch — NORMAL.
22. TACAN power switch — NORMAL (if installed).
23. Data Link panel - Set (as required).
    a. TEST ADDRESS/DISPLAY Switch - TEST ADDRESS.
    b. Data link channel - Select.
    c. ADDRESS SELECT - Set numerical call sign.
24. Rain removal switch — OFF.
▲ 25. Thunderstorm lights switch — Check, then as desired.
26. Warning lights dimmer switch—As desired.
▲ 27. All external and internal light switches — Check and set, as required. (FP-RP)
    Turn lights on and confirm proper operation and dimming with aid of crew chief, then turn off if not required.

NOTE

The anticollision lights will illuminate when the formation-navigation lights switch is in the NAV ON position. Use of the anticollision lights on the ground shall be kept to an absolute minimum. The excessive heat created on the ground is detrimental to bulb life. During ground emergencies the operating light could confuse rescue operations since emergency ground vehicles use a similar light.

28. Formation navigation lights switch — FORM ON.
▲ 29. Pitot heat switch — Check, then OFF (ON for cocking).
30. Canopy antifog switch — Climatic.
    Place canopy antifog switch ON if required to electrically heat canopy glass panels.
31. Surface and engine anti-icing switch — AUTO ON.
32. Cabin temperature control knob — AUTOMATIC.
33. Compass control panel — Set.
    a. Latitude — Set.
    b. Function selector switch — SLAVED (MAG some airplanes).

CAUTION

The synchronization knob may be used at any time to obtain synchronization, but should not be operated for more than 30 seconds at a time to prevent overheating of the slewing motor.

Ⓑ 34. Air refuel switch (if installed) — OFF (guard closed).

Ⓑ 35. Refuel select switch (if installed) — ALL TANKS (guard closed).

Ⓑ 36. Emergency slipway door open switch (if installed) — NORM (guard closed).

37. Fuses — Check. (FP-RP)

38. UHF communication circuit breaker switch (if installed) — On (up position).

BEFORE STARTING ENGINE

An external electrical power source may be connected for the starting procedure. In addition, an external air pressure source may be used to pressurize the hydraulic reservoirs, thereby providing for subsequent system check after start has been accomplished. On Ⓑ airplanes engine start is made from the front cockpit.

CAUTION

Before starting engine, determine that wheels are firmly chocked, and hold wheel brakes on. Parking brakes are not installed on the airplane.

WARNING

Determine that danger areas fore, aft, and under the airplane are clear of personnel, aircraft, and vehicles. Refer to figure 2-4. Suction at intake ducts is sufficient to kill or seriously injure personnel pulled against or drawn into the ducts. The danger area aft of the airplane is created by the exhaust velocity and temperature.

STARTING WITHOUT GROUND SUPPORT EQUIPMENT

An outside air and electrical power source is normally used to start the engine. However, the start may be accomplished by using the airplane battery and either an external air pressure source or the airplane high-pressure pneumatic system.

NOTE

- All preflight check items that require electrical power should be checked after the engine has been started. Refer to After Battery Start, this Section.
- A combustion start, utilizing the airplane's air supply, is dependent upon air stored in the main system flasks and can be satisfactorily accomplished if pressure is at least 1500 psi.
- If the battery start is aborted for any reason and a second start is desired, it will be necessary to allow at least 30 seconds for the starter fuel flask to refill prior to the second start.
- After throttle is moved to idle, place emergency ac generator switch to START.

PNEUMATIC START

Combustion starts are normally used for engine starting. However, electrical power can be interrupted to the starter ignition system and the engine can be pneumatically started without firing the combustion part of the starter system.

1. Starter ignition disarm switch — OFF.

   Have the ground crew place the starter ignition disarm switch OFF.

NOTE

- Pneumatic starts will be used only after failure to obtain a combustion start.
- An external compressed air source must be used for all pneumatic starts.

2. Use normal start procedure.

NOTE

The throttle should not be advanced from OFF to IDLE until rpm reaches a minimum of 10%, to prevent excessive exhaust gas temperatures.

STARTING ENGINE

NOTE

Items marked with the symbol ▲ preceding the step cannot be performed if starting with the battery.

1. Clear to start — Check. (FP-RP)

WARNING

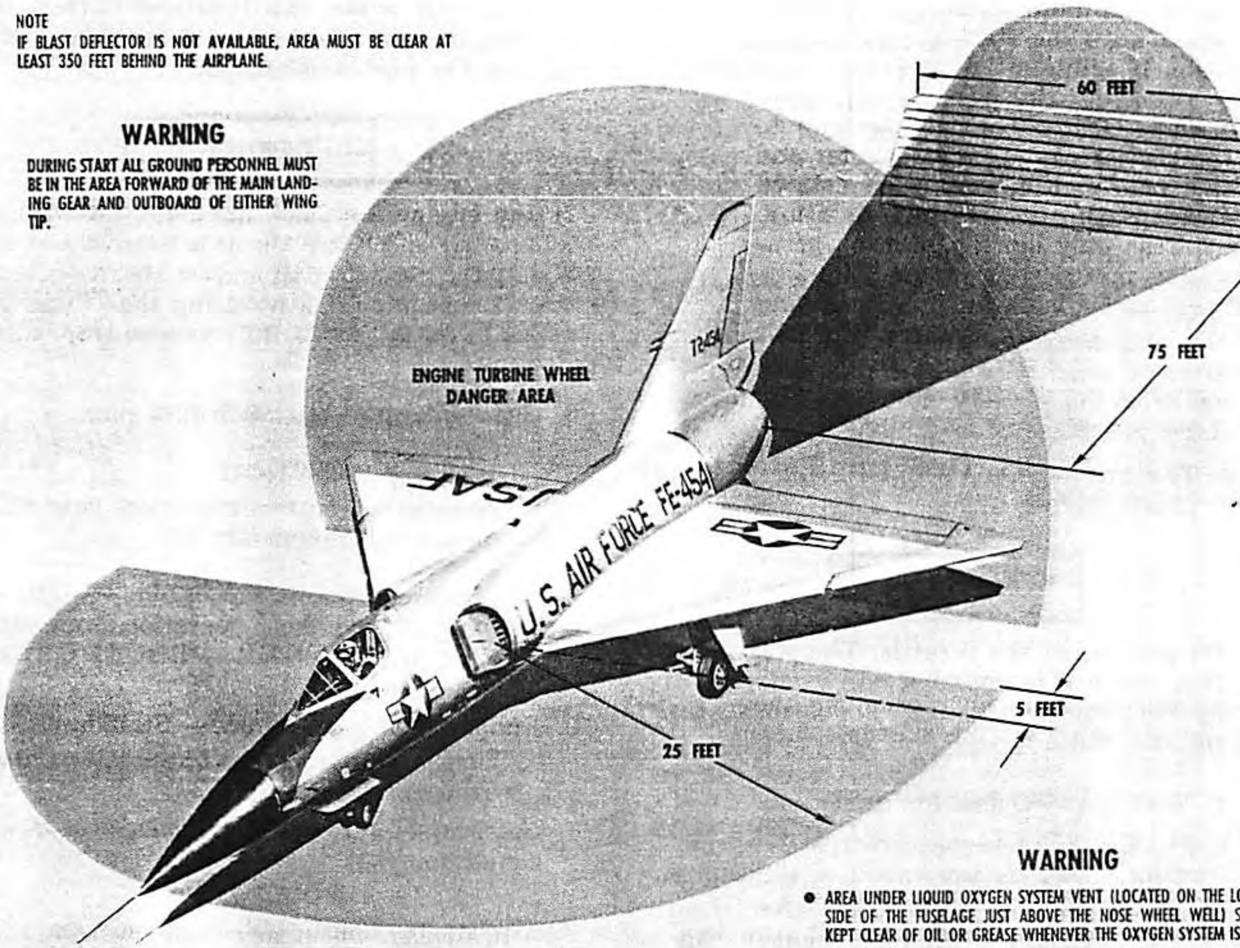
Prior to any ground start make sure that all ground personnel are forward of the main landing gear and outboard of either wing tip (figure 2-4) before moving the throttle out of the OFF position. Ground crew should not re-enter the wheel well area until the starting cycle is completed.

danger areas

NOTE
IF BLAST DEFLECTOR IS NOT AVAILABLE, AREA MUST BE CLEAR AT LEAST 350 FEET BEHIND THE AIRPLANE.

WARNING

DURING START ALL GROUND PERSONNEL MUST BE IN THE AREA FORWARD OF THE MAIN LANDING GEAR AND OUTBOARD OF EITHER WING TIP.



WARNING

- AREA UNDER LIQUID OXYGEN SYSTEM VENT (LOCATED ON THE LOWER LEFT SIDE OF THE FUSELAGE JUST ABOVE THE NOSE WHEEL WELL) SHOULD BE KEPT CLEAR OF OIL OR GREASE WHENEVER THE OXYGEN SYSTEM IS SERVICED.
- VARIABLE RAMP EMERGENCY OPERATION WILL EJECT HYDRAULIC FLUID WITH CONSIDERABLE FORCE FROM AN OVERBOARD DRAIN LINE LOCATED UNDER THE LEFT HAND ENGINE INTAKE DUCT.
- THE ENGINE BLEED AIR DISCHARGE DUCTS (LOCATED ON THE LEFT AND RIGHT FUSELAGE SIDES ABOVE THE WINGS) DISCHARGE HOT AIR DURING ENGINE OPERATION.

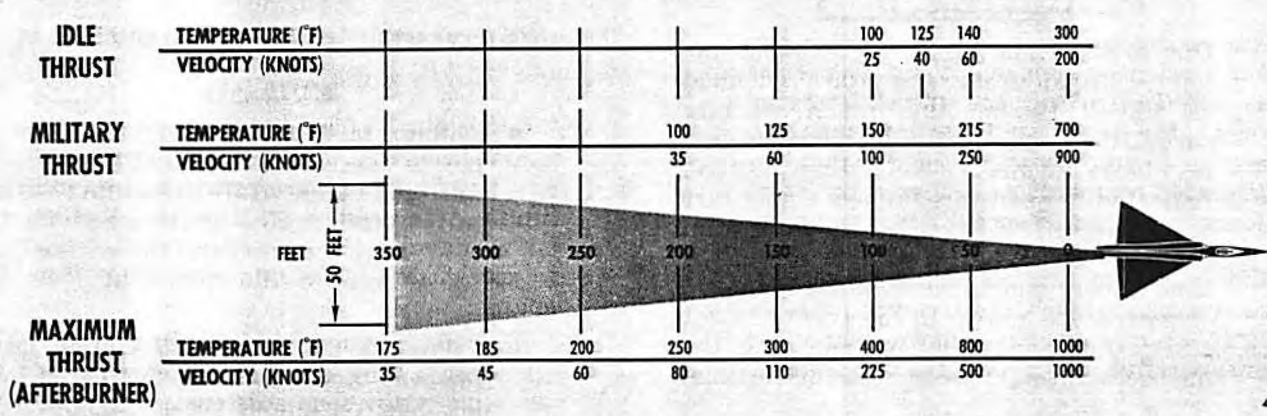


Figure 2-4

2. Engine ignition button—Depress and hold.

CAUTION

To avoid closing the starter shutoff valve and terminating ignition, once depressed the engine ignition button must be held depressed until the starting sequence is complete. If ignition button is inadvertently released, the throttle must be returned to OFF to shut off fuel flow to the combustion chambers, as starter and ignition operation cannot be reinstated without repeating the starting procedure. This is true only during ground operation since ignition for an air start is obtained immediately upon depressing the engine ignition button. Do not attempt another start until fuel drainage from the engine combustion chamber drain has ceased.

3. Throttle—START, OFF (until starter fires), then IDLE.

CAUTION

Do not jockey the throttle. The starting fuel schedule is automatically controlled by the fuel control unit. Jockeying the throttle will interrupt this schedule.

▲ 4. Fuel flow—Check for indication.

5. Exhaust gas temperature rise—Check.

   During a satisfactory start, a lightup occurs within 20 seconds after throttle is advanced to IDLE. Lightup can be noted by an indication of exhaust gas temperature.

CAUTION

Due to the time lag in the exhaust gas temperature system, a hot start is defined as 400°C. During the initial starting surge, the actual tailpipe temperatures may be as much as 200°C higher than the indicated temperature. When a hot start occurs, record on Form 781 the maximum exhaust temperature reached, and the time during which the exhaust temperature exceeded 400°C. In the event of a hot start, shut down the engine using the unsuccessful start procedure, this Section.

6. Hydraulic pressure-low warning light—Out (approximately 8 to 10% rpm).

7. Engine ignition button—Release, at approximately 30% rpm.

NOTE

Starter will automatically cut out at approximately 35% rpm by means of a centrifugal switch. The starter will also cut out when the ignition button is released, or after 13 to 16 seconds when starter fuel is exhausted.

CAUTION

If the engine does not light up within 20 seconds after throttle is advanced to IDLE, the start should be aborted. Shut down the engine by using the UNSUCCESSFUL START procedure, this Section.

8. Check idle rpm—59 to 61% rpm.

NOTE

With the exhaust nozzle open, rpm will increase approximately 2%.

9. Oil pressure-low warning light—Out.

   For cold weather operation, refer to COLD WEATHER PROCEDURES, Section IX.

10. Exhaust gas temperature—Stabilized.

    Check that exhaust gas temperature rises within limits.

▲ 11. Compressed air and electrical power—Disconnected.

NOTE

If airplane pneumatic power was used in starting, the combustion starter manual air shutoff valve should be closed after engine is started to prevent loss of pneumatic pressure through leakage.

UNSUCCESSFUL, HUNG, OR SLOW START

NOTE

- A hung start is indicated by failure of rpm to increase after lightup with exhaust gas temperature remaining within limits. After engine ignition and starter cutout have occurred and the engine fails to accelerate to idle rpm, shut down the engine.
- A slow start is similar to a hung start. However, a slow start is evidenced by a slow but continuous acceleration of rpm to idle after lightup. When a slow start is experienced, shut down the engine.

1. Throttle — OFF.

   Move throttle to OFF to shut off fuel flow to the engine to reduce the possibility of fire.
2. Check for fire.

NOTE

Do not attempt another start until cause has been determined. Wait 30 seconds for fuel drainage before attempting another engine start. Failure to obtain a combustion start shall be entered on Form 781.

CLEARING ENGINE

Refer to EXCESSIVE EGT OR FIRE IN TAILPIPE DURING GROUND OPERATIONS, Section III, for clearing engine.

BATTERY STARTING

1. All generators—OFF.
2. All boost pumps—OFF.
3. Manual start valve—START (if required).

NOTE

It may be necessary to brief ground personnel on how to disconnect pneumatic source or how to place manual start valve in NORMAL position after engine is started.

4. Master electrical power switch—ON.
5. Move throttle through normal starting sequence.
6. After engine is started, turn on necessary electrical equipment.
7. Manual start valve—NORMAL (if used).

ENGINE GROUND OPERATION

After the engine stabilizes at idle, it may be operated at full thrust; however, cooling limitations (ground operations) must not be exceeded.

NOTE

Refer to ENGINE COOLING, Section V, for ground cooling limitations.

CAUTION

Insufficient tire traction requires the use of an airplane restraining bridle when using afterburner above the minimum afterburner range if the airplane is to remain in a stationary position on the ground.

BEFORE TAXIING

Make ground tests with external power disconnected.

ELECTRICAL POWER SUPPLY SYSTEM CHECK

1. Master electrical power switch—ON.

   Check to see that the ac and dc power failure warning lights are illuminated.
2. Emergency ac generator—Check.
   a. Emergency ac generator switch—START.

      Momentarily place the emergency generator switch to START.

      It is not necessary to hold the switch in START position.
   b. Fuel quantity selector switch—TOT, then RH (or LH) then back to TOT (check for indicator movement).
   c. Move control stick; hydraulic pressure gage — Check for movement.
   d. Observe that the ADI warning flags retract indicating proper operation of the emergency ac generator.
3. DC generator switch—ON; check dc power failure warning light out.
4. MA-1 power switch—ON.

   Check illumination of the radar range lights on the radar scope to ascertain that MA-1 power is on. Monitor lights and check for 5 or 50 second power dump.

NOTE

The flight mode selector switches and automatic mode selector switches must be in the same positions in both cockpits in order that the "Will Transfer" annunciators will display "YES" or "OK." The transfer function is then operable.

5. ATG—Check.
   a. Throttle—75%.

WARNING

Perform check as quickly as possible. Check area behind airplane to see that personnel and equipment are clear of jet blast.

b. AC power failure warning light and fuel boost pressure-low warning lights — On.

c. ATG switch—AUTO.

d. Fuel boost pressure—low warning lights—Out.

e. thru f. Deleted.

6. AC generator switch—ON.

a. Throttle—IDLE.

b. Check AC power failure warning light OUT.

CAUTION

If the ac and dc power failure warning lights remain illuminated, it is an indication of a defective constant-speed drive unit, or that it is in an under drive condition due to improper oil circulation. If the lights do not extinguish, have maintenance personnel investigate before shutting engine down.

7. Armament selector switch - SPL WPN (if AIR-2A is not loaded).

8. Radar scope switch - ON.

9. Altimeter-Reset (conventional instrument display).

INTERIOR INSPECTION AFTER BATTERY START

Except for the following items, the interior check should have been completed prior to starting the engine. The following checks should be performed:

1. Armament recycle button—Depress for 5 seconds.

2. Boost pump switches—Check, then ON.

   Turn fuel boost pump switches ON, one at a time, then OFF. Check that the respective fuel boost pump warning light goes out when the pump is operating, then turn all fuel boost pump switches ON.

CAUTION

If the warning light does not extinguish and/or the warning light extinguishes on the opposite wing, the cause should be established and corrected prior to flight.

3. Special weapon armed light—OFF, press-to-test.

Ⓐ 4. CG transfer test failure light — Press-to-test.

5. Landing and taxi light switch—Climatic.

6. Oxygen quantity—Check.

7. Marker beacon light — Press-to-test. (FP-RP)

8. Fuel quantity—Check.

   Select FWD, LH, RH, TOT, and No. 3 positions with the fuel quantity gage selector switch(es) and check the fuel quantity gage for proper fuel quantities. Leave the switch in the TOT position.

NOTE

No. 3 tank fuel quantity cannot be obtained with the fuel quantity gage selector switch in the FWD position.

9. All external and internal light switches — Climatic.

10. Pitot heat switch — Check, then OFF.

11. Seat and rudder pedals—Adjust. (FP-RP)

HYDRAULIC AND FLIGHT CONTROL SYSTEM CHECK

1. Throttle — IDLE.

2. Speed brakes switch—IN, then center (off).

   Retract speed brakes and check with crew chief for proper brake operation. Secondary hydraulic system pressure may drop momentarily but must return to normal.

3. Hydraulic pressure — Check 3000 (±100) psi.

CAUTION

Any pressure over 3100 psi, when there is no demand on the system, indicates possible hydraulic system component failure.

4. Flight mode selector switch — PITCH.

   Note that engagement has no noticeable effect on controls or surfaces.

5. System check.

   a. Control surface movement.

   Check for control surface movement. With the stick in the full aft position, check elevon surfaces. Move stick to the full forward position and check for binding in either stick or elevator movement. Verify each control surface movement. Check full rudder application in both directions to insure that full control is available and that brakes are not being inadvertently depressed during rudder pedal movement.

b. Hydraulic system recovery.

Check operation of hydraulic system by moving the control stick from the left forward corner to the right aft corner in approximately three seconds. If system pressure drops, the hydraulic pressure gages must return to system pressure within two seconds or less after control stick movement has stopped. Repeat the procedure by moving the stick from the aft left corner to the forward right corner in approximately three seconds and check for normal system pressure within two seconds. If the hydraulic system does not recover within two seconds or less the flight should be aborted.

6. Emergency direct manual button—Depress.

   Check that flight mode selector switch returns to DIR MAN.

7. Manual mode trigger — Depress.

   Note that the flight mode failure warning light extinguishes.

8. Trim — Check and set for takeoff trim.

   Trim nose down, check for forward stick movement; trim RWD with stick full depressed to the right and check for left control pressure; trim LWD with stick full depressed to the left and check for right control pressure, rudder trim; then depress takeoff trim button. Rudder should return to neutral. Control stick should move aft to the takeoff position and the takeoff light should illuminate. Check light out when button is released.

AIR REFUELING SYSTEM CHECK (AIR REFUELING PLANNED)

Prior to any planned air refueling mission, the following operational ground check of the air refueling system should be performed. Operation of the slipway door, boom latches, and slipway lights must be visually verified.

NOTE

The items for visual check can be accomplished by the crew chief while standing on the ladder.

WARNING

Care must be taken by personnel operating near engine inlet ducts to avoid injury or foreign object damage while the engine is running.

1. Air refuel switch—ON.
   a. Slipway door—Fully open.
   b. Boom latches—Retracted.
   c. Slipaway lights—On.
   d. Ready light — On.

NOTE

- On A airplanes, tank pressure blow down results in illumination of the F tank and T tank shutoff valve warning lights, "FUEL VALVE CLOSED" warning light, and master warning light. These lights will remain on until the air refuel switch is turned off.
- On B airplanes, with fuel in the F tank, tank pressure blow down results in illumination of the "F TANK PRESS" warning light and the master warning light. These lights will remain on until the air refuel switch is turned off.
- On B airplanes, if the F tank is empty, the fuel low level float switch prevents illumination of the "F TANK PRESS" warning light and master warning light. During inflight refueling, when fuel enters the F tank, the "F TANK PRESS" warning light and master warning light will illuminate.

2. Reset/MBL switch—MBL.
   a. Boom latches—Extended.
3. Manual disconnect switch — Depress and hold.
   a. Boom latches—Retracted.
   b. Disconnect light — On.
4. Manual disconnect switch — Release.
   a. Boom latches—Extended.
   b. Disconnect light — Off.
5. Reset/MBL switch—NORM.
   a. Boom latches—Extended.
   b. Contact light — On.
6. Manual disconnect switch — Momentarily depress.
   a. Boom latches—Retracted.
   b. Ready light — Off.
   c. Contact light — Off.
   d. Disconnect light — On.
7. Reset/MBL switch—RESET.
   a. Boom latches—Retracted.
   b. Ready light — On.
   c. Disconnect light — Off.

8. Air refuel switch — Off.

   A delay of a few seconds will occur between the time that the switch is placed in OFF and refueling receptacle door closure. During this delay, the disconnect light will be on and the ready light will be off.

   a. Slipway door — Closed.

   Check that all indicator and warning lights show that the fuel system has been returned to normal operation.

GENERAL

Observe the following instructions before taxiing:

1. Oxygen supply switch — ON. (FP-RP)

CAUTION

When using oxygen through the survival kit regulator, do not turn the oxygen supply switch to ON until immediately after the oxygen mask is fastened in place or the pressure helmet faceplate is closed. If oxygen supply is on without the system being fully connected, the free flowing oxygen will result in low temperatures which may damage oxygen system components or cause personal injury.

2. Engine anti-ice warning test button — Depress and hold for approximately three seconds.

   The warning light requires approximately three seconds to illuminate, and will extinguish after the test button is released. The test must be performed with the surface and engine anti-ice switch in AUTO ON or MAN ON.

3. Rain removal switch — ON, then OFF.

   Place the rain removal switch ON and check for positive air flow, then move the switch to OFF.

4. Canopy — Close (as required).

CAUTION

- On Ⓑ airplanes the canopy must be closed to within 12 inches of the canopy sill to engage the canopy sway braces, prior to taxiing.
- Refer to OTHER OPERATING LIMITATIONS, Section V, for maximum taxi speed with canopy open.
- If binding is evident when actuating the canopy latch handle, an entry should be made on Form 781. Continued operation could damage the latching mechanism sufficiently to preclude unlatching of the canopy hooks either manually or during emergency canopy jettison.

5. Flight instruments — Check and set. (FP-RP)

   a. Check the following instruments on the conventional instrument display:

      (1) Altimeter reading corresponds to pressure conditions.

WARNING

It is possible to misset the altimeter by 10,000 feet and still have the correct indication on the barometric scale. This happens when the barometric set knob is continuously rotated after the barometric scale is out of view until eventually the numbers reappear in the barometric scale. If the correct altimeter setting is then established, the altimeter will read 10,000 feet in error. To avoid the possibility of this error, pay particular attention to the ten-thousand foot pointer when setting the altimeter.

      (2) Vertical velocity indicator, airspeed-angle of attack indicator, Mach indicator, accelerometer, and turn-and-slip indicator, indicating proper static conditions.

      (3) Heading indicator (slaved) stabilizing (azimuth ring on the course indicator).

      (4) Attitude indicator.

      Check attitude warning flag retracts within 1 minute ±10 seconds and horizon bar is free from oscillation and has proper attitude and response to trim knob. Set 5° low.

WARNING

If the attitude warning flag requires longer than 1 minute ±10 seconds to retract, or any oscillations are noted after the attitude warning flag retracts, a malfunction exists and the indicator will not be reliable for flight. Note the nature of the malfunction on Form 781.

(5) Set altimeter to field elevation.

b. Check the following instruments in the integrated flight instrument system:

(1) Compass card — Slaved to airplane heading.

(2) AMI, AVVI, and turn-and-slip indicator, indicating proper static conditions. With angle of attack vane full up, check lubber line between words MIN and SAFE on angle of attack indicator.

(3) Set altimeter to field elevation.

(4) Standby altimeter — Set.

(5) ADI.

Check attitude warning flag retracts within 1 minute ±10 seconds and horizon bar is free from oscillation and has proper attitude and response to trim knob. Set 5° low.

WARNING

If the attitude warning flag requires longer than 1 minute ±10 seconds to retract, or any oscillations are noted after the attitude warning flag retracts, a malfunction exists and the indicator will not be reliable for flight. Note the nature of the malfunction on Form 781.

6. Barometer setting indicator — Set to altimeter setting. (FP-RP)
7. Engine pressure ratio gage—Set. (FP-RP)

TAKEOFF CHECK TABLE

TEMPERATURE		PRESSURE RATIO SETTING
°F	°C	
113	45	1.97
108	42.5	1.98
104	40	2.00
99	37.5	2.01
95	35	2.03
90	32.5	2.05
86	30	2.06
81	27.5	2.08
77	25	2.09
72	22.5	2.11
68	20	2.12
63	17.5	2.14
59	15	2.15
54	12.5	2.16
50	10	2.18
45	7.5	2.19
41	5	2.20
37	2.5	2.22
32	0	2.23
27	−2.5	2.24
23	−5	2.25
18	−7.5	2.26
14	−10	2.27
9	−12.5	2.28
5	−15	2.29
0	−17.5	2.30
−4	−20	2.31
−8	−22.5	2.32
−13	−25	2.33
−18	−27.5	2.33
−22	−30	2.34
−27	−32.5	2.34
−31	−35	2.35
−35	−37.5	2.35
−40	−40	2.35

8. Ejection seat ground safety pin — Remove and display to crew chief. (FP-RP)

9. Deleted.

10. Power annunciator—"OK."

11. Chocks—Removed.

STARTING ENGINE (SCRAMBLE)

1. Normal start.
2. All personal equipment — Attach. (FP-RP)
3. Canopy support — Remove.

ELECTRICAL POWER SUPPLY SYSTEM (SCRAMBLE)

1. External power — Disconnected.
2. Master electrical power switch — ON.
3. Emergency AC generator—Check.
4. DC generator switch—ON.
5. MA-1 power switch—ON.
6. ATG—Check.
7. AC generator switch—ON.
8. Altimeter—RESET (conventional instrument display).

HYDRAULIC POWER SUPPLY SYSTEM (SCRAMBLE)

1. Hydraulic and flight control system—Check.

2. Takeoff trim button — Depress.

BEFORE TAKEOFF (SCRAMBLE)

1. Ejection seat ground safety pin — Remove. (FP-RP)

2. Canopy — Close, lock, & light out.

3. Idle thrust control switch — OFF.

4. AIR-2A Arm/Safe/Monitor circuit breaker —Push in.

5. Cabin air selector switch—PRESS.

6. Emergency fuel—Check.

RADAR/IR GROUND CHECKOUT PROCEDURES

WARNING

Do not point the antenna at personnel as excessive microwave radiation can be dangerous to life or parts of the body.

1. IR SEEKERHEAD - Check extension.

   As STANDBY power is applied, the Seekerhead will extend and the IR compressor will begin to run. Cell cooldown can take up to 5 minutes according to specifications but will generally take much less time. The radar scope should illuminate within 30 seconds. To see the attack displays, close and lock the canopy or depress the scope "ON" button for about 3 seconds to get the attack displays on. Using the latter method, the attack displays will remain on the scope for 3 minutes and then disappear. The cycle will then have to be repeated to regain the displays.

1A. Photographic Recorder—Check for proper operation.

2. Search and Attack Displays - Check centering.

   With ILS selected, the B-sweep should appear at 0 degrees azimuth. If it appears in another position, this will be the position on the scope that a target dead ahead will appear. Check the left hand side of the "Z" marker for vertical centering with the scope etchings. If it is too high or low, the firing bar will be displayed in error by an equal amount. There is a built in problem with parallax, since the etchings are displaced well in front of the tube face.

3. DIPS/AUTO MODE switch - MAN NAV or AUTO NAV.

NOTE

Remaining in ILS will prevent the MA-1 system from entering PURSUIT.

4. Artificial horizon - Adjust 5 degrees high.

5. Firing range bar - 3.3 miles. (If SPL WPN selected).

   If no firing range bar is displayed at time-in to STBY, pursuit will not be entered with an MA-1 power dump.

6. Nose/Tail switch - Tail.

7. Firing range bar - 1.8 miles. (If SPL WPN selected).

8. Armament Selector switch - RAD.

9. LC/PUR switch - Depress momentarily.

The Man Pur relay drops out passing through VI.

10. Firing range bar - 0.4 mile.

11. Nose/Tail switch - Nose.

12. Firing range bar - 2.1 miles.

13. Range switch - 16 miles.

The firing bar should still be in the vicinity of 2 miles.

14. LC/PUR switch - Depress momentarily.

Note the disappearance of the firing bar. If the radar is not searching yet, proceed with the IR check.

15. RDR/IR Select button. Depress and release.

Note the appearance of the Az-El dot sweeping in broad scan (±53.5°). Visually check the ANT ELEV control in the detent position. The C-scan should bracket the center etchings on the scope (2 bars of the 4 bar scan either side of the etchings). The top bar of the 4 bar scan should be 3.75° above the etchings and the bottom bar 3.75° below them. If the elevation scan is wider than this, it could mean that the scan rasters are too far apart and do not overlap. The Az-El dot should leave a stored trace of about 1 bar in broad scan.

16. Elevation scaling - Check.

With the hand control in the azimuth detent, depress full action switch and move the vernier to the full up position. The AZ-EL dot should go 46.5° up. Then move the vernier to the full down position. The AZ-EL dot should go 30° down.

17. IR Tone and Video - Adjust.

IR Tone volume should be turned cw to a comfortable level. The tone threshold should then be turned cw through the null and stopped at a low growl on the other side. Turn the IR video threshold cw until a small amount of "grass" appears on the "C" scan. Check for IR targets (Targets will not be visible if seekerhead cool down is insufficient.)

18. IR Lock-on - Accomplish.

a. Note that the short firing bar appeared at full action switch depression and the IR range mark jumped to the top left side of the scope.

b. IR Track display - Check. The IRT display will appear when the action switch is released during the lock-on process. A long firing bar, steering dot and reference circle will appear at this time. The expanded "C" display will be present for 8.5 seconds. A flat base line would indicate no lock-on and the tone would be lower in pitch.

c. RDR SCAN display - Check.

The scan limits in broad are now (± 46.5°).

d. RDR SLVD - Select.

If a radar target is visible in the sweep, go half-action and put the gate on the target. The range scale lights should illuminate at ROT. At release of action switch, the gate should remain visible. Recall the expanded "C". It should appear at the elevation of the seekerhead.

19. RDR/IR Select button - Depress and release.

If locked on from the previous step, a lead collision display should appear at crossmode. If no lock-on was possible in IR, you must return to search before--crossmoding. Obtain radar lead collision lock-on. Check the lead collision displays for calibration. Check the B-sweep centering and length. These adjustments can seriously affect the use of the firing bar. Also, check the TTG circle for roundness and size. Attack display size can affect the firing bar range. Check the "Z" marker - the left side should be adjacent to the center etchings and the right side should be within several degrees of target elevation. If the "Z" marker is not properly calibrated, you will have to apply the error when you use the elevation scale of the scope.

20. Auto Search button - Depress momentarily.
    Note that the B-sweep is searching (± 53.5°). Check that the antenna elevation marker is stepping on the 4-bar scan.

21. Radar scope - Adjust.

    a. Erase intensity, adjust and verify desired rate of target erasure. Attack intensity, adjust to minimum viewing level.

CAUTION

Excessive attack intensity will cause damage to the MMST.

DIMMER, adjust for desired brightness level. The IF gain should normally be fully cw. The video gain should normally be fully cw. With excessive background noise level turn the video gain knob ccw slightly.

    b. B-sweep appearance - Check intensity in 4, 16 and 40 mile ranges while in search, supersearch and manual search.

    c. Tune switch -- F-MAX, F-MIN, NORM.
       Be sure that the video does not break up or disappear.

22. Radar Lock-on - Accomplish.

    a. CADJ switch - On. (Check for reduced steering dot activity).

    b. Chaff switch - GATE then ALL. Note that the radar remains locked-on and when ALL is engaged check that the CADJ switch drops OFF.

    c. Radar Mode switch - HOM, check ATOT sensitivity.
       Go half action and move the gate off the target. If the ATOT sensitivity adjustment is too high, the lead collision extrapolate display will appear immediately.

    d. Radar Mode switch - Norm.

    e. Auto search button - Depress momentarily.

23. IR seekerhead - Stow.

24. Armament selector switch - VI.

25. DL Test Pattern - Check.

    a. DISP/AUTO MODE switch - MAX RNG or MIN TIME.

    b. DL Volume Control - Increase. (Listen for motor boating sound).

    c. DL Displays - Check for the following:

       (1) Command heading - 045° and 225° (corrected).
       (2) Command and target altitudes - 15,000, 60,000 and 30,000 feet.

(3) Command Mach - 0.53, 1.93 and 0.98.

(4) Scope displays - Check proper displays. TTG count down from 20 seconds and then after offset indications.

(5) MCC - Check indicators.

NOTE

MCC indications will not be displayed in the Universal Test Message if tactics are not set in CC.

All indicators which reflect MCC information will show the position, range, bearing and elevation of the local division test target. (Briefing on the location of the test target will have been necessary.)

NOTE

The test pattern values alternate approximately every 15 seconds. Normally the altitude indications will not reach 60,000 feet before the next message of 30,000 feet is received.

d. TEST ADDRESS/DISPLAY switch - Set to DISPLAY OFF or DISPLAY ON.

26. Antenna Stabilization and Range Gate drift - Check.
While taxiing out, run the radar antenna up in hand control. Release the action switch and note that the antenna remains stable in azimuth and elevation for at least 15 seconds. The range gate should not drift in or out.

RADAR GROUND CHECK – MSR OR WSEM'S ABOARD

1. With V1 selected, armament recycle button depress and hold for 5 seconds.

2. Armament selection indicator – OK (for SPL WPN, RAD, and ALL).

3. Taxi and takeoff – VIS IDENT selected.

NOTE

Armament switches will not be safety wired with MSR or WSEM aboard.

TAXIING

WARNING

If taxiing with the canopy partially open, do not place arms or hands on the canopy sill. Should the canopy fall, serious injury could result.

CAUTION

When taxiing the airplane with fully fueled 360 gallon external tanks, the following should be closely observed to prevent damage to the landing gear structure:

a. Taxi speed should be held to the minimum practical.

b. Do not apply wheel brakes during turns. Slow the airplane prior to reaching the turn point.

c. All turns should be made with as large a radius as practical and at a reduced speed.

d. Do not apply wheel brakes in such a manner as to skid the tires on the pavement.

e. In the event of nose wheel steering failure, do not attempt to taxi using wheel brakes for steering.

NOTE

For minimum turning radius and ground clearances, see figure 2-5.

1. Brakes and nose wheel steering—Check.
Check brakes and momentarily depress nose wheel steering button to engage steering. Maintain directional control by operating rudder pedals as required.

2. Idle thrust control switch — As desired.

3. Flight instruments — Recheck and set (if necessary). (FP-RP)

4. Navigation equipment — Check. (FP-RP)
Check operation of MA-1 navigation equipment and heading indicator (slaved) for turn indication during taxiing. Compass should conform to actual heading changes.

minimum turning radius and ground clearances

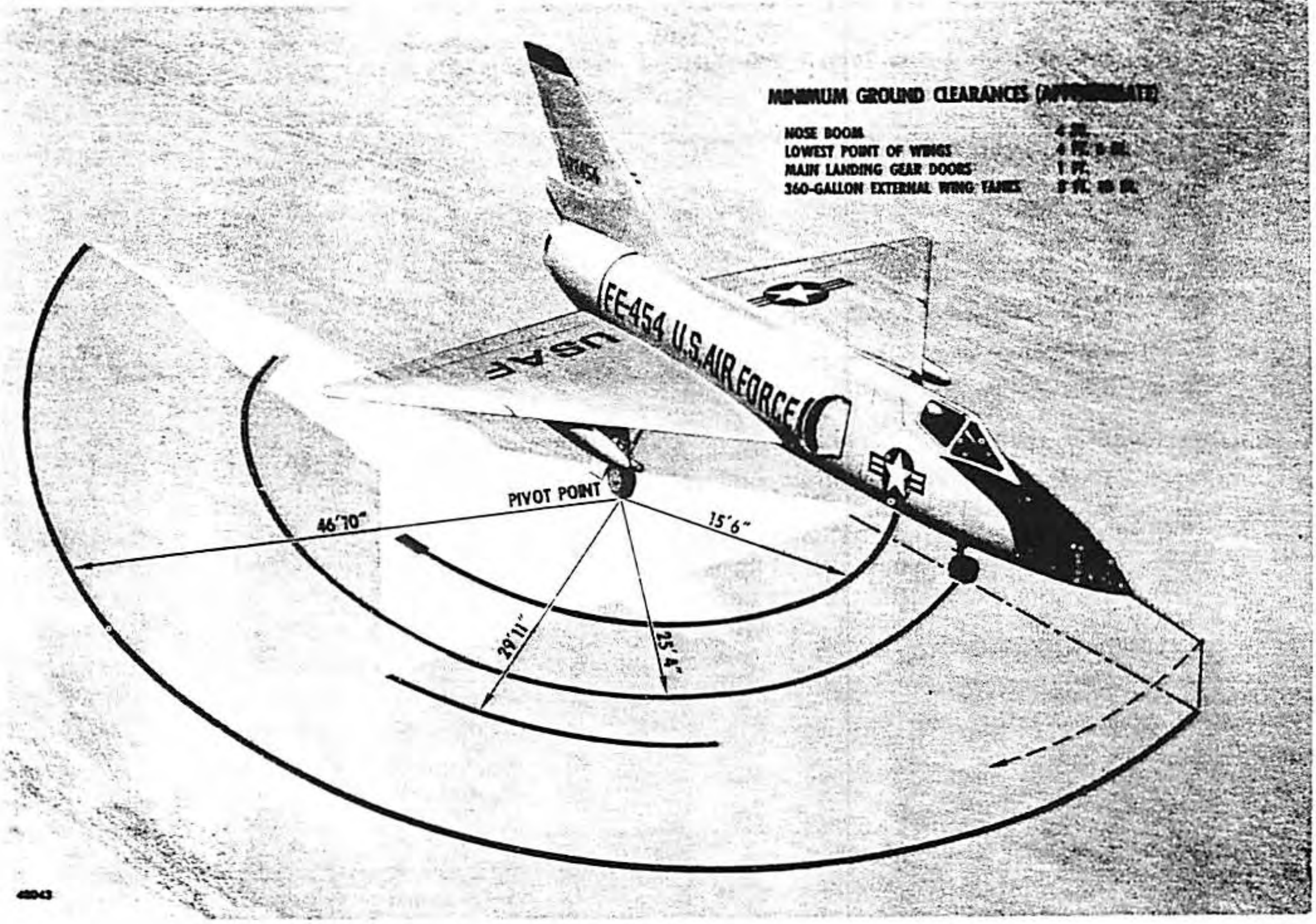


Figure 2-5

5. Deleted
6. Deleted.

BEFORE TAKEOFF

AIRPLANE CHECK

1. Canopy—Close and lock, check light out.

   On Ⓐ airplanes check the canopy clutch disengaged by listening for the "clunk" when the lock handle is actuated. On Ⓑ airplanes, notify the other crew member that the canopy is to be closed and wait for confirmation that he is clear before closing the canopy.

CAUTION

- To prevent breakage of the canopy shear pin, do not close the canopy in one complete motion. Hold the canopy switch at CLOSE until the canopy is within two inches of the canopy sill, momentarily release the canopy switch, then hold the canopy switch at CLOSE until the canopy has stopped moving and is completely flush with the canopy sill.

- If canopy clutch does not disengage when the canopy is locked, the canopy cannot be raised manually with the canopy latches released.

- Do not fly with the canopy clutch engaged. The canopy actuator shear pins will shear when climbing to altitude with the cockpit pressurized and upon landing the canopy may not open fully.

2. AIR-2A Arm/Safe/Monitor power circuit breaker — Push in.
3. Cabin air selector — PRESS.
4. Pressure suit vent control handle—Cracked.

   Move the pressure suit vent control handle to the OFF position and crack approximately one-half inch when wearing the full pressure suit.

5. Ejection seat ground safety pin — Check removed. (FP-RP)
6. Idle thrust control switch — OFF.

WARNING

Positively ensure that the idle thrust control switch is in the OFF position. Failure to close the exhaust nozzle will result in excessive thrust loss and ground roll during a military thrust takeoff. Although afterburner (maximum) thrust is not affected by the idle thrust control switch, the switch must be placed OFF to permit the nozzle to close in event afterburner blowout occurs during an A/B takeoff and takeoff is continued using military thrust.

NOTE

The idle thrust control switch must be OFF before preflight engine checks to obtain correct pressure ratio and EGT.

WARNING

With the idle thrust control switch ON, approximately 35% loss of thrust will occur at FULL MIL POWER when the weight of the airplane is on the landing gear.

7. Flight mode selector switch — DIR MAN.

WARNING

High-speed, low-altitude flight without dampers can cause control problems and should be avoided except where required to accomplish the assigned mission.

8. All warning lights — Out.
9. Formation-navigation lights switch — Climatic.
10. Pitot heat switch — ON.
11. Line up with nose wheel steering engaged.
12. Gyro erect button — Press to erect (if required)
13. IFF/SIF control panel — As required.
14. Deleted.

ENGINE CHECK

1. Throttle—IDLE.
2. Fuel control emergency system—Check.
   a. Fuel control switch—EMER and check fuel control warning light on.
      During changeover from NORMAL to EMER a momentary fuel flow fluctuation will occur.
   b. Fuel flow at IDLE—Check 1150 pph (minimum) to 1650 pph.
   c. Throttle—85% RPM.
   d. Fuel control switch—NORM and check fuel control warning light out.
3. Throttle—FULL MILITARY POWER: allow engine RPM to stabilize.
4. EGT spread—Check.
   a. EGT. and rpm—Note.
   b. EGT spread button—Depress and check that EGT overtemperature warning light illuminates as EGT reading decreases.
      Note EGT spread. If spread is above limits. abort and record reading in Form 781.
   c. EGT spread button—Release.
5. Oil quantity gage—Check needle in the green range.
6. Engine instruments—Check (FP-RP)
   Check tachometer, exhaust gas temperature gage, and fuel flow indicator for normal operating limits. Check EPR gage pointer within the arc of the takeoff thrust index marker.

TAKEOFF

CAUTION

Avoid wake turbulence. Allow a minimum of two minutes before takeoff behind a heavy aircraft or helicopter and 4 minutes behind extremely heavy aircraft such as C-5A and Boeing 747. With effective crosswinds of over 5 knots, the interval may be reduced, but attempt to remain above and upwind of the preceding aircraft's flight path.

NORMAL TAKEOFF

A typical takeoff is illustrated in figure 2-6. Refer to T.O. 1F-106A-1-1 for takeoff charts showing distances required at varying gross weights, temperatures, and field elevations. Use the following procedures for normal takeoff:

1. Throttle — FULL MIL POWER.
2. Deleted.
3. Brakes — Release.
   Release brakes and establish a straight takeoff roll.

takeoff
(typical)

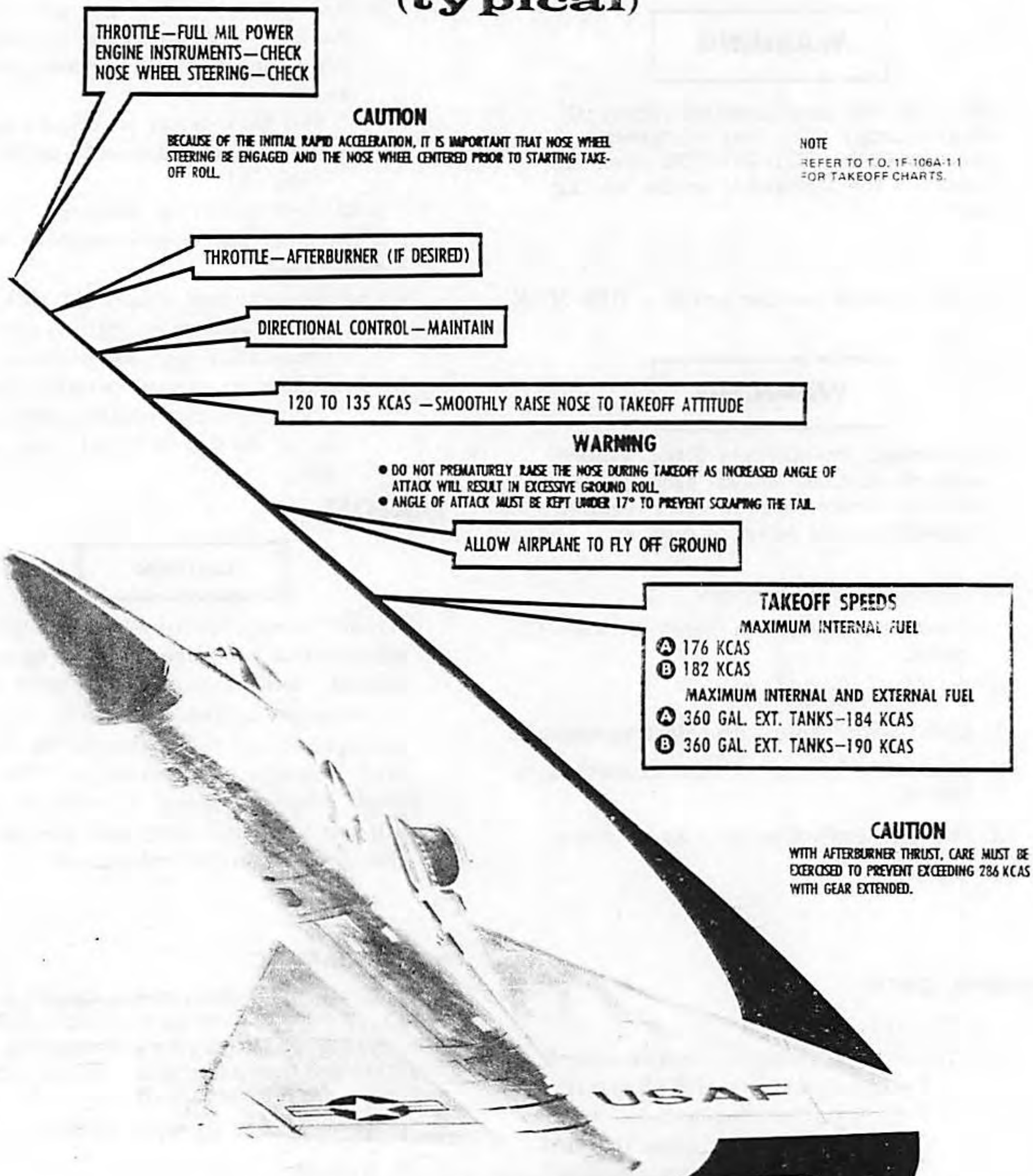


Figure 2-6

4. Nose wheel steering — Check.
   Move rudder pedals until it is ascertained that nose wheel steering is engaged.

CAUTION

Because of the initial rapid acceleration, it is important that nose wheel steering be engaged prior to starting takeoff roll.

5. Throttle — AFTERBURNER.
   Afterburner should light within two seconds.

WARNING

- When the engine is not in alignment yaw may occur either gradually or abruptly between 50 and 350 KCAS and should be corrected with rudder application, not with rudder trim. The yaw condition is more noticeable during cold weather and during high angles of attack. If the yaw condition becomes aggravated, reduce to military thrust and note the condition on Form 781 upon completion of the flight.
- Takeoff should be aborted immediately if any directional change is noted when the afterburner is ignited. A directional change at this time could indicate possible afterburner nozzle malfunction which could cause side forces to be applied to the extent that rudder would be insufficient to control the airplane immediately after leaving the ground.

CAUTION

- When making an afterburner takeoff, check that the pressure ratio gage returns to the takeoff reading after dropping momentarily.
- Angle of attack must be kept under 17° to prevent scraping the tail and inadvertent tailhook engagement.
- On Ⓑ airplanes, when operating with F tank empty, caution should be used when rotating to takeoff attitude to prevent scraping the tail by overrotating.

NOTE

- Nose wheel shimmy may be encountered at speeds between 50 and 80 KCAS. If shimmy is encountered, disengage nose wheel steering and maintain directional control with rudder and brakes.
- If nose wheel shimmy becomes excessive, abort the takeoff.

6. 120 to 135 KCAS—Smoothly raise nose to takeoff attitude and allow aircraft to fly off ground. For a maximum thrust takeoff, nose should be raised to horizon, or slightly below horizon for military thrust takeoff. Lower angles of attack will result in high speed and longer ground rolls. Higher angles of attack will result in lower airspeeds and lower rates of climb immediately after takeoff. While still on the runway with the nose on the horizon, the angle of attack will be approximately 12 degrees. The airplane should be allowed to fly off the ground, with lift-off occuring at the following speeds:

TAKEOFF SPEEDS

Full Internal Fuel	Full Internal Fuel Plus External Fuel
Ⓐ176 KCAS	Ⓐ184 KCAS
Ⓑ182 KCAS	Ⓑ190 KCAS

WARNING

Do not prematurely raise the nose during takeoff as increased angle of attack at low speed will result in excessive ground roll during takeoff.

7. Attitude indicator—10° nose-up indication.
   After landing gear is retracted, maintain positive climb until intercepting the climb schedule.

CAUTION

With afterburner thrust, care must be exercised to avoid exceeding gear limit speeds before landing gear is fully retracted. Gear retraction time is approximately four to six seconds.

MILITARY THRUST TAKEOFF

If takeoff is to be made without the use of the afterburner, normal takeoff procedures should be utilized up to the point of breaking ground. Due to the possibility of increasing drag excessively by a high angle of attack immediately after takeoff, the nose should be raised to a point just below the horizon and the airplane allowed to fly off. When using military thrust for takeoff, it is relatively easy to scrape the tail if the nose is raised excessively just prior to breaking ground. After breaking ground, the airplane should be allowed to accelerate until certain the airplane will remain airborne before retracting the landing gear.

MINIMUM RUN TAKEOFF

A minimum run takeoff can be accomplished by using the normal takeoff procedures and engaging afterburner as soon as possible after brake release.

CAUTION

Angle of attack must be kept under 17° to prevent scraping the tail and inadvertent tailhook engagement.

CROSS-WIND TAKEOFF

Cross-wind takeoffs present no particular problems in this airplane. In addition to the Takeoff Distances and Speeds Charts, check the Takeoff and Landing Cross-Wind Chart in the Appendix (figure A2-8) to determine the cross-wind component and minimum nose wheel lift-off speed. After lift-off, establish the crab angle necessary to maintain the desired flight path.

AFTER TAKEOFF — CLIMB

When airplane is definitely airborne:

1. Landing gear handle — UP and check lights.
   Ensure handle is in fully UP position, and check landing gear warning light out.

CAUTION

To prevent airloads from inflicting structural damage, the landing gear should be raised and checked up and locked before exceeding gear limit speed. Refer to AIRSPEED LIMITATIONS, Section V.

WARNING

If external wing tanks are installed, particular attention must be given to the operating limitations for flight with external tanks. Refer to OPERATING LIMITATIONS, Section V. Turbulent air can cause rapid increases in g loads. Keep a safe margin where possible. Monitor the accelerometer.

2. IFF/SIF—Checked.

NOTE

As soon after take-off as flight conditions permit, positive operation of the IFF should be established with an Air Traffic Control Facility if the route of flight will require an operative IFF. Consult appropriate FLIP documents for IFF/SIF traffic control requirements and procedures.

3. Flight mode selector switch—PITCH (above 5000 feet).
4. Altimeter – Set area altimeter below FL180 and 29.92 IN. HG. at FL180 and above.
5. Standby flight instruments—Check for normal operation.
   Reset standby altimeter at designated altitude and compare standby flight instruments with primary flight instruments.

CLIMB

The recommended climb speeds as given in the Appendix should be followed.

CAUTION

Exhaust gas temperatures should be closely monitored during all phases of flight, particularly during climb. Throttle should be retarded to maintain EGT limits.

ARMAMENT SAFETY (TRIGGER) CHECK

WARNING

- If at any time the missile bay doors begin to open, release trigger (if pressed), select VIS IDENT, wait a minimum of 75 seconds, then close missile bay doors. With armament on board, return to base and DO NOT depress the trigger.
- This check should be accomplished on all flights with the nose of the airplane pointed away from populated areas.

1. Arm-safe switch – SAFE.
   The guard must be safetied and sealed if armament is aboard.
2. Special weapon release lock switch — LOCK.
   The guard must be safetied and sealed if primary armament is aboard.
3. Armament selector switch — MISSILES RAD.
4. LC/PUR Switch - Depress.

5. Radar scope — Check for firing bar.
6. Armament trigger — Wait 10 seconds, then press to second detent.

   Check that there is no missile bay door operation. Wait for "X" on scope.

NOTE

On Ⓑ airplanes when UHF control is in the rear cockpit, both triggers must be pressed to make check valid.

7. Armament selector switch — VIS IDENT.
8. Deleted.

NOTE

If practice AIR-2A intercepts are conducted with secondary armament aboard, the following safety steps shall be accomplished instead of the above steps. Do not attempt practice intercepts in missile mode if loaded with secondary armament.

1. Arm-safe switch — SAFE.

   The guard must be safetied and sealed if armament is aboard.
2. Special weapon release lock switch — LOCK.
3. Armament selector switch — SPL WPN.
4. LC/PUR switch - Depress.
5. Radar scope — Check for firing bar.
6. Armament trigger — Press to second detent.

   Check that there is no missile bay door operation. Check for "8" on scope.

NOTE

On Ⓑ airplanes when UHF control is in the rear cockpit, both triggers must be pressed to make check valid.

7. Armament selector switch — VIS IDENT.
8. Deleted.

NAVIGATION

RADAR AIRBORNE CHECK

1. ANT ELEV — Check for level sweep.

   a. Miles vs altitude (15 miles - 15,000 feet) above the terrain.

   If you did not have the time before takeoff or were not satisfied with your ANT ELEV antenna elevation check on the ground, recheck it in the air. A rough rule of thumb is this: ground clutter should appear on the bottom sweep of the raster on the basis of approximately 1 mile to each 1,000 feet of altitude. If you are at 15,000 feet above the terrain, you should have ground return, not the altitude line, at slightly less than 15 miles on the scope, using the bottom bar of the 4-bar pattern.

   b. Level with other aircraft — Target on both sweeps, 2-Bar.

   If your position is No. 2, you have an excellent check on the leader after you both level off at altitude. Adjust the antenna until you are painting evenly on both bars of the two bar pattern. Note the displacement of the ANT ELEV knob index and the Z-marker position. Use this as the center position throughout the flight.

2. Antenna — Depress.

   During automatic search depress the antenna scan pattern so that the ground return is visible. Check whether the dark ground return area moves in a straight line across the scope even if aircraft is in a turn. Uneven or sudden drops can cause blank areas in the search pattern.

3. Radar horizon — Drift or tilt.

   A small amount of tilt can reduce contact ranges appreciably because the antenna may be sweeping above or below the target. If time permits, reset your grid reference and re-erect with aircraft wings level and unaccelerated flight. If time does not permit, hit erect button only, realizing, that any subsequent data link or AUTO NAV steering information will be inaccurate. When the attack is complete, remember to set in the correct grid reference and re-erect. If the tilt is not much more than 5 or 6 degrees, you can compensate during search by using the ANT ELEV knob and moving the antenna up or down a bit

4. Range gate drift – Recheck.

   If the ground check showed a range gate drift, recheck it now. Use the same procedure for the ground check. If the drift is down, lock on from the top. Conversely, if drift is out, lock on from the bottom.

5. Antenna tracking.

   Lock on in a turn or make a turn if straight and level. Check that the B-sweep moves with the target position on scope. Ensure that dot movement checks OK.

6. Computer operation – Check throughout flight.

WARNING

If the computer has been operating in an intermittent manner in the NAV/DL mode as evidenced by erroneous displays, (i.e., frozen DL dot, improper NAV commands) then the pilot should EXPECT computer malfunctions in the attack mode. These attack mode malfunctions (no 20 second time, no F-pole, no fire signal) could and do occur with no warning. Knowledge of these critical ranges (20 seconds, firing range) is MANDATORY if the pilot is to recognize and cope with a malfunctioning computer. The pilot must CONSTANTLY MONITOR his range to the target on the B-sweep. When within 4 miles the pilot should GO TO THE 4 MILE SCOPE. That is the BEST WAY to avoid hitting the target. IF radar range is not available because of ECM, clutter, or confusion then the pilot should NOT PRESS THE ATTACK without a tallyho on the target.

IR AIRBORNE CHECK.

1. IR STOW switch – RDR SLVD or RDR SCAN.
2. AUTO SEARCH button – Press and release.
3. Armament selector switch – Not VIS IDENT.
4. RDR IR SELECT button – Press and release.
5. IR THRESHOLD VIDEO control – Adjust.

   This control may require readjustment when passing over different types of terrain to allow for changes in background radiation.

   Adjust so that small deflections due to stray IR radiation appear on the C-scan. The change in radiation level may also require a readjustment of the IR THRESHOLD TONE control.

6. IR lockon – Accomplish.

   When a target appears visible on the C-scan, press the action switch to the first detent during the scan of the bar on which the target appeared. Check that the target appears on the supersearch display. A lateral adjustment of the hand control may be necessary if the target was outside the 33-degree limits of the supersearch scan. Press action switch to second detent, spotlight target, and release to accomplish lockon. Check that the IR expanded C-scan appears for 8.5 seconds. After 8.5 seconds, the IR expanded C-scan should be replaced by the B-sweep. The motion of the B-sweep will be dependent upon the position of the IR STOW switch.

7. AUTO SEARCH button – Press and release.
8. RDR IR SELECT button – As desired.

VISUAL IDENTIFICATION PROCEDURE.

1. Arm-safe switch – SAFE.

2. Armament selector switch – VIS IDENT.

3. DISP/AUTO MODE selector switch – Any position except ILS or ILS APCH.

4. Radar range lockon - Accomplish.

5. FLT MODE selector switch – As desired.

6. Radar scope – Monitor.

In the VI mode, the steering dot and reference circle are positioned by the radar antenna and represents the azimuth and elevation position of the target with respect to the interceptor.

WARNING

During a VIS IDENT pass, the steering dot represents target position and NOT the course to be flown. If flying manually, OFFSET THE STEERING DOT MANUALLY TO AVOID COLLISION.

7. Steering dot – Moves up and to the left from the radar scope center.

If in manual flight, fly the interceptor to offset the steering dot up and to the left when the range-to-target circle starts to shrink, 1 mile range-to-go. This manual steering action positions the interceptor down and to the right of the target.

If in automatic flight, the AFCS positions the interceptor down and to the right of the target. Since the steering dot represents the target position, the steering dot moves up and to the left.

8. Closing rate – Adjust to 50 knots or less.

9. VI WARN, MASTER WARNING, and FLT MODE FAIL lights – Monitor.

At a target range of approximately 0.25 nautical miles, VI WARN, MASTER WARNING, and FLT MODE FAIL lights illuminate and the FLT MODE switch steps down to ASSIST.

WARNING

After the FLT MODE selector switch steps down to ASSIST, the aircraft attitude at the moment of stepdown will be maintained. Because of last minute corrections by the AFCS, this could result in a collision hazard. Take immediate manual control of the aircraft.

10. FLT MODE selector switch – Monitor for step-down to ASSIST.

FUEL QUANTITY CHECKS

To avoid inaccurate inflight fuel quantity indications, do not check the fuel quantity gage when activating switches which utilize ac power. Fuel quantity checks should be made only when airspeed is stabilized and while in average cruise conditions (six degrees nose-up, wings level attitude). During flight, since the fuel quantity indicating system does not indicate individual fuel quantities with the exception of the F tank, fuel should be monitored as follows:

1. Monitor the individual (LH, RH, No. 3, and F) fuel quantities frequently during flight to verify symmetrical fuel flow and proper F tank feeding.

Ⓐ a. Check initial F tank feeding at approximately 8000 pounds.

At approximately 8000 pounds of total fuel remaining, initial F tank feeding should be checked (about 1000 pounds will remain in the F tank after completion of initial feeding). Initial F tank feeding is an indication that the T tanks have emptied.

Ⓑ b. Check for F tank feeding at approximately 8000 pounds. F tank quantity should read zero.

NOTE

On Ⓐ airplanes, fuel system "hammering" may occur during the fuel transfer sequence (at approximately 8000 pounds total fuel) and may last until first F tank feeding has been completed. The "hammering" is caused by pressure surges in the system and may be described as a series of "thumps" near the aft portion of the cockpit, where the F tank is located. Although loud and startling, the "hammering" does not jeopardize the structural integrity of the airplane.

c. Check second sequence-F tank feeding at approximately 3500 pounds.

At approximately 3500 pounds of total fuel remaining, the F tank should be checked for proper feeding into the left and right No. 3 tanks.

NOTE

The F tank should be empty at approximately 2500 pounds of total fuel remaining. If this fuel is in the T tanks during supersonic flight, there will be no indication of proper transfer of this fuel into the No. 3 tanks.

WEAPON DELIVERY

WARNING

If MA-1 power is lost during an attack and any armament has been selected, SAFE and VIS IDENT must be selected in order to recycle the system and prevent inadvertent door operation which would result in loss of armament attack capability.

AIR-2A ATTACK PROCEDURES

AIR-2A Armament Selection

In subsequent procedures in this section, when instructed to select the AIR-2A, accomplish the following steps:

1. Arm-safe switch – SAFE.
2. AIR-2A power circuit breaker - checked, closed.

   Before changing the position of the armament selector switch, the arm-safe switch must be in SAFE position.

3. Armament selector switch – SPL WPN.
4. Armament selection indicator – OK.
5. Arm-safe switch – ARM.

5A. MMST confirm light number 5–CHECK ILLUMINATED.

6. Special weapon-armed light – SPL WPN ARMED.

   If the special weapon-armed light fails to illuminate, press the light housing to test the light. If the light does not illuminate when the housing is pressed, continue the AIR-2A rocket attack. If the light illuminates when the housing is pressed, abort the AIR-2A rocket attack and if possible make a missile attack.

7. Special weapon release lock switch – UNLOCK.
8. Special weapon release lock indicator – UNLOCK.
9. Photographic Recorder–CHECK OPERATION.

AIR-2A Post-Attack Procedures

In subsequent procedures in this section, when instructed to perform AIR-2A post-attack procedures, accomplish the following steps.

NOTE

Wait 15 seconds after releasing armament trigger before placing arm safe switch to SAFE or armament selector switch to VIS IDENT to prevent an unscheduled reopening of the missile bay doors.

1. Arm safe switch – SAFE.
2. Armament selector switch – VIS IDENT.
3. Special weapon release lock switch – LOCK.
4. Special weapon release lock indicator – Striped.
5. AUTO SEARCH button – Press and release.

ROCKET LEAD COLLISION ATTACK

NOTE

The automatic flight control system (AFCS) may be used for automatic attack steering. Following an automatic attack, escape maneuver must be flown manually.

1. AIR-2A armament selection – Accomplish.
2. Radar range lockon – Accomplish.

3. Attack steering – Accomplish.

   If flying manually, fly to center steering dot in reference circle.

4. Armament trigger – Press to second detent and hold at 20 seconds before firing time.

   At 20 seconds before time to fire (indicated by shrinkage of time-to-go circle), press and hold the armament trigger to the second detent. This will allow the fire signals to be supplied to the AIR-2A launching system by the computer.

5. Radar Scope – Monitor for "8" pullout signal.

6. Escape maneuver – Accomplish.

   Refer to ROCKET ESCAPE MANEUVER PROCEDURES, this Section.

7. Armament trigger - Release.

8. AIR-2A post-attack procedures – Accomplish.

ROCKET RADAR PURSUIT ATTACK

1. AIR-2A armament selection – Accomplish.

2. LC/PUR switch – Press and release.

3. RDR switch – NORM.

4. Radar range lockon – Accomplish.

5. Attack steering – Accomplish.

   If flying manually, fly to center steering dot in reference circle.

6. Deleted.

7. Armament trigger – Press to second detent and hold at cycling firing bar indication.

   The AIR-2A will be ejected after a 1.20-second delay.

8. Radar scope – Monitor for "8" pullout signal.

9. Escape maneuver – Accomplish.

   Refer to ROCKET ESCAPE MANEUVER PROCEDURES, this Section.

10. Armament trigger – Release.

11. LC/PUR switch – Press and release.

12. AIR-2A post-attack procedures – Accomplish.

ROCKET RADAR PURSUIT/LEAD COLLISION ATTACK

1. Steps 1 thru 5 of ROCKET RADAR PURSUIT ATTACK – Accomplish.

2. LC/PUR switch – Press and release.

3. Steps 3 thru 8 of ROCKET LEAD COLLISION ATTACK – Accomplish.

ROCKET RADAR PURSUIT ATTACK, HOM MODE

1. AIR-2A armament selection – Accomplish.

2. RDR switch – HOM.

   If jamming is detected and is too severe to permit normal lockon, place RDR switch to HOM to permit angle tracking (ATOT).

3. ATOT – Accomplish.

4. Attack steering – Accomplish.

   If flying manually, fly the steering dot to provide elevation separation until burn through.

5. Target at firing range bar – Observe on 4 mile radar scope.

   If the target can be detected through jamming, observe target blip coincident with firing range bar. If target cannot be detected, attempt optical sight ranging.

6. Armament trigger – Press to second detent and hold.

When target reaches the firing range bar, press the armament trigger to the second detent and hold. The AIR-2A will be ejected after a 1.20-second delay.

7. Radar scope – Monitor for "8" pullout signal.

8. Escape maneuver – Accomplish.

   Refer to ROCKET ESCAPE MANEUVER PROCEDURES, this Section.

9. Armament trigger – Release.

10. AIR-2A post-attack procedures – Accomplish.

ROCKET RADAR PURSUIT/LEAD COLLISION ATTACK, HOM MODE

1. Steps 1 thru 4 of ROCKET RADAR PURSUIT ATTACK, HOM MODE – Accomplish.

2. Radar scope – Monitor for target burn-through.

   As the range to target decreases, radar range lockon may be possible. When target is visible, accomplish range lockon.

3. Steps 3 thru 8 of ROCKET LEAD COLLISION ATTACK – Accomplish.

ROCKET IR PURSUIT ATTACK

1. AIR-2A armament selection – Accomplish.

2. RDR switch – NORM.

3. IR STOW switch – RDR SCAN or RDR SLVD.

4. IR lockon – Accomplish.

5. Attack steering – Accomplish.

   Fly to center steering dot in reference circle.

6. Deleted.

7. Armament trigger – Press to second detent and hold when the target is at the proper range.

   The AIR-2A rocket will be ejected after a 1.20-second delay.

8. Radar scope – Monitor for "8" pullout signal.

9. Escape maneuver – Accomplish.

   Refer to ROCKET ESCAPE MANEUVER PROCEDURES, this Section.

10. Armament trigger – Release.

11. AIR-2A post-attack procedures – Accomplish.

ROCKET IR PURSUIT/RADAR LEAD COLLISION ATTACK

1. Steps 1 thru 5 of ROCKET IR PURSUIT ATTACK – Accomplish.

NOTE

The IR STOW switch must be set to RDR SLVD at this point. The antenna will then be slaved to the seeker head and radar lock-on may be possible.

2. Radar scope – Monitor for radar contact.

   When radar target blip becomes visible, accomplish radar range lockon.

3. Radar IR SELECT button – Press and release.

   After radar lockon is accomplished, this action will effect the transfer to a radar dominant mode.

4. Steps 3 thru 8 of ROCKET LEAD COLLISION ATTACK – Accomplish.

ROCKET OPTICAL PURSUIT ATTACK

If the target can be detected visually and no other means can be utilized to determine firing range (as in lead collision or pursuit attack) the optical sight may be used for firing range determination. For optical sight firing, use the following procedure:

1. Optical sight – Unstow.
2. Reticle selector ring – As required.

   Set the reticle selector ring to the reticle corresponding to target size and aircraft altitude.

   WARNING

   Anytime the system is in manual pursuit (using LC/PUR switch), optical sight is unstowed, and SPL WPN selected, the reticle selection determines AIR-2A time of flight. Proper reticle selection must be made to insure proper AIR-2A time of flight. Failure to do so may put the interceptor within the AIR-2A lethal zone.

3. AIR-2A armament selection – Accomplish.
4. LC/PUR switch – Press and release.
5. Pursuit course to firing position – Accomplish.

   Fly the interceptor to a point where the target wing span fills the reticle.

6. Armament trigger – Press to second detent and hold.

   When target fills reticle, press the armament trigger to the second detent and hold. The AIR-2A rocket will be ejected after a 1.20-second delay.

7. Escape maneuver – Accomplish.

   Refer to ROCKET ESCAPE MANEUVER PROCEDURES, this Section.

8. Armament trigger – Release.
9. LC/PUR switch – Press and release.
10. Optical sight – Stow.
11. AIR-2A post-attack procedures – Accomplish.

ROCKET ESCAPE MANEUVER PROCEDURES

WARNING

- The following safe escape procedures must be used when the AIR-2A rocket is launched. Failure to use these procedures may result in destruction of, or severe damage to, the aircraft.
- To prevent flash blindness, concentrate on the instrument panel after AIR-2A rocket launch.

LOW ALTITUDE ESCAPE MANEUVER

Immediately after AIR-2A launch below 2000 feet AGL (2000-5000 feet AGL optional) execute following escape maneuver, (figure 2-6A).

1. AIR-2A Rocket Launch – First establish a 60-degree bank angle, then pull maximum load factor as rapidly as possible.
2. Escape Maneuver Completion – Continue maximum G climbing turn for at least 135 degrees of turn.

NOTE

When executing the low altitude escape maneuver, the aircrew may receive a nuclear radiation dose of not greater than 20 REM. This dosage is safe and well below the ADC limit of 100 REM.

LEVEL TURN ESCAPE MANEUVER

Immediately after AIR-2A rocket launch at coaltitude between 2,000 and 45,000 feet (figure 2-6B), execute the following maneuver.

WARNING

When flying between 5,000 and 30,000 feet, do not exceed 4 g's, as damaging gust loads on the aircraft may result.

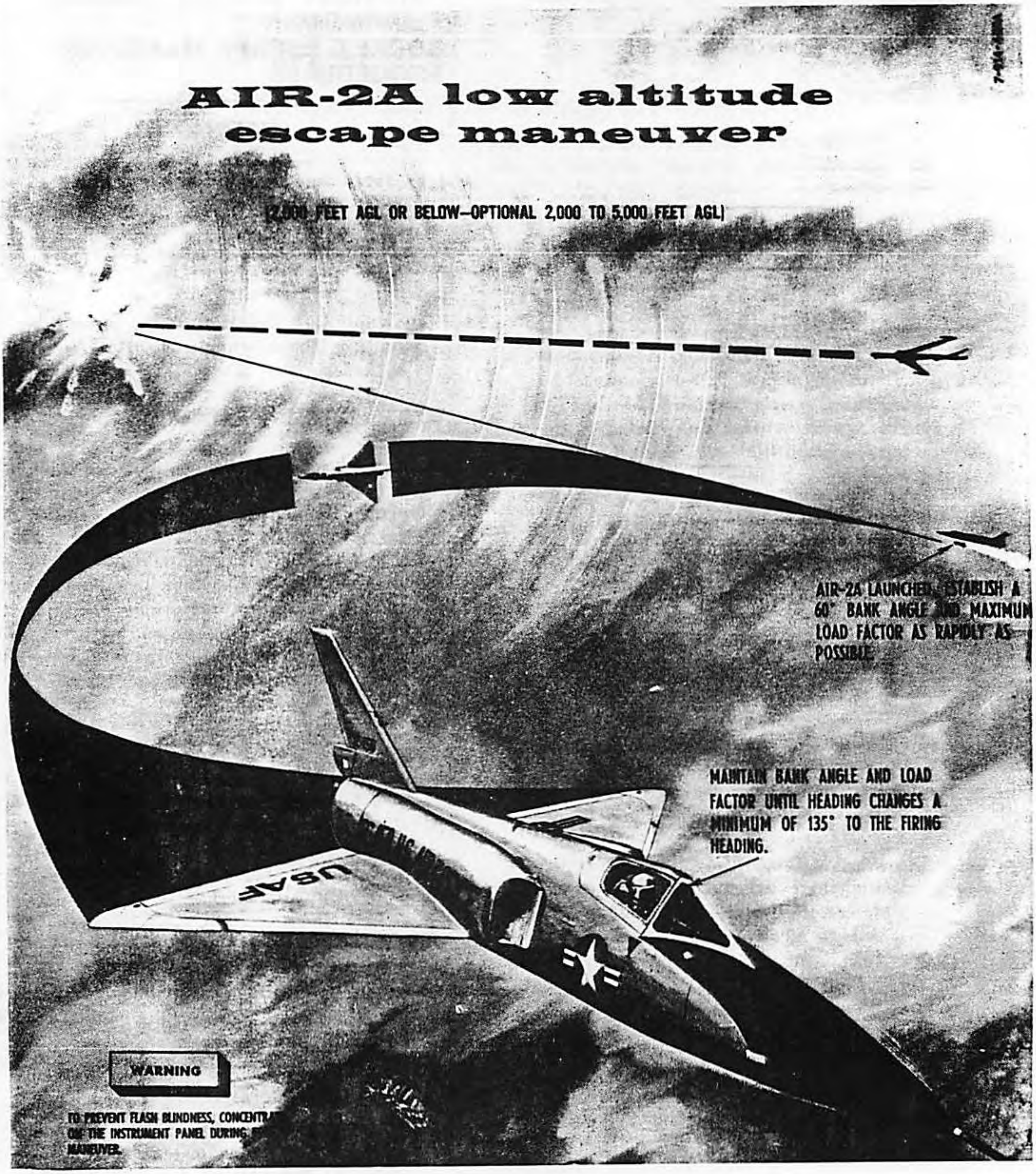

AIR-2A low altitude escape maneuver
(2,000 FEET AGL OR BELOW—OPTIONAL 2,000 TO 5,000 FEET AGL)
AIR-2A LAUNCHED. ESTABLISH A 60° BANK ANGLE AND MAXIMUM LOAD FACTOR AS RAPIDLY AS POSSIBLE.
MAINTAIN BANK ANGLE AND LOAD FACTOR UNTIL HEADING CHANGES A MINIMUM OF 135° TO THE FIRING HEADING.
USAF
WARNING
TO PREVENT FLASH BLINDNESS, CONCENT
ON THE INSTRUMENT PANEL DURING
MANEUVER.

Figure 2-6A.

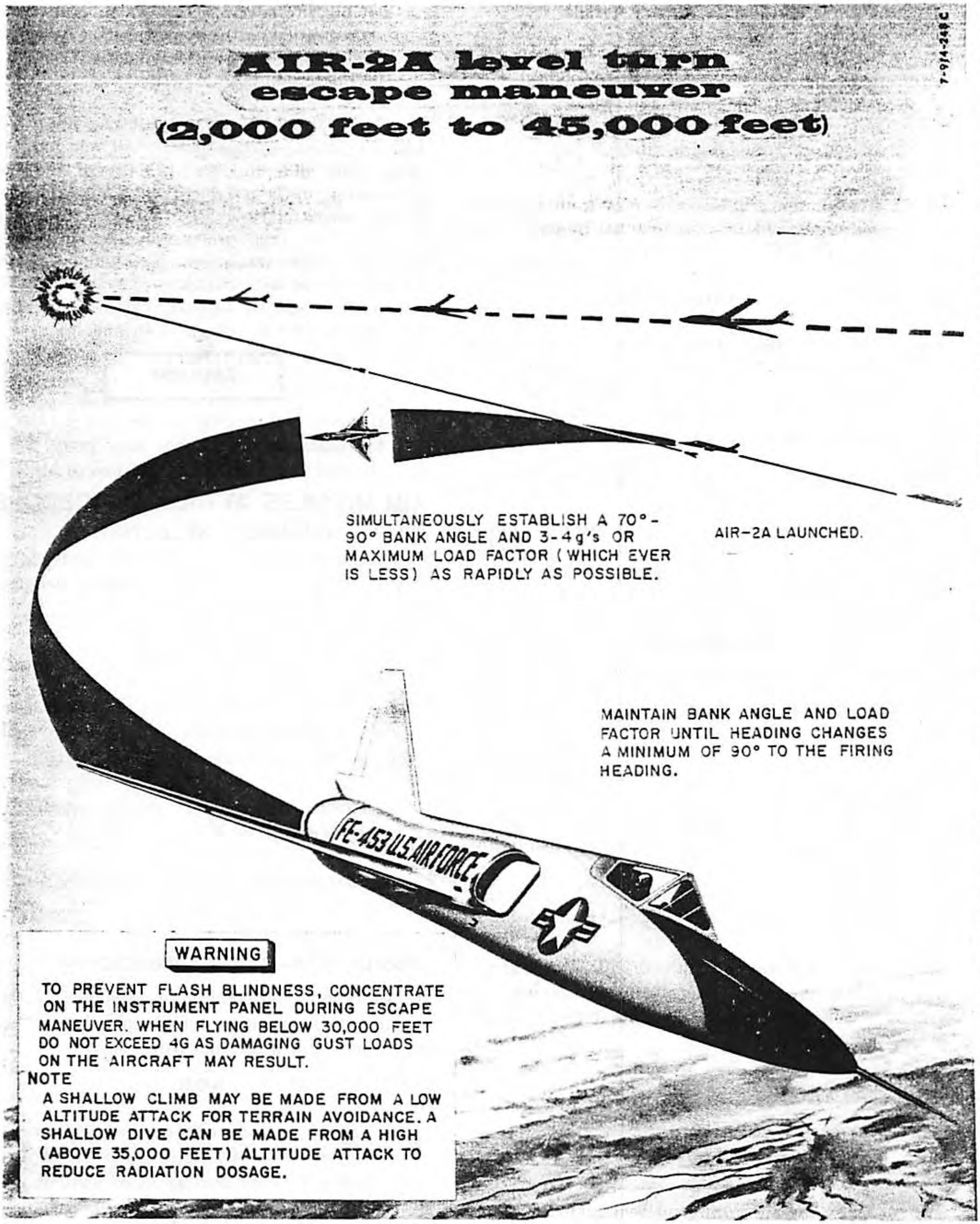


Figure 2-6B. 2-6B

1. Bank angle and load factor— Establish as rapidly as possible.

   Simultaneously establish a 70-90 degrees bank angle and 3-4 g's or maximum load factor (whichever is less) as rapidly as possible.

2. Bank angle and load factor – Maintain until minimum 90 degrees heading change from firing heading is accomplished.

NOTE

A shallow climb may be made from a low-altitude attack for terrain avoidance. A shallow dive may be made from a high (above 35,000 feet) altitude attack to reduce radiation dosage.

MODIFIED SPLIT–S ESCAPE MANEUVER

Immediately after AIR-2A rocket launch from a snap-up attack or from an altitude above 45,000 feet (figure 2-6C), execute the following escape maneuver:

WARNING

To prevent flash blindness, concentrate on the instrument panel during escape maneuver.

1. 135-180 degrees bank angle – Establish as rapidly as possible.
2. Maximum load factor – Establish as rapidly as possible.
3. Dive attitude and recovery – Accomplish.

   When a dive attitude of 30 - 40 degrees has been reached, roll out and recover to level flight, using approximately 2-g's load factor.

MSR ATTACK

WARNING

When using the MSR, no armament will be aboard. If armament panel safety wire has not been removed prior to flight, the - pilot must not break any wire or seals. The flight must be conducted as if the aircraft is loaded.

The MSR attack pass is flown exactly as is the AIR-2A attack except that at 10 seconds before firing time the pilot should observe and record interceptor mach and target altitude. The system should be armed at least 15 seconds prior to initiation of the fire signal during an MSR intercept. After the attack pass when the special weapon release lock switch is placed in the LOCK position, the special weapon release lock indicator will indicate LOCK instead of striped.

CAUTION

If the special weapon release lock indicator indicates stripped, the pass should be aborted to avoid inadvertent loss of MSR.

AIM MISSILES ATTACK PROCEDURES

MISSILE ARMAMENT SELECTION

In subsequent procedures in this section, when instructed to select missiie armament, accomplish the following steps:

1. Arm-safe switch – SAFE.

   Before changing the position of the armament selector switch, ensure that the arm-safe switch is in SAFE position.

2. Armament selector switch – MISSILES RAD, ALL, or IR.
3. Armament selection indicator – OK.
4. Arm-safe switch – ARM.

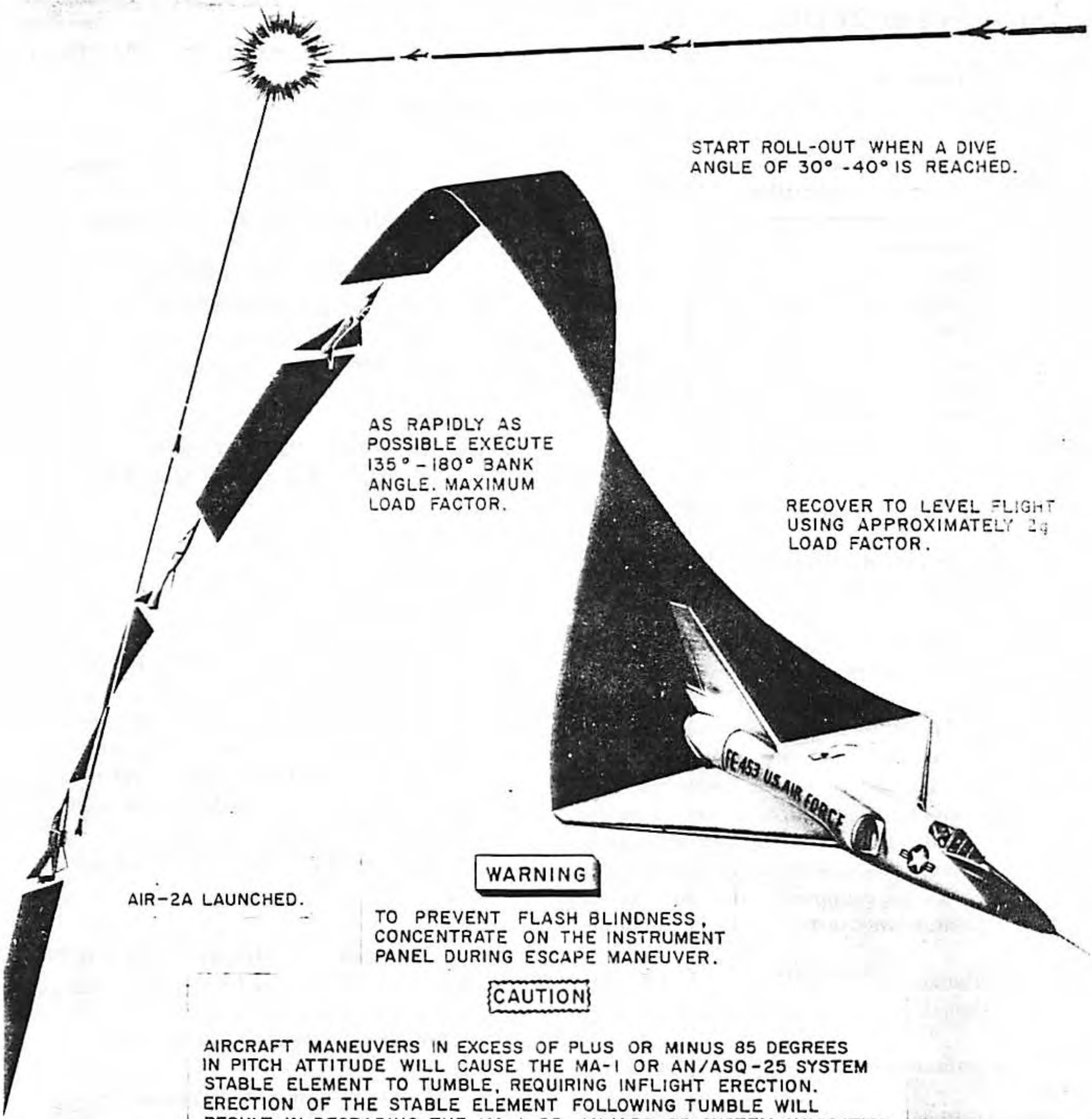
MISSILE POST–ATTACK PROCEDURES

In subsequent procedures in this section, when instructed to perform missile post-attack procedures, accomplish the following steps:

NOTE

Wait 15 seconds after releasing armament trigger before setting the arm-safe switch to SAFE or the armament selector switch to VIS IDENT to prevent an unscheduled reopening of the missile bay doors.

AIR-2A modified split-s escape maneuver (snap-up or above 45,000 feet)



WARNING

TO PREVENT FLASH BLINDNESS, CONCENTRATE ON THE INSTRUMENT PANEL DURING ESCAPE MANEUVER.

CAUTION

AIRCRAFT MANEUVERS IN EXCESS OF PLUS OR MINUS 85 DEGREES IN PITCH ATTITUDE WILL CAUSE THE MA-1 OR AN/ASQ-25 SYSTEM STABLE ELEMENT TO TUMBLE, REQUIRING INFLIGHT ERECTION. ERECTION OF THE STABLE ELEMENT FOLLOWING TUMBLE WILL RESULT IN DEGRADING THE MA-1 OR AN/ASQ-25 SYSTEM NAVIGATION AND ATTACK CAPABILITIES.

Figure 2-6C.

1. Arm-safe switch – SAFE.
2. Armament selector switch – VIS IDENT.
3. AUTO SEARCH button – Press and release.

MISSILE LEAD COLLISION ATTACK

1. Missile selection – Accomplish.
2. Radar range lockon – Accomplish.

CAUTION

The action switch should not be held pressed except when required to effect lockon. Indiscriminate use of the action switch causes needless missile preparation. Approximately 20 minutes of continuous preparation can result in overheating the missiles.

3. Attack steering – Accomplish.

   If flying manually, fly to center steering dot in reference circle.

4. Armament trigger – Press to second detent and hold.

   At 20 seconds before time to fire, indicated by the time-to-go circle beginning to shrink, press the armament trigger to the second detent and hold. This will complete missile preparation and allow the fire signal to be supplied to the missile launching system by the computer.

5. Radar scope – Monitor for "X" fire signal.
6. Pullout maneuver – Accomplish.

   Refer to MISSILE PULLOUT MANEUVER PROCEDURES, this Section.

7. Armament trigger – Release.
8. Missile post-attack procedures - Accomplish.

MISSILE RADAR PURSUIT ATTACK

1. Missile selection – Accomplish.
2. LC/PUR switch - Press and release.
3. RDR switch – NORM.
4. Radar range lockon – Accomplish.
5. Attack steering – Accomplish.

   If flying manually, fly to center steering dot in reference circle.

6. Deleted.
7. Armament trigger – Press to second detent and hold at cycling firing bar indication.
8. Radar scope – Monitor for "X" fire signal.

   Appearance of the fire signal indicates that the selected missiles have been launched. The signal will remain for the computed missile time of flight.

9. Pullout maneuver – Accomplish.

   Refer to MISSILE PULLOUT MANEUVER PROCEDURES, this Section.

10. Armament trigger – Release.
11. LC/PUR switch – Press and release.
12. Missile post-attack procedures – Accomplish.

MISSILE RADAR PURSUIT ATTACK, HOM MODE

1. RDR switch – HOM.

   If jamming is detected and is too severe to permit radar range lockon, set mode selector switch to HOM to permit ATOT.

2. Missile selection – Accomplish.

3. Radar angle lockon – Accomplish.

4. Attack steering – Accomplish.

   If flying manually, fly to center steering dot in reference circle.

5. Target at firing range bar – Observe on radar scope.

   If the target can be detected through jamming, observe target blip coincident with firing range bar. If target cannot be detected, attempt optical sight ranging.

6. Armament trigger – Press to second detent and hold.

   When the target coincides with the firing range bar, press the armament trigger to the second detent and hold.

7. Radar scope – Monitor for appearance of "X" fire signal.

   Appearance of the fire signal indicates that the selected missiles have been launched. The signal will remain for the computed missile time of flight.

8. Pullout maneuver – Accomplish.

   Refer to MISSILE PULLOUT MANEUVER PROCEDURES, this Section.

9. Armament trigger – Release.

10. Missile post-attack procedures – Accomplish.

MISSILE IR PURSUIT ATTACK

1. RDR switch – As desired.

2. IR STOW switch – RDR SCAN or RDR SLVD.

3. Missile selection – Accomplish.

4. IR lockon – Accomplish.

5. Attack steering – Accomplish.

   In the IR pursuit attack there is no provision for automatic flight. The pilot must manually fly to center the steering dot in the reference circle.

6. Deleted.

7. Armament trigger – Press to second detent and hold, when at proper range.

8. Radar scope – Monitor for fire signal.

9. Pullout maneuver – Accomplish.

   Refer to MISSILE PULLOUT MANEUVER PROCEDURES, this Section.

10. Armament trigger – Release.

11. Missile post-attack procedures – Accomplish.

MISSILE IR/RADAR PURSUIT ATTACK

1. Steps 1 thru 5 of MISSILE IR PURSUIT ATTACK – Accomplish.

NOTE

At this point the IR STOW switch must be set to RDR SLVD. The antenna will then be slaved to the IR seeker head and radar lockon will be possible.

2. Radar lockon – Accomplish.

If radar target becomes visible, accomplish radar lockon in DROT and/or ATOT.

3. Steps 6 thru 10 of MISSILE IR PURSUIT ATTACK – Accomplish.

MISSILE OPTICAL PURSUIT ATTACK

If the target can be detected visually and no other means can be utilized to determine firing range (as in lead collision or pursuit attack), the optical sight may be used for firing range determination. For optical sight firing, use the following procedure:

1. Optical sight – Unstow.
2. Reticle selector ring – As required.

   Select the reticle corresponding to target size and interceptor altitude.

3. Missile selection – Accomplish.
4. LC/PUR switch – Press and release.
5. Attack steering – Fly manually to position interceptor so that target is centered in selected reticle.
6. Armament trigger – Press to second detent and hold.

   The missiles will be launched 4.3 seconds later provided 10 seconds of missile preparations have been accomplished.

7. Radar scope – Monitor for fire signal.
8. Pullout maneuver – Accomplish.

   Refer to MISSILE PULLOUT MANEUVER PROCEDURES, this Section.

9. Armament trigger – Release.
10. LC/PUR switch – Press and release.
11. Optical sight – Stow.
12. Missile post-attack procedures – Accomplish.

MISSILE PULLOUT MANEUVER PROCEDURES

1. Level turn – Initiate a smooth turn at missile launch.

   The turn should be smooth and gentle to avoid pulling the radar antenna off the target in any heavy ground clutter or chaff environment. The turn should be small enough to avoid driving the antenna into the bumper ring prior to missile impact.

NOTE

When radar missiles are fired, the interceptor's radar must illuminate the target for missile guidance until impact.

2. Modified split-S – Accomplish.

   When missiles are fired in a snap-up configuration the pullout procedure should include a modified split-S.

WSEM ATTACK

WARNING

When using the WSEM, no armament will be aboard. If armament panel safety wire has not been removed prior to flight, the pilot must not break any wire or seals. The flight must be conducted as if the aircraft is loaded.

The WSEM attack pass is flown exactly as is the missile attack except that at firing time the pilot should observe and record the following:

a. Target altitude.

b. Interceptor mach.

c. Closing rate.

d. Antenna train angle.

Do not return to search, disarm or select vis ident for 25 seconds after the doors have closed to insure zero calibrate. This will permit adequate WSEM tape evaluation. Following an attack pass, after returning to search, setting the arm-safe switch to SAFE and selecting VIS IDENT, press and hold the armament recycle switch for 5 seconds, then release.

CAUTION

Do not make WSEM pursuit passes of greater than 4 minutes duration or excessive heating will damage the WSEM. After each pass, allow 20 minutes cooling time for the WSEM before beginning another attack.

The WSEM has a usable tape capability of approximately 130 seconds. It will start at A-time and run for 20 seconds. At B-time, in a lead collision attack (B and C by trigger in pursuit), it will again start and continue until the return to search, disarm, selecting VIS IDENT or 22 seconds after the fire signal. A normal pass uses approximately 30 to 45 seconds of tape.

ABORTED ATTACK PROCEDURES

Immediately after an aborted attack, accomplish the following procedures:

NOTE

Wait 15 seconds after releasing armament trigger before placing arm-safe switch to SAFE or armament selector switch to VIS IDENT, to preclude unscheduled reopening of the missile bay doors.

1. Armament trigger – Release.
2. AUTO SEARCH button – Press.
3. Arm-safe switch – SAFE.
4. Special weapon release lock switch – LOCK.
5. Special weapon release lock indicator – LOCK.
6. Armament selector switch – VIS IDENT.

NOTE

If an attack is aborted after C-time (doors open), the thermal batteries are expended; therefore, armament internal power is not available for a second attack. Make appropriate entry on Form 781.

PRACTICE AIR-2A INTERCEPTS WITH SECONDARY ARMAMENT

WARNING

Make only practice AIR-2A intercepts with secondary armament aboard. Attempt no missile attacks. An intercept will not be attempted if there is any indication of a fire control system malfunction. Attempt no practice intercepts with an AIR-2A aboard.

1. ARM-SAFE switch SAFE. Guard safetied and sealed.
2. Special weapon release lock switch – LOCK. Guard safetied and sealed.
3. Armament selector switch – SPL WPN.
4. Perform practice AIR-2A intercept.

WARNING

If, with secondary armament aboard, the missile bay doors begin to open, immediately release the trigger (if pressed), return radar to search, select VIS IDENT, wait 75 seconds minimum, then close missile bay doors and immediately return to base.

5. At fire signal perform escape maneuver.

6. Armament selector switch – VIS IDENT.

7. AUTO SEARCH button – Press and release.

DESCENT

The recommended descent speeds as given in T.O. 1F-106A-1-1 should be followed.

1. Fuel quantity — Check.
2. Cabin temperature control knob — AUTOMATIC HEAT or HOT.

   Cabin heat should be as warm as possible to reduce possibility of fogging.
3. Altimeter—Reset at designated altitude.
4. Standby flight instruments – Check for normal operation.

   Reset standby altimeter at designated altitude and compare standby flight instruments with primary flight instruments.
5. IFF/SIF – Checked.

   Within one hour prior to the estimated time of landing, a positive IFF check should be made with an air traffic control facility for normal IFF operation.
6. Boost pump switches — ON.
7. Arm-safe switch — SAFE.
8. Armament selector switch — VIS IDENT.
9. Special weapon release lock switch — LOCK.
10. Special weapon release lock indicator — "LOCK."
11. Hydraulic pressures — Check.
12. Cockpit no-fog and ventilated suit switch — As desired.

12A. Canopy antifog switch–As required.

13. Pressure suit vent control handle—Cracked.

    Move the pressure suit vent control handle to the OFF position and crack approximately one-half inch when wearing the full pressure suit.
14. Idle thrust control switch – Check OFF.

    Although there is no necessity to actuate this switch during flight, make a positive check, before landing, that it has not been inadvertently moved from the OFF position.

WARNING

If the idle thrust control switch is ON and main landing gear touchdown occurs, the exhaust nozzle will open. Should this occur during a military thrust go-around, the resulting thrust loss will seriously affect the success of the go-around.

15. Shoulder harness inertial reel handle – AUTOMATIC. (FP-RP)
16. Warning lights dimmer switch – As necessary.

BEFORE LANDING

The normal landing pattern entry speed can vary over a wide range; however, the most desirable speed range is from 275 to 350 KCAS. At the break, altitude should be 1500 feet above field elevation and speed brakes extended if desired. Thrust should be reduced to obtain desirable gear-down speed. The downwind leg should be flown at 240 KCAS and altitude maintained at 1500 feet above field elevation. Prior to turning base, extend the landing gear and check for a safe indication. As the turn-on to base leg is started, a descent should be established and airspeed should be allowed to reduce to 200 KCAS (minimum) on the base leg. Refer to the Landing Distances Charts in T.O. 1F-106A-1-1 for recommended final approach, prior to flare, and touchdown speeds. See figure 2-7 for normal landing pattern.

1. Landing gear handle — DOWN and check lights. (FP-RP)

   Put the landing gear down and check landing gear green light illuminated, landing gear red warning light out, and audio warning off.
2. Landing and taxi light switch — LANDING LIGHTS.
3. Flight mode selector switch – DIR MAN.

NOTE

Refer to Section V for maximum landing gear extension speeds and tire ground limit speeds.

LANDING

CAUTION

Avoid wake turbulence. Allow a minimum of two minutes separation before landing behind a heavy aircraft or helicopter and 4 minutes behind extremely heavy aircraft such as C-5A and Boeing 747. With effective crosswinds of over 5 knots, the interval may be reduced, but attempt to remain above and upwind of the preceding aircraft's flight path. Wake turbulence is most dangerous during the approach and flare prior to touchdown with calm or light crosswinds.

LANDING WITH EXTERNAL WING TANKS

Landings can be safely accomplished with external wing tanks full or partially full. Pay close attention to sink rate and keep rate of descent at touchdown below 300 fpm.

NORMAL LANDING

Refer to T.O. 1F-106A-1-1 for charts showing recommended approach and touchdown speeds and landing distances for various gross weights. If airspeed becomes excessively low, a high rate of sink may develop, resulting in a hard landing. During the flare, thrust is reduced to idle and touchdown is made with approximately 12° angle of attack (nose on the horizon). See figure 2-7 for typical landing pattern. The following procedures should be employed:

1. Throttle — IDLE, during flareout.
2. Touchdown speed — As required.

CAUTION

- Angle of attack must be kept under 17° to prevent scraping the tail and inadvertent tailhook engagement.
- During touchdown with higher than recommended rates of descent, the tail section or tailhook guards may scrape the runway at angles of attack less than 17°. This could cause inadvertent tailhook engagement of the BAK-6, -9 or -12 arresting cable.

WARNING

Be careful not to inadvertently depress the brake pedals as locked brakes and blown tires may result from very slight brake application on touchdown.

3. Drag chute handle — Pull.

   The drag chute may be deployed as soon as the airplane touches down; the chute requires approximately three seconds for deployment.

CAUTION

Be careful not to depress the tailhook down button while deploying the drag chute.

4. Lower nose wheel to runway.

   The nose wheel should be lowered to runway at approximately 90 to 100 KCAS while elevator control remains effective. For other considerations when lowering the nose, refer to CROSS-WIND LANDING, LANDING ON SLIPPERY RUNWAYS, and MINIMUM RUN LANDING, this Section.

NOTE

- Lower nose gear to runway before applying brakes.
- Rudder is effective for directional control during the early, high-speed portion of the landing roll; however, nose wheel steering may be used as necessary any time after nose wheel touchdown.
- If nose wheel shimmy is encountered, disengage nose wheel steering, and maintain directional control with rudder and brakes. (Nose wheel steering may be reengaged below 50 KCAS if shimmy has ceased.)

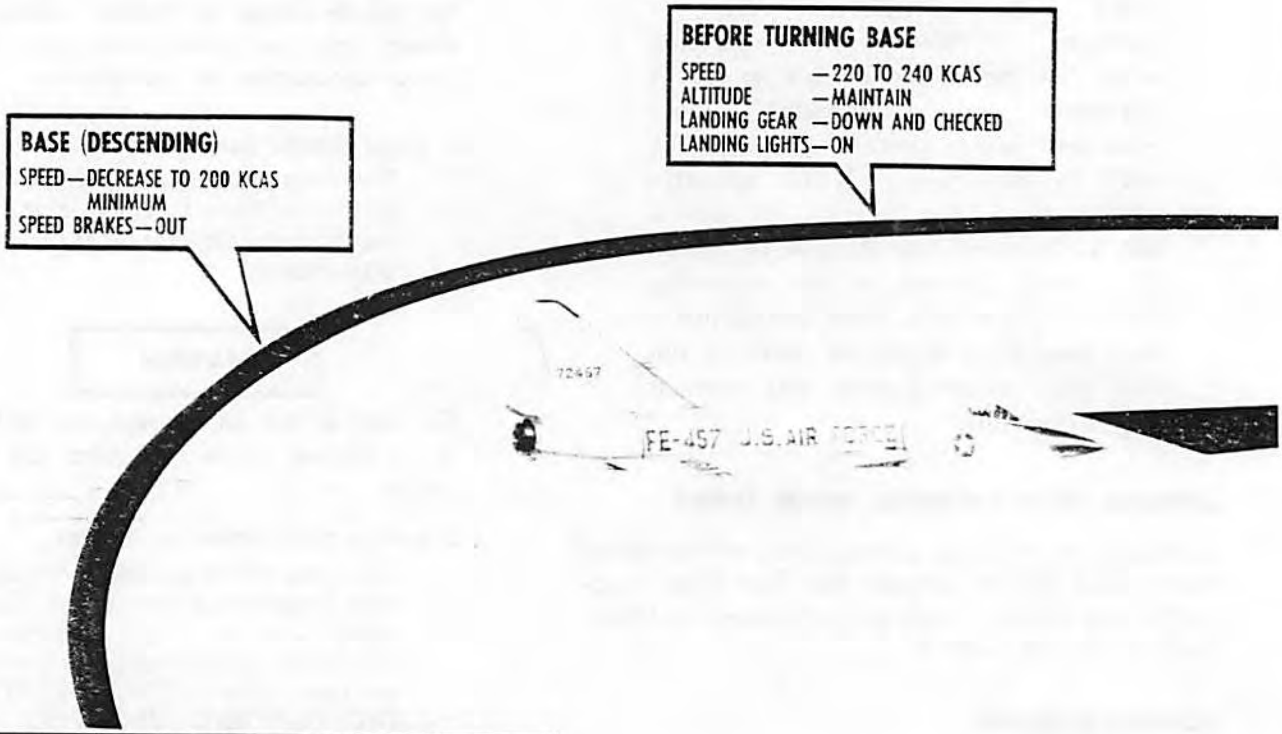
5. Idle thrust control switch – ON.

WARNING

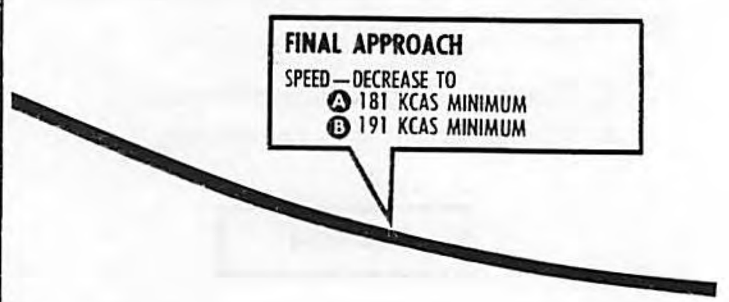
The switch must not be placed ON until it is positively determined that go-around will not be attempted.

6. Braking — Apply smoothly as necessary.

normal landing



RECOMMENDED MINIMUM SPEEDS - KCAS						
FUEL REMAINING LBS	F-106A			F-106B		
	FINAL APPROACH	PRIOR TO FLARE	TOUCH-DOWN	FINAL APPROACH	PRIOR TO FLARE	TOUCH-DOWN
1000	179	166	147	190	177	156
2000	181	168	149	191	178	157
3000	184	171	151	192	178	158
4000	186	173	154	192	179	159
6000	191	177	158	193	180	160
8000	195	181	161	202	187	166
10,000	199	185	165	208	193	172
12,000	203	189	169	212	197	175
14,000	207	192	172	216	200	178



NOTE

- THESE SPEEDS INCLUDE COMPENSATION FOR FULL ARMAMENT. IF ARMAMENT LOAD IS EXPENDED, THESE SPEEDS CAN BE DECREASED BY 6 KNOTS.
- LANDING SPEED DIFFERENCES AT VARIOUS GROSS WEIGHTS BETWEEN THE A AND B AIRPLANES CAN BE ATTRIBUTED TO THE DIFFERENCE IN THE LOCATION OF THE CENTER OF GRAVITY OF EACH AIRPLANE.

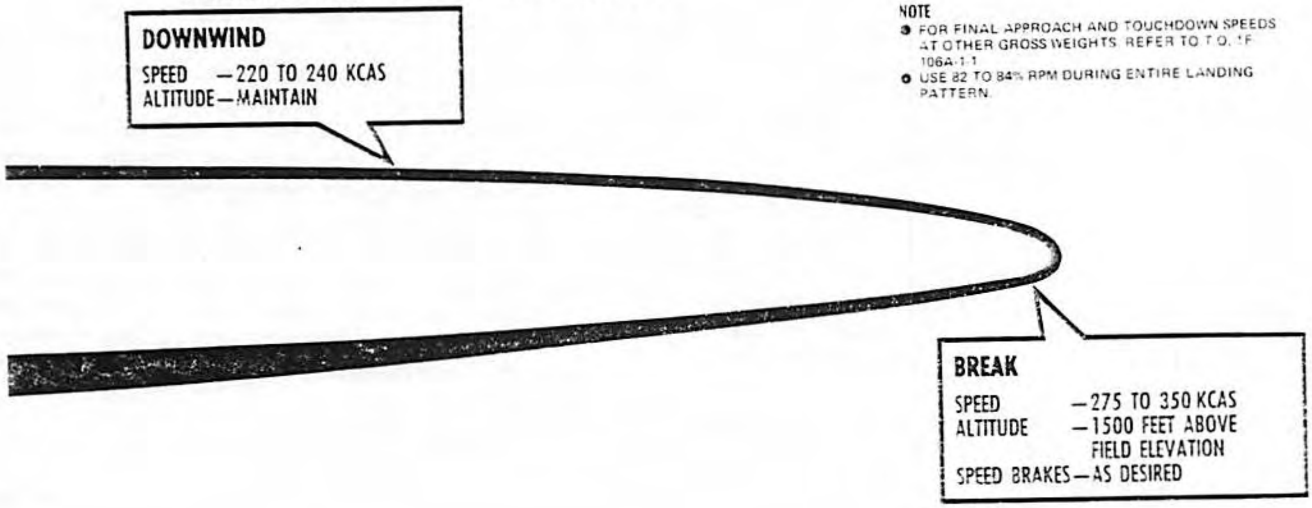
48044-1

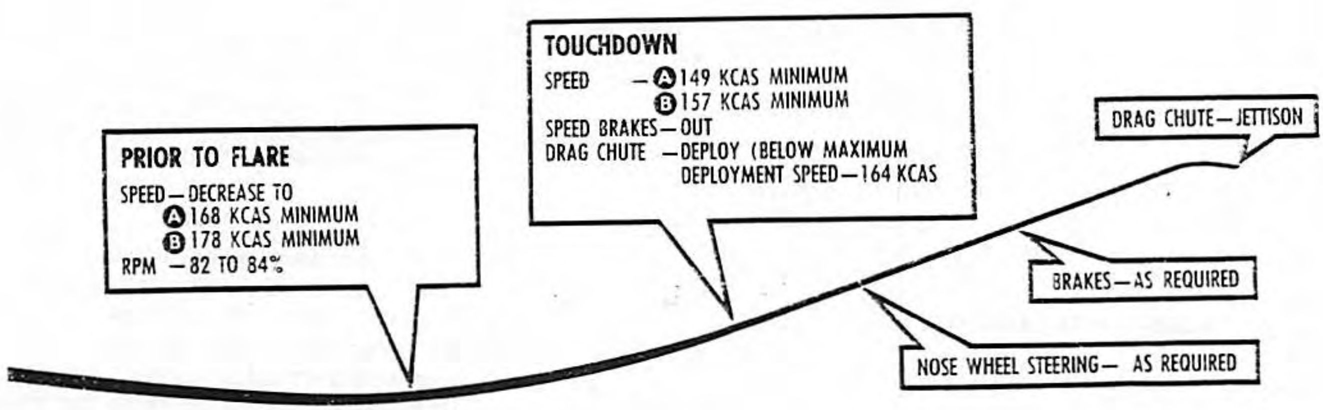
Figure 2-7

pattern (typical)

BASED ON 2000 POUNDS OF TOTAL FUEL REMAINING INCLUDING FULL ARMAMENT LOAD

NOTE
- FOR FINAL APPROACH AND TOUCHDOWN SPEEDS AT OTHER GROSS WEIGHTS REFER TO T.O. 1F-106A-1-1
- USE 82 TO 84% RPM DURING ENTIRE LANDING PATTERN.





WARNING

DO NOT REDUCE RPM BELOW 82% UNTIL STARTING FLARE.

48044-2

WARNING

To prevent skidding the tires and causing flat spots and possible blowout, do not apply brakes immediately after touchdown or at any other time when there is considerable lift on the wings. If maximum braking is required immediately after touchdown on a dry runway, lift should first be decreased as much as possible by lowering the nose before applying brakes. A heavy brake pressure can result in locking the wheels more easily if the brakes are applied immediately after touchdown than if the same pressure is applied after the full weight of the airplane is on the wheels. With the nose wheels on the ground, the wings provide negative lift which forces the wheels down against the runway, thus decreasing the possibility of locking the wheels by heavy braking.

NOTE

When practicable the full length of the runway should be used during landing roll to reduce brake heating and wear.

Directional Control During Landing Roll

Rudder, wheel brakes, and nose wheel steering should normally be used for directional control during the landing roll. However, directional control during the landing roll can be maintained by use of the ailerons. After the main landing gear is firmly on the runway, the airplane will turn in the direction of aileron selection.

NOTE

Push stick full forward for more positive nose wheel steering and to afford more aileron control.

CROSS-WIND LANDING

To determine recommended landing speeds for various cross-wind conditions, refer to the Takeoff and Landing Crosswind Chart in Section II, T.O. 1F-106A-1-1.

Before Touchdown

WARNING

- Anticipate restricted lateral control stick movement.
- Landing with a crosswind component of 15 knots or above is not recommended when wearing a full pressure suit, anti-exposure suit, or heavy winter flying clothes.

The traffic pattern for a cross-wind landing should be normal, making proper allowances for strength and direction of the crosswind. Proper runway alignment on the final approach can be maintained by crabbing or dropping one wing; however, a combination of the two is recommended just prior to flare. For dry runway landing, remove crab before touchdown using wing low technique to prevent side drift. Reduce sink rate to a minimum to accomplish smooth touchdown. At increased cross-wind components, sink rate must be minimized due to increase of side loads imposed on the landing gear. For slippery runways, touch down with a combination of crab and bank.

After Touchdown

Accomplish touchdown in the center of the runway. If desired, deploy the drag chute immediately after touchdown. Prior to nose wheel touchdown, directional control should be accomplished by coordinating ailerons and rudder rather than using crossed controls as used during the approach.

CAUTION

- If excessive weathervaning occurs after touchdown, lower the nose immediately and maintain directional control by using nose wheel steering, brakes, aileron, and rudder as necessary. Drag chute jettisoning may be necessary to retain directional control.
- If extreme crosswind conditions exist on landing, the nose should be lowered before deploying the drag chute.

NOTE

With the nose wheel on the ground, a combination of downwind aileron (left aileron to turn left) and nose wheel steering is the most effective means of correcting weathervaning during landing roll. Downwind aileron corrects for weathervaning and nose wheel steering compensates for side drift and maintains runway alignment.

GUST CORRECTION

When gusty wind conditions exist, regardless of wind direction, a correction factor should be added to the final approach, flare, and touchdown airspeeds to prevent inadvertent increase of the sink rate. An increased sink rate may be induced when correcting for gusts if the airspeed is at or below the minimum approach speeds for the gross weight. This gust correction factor is determined by increasing the final approach, flare, and touchdown speed by one-half the difference between steady wind and peak gust. (Wind 10 knots, gusts to 20 knots – difference = 10 knots × ½ = 5 knots gust factor.) This gust factor is added to the gross landing weight airspeed.

NOTE

- The gust factor is added to provide a safety margin to maintain a desired sink rate while flying the airplane through a series of accelerations. The accelerations can be equally severe whether they are produced by headwind, crosswind, or tailwind.
- Since the frequency or timing of gusts cannot be estimated with practical accuracy, it is possible for the airplane to arrive at the flare point with gust corrections added during an interval when gusts have stopped momentarily. Under such conditions, the touchdown point may occur further down the runway than planned. Therefore, wherever a correction factor is added for gusts or other accelerations, be prepared to accept a correspondingly higher flare speed with increased landing distance.

LANDING ON SLIPPERY RUNWAYS

Prior to landing on a slippery runway, determine the corrected stopping distance using the latest RCR (runway condition reading). The ground roll which is to be corrected shall be determined from the Landing Distance Chart in T.O. 1F-106A-1-1 and based upon "good" runway conditions.

NOTE

- If no RCR is available, use representative values of 23 for DRY runways, 12 for WET runways, and 5 for ICY runways.
- For ICAO reports, use RCR values of 23 for GOOD, 12 for MEDIUM, and 5 for POOR.

Maximum decelerating forces after touchdown are obtained during the landing roll by holding the nose high touchdown attitude, utilizing maximum aerodynamic braking until forces available from wheel braking become greater than the aerodynamic drag. The use of speed brakes or drag chute does not significantly affect the speed at which the nose should be lowered. Therefore, to obtain minimum ground roll, the landing technique is the same with or without a drag chute or whether the runway is wet, dry, or icy. The only variable is the nose lowering speed which is determined by runway condition. The highest practical angle of attack that can be used without scraping the tail is 16°. Assuming 16° angle of attack is reached and the RCR is 12 or greater, minimum ground roll is obtained by lowering the nose at 115 KCAS and applying maximum braking. If the RCR is less than 12, the nose high attitude should be held as long as possible — approximately 80 KCAS for Ⓐ airplanes and 100 KCAS for Ⓑ airplanes—to utilize maximum aerodynamic braking. After the nose wheel is on the runway, begin maximum braking. Since the landing touchdown angle of attack is approximately 12° at normal landing weight, 16° angle of attack must be obtained by gradually raising the nose after touchdown as the airplane decelerates. If slippery runway conditions are aggravated by a large crosswind component, the additional problems of sidedrift (due to hydroplaning) and weathervaning must be considered. Usually, the best course of action in this circumstance is to proceed to a more suitable alternate.

Wet Runways

Minimum run landing on a wet runway is obtained by using the same procedure as for a dry runway provided the RCR is 12 or above. If RCR is below 12 the nose high attitude should be maintained as long as possible after touchdown to utilize maximum aerodynamic braking. Maintain directional control with rudder and aileron adding nose wheel steering after the nose wheel is on the runway. During wet runway landings, dynamic and viscous hydroplaning becomes a factor to be considered. In order to develop dynamic hydroplaning, the runway must be covered or puddled with at least 0.1 to 0.3 inch of water. With the right conditions the tire can actually ride on the surface of the water and render braking and nose wheel steering completely ineffective. The dynamic hydroplaning speed is basically a function of tire pressure; therefore, the higher the tire pressure, the higher the speed required to hydroplane. With the tire design and pressures used for the F-106, hydroplaning speed is well above the 115 KCAS maximum recommended braking speed used for wet runways. Above this speed, the rudder is still effective for maintaining directional control. Partial hydroplaning may occur at lower speeds, but due to the high tire pressures, the dynamic hydroplaning problem is significantly reduced. Viscous hydroplaning is controlled by the slickness of the runway surface. It can occur at any speed with very small amounts of moisture on the runway. It is a factor in the touchdown area of the runway where a rubber buildup has made the runway surface relatively slick. Braking effectiveness may be intermittent and control difficulties may develop, even at very low ground speeds.

Icy Runways

The procedure for an icy runway (RCR below 5) landing is that used for a wet or dry runway minimum run landing. However, since wheel braking effectiveness is reduced further from that for wet runway, maximum aerodynamic braking down to 80 KCAS provides the shortest stopping distance. Therefore, use maximum aerodynamic braking by maintaining the nose high attitude, 16° maximum, until the control stick is in the full aft position. Hold the full aft position until the nose lowers to the runway at approximately 80 KCAS for Ⓐ airplanes and 100 KCAS for Ⓑ airplanes. Begin optimum braking after the nose wheel contacts the runway.

NOTE

- With the nose wheel off of the runway, good directional control is available by using rudder and ailerons for normal crosswind conditions.
- With the nose wheel on the runway, directional control may be more difficult due to reduced nose wheel steering and control surface effectiveness. Combinations of nose wheel steering, aileron deflection, rudder and wheel braking may be required.

HEAVY-WEIGHT LANDING

Refer to T.O. 1F-106A-1-1 for approach and touchdown speeds at varying gross weights, and note increase in rollout required at higher gross weights.

CAUTION

During heavy weight landings on Ⓑ airplanes, the cg may be near the aft limit. This condition can cause critical angle of attack and result in scraping the tail on landing unless extreme caution is used. This may also cause the nose wheel strut to extend and make nose wheel steering intermittent. This condition is most prevalent when the cg of the airplane has moved aft due to F tank feeding.

Use of Tail Hook

Although the tail hook is primarily an emergency system, it should be utilized without hesitation anytime that runway length, braking conditions, and/or heavy gross weights combine to extend landing rollout distances. It is far safer to engage the barrier squarely with the airplane under control than to blow a tire by excessive braking. The hook should be extended approximately 2000 feet from the barrier.

MINIMUM-RUN LANDING (DRY RUNWAY)

Use normal approach pattern and speeds. Retard the throttle to IDLE during flare. The drag chute may be deployed the instant before touchdown. After touchdown, place the idle thrust control switch ON. Maintain landing attitude angle of attack (16° maximum) to obtain maximum aerodynamic braking. At 115 KCAS, lower the nose wheel to the runway and apply optimum braking. After the nose wheel is on the runway and optimum braking is applied, additional aerodynamic braking is obtained by holding maximum back stick (without raising the nose wheel from the runway).

CAUTION

- Braking is permitted above 100 KCAS; however, caution should be used to prevent wheels from sliding, as brake action is extremely difficult to feel above this speed.
- Excessive use of the brakes at high speeds will cause overheating of the brakes and tires. If the heat is great enough it will cause the tires to weaken and blow out.

TOUCH-AND-GO LANDINGS

Touch-and-go landings should be held to a minimum because of excessive tire wear.

WARNING

During touch and go landings, ascertain that the idle thrust control switch is OFF. Otherwise if the switch is ON when main landing gear touchdown occurs, the exhaust nozzle will open. The resulting thrust loss will seriously affect the success of a subsequent military thrust takeoff.

GO-AROUND

A go-around may be made from any point in the approach. Inasmuch as a relatively high engine rpm is maintained throughout most of the pattern, thrust will usually be readily available when the throttle is advanced. However, this may not be true when go-around is initiated after thrust has been reduced to near idle. Under this condition, or at any time when the go-around is initiated after landing flare has been started, the application of thrust may not be able to overcome the sink rate that has been established and the airplane may be committed to a touchdown. If such is the case, normal flare and touchdown speeds should be maintained resulting in a normal touchdown followed by a normal takeoff after go-around thrust has adequately increased airspeed. No appreciable trim change is experienced when applying thrust or retracting landing gear. If the drag chute handle was pulled, the handle must be pushed in (to jettison the drag chute) and the speed brakes should be retracted when the throttle is advanced.

1. Throttle—Full military or maximum.

WARNING

- If go-around is initiated during the flare or after thrust has been reduced to near idle, any attempt to hold the airplane off the ground may result in critical angle of attack and unsafe airspeed. This condition will be shown on the angle of attack indicator (approach symbol below the lubber line) and can result in excessive sink rate or possible loss of airplane control. The same considerations apply if the airplane is prematurely pulled off the runway during a go-around after touchdown. If committed to touch down, maintain normal approach and touchdown speeds, accomplish a normal touchdown, lower the nose to allow airspeed to increase adequately, and continue with a normal takeoff. Use smooth and deliberate control applications throughout this phase of the go-around. Refer to STALLS, Section VI, for additional information.
- In ground effect (from ground level to approximately 40 feet), it is possible to lift off at speeds which can result in a critical angle of attack as the airplane leaves ground effect. If this situation develops, there may be insufficient time to reduce up-elevator adequately to avoid excessive sink rate or post-stall gyrations. Refer to Section II, T.O. 1F-106A-1-1 for recommended takeoff speeds at various gross weights. After initiating a go-around, do not increase altitude until the appropriate takeoff speed (or higher speed) has been achieved. This will assure that control during climbout can be maintained after leaving ground effect.

2. Drag chute handle — In.
3. Speed brakes — Closed.
4. Idle thrust — OFF.
5. Landing gear — UP (when airplane is definitely airborne).

NOTE

Approximate fuel required for go-around in which a distance of nine miles is traveled is 350 pounds for maximum thrust, or 300 pounds for military thrust. The amount of fuel required is based on maintaining the specified thrust until traffic pattern altitude and speed are attained; after which reduced thrust is utilized.

AFTER LANDING — CLEAR OF RUNWAY

1. Drag chute — Jettison.
2. Ejection seat ground safety pin — Install. (FP-RP)
3. Parachute—Disconnect and safety. (FP-RP).

WARNING B

Rear pilot should assure that front pilot has installed his ejection seat ground safety pin prior to pulling ditching control handle.

   a. Ditching control handle—Pull.
      Check that the firing lanyard disengages from the parachute disconnect.
   b. Ditching control handle—Restow (forward).
   c. Safety pin—Install in firing lanyard shell.
   d. Dust cap—Install.
   e. Firing lanyard—Stow in snap tab on harness.

4. Mode 4 selector knob—HOLD for 2 - 3 seconds, if required.

4A. IFF/SIF Master control knob—OFF.

NOTE

If the mode 4 code selector has been moved to HOLD wait at least 15 seconds before moving the IFF/SIF master control knob to OFF to prevent the code setting from being ZEROIZED.

5. Deleted.
6. Takeoff trim button — Depress.
7. RAT—Extend.
   The ram air turbine should be extended prior to reaching the parking area to preclude possible injury to ground personnel.
8. RAT handle—Up.

NOTE

The handle should be returned to the "in" or "up" position to prevent injury to ground personnel when the RAT door is closed.

9. Arm-safe switch—SAFE.
10. AIR-2A Arm/Safe/Monitor power circuit breaker — Open.
11. Armament Selector Switch—VIS IDENT.
12. Special weapon release lock switch—LOCK.

WARNING

If missiles are retracted after an aborted attack, advise the ground crew of the possibility of trapped missile power plant exhaust fumes in the missile bay. The exhaust fumes are toxic and must not be inhaled.

13. Cockpit no-fog and ventilated suit switch—OFF.
14. Landing and taxi light switch—Climatic.
15. Oxygen supply switch—OFF (immediately before removing mask or face plate). (FP-RP)
16. Radar intensity controls—Minimum. (FP-RP)
17. Cabin air selector switch—OFF.
18. Canopy—As desired.
19. Formation-navigation lights switch—Climatic.
20. Anti-icing, antifog, rain removal and pitot heat switches—OFF.

ENGINE SHUTDOWN

To shut down engine, proceed as follows:

CAUTION

Whenever the engine has been operated at high thrust settings for an appreciable length of time, it must be operated at idle for up to five minutes prior to shutdown in order to prevent seizure of the rotors.

1. Wheel chocks—Installed.
2. Compressed air—Connected or selected.
   External air will be connected by the crew chief. If not available, the crew chief will select internal air prior to engine shutdown.

NOTE

In the event of engine fire during shutdown, compressed air will be necessary to motor the engine.

3. Canopy—Fully open.
4. ATG switch—OFF
5. MA-1 power switch—OFF.

NOTE

Turn MA-1 power switch to OFF prior to engine shutdown to prevent MA-1 electrical power system surges which may trip some system circuit breakers.

6. Boost pumps—OFF.
7. T tank shutoff switch—CLOSE.

CAUTION

Excessive air pressure in the fuel pressurization system due to the tripping of the emergency air pressure regulator may cause structural damage to the wing if the T tank switch is not closed prior to engine shutdown.

8. Idle thrust control switch—OFF.
9. AC and dc generator switches—OFF.
   a. Emergency ac generator switch—START.
10. Throttle—OFF.
   Check fuel flow for zero reading to insure that the fuel has been completely shut off. A fuel flow indication during engine shutdown would be the first indication of possible fire.

NOTE

Under no-load conditions, the emergency ac generator should remain on the line until engine rpm falls below 14%.

   a. Observe the "OFF" flag on the attitude indicator.
   b. Monitor fuel quantity gage.

NOTE

Check that engine decelerates freely by listening for any excessive engine noises during shutdown.

11. Master electrical power switch—OFF.

NOTE

If the master electrical power switch is left in the ON position, the battery circuits will be closed and battery power will be depleted.

BEFORE LEAVING AIRPLANE

Before leaving the airplane, check the following:

WARNING

The ditching control handle must be operated before the safety belt is released. Attempts to leave the airplane without disconnecting and safetying the parachute may result in inadvertent firing of the parachute and damage to the parachute and associated hardware or injury to personnel.

1. Ejection seat ground safety pin — Check installed. (FP-RP)
2. Parachute firing lanyard—Check released and safetied.

NOTE

The parachute should not be left in the airplane with the firing lanyard connected to the parachute disconnect assembly unless the airplane is on alert status.

3. All electrical switches—OFF.
4. Canopy hold-open support(s) —Installed.
   Before leaving the airplane the canopy hold-open supports must be properly installed by ground personnel.

WARNING Ⓑ

The canopy hold-open supports do not prevent momentary upward movement of the canopy. If the shear pin has failed and a wind gust (such as caused by prop or jet wash) strikes the canopy, the canopy could be moved off the actuator rod and hold-open supports, and fall to the sill.

5. Throttle quadrant dust cover—Install.
6. Form 781—Complete.
   Enter total fuel remaining and any discrepancies which were noted during flight.

CAUTION

- Make appropriate entries on Form 781 covering any system defects or any limits in the Flight Manual that have been exceeded during the flight. Entries must also be made when the airplane has been exposed to unusual or excessive operations *such as hard landings,* excessive braking action during aborted takeoffs, long and fast landings, long taxi runs at high speeds, etc.
- Insure the security of the airplane (ground safety pins, including external tank ground safety pins, chocks, etc.) before leaving the area.

STRANGE FIELD PROCEDURES

If it is necessary to land at an airfield where normal ground support is not available, the pilot will be responsible for performing or closely supervising the required airplane servicing. There are several items which must be serviced immediately after engine shutdown, and additional items of servicing and inspection are required prior to takeoff. During cold weather operation, it may be necessary to store the battery in a warm place if a suitable electrical power source is not available for starting. It is strongly recommended that pilots become familiar with the servicing procedures for all items listed on the Servicing Diagram, figure 2-8. Detailed information is available in Ground Handling, Servicing, and Airframe Group Maintenance Manual, T. O. 1F-106A-2-2. Installation of the drag chute is covered under Flight Control Systems, T.O. 1F-106A-2-7. The following check list supplements the normal operating procedures and includes items that would normally be handled by the ground crew.

IMMEDIATELY AFTER ENGINE SHUTDOWN

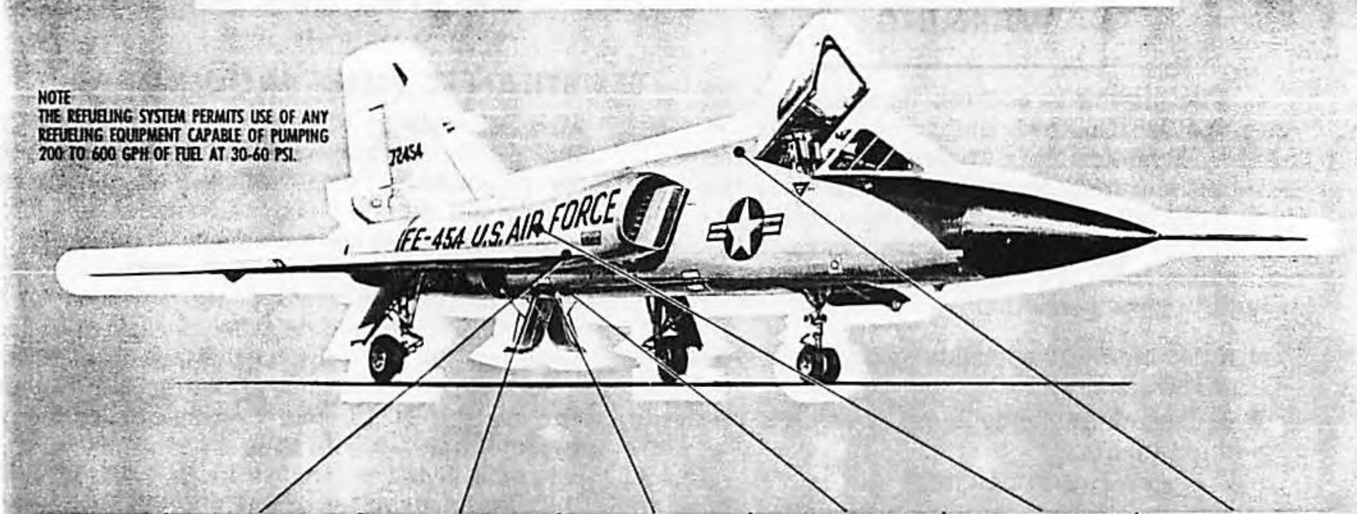
1. Engine oil level—Approximately two inches below filler neck. (Service with oil MIL-L-7808 or MIL-L-23699 according to season.)

NOTE

If this check is delayed, oil will drain from the tank into the accessory gear case and it will be impossible to obtain an accurate reading. If the level is to be checked on a cold engine, it will be necessary to operate the engine for approximately two minutes to pump excess oil out of the accessory gear case. An accurate reading can then be obtained immediately after engine shutdown. Oils MIL-L-7808 and MIL-L-123699 are compatible and can be mixed. However, these oils should never be mixed as a routine practice; mix them only in an emergency for one flight.

2. Constant-speed drive oil level — Near full mark on dipstick. (Service with MIL-L-7808. Add oil very slowly, wait two minutes and recheck.)
3. Hydraulic system reservoirs.
   a. Relieve system pressure by operating flight controls.
   b. Hydraulic accumulators — 750 (±25) psi precharge.

servicing diagram



NAME	Fuel Tanks	Hydraulic System Accumulator (Primary and Secondary Systems)	Hydraulic System Reservoir (Primary System)	Hydraulic System Reservoir (Secondary System)	Constant Speed Drive Oil Tank	Refrigeration Unit
REPLENISHING AGENT	Fuel	Dry Nitrogen or Clean Dry Air	Red Hydraulic Fluid	Red Hydraulic Fluid	Oil	Oil
SPECIFICATION	MIL-T-5624 Grade JP-4 ⚠1		MIL-H-5606 ⚠3 H-515	MIL-H-5606 ⚠3 H-515	MIL-L-7808 ⚠3 0-148	MIL-L-6085
CAPACITY	Refer to Fuel Quantity Data Table	750 PSI	1.72 U.S. Gallons	1.9 U.S. Gallons	2.37 U.S. Gallons	Maintain Level Between "ADD" and "FULL"
SERVICING LOCATION	Right Side Under Engine Inlet Duct and Upper Aft of External Tanks ⚠2	Forward Bulkhead Hydraulic Compartment	Left Side Hydraulic Compartment	Right Side Hydraulic Compartment	Fuselage Over Right Wing	Refrigeration Compartment

48028-1

Figure 2-8

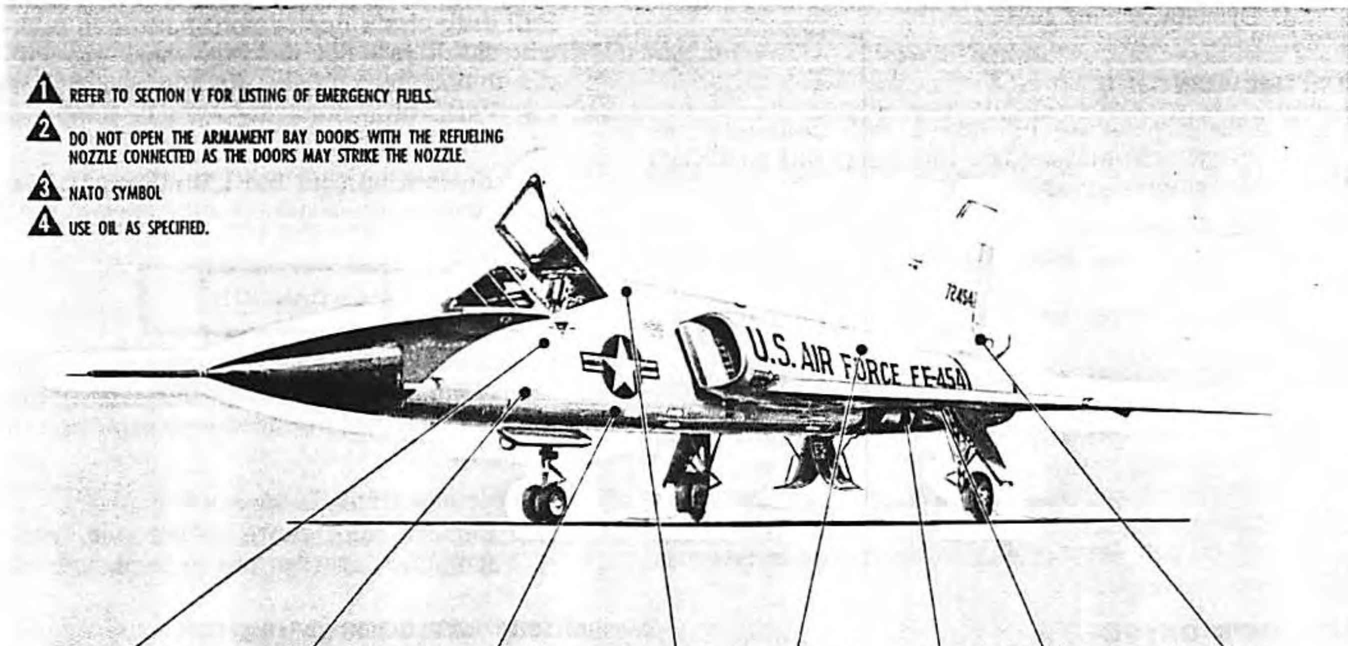
c. Fluid level—Not more than ¾ of an inch below the full mark corresponding to temperature on reservoir temperature gage.

d. Service with MIL-H-5606.

Close the reservoir pressure shutoff valve. Relieve reservoir air pressure by depressing button on filler cap. Remove cap and fill to mark corresponding to reservoir temperature gage.

e. Open the reservoir pressure shutoff valve; the reservoir pressure gage should indicate 55 psi. Check the bleed indicator, and if bleed is indicated, open the bleed fitting on the reservoir servicing panel. When clear fluid flows, tighten the bleed fitting.

4. Engine hot-section data recorded—Check and reset.
   a. Check recorder for recorded engine temperature performance data; enter in Form 781.
   b. Energize ac or dc essential bus. Check that EGT gage power failure flag is not on display.



Emergency Oxygen Bottles	Oxygen	External Power	Glycol Tank	Engine Oil Tank	Air Turbine Generator	Air Flasks (High Pressure Pneumatic System)	Drag Chute
Gaseous Oxygen	Liquid Oxygen	208 Volt 400 CPS AC 28 Volt DC, MA-1 Electrical Power	Glycol (by volume) -Distilled Water Mixture-2 Parts Glycol Fluid To 1 Part Distilled Water	Oil	Oil	Clean Dry Air	Drag Chute Pack
MIL-0-27210 Grade A Type 1	MIL-0-27210 Grade A Type II		MIL-A-8243 (Glycol)	MIL-L-7808 4 3 0-148 MIL-L-23699 4	MIL-L-7808 3 0-148		Pack in Accordance with T.O.14D1-3-112
1800 PSI	A 5 Liters B 10 Liters		2.0 U.S. Gallons	Fill to 2" Below Top of Filler Neck	Fill to Full Mark on Sight Glass	3000 PSI	
Survival Kit or Ejection Seat Assembly	Fuselage Left Side, Just Aft of Forward Electronics Compartment	Fuselage Left Side Below Cockpit	Refrigeration Compartment	Upper Fuselage, Left Side at Leading Edge of Vertical Fin	Left Engine Access Compartment	Wing Section of Left Main Wheel Well	Above Tail Cone, Enclosed by Speed Brakes

48028-2

c. Remove recorder reset button caps; reset recorder overtemperature flags and clocks as required.

d. Reinstall reset button caps; turn off electrical power.

REFUELING

1. Fuel—JP-4. (Refer to EMERGENCY FUEL, Section V, for listing of Emergency Fuels.)
2. Refueling truck pressure—30 to 60 psi.
3. Refuel until truck gage indicates that flow has stopped.

CAUTION

While fuel is flowing into the tanks, check for air flow from the vent outlet in the bottom of each wing. If air is not detected flowing from each vent, shut off fuel flow and investigate the trouble.

4. Disconnect fuel hose.

TIRE SERVICING

1. Nose tires—140 psi 41,000 pounds gross weight and below; 150 psi 41,001 to 42,720 pounds gross weight.
2. Main tires:
   225 psi (33,000 pounds gross weight and below).
   240 psi (33,001 to 35,000 pounds gross weight).
   260 psi (35,001 to 37,500 pounds gross weight).
   275 psi (37,501 to 39,750 pounds gross weight).
   285 psi (39,751 to 41,000 pounds gross weight).
   295 psi (41,001 to 42,720 pounds gross weight).

LIQUID OXYGEN

1. Service with liquid oxygen MIL-0-27210A, Grade A, Type II.

HIGH-PRESSURE PNEUMATIC SYSTEM

1. Use Besler 56150-17 quick-disconnect assembly and MC-11 or equivalent compressor. If quick-disconnect assembly is not available, disconnect tubing aft of quick-disconnect fitting in airplane and connect compressor or tubing. Some airplanes incorporate a tee fitting in the servicing line for alternate servicing. Remove cap to service.
2. Charge pneumatic system to 3000 psi.

DRAG CHUTE INSTALLATION

1. Insert the drag chute into the canister so that the riser lies flat under the deployment bag and the closing flaps of the pilot chute are positioned at 45° angle from vertical.
2. Check that keeper on riser is on top riser and under pilot chute section of deployment bag and that riser is positioned on guide bracket.
3. Place the three drag chute canister straps over the pilot chute and insert ripcord pin in cone of retaining straps. Upper retaining strap to be attached last.
4. Pull riser until positioning keeper is against the canister straps and insert D-ring in jaw clamp. (See figure 2-9.)
5. Release locking pawl to allow release arm to rest against top of jaw clamp. (See figure 2-9.)
6. Insert pin in jettison switch. Check that pin fits snugly.
7. Pull drag chute deployment handle in cockpit, check jaw for locking, check rip cord pin to determine if the pin has been pulled.
8. Push drag chute deployment handle to the "IN" position.
9. Release locking pawl to allow arm to rest against top of jaw clamp. (See figure 2-9).

CAUTION

Failure to release locking pawl may result in drag chute automatically jettisoning when deployed or speed brake damage in flight.

10. Repeat steps 2 and 3 above.
11. Remove pilot chute spring pin (pin with streamer); return pin to parachute loft.

MISSILE BAY DOOR OPERATION

To Open Missile Bay Doors

1. Check the high-pressure pneumatic system pressure gage in the left wheel well, to determine that system pressure is between 2000 and 3000 psi. Charge the pneumatic system if necessary.
2. Clear the missile-bay area and post personnel to warn others that doors are to be operated.

CAUTION B

Do not operate the missile bay doors with the aft electronic compartment door open. Damage to the electronic door may result.

3. Place the manual door-control valve handle (left main wheel well bulkhead) in the OPEN position.
4. Lock the manual door-control valve handle in the OPEN position with the red streamered ground safety pin.
5. Install door safety locks on all door actuating cylinders, if available.

WARNING

- If door safety locks are not available, extreme caution should be exercised when checking equipment in the exposed missile bay area.
- Avoid movement of any of the door control valve indicator pins, as damage to the missile bay doors will result.

To Close Missile Bay Doors

1. Check that the manual door-control valve handle is safetied in the OPEN position.
2. Remove door safety locks, if installed.
3. Clear the missile-bay area of personnel.
4. Remove and stow the manual door-control valve handle safety pin.
5. Place the manual door-control valve handle in the CLOSE position.
6. Check that pneumatic system pressure is between 2000 and 3000 psi. Charge the pneumatic system, if necessary.

RETRACTING AND LATCHING TAILHOOK

The tailhook is manually retracted and latched. The latching procedure generally requires two men and a special tailhook retracting tool (8-96515). However, the special tool may not be available, and the following alternate retraction method may be used:

1. Disconnect battery.
2. Position a man on each side of extended tailhook.
3. Raise and hold the tailhook in the retracted position.

WARNING

To prevent serious injury to personnel, do not allow any part of the body to extend into the tailhook extension area.

4. Engage the latch shaft with a 5/8-inch open end wrench, and rotate latch shaft approximately 60° aft (clockwise, looking up). Listen for a definite, audible click which occurs as the latch lever seats in the trigger. (If click is not heard, lock may not be fully engaged. Check mechanism for binding, defective solenoid, etc.)
5. Remove wrench from latch shaft, and check that shaft remains in latched position.
6. Slowly ease off pressure applied to tailhook, until it can be determined that tailhook is firmly latched, then release tailhook.
7. Install the tailhook safety pin.
8. Check that tailhook shoe is held against upper stop with safety wire.

LANDING GEAR STRUT SERVICING

NOTE

The P/N MS28889-1 high-pressure air valve is being issued in place of AN6287-1 valve on attrition basis. The MS28889-1 valve has a 3/4-inch hex swivel nut and

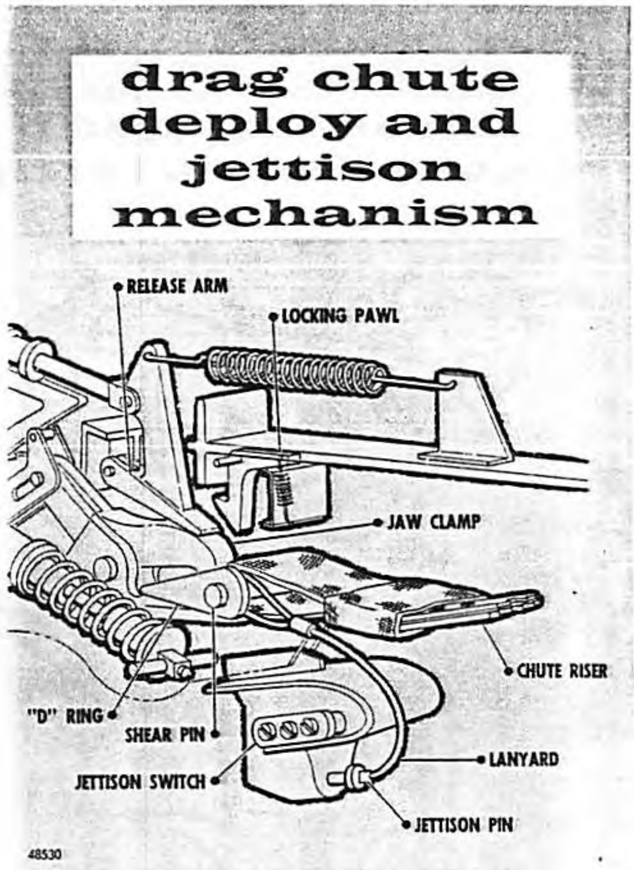


Figure 2-9

does not have a valve core as a secondary seal. Prior to inflating or deflating shock strut determine which valve is installed.

WARNING

Make sure that all personnel are clear of airplane. Determine that all obstructions, which might cause damage if airplane is to be lowered, are removed from under fuselage and wings. Do not inflate shock struts in order to install or remove jacks. Do not inflate shock struts when jacks are installed; serious injury to personnel and/or structural damage to the airplane may result.

Servicing Main Gear Shock Strut With Air

The distance between torque arm centers on the main landing gear shock strut should be 5 5/16 inches. To inflate or deflate the main gear shock struts to the 5 5/16-inch dimension use the following procedure:

a. Remove dust cap from air valve at top of strut (see figure 2-10).

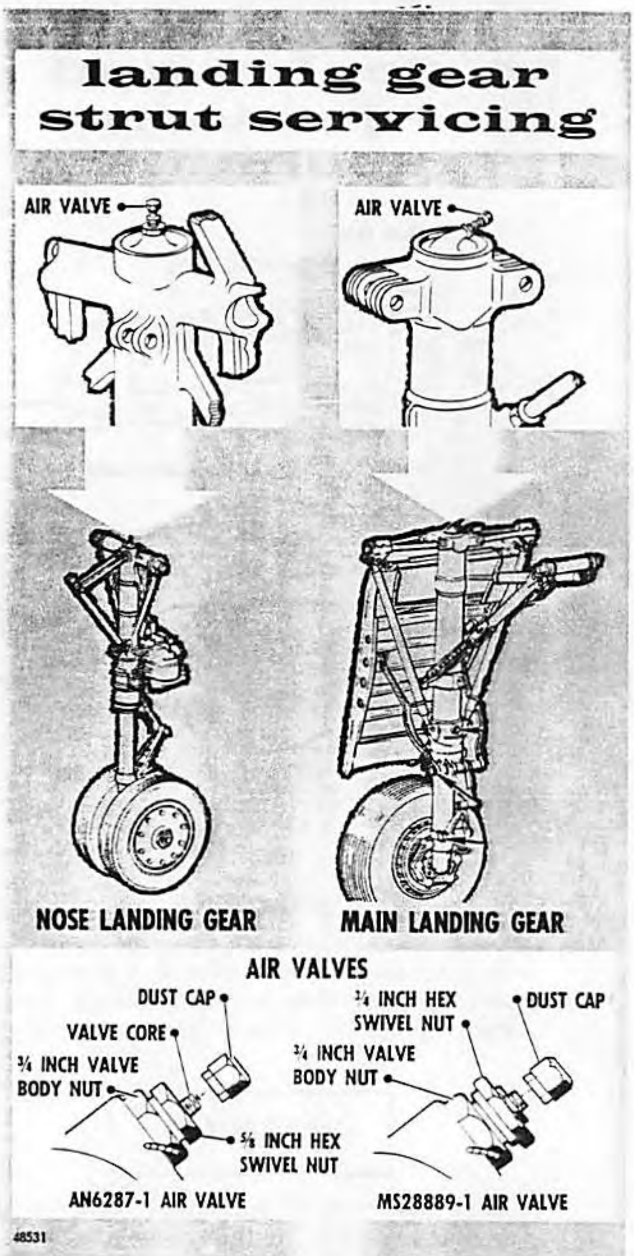


Figure 2-10

CAUTION

The MS28889-1 valve does not contain a valve core. Loosen dust cap about one turn to allow air, which may be trapped between the metal-to-metal seal of the valve and cap, to escape; then remove cap.

NOTE

Oscillate the wing, when inflating or deflating shock struts, to minimize piston friction.

b. To inflate main gear shock strut:
   (1) Attach dry air or nitrogen supply hose to valve.

CAUTION

Use a regulated source of air pressure; inflate strut slowly. The maximum servicing pressure, with the airplane gross weight resting on the strut, is 1300 psi. Do not service a fully extended strut to more than 203 psi.

   (2) Loosen ⅝-inch hex swivel nut (top nut) installed on AN6287-1 air valve or ¾-inch hex swivel nut (top nut) installed on MS28889-1 valve to a maximum of ¾ turn.

NOTE

To keep the ¾-inch safetied valve body (bottom nut) from turning, hold valve body with another wrench.

   (3) Inflate slowly to 5$\frac{5}{16}$-inch dimension.
   (4) Tighten ⅝-inch or ¾-inch swivel nut (top nut), to 50-70 inch-pounds torque.
   (5) Remove dry air or nitrogen supply hose from valve.
   (6) Replace and tighten valve cap to extreme finger tightness.

c. To deflate main gear shock strut:
   (1) On AN6287-1 valve, loosen ⅝-inch hex swivel nut (top nut) 1/10 of a turn and depress valve core. ON MS288889-1 valve, loosen ¾-inch swivel nut (top nut) slowly. The amount the ¾-inch nut is loosened will govern the rate of air discharge.

NOTE

To keep the ¾-inch safetied valve body (bottom nut) from turning, hold valve body with another wrench.

   (2) Deflate shock strut below the 5$\frac{5}{16}$-inch dimension and tighten the swivel nut.
   (3) Inflate shock strut using procedures in step b.

Servicing Nose Gear Shock Strut With Air

Due to airplane cg variations caused by different airplane weight configurations, it is necessary to check shock strut air pressure as well as extension distance. To service the nose gear shock strut proceed as follows:

a. Remove dust cap from air valve and attach dry air or nitrogen service line, with 0 to 1000 psi air pressure gage (see figure 2-10).

CAUTION

The MS288889-1 valve does not contain a valve core. Loosen dust cap about one turn to allow air, which may be trapped between the metal-to-metal seal of the valve and cap, to escape; then remove cap.

NOTE

Use a regulated source of air pressure; inflate the strut slowly. The maximum servicing pressure, with the airplane gross weight resting on the strut, is 1100 psi on Ⓐ airplanes and 850 psi on Ⓑ airplanes. Do not service a fully extended strut to more than 75 psi.

b. Loosen 5/8-inch hex swivel nut (top nut) installed on AN6287-1 air valve or 3/4-inch hex swivel nut (top nut) installed on MS288889-1 valve to a maximum of 3/4 turn. Take a pressure reading and tighten hex swivel nut.

NOTE

To keep the 3/4-inch safetied valve body (bottom nut) from turning, hold valve body with another wrench.

c. Find a pressure, on the inflation chart mounted on the shock strut, that corresponds to the gage reading and note proper strut extension for that pressure.

d. Measure the distance between torque arm centers. The distance should be within 1/4-inch of the noted chart distance. If measurement is not within tolerance, inflate or deflate the strut to the proper distance.

e. Tighten hex swivel nut to torque of 50-70 inch-pounds and remove servicing line. Install valve cap and tighten to extreme finger tightness.

WEATHER PROCEDURES

Canopy

The canopy and compartment doors should be kept closed whenever the airplane is unattended, to prevent the entry of sand and dust, even though the inside temperature may be 10 to 20 degrees lower with them open. When attended by maintenance personnel, the canopy and compartment doors may be opened for ventilation and to prevent heat warping of delicate equipment.

Fuselage

All openings such as engine air intake ducts, tailpipe, vents, exhaust outlets, pitot and artificial feel system intake heads, should be covered and protected with tight-fitting covers to keep out all sand and dust.

RADOME ANTI-ICING TANK SERVICE

The radome anti-icing system is serviced with a mixture of two parts ethylene glycol, Specification MIL-A-8243, and one part water. The anti-icing tank is located at the upper right side of the air-conditioning compartment. Use the following procedure to fill anti-ice fluid tank:

a. Depressurize tank by pressing manual bleed valve in center of filler cap. Hold cloth over cap to prevent spillage of fluid which may escape with released air.

b. Remove filler cap and fill tank. Tank capacity is two U. S. gallons.

c. Replace filler cap and tighten to extreme finger tightness.

ELECTRICAL POWER REQUIREMENTS

1. AF/ECU-10/M, AF/M32A-13, and AF/M32M-2 will provide sufficient power for starting and operation of the MA-1 system.
2. MC-1 and MD-3 with adapter cable 8-96052-801 (6115-690-4050) will provide sufficient power for starting only.

CAUTION

Insure that the MC-1 or the MD-3 have three-phase ac power only. If single phase power is applied, the airplane electrical system can be seriously damaged.

NOTE

If the above units are not available, a battery start will be necessary. If the airplane is to be parked for more than approximately one hour in freezing temperatures, and a battery start is expected, it will be necessary to store the battery in a warm place until ready to start.

emergency procedures

Section III

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INTRODUCTION

This section includes procedures to be followed to correct an emergency condition. The procedures, if followed, will insure safety of the pilots and aircraft until a safe landing is made or other appropriate action is accomplished. Multiple emergencies, adverse weather, and other peculiar conditions may require modification of these procedures. Therefore, it is essential that pilots determine the correct course of action by use of common sense and sound judgment. Procedures appearing in bold face capital letters are considered critical action. Procedures appearing in small letters are considered noncritical action. Each is defined as follows:

Critical Actions: Those actions which must be performed immediately without reference to a checklist if the emergency is not to be aggravated and injury or damage is to be avoided. These critical steps will be committed to memory.

Noncritical Actions: Those actions which contribute to an orderly sequence of events, and where there is time available to consult a checklist. To assist the pilot when an emergency occurs, three basic rules are established which apply to most emergencies occurring while airborne. They should be remembered by each pilot. The rules follow:

1. Maintain aircraft control.
2. Analyze the situation and take proper action.
3. Land as soon as practicable.

GROUND OPERATIONS

ENGINE FIRE DURING START

If the engine fire warning light illuminates or there is visible evidence of fire during starting, proceed as follows:

1. **THROTTLE—OFF.**
2. Fuel shutoff switches—Close.
3. Master electrical power switch—OFF.

EXCESSIVE EGT OR FIRE IN TAILPIPE

Excessive EGT is indicated by the exhaust gas overtemperature warning light. If excessive EGT is observed or if excessive fire occurs in the tailpipe during start or shutdown, proceed as follows:

1. Throttle—Off.
2. Compressed air — Connected.

NOTE

In the event a ground cart is not available, air for motoring the engine may be taken from the airplane pneumatic pressure supply by opening the manual shutoff valve in the main wheel well.

3. Fuel shutoff and boost pump switches—OFF.
4. Master electrical power switch—ON.
5. Engine ignition button—Depress and hold.
6. Throttle—START, then OFF.
   Move throttle to START, check for positive rpm, then move throttle to OFF. This will fire the combustion starter and aid in clearing the engine.
7. Engine ignition button—Release.
   Release ignition button when fuel in starter flask is expended.
8. Throttle—OFF.
9. Master electrical power switch—OFF.

EJECTION DURING TAXI AND GROUND ROLL

The egress system provides safe escape capability under most conditions on the ground with the canopy closed, i.e., static, taxi, and ground roll prior to takeoff and after landing. In addition to normal attitude, successful static tests have been accomplished with a simulated aircraft attitude of a collapsed nose gear and a combination of both the nose gear and the right main gear collapsed. Refer to INFLIGHT for ejection procedures.

WARNING

- The canopy must be closed to ensure safe removal during the automatic ejection sequence.
- Before a ground level ejection is attempted, consideration should be given to the possibility of manually disconnecting the parachute, personal equipment, and lap belt and "going over the side."
- Do not attempt to beat the system by manually pulling the D-ring prior to automatic chute deployment by the drogue gun. At airspeeds below 150 knots, the automatic system must be depended upon for successful ground level ejection.

EMERGENCY GROUND EGRESS

The following procedure prescribes the fastest method of egress from a disabled airplane on the ground. It also provides additional pilot protection from fire. After the airplane is stopped, proceed as follows:

1. OXYGEN SUPPLY LEVER – OFF.
2. SURVIVAL KIT EMERGENCY RELEASE HANDLE-PULL

WARNING

Extreme caution must be exercised in the use of the emergency release handle because of its similarity and close proximity to the right ejection handgrip.

3. **DITCHING CONTROL HANDLE—PULL.**

WARNING

Failure to actuate the ditching control handle may result in firing of the parachute deployment gun and resultant entanglement of the parachute with the airplane.

4. **SAFETY BELT AND SHOULDER HARNESS—RELEASE.**
5. **CANOPY — JETTISON** (if necessary).
6. Depart airplane.

WARNING

If the nose gear is extended, there is approximately an eight foot drop to the ground. Exercise caution to prevent an incapacitating injury.

Should canopy opening or removal methods fail and ground egress is necessary, proceed as follows:

a. Ejection seat ground safety pin—Install.

b. Release safety belt and shoulder harness and remove parachute.

c. Remove the canopy breaker tool from its brackets and firmly grasp the handle with both hands.

d. Chop a hole in the upper forward area of the canopy plexiglass (see figure 3-1).

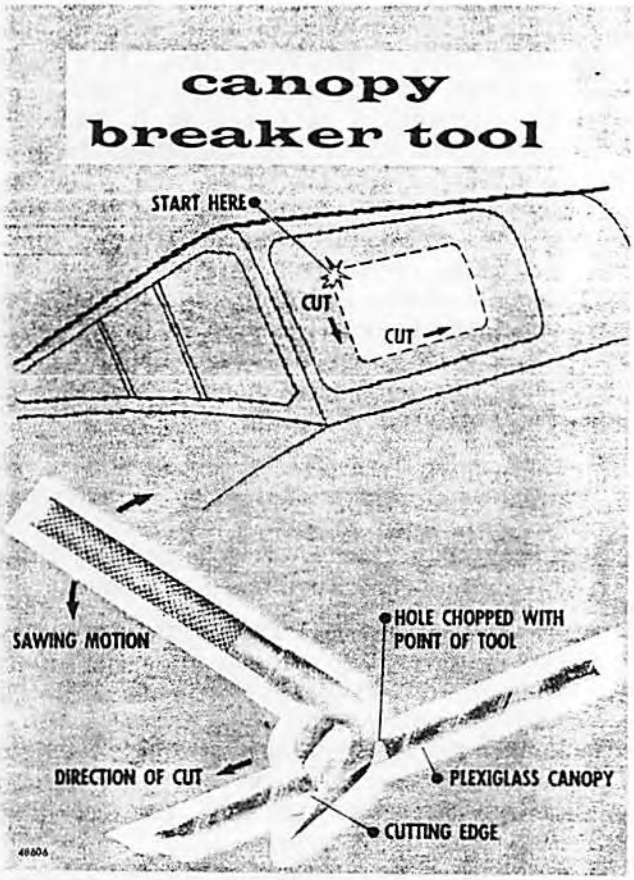


Figure 3-1

e. Insert the point of the canopy breaker tool in the hole and begin a sawing motion using short, rapid strokes (see figure 3-1). Cut a hole approximately 12 inches high and 20 inches wide.

CAUTION

Attempting to break off large pieces of the canopy instead of sawing as directed will increase the time required for escape and will quickly fatigue the pilot.

f. Rub the jagged edge of the opening with the serrated portion of the canopy breaker tool handle to remove as much of the sharp edge as possible.

g. Egress through the hole.

EMERGENCY ENTRANCE

The procedure to be used by rescue personnel assisting a disabled pilot from the airplane following a crash landing is contained in figure 3-2.

TAKEOFF

ABORT

An aborted takeoff should always be made with the landing gear extended and in accordance with those procedures necessary to stop the airplane.

1. **THROTTLE—IDLE.**

NOTE

- With the throttle in the OFF position, nose wheel steering will not be available and normal drag chute deployment may not be possible.
- If the airplane leaves the runway, stopcock the throttle.

2. **DRAG CHUTE HANDLE — PULL (EMERGENCY DEPLOY).**
3. **EXTERNAL TANKS—JETTISON** (if necessary).
   External tanks containing fuel should be jettisoned as soon as the emergency is known to allow the tanks to clear the vicinity of the airplane. Empty tanks should be retained.
4. **TAILHOOK DOWN BUTTON—DEPRESS** (if necessary).
   If a barrier engagement is anticipated, extend the tailhook at least 2000 feet from the barrier if possible.
5. Maintain aircraft control.
   a. Nose wheel steering—As necessary.
   b. Rudder and elevon—As necessary.
   c. Differential braking—As necessary.

CAUTION

Avoid use of wheel brakes above 115 KCAS since braking action is difficult to feel and a blown tire could cause loss of directional control.

5A. Deleted.

6. Idle thrust control switch—ON.
7. Brakes—As necessary.
   Lower the nose wheel to the runway and brake as required. Hold maximum possible backstick (up elevons) without raising the nose wheel from the runway, to obtain additional braking.

CAUTION

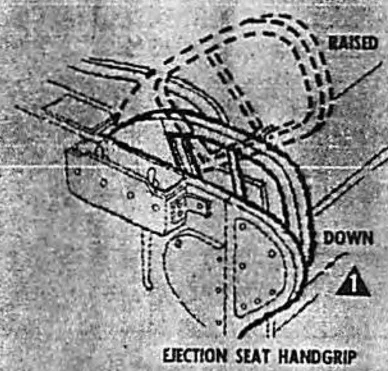
Excessive braking prior to barrier engagement should be avoided to reduce the possibility of a tire blowout which could cause loss of directional control and possibly prevent barrier engagement.

8. Shoulder harness inertia reel handle—MANUAL LOCK. (FP-RP).
9. Canopy—Retain.

WARNING

- This airplane contains a one-motion ejection seat system. Do not raise either ejection seat handgrip to jettison the canopy as such motion will also eject the seat.
- If canopy is to be jettisoned, make sure it is jettisoned before the airplane comes to a complete stop. Otherwise, sparks from the canopy remover may cause a fire if fuel is spilled in the vicinity of the airplane.
- If the canopy is jettisoned below 175 KCAS (airplanes without clear top canopy) 208 KCAS (airplanes with clear top canopy) while airborne, it may strike the vertical fin. If the canopy is to be jettisoned at any time, it must be in the closed position.
- If the canopy is not jettisoned, do not unlock the canopy until the airplane stops. On rough terrain the aft canopy mounts may become damaged, making canopy raising difficult.
- Because of the height of the cockpit above the ground (approximately eight feet), care should be exerted when abandoning the airplane without a ladder.
- Unless circumstances warrant shutting down the engine, do not stopcock the engine, but maintain idle rpm until fire equipment arrives. Stopcocking the engine allows fuel to vent near the wheel brakes, thus creating a fire hazard.
- After an aborted takeoff, wheel brakes will reach high temperatures and tire explosion may result. Park airplane in an isolated area to minimize danger to ground crews and other airplanes. Refer to USE OF WHEEL BRAKES, Section VII.

emergency entrance



(COCKPIT ENTRANCE)

BEFORE ATTEMPTING TO ENTER COCKPIT CHECK POSITION OF EJECTION SEAT HANDGRIPS FOR DOWN AND STOWED POSITION

WARNING

IF THE HANDGRIPS ARE RAISED, DO NOT PULL THE CANOPY EXTERNAL JETTISON HANDLE OR THE PILOT AND SEAT WILL BE EJECTED.

IF EJECTION SEAT HANDGRIPS ARE DOWN.

1. REMOVE CANOPY EXTERNAL JETTISON HANDLE ACCESS DOOR.
2. GRASP CANOPY EXTERNAL JETTISON HANDLE AND PULL OUTBOARD APPROXIMATELY 6 FEET TO FIRE CANOPY.

NOTE

THE CANOPY SHOULD TRAVEL UP AND AFT SUFFICIENTLY TO CLEAR THE COCKPIT. HOWEVER, IT WILL PROBABLY STRIKE THE AIRPLANE MIDSECTION.

WARNING

SPARKS FROM THE CANOPY JETTISON CHARGE MAY IGNITE FUEL SPILLED IN OR NEAR THE AIRPLANE.

3. IF CANOPY FAILS TO JETTISON BY BALLISTIC CHARGE, OR IF PRESENCE OF FUEL FUMES MAKES JETTISON INADVISABLE, USE PROCEDURES AS GIVEN BELOW.

IF EJECTION SEAT HANDGRIPS ARE RAISED.

1. OPEN EXTERNAL CANOPY CONTROL ACCESS DOOR LOCATED JUST BELOW THE LEFT WINDSHIELD.

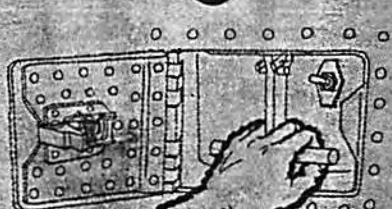
A 2. GRASP CANOPY EXTERNAL LATCH HANDLE AND PULL OUTBOARD APPROXIMATELY 6 INCHES TO RELEASE CANOPY LATCHES.

B PULL LATCH RELEASE HANDLE FULL OUT; ROTATE UP AND AFT AS FAR AS POSSIBLE (ABOUT 100 DEGREES); PULL OUT AGAIN (ABOUT ½ INCH); THEN ROTATE HANDLE BACK DOWN TO ORIGINAL POSITION TO UNLOCK CANOPY.

A 3. RAISE THE CANOPY MANUALLY.

B HOLD THE SPRING-LOADED TOGGLE SWITCH IN THE UP POSITION UNTIL THE CANOPY IS OPEN AS FAR AS DESIRED. IF CANOPY DOES NOT OPEN WHEN THE SWITCH IS HELD IN THE UP POSITION, PULL THE "CANOPY MOTOR BRAKE RELEASE HANDLE" DOWN THROUGH AN ARC OF 60 DEGREES, THEN MANUALLY OPEN THE CANOPY.

A



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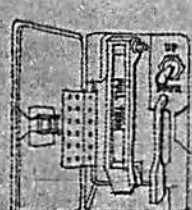


CAUTION A

DO NOT ATTEMPT TO OPEN THE CANOPY WITH THE EXTERNAL CANOPY SWITCH, AS POSSIBLE DAMAGE TO THE CANOPY ACTUATOR OR LOSS OF ELECTRICAL POWER DURING THE OPENING CYCLE WOULD RESULT IN THE CANOPY JAMMING PARTLY OPEN.

WARNING

IF THE HANDGRIPS ARE RAISED, EXERCISE CAUTION TO PREVENT INJURY IN THE EVENT THE CANOPY JETTISON CHARGE FIRES WHILE THE CANOPY IS BEING REMOVED.

IF UNABLE TO OPEN CANOPY

1. SHATTER PLEXIGLASS ALL THE WAY AROUND THE CANOPY FRAME, AS NEAR THE FRAME AS POSSIBLE.
2. CUT DEFOG PANEL FROM EDGE OF CANOPY FRAME USING AXE OR SHARP HOOK KNIFE.

48053-1

Figure 3-2

RUNWAY OVERRUN BARRIER ENGAGEMENT

The tailhook system is installed for use with the BAK-6 "water squeezer" overrun barrier, the BAK-9 or BAK-12 "rotary friction unit" overrun barrier, or the MA-1A modified "chain" overrun barriers which are modified to include an additional "across-runway" cable for tailhook arrestment. Each of these installations has an "across-runway" cable that is supported approximately three inches above the runway. A chain (webbing) barrier may be used as a back-up installation for the BAK-6, BAK-9 or BAK-12 barrier. When this combination is installed, the "across-runway" cable is placed some distance in front of the landing gear webbing engagement cable. Any of these installations will provide at least 90% probability of the tailhook engaging the supported cable. Combinations of installations mentioned above will

(PILOT REMOVAL)

1. REACH INTO THE COCKPIT AND DISARM BOTH SEATS. CUT CABLE WITH HEAVY DUTY CUTTERS AND MANUALLY TRIP DISCONNECT FITTINGS (F-106B, 4 FITTINGS—2 LH, 2 RH; F-106A, 3 FITTINGS—2 LH, 1 RH).

2. IF HANDGRIPS ARE RAISED, RETURN HANDGRIPS TO THE DOWN, AND STOWED POSITION (IN DETENT). PUSH THE HANDGRIP UPLOCK RELEASE LEVER, LOCATED ON THE SEAT BEHIND THE RIGHT HANDGRIP, FORWARD AND THEN PUSH THE HANDGRIPS DOWN UNTIL THE DETENT CLIPS ENGAGE.
3. INSERT THE SAFETY PIN (LOCATED ON THE RIGHT-HAND CONSOLE) INTO THE RIGHT SIDE OF EACH SEAT, JUST BELOW THE RIGHT HANDGRIP.

NOTE

IF THE CANOPY JETTISON HANDLE HAS BEEN USED, THIS HANDLE MUST BE RETURNED TO THE STOWED POSITION IN ORDER TO INSERT THE SEAT SAFETY PIN.

WARNING

ALWAYS OPERATE THE DITCHING CONTROL HANDLE BEFORE OPENING THE SAFETY BELT.

4. DISCONNECT THE PILOT'S PERSONAL EQUIPMENT LEADS.
5. RAISE DITCHING CONTROL HANDLE.
6. RELEASE THE SEAT BELT AND SHOULDER HARNESS.
7. SHUT OFF THE OXYGEN SUPPLY AT THE OXYGEN CONTROL PANEL (LOCATED ON THE LEFT-HAND CONSOLE).

48053-2

provide an engagement probability of from 93% to 99.6%. There is no minimum speed for tailhook engagement. If the barrier installation includes both the landing gear barrier webbing cable and the tailhook "across-runway" cable, the tailhook cable will provide the most reliable operation. Normally, the webbing should be lowered for engagement. On barrier installations with webbing, if the webbing is up, there is the possibility of a one-strut engagement after a successful tailhook engagement. Therefore, whenever a remote barrier control exists in the control tower, request that the barrier webbing be lowered. If during flight, a barrier engagement is anticipated, external wing tanks should be jettisoned prior to lowering the gear for landing to preclude damage to the landing gear doors. In the event of an aborted takeoff or an emergency on the landing roll, jettison external tanks as soon as the need for barrier engagement is known. Excessive braking prior to engagement should be avoided to reduce the possibility of a tire blowout, which could cause loss of directional control and possibly prevent barrier engagement. Deploy the tailhook at least 2000 feet from the barrier if possible and steer the airplane toward the center of the barrier. During arrestment, airplane behavior is satisfactory, and nose wheel steering or brakes may be used for directional control. Be prepared to correct for yaw after engagement which might cause the airplane to swerve off the edge of the overrun.

WARNING

- Do not hold the brake on a blown tire prior to barrier engagement. This may cause damage to the underside of the wing, but it will insure that the barrier will not be severed by the flat tire rim, thus increasing the success of barrier engagement.
- To reduce the possibility of fire if landing with a blown tire or tires, do not stop-cock the throttle until signalled to do so.

Barrier Engagement Procedures

1. Call for barrier webbing to be lowered.
2. Tailhook down button — Depress (at least 2000 feet from barrier.)
3. Steer airplane toward center of barrier and follow ABORT procedures.

ENGINE FAILURE DURING TAKEOFF

If engine failure occurs during takeoff but after becoming airborne, an immediate touchdown may be practical if sufficient runway remains. If there is not sufficient runway, an immediate relight may be possible with the emergency fuel control and all-points ignition. The course of action depends on remaining runway, altitude at time of failure, automatic escape equipment capabilities (refer to EJECTION), and terrain features of the overrun landing area.

If decision is made to stop:

1. **FOLLOW ABORT PROCEDURE.**

If takeoff is continued:

1. **EXTERNAL TANKS—JETTISON.**
2. **ZOOM (IF POSSIBLE) AND EJECT.**

ENGINE FIRE DURING TAKEOFF

If decision is made to stop:

1. **FOLLOW ABORT PROCEDURE.**

If takeoff is continued:

1. **THROTTLE—MAXIMUM THRUST TO SAFE EJECTION ALTITUDE.**
2. **EXTERNAL TANKS—JETTISON** (if necessary).
3. **IF ON FIRE—EJECT.**

   Check for positive indication of fire such as trailing smoke or flame or request a visual check from another aircraft or from the ground.

If fire cannot be confirmed:

4. Throttle—Minimum practical thrust.
5. Land as soon as possible.

EJECTION DURING TAKEOFF

The zero-zero egress system provides safe escape capability under all conditions at and immediately after takeoff within an envelope defined by a nose high airplane and up to 40° of bank. At 150 feet altitude with a nose high airplane, safe ejection can be accomplished at up to 65° of bank. At 500 feet altitude, safe ejection can be accomplished at up to 85° of bank. Refer to INFLIGHT for ejection procedures.

FLAT TIRE DURING TAKEOFF

If a tire is blown on takeoff and airspeed is below nose wheel lift-off speed, discontinue the takeoff (refer to FLAT TIRE ON LANDING). If the tire is blown at or above nose wheel lift-off speed (120-135 KCAS), proceed as follows: Continue the takeoff and leave the landing gear extended.

WARNING

Do not raise the gear as further damage to the airplane may be incurred.

External tanks (if installed) should not be jettisoned on takeoff, but will be jettisoned while airborne prior to landing if they contain fuel. Plan to land at minimum gross weight unless the damage incurred on takeoff necessitates an immediate landing. Directional control is more difficult, and braking efficiency is greatly reduced at high gross weights. A normal approach and landing should be accomplished with touchdown on the side of the runway corresponding to the good tire, at the recommended airspeed. After touchdown proceed as directed in FLAT TIRE ON LANDING.

AFTERBURNER FAILURE DURING TAKEOFF

Afterburner failure is noted by a sudden loss of thrust. If decision is made to stop:

1. **FOLLOW ABORT PROCEDURE.**

If takeoff is continued:

1. **THROTTLE—INBOARD TO FULL MIL POWER.**

WARNING

If the takeoff is to be continued with an afterburner blowout, the throttle must be moved out of the AFTERBURNER position immediately to enable the afterburner nozzle to close. Nonafterburner engine operation with the nozzle in the open position will result in an appreciable loss of thrust.

AFTERBURNER EXHAUST NOZZLE FAILURE DURING TAKEOFF

If the afterburner exhaust nozzle fails to open at the time of afterburner ignition, a rapid rise in exhaust gas temperature, a reduction of RPM, and engine compressor stall will occur.

If an EPR drop or loss of thrust does not occur when afterburner is selected:

1. Throttle—Inboard, and abort takeoff.

LANDING GEAR WARNING LIGHT REMAINS ILLUMINATED ON LANDING GEAR RETRACTION OR ILLUMINATES ABNORMALLY DURING FLIGHT

Leave the landing gear control handle in the up position, remain below or slow to 280 KCAS, and request a visual inspection. If the main landing gear doors are closed and main landing gear is extended, or if unable to obtain a visual inspection, use the procedure for LANDING GEAR EMERGENCY EXTENSION. This insures positive "door open" and "gear down and locked" positions.

CAUTION

An attempt to make a normal landing gear extension or to recycle the landing gear with a "gear down—doors up" configuration will result in damage to the landing gear doors by retracting the landing gear into the closed doors.

If the visual inspection reveals that the landing gear has apparently retracted normally and the gear is not down, recycle the landing gear handle. If unable to obtain a safe gear up indication, extend the landing gear and land as soon as practical.

CANOPY RETENTION VERSUS JETTISONING

The canopy should be retained (not jettisoned) during landing emergencies (except ditching), aborted takeoff, and barrier engagements. Under such emergency conditions, probability of survival is enhanced with canopy retention. The canopy is protection against being saturated with flaming fuel and is temporary protection against the direct effects of fire; also, retention of the canopy may prevent barrier cables and wires from entering the cockpit. Following an emergency in which the canopy is retained, normal opening should first be attempted with jettisoning used as an alternate method. The application of these considerations does not preclude the pilot from exercising sound judgment in jettisoning the canopy when he deems it necessary.

INFLIGHT

BEFORE EJECTION (IF TIME AND CONDITIONS PERMIT)

Ⓑ1. Inform other pilot of ejection necessity by use of intercom.
2. Aim airplane toward uninhabited area.
3. Give location to nearest radio facility and turn IFF on emergency, Mode 3, Code 77.

Canopy ejection activates emergency IFF if the selector switch is in NORM or LOW position.

4. Stow all loose equipment. (FP-RP)
5. Actuate bailout oxygen bottle (if applicable) (FP-RP).
6. Cabin air selector switch—RAM.
7. Helmet visor—Down. (FP-RP)
8. Airplane—Trim for level flight.

WARNING

If the airplane is not trimmed for level flight it will pitch down when the control stick is released for ejection. This greatly decreases chances for a successful low-altitude ejection.

9. Throttle—OFF; reduce speed.
10. Sit erect, place elbows tightly against body while grasping the ejection seat handgrips. Position head back firmly against headrest with chin tucked in.

EJECTION

NOTE

Every emergency in which ejection is considered will have its particular set of circumstances involving such factors as airplane speed, attitude and control, altitude, and sink rates. The information presented below indicates capabilities under various combinations of these circumstances and some desirable techniques and procedures to improve the probability of a successful ejection. It must be emphasized, however, that the decision to eject must be based on the particular circumstances at hand and not just on the specification 0-0 to 450 KCAS limits of the system.

The escape system is designed for ejection in a zero altitude—zero speed condition to 450 KCAS with certain limitations under various combinations of critical attitudes and sink rates. Pitch attitude has the most influence on the recovery capability. Ejection while the nose of the airplane is above the horizon, results in a more nearly vertical trajectory of the seat, thus providing more altitude and time for seat separation and parachute deployment. With an airspeed of 180 KCAS and wings level, the escape system will provide safe ejection under an additional 1500 feet/minute sink rate if the nose is rotated from -10° pitch to +10° pitch. Therefore, when at all possible, maintain sufficient airspeed to employ the "zoom-up" maneuver and eject with the nose above the horizon (limit +50° pitch) while there is still a positive rate of climb. Any bank angle conditions other than zero or a wings level airplane will degrade the escape system capability. Computer studies indicate that with a nose high airplane, the wings level ejection capability is degraded by 20% with a 20° bank angle and 60% with a 40° bank angle. The ejection seat should be used to abandon the airplane in flight. The airplane should be slowed down as much as possible if at high airspeeds, as forces on the body and injury hazard are decreased when airspeed is reduced. In addition, the ballistically deployed parachute lends itself to better performance at lower airspeeds. The maximum speed for an inflight ejection is 450 KCAS. Slow the airplane to below 450 KCAS prior to ejection if possible. The "zooming" maneuver is encouraged at all points in the flight envelope. The zoom will exchange airspeed for altitude which will result in the desired condition of a nose high/minimum speed ejection. Inflight capabilities are as follows:

General

Conditions	*Minimum Safe Ejection Altitude*
Sink Rate: 0 to 10,000 feet per minute Pitch Attitude: 0 to −12° Bank: 0 to 85° Airspeed: 0 to 450 KCAS	**2000** feet above terrain

Specific Examples:

a. The absolute minimum ejection altitude using the maximum capability of the system for an airplane with a relatively slow forward velocity of 180 KCAS, -10° nose down, wings level, and with a 5000 ft/min sink rate is 200 feet above ground level.

b. With at least a +10° pitch attitude, 180 KCAS velocity and a wings level airplane, ejection can be accomplished at any altitude with a sink rate of 3500 ft/min or less.

WARNING

- When the airplane is uncontrollable and cannot be leveled, ejection should not be delayed as this will reduce the probability of success.
- When in the landing configuration below 2000 ft AGL, do not delay ejection by attempting to airstart a failed engine. Eject immediately unless landing is assured. There is not enough time available for the engine to accelerate from windmilling RPM to military power prior to ground impact.
- Under spin or dive conditions or any uncontrollable maneuver, eject at least **10,000** feet above the terrain whenever possible.

The egress system has been extensively tested and has been demonstrated to be a reliable system. The automatic system featuring the automatic opening safety belt, seat-man separator, and ballistically deployed parachute are sequenced in operation to provide the desired results more reliably than manual operation. Therefore, to assure survival from both low and high altitude ejections, the automatic features of the equipment must be used and depended upon. Refer to figure 3-3 for ejection parameters during flameout landing.

Ⓑ1. **ORDER CREWMEMBER TO EJECT.**

Ⓑa. Bailout warning switch—ON (if necessary).

2. **EJECTION SEAT HANDGRIPS—RAISE.**

WARNING

- On Ⓑ airplanes, unless circumstances dictate otherwise, the rear seat occupant should initiate his own ejection. If the front pilot initiates ejection, the aft seat will be ejected followed, one second later by the front seat. The aft seat handgrips and arm guards will not be raised and the front seat will eject after a one-second delay, regardless of aft seat position.
- To prevent injury to arms during ejection, keep arms tightly against body when raising handgrips.
- Exert a positive pull on the handgrips to insure they reach the full up and locked position.
- Continue to hold the handgrips throughout the ejection to prevent arms from flailing.

If canopy fails to jettison:

1. Ejection seat handgrips—Return to normal down position.

NOTE

Push the handgrip uplock release lever to return the handgrips to the down (in detent) position.

flameout landing ejection capabilities

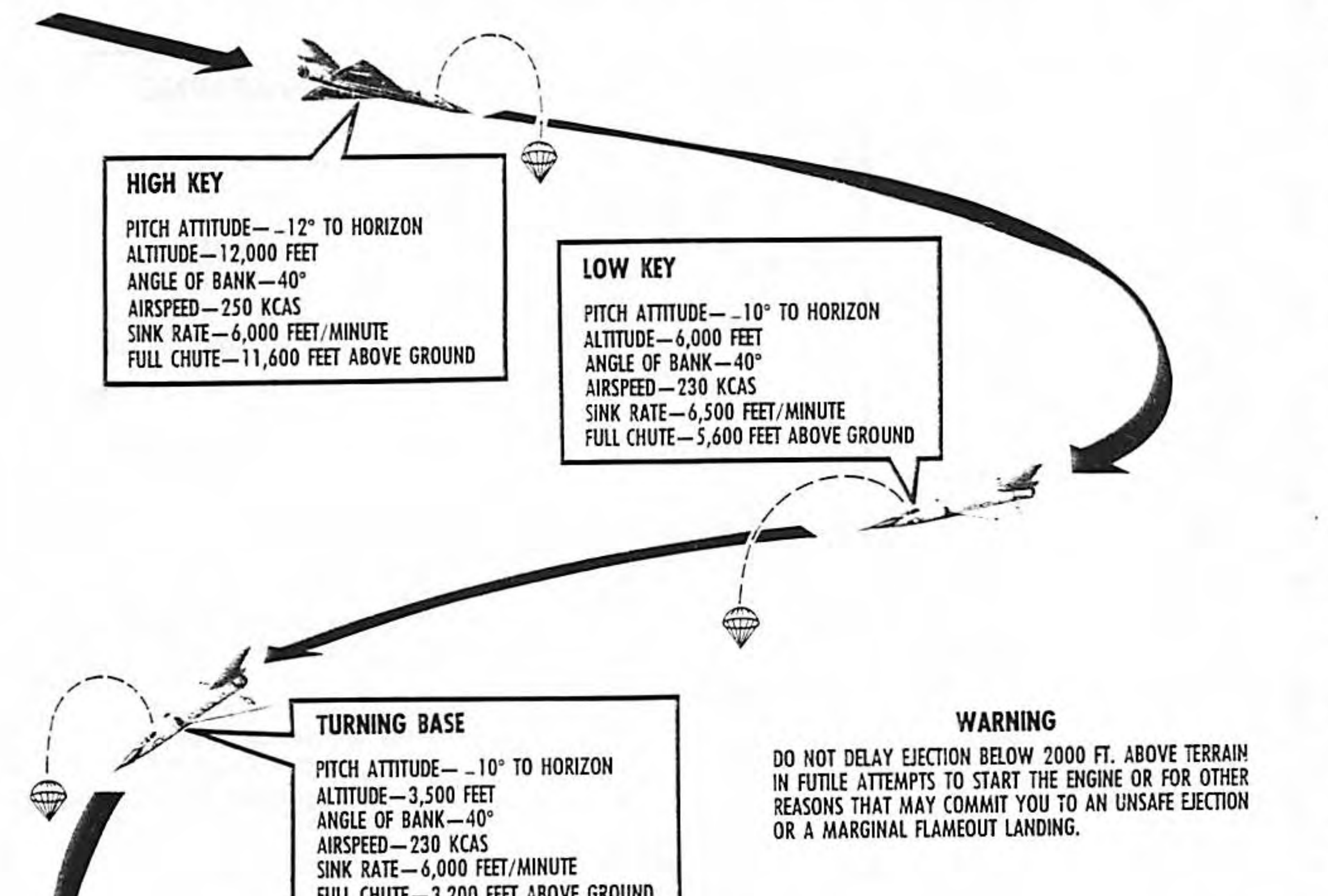


Figure 3-3

NOTE

On Ⓑ airplanes, the canopy cannot be manually pushed open from inside the cockpit.

3. Canopy switch—Open (if necessary).
4. Ejection seat handgrips—Raise.

AFTER EJECTION

NOTE

- The zero-zero egress system is a reliable, completely automatic system from the time the pilot pulls the ejection handgrips until there is a full canopy. A failure in the automatic system after ejection can be identified by the following:
  a. Failure of the safety belt to open.
  b. Failure of the seat-man separator to reel in the strap pushing the pilot away from the seat.
- The lap belt should open and the seat-man separator should be reeled in within two seconds after ejection. If these events do not occur, manual activation of the system shall be accomplished as follows:
  a. Raise ditching control handle.
  b. Open safety belt and push away from the seat.
  c. Pull rip cord.

WARNING

- Always operate the ditching control handle before opening the safety belt.
- Do not pull rip cord until at or below 14,000 feet.
- If automatic actuator malfunctions do not release the survival kit until after parachute deployment to prevent the kit or lanyard from fouling the parachute. However, the survival kit should be released as soon as the parachute is stabilized to prevent possible injury when contacting the ground with the survival kit attached.
- If automatic actuator malfunctions do not raise emergency release handle until after descent to an altitude not requiring oxygen. The oxygen supply will be cut off when the survival kit is released.

OVER-THE-SIDE BAILOUT

In an attempted ejection, when all possible means to eject the seat have failed, an "over-the-side" bailout can be attempted as an alternative to a forced landing or ditching. An "over-the-side" bailout should be accomplished in the following manner:

WARNING

Bailout should be accomplished above 2000 feet terrain altitude.

NOTE

To insure successful bailout, the decision to bail out should be made at 10,000 feet or above, and not unnecessarily delayed.

a. Establish a glide speed (if possible) between 200 and 250 KCAS.

b. Actuate bailout oxygen bottle (if installed).

NOTE

An additional source of oxygen is available if the parachute contains an oxygen bailout bottle.

c. Raise survival kit emergency release handle. Raising the handle releases the personal leads bundle and the parachute attaching wedges, thereby completely separating the pilot from the survival kit.

NOTE

The oxygen supply in the survival kit will be cut off when the survival kit is released.

d. Raise the ditching control handle, open the safety belt, release the control stick, and push away from the seat and airplane.

e. Trim rudder full right or full left (the resulting yaw will aid in clearing the vertical fin). Trim nose full down while holding airplane attitude with control stick back pressure. If altitude permits, the airplane should be rolled to the inverted position. Roll should be toward the direction that the rudder is trimmed, with positive g forces maintained during the roll.

f. After bailout and descent to 14,000 feet, pull rip cord.

NOTE

Pulling the rip cord does not fire the deployment gun. In this case the parachute is deployed by the pilot chute.

ENGINE MECHANICAL FAILURE DURING FLIGHT

NOTE

A rapid drop of rpm and fuel flow to zero accompanied by simultaneous loss of generators and hydraulic pressure indicates failure of the engine accessory drive system. This failure is not accompanied by vibration, rough engine, or unusual noises normally associated with other types of structural failure. Engines failing from accessory system loss cannot be restarted. If this type of failure occurs, attention should be directed to ejection or successful flameout landing as repeated airstart attempts will be futile.

WARNING

Air starts should not be attempted when engine damage is apparent; i. e., overtemperatures, increasing fuel flow, no throttle response. rapid engine deceleration, and scraping or minor explosion noises. Major explosion or airplane disintegration may result.

1. Throttle — OFF.
2. Airspeed—250 KCAS (gear up, speed brakes closed).
   Establish a glide speed of 250 KCAS with gear up and speed brakes closed.
3. RAT — Extend (if necessary).
4. Fuel shutoff and boost pump switches — Off.
   Check that all fuel shutoff valves are closed and all boost pump switches are off.

WARNING

- If hydraulic pressure for flight control operation becomes marginal at low engine rpm, extend the ram air turbine.
- With a frozen engine, the ram air turbine is the only source of hydraulic pressure.
- Pilot's discretion should determine the action to be taken following an engine failure. Refer to EJECTION VS FLAMEOUT LANDING.

NOTE

- During glide with engine windmilling above approximately 41% rpm, all normal electrical power should be available provided the "AC POWER FAIL" warning light is not illuminated.
- If engine rpm has dropped below 41%, boost pumps will not be operative.
- With engine windmilling below approximately 41% rpm, only the ac and dc essential buses will be energized. With the emergency ac generator supplying power to the ac essential bus, expect intermittent operation of the UHF command radio, and other equipment powered by this bus, while making excessive demands on the secondary hydraulic system; i.e., rapid flight control movement, lowering the gear by normal means, etc.
- During glide with a "frozen engine," ac power will not be available. The battery will be the only source of power; therefore, only the dc essential bus and the battery bus will be energized.

FROZEN ENGINE

Any failure which results in a frozen engine should normally be a progressive type failure and may be noticed by indications such as no or low engine oil pressure, or engine vibrations. Should engine vibration become moderate to heavy, complete engine failure is only seconds away and the engine should be shut down to preclude destructive failure that would jeopardize a successful ejection or forced landing.

WARNING

- When engine failure results in a frozen engine it is necessary to extend the ram air turbine to obtain hydraulic pressure for flight control operation.
- If airspeed is above RAT maximum extension speed and the engine is frozen, do not extend the speed brakes to slow the airplane. Speed brake extension will cause immediate depletion of remaining secondary hydraulic system pressure. Deceleration should be accomplished by moving the flight controls a minimum amount to establish a flight attitude which provides deceleration.

With a frozen engine, the battery is the only source of electrical power and the turn-and-slip indicator is the only electrically powered flight instrument which will operate. If a forced landing is to be made, the emergency landing gear and drag chute systems must be used.

ENGINE FAILURE DURING FLIGHT AT LOW ALTITUDE

In the event of engine failure during flight at low altitude, and with sufficient airspeed available, the airplane should be pulled up (zoom-up) to exchange airspeed for an increase in altitude. This will allow more time for accomplishing subsequent emergency procedures (air start, ejection, establishing forced landing pattern, etc.).

WARNING

If the flameout is due to temporary interruption of fuel flow from the fuel tanks, restart may take up to four minutes and may preclude the possibility of a low-altitude air start. This condition can be detected by the absence of fuel flow indication when the throttle is open.

NOTE

The point at which climb should be terminated will depend on whether the pilot intends to eject or whether he intends to continue attempting air starts, establish forced landing pattern, etc. In any event, it is recommended that air start be attempted immediately upon detection of engine flameout and repeated as many times as possible during the zoom-up. If the decision is to eject, the airplane should be allowed to climb as far as possible. Ejection should be accomplished while the nose of the airplane is above the horizon but prior to reaching a stall or sink.

In the zoom-up maneuver, more altitude can be gained if external tanks are jettisoned. Maximum altitude gain can be achieved by jettisoning external tanks prior to zoom-up. However, when jettisoning tanks consideration must be given to factors such as sufficient airspeed to allow time for pilot reaction, and the proximity to populated areas where tanks will fall. In any event, the decision to jettison or retain external loads must be made on the basis of an evaluation of the above factors and conditions existing at the time of the emergency.

FLAMEOUT

If there is no indication of mechanical failure within the engine, an air start should be attempted as soon after flameout as possible, altitude permitting. Refer to figure 3-4 for maximum glide time and distance.

WARNING

If a flameout occurs when the airplane is in the landing configuration at less than 2000 feet above ground level, EJECT if landing is not assured. Do not attempt an airstart as there will not be sufficient altitude available to allow time for the engine to accelerate to military thrust prior to ground impact.

NOTE B

Air starts can be made from either the forward or the aft cockpit.

AIR START

1. **THROTTLE — INBOARD FROM AFTERBURNER.**
2. **FUEL CONTROL SWITCH — EMERGENCY.**

   Check that the emergency fuel control warning light illuminates.
3. **IGNITION BUTTON DEPRESSED AND THROTTLE —AS REQUIRED.**

   Attempt to match throttle position with engine rpm and simultaneously depress ignition button. The throttle must be retarded to below military power when attempting an airstart above 30,000 feet to preclude engine overtemp.
4. RPM — 60-80%, then release ignition button.

NOTE

- An increase in rpm and fuel flow are the first indication of a successful air start, since the exhaust gas temperature system has a relatively low response rate.
- Retard the throttle to OFF if during an air start lightup does not occur within 20 seconds when there is a positive fuel flow indication, or if the engine fails to accelerate to idle rpm within approximately 45 seconds after lightup.

Immediate relight not obtained:

1. Throttle—OFF.

WARNING

If the flameout is due to temporary interruption of fuel flow from the fuel tanks, restart may take up to four minutes. This condition can be detected by the absence of fuel flow indication when the throttle is open.

NOTE

Loss of main ac generator will cause loss of boost pump operation.

2. Fuel control switch—EMERGENCY.
3. All fuel switches—Check.

   Check that all fuel shutoff switches are OPEN and boost pump switches are ON.
4. Airspeed—250 KCAS.
5. Ignition button—Depress and hold.
6. Throttle—START, OFF, then IDLE.

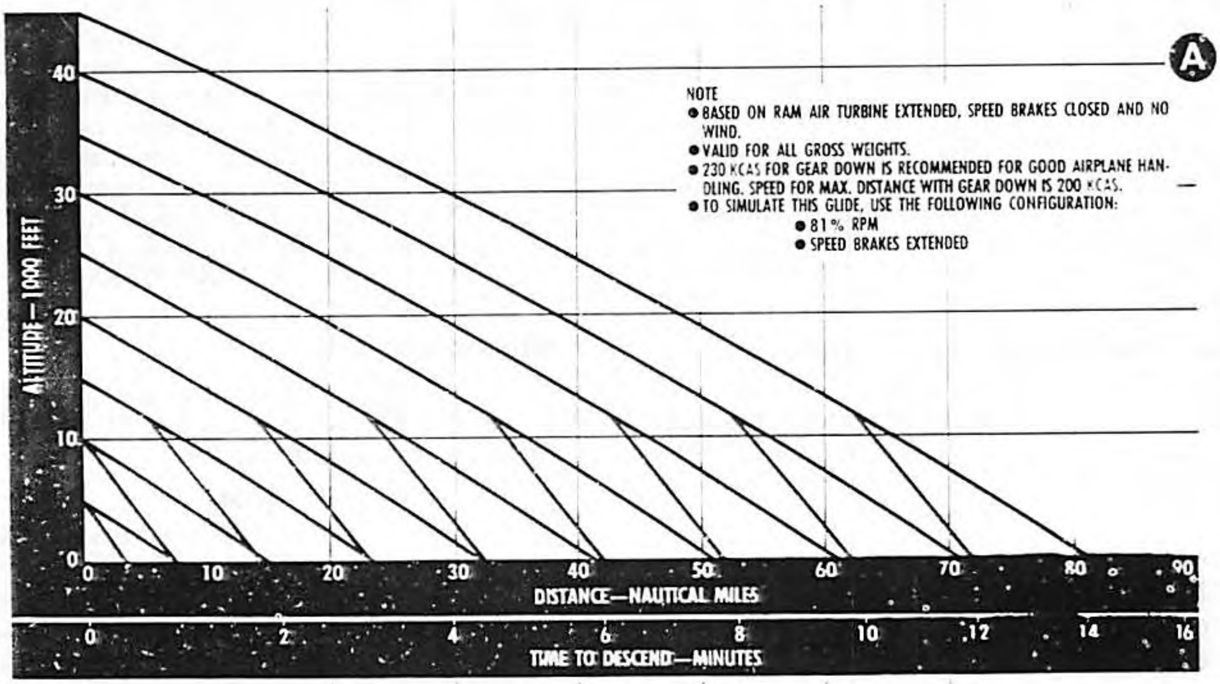
   Use ground start procedure to insure ignition in the event ignition circuit is malfunctioning.
7. RPM—60-80%, then release ignition button.

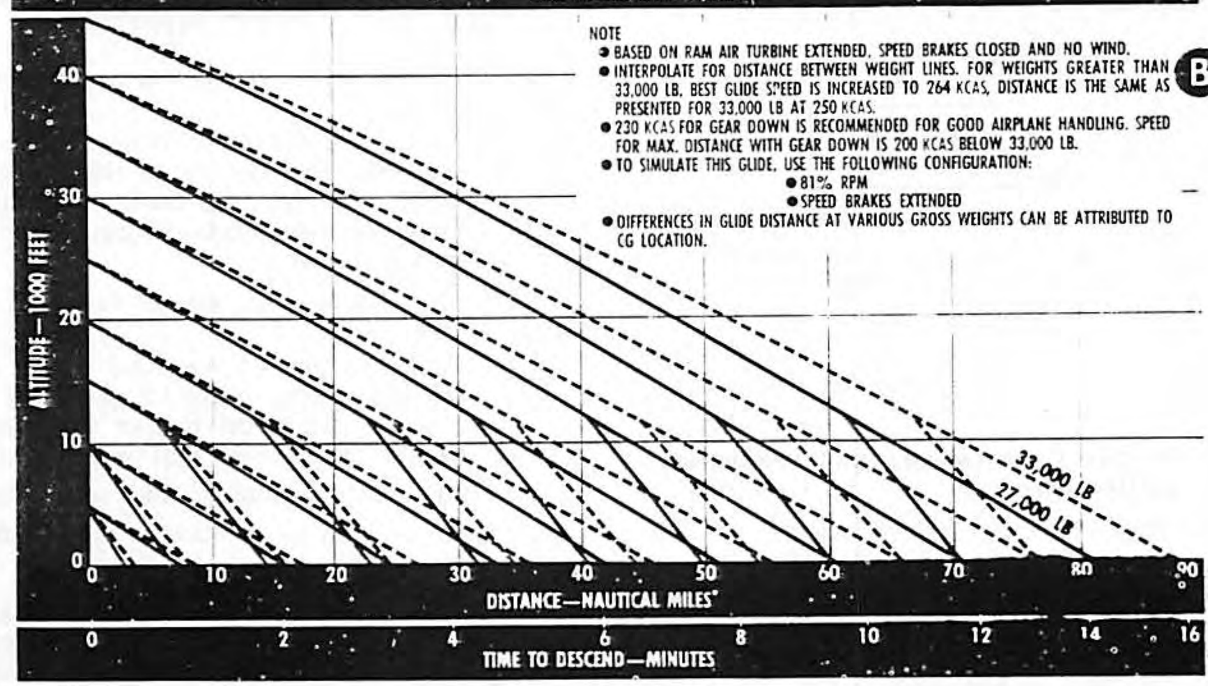
maximum glide time and distance

DATE: 1 SEPTEMBER 1961
DATA BASIS: FLIGHT TEST

GEAR UP • BEST GLIDE SPEED 250 KCAS NOT TO EXCEED .9 MACH
GEAR DOWN (BELOW 12,000 FEET) • GLIDE SPEED 230 KCAS • STANDARD DAY

WINDMILLING OR FROZEN ENGINE





48050A

Figure 3-4

WARNING

- If hydraulic pressure for flight control operation becomes marginal at low engine rpm, extend the ram air turbine.
- When the engine fails and rpm has dropped below 41% the boost pumps will be inoperative. If the fuel level is below 2800 pounds, maintain a glide speed of 250 KCAS to assure uninterrupted fuel flow for a restart. Avoid unusual or uncoordinated maneuvers, and do not lower the landing gear or open the speed brakes.

NOTE

If the engine cannot be restarted on the normal or emergency fuel system, consideration should be given to ejection. Refer to EJECTION VS FLAMEOUT LANDING.

ENGINE FIRE—STEADY OR FLASHING WARNING LIGHT

1. **THROTTLE—MINIMUM PRACTICABLE THRUST.** Reduce thrust to the minimum necessary to maintain safe ejection altitude and check for fire.

If fire is confirmed:

2. **THROTTLE—OFF.**
3. **FUEL SHUTOFF SWITCHES—CLOSE.**
4. Deleted.
5. **IF FIRE CONTINUES—EJECT.**

WARNING

The fire warning system is deactivated by turning off the master electrical power switch thus causing the fire warning light to go out.

If fire cannot be confirmed, continue flight at minimum safe thrust and land as soon as practicable.

EXHAUST GAS OVERTEMPERATURE WARNING LIGHT ILLUMINATED

1. Thrust—Reduce (when practical).
   The EGT overtemperature warning light is set to illuminate at 630°C. A reduction in thrust should extinguish the light.
2. If the light extinguishes:
   a. Continue the mission and monitor the exhaust gas temperature and other engine instruments.
3. If the light stays on:
   a. Monitor the exhaust gas temperature and other engine instruments. If the exhaust gas temperature remains above 630°C and/or the other engine instruments indicate a malfunction, land as soon as practicable.

NOTE

Illumination of the exhaust gas overtemperature warning light in flight could be caused by several factors. Among these are engine fuel control failure, afterburner exhaust nozzle failure, compressor stall, or improper throttle management under certain flight conditions. As dictated by other engine instrument indications or the flight condition at the time of overtemperature, refer to ENGINE FUEL CONTROL FAILURE, this section, AFTERBURNER FAILURE, this section, or COMPRESSION STALL, Section VII.

ELECTRICAL FIRE

There is no system to warn of electrical fire in this airplane. Fuses protect most of the circuits and tend to prevent electrical fire.

NOTE

In the event of a TSD fire, turn the TSD light intensity rheostat off.

If an electrical fire occurs, however, and its source cannot be readily determined visually, attempt to isolate and eliminate the fire as follows:

1. MA-1 power switch—EMER.
2. If fire continues, ac and dc generator switches —OFF.
   Turn the generator switches OFF to eliminate electrical power to the nonessential buses. The essential buses will automatically become energized through the emergency system.
3. If fire still persists, master electrical power switch—OFF.
   With the master electrical power switch OFF, the drag chute (through the emergency system), the bailout warning light Ⓑ, and the tailhook will be available.
4. If fire continues and becomes severe—Eject.
5. Land as soon as practicable if fire subsides.

NOTE

The gear can be lowered by the emergency system when the master electrical power switch is OFF; however, to check for a gear-down indication, the master electrical power switch must be turned ON momentarily.

SMOKE OR FUMES

Accomplish those procedures necessary to eliminate smoke or fumes.

1. Cabin air selector switch — RAM (below 25,000).
   Cabin air selector switch — OFF (above 25,000).
2. Start immediate descent to 25,000 or below (if practicable).
3. MA-1 power switch—EMER.
4. Cabin temperature control knob—MAN COLD.
5. When smoke or fumes are eliminated, cabin air selector switch — PRESS.
6. MA-1 power switch—ON (If not cause of smoke or fumes).

FOG

1. Cabin air selector switch—RAM.
2. Cabin air selector switch — PRESS, when cockpit clears.
3. Cabin temperature control knob — HOT.

NOTE

The cabin temperature control knob is bypassed and cannot be used to select cockpit temperature if the cockpit no-fog and ventilated suit switch is ON.

CONTROL LOSS

(Stall, Post-Stall Gyration, and Spin Recovery).

At the first sign of a control loss, neutralize the ailerons and unload the airplane to approximately zero "G." If the airplane does not recover:

1. **CONTROL STICK—CENTER FORWARD OF NEUTRAL.**

   Hold the above control stick position until recovery is evident or a spin developes.

   WARNING

   - Do not attempt to oppose roll oscillations that occur during a post-stall gyration. A small aileron deflection may force the airplane into a spin in a direction opposite to control stick movement.
   - Do not apply full forward stick. Full forward stick may cause the airplane to roll inverted into a negative stall or an inverted spin. Either of these conditions can be recognizes by a sustained negative load factor. If at any time during recovery a negative G (less than zero G) condition occurs, neutralize the controls. If required, roll upright using rudder only. Any aileron input at this point may force the airplane back into a control loss condition.

   NOTE

   Rudder inputs under negative G conditions will result in a roll opposite to the direction of rudder application.

2. **EMERGENCY DIRECT MANUAL BUTTON - DEPRESS.**

If the above does not produce immediate signs of recovery and control loss continues:

3. **THROTTLE—IDLE.**

   Retarding the throttle to IDLE will reduce the possibility of engine overtemp and compressor stall.

If a spin developes:

WARNING

Oscillations in roll and yaw associated with a post-stall gyration can create the sensation of being in a spin. A spin is characterized by a constant rotation in one direction about the yaw axis. Definitely establish the direction of spin rotation by reference to the turn needle. When high yaw rates and lateral oscillations exist, it may not be possible to determine the direction of rotation visually by any means other than the turn needle.

4. **HOLD FULL AILERON IN THE DIRECTION OF SPIN ROTATION MAINTAINING FORWARD STICK UNTIL THE SPIN IS BROKEN OR EJECTION ALTITUDE IS REACHED.**

   WARNING

   After applying recovery controls, four or more turns may be required to effect recovery. If spin rotation is not stopped by 10,000 feet above the terrain - EJECT.

   NOTE

   Termination of the spin is characterized by an abrupt nose down pitch change to the near vertical attitude and airspeed increasing above 140 KCAS.

When spin is broken:

5. NEUTRALIZE AILERONS—MAINTAIN DIVE ATTITUDE UNTIL AIRSPEED REACHES 140 KCAS.

NOTE

- A rolling motion may accompany the recovery and may appear to be a continuation of spin rotation. As airspeed builds, the rolling motion will cease after ailerons are neutralized.
- After recovery from a spin during which excessive exhaust gas temperatures may have been encountered, the airplane should be recovered using minimum thrust to maintain flight. If the airplane has been fully stalled or spun, a write-up is required on the AFTO Form 781 so the engine can be inspected for possible damage.

VARIABLE RAMP FAILURE

During normal operation, the variable ramps should automatically retract when the airplane is slowed below Mach 1.2. If the ramps do not automatically retract when speed is reduced below Mach 1.2, as indicated by illumination of the variable ramp warning light, use the following procedure:

1. Speed — Decrease to less than Mach 1.1 (or 500 KCAS), and IDLE rpm below 25,000 feet.
2. Variable ramp switch — EMER OPEN (retracted).
3. Check other systems for failure:
   a. AC and dc power — Check.
   b. Secondary hydraulic pressure — Check.
   c. Air data computer — Check "OFF" flag in Mach indicator, or airspeed-Mach indicator (denotes possible failure).
4. Fuel flow—Monitor for range considerations.
5. Land as soon as practicable maintaining a minimum of 200 KCAS until starting flare.

WARNING

Should the variable ramps malfunction and remain in the fully extended position, a thrust loss will occur. The maximum altitude which can be maintained with full military thrust will be approximately 20,000 feet.

NOTE

- The emergency retraction system is designed only to retract the variable ramps. The variable ramp switch must be left in EMER OPEN (retracted) position and use of the ramps should not be attempted during the remainder of the flight. After landing, make emergency ramp retraction entry on Form 781.
- With variable ramps in the fully retracted position a thrust degradation will exist at velocities in excess of Mach 1.4 due to subcritical inlet operation. The characteristic sound of subcritical duct pressure fluctuations will increase in intensity with flight velocity. The engine compressor stall margin is generally sufficient to permit operation as attainable in level flight.

AFTERBURNER FAILURE

If the afterburner exhaust nozzle fails to open at the time of afterburner ignition, a rapid rise in exhaust gas temperature and a reduction of rpm will be noted. If the afterburner exhaust nozzle does not open as soon as the afterburner starts to operate, discontinue afterburning immediately to prevent engine overtemperature. If the afterburner exhaust nozzles fail to close, a drop in exhaust gas temperature and a rise in rpm will be noted. The throttle should be adjusted to maintain exhaust gas temperature. Nonafterburner operation with the exhaust nozzles open will result in approximately a 30% loss of thrust and a 30% increase in fuel consumption. If afterburner fails to ignite when throttle is moved outboard:

1. Throttle — Inboard.
   Move the throttle inboard to shut off fuel to the afterburner and close exhaust nozzles.
2. Wait five seconds to clear afterburner.
3. Throttle — AFTERBURNER (if desired).
   Place throttle outboard to re-ignite the afterburner if desired.
4. If afterburner fails to light within five seconds, throttle — Inboard.

NOTE

If a relight is attempted above 45,000 feet the throttle should be retarded approximately one fourth in the afterburner range prior to second relight attempt.

AFTERBURNER CUTOFF FAILURE

In the event of an electrical system failure or if the afterburner cannot be cut off through use of normal throttle motion, retard the throttle to a point aft of the aft afterburner stop to terminate afterburner operation. Nonafterburning thrust can then be used as desired. When the emergency cutoff procedure has been used to terminate afterburning, a relight cannot be obtained unless electrical power has been restored. If the failure is of a temporary nature an afterburner relight may be made again by advancing the throttle above the minimum afterburner actuating position and moving the throttle outboard.

ENGINE OIL SYSTEM FAILURE

Failure of the engine oil system to supply sufficient operating pressure is indicated by the oil pressure gage and illumination of the oil pressure-low warning light (master warning system). If an oil system malfunction has caused prolonged oil starvation of engine bearings, the result will be a progressive bearing failure, engine roughness, oil seal failure and loss of oil, and possible subsequent engine seizure. This progression of bearing failure starts slowly and will normally continue at a slow rate up to a certain point, at which time the progression of failure accelerates rapidly to complete bearing failure. The time interval from the moment of oil starvation to complete failure depends on such factors as bearing condition prior to oil starvation, operating temperatures of bearings, and bearing loads. A good possibility exists for several minutes of operation after experiencing a complete loss of lubricating oil. Bearing failure due to oil starvation is generally characterized by a rapidly increasing vibration. When the vibration becomes moderate to heavy, complete failure is only seconds away and in most instances the chances of a successful ejection or no-thrust landing will be increased by shutting down the engine. The end result of oil starvation is engine seizure.

OIL QUANTITY LOW OR OIL LOW WARNING LIGHTS ILLUMINATED OR OIL QUANTITY BELOW ONE HALF FULL.

The Oil Quantity Low Warning Light on Ⓐ aircraft or Oil Low Warning Light on Ⓑ aircraft will illuminate when usable oil in the oil tank decreases to approximately one half full. Oil loss may be due to high oil comsumption, oil line failure, engine beraing or seal failure, or other oil system malfunctions.

If the "Oil Quantity Low" or "Oil Low" warning lights illuminate during flight or oil quantity decreases below one half full, follow the procedure entitled "Oil - Pressure - Low Warning Light Illuminated," this section.

OIL-PRESSURE-LOW WARNING LIGHT ILLUMINATED

If the oil pressure-low warning light illuminates, or loss of oil pressure is indicated by the oil pressure gage, attempt to forestall engine seizure as long as possible as follows:

1. Thrust—Minimum required for flight.

   High thrust setting should be avoided if at all possible in order to keep bearing loads at a minimum. Upon detection of an oil system malfunction, a minimum thrust setting should be established depending on airplane configuration, gross weight, and altitude. This setting should be sufficient to maintain level flight and allow for safe approach maneuvers (subsequent thrust variations should be avoided if possible).
2. Airspeed—Reduce below 345 KCAS.

   Airspeed should be reduced below the maximum ram air turbine speed in the event the engine freezes and it is necessary to extend the ram air turbine.
3. External tanks—Jettison (if necessary).
4. G forces — Minimize.

   Avoid all abrupt maneuvers causing high g forces.
5. If engine vibrations become excessive, shut down engine.
6. Land as soon as possible; use flameout landing pattern if practicable and minimum thrust.

   If engine seizure does not occur, shut down the engine immediately after taxiing off the active runway.

WARNING

- If a high thrust setting is required to climb to a higher altitude, minimum afterburner will impose less bearing loads than a full military thrust setting.
- If engine oil system fails and engine seizes, battery power and hydraulic pressure from the RAT will be the only available power sources.
- Complete engine failure will normally be indicated by a steadily increasing vibration. At this indication, the engine should be shut down to preclude such a destructive failure as to jeopardize a successful ejection or forced landing.

IMPROPER FUEL FEEDING

If any individual system is not feeding properly, as detected from the fuel quantity gage, the total (TOT) reading on the gage will give total fuel aboard, but not total fuel that can be used.

NOTE

Monitor the No. 3 tank fuel quantity to determine usable fuel remaining.

WARNING

- If a fuel quantity-low warning light illuminates but the respective individual system fuel quantity gage reading is considerably greater than 570 pounds, assume the light is accurate and that improper feeding has occurred in that system. Rapid depletion of fuel in the respective No. 3 tank will result.
- If the LH, RH, or No. 3 quantity gage readings indicate less than 570 pounds prior to illumination of the fuel quantity-low warning light, assume that a malfunction has occurred in the low-level fuel warning system.

In any event, upon detection of improper fuel feeding, land as soon as practicable.

ENGINE FUEL CONTROL FAILURE

Engine fuel control failure may be recognized by a drop or rise from normal in exhaust gas temperature, rpm, and fuel flow. If fuel control system fails during takeoff, the takeoff should be aborted if conditions permit (runway length, overrun condition, barrier availability, etc.). If, however, the takeoff has progressed to the point that an abort is not feasible, or if fuel control fails at any time, use the following procedure:

1. Throttle—As required.
   Attempt to match throttle position to engine rpm.
2. Fuel control switch—EMER.
3. Control engine speed as necessary.
4. Land as soon as possible.

WARNING

- Avoid rapid throttle movements.
- Following an inflight normal fuel control failure, do not return to NORMAL for the duration of the flight as flameout will result.

AFTERBURNER OR ENGINE STAGE FUEL PUMP FAILURE

When operating in afterburner, failure of one of the afterburner stage fuel pumps or the engine stage fuel pump is indicated by a .3 to .5 pressure ratio drop. Afterburner operation will continue with a thrust loss of about 23% at sea level on a standard day.

FUEL BOOST PUMP FAILURE

In the event complete boost pump failure is experienced during takeoff, but fuel tank pressurization can be maintained, maximum engine and afterburner thrust can be sustained at any altitude from sea level to 36,000 feet with any fuel temperature up to 100°F in the tanks at the time of takeoff. With complete fuel boost pump failure the airplane must be operated in a nose-high attitude. Nose-down conditions and uncoordinated or unusual maneuvers should be avoided, to prevent uncovering the open fuel line in the aft end of No. 3 tanks.

NOTE

- Unless the ATG is operating, a complete boost pump failure will occur with loss of ac nonessential power.
- The fuel flow equalizer will automatically go into a bypass condition in the event of boost pump failure.

In the event of complete boost pump failure, the following procedures should be followed:

1. ATG switch—Check AUTO.
2. Operate in a nose-high attitude.
   Normal cruise attitude, recommended glide speed attitude (250 KCAS, gear up and speed brakes closed), and traffic pattern attitudes, will be within safe limits.
3. Avoid negative g maneuvers.
4. Avoid uncoordinated maneuvers.
5. Avoid rapid decelerations (combinations of large thrust changes, speed brakes and landing gear extensions).
6. Fuel quantities—Monitor.
7. When asymmetrical fuel feeding occurs, close the fuel shutoff valve on the low side when fuel quantity on that side reaches approximately 200 pounds.
8. Land as soon as practicable.

FUEL BOOST PRESSURE-LOW WARNING LIGHT ILLUMINATED: "FUEL BOOST PRESS L OR R"

Since the boost pump control system is completely independent of the fuel valve control and indicating system, illumination of a fuel boost pressure-low warning light confines the problem to the boost pump system only. In the event of warning light illumination during flight the following procedures should be followed:

1. Fuel quantities—Monitor.
   Check for fuel feeding from the left and right sides.

2. If fuel flow is symmetrical, assume an erroneous warning indication.
3. If fuel flow is asymmetrical, this confirms boost pump failure on one side.
4. Land as soon as possible using fuel available on the side with operating fuel boost pumps.
5. If sufficient fuel is not available on the side with operating boost pumps for recovery, all fuel boost pumps should be turned OFF and the procedure for FUEL BOOST PUMP FAILURE should be followed.

NOTE

To avoid uncovering the aft fuel intakes in No. 3 tanks, fly the airplane in a nose-high attitude. Avoid combinations of large thrust changes, speed brakes and landing gear extensions. The recommended glide speed of 250 KCAS (gear up and speed brakes closed) and normal traffic pattern attitudes provide safe conditions.

FUEL IMBALANCE

Minor fuel imbalance can result from improper operation of the fuel flow equalizer and can be expected during flight. Indications of fuel imbalance are wing heaviness, one external tank failing to feed, or fuel quantity gage readings. Should the imbalance progress to a 500 pound differential, then corrective action should be taken to insure that a more serious malfunction has not occurred. A failure of the fuel flow equalizer, fuel shutoff valve, or a fuel line restriction may stop the flow from one wing and efforts to balance fuel may not be successful. Early detection of this problem is imperative to preclude the aircraft proceeding to a point where recovery is not possible with the fuel remaining. A failure of this nature can first be confirmed on the fuel quantity gage at 7400 pounds fuel remaining when a 500 pound imbalance will develop. Should this occur plan on only the remaining F tank and low side wing fuel being available and land as soon as practicable.

When a fuel imbalance reaches 500 pounds differential, attempt to balance fuel by the following procedure:

1. Fuel quantity — Check.

   In the event the fuel quantity-low warning light illuminates on one side only, or abnormal wing heaviness is noted, LH, RH, or No. 3 fuel quantity should be verified by the fuel quantity gage.

2. Boost pump switches — OFF (on the side with the lowest fuel).

   If it is found that the fuel level is low on one side while there is a relatively large quantity of fuel on the opposite side, move the boost pump switches on the low side to OFF, permitting high side to reduce the unbalanced condition.

3. Boost pump switches — ON (on the side with the highest fuel).

   The boost pump switches on the side having the larger fuel quantity should be ON to force fuel usage from this side.

4. Fuel quantity – Monitor LH, RH, or No. 3 tank selections to insure that the high side fuel decreases and the low side quantity remains constant. If the low side continues to decrease with the boost pumps inoperative, suspect failure of one wing to supply fuel. Depend on low side wing and fuselage tank fuel only. Land as soon as practicable.

5. When fuel quantities are balanced or when the fuel quantity-low warning light on the high side illuminates turn all boost pump switches—ON.

WARNING

- Unless a fuel shutoff valve is closed, all boost pump switches should be turned ON prior to landing regardless of the fuel quantity or asymmetrical fuel loading, to insure positive fuel supply in the event of a go-around.
- The respective fuel boost pump switches must be placed in the OFF position prior to closing the fuel shutoff valve. If the fuel boost pumps are not turned OFF, failure of the equalizer absolute pressure switch may result in engine fuel starvation.

WARNING

If operation with a known empty tank is anticipated, turn the boost pump switch on the empty side OFF, then close the shutoff valve on the empty side to minimize pump operation without fuel cooling and to preclude possible engine flameout. When the fuel boost pumps are used on the side with the fuel quantity-low warning light illuminated, an empty No. 3 fuel tank is indicated when the fuel boost pressure-low warning light is illuminated.

WING TANK PRESSURIZATION FAILURE

If wing tank pressurization fails, normal fuel transfer cannot be expected. Some fuel may transfer into No. 3 tank, but the quantity which transfers will depend on flight attitude, altitude, etc. Therefore, from the time of pressurization failure, only the fuel in No. 3 tank can be used. Wing tank pressurization failure is indicated by illumination of the appropriate (L or R) fuel tank pressure-low warning light. Should either warning light illuminate, proceed as follows:

1. Land as soon as practicable.
2. On the side of the illuminated warning light, depend on No. 3 wing tank fuel only.
3. Monitor fuel quantity-low warning lights and fuel quantity in each wing.
4. Boost pump switches—OFF, on the pressure failed side when the fuel quantity low warning light illuminates.

FUSELAGE TANK PRESSURIZATION FAILURE Ⓐ

Failure of components in the fuel transfer system other than the F tank pressure regulator could prevent fuel transfer from the F tank. For this reason, frequent fuel quantity checks should be made to insure proper feeding. At approximately 8000 pounds of total fuel remaining, if initial F tank fuel sequence (about 400 pounds) has not transferred, plan on usable fuel in tanks No. 1, No. 2, and No. 3 only. On airplanes with the emergency F tank pressure switch or emergency F tank boost pump switch, an attempt should be made to transfer the fuel from the F tank to the wing tanks using the procedure below.

NOTE

Transfer of F tank fuel is predicated on proper feeding and depressurization of the T tanks. Therefore, some analysis may be obtained by shutting off the T tanks, then checking for F tank feeding, prior to attempting F tank emergency transfer.

1. Airplanes with F tank emergency pressure switch:
   a. F tank shutoff switch—Check OPEN.
   b. If F tank does not feed, T tank shutoff switch—CLOSE.
   c. F tank emergency pressure switch — ON.

      Leave switch ON as long as fuel is transferring. Return to the OFF position when F tank is empty, or if fuel fails to transfer.

NOTE

On airplanes with the emergency F tank boost pump, initial F tank sequence will not occur when using the emergency boost pump. The emergency boost pump will only transfer fuel from the F tank when the fuel level in each No. 3 tank reaches approximately 1200 pounds.

At approximately 3500 pounds of total fuel remaining, if the remaining fuel in the F tank (approximately 1000 pounds) fails to feed into the No. 3 tanks, use the following procedure:

2. Airplanes with F tank emergency boost pump switch:
   a. F tank shutoff switch—Check OPEN.
   b. If F tank does not feed, T tank shutoff switch—CLOSE.
   c. F tank shutoff switch—CLOSE.
   d. F tank emergency boost pump switch—ON.

      Leave switch ON as long as fuel is transferring. Return to the OFF position when F tank is empty, or if fuel fails to transfer.

NOTE

Fuel transfer is slower than normal when using the emergency system.

3. Apply positive and negative g's to the airplane in an attempt to free any vent valves that may be stuck in an open or partially opened position.
4. Deleted.
5. If, after accomplishing the above, the F tank still fails to transfer, only the fuel remaining in the No. 3 tanks will be available and the flight should be planned accordingly.

FUSELAGE TANK PRESSURIZATION FAILURE B

Failure of components in the fuel transfer system other than the F tank pressure regulator could prevent fuel transfer from the fuselage tank. For this reason, frequent fuel quantity checks should be made to insure proper feeding. When total fuel quantity remaining decreases to 8000 pounds, place the fuel quantity gage to the FWD position and determine if fuselage fuel is transferring. If the F tank pressure-low warning light illuminates, or if the F tank is not feeding when total fuel quantity reaches 8000 pounds, proceed as follows:

1. Airplanes with F tank emergency pressure switch:
   a. F tank shutoff switch — OPEN.
   b. F tank emergency pressure switch — ON.

      Leave the switch ON as long as fuel is transferring. Return it to the OFF position when F tank is empty, or if fuel fails to transfer.
2. Airplanes with F tank emergency boost pump switch:
   a. F tank shutoff switch — CLOSE.
   b. F tank emergency boost pump switch — ON.

      Leave the switch ON as long as fuel is transferring. Return it to the OFF position when F tank is empty, or if fuel fails to transfer.

NOTE

Fuel transfer is slower than normal when using the emergency system.

3. Apply positive and negative g's to the airplane in an attempt to free any vent valves that may be stuck in an open or partially opened position.

4. If, after accomplishing the above, the F tank still fails to transfer, F tank fuel will not be available, and the flight should be planned accordingly.

NOTE

If the F tank pressure-low warning light illuminates and the F tank is empty, assume an erroneous indication.

FUEL QUANTITY-LOW WARNING LIGHT ILLUMINATED

When the usable quantity of fuel in a No. 3 tank reaches approximately 570 pounds, the appropriate fuel quantity-low warning light will illuminate. Should this occur, proceed as follows:

1. Check for asymmetrical fuel loading.
   a. If an asymmetrical fuel condition exists, refer to Fuel Imbalance, this section.
2. Land as soon as possible.

FLIGHT OPERATIONS WITH LOW FUEL QUANTITY

Additional care must be taken when conducting flight operations with fuel quantity-low warning lights illuminated or when total fuel indicated is less than 1000 pounds. Prolonged turns, decelerations, or negative g maneuvers should be avoided in order to preclude fuel flow interruption and subsequent engine flameout. The hopper tank will supply fuel for at least 30 seconds engine operation at 4000 pounds per hour flow with 650 pounds in each number 3 tank during the above maneuvers. However, at number 3 tank quantities below 650 pounds, the 30 second operating time has not been verified.

Should a fuel quantity-low light illuminate or total fuel quantity indicated reach 1000 pounds, proceed as follows:

1. Avoid prolonged turns, decelerations, or negative g maneuvers and maintain a nose high attitude, if practicable.

2. Land as soon as possible from a straight-in approach.

FUEL VALVE-CLOSED WARNING LIGHT ILLUMINATED

When any one of the shutoff valves (except F tank on Ⓑ airplanes) is in any position other than fully open, the fuel valve-closed warning light will illuminate. Should this occur, proceed as follows:

1. Check the fuel control panel to determine which valve is closed.
2. Check for asymmetrical fuel loading.
3. Land as soon as possible.

AC POWER FAILURE WARNING LIGHT ILLUMINATED

The ac power failure warning light will illuminate when the ac generator fails. To verify the warning indication, and to preclude engine flameout, use the following procedure:

1. Fuel boost pressure-low warning lights—Check (not illuminated).

   If fuel boost pressure-low warning lights illuminate, check ATG switch in AUTO position.
2. Throttle—83% minimum.
3. Altitude—Establish 35,000 feet or below.

NOTE

- If operating above 35,000 feet at the time of ac power failure, descend to 35,000 feet or below within two minutes.
- Starting loads of the ATG may require power settings of up to full military power. If the ATG does not start immediately, increase rpm to military power and descend with this throttle setting to 35,000 feet. Below 35,000 feet, 83% rpm will sustain starting and operating loads.

4. AC generator switch—OFF, then ON.
5. Cabin temperature control knob—HOT.
6. Rain removal switch—ON (if subsonic).
7. If warning light remains on:
   a. AC generator switch — OFF.
      Emergency ac generator power is available automatically.
   b. Land as soon as practicable.
8. Important system losses:
   a. Automatic vari-ramp control.
   b. Windshield and canopy anti-icing.
   c. Pitch and yaw dampers.
   d. If ATG is inoperative:
      (1) Fuel boost pumps.
          The fuel flow equalizer will automatically go into bypass with the loss of boost pumps.
      (2) Deleted.
      (3) Deleted.

(4) Deleted.

(5) Pitot heat (Nose Boom).

9. Land as soon as practicable. Avoid high Mach numbers. Make a nose-high, straight-in approach.

NOTE

If at any time both fuel boost pressure-low warning lights illuminate steadily, utilize procedures for fuel boost pump failure. It is a normal condition if the fuel boost pressure-low warning lights illuminate momentarily during change-over from ac power to ATG power. If it is determined that the ATG is not operating, place the ATG switch in the OFF position to preclude unnecessary damage to the unit.

DC POWER FAILURE WARNING LIGHT ILLUMINATED

The dc power failure warning light will illuminate when the dc generator fails. To verify the warning indication, and best utilize available electrical power, use the following procedure:

1. DC generator switch — OFF, then ON.
2. If warning light remains on:
   a. DC generator power switch — OFF.
   b. Land as soon as practicable.

NOTE

If both ac and dc generators have failed, the transformer-rectifier will assist the battery in powering dc essential buses.

3. Important system losses:
   a. Automatic vari-ramp control.
   b. Automatic cg fuel transfer.
   c. F tank emergency boost pump control.
   d. Taxi and landing lights.
   e. Anticollision lights.
   f. Pitch and yaw dampers.
   g. Automatic ice detector.
   h. Fuel in external wing tanks.

AC AND DC POWER FAILURE WARNING LIGHTS ILLUMINATED

If the ac and dc generators fail, or the CSD unit fails, all items on the ac and dc nonessential buses will be lost. The items on the ac essential bus will be sustained by the emergency ac generator which cuts in automatically. The items on the dc essential bus will be sustained by the emergency dc power package, which automatically assumes the load. The ATG will sustain the boost pumps and pitot heat.

NOTE

- The CSD failure sequence causes a momentary mismatch of the N_1 and N_2 compressor speeds. This mismatch combined with the momentary fuel boost pump pressure reduction may cause engine flameout. Airstart should be successful using the normal fuel control system.
- The emergency ac generator, through the transformer-rectifier, will assist the battery in powering the dc essential bus.

See figure 3-5 for effect of ac and dc power failure on various components.

COMPLETE GENERATOR FAILURE

If electrical power failure results in loss of the dc generator, the ac generator, the emergency ac generator and ATG, the 28-volt battery will still be available to supply electrical power for the dc essential functions.

NOTE

When operating from the 28-volt battery only equipment needed to maintain flight should be utilized. See figure 3-5 for effect of complete generator failure on various components.

The following are either independent of electrical power, or require battery power only and will continue to function with complete generator failure:

1. All airplanes:
   a. Armament salvo.
   b. Cabin pressure altitude gage.
   c. Canopy jettison.
   d. Drag chute.
   e. Emergency fuel transfer.
   f. Exhaust gas temperature gage.
   g. Landing gear and tailhook down warning lights.
   h. Manual engine anti-icing.
   i. Speed brakes.

It should be noted that several instruments are completely independent of the electrical power sources and will continue to function with a complete loss of electrical power. The instruments are:

1. Conventional instrument display:
   a. Airspeed phase of the airspeed-angle of attack indicator.
   b. Cabin pressure altitude gage.
   c. Accelerometer.
   d. Tachometer.
   e. Vertical velocity indicator.
   f. Altimeter.
   g. Standby magnetic compass.
2. Integrated flight instrument system:
   a. Tachometer.
   b. Standby altimeter.
   c. Standby airspeed indicator.
   d. Standby magnetic compass.
   e. Cabin altitude marker.

HYDRAULIC POWER SYSTEM FAILURE

NOTE

Make an entry on Form 781 if the ram air turbine has been operated under a cavitated condition. This condition would occur if all hydraulic fluid leaked out due to a leak in a primary hydraulic system line.

In the event of failure of one hydraulic system, the other system will remain in operation, but will produce only approximately half the power to the flight controls, since both hydraulic systems are almost identical in capabilities. Loss of both hydraulic systems will require displacement of the ram air turbine for flight control operation. With complete ac power failure (which may accompany failure of both hydraulic systems if the cause is a "frozen" engine), two of the instruments which will be rendered inoperative are the hydraulic pressure gages. The pressure gages in this instance may not necessarily return to a reading of zero pressure but may continue to indicate the pressure readings at the time of failure. After displacement of the ram air turbine, the only indication of hydraulic pressure will be the dc (battery) operated hydraulic pressure-low warning light and physical movement of the flight controls.

CAUTION

During flight, an above-normal increase in hydraulic pressure on either system (i.e., over 3100 psi), with no demand on the system, is an indication of possible hydraulic system component failure, and a landing should be accomplished as soon as practicable.

HYDRAULIC PRESSURE-LOW WARNING LIGHT ILLUMINATED (FLASHING)

If the hydraulic pressure-low warning light flashes, it is an indication that one of the hydraulic systems has failed. To best utilize the remaining system, proceed as follows:

1. Airspeed—Reduce below 345 KCAS.
   Reduce airspeed to below ram air turbine maximum extension speed by immediate thrust reduction as required so that the emergency system will be readily available in the event of failure of the remaining system.

   CAUTION

   Do not use the speed brakes, as this will reduce the pressure available for flight control.

2. Hydraulic pressure gages — Check.
   Check the hydraulic pressure gages to determine which system gives the low pressure indication, then proceed with applicable procedures.

Failure of Primary Hydraulic System

1. Avoid violent maneuvers, dives and unnecessary use of speed brakes.
2. Land as soon as practicable.
   a. Landing gear handle — DOWN.
      Extend the landing gear with the normal system.

      WARNING

      Landing gear extension should be made at a time when minimum use of flight controls is required. Speed brakes should not be used during gear extension since the secondary hydraulic system will be used for both landing gear extension and flight control operation. Maintain at least 80% engine rpm during landing gear extension to increase secondary hydraulic system capabilities.

   b. RAT — Extend.
      Extend the ram air turbine after entering the landing pattern and after extending the gear.

electrical power failure

MA-1 FAILED

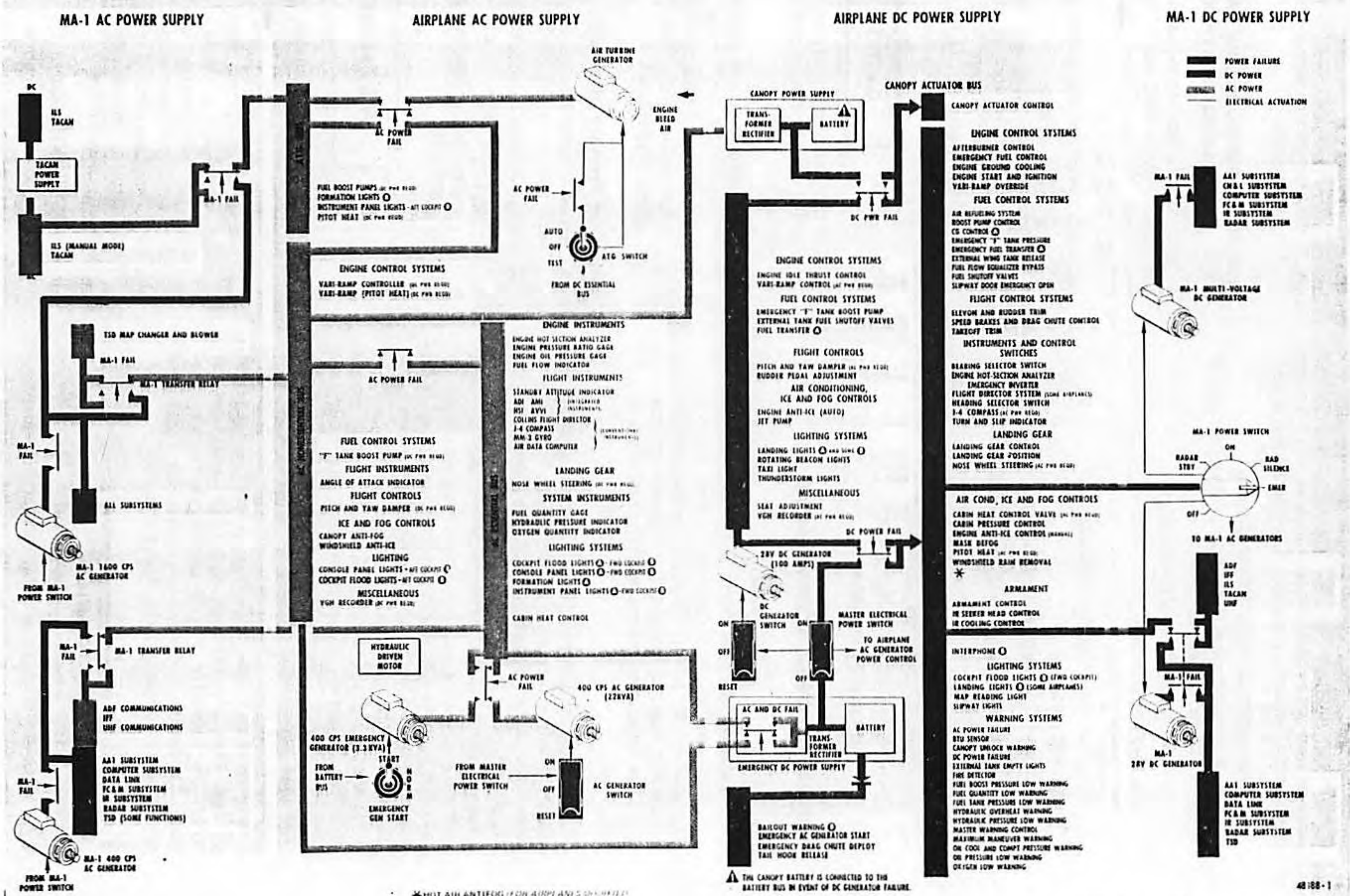


Figure 3-5 (Sheet 1 of 3)

electrical power failure

AC AND MA-1 FAILED

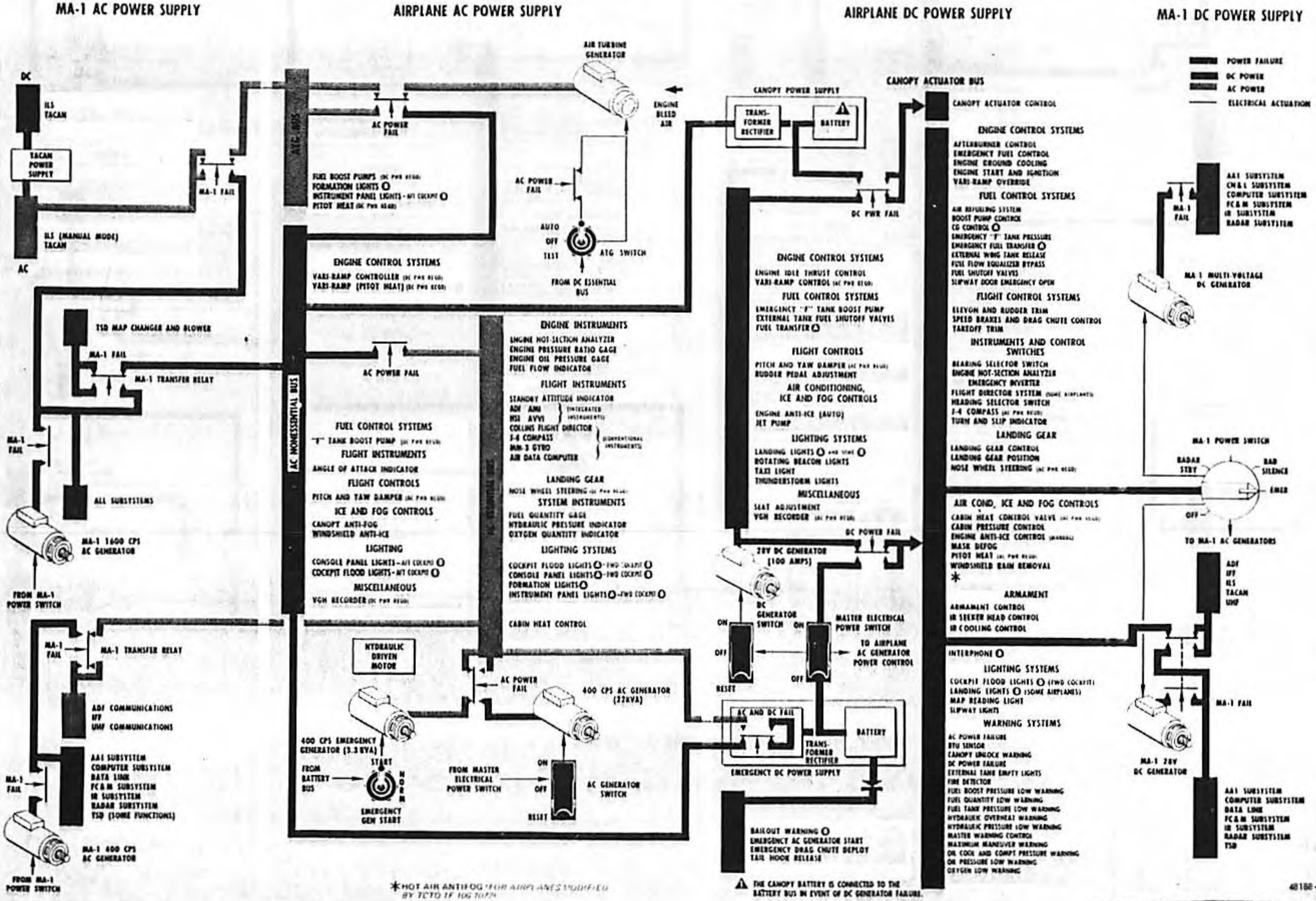


Figure 3-5 (Sheet 2 of 3)

electrical power failure

AC, DC AND MA-1 FAILED

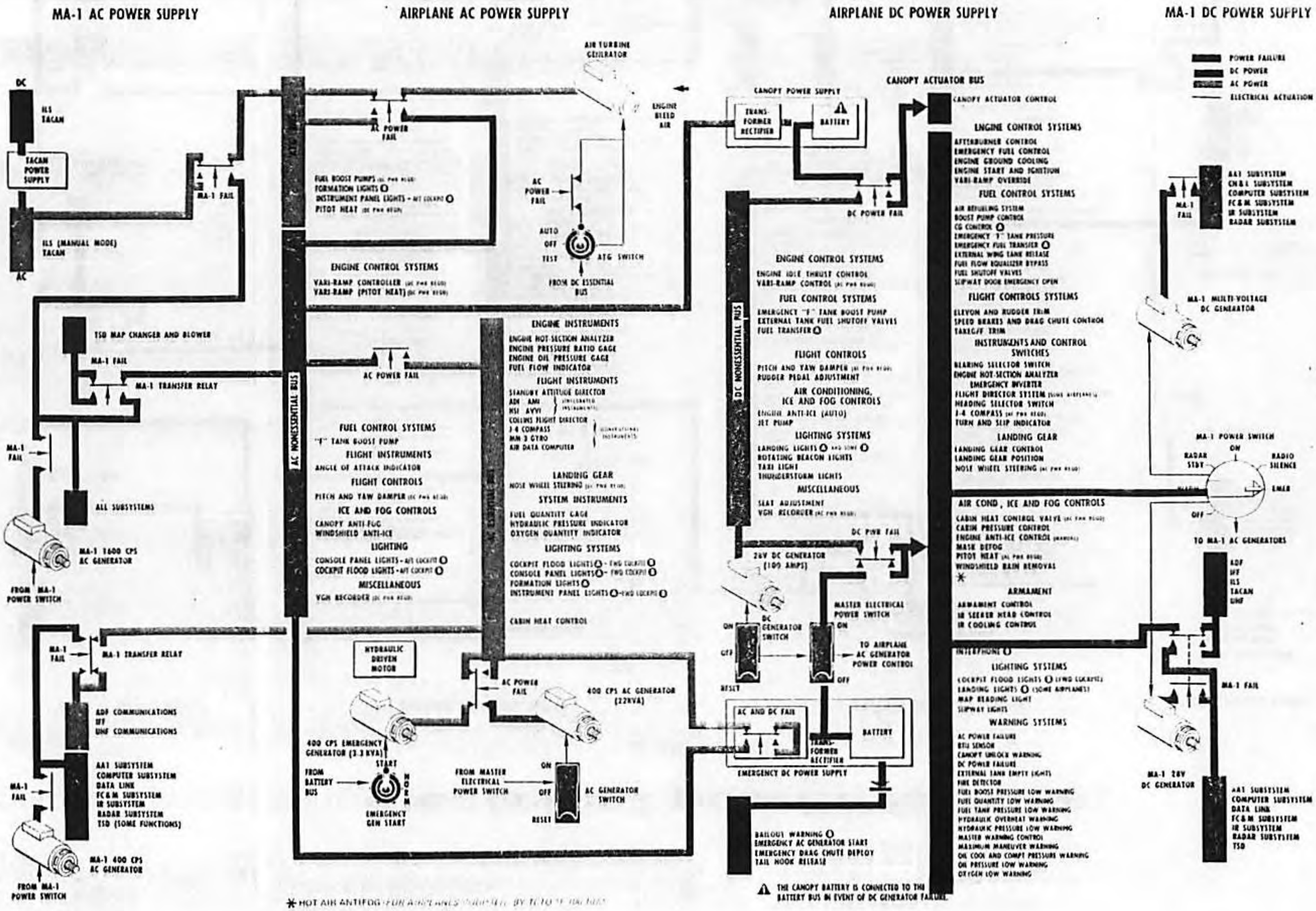


Figure 3-5 (Sheet 3 of 3)

Failure of Secondary Hydraulic System

NOTE

If the secondary hydraulic system fails, the following will be inoperative: nose wheel steering, pitch damper, emergency ac generator, normal speed brake operation, normal variable ramp operation, normal landing gear operation and normal slipway door operation (if installed). With the secondary hydraulic system failed, control stick breakout forces will increase, and positive centering of the stick may not be possible. In some instances failure of the secondary system results in an increased sensitivity of the flight controls.

1. Flight mode selector switch — YAW.
   Place the flight mode selector switch to YAW to prevent "wandering" of the control surfaces.
2. Avoid violent maneuvers, dives, and use of speed brakes.
3. Land as soon as practicable.
   a. Landing gear emergency extension handle —Press down, then pull (250 KCAS).

      An emergency gear extension should be recorded on Form 781.
   b. Landing gear handle — DOWN.
      Place the landing gear handle DOWN to avoid landing gear warning light indication and audio warning.

WARNING

Do not extend the ram air turbine. Extension will serve only to increase the temperature of the hydraulic system. An increase of temperature could have an adverse effect on primary hydraulic system operation.

   c. Drag chute handle—Emergency deploy (on landing).

WARNING

If secondary hydraulic system failure has been caused by a known malfunction in the speed brake system, asymmetrical speed brake extension may occur if the drag chute is deployed. Therefore, a no-drag-chute landing should be planned if runway conditions, length and barrier availability permit. If use of drag chute is necessary, nose wheel should be lowered to the runway prior to deploying the drag chute.

HYDRAULIC PRESSURE-LOW WARNING LIGHT ILLUMINATED: "HYD FAIL" (STEADY)

If the hydraulic pressure-low warning light illuminates steadily, it is an indication that both systems have failed. Proceed as follows:

1. **RAT—EXTEND.**

WARNING

- Minimum airspeed is 174 KCAS when the ram air turbine is the only source of hydraulic pressure. However, this will not assure control for other than wings level flight.
- If airspeed is above ram air turbine maximum extension speed and engine is "frozen," do not extend the speed brakes to slow the airplane. Speed brake extension will cause immediate depletion of remaining secondary hydraulic system pressure. Deceleration should be accomplished by moving the flight controls a minimum amount to establish a flight attitude that will provide deceleration.

2. **IF FLIGHT CONTROL OPERATION IS NOT POSSIBLE — EJECT.**
3. Check flight control operation by:
   a. Moving the control stick and checking control response.
   b. Checking primary hydraulic pressure gage (if ac power is available).
   c. Checking that hydraulic pressure-low warning light starts to flash indicating that the ram air turbine is providing pressure through the primary system.

NOTE

In the event that hydraulic system failure is the result of a frozen engine, the hydraulic pressure gage will be inoperative due to complete ac power failure and, therefore, will not indicate hydraulic pressure even though the emergency system is operating. In this event the pressure gage will not necessarily return to a reading of zero pressure. At this time, the dc (battery) operated hydraulic pressure-low warning light will be the only instrument for indication of hydraulic pressure.

4. If pressure is available for flight control operation, use the following procedure:
   a. Avoid violent maneuvers and use of speed brakes.
   b. Extend landing gear by emergency system.
5. Land as soon as practicable.
   Plan landing to provide for straight-in final approach, using a minimum of control movement for a few seconds before flareout. This is necessary to assure adequate hydraulic pressure for normal elevon operation during the flareout.

FLIGHT CONTROL SYSTEM MALFUNCTIONS OR OSCILLATIONS

Flight control oscillations are primarily due to failure of the damper system, trim system, or of the artificial feel system. Gradual failure of either hydraulic system may be accompanied by the excessive generation of heat within the system. Elevon surface oscillations can result from overtemperature of the hydraulic fluid at the elevon control valve. If the oscillations are due to overtemperature there will be movement of the control stick; however, if the oscillations are due to the damper system there will be no accompanying movement of the control stick. Failure of flight control system components may be accompanied by a partial or full deflection of the rudder or elevons. Control may be regained as the failed component is disconnected electrically by following the prescribed emergency procedure. Heavy control stick forces may be encountered which must be overcome physically by the pilot. A system failure that causes an intermittent control surface deflection can creat an extreamely hazardous situation by allowing the pilot to regain control only to lose it again during a critical phase of the flight. Failures of this nature are rare; however, when they occur and full control of the plane cannot be regained, abandoning the aircraft is the only alternative.

FLIGHT CONTROL SYSTEM OSCILLATIONS, MALFUNCTION, AND/OR HYDRAULIC FLUID OVERHEAT WARNING LIGHT ILLUMINATED

On some airplanes a warning light, located on the master warning light panel, is provided to give indication of hydraulic system overtemperatures. If the hydraulic fluid overheat warning light is on, or uncontrollable oscillations or maneuvers are encountered, proceed as follows:

1. **REDUCE AIRSPEED TO 230 KCAS.**
   Reduce engine RPM to the minimum required to maintain this speed.
2. **EMERGENCY DIRECT MANUAL BUTTON—DEPRESS.**
3. **FLIGHT MODE SELECTOR SWITCH—DIR MAN.**

NOTE

Pilot induced oscillations, which may occur at transonic speeds when the flight mode selector switch is in the DIR MAN position, are a phenomenon not related to flight control system oscillations.

4. If uncontrollable maneuvers are encountered:
   a. **DC GENERATOR SWITCH—OFF.**
   b. **IF CONTROL IS NOT REGAINED, MASTER ELECTRICAL POWER SWITCH—OFF, CLIMATIC CONDITIONS PERMITTING.**
   c. Attempt to maintain control of the airplane by overcoming any heavy control stick forces encountered.
   d. If control cannot be regained—EJECT.
   e. If control is regained, land as soon as practicable.
5. If oscillations are encountered:
   a. Activate alternate flight modes (yaw and pitch damper modes) and remain in the flight mode that gives the best stability.

NOTE

To aid in reducing hydraulic system fluid temperatures, use minimum control movements at all times. If oscillations occur, they may be more rapidly reduced in magnitude by releasing the control stick than by correcting for the oscillations with stick input.

6. If oscillations persist or increase:
   a. DC generator switch—OFF.
   b. If oscillations continue, master switch—OFF.
   c. If oscillations continue, master switch—ON.
   d. Maintain safe altitude while severe oscillations persist.
   e. When oscillations reduce to a safe minimum, land as soon as practicable.
   f. If an airplane becomes uncontrollable—EJECT.

WARNING

Do not extend the ram air turbine. Extension of the ram air turbine will serve only to increase the hydraulic temperatures in the primary hydraulic system.

7. During low fuel conditions attempt to keep nose level attitude.

NOTE

The heat exchanger utilizes fuel from the No. 3 tank to cool the hydraulic fluid; therefore to maintain the most effective cooling during low fuel conditions (1000 pounds or less) the airplane should not be flown in a 5° or greater nose-low attitude except as required for a descent to a landing.

8. If an overheated condition exists, maintain altitude to evaluate the nature of the system overheat.

NOTE

It is possible that an overheated system may cool if flight is continued under conditions dictated by the above procedures.

TRIM FAILURE

If any of the three trim actuators should fail to operate, the force required to perform normal maneuvers should be well within normal physical capabilities. Complete loss of dc power will cause the trim system to be inoperative.

DAMPER MODE FAILURE

Both damper systems become inoperative if either the ac or dc generator fails. Yaw damper is lost when primary hydraulic system pressure fails, and secondary hydraulic pressure failure will result in the loss of pitch damper. If any of these conditions occur, the hydraulic control valves permit normal pilot control without stability augmentation.

ARTIFICIAL FEEL SYSTEM INOPERATIVE

Failure of the artificial feel system may be caused by a leak in the pneumatic pressure line, a leak in the feel force cylinder, internal failure of the feel force regulator, or by electrical failure to either or both of the ram air ("q") intakes, allowing them to become clogged by ice while operating in icing conditions. Failure of the artificial feel system may be recognized by either high or low stick and/or rudder forces, or pilot-induced oscillations. In the event of artificial feel system failure, proceed as follows:

1. Airspeed—Reduce to 230 KCAS.
2. Use minimum control stick movement.
3. Flight mode selector switch—DIR MAN.

WINDSHIELD HEATING FAILURE

The laminated windshield is electrically heated to prevent fogging. In the event of loss of electrical power to the laminated windshield, windshield defogging may be obtained by the following procedure:

1. If the windshield is heated and ac power failure is noted, immediately proceed as follows:
   a. Rain removal switch — ON (below supersonic speed).

      If the system has functioned normally, maintaining a heated windshield, and ac power failure is noted, the rain removal system should be turned on immediately. Activation of the rain removal system will provide an effective "heat block" for the heat energy stored in the left-hand windshield panel at the time of power failure and maintain the inside surface temperature of the left-hand panel.
2. If system power failure is not noted until condensation has formed, proceed as follows:
   a. Rain removal switch — ON (below supersonic speed).

      Activation of the rain removal system will initiate a heat addition process on the left-hand windshield outer surface.
   b. Cabin temperature control knob — HOT.

      Regulate cabin temperature to maximum tolerable. This action will start a heat addition process on the inside surface of the windshield.

NOTE

The desired heating on a windshield that has cooled, or was not preheated electrically, will require from two to four minutes when using the above procedure.

CRACKED WINDSHIELD

At high altitudes and low temperatures, uneven heat distribution, caused by possible malfunction of the electrical input to the defogging system, could cause a windshield panel to crack. A hair-line crack, which leaves the panel transparent, indicates an outer layer failure only. However, if the panel becomes white or opaque, the inner layer, which is the stress-bearing part of the windshield, has probably failed. Inner layer failure may further be evidenced by feeling the glass, and by observing particles of shattered glass within the cockpit. In the event of a cracked windshield, proceed as follows:

Outer Layer Crack

1. Windshield anti-icing, antifog switch — OFF (on side corresponding to the cracked panel).

CAUTION

Subsequent cracking may short the anti-icing, antifog circuit if the windshield failure is due to excessive heat.

2. Ascertain that only the outer layer has cracked.

NOTE

Cabin pressure need not be dumped if only the outer layer is cracked.

3. Land as soon as practicable.

Inner Layer Crack

1. **HELMET VISOR(S) — DOWN. (FP-RP)**
2. **CABIN AIR SELECTOR SWITCH — RAM.**

WARNING

Cabin pressure must be dumped immediately in the event of an inner layer crack.

3. Windshield anti-icing, antifog switch — OFF (on side corresponding to the cracked panel).

CAUTION

Subsequent cracking may short the anti-icing, antifog circuit if the windshield failure is due to excessive heat.

4. Airspeed — Reduce.
5. Descend, and land as soon as practicable.

CANOPY UNLOCKED WARNING LIGHT ILLUMINATED

Illumination of the canopy unlocked warning light in flight indicates that the canopy latch hooks are not fully engaged, and loss of the canopy could ensue. If the light comes on in flight, observe the following:

1. If the canopy unlocked warning light illuminates in flight when the canopy latch handle is in an apparent full forward position, and the canopy latches are engaged, carefully push the handle (fully) to the LOCK position. Any slight movement of the latch mechanism (.060 inches) in the aft direction would cause the warning light to illuminate.

NOTE

Emergency IFF is activated when the canopy warning light is illuminated.

WARNING

Under no circumstances should the canopy latch handle be moved aft. Windblast would jettison the canopy.

2. If warning light fails to go out, handle is not in an apparent locked position, or canopy latches are not engaged:
   a. Airspeed—Reduce to 230 KCAS.
   b. Cabin air selector switch — OFF.
   c. Land as soon as practicable.

ENGINE COMPARTMENT OVERPRESSURE WARNING LIGHT ILLUMINATED

Illumination of the engine compartment overpressure warning light while in flight indicates that pressure has reached 3.0 psi above ambient in the engine accessory compartment. If the light illuminates, proceed as follows:

1. Airspeed — Reduce to below Mach 1.0 as soon as practicable.

   Gradually reduce airspeed below Mach 1.0 within 2½ minutes to prevent possible damage to the fuselage from excessive pressure, and to conserve high-pressure pneumatic system pressure. Continued subsonic operation with the light on presents no hazard.

NOTE

Once illuminated, the light will remain on for the duration of the flight.

PNEUMATIC PRESSURE-LOW WARNING LIGHT ILLUMINATED

When pressure in the two main pneumatic pressure storage flasks drops to 1700 (±50) psi, the pneumatic pressure-low warning light will illuminate. To conserve remaining pressure, proceed as follows:

1. Do not recycle the armament system.
2. Use rudder control with caution.
3. Plan on no-drag-chute landing.
4. There is a possibility of loss of one or both brakes.

OXYGEN-LOW WARNING LIGHT ILLUMINATED

When the oxygen supply in the liquid oxygen system falls below one-half liter on airplanes with a five liter converter and one liter on airplanes with a ten liter converter, the oxygen-low warning light illuminates. Should this occur, proceed as follows:

1. Descend to an altitude where oxygen is not required.
2. Actuate emergency bailout bottle, if necessary.

CABIN PRESSURE-LOW WARNING LIGHT ILLUMINATED

When the cabin altitude rises to 44,000 (±2000) feet the cabin pressure-low warning light will illuminate and the emergency cabin pressure system will be energized. As cabin pressure increases the warning light and the emergency pressure system will be deenergized when the cabin altitude drops to 26,000 (±2000) feet. In the event this light illuminates during flight, proceed as follows:

1. If practicable, descend to 24,000 feet cabin altitude to shut off the emergency cabin pressure system.
2. Land as soon as practicable.

HOT AIR BLEED LINE/DUCT FAILURE

Problems detected in the air conditioning and pressurization system, electronic cooling, and electrical system (when persistent), or any combination of these problems, may be an indication of hot air bleed line/duct failure. When such a failure is suspected on the ground, the flight should be aborted and the engine shut down. When failure is suspected while airborne, hot air bleed line/duct failure should be treated as an emergency requiring minimum thrust and landing as soon as possible. In a malfunction caused by rupture of hot air bleed line, continued flight can be extremely hazardous.

WARNING

After landing, when clear of the runway, open the canopy, shut down the engine, and abandon the airplane. Do not turn off any electrical switches to preclude the possibility of explosion of fuel fumes.

ARMAMENT EMERGENCY PROCEDURES

ARMAMENT JETTISON PROCEDURE

The AIR-2A rocket and the AIM-4F and AIM-4G missiles may be jettisoned (salvoed) in one operation. The salvo selection is provided to insure jettisoning of all armament in an emergency. The salvo operation is completely independent of MA-1 functions, and to ensure this capability in event of a d-c generator failure, the electrical power required is taken from the essential d-c bus. The special weapon is jettisoned first followed by the two aft bay missiles and then the two forward bay missiles. The special weapon is ejected inert with no rocket motor iginition. The missile motors will function but the missile is unarmed. The maximum time required for the salvo cycle is about 6.5 seconds during which the armament trigger must be held pressed. If the missiles are desired to be salvoed and the special weapon retained aboard, the special weapon release lock switch must be kept in the LOCK position. In the event of a misfire of the AIR-2A rocket ejection system, the aft missiles are extended approximately 3.25 seconds after the ejection signal. The missile sequence will then continue normally, but the doors will not close automatically. If a missile misfires during SALVO, the cycle stops as it would for a normal 4-missile pass. The SALVO circuits also provide for complete operation to jettison all remaining armament if some of the load has been previously fired or misfired. Use the following procedure for armament jettison.

1. Armament selector switch – SALVO.
2. ARM-SAFE switch – ARM.
3. Special weapon release lock switch – UNLOCK.
4. Special weapon release lock indicator – UNLOCK.
5. Armament trigger – Press to second detent.

   When the trigger is pressed, the missile bay doors open, the armament is jettisoned, and the doors close. The trigger must be held pressed through the cycle.

6. MISFIRE warning light – Monitor.

   The MISFIRE and MASTER WARNING lights flash briefly when the AIR-2A rocket is ejected, again when the aft missiles are launched, and finally when the forward missiles are launched.

7. Armament trigger – Release.

MISFIRE OR DOOR OPEN WARNING LIGHT ILLUMINATED

Use the following procedure if the missile bay DOOR OPEN or MISFIRE warning light remains illuminated:

1. Armament trigger – Release (if pressed).

2. AUTO SEARCH button – Press to break lockon.

3. Special weapon release lock switch – LOCK.

4. Special weapon release lock indicator – LOCK or striped.

   If the rocket has been launched, the special weapon release lock indicator will display a barber pole. If the rocket is still aboard, the indicator will indicate LOCK.

5. MISSILE DISPLACED warning light – Check NOT illuminated.

   If this light is illuminated, follow the MISSILE DISPLACED procedure for applicable configuration.

6. 75 seconds – Wait (if missiles are aboard).

WARNING

If emergency procedures are necessary for retraction, the missiles must not be retracted less than 75 seconds after full extension. A potential hazard of an "in bay" explosion, resulting in severe aircraft damage and pilot danger exists if AIM missiles are retracted into the missile bay with the electrical-hydraulic power supply still operating.

7. ARM-SAFE switch – SAFE.

8. Armament selector switch – VIS IDENT.

9. Missile bay DOORS CLOSE button – Press and hold until misfire and door-open warning lights extinguish.

MISSILE DISPLACED WARNING LIGHT ILLUMINATED

Illumination of the MISSILE DISPLACED warning light indicates that a missile is displaced, i.e., not properly seated on a launcher. This light may be accompanied by illumination of the MISFIRE and DOOR OPEN warning lights. If the light illuminates during ground operation, the cause should be determined and the condition corrected prior to flight. Use the following procedures for inflight illumination of the warning light:

MISSILE DISPLACED WARNING LIGHT ILLUMINATED WITH MISSILE BAY DOORS CLOSED

1. If the MISSILE DISPLACED warning light illuminates in flight and the launchers are known to be retracted (indicated by doors being closed), the aircraft should be landed without further armament operation.

MISSILE DISPLACE WARNING LIGHT ILLUMINATED WTH LAUNCHERS EXTENDED

1. If a clean aircraft configuration is not mandatory, proceed as follows:

   a. Fuel consumption – Monitor for range considerations.

   b. Land as soon as practicable with the launchers extended.

CAUTION

Retraction of missiles with the MISSILE DISPLACED warning light illuminated can result in structural damage to the aircraft and missiles. Retraction under these conditions should be performed only when absolutely necessary for safe recovery of the aircraft.

2. If a flight condition exists wherein a clean aircraft configuration is mandatory, use the following procedure to retract the launchers:

   a. AUTO SEARCH button – Press to break lockon.

   b. 75 seconds – Wait.

   c. Arm-SAFE switch – SAFE.

   d. Armament selector switch – VIS IDENT.

   e. Missile bay DOORS CLOSE button – Press and hold until DOOR OPEN warning light extinguishes.

THROTTLE HANGUP

NOTE

Throttle hangup may occur due to freezing of moisture in the throttle teleflex cable. This problem can be eliminated by descending to below the freezing level.

With throttle hangup or throttle linkage failure at high rpm where throttle control cannot be regained, the only method of stopping the engine is with the fuel shutoff switches or by fuel starvation. See Section I, FUEL SHUTOFF SWITCHES, for time between fuel shutoff and thrust loss at various power settings. When attempting to land with throttle hangup, the following procedure should be used.

1. Retain drop tanks and burn off excess fuel in the vicinity of base of intended landing.
2. Press to test fuel shutoff valve warning lights.
3. Slow aircraft to below maximum gear lowering speed by use of speed brakes and climbing.
4. Descend to traffic pattern altitude and fly a long straight-in final approach. Anticipate a higher than normal approach speed if rpm is above that required for normal approach speed.

NOTE

- Selection of emergency fuel may reduce thrust.
- "S" turns on final approach may be required to reduce speed if the throttle cannot be retarded below 90% RPM.

5. At approximately 1 to 2 miles from the approach end of the runway, move boost pump switches to OFF and close the fuel shutoff switch on one side. Check fuel warning light on.
6. When landing is assured, close the remaining fuel shutoff switch. Check fuel warning light on.
7. Execute normal touchdown and landing. Anticipate landing long and be prepared to engage the barrier.

NOTE

Landing roll can be reduced by selecting afterburner immediately after touchdown. This dumps fuel into the afterburner lines and rapidly depletes fuel downstream from the fuel shutoff valves thus accelerating engine coastdown. The afterburner will not light if the fuel shutoff valves are closed.

STRUCTURAL DAMAGE

If structural damage occurs in flight, the pilot must decide whether to leave the aircraft at that point, proceed to an area that is more favorable for bailout, or to attempt a landing. If the aircraft remains controllable, the pilot must attempt to ascertain what damage has occurred and to what degree it will affect continued flight or attempted landing. Obvious indications to the pilot will be control forces (response) necessary to continue flight and engine or systems instruments in the cockpit. If another aircraft is available, that pilot should position himself to safely observe and report the damage that has occurred. If the pilot elects to attempt a landing, he should check for aircraft controllability by simulating a landing approach. If time, fuel and weather permit, the pilot should position the airplane near the field of intended landing at a minimum of 15,000 feet. At this point he should:

a. Simulate a landing approach with gear down, speed brakes retracted (switch neutral) at recommended approach speed for fuel aboard.

b. If aircraft is not controllable at approach speed, eject.

c. If aircraft is controllable at approach speed, plan for a straight-in approach without speed brakes. Once the gear is down on the simulated approach, leave it down.

d. Maintain recommended approach speed until landing is assured.

e. Runway conditions permitting, lower nose wheel to the runway prior to deploying drag chute.

SPEED BRAKE STRUCTURAL FAILURE

If an inflight loss of one or both speed brakes occurs, the speed brake switch must be returned to the neutral position to prevent failure of the secondary hydraulic system due to fluid loss. Loss of both speed brakes is evidenced by a slight nose down pitch change and loss of deceleration force as though the speed brakes were closed. Loss of a single speed brake may induce a severe yaw and roll in the direction of the remaining speed brake. Flight tests indicate that at an airspeed of 150 to 200 KCAS, 90% of full rudder is required to maintain zero sideslip with a single speed brake extended. However, no flight tests have been conducted at speeds above 200 KCAS. Operational experience indicates that at speeds above 200 KCAS, the airplane is uncontrollable until the remaining speed brake is closed.

If an inflight loss, or suspected loss, of one speed brake occurs, as evidenced by a severe yaw and rapid roll, immediately close the remaining speed brake and return the switch to the neutral position. Adequate rudder is available to counter the yaw and roll tendency at airspeeds below 200 KCAS, however, above this speed airplane control may not be regained until the remaining speed brake is closed.

NOTE

Refer to Section IV for the following:

- **ENGINE ANTI-ICING WARNING LIGHT ILLUMINATED: "ENGINE ANTI-ICE"**
- **HYPOXIA SYMPTOMS**
- **OXYGEN SYSTEM DEPLETED OR CONTAMINATED**
- **LOSS OF CABIN PRESSURE**
- **ELECTRONIC COOLING WARNING LIGHT ILLUMINATED: "ELECTRONIC COOLING"**
- **REFRIGERATION UNIT FAILURE**

LANDING

LANDING GEAR WARNING LIGHT REMAINS ILLUMINATED ON LANDING GEAR EXTENSION

Recycle the landing gear handle. After recycling attempts, if the landing gear warning light does not go out, the landing gear green indicator lights do not illuminate, or the audio warning signal continues with the landing gear handle in the DOWN position, use the procedure for LANDING GEAR EMERGENCY EXTENSION.

NOTE

On night missions, in addition to pressing to test the green indicator lights, move the warning lights dimmer switch to the BRT position. If a safe indication is obtained on BRT, a resistor in the dimming circuit is defective.

LANDING GEAR EMERGENCY EXTENSION

1. Speed—Below 250 KCAS.
2. Landing gear emergency extension handle—Press down, then pull.

WARNING

Insure that the landing gear emergency extension handle is fully extended when actuated. Recycling the handle after it has once been fully extended will induce air into the secondary hydraulic system and may fail the system, cause flight control oscillations and deplete the pneumatic system.

NOTE

- Time required for emergency extension of the landing gear is four to six seconds.
- Nosewheel steering is inoperative when the landing gear has been extended with the emergency system.

3. Landing gear handle—DOWN.

NOTE

Emergency landing gear extension can be accomplished with the landing gear handle in either the up or the down position; however, if the handle is in the up position, the red warning light will illuminate and the audio warning will sound even though the landing gear is extended.

4. Landing gear—Check down and locked.

NOTE

- Should use of the emergency extension system fail to provide a safe indication because of a hung landing gear door, the gear door may be removed by airloads allowing the landing gear to fall to the down and locked position. It should first be verified visually by a chase airplane or from the ground that a main landing gear door is partially opened, preventing landing gear extension. If the door is partially opened, accelerate to above 310 KCAS at which time the hung door should depart the airplane and a down and locked indication should appear on the landing gear position indicator. This procedure should be used only as a last resort prior to landing with one or both main landing gears retracted. Substantial damage will occur to the door actuating system and loss of secondary hydraulic system is probable.
- If a safe indication cannot be obtained on the main landing gear, then an approach end barrier engagement should be considered. Bouncing the airplane on the runway in an attempt to jar loose a hung landing gear is not recommended. This procedure has had little success in the past and exposes the pilot and airplane to hazards more critical than landing with an unsafe gear.

5. Landing or taxi lights—On.

   Place the landing and taxi light switch to LANDING LIGHTS if a main gear is unsafe, or to TAXI LIGHTS if the nose gear indicates unsafe. The light on each respective landing gear will illuminate if the gear is down and locked. Have mobile or the tower check to determine if the corresponding lights have illuminated.

NOTE

- Maximum allowable airpseed after the gear is extended is 280 KCAS.
- Record emergency gear extension on Form 781.

RUNWAY APPROACH END ARRESTMENT (BAK-6, BAK-9, BAK-12)

Engagement of the BAK-6, 9, or 12 type arresting gear at the approach end of the runway is recommended when a condition occurs which is likely to cause loss of directional control after touchdown or when extended runout can cause progressive failure, such as a blown main gear tire or landing with one or both main gear up or unlocked.

CAUTION

Approach end arrestments are not recommended if the only problem is failure of the nose gear to extend. Emergency landings utilizing a normal approach and landing roll, have experienced no control difficulty and only minor damage.

Engagement of the BAK-6, 9, or 12 barrier at the approach end of the runway will provide the following:

a. Additional directional stability.

b. A reduction in time that the pilot and airplane are exposed to possible injury or damage.

c. Limit airplane rollout or uncontrolled skid to 1500 feet or less.

d. A specific location where emergency vehicles can plan to meet the airplane.

Refer to figure 5-4 for maximum barrier engagement ground speed. The decision to attempt an approach end arrestment must be dictated by local conditions. Where the arresting cable is installed at the extreme end of the runway or in the overrun, the ability of the overrun area to accommodate touchdown loads should be considered. Touchdown on poorly paved overrun may prevent an arrestment through loss of directional control, hook skip, or damage to the airplane. During nose gear up arrestment premature lowering of the nose may cause the arresting cable to be picked up by the pitot boom; late lowering of the nose results in a severe nose down reaction at engagement. At least 750 feet of rollout is required, from an on-speed touchdown, to lower the nose wheel to the runway. Although desirable, lowering the nose prior to engaging the barrier is not mandatory. In all cases, when a condition occurs which is likely to cause loss of directional control after touchdown, an approach end arrestment is recommended whether or not the nose wheel can be lowered to the runway prior to engagement. There have been several inadvertent approach end arrestments in the F-106 where the aircraft was still in the landing attitude when the tailhook engaged the cable. There has been no injuries as a result of these engagements and damage has been minor with most instances being only a nose strut failure. This damage is minimal when compared to the risks involved with an uncontrollable aircraft at landing speeds. If it is decided that an approach end barrier arrestment is to be made, proceed as follows:

WARNING

Do not attempt to engage the MA-1 1A type barrier on the approach end as injury and damage will result. If time allows, the MA-1 1A cable should be removed before an approach end engagement of either the BAK-6 or BAK-9 barrier is attempted. If time does not allow for removal of the MA-1/1A cable, the tailhook must be deployed after passing the MA-1 1A barrier cable(s).

1. Expend excess fuel to lighten airplane and minimize fire hazard. Retain empty drop tanks.

NOTE

- Alert ground personnel and request BAK-9 or BAK-6 arresting gear be installed on approach end of runway.
- Request that the runway be foamed in arresting gear runout area to decrease fire hazard and landing gear side loads (time and conditions permitting). (BAK-9 runout distance is 950 feet, BAK-6 is 1500 feet.)
- Based on engaging the barrier at recommended touchdown speed with armament in, maximum fuel on board should be:

	BAK-6	BAK-9	BAK-12
Ⓐ	6500 lbs	11,000 lbs	9000 lbs
Ⓑ	4400 lbs	9000 lbs	7200 lbs

2. Perform airborne procedures listed in the flight manual for the respective conditions.
3. Tailhook down button—Depress.

CAUTION

Do not deploy the tailhook in flight where obstructions such as power lines, fences, approach lights, etc., could be engaged by the arrest hook.

NOTE

- Plan touchdown point according to type of emergency. A normal, smooth landing flare and touchdown is of primary importance. Therefore, do not "force on" or "hold off" airplane to achieve planned touchdown point.
- If conditions permit, touchdown point should be sufficiently short of the arresting cable to permit lowering of nose gear to runway before arrestment.
- Plan touchdown and barrier engagement to be as near the runway centerline as possible.

CAUTION

Premature touchdown and lowering of the nose with one main gear up will cause the wing to drop, resulting in loss of directional control which may permit airplane to veer off runway before reaching the arresting cable.

4. Shoulder harness inertia reel handle—MANUAL LOCK. (FP-RP)
5. Drag chute—Deploy.
6. Nose wheel steering—Engage.
7. Avoid using brakes just prior to engagement.

CAUTION

Do not attempt to lock brakes on wheel with flat tire until after crossing arresting cable. The wheel should be rolling when passing over the cable to avoid snagging or cutting the cable.

8. Throttle—IDLE or OFF.

CAUTION

When landing with a flat main gear tire, keep engine running until fire equipment arrives. Stopcocking the engine allows fuel to vent overboard near the wheel brakes, thus creating a fire hazard.

NOTE

Throttle advancement may occur during rapid deceleration if the pilot's hand is on the throttle.

9. Fuel shutoff switches—CLOSE.
10. Master electrical power switch—OFF (if required).

CAUTION

Master electrical power switch must be ON if nose wheel steering is required.

RUNWAY OVERRUN BARRIER ENGAGEMENT

Refer to TAKEOFF for overrun barrier engagement procedures.

LANDING WITH TAILHOOK EXTENDED

If the tailhook is inadvertently extended in flight, it will not introduce any problems while airborne. A safe touchdown can be made with the hook extended.

WARNING

If the tailhook is inadvertently extended in flight, and there is a barrier on the approach end of the runway, the airplane touchdown point must be compensated to avoid inadvertent engagement with the barrier. When extended, the tailhook may drag for 1000 feet before touchdown during a normal landing. When landing with the tailhook extended, fly a high final approach and touch down well past the arresting cable on the approach end of the runway, or the tailhook may engage the barrier and severely damage the airplane.

FLAT TIRE ON LANDING

WARNING

- If a known bare wheel exists, lock that wheel prior to touchdown and keep that wheel locked throughout landing roll to prevent wheel fragments from penetrating wing tanks.
- Do not hold the brake on the blown tire prior to barrier engagement. This may cause damage to the underside of the wing, but it will insure that the barrier will not be severed by the flat tire rim, thus increasing the success of barrier engagement.

If a tire is blown prior to landing or blows out on landing: move the throttle to idle, deploy the drag chute and move the idle thrust switch to the ON position. If the failed tire is on the main landing gear, the nose should be lowered immediately to the runway, nosewheel steering engaged, and the control stick moved away from the blown tire to relieve as much weight as possible from the wheel. Keep the nosewheel steering button depressed to maintain positive engagement. Differential braking and nosewheel steering should be used to maintain directional control. As vibration increases, the brake on the wheel with the blown tire should be held and locked. The good tire should not be deliberately blown as braking efficiency is reduced and the hazard of particles being thrown into the wings is increased. If the tire failure is on the nose gear, hold full up elevator to relieve as much weight as possible from the nosewheel. If nosewheel steering becomes uncontrollable, it should be disengaged.

NOTE

- If nosewheel steering is not available, or has been intentionally disengaged, and the airplane is above normal taxi speed, shut the engine down and follow ABORT procedure. Unless the situation dictates, do not stopcock the engine at speeds less than normal taxi speed, but maintain idle rpm until fire equipment arrives. Stopcocking the engine at slow airspeeds allows fuel to vent in the vicinity of the wheel brake area, creating a fire hazard.
- If a main landing gear tire is blown prior to landing, an approach end barrier engagement is the recommended landing procedure.

PARTIAL GEAR LANDING

If landing is to be made with partial gear extension, proceed as follows:

1. External wing tanks release button—Depress (if required).

    Depress the external tanks release button to jettison the external tanks if tanks are installed and contain fuel. If external tanks are empty, they should be retained to cushion the impact unless landing is to be made on an unprepared surface.

2. Plan normal approach and landing with speed brakes out.

    If either main gear is unsafe, plan the touchdown for side of runway corresponding to the safe gear.

3. Shoulder harness inertia reel handle—MANUAL LOCK. (FP-RP)

4. Immediately before touchdown:
   a. Throttle — OFF.
   b. Fuel shutoff switches — CLOSE.
   c. Master electrical power switch — OFF.
   d. Canopy—Retain.

WARNING

If the nose gear has not extended or has collapsed, do not jettison or open the canopy prior to passing all runway barriers as the arresting cable may slide up the nose and into the cockpit.

5. After touchdown:
   a. Drag chute handle — Pull (emergency deploy).
   b. Hold faulty gear off the ground.
      Hold the faulty gear off the ground, but ease it down before positive control is lost.
   c. Braking — As required.
      Do not use brakes if a safe stop can be made without them when the faulty gear is an uncollapsed nose gear.
6. Abandon the airplane when it stops.

BELLY LANDING

If forced to make a gear-up landing, proceed as follows:

CAUTION

If conditions permit, salvo the armament.

1. External wing tanks release button—Depress (if required).
   Depress the external tanks release button to jettison the external tanks if tanks are installed and contain fuel. If external tanks are empty, they should be retained to cushion the impact unless landing is to be made on an unprepared surface.
2. Make normal approach.
3. Speed brakes switch — As required.
   Open speed brakes prior to touchdown to ensure drag chute deployment.
4. Shoulder harness inertia reel handle—LOCKED. (FP-RP)
5. Immediately before touchdown:
   a. Throttle — OFF.
   b. Fuel shutoff switches — CLOSE.
   c. Canopy—Retain.

WARNING

If the canopy is to be jettisoned, make sure it is jettisoned before the airplane comes to a complete stop. Otherwise, sparks from the canopy remover may cause a fire if fuel is spilled in the vicinity of the airplane.

6. Normal landing attitude for touchdown.
7. Drag chute handle—Pull (emergency deploy).
8. Master electrical power switch—OFF.
9. Abandon the airplane when it stops.

EJECTION VS FLAMEOUT LANDING

Normally, ejection is the best course of action in the event of complete engine flameout (windmilling or frozen), or if positive control of the airplane cannot be maintained. Because of the many variables encountered, the final decision to attempt a flameout landing or to eject must remain with the pilot. It is impossible to establish a predetermined set of rules and instructions which would provide a ready-made decision applicable to all emergencies of this nature. The basic conditions listed below, combined with an analysis of the condition of the airplane, type of emergency, and pilot proficiency, are of prime importance in determining whether to attempt a flameout landing, or to eject. These variables make a quick and accurate decision difficult.

NOTE

No attempt should be made to land a flamed-out airplane at any field where approaches are over heavily populated areas. If possible, prior to ejection, an attempt should be made to turn the airplane toward an area where injury to persons or damage to property on the ground or water is least likely to occur.

Before a decision is made to attempt a flameout landing, the following basic conditions should exist:

a. Flameout landings should be attempted only on a prepared or designated suitable surface.
b. Approaches to the runway should be clear.
c. Weather and terrain conditions must be favorable. Cloud cover, ceiling, visibility, turbulence, surface wind, etc., must not impede in any manner the establishment of a proper flameout landing pattern.

NOTE

Night flameout landings, or flameout landings under poor lighting conditions such as at dusk or dawn should not be contemplated regardless of weather or field lighting.

d. Flameout landings should be attempted only when either a satisfactory high key or low key position can be achieved.

e. If at any time during the flameout approach, conditions do not appear ideal for successful completion of the landing, ejection should be accomplished. Eject not later than the low key altitude.

AIR START ATTEMPTS DURING FLAMEOUT LANDING PATTERN

It is suspected that a contributing factor in some recent unsuccessful flameout landings may have been that the pilot did not devote his full attention to the maintenance of the required flameout pattern once high key was reached. The distracting influence in these instances was air start attempts. The solution to this problem is not simply to prohibit air start attempts once high key is reached. Past experience has shown that flameout landings are dangerous and should be attempted only under ideal conditions. Accordingly, it would be dangerously misleading to create the impression that air starts should be abandoned at any specific time and all concentration be directed towards a flameout landing. If several air starts have already been attempted or if the nature of the flameout indicates that any further air start attempts will be futile, then obviously no further attempts should be made even long before the high key point is reached. But if flameout has occurred near high key point, the pilot can attempt an air start below high key provided he is not dangerously distracted. Regaining power must be afforded a high priority since, as already pointed out in previous instructions, flameout landings should be avoided to the extent that ejection is normally considered the best course of action. If it is decided to attempt a forced landing, the following considerations should be observed:

a. In the event of a flameout, attempt to complete all air start efforts before high key is reached so that full attention may be devoted to accomplishing a successful flameout landing.

b. If the circumstances of flameout have precluded conclusive air start attempts prior to high key, further air starts may be attempted, but primary attention should be devoted to proper execution of the flameout landing.

c. Do not attempt air starts after low key is reached.

NOTE

This does not preclude air start attempts when flameout occurs below low key.

d. These instructions in no way alter previously established requirements for ejection versus forced landing. (Refer to EJECTION VS. FLAMEOUT LANDING.)

FLAMEOUT LANDING (ALL GROSS WEIGHTS)

All landing emergencies involving landing on prepared or unprepared surfaces should be made with the landing gear extended. The extended gear, even on reasonably rough terrain, provides an absorption of the initial shock, resulting in less injury and damage to the airplane. The inherent nose-high landing attitude of this airplane will result in severe "slap" to the ground if the tail section is permitted to take the initial shock of the wheels-up landing. Therefore, to reduce vertebral injuries, all landings should be made with the gear extended.

WARNING

- If terrain is unknown or unsuitable for forced landing, eject.
- The canopy should be retained if a crash landing is to be made.
- All landings should be made with landing gear extended, even in the event of a damaged or missing tire or wheel, or when only partial gear extension is possible.
- During crash landings or ditching, the helmet visor should be placed in position over the face as the visor affords protection to the eyes and exposed areas.

See figure 3-6 for recommended procedures and techniques for a flameout landing on suitable terrain.

SIMULATED FLAMEOUT LANDING

See figure 3-6 for simulated flameout landing configuration.

EJECTION

The egress system provides safe ejection capability at all points in a normal or flameout landing pattern. It is essential that proper airspeed be maintained to provide the capability to "zoom" the airplane prior to an ejection in the landing pattern. Figure 3-3 shows ejection parameters during a flameout landing. Refer to EJECTION IN FLIGHT for ejection procedures.

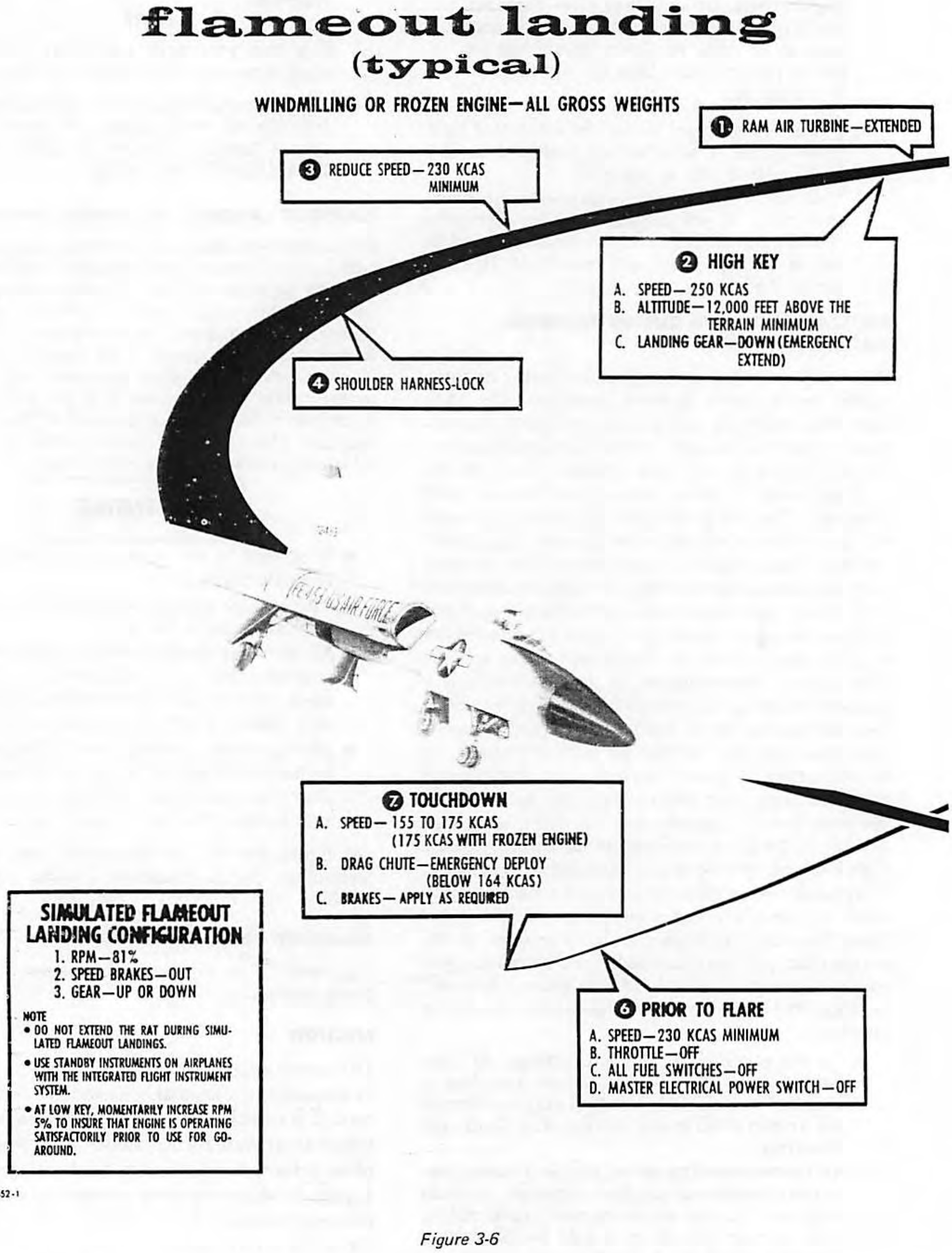

flameout landing
(typical)
WINDMILLING OR FROZEN ENGINE—ALL GROSS WEIGHTS
1 RAM AIR TURBINE—EXTENDED
3 REDUCE SPEED—230 KCAS MINIMUM
2 HIGH KEY
A. SPEED—250 KCAS
B. ALTITUDE—12,000 FEET ABOVE THE TERRAIN MINIMUM
C. LANDING GEAR—DOWN (EMERGENCY EXTEND)
4 SHOULDER HARNESS-LOCK
7 TOUCHDOWN
A. SPEED—155 TO 175 KCAS (175 KCAS WITH FROZEN ENGINE)
B. DRAG CHUTE—EMERGENCY DEPLOY (BELOW 164 KCAS)
C. BRAKES—APPLY AS REQUIRED
SIMULATED FLAMEOUT LANDING CONFIGURATION
1. RPM—81%
2. SPEED BRAKES—OUT
3. GEAR—UP OR DOWN
NOTE
• DO NOT EXTEND THE RAT DURING SIMULATED FLAMEOUT LANDINGS.
• USE STANDBY INSTRUMENTS ON AIRPLANES WITH THE INTEGRATED FLIGHT INSTRUMENT SYSTEM.
• AT LOW KEY, MOMENTARILY INCREASE RPM 5% TO INSURE THAT ENGINE IS OPERATING SATISFACTORILY PRIOR TO USE FOR GO-AROUND.
6 PRIOR TO FLARE
A. SPEED—230 KCAS MINIMUM
B. THROTTLE—OFF
C. ALL FUEL SWITCHES—OFF
D. MASTER ELECTRICAL POWER SWITCH—OFF
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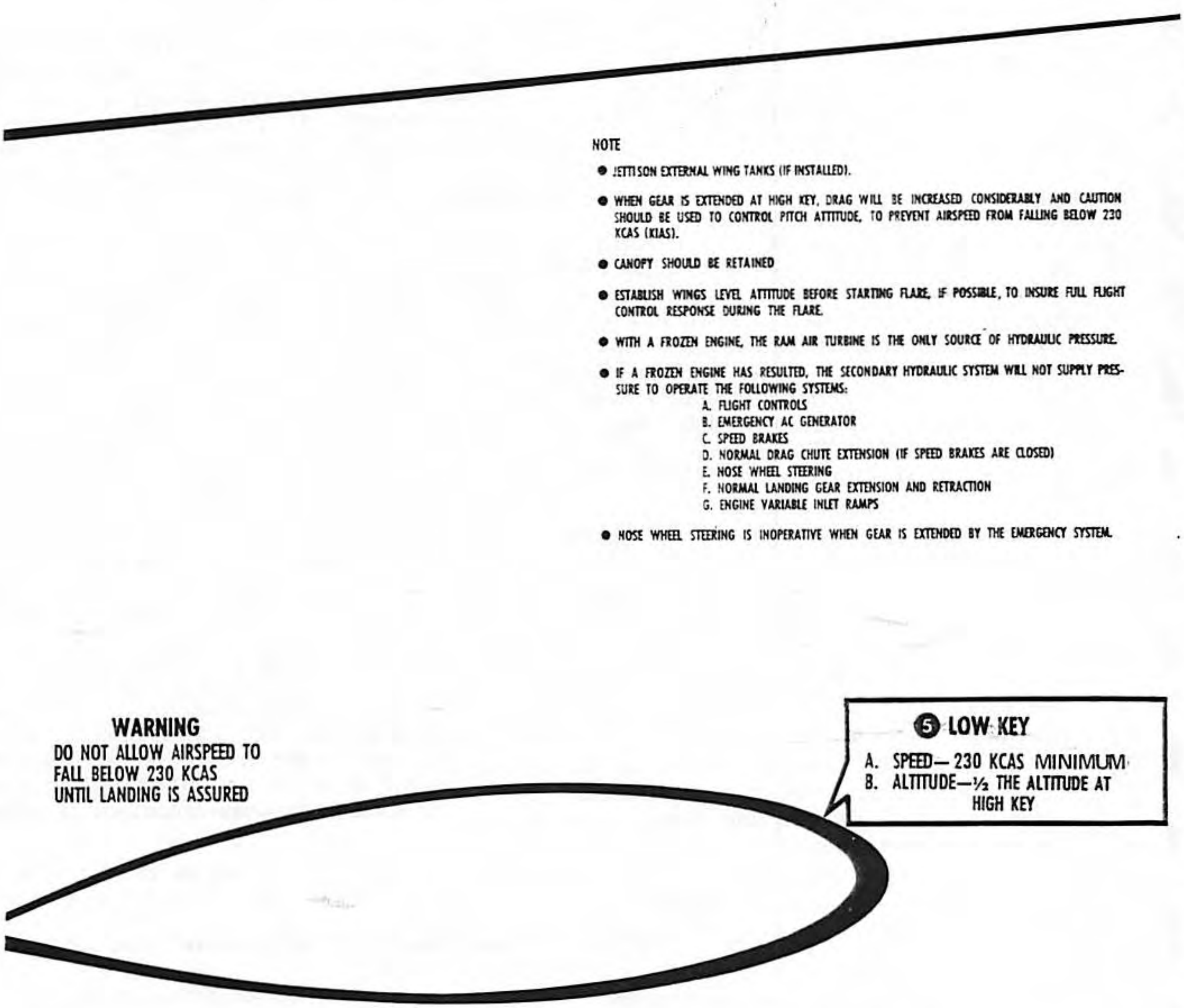
Figure 3-6



NOTE

- JETTISON EXTERNAL WING TANKS (IF INSTALLED).
- WHEN GEAR IS EXTENDED AT HIGH KEY, DRAG WILL BE INCREASED CONSIDERABLY AND CAUTION SHOULD BE USED TO CONTROL PITCH ATTITUDE, TO PREVENT AIRSPEED FROM FALLING BELOW 230 KCAS (KIAS).
- CANOPY SHOULD BE RETAINED
- ESTABLISH WINGS LEVEL ATTITUDE BEFORE STARTING FLARE, IF POSSIBLE, TO INSURE FULL FLIGHT CONTROL RESPONSE DURING THE FLARE.
- WITH A FROZEN ENGINE, THE RAM AIR TURBINE IS THE ONLY SOURCE OF HYDRAULIC PRESSURE.
- IF A FROZEN ENGINE HAS RESULTED, THE SECONDARY HYDRAULIC SYSTEM WILL NOT SUPPLY PRESSURE TO OPERATE THE FOLLOWING SYSTEMS:
  A. FLIGHT CONTROLS
  B. EMERGENCY AC GENERATOR
  C. SPEED BRAKES
  D. NORMAL DRAG CHUTE EXTENSION (IF SPEED BRAKES ARE CLOSED)
  E. NOSE WHEEL STEERING
  F. NORMAL LANDING GEAR EXTENSION AND RETRACTION
  G. ENGINE VARIABLE INLET RAMPS
- NOSE WHEEL STEERING IS INOPERATIVE WHEN GEAR IS EXTENDED BY THE EMERGENCY SYSTEM.

WARNING

DO NOT ALLOW AIRSPEED TO FALL BELOW 230 KCAS UNTIL LANDING IS ASSURED

5 LOW KEY

A. SPEED— 230 KCAS MINIMUM
B. ALTITUDE—½ THE ALTITUDE AT HIGH KEY

WARNING

- ALL LANDINGS SHOULD BE MADE WITH LANDING GEAR EXTENDED, EVEN IN THE EVENT OF A DAMAGED OR MISSING TIRE OR WHEEL, OR WHEN ONLY PARTIAL GEAR EXTENSION IS POSSIBLE.
- IF TERRAIN IS UNKNOWN OR CONDITIONS ARE UNSUITABLE FOR FORCED LANDING, EJECT. (SEE EJECTION VS FORCED LANDING THIS SECTION.)
- ON AIRPLANES WITH THE INTEGRATED FLIGHT INSTRUMENT SYSTEM, THE STANDBY INSTRUMENTS SHOULD BE USED DURING ALL SIMULATED FLAMEOUT APPROACHES TO ACCUSTOM PILOTS WITH THEIR USE PRIOR TO ACTUAL EMERGENCIES.
- WHILE PRACTICE FLAMEOUT APPROACHES IMPROVE PROFICIENCY, SUCCESSFUL COMPLETION OF SFO's WILL NOT ASSURE ACTUAL FLAMEOUT LANDING SUCCESS. DURING AN ACTUAL FLAMEOUT, IF CONDITIONS AT ANY TIME DO NOT APPEAR IDEAL—EJECT. TURN AIRCRAFT AWAY FROM POPULATED AREAS. REFER TO EJECTION VS FLAMEOUT LANDING, THIS SECTION.

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WARNING

- Do not delay decision to eject if airplane becomes uncontrollable at any point in the landing pattern.
- For Ⓑ airplanes, the forward seat is ejected one second later than the aft seat. This fact must be considered when planning a flameout landing.

DITCHING

In all cases where the capability of successfully ditching the airplane is available to the pilot, i.e., controllable airplane, the capability also exists for a successful ejection regardless of altitude. Therefore, eject rather than ditch the airplane as spinal injury is probable during water landing. The optimum circumstance for a low altitude ejection is to achieve a wings-level, nose-high attitude, then eject. Refer to EJECTION, this section. If ditching is unavoidable, proceed as follows:

1. External tanks—Jettison (if installed).
2. Make normal approach with speed brakes out.

   Plan approach parallel to swells unless wind is in excess of 25 knots, in which case a landing into the wind is recommended.
3. Shoulder harness inertia reel handle—MANUAL LOCK. (FP-RP)
4. Immediately before touchdown:

   If landing into the wind, touch down on falling side of the wave, if possible. Maintain nose-high attitude and touch down slowly with low rate of descent.

   a. Throttle—Off.
   b. Fuel shutoff switches — CLOSE.
   c. Master electrical power switch — OFF.
   d. Canopy—Jettison.
   e. Drag chute handle — Pull (emergency deploy).
5. Abandon the airplane as soon as forward motion stops.

Upon contact with the water the airplane will usually bounce. During this time, the sensation of being stopped may be experienced. Do not attempt to exit from the seat until water spray is noticeable, deceleration forces have stopped, and water begins to enter the cockpit. Exit from the seat by first operating the ditching control handle and opening the safety belt. Retain the survival kit when leaving the airplane.

DRAG CHUTE FAILURE

Landing distances for a dry runway without a drag chute as shown in the Appendix can be obtained by using the minimum run landing techniques described in Section II. For a dry or wet runway (RCR 12 or above), maintain the nose high touchdown attitude utilizing maximum aerodynamic braking until 115 KCAS. Then lower the nose and apply optimum wheel braking. Below 115 KCAS wheel braking is more effective than aerodynamic braking and the danger of skidding the tires is reduced. Drag chute failure on a slippery runway (RCR below 12) presents additional problems over which the pilot has little control. The amount of water, snow or ice on the runway, type and condition of runway, and tire tread depth induce many variables in stopping distance. The following procedure will provide the shortest landing roll under the most adverse conditions.

a. Maintain the nose high touchdown attitude after touchdown. As the aircraft decelerates raise the nose as high as possible without scraping the tail. Hold this attitude as the aircraft decelerates and maintain directional control with rudder and aileron.

b. As the speed reaches approximately 80 KCAS for Ⓐ and 100 KCAS for Ⓑ aircraft, the nose wheel will lower to the runway while holding full aft stick. Do not release back stick pressure until the nose wheel contacts the runway.

c. With the nose wheel on the runway, begin optimum braking.

LANDING WITH A SPEED BRAKE FAILURE.

In the event of a speed brake malfunction or suspected speed brake structural failure, a no speed brake, no drag chute landing should be planned. Past experience has shown that with a speed brake failure, the drag chute probably will not be available. Runway conditions and length, plus availability of a barrier should be considered. If use of the drag chute is required, the nose wheel should be lowered to the runway prior to attempting deployment. Pilots should anticipate the yaw that will be induced by a single speed brake extension when the drag chute is deployed and should be prepared to counter this yaw by a combination of flight controls and nose wheel steering, if available.

auxiliary equipment

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AIR-CONDITIONING AND PRESSURIZATION SYSTEM

The air-conditioning and pressurization system (figure 4-1) uses hot bleed air from the engine high-pressure compressor. The hot air is first cooled in a heat exchanger and then refrigerated in a refrigeration turbine. To control cockpit temperature, hot air bypassed around the heat exchanger and the refrigeration turbine is mixed with cold air from the refrigeration turbine and discharged into the cockpit through outlets located under the canopy sills behind the ejection seat, on the floor outboard of the rudder pedals, and through diffusers (one on each side of the instrument panel). Normally, air is discharged into the cabin through the vertical outlets located behind the ejection seat. The horizontal outlets under the canopy sill are designed for use when flight conditions make the cockpit extremely hot and better circulation of air around the body is desired. A modulating temperature control valve, controlled either by a thermostat which measures cockpit air temperature or by a thermostat which measures cockpit inlet duct temperature, mixes the proper amounts of hot and cold air to maintain the selected inlet duct or cockpit air temperature. Cockpit pressurization (figure 4-2) is provided by regulating the rate of discharge of the conditioned air from the cockpit. An airflow control valve maintains airflow into the cockpit at a constant rate, and a pressure regulator valve controls cockpit air discharge to maintain an unpressurized cockpit up to an altitude of 12,500 feet, a cockpit altitude of 12,500 feet up to 31,000 feet, and a cockpit pressure five psi above ambient at altitudes above 31,000 feet. An emergency cabin pressurization system uses high-pressure pneumatic supply system air for cabin pressurization and includes a cabin pressure-low warning light. If the cabin air selector switch is in PRESS or OFF position when the cabin pressure altitude rises above approximately 44,000 (±2000) feet, a pressure switch in the cockpit illuminates the cabin pressure-low warning light and electrically (dc essential bus power) opens a valve which admits air from the pneumatic supply system to the canopy seal and to the cabin for cabin pressurization. The pressure switch cuts off electric power, extinguishing the cabin pressure-low warning light and closing the valve to shut off pneumatic system pressure when cabin pressure altitude is decreased to approximately 26,000 (±2000) feet. Pneumatic supply system air for emergency cabin pressurization is taken from the two main air supply bottles. If desired, the cockpit can be ventilated by ram air;

air conditioning and

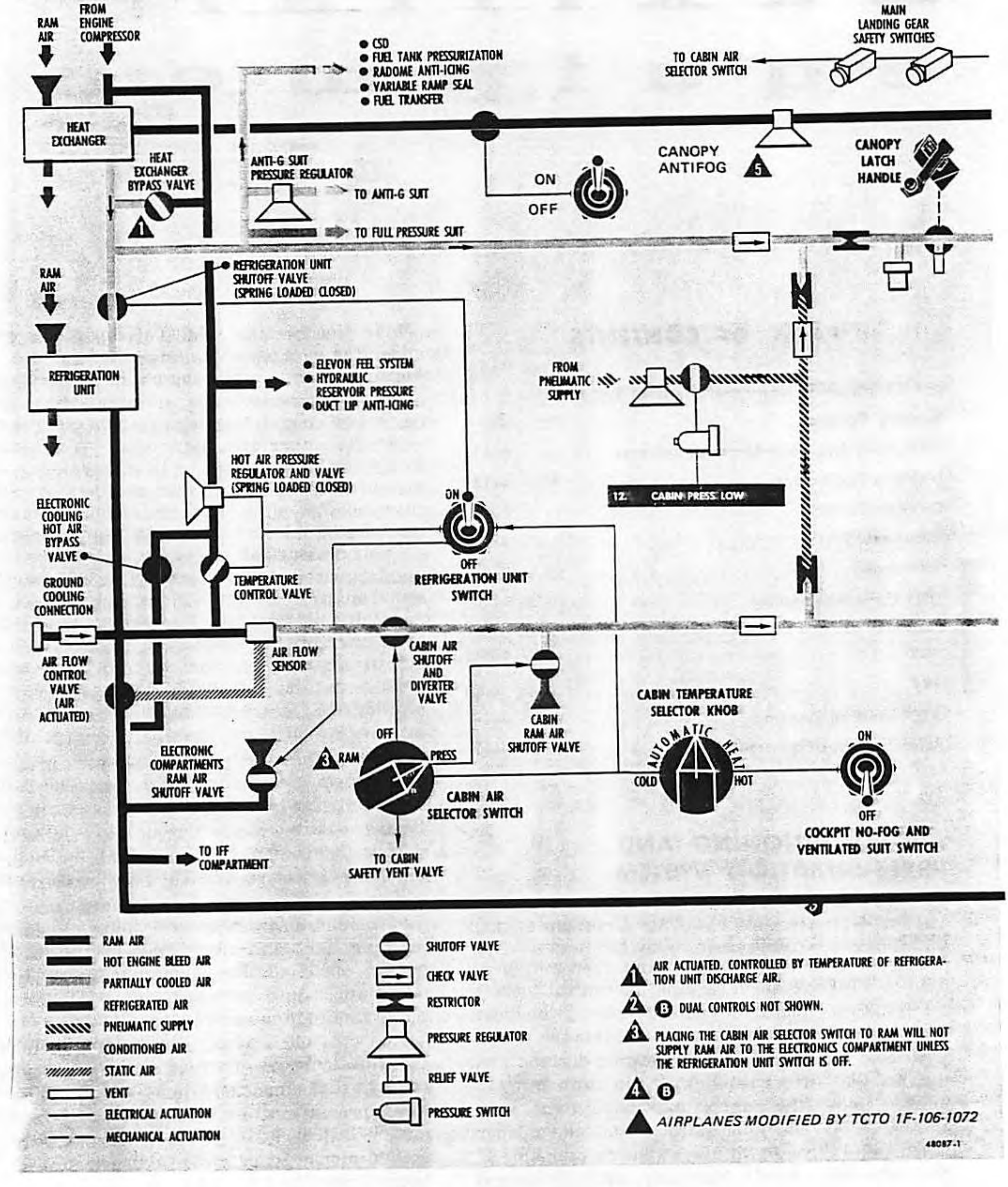
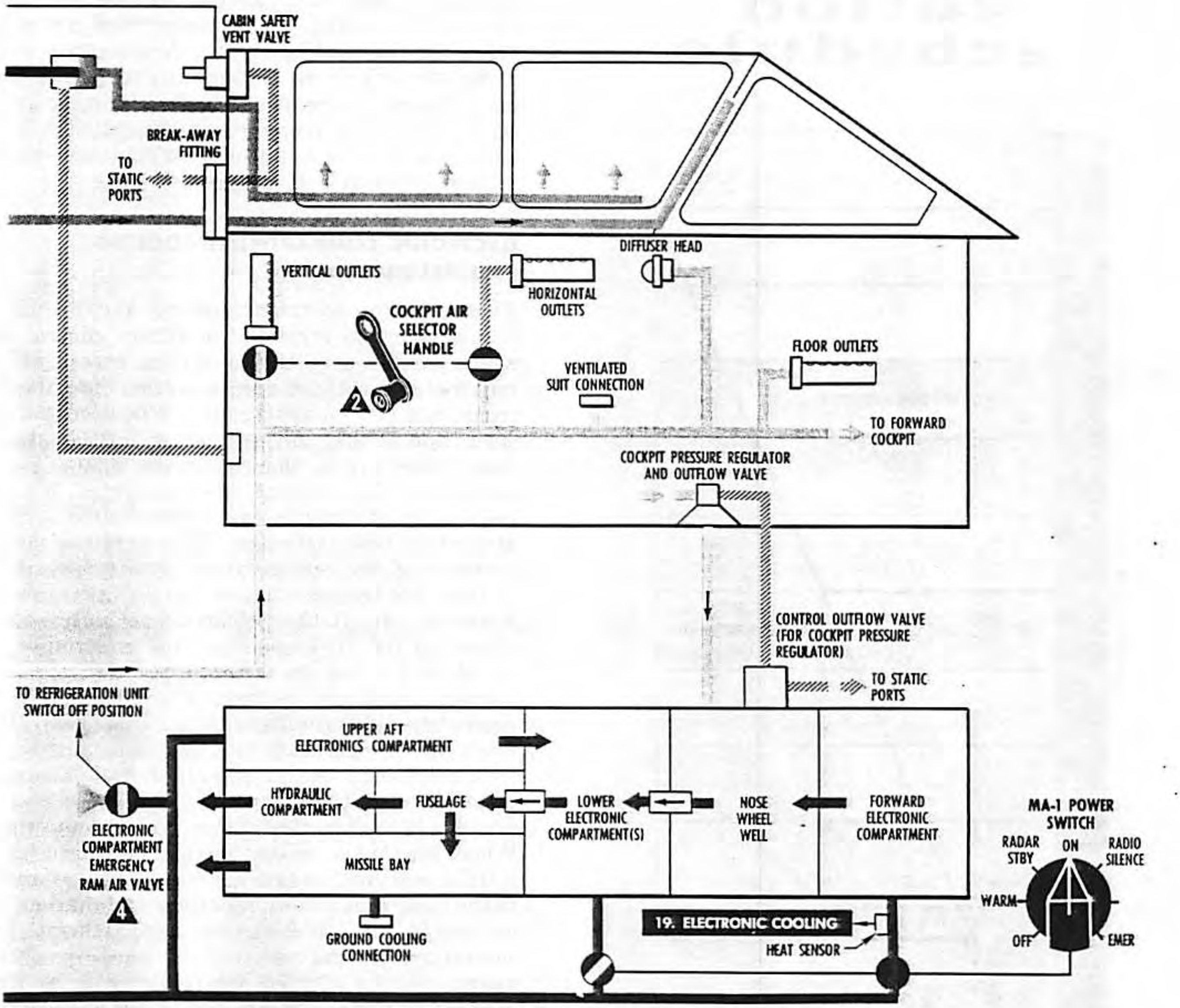


Figure 4-1

pressurization system



		PRESS		OFF		RAM	
SWITCH POSITION	CABIN AIR SELECTOR KNOB	PRESS		OFF		RAM	
	REFRIGERATION UNIT SWITCH	ON	OFF	ON	OFF	ON	OFF
VALVE POSITION	REFRIGERATION UNIT	OPEN	CLOSED	OPEN	CLOSED	OPEN	CLOSED
	COCKPIT RAM AIR SHUTOFF	CLOSED	CLOSED	CLOSED	CLOSED	OPEN	OPEN
	HOT AIR PRESSURE REGULATOR	REGULATING	CLOSED	CLOSED	CLOSED	CLOSED	CLOSED
	TEMPERATURE CONTROL	OPERATING	CLOSED	CLOSED	CLOSED	CLOSED	CLOSED
	CABIN SAFETY VENT	CLOSED	CLOSED	CLOSED	CLOSED	OPEN	OPEN
	AIRFLOW SENSOR AND CONTROL	OPEN	OPEN	CLOSED	CLOSED	CLOSED	CLOSED
	ELECTRONIC COMPT. RAM AIR SHUTOFF	CLOSED	OPEN	CLOSED	OPEN	CLOSED	OPEN
	COCKPIT AIR SHUTOFF AND DIVERTER VALVE	OPEN	CLOSED	CLOSED	CLOSED	CLOSED	CLOSED
	ELECTRONIC COMPT. EMER. RAM AIR VALVE	CLOSED	OPEN	CLOSED	OPEN	CLOSED	OPEN

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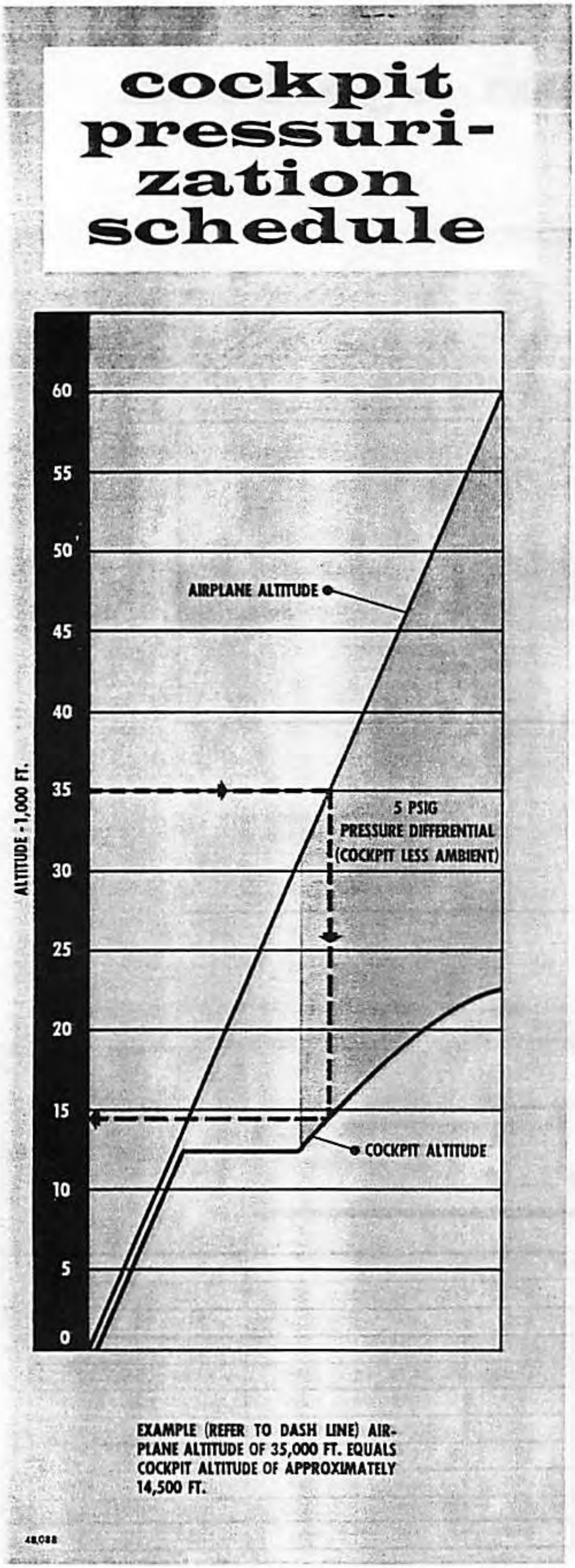


Figure 4-2

however, pressurization and temperature control are not operative when ram air ventilation is used. Partially conditioned air which has been cooled in the air-conditioning system heat exchanger is always available for use in the following systems: fuel transfer, fuel tank pressurization, canopy seal, variable ramp seal, radome anti-ice and anti-g suit pressurization. Fully conditioned air is provided for the ventilated suit. A connector located under the left wing is provided to permit using an auxiliary source of refrigerated air for ground cooling of the electronic compartment, missile bay, and cockpit. For additional information on this system, refer to T.O. 1F-106A-2-6.

ELECTRONIC COMPARTMENT COOLING AND PRESSURIZATION

Electronic compartment cooling and pressurization is provided through the airflow control valve which routes conditioned air in excess of that required for cockpit pressurization into the electronic and IFF compartments. When cockpit pressurization is shut off, a diverter valve routes the conditioned air to the electronics and IFF compartments. To prevent overcooling of the electronics compartments, engine bleed air is bypassed around the heat exchanger. This increases the temperature of the refrigeration unit intake air and, in turn, the temperature of the refrigeration unit discharge air. If the refrigeration unit switch is placed in the OFF position, the electronics ram air shutoff valve opens to supply ram air to the forward and aft electronics compartments for emergency electronics cooling. On B airplanes when the refrigeration unit switch is placed in the OFF position, a second ram air shutoff valve opens to supply additional ram air for cooling of the upper aft electronics compartment equipment. When the MA-1 power switch is placed in the EMER position, cooling air will be routed directly to the electronic communications installation area, located in the aft electronic compartment. Pressurization of the electronics compartments is maintained by a fixed restrictor through which electronics compartment cooling air is discharged into the fuselage from where it flows into the missile bay and into the hydraulic compartment where it is dumped overboard. With the nose wheel well door open, air from the forward electronics compartment dumps overboard through the nose wheel well. A check valve prevents the flow of air from the aft electronics compartment into the nose wheel well so that with the nose wheel well door open, aft electronics compartment air flows through the fixed restrictor valve as it does during flight. For electronic compartment ground cooling, an auxiliary source of refrigerated air may be utilized through a connector located under the left wing.

NOTE

If radar pressurization fails at high altitude, a check valve will maintain pressurization in the UHF radio transmitter, permitting continued communications for a limited time until pressure decreases due to leakage.

CAUTION

For electronic equipment ground cooling limitations, refer to COOLING LIMITATIONS (GROUND OPERATIONS), Section V.

REFRIGERATION UNIT SWITCH

The refrigeration unit switch (figure 4-3) is located on the right console. On B airplanes the refrigeration unit switch is located in the forward cockpit only. The switch is placarded "Refr Unit," and has two positions, ON and OFF. With the switch in the ON position, the refrigeration unit shutoff valve is open to route engine high-pressure compressor bleed air through the refrigeration turbine and the electronics ram air shutoff valve is closed. The air-conditioning and pressurization system is under control of the cabin air selector switch. With the refrigeration unit switch in the OFF position, the electronics ram air shutoff valve is open and the refrigeration unit shutoff valve, the cabin temperature control valve and the hot-air regulator are closed. On B airplanes when the refrigeration unit switch is placed in the OFF position, a second ram air shutoff valve opens to supply additional ram air for cooling of the upper aft electronics compartment equipment. Engine high-pressure compressor bleed air cannot flow into the refrigeration turbine nor through the refrigeration unit bypass; therefore, the cockpit, electronics compartments, and IFF compartment cannot be pressurized and temperature control is not possible.

NOTE

- The refrigeration unit switch contains a locking feature which prevents the switch from being inadvertently moved to the OFF position. It is necessary to pull out on the switch before it can be moved to OFF.
- With the refrigeration unit switch in the OFF position, cockpit and electronic compartment pressurization and temperature control is not possible; however, ram air is supplied to the electronics compartment for emergency electronics cooling, and air which has been partially cooled in the heat exchanger is still available for the following functions: fuel transfer, fuel tank pressurization, canopy seal, variable ramp seal, radome anti-ice, and anti-g suit pressurization. On B airplanes a second ram air shutoff valve opens to supply additional ram air for cooling of the upper aft electronics compartment equipment.
- For ground cooling, with the ground cart, the refrigeration unit switch must be in ON position. This will close the emergency electronic cooling valve, and allow normal cooling.

CAUTION

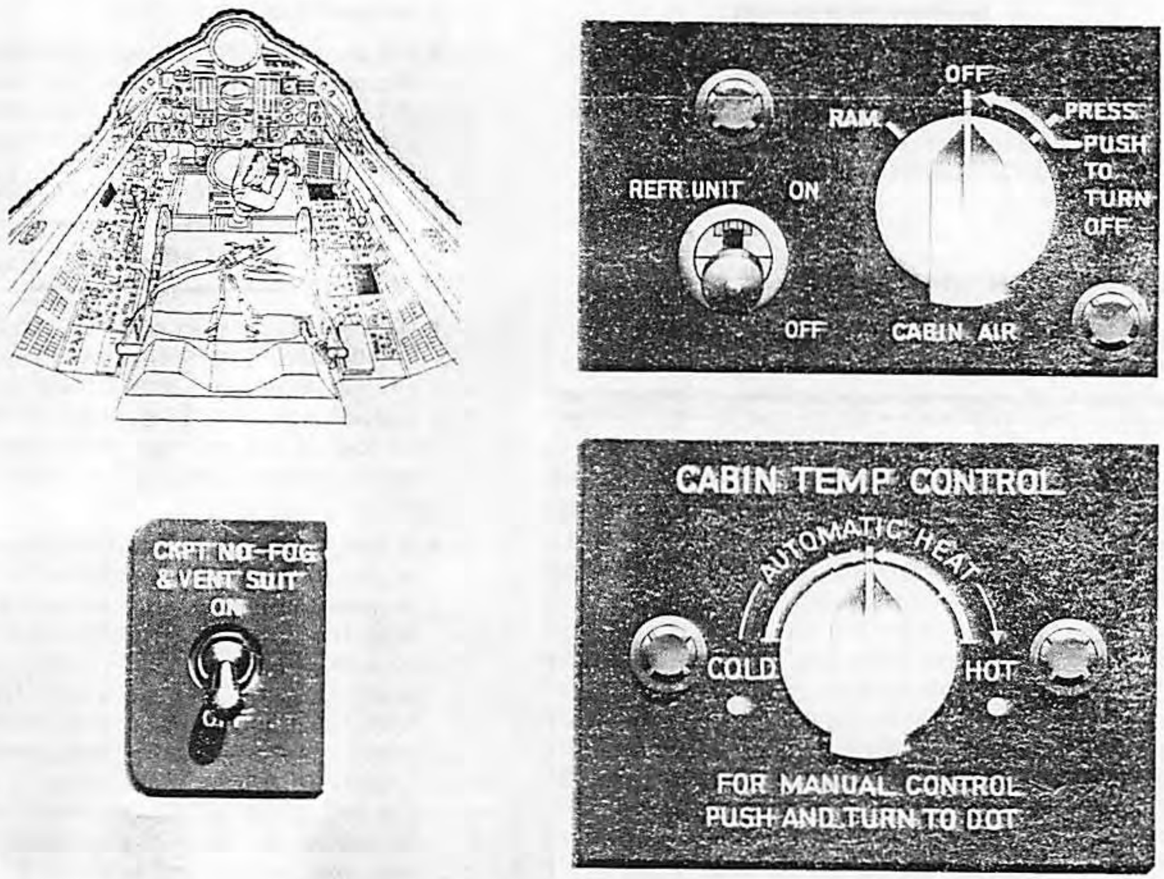
- If the refrigeration unit switch is placed in the OFF position, the MA-1 power switch must be immediately placed in EMER position. To prevent overheating of the MA-1 system electronics equipment, airspeed should be reduced to 250 KCAS.
- If the refrigeration unit switch is placed in the OFF position and cabin altitude is above approximately 44,000 (±2000) feet, the emergency cabin pressurization system will be actuated if the cabin air selector switch is not in the RAM position. Continued operation with the emergency pressurization system actuated will depiete the pneumatic power supply for the following: Constant-speed drive air-oil cooler shutoff valve, rudder feel system, and armament system.

The refrigeration unit switch receives power from the dc essential bus.

CABIN AIR SELECTOR SWITCH

The cabin air selector switch (figure 4-3) is located on the right console. On B airplanes the cabin air selector switch is located in the forward cockpit only. The switch is placarded "Cabin Air" and has three positions, RAM, OFF, and PRESS. With the switch in the OFF position, the diverter valve is open to the electronics compartment while the ram air shutoff valves, the cabin safety vent valve, the cabin temperature control valve and the hot-air pressure regulator are closed. If the refrigeration unit switch is in the ON position, refrigerated air is supplied to the electronics and IFF compartments. Moving the cabin air selector switch to the PRESS position opens the diverter valve to the cabin, activates the hot-air pressure regulator, and permits automatic temperature control as selected

air conditioning and pressurization controls



AUTOMATIC CONTROL:

- TURN KNOB TO DESIRED POSITION IN AUTOMATIC HEAT RANGE.

MANUAL CONTROL:

- DEPRESS KNOB AND TURN THROUGH EITHER DOT TO NEUTRAL (DOWN) POSITION.
- INCREASE OR DECREASE TEMPERATURE BY TURNING KNOB TOWARD APPROPRIATE DOT UNTIL THE STOP IS FELT.

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Figure 4-3

by the cabin temperature control knob. Placing the cabin air selector switch in the RAM position opens the diverter valve to the electronics and IFF compartments, closes the hot-air pressure regulator, opens the cabin safety vent valve, and opens the ram air shutoff valve to route ram air to the cockpit. Whenever the cabin air selector switch is in the OFF or PRESS position and cabin altitude is above approximately 44,000 ($\pm$2000) feet, the emergency cabin pressurization system will be actuated. The cabin air selector switch receives power from the dc essential bus.

CABIN TEMPERATURE CONTROL KNOB

The cabin temperature control knob (figure 4-3), located on the right-hand console, provides for automatic control of the cockpit temperature or for manual control of the system in the event of malfunction of the automatic control. On Ⓑ airplanes the cabin temperature control knob is located in the forward cockpit only. The knob can be positioned through an AUTOMATIC HEAT range having a COLD position at one extreme and a HOT position at the other. With the knob set within this range, temperature is automatically controlled by modulating the temperature control valve to mix hot bleed air with refrigerated air in the proportions required to maintain the selected cabin temperature. To override the automatic system, the knob is depressed and is moved past either the HOT or COLD extreme of the automatic range into a neutral zone opposite the AUTOMATIC HEAT band. When the knob is in the neutral zone, automatic temperature control is not provided and the temperature control valve is no longer modulated but remains as it was when the knob was moved into the neutral zone. After the knob is in the neutral zone, temperature may be increased or decreased by turning the knob toward the dot below either the HOT or COLD position until the stop is felt. Temperature will then be increased (HOT position) or decreased (COLD position) as long as the knob is held at the dot. When the knob is released, it will return to the neutral zone.

CAUTION

It is possible to force the knob beyond the stop. If this is done, the knob will not return to the neutral position when released, and erratic operation of the temperature control valve will result.

To return to automatic control, the knob is depressed and moved past either the HOT or COLD position and into the AUTOMATIC HEAT range. Power is supplied from the dc essential bus and the ac essential bus.

CABIN-AIR SELECTOR HANDLE

The cabin-air selector handle (34, figure 1-10) is located at the rear of the cockpit to the left of the seat. The handle mechanically operates a butterfly valve which directs discharge air either to the horizontal canopy sill outlets, to the vertical outlets behind the seat, or to both outlets simultaneously. Some airplanes have placards adjacent to the handle which read "Vertical Piccolo-Handle Down" and "Horizontal Piccolo-Handle Up." Other airplanes have placards reading "Vertical Piccolo-Handle Up" and "Horizontal Piccolo-Handle Down." The handle may be positioned at an intermediate point to simultaneously direct air to the canopy sill outlets and the outlets behind the seat. The flow of conditioned air to the ventilated suit through the diffusers, and through the discharge tubes near the rudder pedals, is not affected by the cabin air selector handle. Normal cabin air inflow should be through the vertical outlets behind the seat, which are designed for low velocity inflow. The horizontal canopy sill outlets direct the flow at higher velocity around the pilot's body for increased circulation at high temperature conditions.

CABIN-AIR DIFFUSER HEADS

On airplanes with the conventional instrument display the cabin-air diffuser heads in the air-conditioning outlets (18, figure 1-10, and 1, figure 1-11) are located on either side of the instrument panel. The heads can be adjusted manually to direct cooling air directly on the pilot, or can be completely shut off.

CABIN PRESSURE ALTITUDE GAGE

On airplanes with the conventional instrument display the cabin (cockpit) pressure altitude gage (7, figure 1-8) is located on the instrument panel. The gage indicates cockpit pressure altitude in feet above sea level. On airplanes with the integrated instrument system, cabin pressure altitude information is presented on the altitude-vertical velocity indicator. Refer to INSTRUMENTS, Section I.

CABIN PRESSURE-LOW WARNING LIGHT

The cabin pressure-low warning light (12, figure 1-30) is located on the master warning light panel. The light illuminates and displays "CABIN PRESS LOW" when cabin pressure altitude rises to 44,000 (±2000) feet. The light extinguishes when cabin pressure altitude drops to 26,000 (±2000) feet. Illumination of the light also indicates that the emergency cabin pressurization system is in operation supplying pneumatic pressure to the cabin. When the light is extinguished it indicates that the emergency cabin pressurization system has been shut off. If the cabin air selector switch has been placed in the RAM position, the warning lights will not illuminate and the emergency cabin pressurization system will not be actuated. The light receives power from the dc essential bus and is tested by depressing the master warning lights test button.

ELECTRONIC COOLING WARNING LIGHT

On some airplanes an electronic cooling warning light (19, figure 1-30) located on the master warning panel will illuminate to display "ELECTRONIC COOLING" when the cooling air directed to the electronic compartments is insufficient for sustained equipment operation. A heat sensor within the cooling air duct to the electronic compartments actuates the light. Refer to NORMAL OPERATION OF ELECTRONIC COMPARTMENT COOLING, this Section. The light receives power from the MA-1 electrical power supply system.

NOTE

When the MA-1 power switch is placed in EMER the electronic cooling warning light will be inoperative.

NORMAL OPERATION OF COCKPIT AIR-CONDITIONING AND PRESSURIZATION SYSTEM

1. Refrigeration unit switch — ON.

NOTE

With the refrigeration unit switch in the OFF position, air-conditioning and pressurization is inoperative with the exception of electronics cooling by ram air, and emergency cabin pressurization; however, air which has been cooled in the heat exchanger is available for fuel transfer, fuel tank pressurization, canopy seal, radome anti-ice and anti-g suit pressurization.

2. Cabin air selector switch — PRESS.
3. Cockpit no-fog and ventilated suit switch — As desired.

WARNING

During high humidity conditions, the cooling effect produced by the refrigeration turbine of the air-conditioning system can create heavy condensation in the cockpit inlet air. This condensation forms a very dense fog within the cockpit. Under extreme conditions cockpit fog can reduce visibility to a point where it is impossible to see the cockpit instruments, as well as completely obscuring outside visibility. Cockpit fog is most likely to occur on takeoff, but may also occur during landing or go-around. To prevent fog formation in the cockpit under most climatic conditions (dew point of less than 80°F), the cockpit no-fog and ventilated suit switch should be placed in the ON position prior to takeoff and landing. Effective prevention of cockpit fog with ambient air dew point in excess of 80°F can be accomplished by placing the cockpit no-fog ventilated suit switch OFF and selecting a high cockpit temperature with the cabin temperature control knob.

4. Cabin temperature control knob — As desired. Adjust temperature as desired by positioning the temperature control knob within the automatic range.

NOTE

The cabin temperature control knob is bypassed and cannot be used to select cockpit temperature when the cockpit no-fog and ventilated suit switch is in the ON position.

5. Cabin air selector handle — Adjust for vertical outlet.
6. Cabin air diffuser heads — As desired.

EMERGENCY OPERATION OF COCKPIT AIR-CONDITIONING AND PRESSURIZATION SYSTEM

WARNING

If wearing an MBU-3/P or MBU-5/P oxygen mask and cabin pressurization is lost, immediate descent to 25,000 feet or below is mandatory.

Loss of Cabin Pressure

If cabin pressure loss occurs:

1. Refrigeration unit switch — ON.
2. Cabin air selector switch — PRESS.

NOTE

The cabin pressure-low warning light will illuminate if the cabin pressure altitude reaches 44,000 (±2000) feet or greater, and emergency pressurization will commence automatically.

3. If cabin altitude is higher than actual altitude, place the cabin air selector switch to OFF and reduce speed.
4. Turn MA-1 power switch to EMER and descend to 24,000 feet.

If cockpit becomes contaminated or depressurization occurs or becomes necessary:

1. Oxygen mask connections—Check. (FP-RP)
2. Cabin air selector switch — RAM.

Excessive Cockpit Temperatures

1. Cabin temperature control knob — Push in for manual operation and position as desired.
2. Face mask — Check for overheat.
3. G suit — Check for excessive hot air flow.
4. If cockpit temperature cannot be decreased manually, move cabin air selector switch to OFF.
5. Canopy antifog switch(es) — OFF.
6. If cockpit temperature remains excessive, move cabin air selector switch to RAM.

Refrigeration Unit Failure

If refrigeration turbine fails (failure of the refrigeration turbine will be evidenced by a rapid increase in temperature and a decrease in pressure), use the following procedures to provide emergency cooling of the MA-1 system electronics equipment:

1. Refrigeration unit switch — OFF.

2. MA-1 power switch — EMER.

CAUTION

When the refrigeration unit switch is placed in the OFF position, the MA-1 power switch must be immediately placed in the EMER position.

3. Airspeed—250 KCAS.

CAUTION

To prevent overheating of the MA-1 system electronics equipment, airspeed should be reduced to 250 KCAS.

4. Land as soon as practicable.
5. MA-1 power switch—OFF (after landing).

NORMAL OPERATION OF ELECTRONIC COOLING SYSTEM

An electronic cooling warning light is located on the master warning light panel. The warning light system measures both cooling air temperature and airflow to the forward electronics compartment. It is possible for the light to illuminate under the following conditions:

GROUND OPERATION

a. Taxiing at idle rpm on warm days.

FLIGHT OPERATION

a. Reduction of engine power.
b. During letdown and approach.
c. Leveling out after a high thrust climb.
d. Zoom maneuvers at high altitude.
e. Subsonic flight above 49,500 feet.

As with all electronic equipment, excessive heating is detrimental. If the electronic cooling warning light illuminates, proceed as follows:

Electronic Cooling Warning Light Illuminated: "ELECTRONIC COOLING"

GROUND OPERATION

1. Cabin air selector switch—OFF and/or increase engine rpm to extinguish the light.

CAUTION

If the electronic cooling warning light cannot be extinguished through the above procedures, the MA-1 power switch must be placed in the OFF or EMER position to avoid damage to the electronic equipment. Return to line and determine cause of malfunction.

FLIGHT OPERATION

1. Throttle—Advance if possible.
2. Flight conditions—Change if possible.
3. Cabin temperature — Increase.
4. Cabin air selector switch—OFF or RAM.

CAUTION

If the above actions fail to extinguish the warning light, follow procedures under REFRIGERATION UNIT FAILURE, this Section.

ANTIFOG SYSTEMS

Electrical current and hot engine bleed air are used for defogging various parts of the airplane. For additional information on this system, refer to T.O. 1F-106A-2-6.

CANOPY ANTIFOG SYSTEM*

Canopy defogging is accomplished by electrically heating a conductive coating embedded in each side panel. The system is identical to the windshield anti-ice system discussed under ANTI-ICING AND RAIN REMOVAL SYSTEMS, this Section, except that the supply voltage is lower and power would not be interrupted during armament firing. Panel heating is controlled by the canopy antifog switch(es) and temperature sensing elements which are embedded in each panel. The elements automatically control the electrical power supply, thus preventing overheating of the panels.

WARNING

Moisture within the canopy panels may cause arcing. If this occurs turn the canopy antifog switch(es) OFF to prevent possible vertigo. If distortion of canopy panels is noted, turn the canopy antifog switch(es) OFF.

CANOPY ANTIFOG SWITCH* Ⓐ

The canopy antifog switch (figure 4-4) is located on the right console. The switch is placarded "Canopy Anti-Fog" and has two positions, ON and OFF. With the switch in the ON position, power is supplied to the canopy antifogging system. The switch receives power from the ac nonessential bus.

CANOPY ANTIFOG SWITCHES Ⓑ

The canopy antifog switches (figure 4-4) are located on the forward right console. The switches are placarded "Canopy Anti-Fog — Fwd, Aft" and control the respective canopy panels. Each switch has two positions, ON and OFF. With

**Prior to TCTO 1F-106-1072*

antifog and anti-icing control panels

(typical)

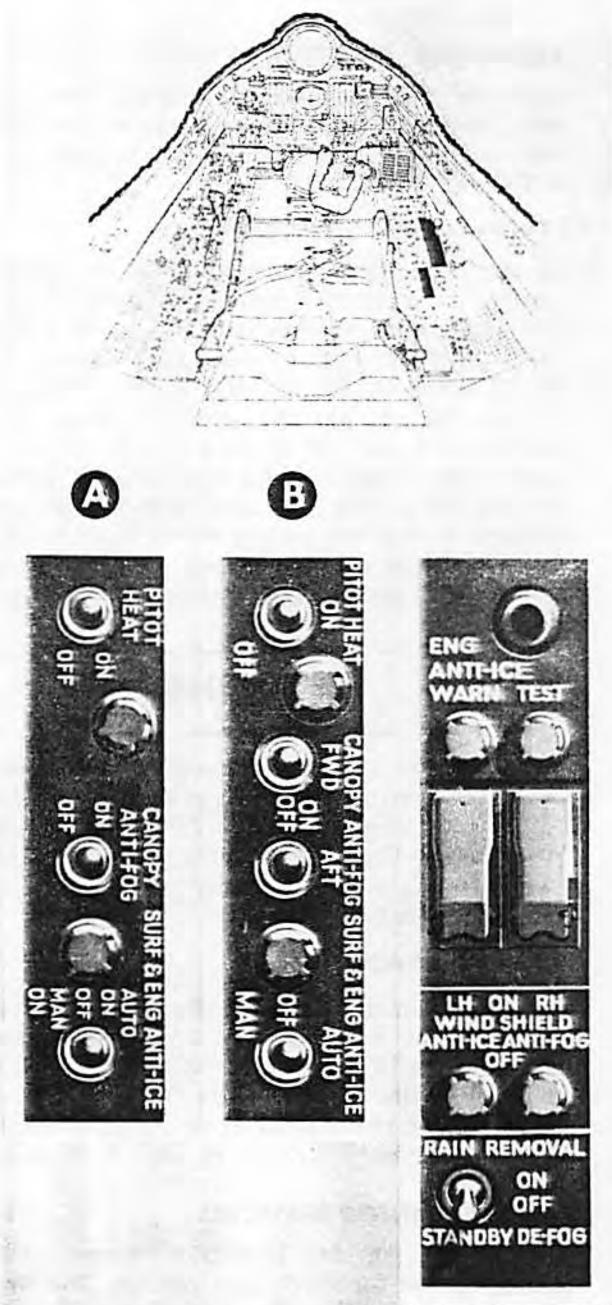

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Figure 4-4

either switch in the ON position, power is supplied to the selected (forward or aft) canopy antifog system. The switches receive power from the ac nonessential bus.

HOT AIR ANTIFOG SYSTEM*

The canopy hot air antifog system uses a mixture of partially conditioned engine bleed air from the heat exchanger and cockpit air to defog the canopy panels. The hot air is distributed along the canopy panels by ducts attached to the canopy frame. (See figure 4-1.) The system is powered by the 28 volt dc essential bus and is controlled by the canopy anti-fog switch.

CANOPY ANTIFOG SWITCH*

When the switch is placed to ON, an electrically operated valve opens and air from the heat exchanger enters the defog ducting to the pressure regulator and defog jet pump. The jet pump, creating a low pressure area, draws air from the cockpit (behind the ejection seat) into the defog ducting. This mixture of air flows up into the manifold duct and forward along each side of the canopy still where it is discharged through corrugated outlets onto each canopy panel. There is no provision for regulating the temperature by the pilot.

WINDSHIELD ANTI-ICING, ANTIFOG SWITCHES

Refer to ANTI-ICING AND RAIN REMOVAL SYSTEMS, this Section.

COCKPIT NO-FOG AND VENTILATED SUIT SWITCH

The cockpit no-fog and ventilated suit switch (figure 4-3) is located above the left console. On B airplanes the cockpit no-fog and ventilated suit switch is located in the forward cockpit only. The switch is placarded "Ckpt No-Fog & Vent Suit" and has ON and OFF positions. Selecting the OFF position places cockpit temperature under control of the cabin temperature control knob. Placing the switch in the ON position transfers temperature control to the duct thermostats which maintain a constant discharge temperature of 80°F for the ventilated suit or to prevent cockpit fog which could occur during landing and takeoff when humidity is high. The system is effective in the prevention of cockpit fog under most climatic conditions (dew point of less than 80°F). Under extreme climatic conditions of dew point in excess of 80°F effective prevention of cockpit fog can be accomplished by placing the cockpit no-fog ventilated suit switch OFF and selecting a high cockpit temperature with the temperature control knob. The cockpit no-fog and ventilated suit switch controls power from the ac essential bus.

**Airplanes modified by TCTO 1F-106-1072*

ANTI-ICING AND RAIN REMOVAL SYSTEMS

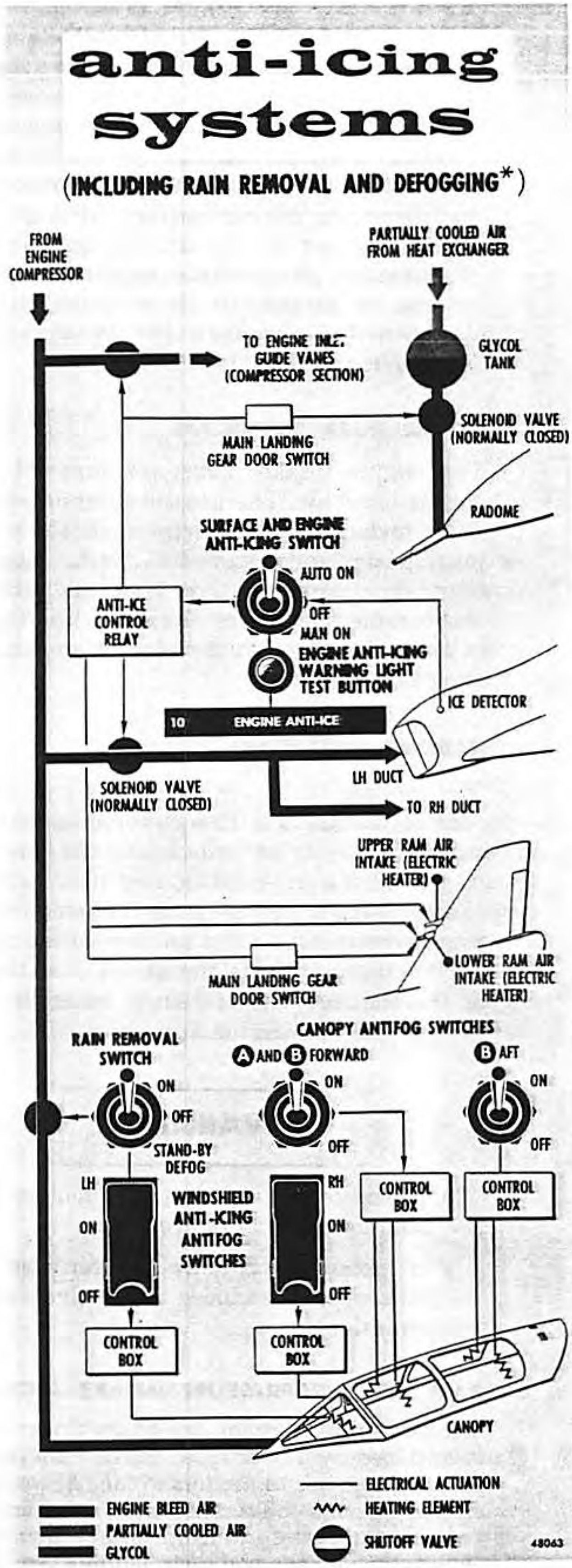
Hot engine bleed air, anti-icing fluid, and electrical current are used to anti-ice different parts of the airplane. The engine and engine inlet duct lips, anti-icing systems and the windshield rain-removal system use unconditioned (hot) engine bleed air. The radome anti-icing system uses anti-icing fluid which is stored in a tank pressurized with partially conditioned bleed air from the heat exchanger. Electrical current is used to anti-ice the windshield, the two artificial feel system intakes on the leading edge of the fin, the pitot-static tube on the nose boom, and the two variable ramp pitot-static tubes in the engine air inlet ducts. Wing and fin anti-icing is not provided. For additional information on this system, refer to T.O. 1F-106A-2-6.

SURFACE AND ENGINE ANTI-ICING SYSTEM

The surface and engine anti-icing system is an automatic anti-icing detection and control system which controls four individual anti-icing systems: the engine, the engine air intake ducts, the radome, and the artificial feel system air intake anti-icing systems. An ice detector probe is mounted in the engine intake duct to sense icing conditions. If the automatic anti-icing system is armed, when the ice detector probe detects icing it automatically turns on the automatic anti-icing system and the probe immediately starts to de-ice. The probe normally de-ices in less than 17 seconds; however, a time delay relay holds the automatic anti-icing system on for one minute after the probe is de-iced. Though the probe may make several "ice — de-ice" cycles at intervals of less than one minute, the anti-icing system will remain on until there is no further indication of ice for a period of one minute after the probe is de-iced. The system will then automatically shut down until the next time the ice detector probe senses ice. If the probe becomes clogged, the probe heater fails, or for any reason the probe does not de-ice within 15 to 20 seconds, the pilot is informed through an indicator light on the master warning panel. The system may then be manually operated. Depressing the engine ignition button heats the ice detector probe to remove any ice which may have accumulated either in flight or on the ground.

ENGINE ANTI-ICING

The engine inlet guide vanes at the face of the compressor and the engine nose fairing are anti-iced by engine bleed air. The thermostatically controlled air flows through the hollow guide vanes into the engine nose fairing and is exhausted at the aft edge of the fairing. This system is a component part of the engine and is controlled through the surface and engine anti-icing switch.



*PRIOR TO TCTO 1F-106-1072

Figure 4-5

NOTE

Operation below approximately 85% rpm may not supply sufficient heat to keep the engine inlet clear of ice under severe conditions. At the relatively high thrust settings used during climb or cruise, ample heat will be provided to assure protection. However, during descent at high airspeeds and low thrust settings, the heat supplied may be inadequate if the ice formation is severe. Under such circumstances, increased thrust should be applied to provide more heat.

INTAKE DUCT ANTI-ICING

The engine intake ducts are supplied with hot engine bleed air. The pressure regulated bleed air flows through the spanwise channels of the duct leading edge and is exhausted partly into the fuselage boundary layer flow area, and partly along the outside surface of the inlet lip. The system is controlled through the surface and engine anti-icing switch.

RADOME ANTI-ICING

The radome anti-icing system uses glycol to prevent ice formation. The glycol supply tank has a storage capacity of two gallons. Engine bleed air cooled by the air-conditioning heat exchanger is used to force the glycol from the tank and through the porous metal ring at the base of the nose boom. Airflow then spreads the glycol over the surface of the radome. The system is controlled through the surface and engine anti-icing switch.

WARNING

- The system is inoperative when the main landing gear doors are open.
- If the system is activated, forward visibility may be reduced due to glycol on the windshield.

RAM AIR "q" PRESSURE INTAKE ANTI-ICING

The ram air pressure intakes on the fin are electrically anti-iced by power from the dc nonessential bus. Electrical power to heaters in the "q" probes is controlled through the surface and engine anti-icing switch. Output of the lower intake anti-icing heater is reduced when the main landing gear doors are open.

WINDSHIELD ANTI-ICING AND DEFOGGING

Anti-icing and defogging of two windshield pane[ls] is accomplished by electrically heating a conductive coating embedded in the panels. For each panel, a relay and a temperature sensing element embedded in the panel automatically control pow[er] to the panels to maintain the proper glass temper[a]ture. Left-hand windshield anti-icing is shut off and cannot be actuated whenever the rain removal system is in operation.

NOTE

Allow approximately two minutes between activation of the switch and actual defogging or anti-icing. Because of this time lag, the switches should be activated prior to takeoff if icing conditions are anticipated.

SURFACE AND ENGINE ANTI-ICING SWITCH

The surface and engine anti-icing switch (figure 4-4), located on the right console, is placarded "Surf & Eng Anti-Ice" and has three positions, AUTO ON, MAN ON, and OFF. On B airplanes the switch is located in the forward cockpit only, is placarded "Sur & Eng Anti-Ice," and has thr[ee] positions, AUTO, OFF, and MAN. When the swit[ch] is in AUTO ON position and the anti-icing system warning light is extinguished, the system is armed and anti-icing of the engine inlet guide vanes, intake duct leading edges, radome, and ram air pressure intakes on the fin is controlled by the automatic anti-icing system. When the switch is in the MAN ON position, the automatic features of the anti-icing system are bypassed and all of the anti-icing systems which are normally controlled by the automatic anti-icing system are activated regardless of icing conditions. When the main landing gear doors are open, the radome anti-icing system is deactivated. The anti-icing switch receives power from the dc essential bus.

ENGINE ANTI-ICING WARNING LIGHT

A light on the warning light panel (10, figure 1-30) will illuminate to display "ENGINE ANTI-ICE" when the surface and engine anti-icing switch in the OFF position, or when the switch is AUTO ON and airflow through the engine intake duct is too low to actuate the ice detector probe or the ice detector probe heater is on for more tha[n] approximately 25 seconds. Illumination of t[he] engine anti-icing warning light indicates that th[e] automatic anti-icing system is inoperative.

antenna locations
(typical)

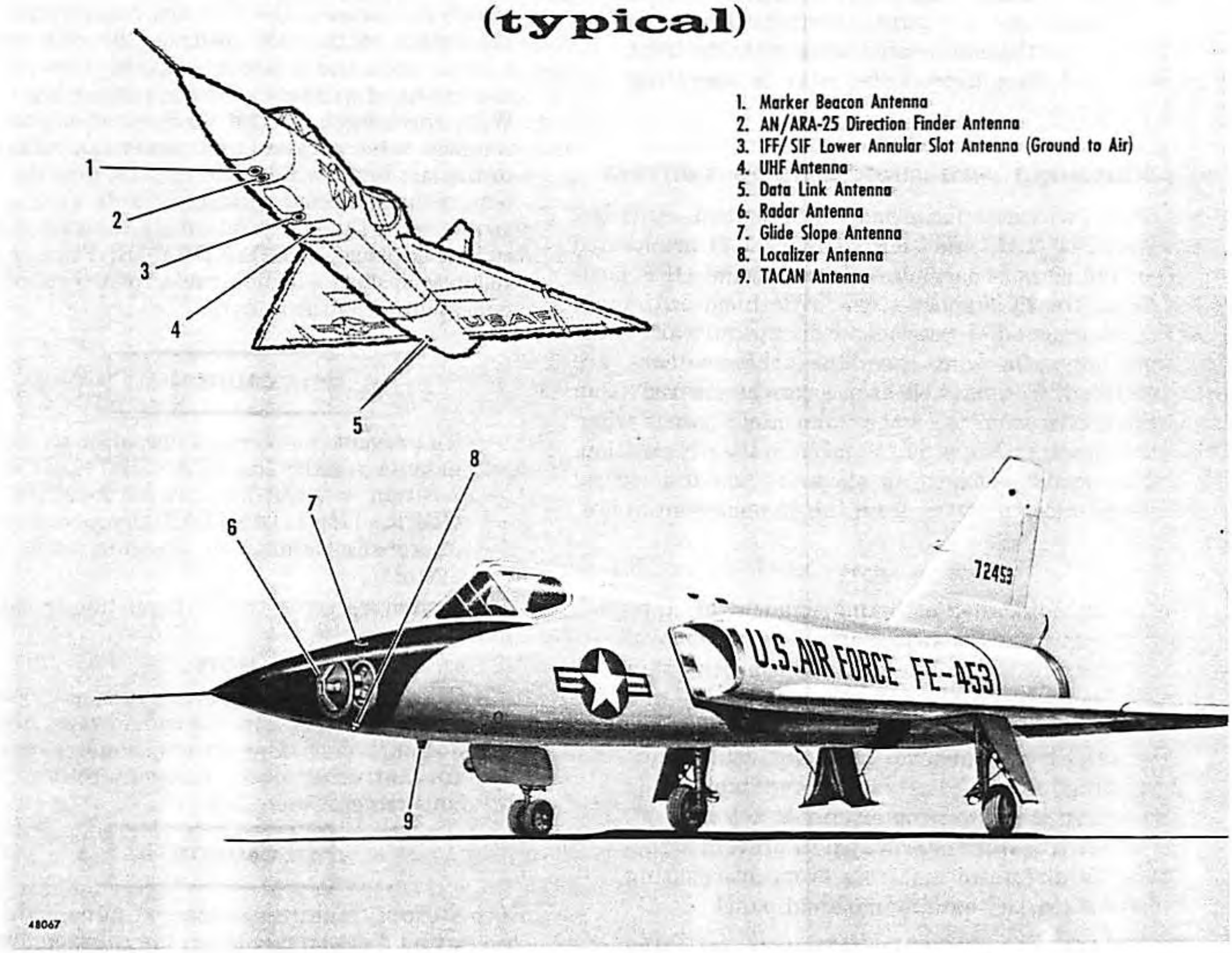


Figure 4-6

NOTE

On airplanes equipped with the engine anti-icing warning test button, illumination of the warning light may indicate a malfunction in the automatic icing detector or a malfunction of the anti-icing shutoff valve. If the warning light extinguishes after placing the surface and engine anti-icing switch in MAN ON, the malfunction is in the icing detector circuits. If the light does not extinguish, the malfunction is in the valve.

The light illuminates when the surface and engine anti-icing switch is in the AUTO ON or MAN ON position and the engine anti-icing shutoff valve is in an intermediate position between open and closed. The light is part of the master warning system and receives power from the dc essential bus.

ENGINE ANTI-ICING WARNING LIGHT TEST BUTTON

On some airplanes an engine anti-icing warning light test button (figure 4-4) is located on the right console and is placarded "Eng Anti-Ice Warn Test." Depressing the button for approximately three seconds will illuminate the engine anti-icing warning light if the surface and engine anti-icing switch is in either the AUTO ON or the MAN ON position.

NOTE

Depressing the engine anti-icing warning light test button will provide an operational check of the three- to five-second time delay relay only. With the button depressed for approximately three seconds, the engine anti-icing warning light will illuminate if the relay is operating properly.

WINDSHIELD ANTI-ICING, ANTIFOG SWITCHES

Two switches placarded "Windshield Anti-Ice, Anti-Fog, LH" and "RH" (figure 4-4) are located on the right console and have ON and OFF positions. On Ⓑ airplanes the windshield anti-icing, antifog switches are located in the forward cockpit only. On some airplanes the switches are guarded. Fog as well as ice can be cleared from the entire surface of the windshield panels when the appropriate switch is placed in the ON position. The windshield heating elements and the control relays receive power from the ac nonessential bus.

NOTE

- If both the left-hand windshield anti-icing, antifog switch and the rain removal switch are in the ON position, the rain removal switch will take precedence and power will be temporarily disrupted to the left-hand windshield anti-icing, antifog circuit. The right-hand windshield anti-icing, antifog circuit is not shut off. This feature is provided to prevent structural failure resulting from overheating of the left-hand windshield panel.
- Allow approximately two minutes between activation of the switches and actual defogging or anti-icing. Because of this time lag, the switches should be activated prior to takeoff.

RAIN REMOVAL SYSTEM

Engine bleed air is used to remove rain and snow from forward sections of the left windshield panel. This system also aids in the prevention of ice or fog formation on this portion of the windshield. Nozzles below and forward of the windshield eject the hot air at high velocity from the rain clearing air duct. Flow of air to the nozzles is controlled by a two-position solenoid operated valve. When the rain removal system is in operation, the electrical power for left-hand windshield anti-icing is shut off.

Rain Removal Switch

The rain removal switch (figure 4-4) is placarded "Rain Removal" and has ON, OFF, and STANDBY DE-FOG positions. The switch is located on the right console. On Ⓑ airplanes the rain removal switch is located in the forward cockpit only. With the switch in the ON position, the rain removal valve is open and a disconnect relay cuts power to the left-hand windshield electric anti-icing system. With the switch in the OFF position, the rain removal valve is closed and power can be supplied to the left-hand windshield electric anti-icing system, if the left-hand windshield anti-icing, antifog switch is in the ON position. If the rain removal switch is placed at STANDBY DE-FOG, the rain removal system will be armed for operation if ac nonessential power is lost.

CAUTION

To prevent inadvertent operation at supersonic speeds, the STANDBY DEFOG position is not utilized. Refer to OTHER OPERATING LIMITATIONS, Section V, for limitations on the rain removal system.

The switch receives power from the dc essential bus.

NOTE

The rain removal system does not clear the entire left-hand windshield when used for anti-icing. The system should be used for anti-icing only if regular electrical anti-icing system fails.

CAUTION

Before making engine runups using ground electrical power, the rain removal switch must be placed in the OFF position to prevent damage to the windshield.

PITOT HEAT SWITCH

The pitot heat switch (figure 4-4), located on the right console, is placarded "Pitot Ht" and has ON and OFF positions. On Ⓑ airplanes the pitot heat switch is located in the forward cockpit only. When in the ON position, the switch supplies dc essential bus power to the pitot heater control relay. The relay in turn supplies ac nonessential or ATG power to the nose boom heater and ac nonessential power to the engine variable ramp pitot heaters. On some airplanes,* the pitot heat switch also controls power to the angle of attack transmitter vane heater circuit.

**Airplanes modified by TCTO 1F-106-850.*

NORMAL OPERATION OF ANTI-ICING SYSTEMS

To place the anti-icing systems in operation:

1. Surface and engine anti-icing switch—AUTO ON.
2. Windshield anti-icing, antifog switches—ON.
3. Pitot heat switch — ON.
4. Rain removal switch — As desired.

EMERGENCY OPERATION OF ANTI-ICING SYSTEMS

Windshield Anti-Icing Failure

1. Rain removal switch — ON (if subsonic).
   Place rain removal switch in ON position to anti-ice left windshield panel.

Engine Anti-Icing Warning Light Illuminated: "ENGINE ANTI-ICE"

1. Surface and engine anti-icing switch — MAN ON.

CAUTION

The engine inlet anti-icing system should not be used in MAN ON at supersonic speeds. During high-speed flight, particularly at low altitude, engine bleed air used for anti-icing may reach temperatures higher than 850°F. The prolonged use of this high-temperature air for anti-icing the engine inlet guide vanes would adversely affect the engine number one bearing area.

NOTE

If the anti-icing warning light illuminates when the surface and engine anti-icing switch is in AUTO ON or MAN ON position, cycle the switch. After cycling the switch, if the light remains on with AUTO ON or MAN ON position selected, the engine anti-icing system has malfunctioned and the engine anti-icing shutoff valve may be inoperative.

LIGHTING EQUIPMENT

EXTERIOR LIGHTING

Exterior lighting consists of two anticollision (beacon) lights on the fuselage, one navigation light near each wing tip, a navigation light near the top of the vertical fin, one landing light on each main landing gear fairing door, a taxi light on the nose landing gear fairing door, and two air refueling slipway lights (some airplanes). The formation and anticollision lights are controlled by a switch in the cockpit (figure 4-7) which allows selection of either formation or navigation lighting display, and a dimming switch to control the brightness of the lights if the formation condition is selected. The landing and taxi lights are controlled by a single selector switch. The air refueling slipway lights are controlled by the air refuel switch. The anticollision lights have dual functions. When the formation-navigation lights switch is in NAV ON, the lights rotate, producing a flashing effect. When the switch is in FORM ON, the lights remain on but do not rotate. The lights appear white from directly above or below, and red from all sides. The lights are manually operated by the formation-navigation lights switch. The lights on the wing tips and fin are illuminated regardless of whether formation or navigation lighting is selected, but can be dimmed by the dimmer switch only if the formation lighting display is selected. The light on the vertical fin is a white light. The light near the left wing tip is red when viewed from forward of the airplane and white when viewed from behind. The light near the right wing tip is green when viewed from forward and white when viewed from behind. For additional information on this system, refer to T.O. 1F-106A-2-10.

FORMATION-NAVIGATION LIGHTS SWITCH

The formation-navigation lights switch (figure 4-7), located on the right console, has FORM ON, center (OFF), and NAV ON positions. On Ⓑ airplanes the formation-navigation lights switch is located in the forward cockpit only. When the switch is in the FORM ON position, the wing, tail, and anticollision lights will operate but the anticollision lights will not rotate. The lights can all be dimmed in the FORM ON position by use of the dimming switch on the lighting control panel. With the switch in the center (OFF) position, the wing, tail, and anticollision lights are inoperative. When the formation-navigation lights switch is in NAV

lighting control panels
(typical)

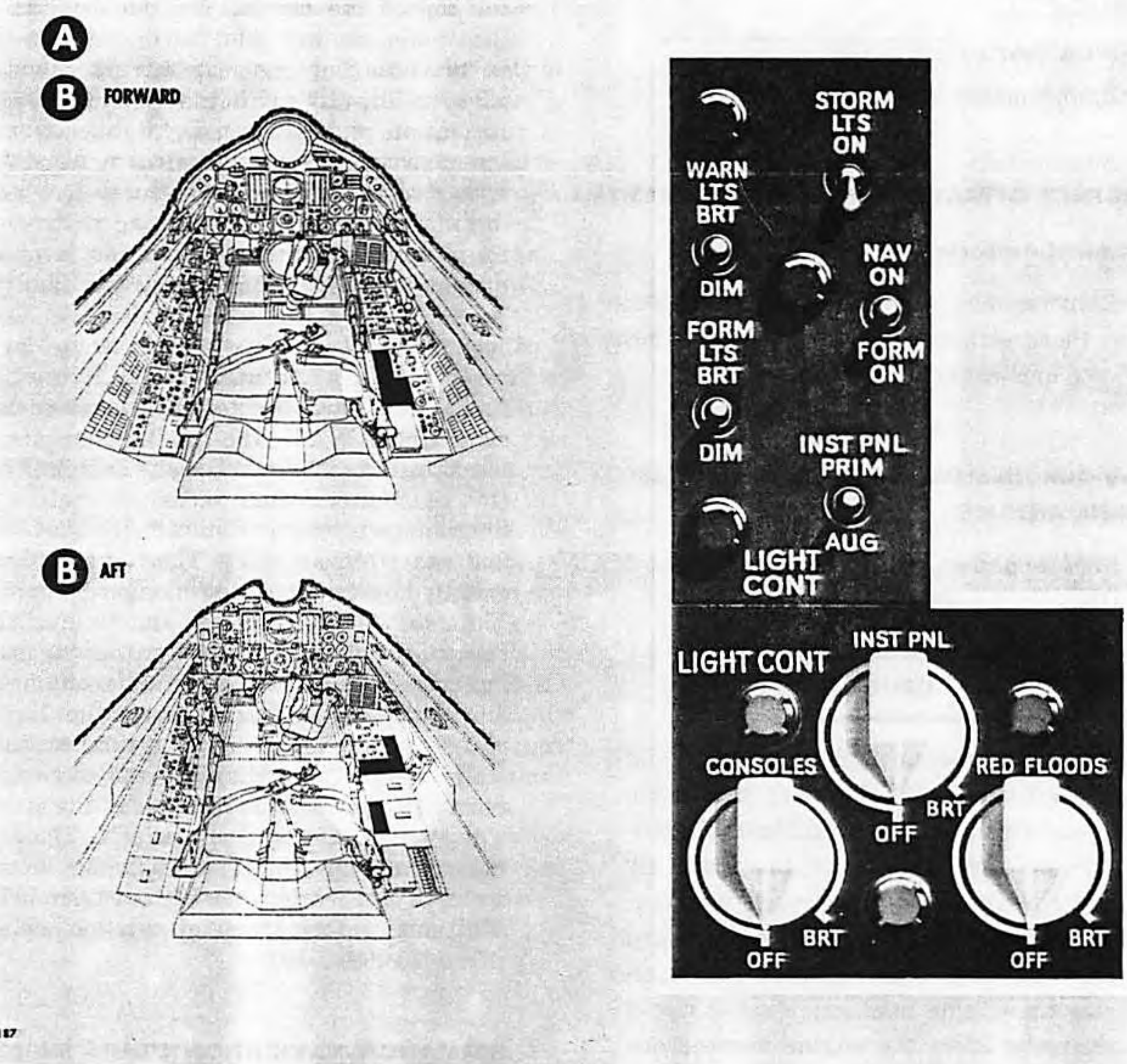


Figure 4-7

ON position the anticollision lights rotate and the wing and tail lights are illuminated. In FORM ON position the formation-navigation light switch receives power from the ac essential bus. In NAV ON position, in addition to ac power, the switch receives dc nonessential bus power for the anticollision light circuit.

NOTE

During flight through actual instrument conditions, the rotating anticollision light should be turned off by moving the formation-navigation light switch to FORM ON. With the light on during instrument conditions, vertigo could be experienced as a result of the rotating reflections of the light against the clouds. In addition, the light would be ineffective as an anticollision light during instrument conditions since it could not be seen by pilots of other airplanes.

FORMATION-NAVIGATION LIGHTS DIMMER SWITCH

A formation-navigation lights dimmer switch (figure 4-7) placarded "Form-Lts" with positions DIM and BRT is located on the right console. On Ⓑ airplanes the formation-navigation lights dimmer switch is located in the forward cockpit only. The switch will dim the formation lights only if the formation-navigation lights switch is in FORM ON position.

LANDING AND TAXI LIGHT SWITCH

The landing and taxi light switch (12, figure 1-10) is located on the cockpit left sidewall. On Ⓑ airplanes the landing and taxi light switch is located in the forward cockpit only. The switch has three positions, LANDING LIGHTS—OFF—TAXI LIGHTS, which allows selection of either landing or taxi lights. A switch on each main landing gear prevents the landing light from operating unless the main landing gear is down and locked. A switch on the nose landing gear prevents the taxi light from operating unless the nose landing gear is down and locked. The landing light and taxi light circuits receive power from the dc nonessential bus.

INTERIOR LIGHTING

Interior lighting equipment includes the edge-lighting for the instrument panels, switch panels and console panels, the thunderstorm lights, cockpit floodlights, the standby compass light, and a map reading light. Edge-lighting of the instrument panel, switch panels, and console panels is accomplished by small lights set into panels. The plastic panel facing has an opaque outer layer and a translucent inner layer. Light from the bulbs in the panels is conducted by the translucent inner layer to the edges of the instruments and the edges of holes through which switches and controls protrude, thereby illuminating instruments and outlining switches and controls. Wherever lettering is cut in the opaque surface material, to identify or mark the positions of a switch or control, light shines through from the translucent layer and illuminates the lettering. Red floodlights are directed at the right and left sides of the instrument panels and the tops of the consoles. White thunderstorm lights provide brilliant illumination of the cockpit to counteract the blinding effects of lightning on the pilot's vision. On airplanes with the integrated instrument system, augmentation lights illuminate the instrument panels to prevent reflection on the glass covering the instruments. Intensity of lighting in ac circuits (which include all of the cockpit lighting except the thunderstorm lights) is controlled by transformers, known as "powerstats." The powerstat control knobs on the lighting control panel have the appearance of rheostat controls. The lighting control panel is located on the right-hand console.

INSTRUMENT PANEL LIGHTS POWERSTAT

The instrument panel lights powerstat (figure 4-7) is located on the lighting control panel and is placarded "Inst Panel." Turning the pointer clockwise from OFF to BRT controls the intensity of the instrument panel lights, and with the pointer at BRT the master warning lights will be at full brilliance. Power is supplied from the ac essential bus.

CONSOLE LIGHTS POWERSTAT

The console lights powerstat (figure 4-7) is located on the lighting control panel and is placarded "Consoles." Turning the pointer clockwise from OFF to BRT controls the intensity of the lights on both consoles and the check list. Power is supplied from the ac essential bus.

COCKPIT FLOODLIGHTS POWERSTAT

The cockpit floodlights powerstat (figure 4-7) is located on the lighting control panel and is placarded "Red Floods." Turning the pointer clockwise from OFF to BRT controls the intensity of the red floodlights. Power is normally supplied from the ac essential bus; however, with loss of ac electrical power, the red floodlights are automatically transferred to the dc essential bus and operate at full brilliance.

AUGMENTATION LIGHTS SWITCH

On airplanes with the integrated flight instrument system, an augmentation lights switch is located on the lighting control panel. The switch (figure 4-7) is placarded "Inst Pnl" and has PRIM and AUG positions. With the switch in the PRIM position, the instruments will be illuminated by integral red lights. With the switch in the AUG position, the instruments will be illuminated by white lights located forward along the cockpit sidewall. These lights will aid in visual adaptation under conditions of extreme glare and reflection. The intensity of the light, with the switch in either position, is controlled by the instrument panel lights powerstat. Power to the augmentation lights switch is supplied by the ac essential bus.

MAP READING LIGHT

A map reading light and switch are located on the right side of the cockpit (7 and 10, figure 1-11). The light is white and is attached to a flexible shaft. The switch has ON and OFF positions, and receives power from the dc essential bus.

THUNDERSTORM LIGHTS SWITCH

The thunderstorm lights switch (figure 4-7) is located on the lighting control panel. On Ⓑ airplanes the thunderstorm lights switch is located in the forward cockpit only. The switch is placarded "Storm Lts" and has an ON position and unplacarded off position. With the switch in the ON position, 15 white thunderstorm lights are energized to illuminate the two consoles and the instrument panel. Moving the switch to the ON position also cuts out the master warning dimming relay so the master warning lights will be at full brilliance when the thunderstorm lights are on. The thunderstorm light switch receives power from the dc nonessential bus.

OXYGEN SYSTEM

The airplane is equipped with a liquid oxygen system (figure 4-8). The major components of the liquid oxygen system are a storage and converter unit, an oxygen regulator, an external filler valve, and a pressure gage and content gage located on the left console. The storage and converter unit is a five-liter on Ⓐ airplanes without air refueling capability and a ten-liter on Ⓑ airplanes and Ⓐ airplanes with air refueling (one liter equals about one quart) insulated storage container. This unit converts the liquid oxygen to gaseous oxygen and then supplies it to the oxygen regulator. The oxygen regulator, which is located in the survival kit, is a pressure-breathing regulator that delivers 100% oxygen at all times and is to be used with the pressure helmet and pressure suit. The liquid oxygen content gage is a capacitance-type and indicates the supply of liquid oxygen in the storage container. Oxygen duration at various altitudes is shown in figure 4-9. At sea level and with average temperature, a full supply of liquid oxygen dissipates through a relief valve in about five days. The liquid oxygen system is serviced through a single-point filler valve located within an access door on the left side of the fuselage below the cockpit. For additional information on this system, refer to T.O. 1F-106A-2-6.

OXYGEN REGULATOR

A pressure-breathing oxygen regulator is mounted in the aft portion of the survival kit in the ejection seat. Gaseous oxygen is supplied from the oxygen converter at approximately 70 psi during normal operation. During emergency operation, oxygen is supplied from the emergency oxygen supply. Emergency oxygen pressure is approximately 1800 psi when fully charged; therefore it is necessary to reduce this pressure by means of a restrictor prior to delivery to the oxygen regulator. The regulator regulates and delivers 100% oxygen to the oxygen mask or pressure helmet and pressure suit. Using the airplane oxygen supply, the regulator supplies oxygen under increasing pressure as altitude increases. The regulator will not function as a diluter (will not mix air with oxygen) and therefore delivers 100% oxygen at all times. The pressure-breathing oxygen system is controlled by a pressure-breathing oxygen supply switch (figure 4-8) located on the left console.

NOTE

When using an MBU-3/P or MBU-5/P oxygen mask with the pressure-breathing system instead of the pressure suit and pressure helmet, it is necessary to use an adapter to reduce the amount of oxygen pressure being supplied to the mask.

OXYGEN CONTROL PANEL

The liquid oxygen control panel (figure 4-8) is located on the left console. The panel contains a pressure-breathing oxygen supply switch, an oxygen quantity gage, an oxygen pressure gage, and an oxygen quantity gage test button. The pressure-breathing oxygen supply switch has ON-OFF positions and controls the flow of oxygen from the airplane supply to the pressure oxygen regulator.

WARNING

Uncontrolled flow of oxygen through open oxygen masks or pressure helmets during ground operation will result in a rapid depletion of oxygen. During uncontrolled flow, frost will appear and may freeze the switch because the system is unable to convert at a rapid pace. This may cause liquid to enter the cockpit. The uncontrolled flow of oxygen occurs most frequently during engine start and taxi operations when the pressure-breathing oxygen supply switch is ON and the oxygen mask or pressure helmet faceplate is not in place.

PRESSURE-BREATHING OXYGEN SUPPLY SWITCH

The pressure-breathing oxygen supply switch (figure 4-8) is located on the oxygen control panel. The switch controls the flow of oxygen to the regulator for the pressure-breathing oxygen system. The switch has two positions, ON and OFF; moving the switch forward to the ON position permits oxygen to flow from the airplane supply to the oxygen regulator in the survival kit. With the switch in the OFF position, no oxygen will flow to the oxygen regulator.

WARNING

Uncontrolled flow of oxygen through open oxygen masks or pressure helmets during ground operation will result in a rapid depletion of oxygen. This uncontrolled flow of oxygen occurs most frequently during engine start and taxi operations when the switch is ON and the oxygen mask or pressure helmet faceplate is not in place.

OXYGEN QUANTITY GAGE

An oxygen quantity gage (figure 4-8) is included on the oxygen control panel installed on the left console. The gage is a capacitance type that indicates the contents of the oxygen converter through a sensing element submerged in the liquid oxygen. The gage is calibrated in liters from 0 to 5 on some Ⓐ airplanes and from 0 to 10 on other Ⓐ airplanes and Ⓑ airplanes. Gage operation can be checked by the oxygen quantity gage test button; if the button is held depressed, the gage pointer should move toward zero, and when the button is released, the pointer should return to its original position. Failure of the pointer to move indicates a faulty system. The indicating system receives power from the ac essential bus.

NOTE

- On Ⓐ airplanes with a five-liter gage, the oxygen quantity gage should read between 4 and 4½ liters when the system is fully charged. It is impossible to charge the liquid oxygen converter to the full five liters.
- On Ⓑ airplanes and Ⓐ airplanes with a ten-liter gage, the oxygen quantity gage should read between eight and nine liters when the system is fully charged. It is impossible to charge the liquid oxygen converter to the full ten liters.

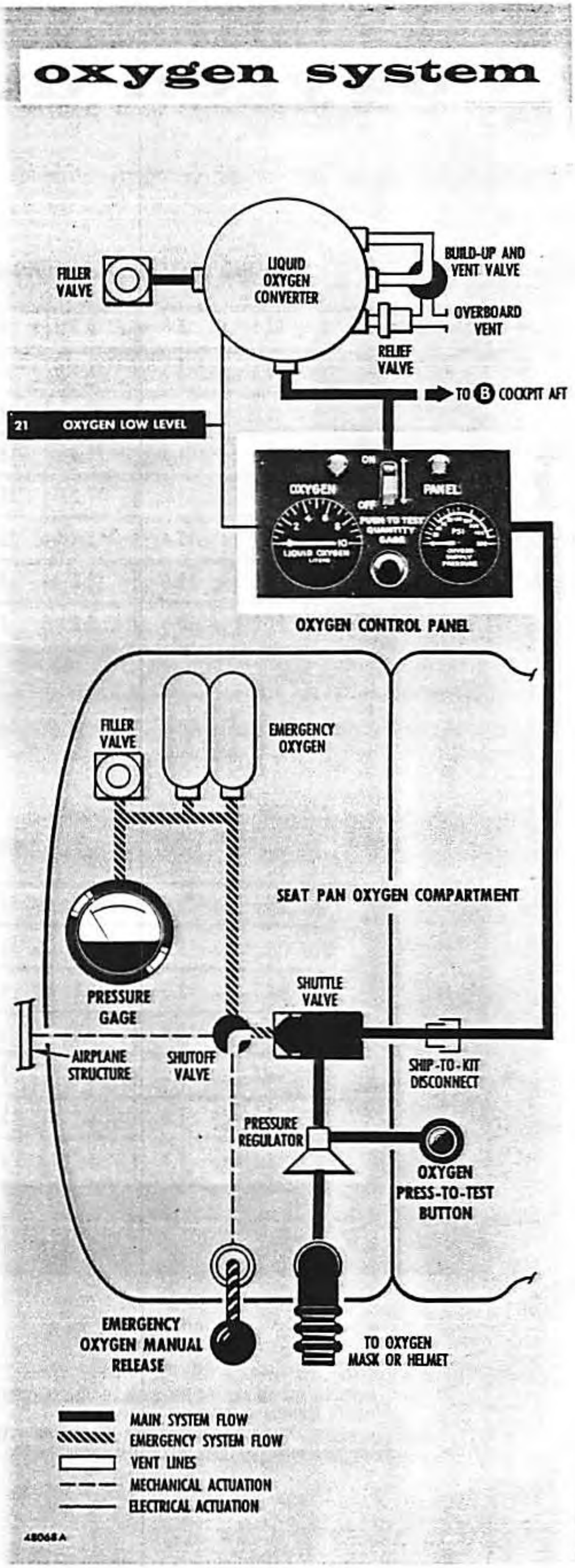


Figure 4-8

oxygen duration–hours

USING PRESSURE BREATHING OXYGEN MASK (TYPE MBU-3/P OR MBU-5/P)

CABIN ALTITUDE—FEET	10	9	8	7	6	5	4	3	2	1
35,000 AND ABOVE	62.8	56.6	50.4	44.0	37.8	31.4	25.2	18.8	12.6	6.2
30,000	45.4	40.8	36.2	31.6	27.2	22.6	18.0	13.6	9.0	4.6
25,000	35.0	31.4	28.0	24.4	21.0	17.6	14.0	10.4	7.0	3.6
20,000	26.6	24.0	21.4	18.6	16.0	13.4	10.6	8.0	5.4	2.6
15,000	21.4	19.2	17.2	15.0	12.8	10.6	8.6	6.2	4.4	2.2
10,000	17.2	15.4	13.8	12.0	10.4	8.6	6.8	5.2	3.4	1.8
5,000	13.6	12.6	10.8	9.8	8.2	7.0	5.4	4.2	2.8	1.4
SL	11.0	10.0	8.8	7.6	6.6	5.6	4.4	3.4	2.2	1.0

QUANTITY GAGE—LITERS

BELOW 1 LITER DESCEND TO ALTITUDE NOT REQUIRING OXYGEN

USING FULL PRESSURE SUIT

CABIN ALTITUDE—FEET	10	9	8	7	6	5	4	3	2	1
30,000 AND ABOVE	25.6	23.0	20.4	18.0	15.4	13.0	10.2	7.6	5.2	2.6
25,000	19.6	18.0	15.6	14.0	11.8	10.0	7.8	6.0	4.0	2.0
20,000	15.2	12.6	12.2	9.8	9.2	8.0	3.0	4.2	3.0	1.4
15,000	12.0	10.8	9.6	8.4	7.2	3.0	4.8	3.6	2.4	1.2
10,000	9.6	9.0	7.6	7.0	5.8	5.0	3.8	3.0	2.0	1.0
5,000	7.8	7.2	6.2	5.6	4.6	4.0	3.2	2.4	1.6	0.4
SL	6.4	5.4	5.2	4.2	3.8	3.0	2.6	1.8	1.4	0.6

QUANTITY GAGE—LITERS

BELOW 1 LITER DESCEND TO ALTITUDE NOT REQUIRING OXYGEN

NOTE

- FIGURES SHOWN ARE FOR USE WITH 10-LITER CONVERTER WITH A ONE-MAN CREW.
- FOR USE WITH 5-LITER CONVERTER OR 10-LITER CONVERTER WITH A TWO-MAN CREW, HALVE THE ABOVE FIGURES.

48069

Figure 4-9

OXYGEN PRESSURE GAGE

An oxygen pressure gage is installed on the oxygen control panel. The gage shows gaseous oxygen pressure from 0 to 500 psi. When oxygen is being used from the system, the gage will normally indicate from 70 to 80 psi. However, under static conditions and on a hot day the gage may indicate as high as 110 psi. Gage operation requires no electrical power.

OXYGEN-LOW WARNING LIGHT

An oxygen-low warning light (21, figure 1-30) on the master warning light panel illuminates and displays "OXYGEN LOW LEVEL" when the oxygen supply in the liquid oxygen system falls below one-half liter in a five-liter converter and one liter in a ten-liter converter. The light is actuated by the liquid oxygen content gage shaft pointer and receives power from the dc essential bus. The light is tested with the lights in the master warning system.

OXYGEN QUANTITY GAGE TEST BUTTON

An oxygen quantity gage test button (figure 4-8) is part of the oxygen control panel located on the left console. When the button is depressed, the oxygen quantity gage pointer will move toward zero, and when the button is released the pointer should return to its original position. Failure of the pointer to move indicates a faulty system. The indicator receives power from the ac essential bus.

OXYGEN PRESS-TO-TEST BUTTON

An oxygen press-to-test button is located on the forward edge of some survival kits. Depressing the button will provide positive pressure to the oxygen mask or the pressure helmet to determine that the oxygen system is operating properly prior to takeoff. The button is the only method of checking for proper operation of the pressure breathing oxygen system (other than a decrease of liquid content and positive flow of oxygen through the system), as there is no blinker or flow indicator installed on this system. A several second delay may occur before pressure buildup.

OXYGEN MASK ADAPTER

When a flight is made with an MBU-3/P or MBU-5/P oxygen mask, an adapter is provided to enable the mask to be connected with the survival kit. The adapter is in two sections, one for the electrical connections and one for the oxygen connection. The adapter for the oxygen lead also provides a pressure restrictor to reduce the oxygen pressure from the regulator to the oxygen mask.

WARNING

If cabin pressure is lost while wearing an MBU-3/P or MBU-5/P oxygen mask, an immediate descent to 25,000 feet or below is mandatory.

PRESSURE-BREATHING OXYGEN SYSTEM PREFLIGHT CHECK (A/P 22S-2 FULL PRESSURE SUIT)

1. Connect the helmet oxygen hose to the suit controller before entering cockpit.
2. Check emergency oxygen bottle gage at 1800 psi minimum.
3. After entering the cockpit, connect ventilation hose to suit connector.
4. Check oxygen pressure gage at 70 to 110 psi.
5. Check oxygen quantity at 3 liters minimum per crew member.
6. Oxygen quantity gage test button—DEPRESS.

   Depress button and hold for 10 seconds, oxygen low level warning light will illuminate at ½ liter (five-liter converter) or 1 liter (ten-liter converter).
7. Connect oxygen line from personal equipment lead bundle to suit controller. Insure that the capstan hose and oxygen-breathing hose are capped.
8. Connect communications lead.
9. Pressure-breathing oxygen supply switch—ON.
10. Lower visor. Check pneumatic seal inflates and oxygen flows through the spray bar.
11. Press the pressure suit press-to-test button. Check breathing and if blow-by is experienced, tighten helmet take-up reel. Check zippers for leakage.
12. Release press-to-test button and check suit deflates to ambient pressure.
13. Adjust vent flow to suit (after starting engine).

PRESSURE-BREATHING OXYGEN SYSTEM PREFLIGHT CHECK (MBU-3/P OR MBU-5/P OXYGEN MASK)

Before takeoff, the oxygen system should be connected and tested as follows:

1. Check emergency oxygen bottle gage at 1800 psi minimum.

pressure suit oxygen connection

(typical)

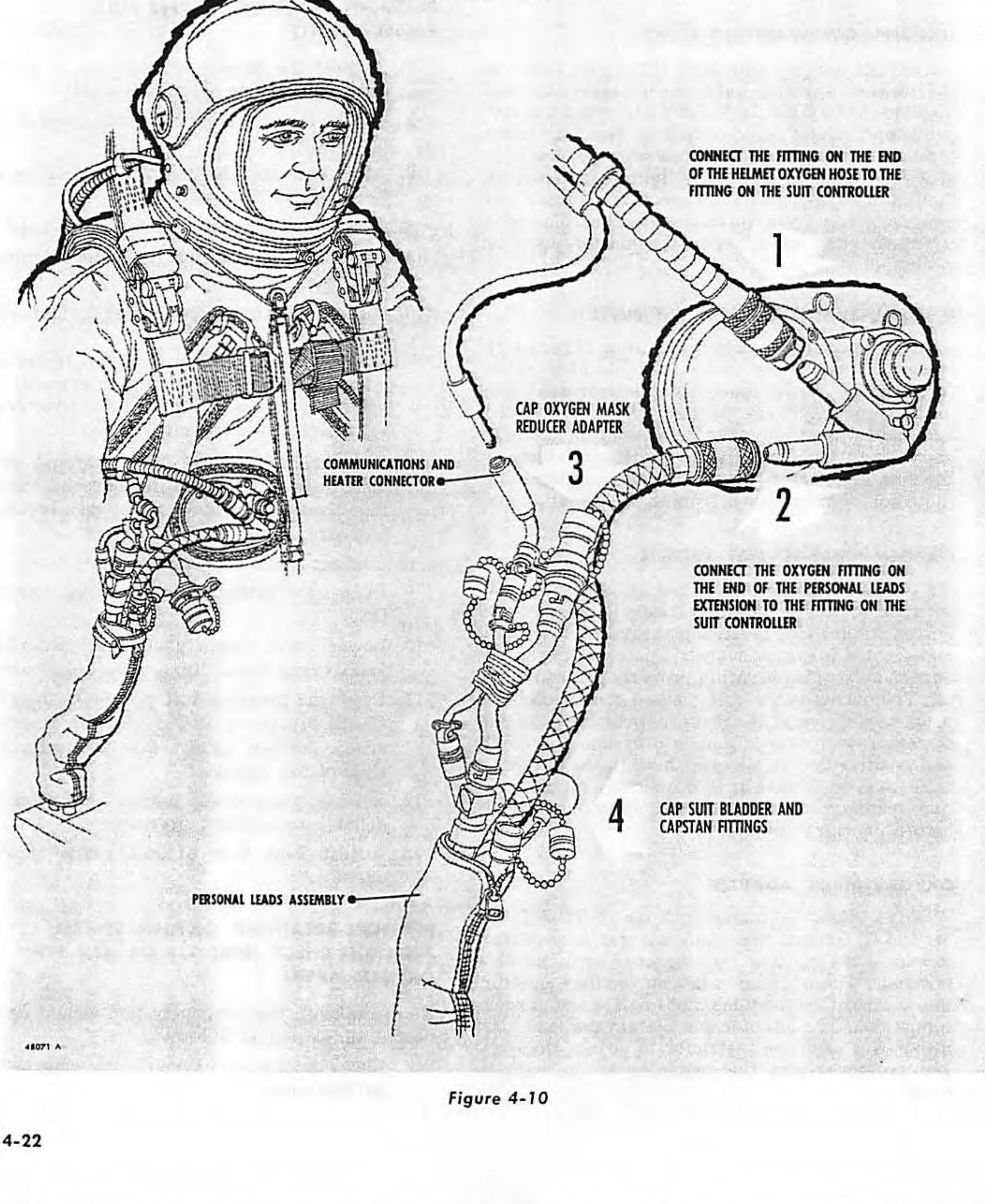


Figure 4-10

2. Connect the oxygen mask adapter to end of oxygen hose in personal equipment lead bundle. Ensure that capstan and pressure suit fittings are capped.
3. Attach oxygen mask and adjust for normal breathing. A slight positive pressure is supplied at all times, but should not cause resistance to breathing.
4. Check bailout bottle hose connected to mask connector (if required).
5. Check oxygen pressure gage at 70 to 110 psi.
6. Check oxygen quantity gage at 3 liters minimum, per crew member.
7. Oxygen-low warning light—Out.
8. Oxygen quantity gage test button—TEST.

   Depress and hold the oxygen quantity gage test button until the oxygen-low warning light illuminates at ½ liter (five-liter converter) or one liter (ten-liter converter).
9. Check pressure-breathing oxygen supply switch—ON.
10. If an MBU-3/P or MBU-5/P mask is worn, breathe normally into the mask. As a slight positive pressure is supplied at all times there should be no resistance to breathing.
11. Depress the oxygen press-to-test button (some airplanes). A definite positive pressure should result within the mask. Hold breath to determine whether there is leakage around the mask or faceplate. Release the oxygen press-to-test button. Check for normal breathing.

NORMAL OPERATION OF PRESSURE-BREATHING OXYGEN SYSTEM

NOTE

Perform oxygen system preflight check as outlined above prior to each flight. After completion of preflight check, oxygen system is ready for normal usage.

1. Check pressure-breathing oxygen supply switch in the ON position.
2. After landing, when oxygen is no longer desired, place the pressure-breathing oxygen supply switch OFF immediately prior to removing oxygen mask or releasing faceplate.

NOTE

To prevent depletion of the oxygen supply, the pressure-breathing oxygen supply switch should not be left ON unless oxygen is being used by the pilot.

EMERGENCY OPERATION OF PRESSURE-BREATHING OXYGEN SYSTEM

1. If symptoms of hypoxia develop, check oxygen hose connections and check pressure-breathing oxygen supply switch ON.
2. If the airplane's oxygen supply is depleted, contaminated, or not supplying oxygen, actuate the emergency oxygen supply by pulling the emergency oxygen manual release (round green knob) on the personal equipment lead bundle. Turn pressure-breathing oxygen supply switch OFF and descend to a cockpit altitude below 10,000 feet within 10 minutes.

NOTE

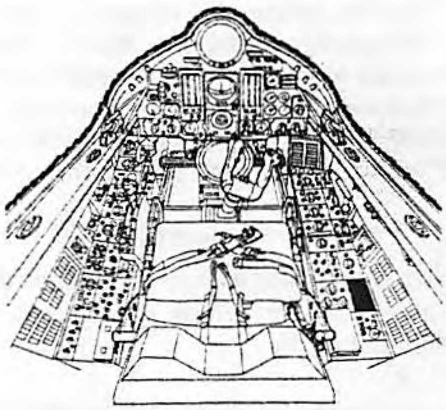
An additional source of oxygen is available if the parachute contains an oxygen bailout bottle.

3. If wearing an MBU-3/P or MBU-5/P oxygen mask and cabin pressure is lost, an immediate descent to 25,000 feet or below is mandatory.

COMPASS SYSTEM

The magnetic compass system may be used as a directional gyro corrected for apparent drift due to the earth's rotation or as a directional, gyro-stabilized magnetic compass. The magnetic compass system consists of a remote directional control gyro, an amplifier-servo assembly, a control panel (figure 4-11) on the right console, and the rotating compass card of the course indicator on the instrument panel. On B airplanes the compass control panel is located in the forward cockpit only. The two modes of operation, magnetic slaved mode and directional gyro, provide accurate directional reference for all latitudes. The directional gyro mode is the most reliable at latitudes near the magnetic poles, since the magnetic slaved mode is subject to severe magnetic distortion near the poles. When in the magnetic slaved mode the system is basically a gyro-stabilized compass slaved to the magnetic azimuth detector transmitter. This mode provides magneitc heading without northerly turning error or oscillations. Directional gyro mode may be used at all latitudes, but is most useful when the magnetic field is weak or distorted or when navigating in the polar regions. When in directional gyro mode, the system is free of magnetic influence and operates

compass control panel



SOME AIRPLANES

OTHER AIRPLANES

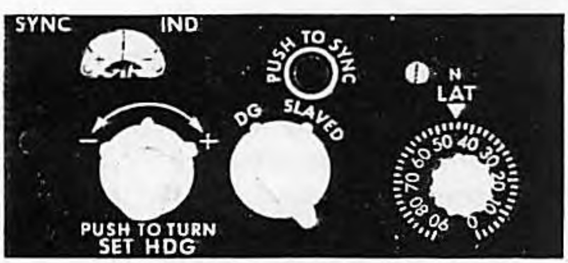


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Figure 4-11

as a directional gyro indicating an arbitrary-gyro heading (corrected for apparent gyro drift due to the earth's rotation) as selected by the pilot. At different latitudes, apparent gyro drift varies, with the smallest amount of drift being at the equator and the greatest amount in the polar regions. In directional gyro mode, with the proper latitude selection made on the control panel, the gyro is made to precess the correct amount required to overcome gyro drift at the selected latitude. The system is powered by the 200/115-volt and 26-volt ac essential bus and the dc essential bus. For additional information on this system, refer to T.O. 1F-106A-2-9.

FUNCTION SELECTOR SWITCH

The two-position function selector switch (figure 4-11), located on the magnetic compass control panel, has positions DG and MAG on airplanes with the conventional instrument display, and DG and SLAVED on airplanes with the integrated flight instrument system. DG position selects directional gyro mode; MAG (SLAVED, some airplanes) selects the magnetic slaved mode. The switch receives power from the dc essential bus.

SYNCHRONIZER KNOB AND SYNCHRONIZATION INDICATOR

The synchronizer knob (figure 4-11), located on the magnetic compass control panel, provides a manual means to fast slave, or synchronize, the rotating compass card of the course indicator or horizontal situation indicator to the correct magnetic heading when the system is in magnetic slaved operation. When in directional gyro mode, the synchronizer knob is used to position the compass card to the desired gyro heading. The knob may be moved to the left, DECR −, position or to the right, INCR +, position. Synchronization rate is controlled by the amount of movement of the synchronization knob from the spring-loaded center position. When in DG operation, the direction of movement of the knob is determined by whether an increased or decreased heading is desired. When in MAG (SLAVED, some airplanes) operation the direction of displacement of the synchronizer knob is determined by a synchronization indicator above the function selector switch on the control panel. When the function selector switch is in MAG (SLAVED, some airplanes) the synchronization indicator is exposed and displays "MAG" ("SLAVED," some airplanes). A center dash mark on the indicator is opposed by a needle which will swing to either a "+" indication on the left or a "–" indication on the right by signals from the flux valve transmitter. When the needle is off center to the "+" side, the synchronizer knob is moved to the INCR + direction to the desired rate of slewing until the synchronization needle swings down to the center position. If the needle is indicating "–," the knob is moved to the DECR – position to center the needle. Centering the needle of the synchronization indicator synchronizes the rotating compass card to the correct magneitc heading. The synchronization knob receives power from the dc essential bus.

Automatic Synchronizer Button

Airplanes with the integrated flight instrument system have an automatic synchronizer button located on the compass control panel and placarded "Push to Sync." Momentary depression of the button when the compass system is in magnetic slaved mode will rotate the azimuth ring on the horizontal situation indicator to the correct magnetic slaved heading. The button may be used for synchronization instead of the synchronizer knob. The button receives power from the dc essential bus.

HEMISPHERE SELECTOR SCREW AND INDICATOR

The hemisphere selector screw (figure 4-11) on the compass control panel is used to select the hemisphere in which the airplane is operating. A small window beside the screw displays "N" or "S" to indicate the hemisphere selected.

LATITUDE SELECTOR KNOB

The latitude selector knob (figure 4-11) on the compass control panel is used to rotate a circular dial at the base of the knob placarded "Lat." The circular dial is numbered from 0 to 90 to indicate degrees latitude, and has a mark for each two degrees. The latitude selector knob and dial are operative in DG mode only and are used to select the latitude in which the airplane is operating. When in DG operation, with the operating latitude selected, the directional gyro will be corrected for apparent drift due to the earth's rotation.

NOTE

The proper corrections will not be made if the hemisphere selector screw is not indicating the correct hemisphere.

The latitude selector knob receives power from the dc essential bus.

NORMAL OPERATION OF THE COMPASS SYSTEM

Magnetic Slaved Mode

1. Function selector switch — MAG (some airplanes).
   Function selector switch — SLAVED (other airplanes).
2. Allow approximately two minutes warmup time after power is applied. When power is initially applied, or when switching from DG, a fast slewing action is applied for the first 15 seconds to synchronize the compass card with the flux valve (remote compass) transmitter. After the initial fast slave cycle, the system returns to the normal slewing cycle of two degrees per minute.

CAUTION

After moving the function selector switch from DG to MAG (SLAVED, some airplanes) wait two minutes before recycling the function selector switch to allow the thermal time delay relay to cool.

3. Before takeoff, check the synchronization indicator to see if the system is synchronized. If the system is not synchronized prior to takeoff, use the synchronizer knob or automatic synchronizer button to cent^r the synchronization needle.

CAUTION

The synchronization knob may be used at any time to obtain synchronization, but should not be operated for more than 30 seconds at a time to prevent overheating of the slewing motor.

Directional Gyro Mode

1. Allow approximately seven minutes warmup time after power is applied.
2. Select the desired hemisphere with the hemisphere selector screw. Check the "N" or "S" indication in the hemisphere selector indicator window.
3. Select the latitude in which the airplane will be operating with the latitude selector knob.

4. After desired heading is established in magnetic mode, switch the function selector switch to DG. The system is then independent of the magnetic compass equipment, and latitude correction for apparent gyro drift is being given to the compass card.

CAUTION

After moving the function selector switch from DG to MAG (SLAVED, some airplanes) wait two minutes before recycling the function selector switch to allow the thermal time delay relay to cool.

NOTE

As the airplane passes through different latitudes in flight, the latitude selector knob should be rotated to the new latitude every two degrees of latitude change.

 All data on pages 4-27 thru 4-28, including figures 4-12 thru 4-12A deleted.

COMMUNICATIONS, NAVIGATION AND LANDING (CN&L) SUBSYSTEM

The communications, navigation and landing (CN&L) subsystem performs the operations in the airplane which are associated with: (1) two-way communication between the airplane and ground; (2) automatic visual display of target; and (3) landing directions upon return to base. Ground-to-air IFF identification is also provided.

Interphone System Ⓐ

An interphone system is installed to permit communications between the pilot and ground crew through the right main wheel well connection (nose wheel well, some airplanes). The microphone is hot and operates on airplane dc essential power. Volume is controlled by the volume control knob on the UHF command radio panel.

Interphone System Ⓑ

The interphone system provides audio communication between the pilots. When the airplane is on the ground with ground interphone equipment connected, the system also allows communication between persons in the cockpit and outside the airplane. The interphone functions as a "hot mike" (the microphone buttons need not be depressed to communicate on interphone). The pilot's microphones and headsets are part of the interphone system. Volume controls, located on the forward and aft left consoles, allow the pilots to select a comfortable volume level for listening. A receptacle on the nose wheel well switch panel provides for connection to the interphone system by the ground crew. The interphone system operates on airplane dc essential power.

Intercom Volume Control Knob Ⓑ. The intercom volume control knob (35, figure 1-10), located on the forward and aft left consoles, is normally set at a comfortable volume level for listening, but can be used to reduce volume below hearing level if so desired.

Operation of the Interphone System

Ⓑ 1. Master electrical power switch—ON or external power connected.

Ⓑ 2. Adjust volume control to desired listening level.

NOTE Ⓑ

The interphone system functions as a "hot mike," and the microphone buttons need not be depressed to communicate on interphone. Depressing a microphone button cuts out interphone communications and actuates the UHF command radio transmitter. When either cockpit microphone button is depressed, either pilot can transmit on UHF communications. When either pilot is transmitting on UHF, the other pilot hears the transmission through the UHF circuits and not through normal interphone circuits.

All data on page 4-28B deleted.

communications, navigation and landing subsystems

(EQUIPMENT IS A COMPONENT OF MA-1 AIRCRAFT AND WEAPONS CONTROL SYSTEM)

TYPE	FUNCTION	PRIMARY OPERATOR	RANGE	LOCATION OF CONTROLS
INTERPHONE EQUIPMENT	CONNECTS AUDIO OF RADIO AND NAVIGATION SYSTEMS AND PILOT'S HEADSET. ALSO PROVIDES COMMUNICATION BETWEEN PILOT AND GROUND CREW WHEN PLANE IS ON THE GROUND. PROVIDES COMMUNICATION BETWEEN PILOTS ON Ⓑ AIRPLANES.	PILOT(S) AND GROUND CREW	COCKPIT TO GROUND CREW AND BETWEEN PILOTS ON Ⓑ AIRPLANES.	GROUND CREW CONNECTION AND AMPLIFIER IN THE NOSE WHEEL WELL. ON Ⓑ AIRPLANES PILOTS' CONTROLS ON FWD AND AFT LEFT CONSOLES.
UHF RECEIVER TRANSMITTER	COMMUNICATIONS FROM AIRPLANE TO AIRPLANE AND FROM AIRPLANE TO GROUND BY UHF COMMUNICATIONS RADIO.	PILOT	LINE OF SIGHT.	LEFT-HAND CONSOLE.
AUTOMATIC DIRECTION-FINDING EQUIPMENT (UHF)	INDICATION OF RADIO STATION BEARING, OR AIRPLANE TO AIRPLANE BEARING.	PILOT	LINE OF SIGHT.	LEFT-HAND CONSOLE. INDICATOR ON INSTRUMENT PANEL.
TACAN	USED IN CONNECTION WITH TACAN SURFACE BEACON TO PROVIDE AIRPLANE GEOGRAPHICAL LOCATION.	PILOT	LINE OF SIGHT 300 MILES	RIGHT-HAND CONSOLE. INDICATOR ON HORIZONTAL SITUATION INDICATOR.
ILS RECEIVER	RECEIVES ILS LOCALIZER AND GLIDE SLOPE TRANSMISSION.	PILOT	LOCALIZER APPROXIMATELY 45 MILES; GLIDE SLOPE APPROXIMATELY 25 MILES.	LEFT-HAND CONSOLE. INDICATOR ON INSTRUMENT PANEL.
IFF/SIF (GROUND TO AIR)	AUTOMATIC RADAR IDENTIFICATION RETURNED IF CHALLENGED BY SURFACE EQUIPMENT. CAN TRANSMIT EMERGENCY IDENTIFICATION SIGNAL.	PILOT	LINE OF SIGHT.	LEFT-HAND CONSOLE.
DATA LINK	GCI DATA RECEIVER.	PILOT	200 MILES LINE OF SIGHT.	RIGHT-HAND CONSOLE.

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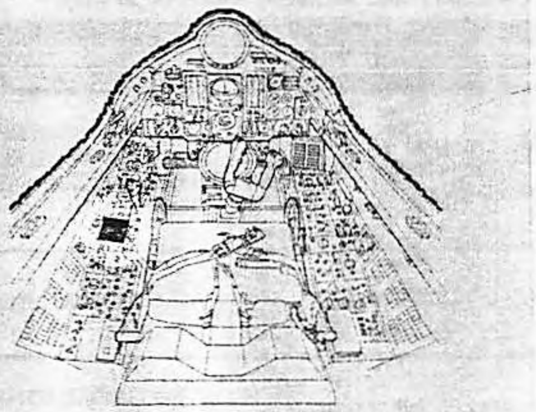
Figure 4-13

communications frequency selector (uhf control panel)



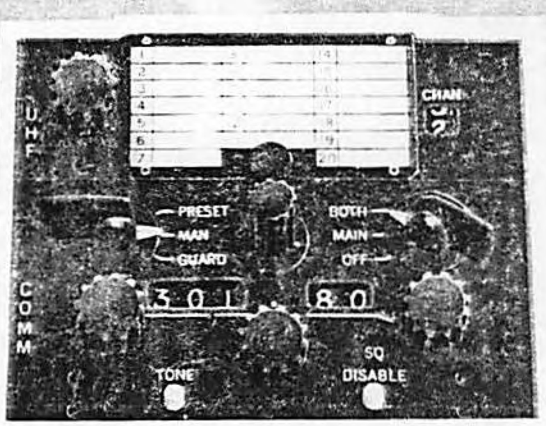


Figure 4-14

control transfer panels (typical)

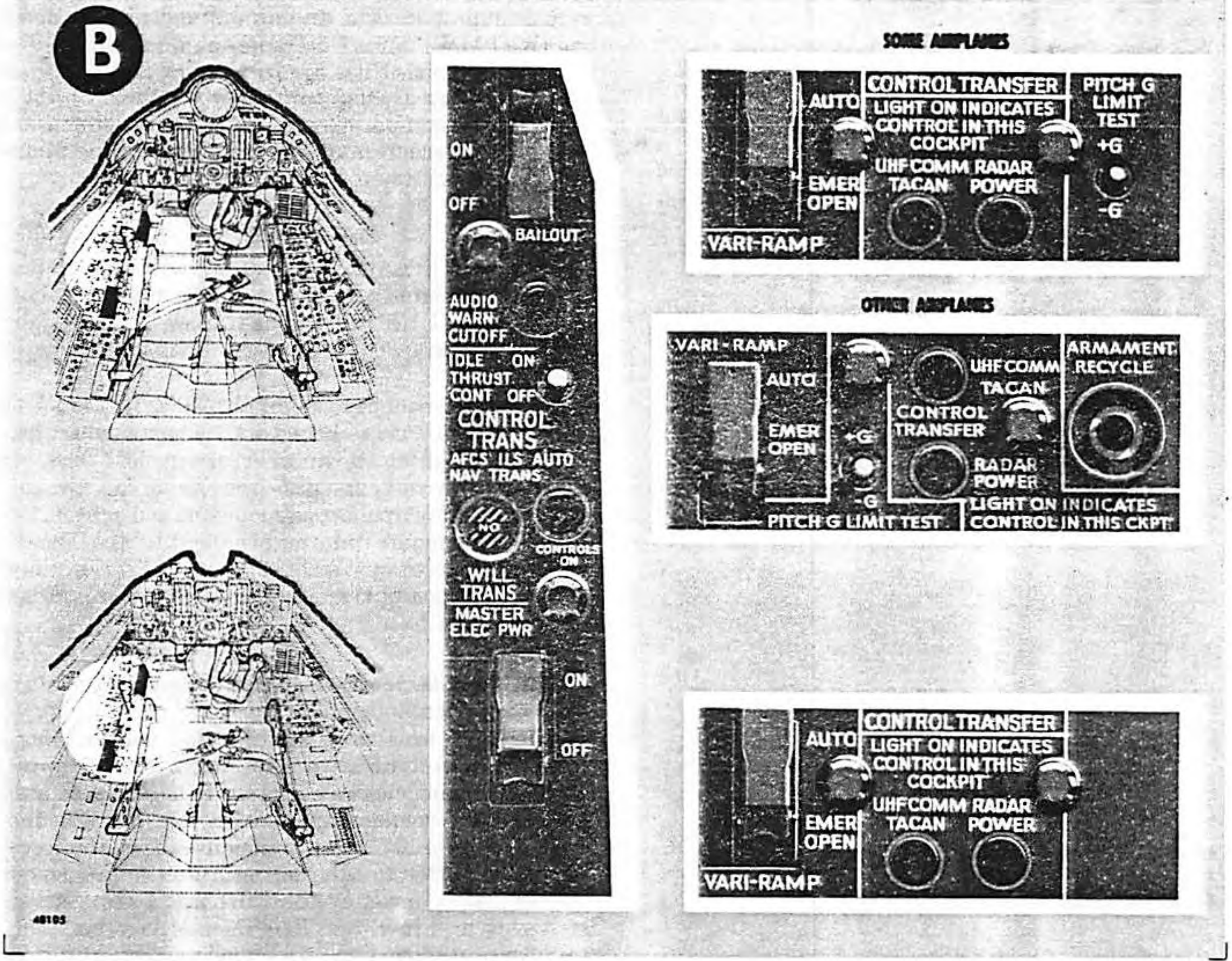


Figure 4-15

UHF Command Radio

The UHF command radio provides air-to-air and air-to-ground voice communications on any of 3500 frequency channels in the range of 225.00 to 399.95 megacycles. The command radio equipment consists of a receiver-transmitter, a control panel, a remote indicator, and an automatic direction finding (ADF) unit. The UHF antenna is located on the bottom of the fuselage forward of the missile bay door. The control panel is located on the left console and permits selection of 20 preset channels, manual selection of any frequency in the operating range, or selection of a preset guard frequency.

NOTE

UHF control transfer switches on the transfer control panels permit control of the UHF radio to be transferred to either cockpit.

Guard frequency range is 238.00 to 248.00 megacycles. A tone is heard in the earphones when the set is tuning to another frequency. The remote indicator indicates the preset channel, the manual frequency, or guard frequency. Normal power for the command radio is provided by the MA-1 electrical power system. If MA-1 power fails, the radio automatically transfers to the airplane ac and dc essential buses.

UHF Mode Selector Switch. The UHF mode selector switch, located on the UHF control panel (figure 4-14), is used to select the desired operating mode. In the PRESET position, any of the 20 preset frequencies can be selected. In MAN position, manual frequencies can be selected. Placing the switch to GUARD position causes the receiver-transmitter to tune to the preset guard frequency of 243.00 megacycles when the function selector switch is in either BOTH or MAIN position. The force required to place the mode selector switch in GUARD position is approximately three times greater than that required for the other selections. This reduces the possibility of inadvertent GUARD selection.

UHF Function Selector Switch. The function selector switch on the UHF control panel (figure 4-14) has placarded positions of BOTH, MAIN, and OFF. In the OFF position, the radio and the remote indicator are inoperative. In the MAIN position, the radio operates on the selected channel or frequency. The BOTH position permits transmission and reception on the selected channel or frequency and simultaneous reception on guard frequency.

UHF Squelch Disable Button. The springloaded squelch disable button, located on the UHF control panel (figure 4-14), is placarded "Sq Disable" and is used to disable receiver squelch circuits. Receiver squelch circuits prevent the receiver from producing an audio output in the absence of signals of a predetermined character. Depressing the squelch disable button allows receiver audio to be heard in the earphones. This provides a check of receiver operation.

UHF Remote Indicator. The UHF remote indicator (41, figure 1-8 and 36, figure 1-9) is located on the instrument panel and is labeled UHF. With the mode selector switch in PRESET, the selected preset channel appears in the remote indicator. When the mode switch is in MAN position, the five-digit manual frequency is displayed. Guard frequency is displayed when the mode switch is in the GUARD position.

Operation of UHF Command Radio

1. Place MA-1 power switch to WARM, RADAR STBY, or ON.
2. Place UHF function selector switch to BOTH position and allow approximately two minutes warmup time.
3. Ⓑ Check UHF-TACAN control transfer monitor light illuminated. If light is not illuminated, depress UHF-TACAN transfer button and recheck light illuminated.
4. Select PRESET with UHF mode selector switch.
5. Select desired channel with the UHF preset channel selector switch.
6. Adjust the UHF volume control knob for desired audio level.

NOTE

This places the radio in an operating condition. When changing channels, wait approximately four seconds between channel selection and transmission. Continuous transmission should not exceed five minutes.

7. To operate on a manually selected frequency:
   a. Select the desired operating frequency using the manual frequency selector switches.
   b. Place the mode selector switch to MAN position.
8. To transmit and receive on guard frequency, place the mode selector switch to GUARD position.

NOTE

No transmission shall be made on emergency (distress) frequency channels except for emergency purposes. For test, demonstration, or drill purposes, the radio equipment shall be operated in a shielded room to prevent transmission of messages that could be construed as actual emergency messages.

9. To turn off the UHF command radio, place the UHF function selector switch in OFF.

Ground Operation Of The UHF Radio With External Power

1. External power connected.
2. MA-1 power switch - OFF.
3. Refrigeration unit switch - OFF.
4. UHF function selector switch - BOTH and allow approximately 2 minutes for warmup.
5. UHF radio frequency - set.

NOTE

The following limitations must be observed in order to prevent overheating of equipment:

a. The transmitter duty cycle shall not exceed 17%: i.e., the transmitter shall not be operated for more than 2.5 minutes during a 15 minute operating period.

b. Total UHF radio ON time shall not exceed 60 minutes.

UHF and TACAN Control Transfer Buttons and Control Monitor Lights Ⓑ

The UHF and TACAN control transfer buttons (figure 4-15), one on the forward and one on the aft transfer control panel, are placarded "UHF Comm TACAN." The buttons are used to shift control of the command radio and the TACAN to the forward or aft control panels. A monitor light in the switch housing illuminates in the cockpit having control. Control is transferred by depressing the button. For example: if control is at the forward panels, as indicated by illumination of the forward monitor light, depressing the aft UHF control transfer button will shift control to the aft control panels and illuminate the aft control monitor light. The control transfer buttons and control monitor lights receive power from MA-1 electrical power supply system. When the MA-1 power switch is placed in EMER position, control is automatically transferred to the forward control panels.

Automatic Direction Finder

The automatic direction finder provides an indication of the bearing of transmitting stations picked up on the UHF command radio. A bearing of GCI transmitting stations can also be received through the data link receiver. The station bearing is indicated by the HSI bearing pointer (airplanes with the integrated flight instrument system) and the TSD command heading pointer (airplanes with the conventional instrument display). The automatic direction finder is on and in standby at any time the command radio is on. On airplanes with the conventional instrument display, controls for the ADF are located on the tactical situation display indicator. On airplanes with the integrated flight instrument system, the controls are incorporated as part of the bearing selector switch. The antenna is located under the fuselage just forward of the nose wheel well. Power is supplied to the automatic direction finder from the MA-1 electrical power supply system. In the event of MA-1 electrical power failure, the automatic direction finder will automatically transfer to the airplane ac essential bus.

Operation of Automatic Direction Finder

NOTE

For warmup time and frequency selection details, refer to OPERATION OF COMMAND RADIO, this Section.

1. TSD mode selector switch — ADF-CMD (some airplanes).

   Bearing selector switch — UHF-ADF or DL-ADF (other airplanes).

2. Select desired command radio frequency.

   Station bearing will be indicated by one of the following methods:

   a. By the grid pointer on the TSD. An arrow on the diametral line of the TSD indicates the bearing on the compass on airplanes with the conventional instrument display.

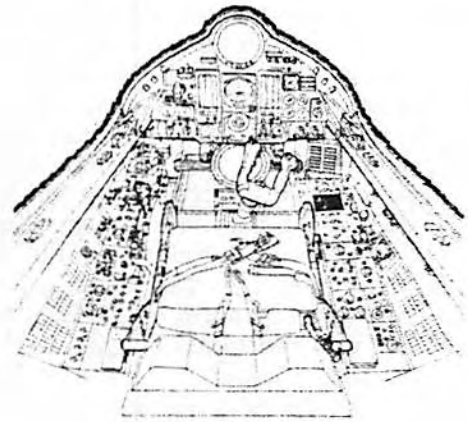
   b. By the bearing selector pointer on the HSI on airplanes with the integrated flight instrument system.

3. The command ADF feature is turned off by changing to any position other than an ADF selection on the TSD mode selector switch or the bearing selector switch (as applicable).

NOTE

- Bearings of the TSD grid pointer are repeated by the HSI bearing pointer.
- UHF audibility will be degraded when using ADF position.
- On B airplanes, ADF function is inoperative unless control of both UHF and AFCS is in the same cockpit.

tacan control panel



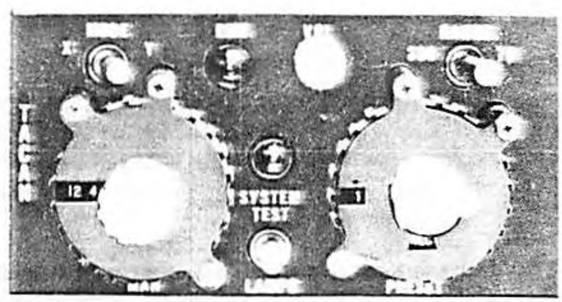


Figure 4-16

TACAN

The TACAN (Tactical Air Navigation) equipment operates in conjunction with a surface navigation beacon or with the TACAN equipment in a similarly equipped airplane. The system enables the airplane to receive continuous indications of range and bearing from a fixed TACAN beacon or line-of-sight distance only to a cooperating airplane within a maximum range of 300 nautical miles. TACAN equipment consists of a receiver-transmitter unit, a control unit, and a signal data converter. Any one of 23 preset TACAN channels or 126 manually selected channels may be selected in each of two operating modes (X and Y). Three TACAN operating functions may be selected: receive, transmit/receive, and air-to-air. In the receive function, TACAN provides continuous bearing information to a fixed ground beacon. In transmit/receive function, both bearing and distance to the selected ground beacon are provided. In air-to-air function, TACAN receives range information from a cooperating airplane and serves as a transponder to a maximum of five cooperating airplanes. In the air-to-ground functions (receive and transmit/receive) when one of the 23 preset TACAN channels is selected, a map of the geographical area surrounding the TACAN beacon is presented on the tactical situation display (TSD). The displays of the 23 TACAN areas are preset and controlled by the MA-1 computer. For each preset channel there is a choice of either a 50-, 200-, or 400-mile radius scale map of the specific area. The scale presented is governed by the position of the radius miles selector switch on the TSD. When a TACAN station is selected, the digital computer receives bearing and range signals from the station and converts the signals into MA-1 grid coordinates of the interceptor. An interceptor projected on, and moving relative to the map display, provides a direct readout of TACAN bearing and slant range. Manual selection is used to select one of the 126 TACAN channels regardless of location. Operation on a manual channel does not present a map of the transmitter area on the TSD. Instead, manual selection provides a choice of any one of three indexes and test maps (see MAP SELECTION and RADIUS MILES SELECTOR SWITCH, this Section). An annunciator on the TSD presents indications of the validity of the TACAN display.

NOTE

TACAN normally receives power from the MA-1 power supply. In event of MA-1 power failure, the TACAN will automatically receive power from the airplane power supply.

TACAN Function Selector Knob and Indicator Window. The function selector knob is located on the TACAN control panel (figure 4-16). It is the smaller of the two controls on the right of the panel. The knob has four positions which are displayed in the indicator window below the knob. The positions in clockwise order are OFF, REC, T/R, and A/A. In the OFF position, power is removed from the set. In REC (receive), TACAN receives bearing data from a ground beacon. In T/R (transmit/receive), distance and bearing data are available from a ground beacon. In A/A (air-to-air), TACAN supplies line-of-sight distance to a cooperating airplane. To cooperate with another airplane in the air-to-air function, both airplanes must have A/A selected and the channels selected must be 63 channels apart. Cooperating airplanes must operate in the same mode (X or Y).

Preset TACAN Channel Selector Knob. The preset channel selector knob, placarded "Preset," is located on the TACAN control panel (figure 4-16). It is the larger of the two knobs on the right of the panel. The knob is used to select one of the 23 preset TACAN channels. A 24th position, labeled M, is used when manual channel selection is desired. As the preset channel selector knob is rotated, the switch position is displayed in a window on the left side of the knob.

Manual TACAN Channel Selector Controls. The manual channel selector controls, placarded "Man," consist of two concentric knobs on the left of the TACAN control panel (figure 4-16). The knobs are used to select one of the 126 manual TACAN channels available. The selected channel is displayed in a window on the left side of the controls. The larger (outer) knob controls positions 1 through 12 with a blank for zero. These selections are displayed as the first or first and second digits in the indicator window. The smaller knob controls positions 0 through 9 which appear as the last digit in the window.

TACAN Volume Control Knob. The volume control knob (figure 4-16) is placarded "Vol." It is used to control the audio level of the TACAN station identification signal.

TACAN Mode Selector Switch. The mode selector switch, placarded "Mode," is located on the TACAN control panel (figure 4-16) and has positions X and Y. In the X position, the 23 preset or 126 manual channels of the X-mode may be selected. In the Y position, 23 preset or 126 manual channels in the Y-mode may be selected. The switch is effective when using manual TACAN channel selection and in the air-to-air function. (When operating on preset channels, the TACAN equipment automatically selects the X or Y mode.)

NOTE

Y mode is not operational for ground stations.

TACAN Range Selector Switch. The range selector switch is located on the TACAN control panel (figure 4-16) and is placarded "Range." It has positions 300 and 70. In the 300 position, the TACAN operates over the full 300-mile range. Placing the switch in the 70 position prevents the processing of range information from stations beyond the 70 miles range.

NOTE

Selecting the 70 position when TACAN distance is greater than 70 nautical miles will not cause the range window on the HSI to shutter. Displayed range will normally be erroneous.

TACAN System Test Switch. The system test switch on the TACAN control panel (figure 4-16) has placarded positions of SYSTEM TEST and LAMPS and is spring-loaded to the center (off) position. The switch provides a go, no-go test of the TACAN. Holding the switch in the SYSTEM TEST position activates self test circuits in the receiver transmitter causing the following self-test sequence: system test indicator light illuminates and an audible tone is heard; system test light goes out and audible tone ceases; bearing pointer drives to 069°; range indicator drives to zero.

NOTE

If TACAN is in the A/A mode when the switch is placed in SYSTEM TEST, the system test indicator lamp will flash erratically unless there is another aircraft cooperating in A/A mode.

Holding the switch in LAMPS position causes the ECM indicator light and the system test indicator light to illuminate as a check of lamp operation.

TACAN System Test Indicator Light. The system test indicator light on the TACAN control panel (figure 4-16) is located immediately above the system test switch. The amber light illuminates and remains illuminated if a no-go condition exists when the system test switch is held in the SYSTEM TEST position. The amber light also illuminates when TACAN is turned on and goes out when warmup of the set is completed. Light operation is checked by holding the TACAN system test switch in the LAMPS position.

TACAN ECM Indicator Light. The ECM indicator light on the TACAN control panel (figure 4-16) is placarded "ECM." The amber light illuminates when a TACAN jamming signal is detected by the decoder in the receiver-transmitter. Light operation is checked by holding the TACAN system test switch in the LAMPS position.

TACAN-Command Altitude Switch-

On airplanes with the conventional instrument display, a TACAN-command altitude switch, located above the right console, is used to switch the function of the command altitude indicator to a TACAN range indicator. The switch has two positions, COMM ALT and TACAN RANGE. When the switch is in the COMM ALT position, the command altitude indicator will display command altitude as provided from data link information. With the switch in the TACAN RANGE position, a green indicator light will illuminate on the instrument panel and the command altitude indicator will display distance in miles of the airplane from a selected TACAN station. For optimum performance of the autonomous navigation and dead reckoning functions, the switch should be placed in the COMM ALT position when TACAN range is not required.

TACAN Range Indicator Light

On airplanes with the conventional instrument display, a green indicator light, located on the instrument panel, will illuminate when the TACAN-command altitude switch is in the TACAN RANGE position. The green indicator light, placarded "TACAN Range," shows that the command

command and target altitude indicator

Figure 4-16A

altitude indicator is functioning as a TACAN distance indicator.

TACAN Annunciator

The TACAN annunciator (figure 4-20) is located on the TSD. The TACAN annunciator indicates the condition of the interceptor information on the TSD. Five displays are possible on the annunciator:

a. "OK"—Valid TACAN information is being received. This consists of range and bearing information used to position the interceptor symbol on the TSD (indirectly by the digital computer), to compute wind, and to make dead reckoning computations.

b. "VR"—Valid range is being received and displayed (HSI). If valid DR position is available, the digital computer will DR the course deviation indicator. If computer DR becomes invalid, the CDI will center in line with the course pointer. TSD interceptor symbol position is being dead reckoned by the computer.

c. "BV"—Valid bearing is being received and displayed (HSI). If valid DR position is available, the digital computer will DR range and the range counter on the HSI will be unmasked. If computer DR becomes invalid, the range counter will be masked. TSD interceptor symbol position is being dead reckoned by the computer.

NOTE

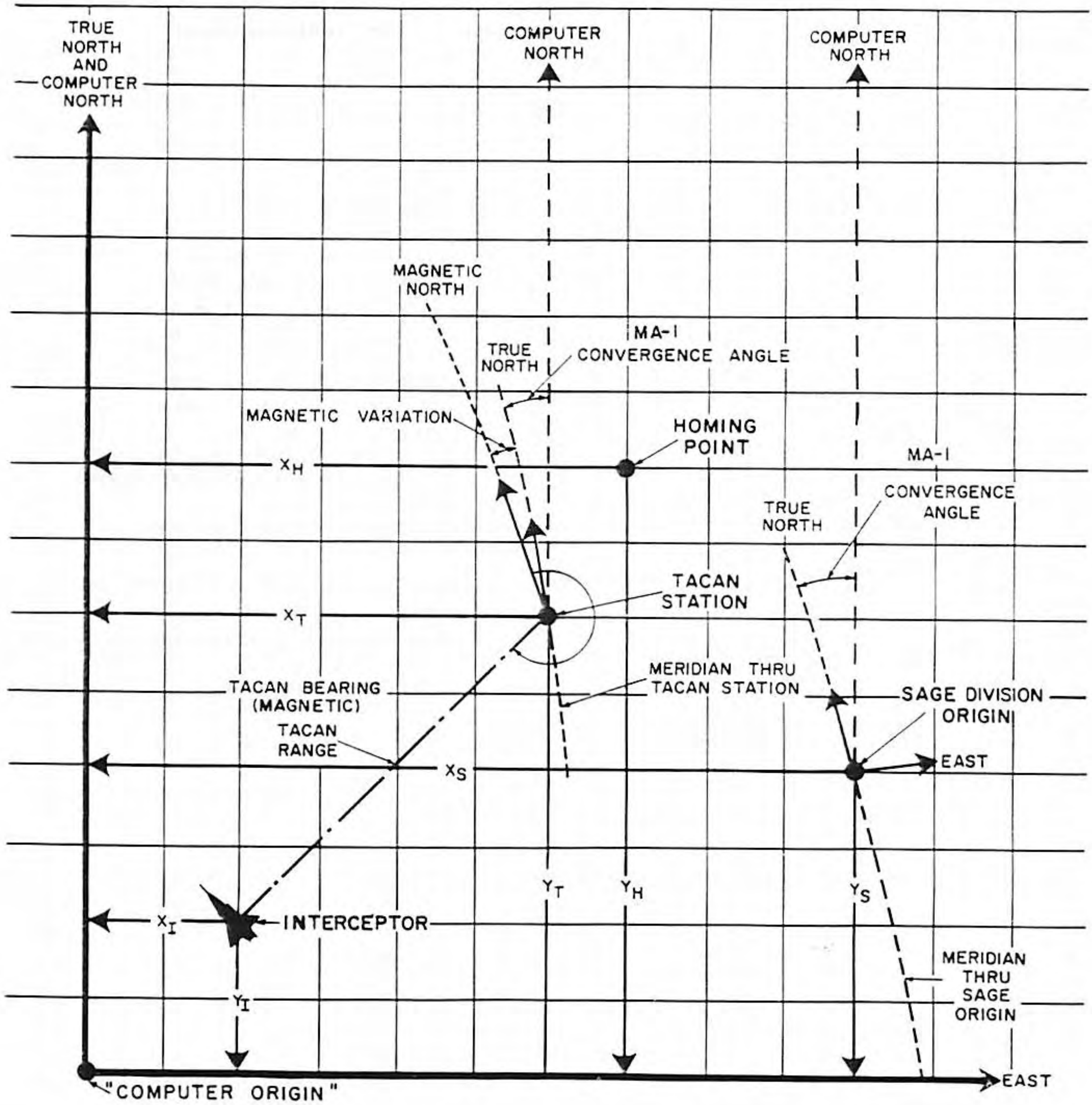
With the display of VR (or BV) the display of bearing (or range) may be dead reckoned from previously received TACAN (OK has been displayed during flight) or from initial position (OK has not been displayed during flight). If it is desirable to determine which is the case, TACAN reception of a valid signal may be interrupted and the display of DR or OFF checked.

d. "DR"—Range and bearing are not being received and interceptor position and HSI range and bearing are being dead reckoned by the digital computer from the last valid TACAN data, using the last wind computation.

e. "OFF"—Reliable TACAN information has never been received.

Valid TACAN must be received to obtain a valid annunciator indication: both range and bearing to display OK; valid range only to display VR and bearing valid only to display BV. Prior to TACAN reception, the TACAN annunciator indicates OFF. However, interceptor position is being dead reckoned by the computer from initial position. If BV is the first indication of TACAN reception, valid bearing is available on the HSI and range (HSI) and interceptor position (TSD) are being dead reckoned from initial position by the computer. The converse is true if VR is the first indication. If this first reception of range or bearing is subsequently lost prior to an OK indication, OFF will again be displayed. When both range and bearing are received simultaneously, OK is indicated. If TACAN is lost after an OK indication DR will be displayed until range or bearing on both are received. Once an OK is obtained, OFF will not be displayed again until the flight is terminated (MACH less than 0.25 and ILS selected). When BV or VR is displayed, the use of the valid information on the HSI should be considered when utilizing the dead reckoned position of the interceptor on the TSD, since the computer will DR from initial position or from last valid TACAN position without consideration of the amount of time received or time since last received.

MA-1 computer grid reference system



LEGEND

TACAN X_T, Y_T - "COMPUTER" COORDINATES OF TACAN STATION

HOMING POINT X_H, Y_H - "COMPUTER" COORDINATES OF HOMING POINT.

GROUND ENVIRONMENT X_S, Y_S - "COMPUTER" COORDINATES OF SAGE DIVISION ORIGIN

Figure 4-16B

8. Bearing selector switch—TAC (airplanes with integrated flight instrument system).

   TACAN-command altitude switch—TACAN RANGE (airplanes with conventional instrument display).

Operation of TACAN Equipment

1. MA-1 power switch—WARM, RADAR STBY, or ON; power annunciator displaying "OK."
2. TACAN function selector knob—As required.

NOTE

When A/A is selected, the bearing pointer is meaningless and the TACAN annunciator initially displays VR. For eight seconds after A/A selection, it is possible that range information from the last selected TACAN ground site is still being presented. After eight seconds, if valid air-to-air range is not available, the TACAN annunciator displays DR. If valid air-to-air range is available, the VR annunciator display will be retained.

3. TACAN mode selector switch—X or Y as required.
4. TACAN range selector switch—300 or 70 as desired.
5. Preset TACAN channel selector knob—Set to desired channel or M (depending on whether preset or manual selection is desired).

6. Manual TACAN-channel selector knobs — Set to desired channel (if manual selection is desired and the preset channel selector knob is in the M position).
7. TACAN volume control knob — Adjust to desired level and identify station.

NOTE

TACAN may lock onto the wrong station. Station identification should be accomplished when making a channel selection.

8. TACAN ECM indicator light — Out.
9. Lamp and system test (if desired):
   a. TACAN system test switch — LAMPS; check ECM indicator and system test indicator lights illuminated.
   b. Select channel that is beyond receiver range.
   c. TACAN system test switch — SYSTEM TEST.
   d. System test light on; audible tone in earphones.
   e. System test light out; audible tone off.
   f. Bearing pointer to 069°.
   g. Range indicator to zero.
   h. TACAN system test switch — Release.

NOTE

System test in air-to-air function provides a test of range reception only.

10. TSD scale selector switch — Set as desired.
11. TSD light intensity rheostat and red filter switch — Set as desired.
12. TACAN annunciator — Check to determine availability of TACAN signals.
13. Bearing selector switch — TAC (airplanes with integrated flight instrument system).
14. TACAN-command altitude switch — TACAN RANGE (airplanes with conventional instrument display).

All data on pages 4-40 thru 4-40A, including figures 4-16C thru 4-16D deleted.

auto nav homing point selector

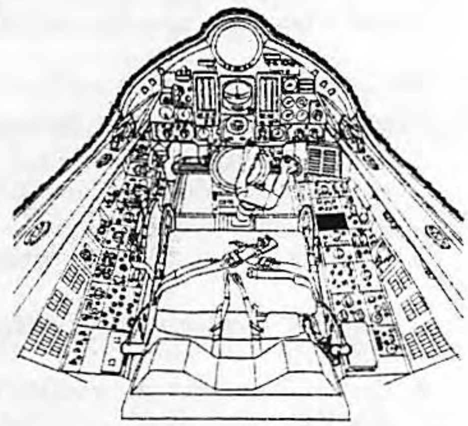


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Figure 4-17

ILS (Instrument Landing and Approach System)

The ILS equipment consists of a glide-slope-marker beacon receiver, a localizer receiver, channel selector, a volume control, a course indicator and an approach horizon. Indications displayed on the course indicator or approach horizon give both vertical and horizontal guidance for an instrument landing system approach. The proper glide-slope and localizer frequency are selected when the ILS channel selector switch is set at the preset ILS channel. The airplane can be manually flown on an ILS approach, or the automatic flight control system can be selected which will automatically fly the airplane down the approach path in response to ILS signals. The ILS equipment is operable at all times in normal flight, but does not supply steering signals to the automatic flight control system unless the flight mode selector switch is in the AUTO position and the automatic mode selector switch is in the ILS position. The ILS equipment receives power from the MA-1 electrical power supply system.

NOTE

With the ILS volume control knob turned up, headset noise may be mistaken for faulty UHF reception.

ILS Channel Selector Switch. An ILS channel selector switch (figure 4-18) is located on the left console. The switch is placarded "Channel Selector" and has 20 positions. Each position represents a preset ILS channel. Power is supplied from the MA-1 electrical power supply system.

ILS Volume Control Knob. A volume control knob is located adjacent to the ILS channel selector switch. The knob varies the level of the localizer audio signals delivered to the headset. The volume control knob receives power from the MA-1 electrical power supply system.

Operation of ILS Equipment

WARNING

Some UHF and data link frequencies create radiation and conduction interference in the ILS. To avoid the possibility of receiving and using erroneous ILS information, do not transmit on UHF frequencies 226.6 mc through 230.2 mc and 247.3 mc through 250.9, or attempt to receive on UHF or data link frequencies 256.2 mc through 263.8 mc during an ILS approach.

1. Conventional instrument display:
   a. MA-1 power switch – WARM, RADAR STBY, ON, or RADIO SILENCE.
   b. ILS channel selector switch — Set to desired channel.
   c. ILS volume control knob — Adjust.

NOTE

- Appearance of the "LOC" (localizer) warning flag in the approach horizon indicates that localizer information is invalid on both the approach horizon and the course indicator.
- Disappearance of the "LOC" warning flag in the approach horizon indicates only that the localizer receiver is receiving a valid signal. In order for valid ILS localizer information to be displayed in the cockpit, continue with the following procedures.

   d. Display automatic mode selector switch — ILS.
   e. Approach horizon function selector knob — ILS.

      Select the ILS position to permit proper operation of the bank steering bar on the approach horizon.

WARNING

If ILS is not selected on the approach horizon selector knob, the information reflected by the course deviation indicator will be from the TACAN rather than the ILS station selected.

f. Heading marker — Set to ILS final approach course.

Set the heading marker on the course indicator to the ILS final approach course to permit correct heading information to be fed to the bank steering bar on the approach horizon.

g. Course arrow — Set to ILS final approach course.

Set the course arrow on the course indicator to the ILS final approach course to permit the course deviation indicator to be displaced in the proper direction.

NOTE

Although it is not necessary to set the course arrow to the ILS final approach course to obtain signal display, it is desirable to do so to provide easy interpretation of left or right deviations from the selected ILS course.

h. "LOC" warning flag—Not visible.

2. Integrated flight instrument system:

For operation of ILS equipment on airplanes with the integrated flight instrument system, refer to INTEGRATED FLIGHT INSTRUMENT SYSTEM, Section VII.

ils channel selector

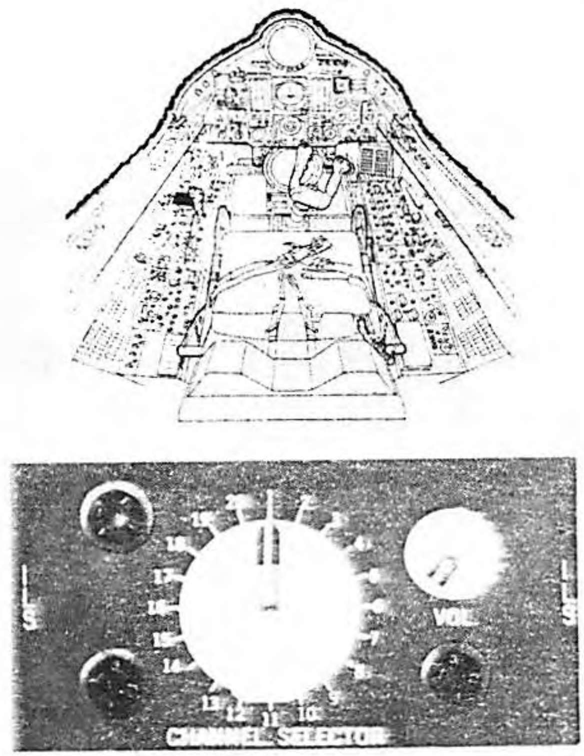

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Figure 4-18

All data on page 4-42 deleted.

IFF/SIF RADAR SYSTEM (AN/APX-72)

The IFF/SIF Radar System is used to automatically identify the aircraft in which it is installed whenever it is properly challenged by suitably equipped air or surface forces. The set has provisions for identifying the aircraft in which it is installed as a specific friendly aircraft within a group of other aircraft. The set has a provision for special signal transmission in an emergency or whenever desired. The set receives challenges and transmits replies to the source of the challenges which are displayed on the radar indicators of the challengers. It also is capable of transmitting aircraft pressure altitude information. The range of the set is line-of-sight. All operating controls are on the left hand console. An ejection relay is actuated if the aircraft canopy is unlocked with the nose wheel in the up position. Regardless of how the APX-72 controls are set, energizing the ejection relay will: (1) switch the APX-72 from low to high sensitivity, (2) enable MODES 1, 2, and 3/A, (3) set the APX-72 in emergency mode of reply, and mode 3/A to code 7700, (4) zeroize mode 4 Codes A and B. The unit receives power from the MA-1 electrical power supply. In the event of MA-1 electrical power supply failure, unit power will automatically be supplied from the AC and DC essential busses.

IFF/SIF MASTER CONTROL KNOB

The IFF/SIF master control knob (7, figure 4-19A) is a five-position, rotary switch. The knob is labeled MASTER and it controls the following functions: the OFF position removes power from the set; the STBY position applies power but does not enable the receiver; the LOW position, selected only upon order of a controlling agency, enables the receiver with reduced sensitivity so that only strong local interrogations are answered; the NORM position enables the receiver with full sensitivity so that reply may be made at maximum range. The EMER position is selected by pulling out on the knob before rotating. When actuated, this mode automatically squawks an emergency reply to interrogation of Mode 1, 2, and 3/A. To disable EMER, rotate the knob out of EMER position. The rotary master knob must be pulled out and rotated to OFF to remove power from the set.

MODE ENABLE/TEST SWITCHES

Modes 1, 2, 3/A and C have three-position toggle switches (2, 4, 6 and 8, figure 4-19A) with an OUT, a center ON, and a spring-loaded TEST position. Placing one of these switches in the ON position will activate the mode associated with the switch. Proper operation of a mode is indicated when the test lamp illuminates while the mode enable/test switch is held in the TEST position. Mode C (Altitude Reporting) enables the APX-72 to report the aircraft pressure altitude to the interrogating agency by the use of a specified 11 bit code through the aircraft air data computer.

MODE 1 CODE SELECT WHEELS

The Mode 1 Code Select Wheels (13, figure 4-19A) are used to set the desired two-digit mode 1 reply code. The wheels are labeled MODE 1.

MODE 3/A CODE SELECT WHEELS

The Mode 3/A Code Select Wheels (11, figure 4-19A) are used to set the desired four-digit mode 3/A reply code. The wheels are labeled MODE 3/A.

IDENTIFICATION OF POSITION ENABLE SWITCH

The switch labeled IDENT-OUT-MIC (10, figure 4-19A) is a three-position toggle switch. OUT position disables the ident feature. At the request of the controlling agency, this switch is momentarily pressed to the IDENT position. The switch spring returns to the OUT position, but the set will initiate a special identification reply for 15 to 30 seconds on the mode and codes selected.

With the switch in the MIC position, the set will initiate the special identification on the mode and codes selected each time the UHF transmitter button is depressed. Codes C and 4 are not affected by this switch.

NOTE

> In the event the IP/MIC switch is in the MIC position and the UHF transmitter is keyed, the transmitter (APX-72) may continue sending after the key is released. Should this occur, the IP/MIC switch should be manually moved to the OUT position to cease transmission.

RADAR TEST/MONITOR ENABLE SWITCH

The switch labeled RAD TEST-OUT-MON (9, figure 4-19A) is a three-position toggle switch. The OUT position disables all functions of this switch. The MON position causes the test indicator light to illuminate when a reply is made to an interrogation of any selected mode. The RAD TEST position is used for ground test operation and requires additional equipment.

TEST INDICATOR LIGHT

The Test Indicator Light (5, figure 4-19A) is labeled TEST and illuminates to indicate proper operation of the IFF/SIF modes when a mode enable switch is held at TEST. This light also illuminates when the radar test/monitor switch is at MON, the master switch is at NORM and the set is replying to an interrogation of any selected mode. It is a green press-to-test type light that may be dimmed for night operation.

MODE 4

Mode 4 is a military secure mode. Controls and indicators for mode 4 are located on the left side and top of the APX-72 control unit (1, 3, 14, and 15, figure 4-19A) outlined by a white line. Mode 4 interrogation is received by a special transponder-computer which encodes and triggers a proper identification response signal. The master rotary switch controls transponder action in all modes. When the Mode 4 Enable Switch is selected ON, mode 4 will operate normally in either NORM or EMERG position of the master control knob and at a reduced receiver sensitivity in LOW. Mode 4 is inoperative in either STBY or OFF position of the master control knob. To operate mode 4, the Mode 4 Enable Switch must be in the ON position, special codes must be inserted into the system, and code A or B selected. Should Mode 4 fail to reply to a valid interrogation, the IFF caution light will illuminate.

IFF CAUTION INDICATOR LIGHT

The amber colored IFF Cauiton Light is located on the pilot's forward right hand console adjacent to the oil pressure gage. It illuminates to alert the pilot that the APX-72 has failed to reply to a valid mode 4 interrogation provided: (1) the aircraft power is on, and (2) the IFF master control knob is not OFF. The IFF Caution Light circuitry monitors for: (1) mode 4 code zeroized, (2) transponder failure to reply to proper interrogation, and (3) automatic self-test function of the computer reveals a computer malfunction. Should the IFF caution light illuminate, check IFF master control knob in NORM, mode 4 Enable Switch in ON, and mode 4 code selector knob in proper A or B code position for current code time period. If light remains illuminated, avoid operation in a known mode 4 interrogating environment or if already in one, take appropriate corrective or emergency action as operationally directed for this condition (inoperative mode 4).

MODE 4 ENABLE SWITCH

The Mode 4 Enable Switch (14, figure 4-19A) is provided for control of mode 4 operation. It is labeled ON and OUT. It is a positive action switch which must be pulled out to be placed ON or OFF.

MODE 4 REPLY INDICATION SELECT SWITCH

The Mode 4 Reply Indication Select Switch (15, figure 4-19A) is a three-position toggle switch with a LIGHT, an OUT (center) and AUDIO position. When the switch is placed in the LIGHT position, only the reply light (3, figure 4-19A) of mode 4 reply is enabled. The AUDIO position enables both the reply light and aural indication. The aural indication in the pilot's headset is not adjustable by the pilot. With the switch in AUDIO position, an aural signal indicates mode 4 interrogations are being received and illumination of the Mode 4 Reply Light indicates replies are transmitted. In the OUT position, both light and audio indications are inoperative. This switch must be in either the AUDIO or LIGHT position when operating mode 4.

REPLY INDICATOR LIGHT

A green Mode 4 Reply Indicator Light (3, figure 4-19A) is provided to indicate mode 4 replies are being transmitted when the mode 4 reply indication select switch is either in the AUDIO or LIGHT position.

MODE 4 CODE SELECTOR KNOB

The Mode 4 Code Selector is a four-position (HOLD, A, B, and ZERO) rotary knob (1, figure 4-19A). A and B codes are preset daily as operationally directed by the single insertion of a Code Changer Key. A is the present code and B is the next succeeding code thereby enabling the set to properly reply to any valid mode 4 interrogation during a given time period. The ZERO position zeroizes the code setting. Both codes are normally zeroized when the master switch is turned to OFF after the aircraft has landed. If a second flight is anticipated during the proper time periods, the code settings may be retained by selecting the HOLD position of the knob. The HOLD position is spring-loaded to return to the A position. To hold codes, the knob must be held momentarily (2-3 seconds) to the HOLD position before power is removed from the transponder. Allow transponder power to remain on for at least 15 seconds after the knob is released, and then turned off as desired. The code setting is now mechanically latched and will be retained when aircraft power is turned off.

NOTE

- To hold the code setting, the aircraft landing gear must be DOWN and LOCKED. The hold feature will remain in effect until the aircraft landing gear is retracted.
- If power is removed from the transponder less than 15 seconds after selecting HOLD, either by turning the transponder off or by turning off aircraft electrical power, the code setting will zeroize when transponder power is lost.

Both A and B codes may be zeroized any time the aircraft has electrical power on and the APX-72 Master Knob is in any position except OFF by placing the Code Selector Knob to the ZERO position. The Code Selector must be pulled out before it can be turned to the ZERO position. Any time (in flight or on the ground) the IFF Master Control Knob is placed in the OFF position, the A and B codes are zeroized unless the HOLD function has been properly actuated. Use Code A or Code B as operationally directed.

OPERATION OF TRANSPONDER

1. MA-1 Power Switch - RADAR STBY, WARM, EMER, or ON and power annunciator displaying "OK."
2. MASTER CONTROL KNOB – STBY (warm up for 2 minutes).
3. MODE ENABLE/TEST SWITCH – ON (desired modes).
4. MODE CODE CONTROL WHEELS – CODE SET as desired.
5. RAD TEST/MON SWITCH – As desired.
6. MASTER CONTROL KNOB – NORM (before Take-off).
7. MODE ENABLE/TEST SWITCH – Hold desired Mode Switch to TEST until test light illuminates. If light does not come on, the selected mode is inoperative. (There is no test for mode 4).

TO OPERATE MODE 4

(IFF MASTER CONTROL KNOB LOW, NORM, OR EMERG)

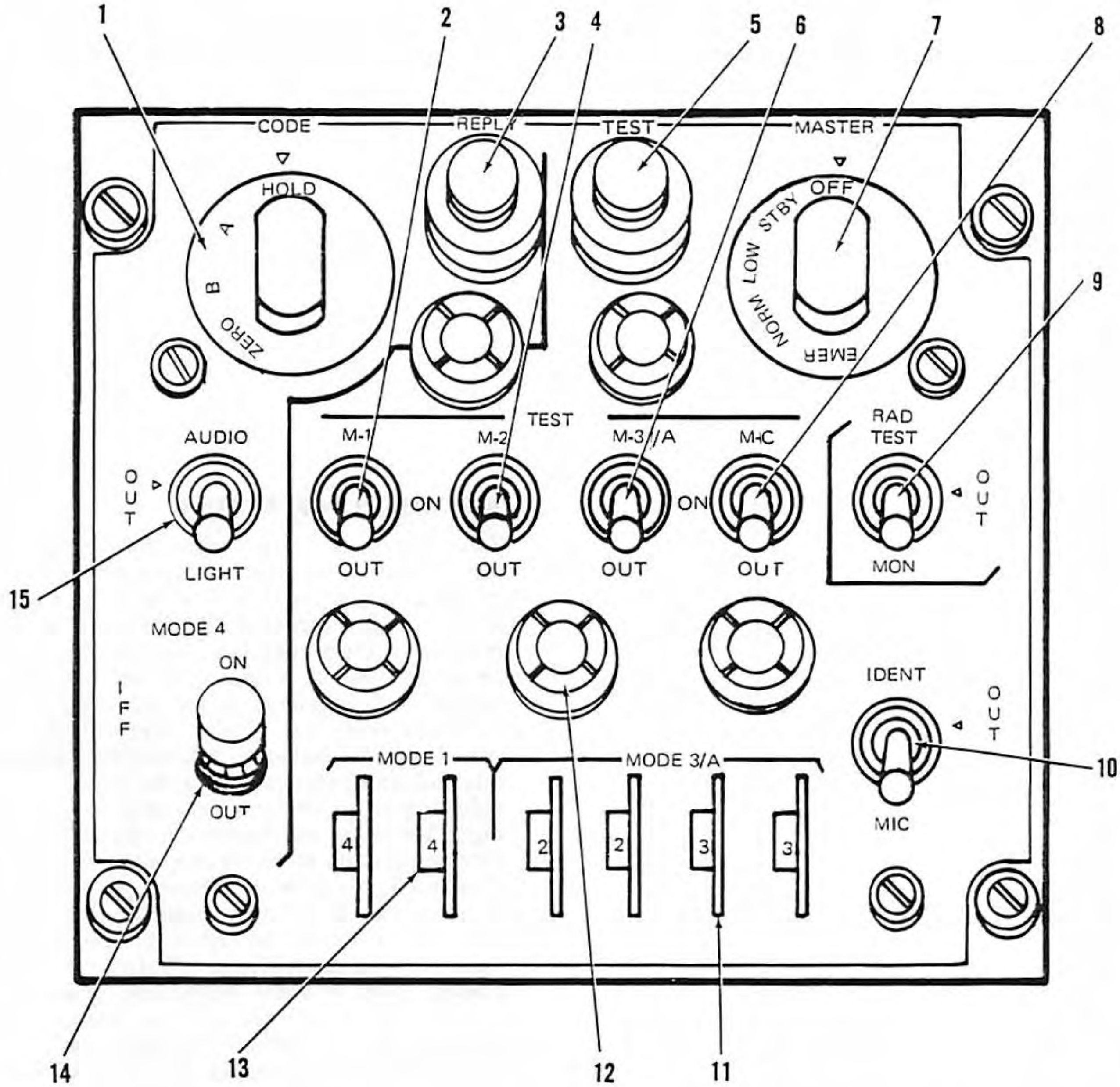
8. MODE 4 ENABLE SWITCH – ON.
9. MODE 4 REPLY/INDICATION SELECT SWITCH – AS DESIRED.
10. MODE 4 CODE CONTROL KNOB – A OR B (AS REQUIRED FOR TIME PERIOD).

Emergency Operation of AN/APX-72 IFF System

For emergency operation, pull upward on the master control knob and rotate to EMER position. Modes 1, 2, and 3/A are enabled automatically. Modes 1 and 2 will reply to interrogations with the reply code selected by their respective code switches. Mode 3/A reply code will be 7700 regardless of the position of the mode 3/A code switches.

All data on page 4-42D including figure 4-19 deleted.

IFF/SIF RADAR SYSTEMS CONTROL PANEL AN/APX-72



1. MODE 4 SELECTOR KNOB
2. MODE 1 ENABLE/TEST SWITCH
3. REPLY INDICATOR LIGHT
4. MODE 2 ENABLE/TEST SWITCH
5. TEST INDICATOR LIGHT
6. MODE 3/A ENABLE/TEST SWITCH
7. IFF/SIF MASTER CONTROL KNOB
8. MODE C ENABLE/TEST SWITCH
9. RADAR TEST/MONITOR ENABLE SWITCH
10. IDENTIFICATION OF POSITION ENABLE SWITCH
11. MODE 3/A CODE SELECT WHEELS
12. PANEL LAMP
13. MODE 1 CODE SELECT WHEELS
14. MODE 4 ENABLE SWITCH
15. MODE 4 REPLY INDICATION SELECT SWITCH

Figure 4-19A.

All data on pages 4-44 thru 4-53, including figures 4-20 thru 4-23 deleted.

AIR REFUELING SYSTEM

Some airplanes are equipped with an air refueling system which permits airborne refueling of internal and external fuel tanks. The system uses the "boom and receptacle" refueling method. This method requires that the airplane be flown to a prescribed in-trail formation position and that the tanker boom be placed in the refueling receptacle for transfer of fuel. Fuel is supplied under pressure from the tanker to internal fuel tanks or to internal and external tanks as selected. Ground refueling transfer lines are used for air refueling. The refueling receptacle (figure 1-1) is located on the top of the fuselage aft of the cockpit. The receptacle area is enclosed by a door which is hinged on the forward end and opens downward at the aft end to expose the receptacle. The open door forms a slipway which guides the refueling boom into the receptacle. The receptacle area is lighted when the slipway door is open. The slipway door is normally actuated by secondary hydraulic system pressure. In event of secondary hydraulic system failure, high pressure pneumatic system pressure is supplied to the door actuator through a shuttle valve to provide emergency door opening. When the refueling boom nozzle is inserted into the receptacle, hydraulically actuated latches lock the nozzle in place. Upon completion of refueling, the boom latches are retracted and the nozzle is released. The latches may be released automatically by a pressure switch when the fuel tanks are full, by cockpit controls, or by a signal from the tanker. In event of secondary hydraulic system failure, there is no provision for emergency operation of the boom latches. However, fuel can be transferred by holding the boom nozzle firmly against the receptacle. The boom latches normally lock automatically when the boom nozzle is inserted. In event of electrical malfunction within the refueling system, a manual boom latching (MBL) feature is provided. When the air refueling system is activated, fuel system pressure valves for F and T tanks Ⓐ or F tank Ⓑ close allowing pressure in these tanks to relieve to 3 psi. This causes the respective fuel valve closed warning lights to illuminate. Simultaneously external tank pressurization is shut off and external tanks are vented to ambient pressure. This allows all tanks to be refueled as required. Three air refueling status lights indicate the condition of the systems during air refueling. Fuel quantity during refueling of internal tanks may be monitored on the fuel quantity gage. External tanks quantity may be monitored by the external tank empty lights. When the lights extinguish, the tanks are full. After air refueling, the boom nozzle is released by automatic disconnect or by cockpit controls, the air refueling system is deactivated, and the fuel system is restored to normal operation.

NOTE

- Refer to T.O. 1-1C-1-17 for information on air refueling operations and procedures.
- When refueling to other than full internal or full internal and external, the fuel load may not be symmetrical.

AIR REFUEL SWITCH

The guarded air refuel switch (figure 4-24) is located on the air refueling panel on the left console Ⓐ or right console Ⓑ (forward cockpit only.) The switch is placarded "Air Refuel" and has ON and OFF positions. Placing the switch to ON accomplishes the following: opens the slipway door; turns on the slipway lights; closes F and T tank Ⓐ or F tank Ⓑ pressurization valves; closes external tanks pressurization valves; and activates the function of the manual mode trigger as a boom nozzle manual disconnect switch. The F and T tank Ⓐ or F tank Ⓑ fuel valve closed warning lights illuminate with actuation of their respective pressurization valves. When the receptacle door is fully open, the blue "READY" light on the status light display illuminates. Placing the air refuel switch to OFF returns the switch functions to their normal conditions and restores the fuel system to normal operation. Electrical power to the switch is supplied by the 28-volt dc essential bus.

AIR REFUEL SELECT SWITCH

The guarded air refuel select switch (figure 4-24) is located on the air refueling panel on the left console Ⓐ or right console Ⓑ (forward cockpit only). The switch is labeled "Refuel Select" and has INT ONLY and ALL TANKS positions. Placing the switch to INT ONLY prior to air refueling closes the external tank refuel selector valve and results in refueling of the internal fuel tanks only. Placing the switch to ALL TANKS permits refueling of both internal and external fuel tanks. The switch receives power from the 28-volt dc essential bus.

EMERGENCY SLIPWAY DOOR OPEN SWITCH

The emergency slipway door open switch (figure 4-24) is located on the air refueling panel on the left console Ⓐ or right console Ⓑ (forward cockpit only). The switch is placarded "Emerg Slipway Door Open" and has NORM and EMERG positions. The NORM position is used for all normal airborne refueling operations. This position allows opening of the slipway door by secondary hydraulic system pressure when the air refuel switch is placed to ON. The EMERG position is used to open the slipway door in event of secondary hydraulic system failure. Placing the switch to EMERG when the air refuel switch is ON routes high pressure pneumatic system air to a shuttle valve at the slipway door actuating cylinder. The shuttle valve, normally positioned for hydraulic pressure flow, allows high pressure pneumatic air to enter the door open side of the slipway door actuator, opening the door. When the switch is returned to NORM, pneumatic system pressure to the door is shut off and pneumatic air in the lines vents to ambient pressure. There is no provision for closing the slipway door pneumatically. The switch receives power from the 28-volt dc essential bus.

RESET/MBL SWITCH

The reset/MBL switch (figure 4-24) is located above the throttle quadrant. On Ⓑ airplanes, the switch is in the forward cockpit only. The switch has a center NORM position and RESET and MBL positions. It is retained in the NORM position by a spring load from RESET and by a detent from MBL. In the NORM position, normal system operation and automatic boom latching are possible. In event system electrical malfunction disables the automatic boom latching capability, placing the switch to MBL activates the manual boom latching capability. This allows the boom latches to be retracted and extended by depressing and releasing the manual disconnect switch on the control stick grip. By this means, the boom nozzle can be manually latched and released. The RESET position is used to reset the system after any disconnect during refueling. Momentary actuation of the switch to RESET resets the air refueling system amplifier and prepares the system for another

air refueling controls and indicators

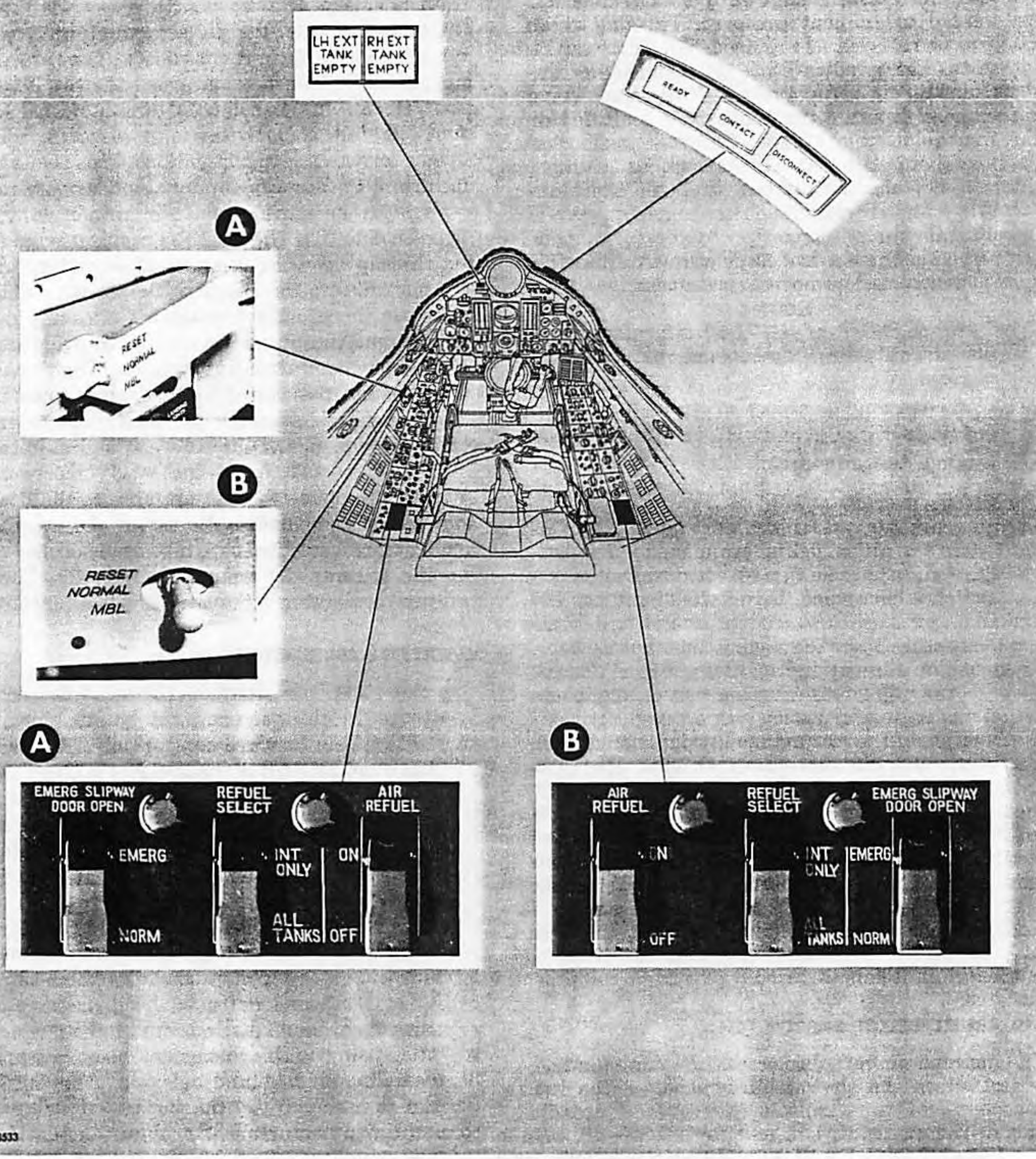


Figure 4-24

contact and additional refueling. The RESET position can also be used as an alternate method of releasing the boom latches when the manual disconnect switch fails to effect a release. When releasing the boom latches by this method, the switch must be held in the RESET position until the boom nozzle is removed from the receptacle. The reset/MBL switch receives power from the 28-volt dc essential bus.

MANUAL DISCONNECT SWITCH

The manual mode trigger (figure 1-18) on the right-hand control stick grip (Ⓑ front cockpit only) serves as an air refueling manual disconnect switch when the air refuel switch is ON. The switch is a two-position switch. When the manual disconnect switch is depressed, the boom latches retract to release the boom from the receptacle. When the reset/MBL switch is in MBL, depressing the manual disconnect switch retracts the latches to effect a manual boom latch. In the MBL mode, the manual disconnect switch is depressed for boom nozzle insertion, then is released to lock the nozzle in place. After refueling is complete, the nozzle is released by depressing the manual disconnect switch. The switch receives power from the 28-volt dc essential bus.

AIR REFUELING STATUS LIGHTS

Three status lights (figure 4-24) are located on a panel on the top right side of the glare shield. The lights are in the forward cockpit only on Ⓑ airplanes. The lights are press-to-test type indicator lights and display the status during different phases of air refueling. Their functions are described in the following paragraphs.

Ready Light

The ready light is the left light on the status light panel and illuminates blue to display a "READY" indication. The light illuminates when the air refuel switch is placed to ON and remains illuminated while the system is ready to accept a contact from the tanker. When the boom nozzle is inserted in the receptacle and the boom latches lock, the ready light goes off. The light will remain off until a new refueling sequence is initiated by turning the air refuel switch to ON or placing the reset/MBL switch to RESET. The ready light receives power from the 28-volt dc essential bus.

Contact Light

The contact light is the center light on the status light panel. It illuminates green to display "CONTACT." The light illuminates when the boom nozzle is inserted into the receptacle. It extinguishes when the nozzle is removed from the receptacle. The contact light receives power from the 28-volt dc essential bus.

Disconnect Light

The disconnect light is the right-hand light on the status light panel and illuminates amber to display a "DISCONNECT" indication. The light illuminates when the boom nozzle is removed from the receptacle. It will then remain on until the reset/MBL switch is placed to RESET or the air refuel switch is placed in OFF and the slipway door closes. If the slipway door remains open with the air refuel switch in OFF, the light will remain on. The disconnect light is powered by the 28-volt dc essential bus.

MISCELLANEOUS EQUIPMENT

ANTI-G SUIT PROVISIONS

Engine bleed air, cooled by the cockpit air-conditioning primary heat exchanger, is used to pressurize the anti-g suit. A regulator maintains pressures of between 9 and 11 psi in the suit at high g loads. A valve in the regulator assembly automatically pressurizes the suit at g loads greater than 1½ to 2 g's. An anti-g suit button (29, figure 1-10), placarded "Push to Test," is used to manually pressurize the suit when desired. When the button is depressed, unregulated engine bleed air pressurizes the anti-g suit. When the button is released, the suit is depressurized. This feature of the regulator may be used during extended flights to prevent fatigue by creating a massaging effect on the body. Operation of the manual control button does not test the automatic feature of the regulator.

SPARE LAMPS

On some airplanes spare lamps are located on the aft end of the left-hand console.

LUGGAGE RACK PROVISIONS

A luggage rack may be installed in the missile bay to provide a container for both luggage and clothing. The missile bay doors may be manually opened when electrical power is not available. The manual missile bay door control handle is located on the forward bulkhead of the left-hand main wheel well, and has OPEN and CLOSE positions. To open the doors, pull the control handle out, then rotate the handle to the right. To close the doors, rotate the handle to the left and push in. Pneumatic pressure is used to operate the doors. A safety pin with streamer must be installed whenever the handle is left in the OPEN position.

CAUTION

- Do not operate missile bay doors with manual missile bay door control handle unless airplane pneumatic pressure is above 2000 psi. Structural damage can occur during system operation if pneumatic pressure falls below 1500 psi.
- Inflight operation of the missile bay doors with the luggage rack installed is prohibited.

MAP AND DATA CASE

The map and data case (25, figure 1-11) is located aft of the right console.

INSTRUMENT FLIGHT TRAINING HOOD

An instrument flight training hood is installed on the rear canopy of some airplanes. Hood may be manually positioned.

operating limitations

Section V

TABLE OF CONTENTS

NOTE

- This section includes airplane and engine limitations which must be observed during normal operation. The high performance of this airplane demands close adherence to these limitations.
- The limitations in this section are exact limits. Limitations in other sections of this manual may show minor differences as a result of being rounded off for simplification.

INSTRUMENT MARKINGS

Careful attention must be given to the instrument markings (figure 5-1), because the limitations shown on these instruments and noted in the captions are not necessarily repeated in the text of this or any other section.

ENGINE LIMITATIONS

Normal engine limitations are shown in figure 5-1 and figure 5-2. Maximum thrust is the thrust obtained by placing the throttle fully forward and outboard for afterburner operation. Military thrust is the thrust obtained by placing the throttle full forward and inboard (nonafterburning).

EGT SPREAD

Limitation for EGT spread check is 100° maximum for runup and 90° maximum after 5 minutes of stabilized operation.

ENGINE OVERSPEED

The maximum permissible engine speed is 106.5% rpm. Any engine speed in excess of this limit must be noted on Form 781. The engine must be inspected for damage.

ENGINE PRESSURE RATIO

The military and maximum thrust checks are based on the takeoff thrust index marker being set according to the outside air temperature. When making the military thrust check during the Before Takeoff, Engine Check, the page pointer should be within the arc of the takeoff thrust index marker. Engine rpm should be stabilized. The maximum thrust check is made at the start of takeoff roll, and is the same as the military thrust check. During takeoff roll, the gage reading will increase above that noted during the maximum thrust check.

OIL PRESSURE

Normal oil pressure is 40 to 50 psi. Except at idle, oil pressures between 35 and 40 psi are undesirable and should be tolerated only for the completion of the flight, preferably at a reduced throttle setting. Oil pressures below normal should be entered on Form 781, and should be corrected before the next takeoff. Oil pressures below 35 psi are unsafe and require that a landing be made as soon as possible, using the minimum thrust required to sustain flight. Although 40 to 50 psi is the allowable pressure limit during all continuous or stabilized engine operation, a maximum of 80 psi is permitted during operation of short duration, such as for takeoff, go-around, or climb.

instrument markings

BASED ON: JP-4 FUEL

NOTE
REFER TO AIRSPEED LIMITATIONS, THIS SECTION, FOR MAXIMUM ALLOWABLE AIRSPEEDS.

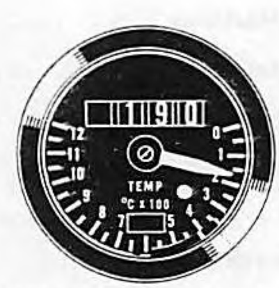
MACH INDICATOR

STRIPED POINTER INDICATES MACH NUMBERS CONSISTENT WITH MAXIMUM DESIGN AIRSPEED OR MAXIMUM STAGNATION TEMPERATURE, WHICHEVER OCCURS FIRST.

AIRSPEED — ANGLE OF ATTACK INDICATOR

287 KCAS MAXIMUM SPEED WITH LANDING GEAR EXTENDED.

256 KCAS MAXIMUM SPEED DURING EMERGENCY LANDING GEAR EXTENSION.



OIL QUANTITY

NORMAL OPERATING RANGE
CAUTION
MINIMUM OPERATING RANGE

EXHAUST GAS TEMPERATURE

340°C TO 575°C CONTINUOUS OPERATION

635°C MAXIMUM FOR MILITARY THRUST, MAXIMUM THRUST—650°C MAXIMUM ON ACCELERATION, NOT TO EXCEED TWO MINUTES

400°C MAXIMUM FOR STARTING

OIL PRESSURE

35 PSI MINIMUM
35 PSI TO 40 PSI (CAUTION) 1
40 PSI TO 50 PSI NORMAL OPERATION
50 PSI MAXIMUM 2

HYDRAULIC PRESSURE GAGE
(PRIMARY AND SECONDARY SYSTEM GAGES IDENTICAL)

1000 PSI MINIMUM OPERATING PRESSURE
1000 PSI TO 2500 PSI (CAUTION) 3
2500 PSI TO 3100 PSI CONTINUOUS OPERATING PRESSURE (NORMAL PRESSURES WITH NORMAL DEMANDS ON THE SYSTEM)
3100 PSI MAXIMUM OPERATING PRESSURE

48090-1

Figure 5-1

FUEL QUANTITY GAGE

- 0-232 LBS UNUSABLE FUEL
- (A) 7700 POUNDS USABLE FUEL REMAINING ⚠4
- (B) 7300 POUNDS USABLE FUEL REMAINING ⚠4

TACHOMETER

- 106.5% MAXIMUM PERMISSIBLE RPM
- 80% TO 100% CONTINUOUS OPERATION

⚠1 REFER TO ENGINE LIMITATIONS, THIS SECTION, FOR ADDITIONAL INFORMATION WHEN OIL PRESSURE IS IN THIS RANGE.

⚠2 A MAXIMUM OF 80 PSI IS ALLOWABLE FOR SHORT DURATION DURING TAKEOFF, GO-AROUND AND CLIMB.

⚠3 PERMISSIBLE WITH HIGH FLOW DEMANDS ON SYSTEM. INDICATES MALFUNCTION WITH NO FLOW DEMANDS ON THE SYSTEM.

⚠4 REFER TO ACCELERATON LIMITATIONS, THIS SECTION, FOR ALLOWABLE LOAD FACTORS ABOVE AND BELOW THIS FUEL QUANTITY.

48090-2

NOTE

The oil pressure gage needle may normally fluctuate as much as ±2.5 psi.

COOLING LIMITATIONS (GROUND OPERATIONS)

Combustion Starter Cooling

At ambient temperatures up to 90°F (32.2°C) inclusive, the combustion starter duty cycle shall be limited to two consecutive combustion runs in rapid succession followed by a cooling time of 30 minutes minimum. Each succeeding run shall then be spaced a minimum of 25 minutes apart. At ambient temperatures above 90°F (32.2°C), the combustion starter duty cycle shall be limited to two consecutive combustion runs in rapid succession followed by a cooling time of 45 minutes minimum. Each succeeding run shall then be spaced a minimum of 40 minutes apart.

Engine Cooling

If the ground operating limits (figure 5-2) are closely approached, five minutes operation at idle is required to allow the engine to cool before it can be shut down or before a schedule which will approach these limits can be repeated.

Electronic Equipment Cooling

Flight will not be conducted or continued with the electronic warning light illuminated.

EMERGENCY FUEL

The following emergency fuels, listed in order of preference, are satisfactory for one time ferry missions only.

TYPE OF FUEL	FREEZE POINT
Commercial JET B	−51.1°C (−60°F)
JP-5 (MIL-J-5624)	−48.3°C (−55°F)
Commercial JET A-1	−50.0°C (−58°F)
Commercial JET A	−40.0°C (−40°F)
AVGAS, lowest grade available (MIL-G-5572)	−60.0°C (−76°F)

Both ground starts and air restarts may be more difficult for engines with fuel controls set to operate on JP-4.

CAUTION

- When using JP-5, Commercial JET A-1, Commercial JET A, or AVGAS, a higher exhaust gas temperature will be reached at lower thrust settings than when operating on JP-4.
- Missions on which the above fuels are used shall be restricted to altitudes where temperatures below the freeze points will not be encountered.

engine operating limits

J75-P-17		MAXIMUM EXHAUST GAS TEMP (C)	TIME LIMITS	
			GROUND OPERATION	FLIGHT OPERATION
OPERATING CONDITIONS	MAXIMUM	635	5 MINUTES	15 MINUTES
	MAXIMUM ON ACCELERATION	650	2 MINUTES	2 MINUTES
	MILITARY	635	15 MINUTES ⚠1	30 MINUTES
	MAXIMUM CONTINUOUS (NON AFTERBURNER)	575	15 MINUTES ⚠1	CONTINUOUS
	IDLE	340	⚠1	———
	STARTING	400	———	———

NOTE
- IF TEMPERATURE LIMITS ARE EXCEEDED, MAKE ENTRY ON FORM 781 STATING DURATION AND PEAK TEMPERATURES.
- TIME LIMITATIONS ARE FOR OPERATING CONDITIONS (THROTTLE SETTINGS) REGARDLESS OF EGT.

⚠1 REFER TO COOLING LIMITATIONS, THIS SECTION.

48094

Figure 5-2

NOTE

NATO F-40 (MIL-J-5624) is the equivalent of JP-4 and normal operating limits of the engine are applicable.

AIRSPEED LIMITATIONS

NOTE

For information concerning AIRSPEED LIMITATIONS, see figure 5-3.

OTHER OPERATING LIMITATIONS

NOTE

For information concerning OTHER OPERATING LIMITATIONS, see figure 5-3.

MAXIMUM BARRIER ENGAGEMENT GROUND SPEED

Figure 5-4 shows maximum barrier engagement speed versus aircraft gross weight. The maximum speed is based on the limit imposed by the yield strength of the tailhook. Both the BAK-9 and BAK-12 have a greater arresting capability than the airplane tailhook. Above 30,000 pounds, the BAK-6 is limited by tailhook strength.

NOTE

The BAK-6 arresting barrier must be engaged as near to the runway centerline as possible. Maximum engagement speed is reduced 5 knots for each 10 feet off center for an offcenter engagement.

MANEUVERS

NOTE

For information concerning MANEUVERS, see figure 5-3.

operating limitations

CONFIGURATION	ACCELERATION LIMITS	AIRSPEED LIMITS
LANDING GEAR EXTENDED AND DURING NORMAL EXTENSION OR RETRACTION	—1 TO 2.5g	287 KCAS DURING EMERGENCY EXTENSION, MAXIMUM ALLOWABLE AIRSPEED IS 256 KCAS. AFTER EXTENSION BY EMERGENCY SYSTEM, MAX AIRSPEED OF 287 KCAS APPLICABLE. TO PREVENT STRUCTURAL DAMAGE TO THE LANDING GEAR, AT LANDING GROSS WEIGHTS IN EXCESS OF 26,891 POUNDS ON Ⓐ AIRPLANES OR 28,060 POUNDS ON Ⓑ AIRPLANES, THE RATE-OF-SINK AT TOUCHDOWN IS LIMITED TO 300 FPM. AT LANDING GROSS WEIGHTS LESS THAN 26,891 ON Ⓐ AIRPLANES OR 28,060 ON Ⓑ AIRPLANES, THE RATE-OF-SINK AT TOUCHDOWN IS LIMITED TO 540 FPM.
DRAG CHUTE DEPLOYMENT	———	164 KCAS
CANOPY OPEN	———	Ⓐ MAXIMUM TAXI SPEED IS 75 KCAS Ⓑ MAXIMUM TAXI SPEED IS 60 KCAS Ⓑ DO NOT TAXI WITH THE CANOPY OPEN HIGHER THAN APPROXIMATELY 12 INCHES.
RAIN REMOVAL SYSTEM	———	PROHIBITED DURING SUPERSONIC FLIGHT
RAM AIR TURBINE EXTENDED	—1 TO 3g	352 KCAS WHEN RAT IS ONLY SOURCE OF HYDRAULIC PRESSURE DO NOT FLY AT AIRSPEEDS BELOW 174 KCAS.
VARIABLE RAMP EMERGENCY RETRACTION	———	MACH 1.1 OR 500 KCAS WHICHEVER OCCURS FIRST AT ALTITUDES BELOW 25,000 FEET, EMERGENCY RAMP OPERATION IS PERMISSIBLE ONLY AT IDLE RPM.
TIRE LIMIT SPEED ON RUNWAY	———	217 KNOTS
ARMAMENT EQUIPMENT		
DOORS OPEN	—1 TO 3g (WITH OR WITHOUT ARMAMENT GEAR EXTENDED)	
ARMAMENT FIRING	———	ARMAMENT FIRING IS PERMISSIBLE WITH EXTERNAL TANKS INSTALLED WITHIN EXISTING EXTERNAL TANK LIMITATIONS.

48104-1

Figure 5-3 (Sheet 1 of 3)

operating

CONFIGURATION	ACCELERATION LIMITS		AIRSPEED LIMITS
	SYMMETRICAL MANEUVERS	ROLLING PULLOUTS	
CLEAN			
(A) LESS THAN 7700 LB OF FUEL:			752 KCAS, MAXIMUM STAGNATION TEMP, OR MACH 2, WHICHEVER OCCURS FIRST
MAX MANEUVER WARN LIGHT OUT	−3 TO 7g	0 TO 5g	
MAX MANEUVER WARN LIGHT ON	−2.3 TO 5g	0 TO 3.6g	NOTE
(A) MORE THAN 7700 LB OF FUEL	−2.4 TO 5g	0 TO 3.9g	A STRIPED POINTER ON THE MACH INDICATOR (ON AIRPLANES WITH THE CONVENTIONAL INSTRUMENT DISPLAY), OR A STRIPED MARKER ON THE AIRSPEED-MACH INDICATOR (ON AIRPLANES WITH THE INTEGRATED INSTRUMENT SYSTEM) INDICATE MAXIMUM DESIGN AIRSPEED (752 KCAS) OR MAXIMUM STAGNATION TEMPERATURE (249°F), WHICHEVER OCCURS FIRST. IF THE STRIPED POINTER IS INOPERATIVE, THE MAXIMUM DESIGN SPEED WILL BE MACH 1.82 OR 670 KCAS, WHICHEVER OCCURS FIRST. (THIS SPEED IS BASED ON AIR FORCE STANDARD HOT ATMOSPHERE.)
(B) LESS THAN 7300 LB OF FUEL:			
MAX MANEUVER WARN LIGHT OUT	−2.4 TO 6g	0 TO 4.3g	
MAX MANEUVER WARN LIGHT ON	−1.8 TO 4.5g	0 TO 3.3g	
(B) MORE THAN 7300 LB OF FUEL	−2.4 TO 5g	0 TO 3.9g	
WITH 360-GALLON EXTERNAL TANKS			
(A)(B) CONTAINING FUEL	−2.4 TO 5g	0 TO 3.9g	SAME AS CLEAN AIRPLANE LIMITS
	NOTE IF MAXIMUM MANEUVER LIGHT COMES ON WHILE TANKS CONTAIN FUEL, OBSERVE MAXIMUM MANEUVER LIGHT ON LIMITS FOR THE CLEAN AIRPLANE.		NOTE DO NOT JETTISON THE TANKS IN THE 0.95 TO 1.2 MACH NUMBER RANGE BELOW 25,000 FEET. TANK JETTISON IN THIS REGION WILL RESULT IN MOMENTARY NEGATIVE LOAD FACTORS APPROACHING OR EXCEEDING NEGATIVE LOAD FACTOR LIMITS.
(A)(B) EMPTY	SAME AS CLEAN AIRPLANE LIMITS		SAME AS CLEAN AIRPLANE LIMITS

48104-2

Figure 5-3 (Sheet 2 of 3)

NOTE

- Symmetrical maneuver limits apply when the airplane bank angle is constant (no roll) during the period of accelerated flight.
- Rolling pullout limits apply when the airplane bank angle is changing during the period of accelerated flight.

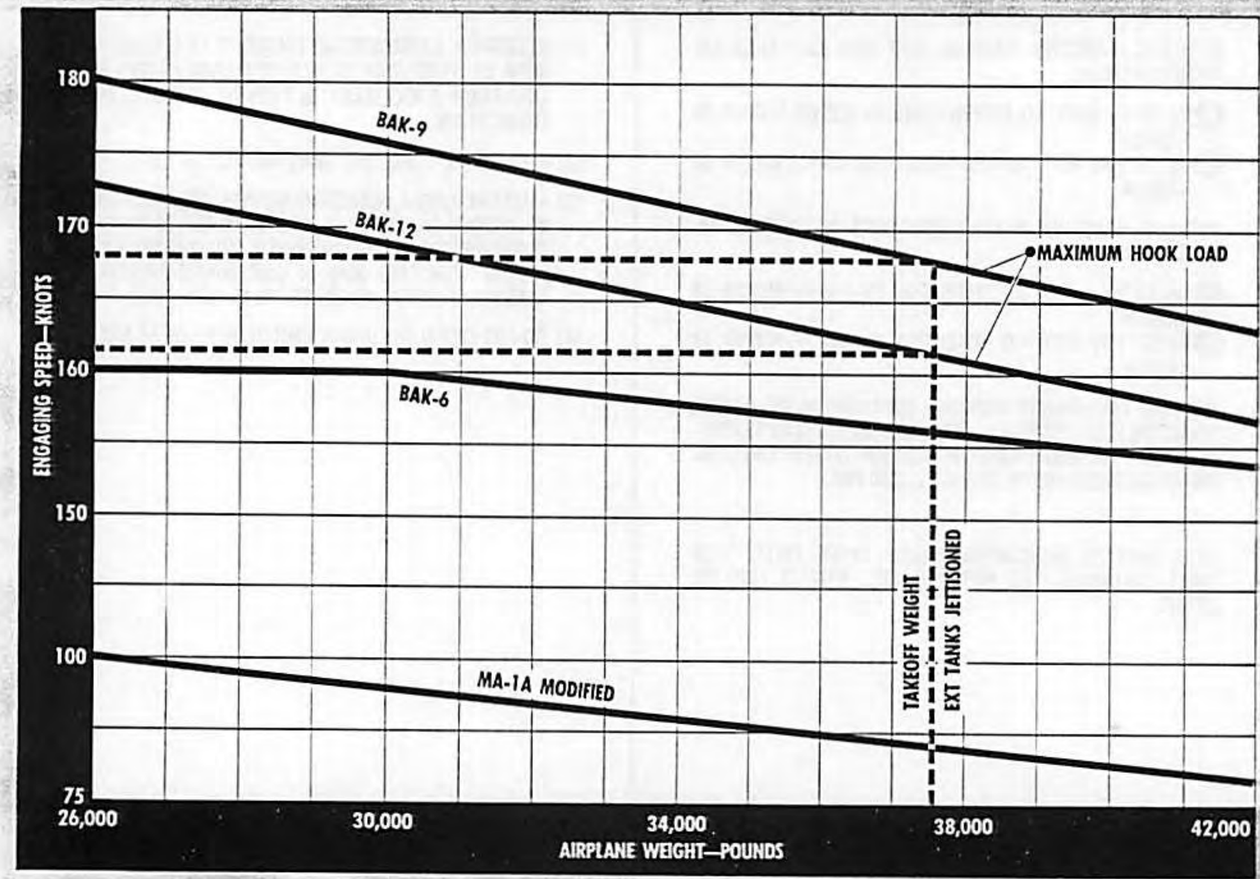
limitations

MANEUVERS	GENERAL MANEUVERING LIMITATIONS
WITH MAX MANEUVER WARNING LIGHT OUT, 360° ROLLS ARE PERMITTED FROM: Ⓐ 0g TO 5g WITH NO RESTRICTIONS ON USE OF RUDDER OR AILERON Ⓑ 0g TO 4.3g WITH NO RESTRICTIONS ON USE OF RUDDER OR AILERON WITH MAX MANEUVER WARNING LIGHT ON, 360° ROLLS ARE PERMITTED FROM: Ⓐ 0g TO 3.6g WITH NO RESTRICTION ON USE OF RUDDER OR AILERON Ⓑ 0g TO 3.3g WITH NO RESTRICTION ON USE OF RUDDER OR AILERON WITH THE YAW DAMPER AND TURN COORDINATOR OFF, AILERON DEFLECTION IS LIMITED TO 5° (APPROXIMATELY ⅓ STICK THROW). HOWEVER, FULL AILERON MAY BE USED FOR TAKEOFF AND LANDING AT AIRSPEEDS BELOW 287 KCAS (280KIAS).	(1) NEGATIVE g MANEUVERS ARE LIMITED TO 15 SECONDS MAXIMUM, OF WHICH ONLY 10 SECONDS MAY BE AT ZERO g. THIS LIMITATION IS NECESSARY TO PROVIDE ADEQUATE ENGINE LUBRICATION. (2) INTENTIONAL STALLS AND SPINS ARE PROHIBITED. (3) EXCESSIVE RUDDER INDUCED MANEUVERS (FISH TAIL) SHOULD BE AVOIDED (INTENT IS TO PERMIT THE USE OF UP TO 180 POUNDS PEDAL FORCE). HOWEVER, FULL RUDDER RAPIDLY APPLIED (FISH TAIL) MAY BE USED DURING TAKEOFF AND LANDING AT SPEEDS BELOW 287 KCAS (280 KIAS). (4) DO NOT EXCEED THE DESIGN LIMIT ALTITUDE OF 65,000 FEET.
CLEAN AIRPLANE MANEUVERING LIMITS APPLY EXCEPT WITH TANKS CONTAINING FUEL MAXIMUM ROLL RATE IS 100° PER SECOND.	

48104-3

Figure 5-3 (Sheet 3 of 3)

maximum barrier engagement ground speed



NOTE
THE BAK-6 ARRESTING BARRIER MUST BE ENGAGED AS NEAR TO THE RUNWAY CENTERLINE AS POSSIBLE. MAXIMUM ENGAGEMENT SPEED IS REDUCED 5 KNOTS FOR EACH 10 FEET OFF CENTER FOR AN OFFCENTER ENGAGEMENT.

48608

Figure 5-4

ACCELERATION LIMITATIONS

See figure 5-5 for maneuvering flight limits plotted for a combat configuration with less than 7700 pounds of fuel remaining on Ⓐ airplanes or 7300 pounds of fuel remaining on Ⓑ airplanes. These charts represent the maneuvering flight limits for symmetrical maneuvers only. For rolling pullouts see figure 5-3. Maneuvering flight limit charts 1 and 2 contain essentially the same information in the areas of positive g acceleration, but chart 1 also contains negative g limits. Chart 1 is plotted against calibrated airspeed while chart 2 is plotted against Mach number. Chart 2 provides acceleration limits for any combination of altitude and Mach number. At the higher altitudes, maneuvering capability is determined by the elevator deflection limit (elevator against the mechanical stops). At lower altitudes in the high speed region, maneuvering capability is limited by the hydraulic power available for control surface deflection. This is referred to as the hinge moment limit. The stall limit, represented by the lines on the left side of chart 2, limits the maneuvering capability at relatively low speeds. An amber light (maximum maneuver warning light) on the instrument panel illuminates to indicate speeds consistent with stagnation temperature of 174°F and represents the maximum speed for maximum maneuvers. In the event the amber light is inoperative, this speed will be considered to be Mach 1.0 below 10,000 feet and Mach 1.55 or 565 KIAS, whichever occurs first, above 10,000 feet. This speed is based on Air Force standard hot atmosphere.

On Ⓐ airplanes, for example, locate the intersection of Mach .8 and 40,000 feet. At this speed and altitude, the airplane will approach a stall at approximately 2.8 g. It will also be noted on chart 2 that a step variation in maneuvering capability occurs above 10,000 feet and Mach 1.14. This is caused by the fuel transfer system. When the cg is moved aft, less elevator deflection is required to pull a given load factor. When operating in regions where elevator deflection is the limiting factor, moving the cg aft provides an increased maneuvering capability. On chart 1, the fuel transfer condition is represented by vertical portions of the various altitude curves. These vertical lines (marked FX) indicate the increased g capability provided by transferring fuel aft. The stall limits for various load factors, both positive and negative, are indicated near the left edge of the green area on chart 1. You will also note on chart 1 that many of the elevator deflection or hinge moment limits fall within the green area. When this occurs, the airplane cannot be overstressed. For an example, follow the flight envelope for 45,000 feet altitude. The entire envelope falls within the green area. This means that the stall limit, the elevator deflection limit, or the hinge moment limit will be reached before the airplane structural limits are exceeded.

CENTER OF GRAVITY LIMITATIONS

CG limitations will not be exceeded during normal flight operation with either reduced internal or full internal fuel, and/or external wing tanks or standard armament. Refer to Handbook of Weight and Balance Data, T.O. 1-1B-40.

WEIGHT LIMITATIONS

The design of the airplane precludes overloading. The maximum gross weight will not be exceeded even when standard armament and full external tanks are carried.

maneuvering

MODEL: F-106A
DATE: 1 OCTOBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN 7700 POUNDS (60%) TOTAL FUEL OR LESS

ENGINE: J75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

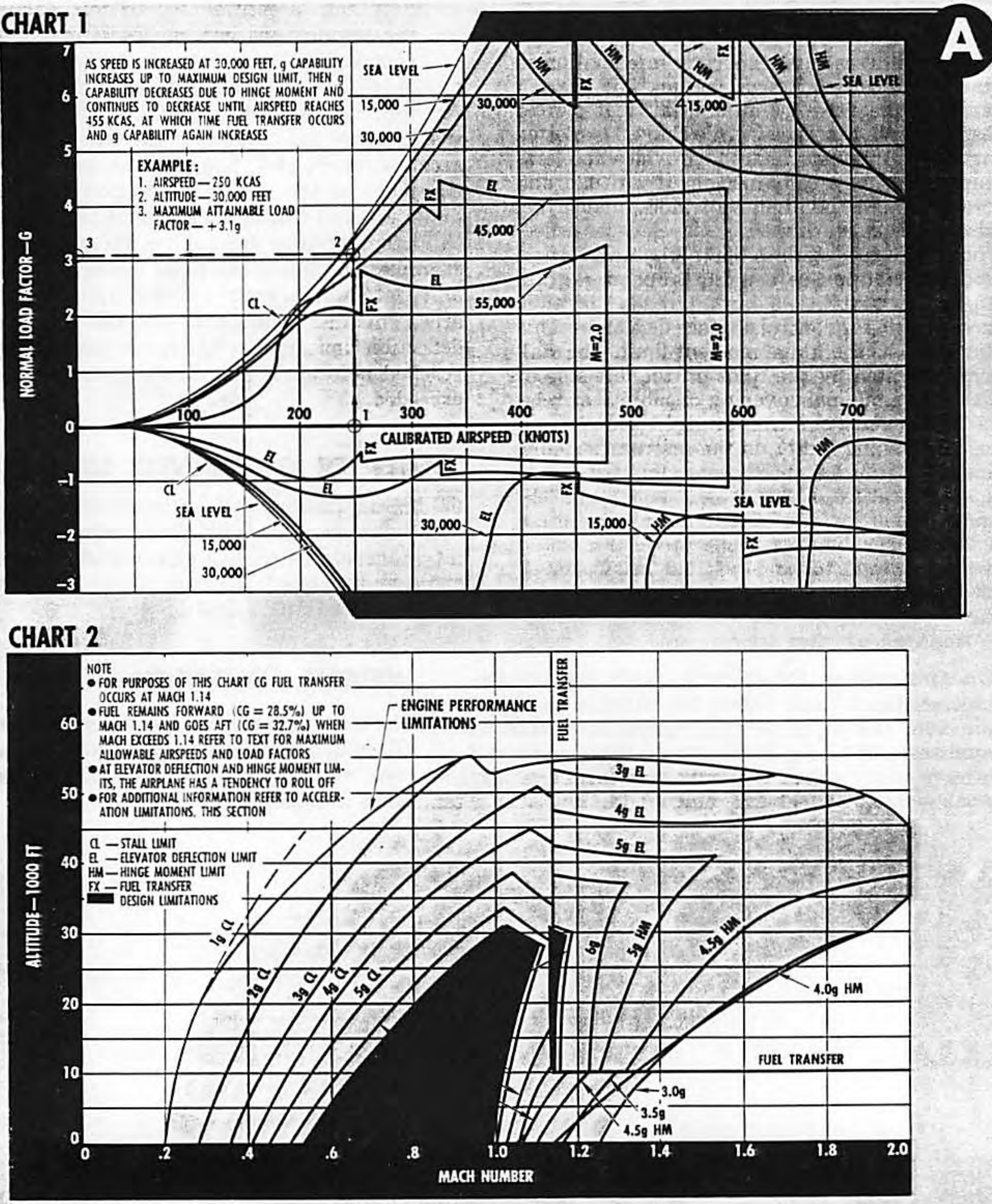


Figure 5-5

flight limits

MODEL: F-106 B
DATE: 1 OCTOBER 1959
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN 7300 POUNDS (60%) TOTAL FUEL OR LESS

ENGINE: J75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

CHART 1

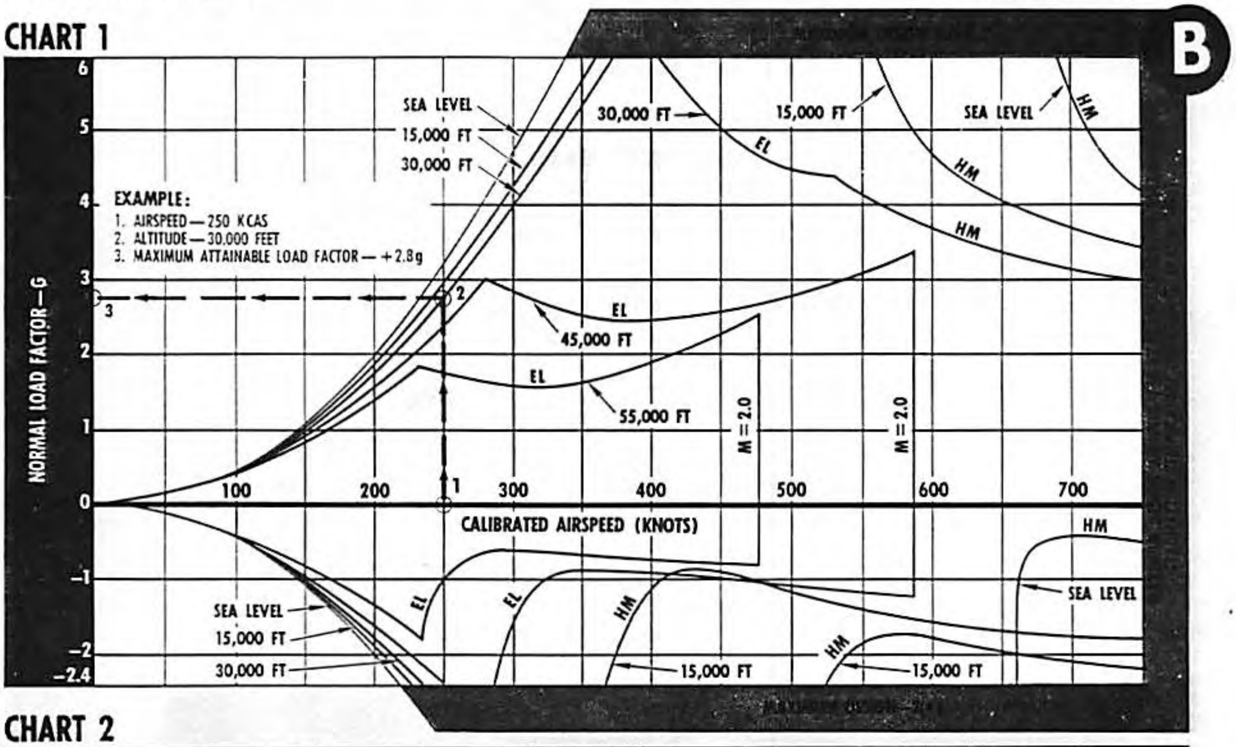
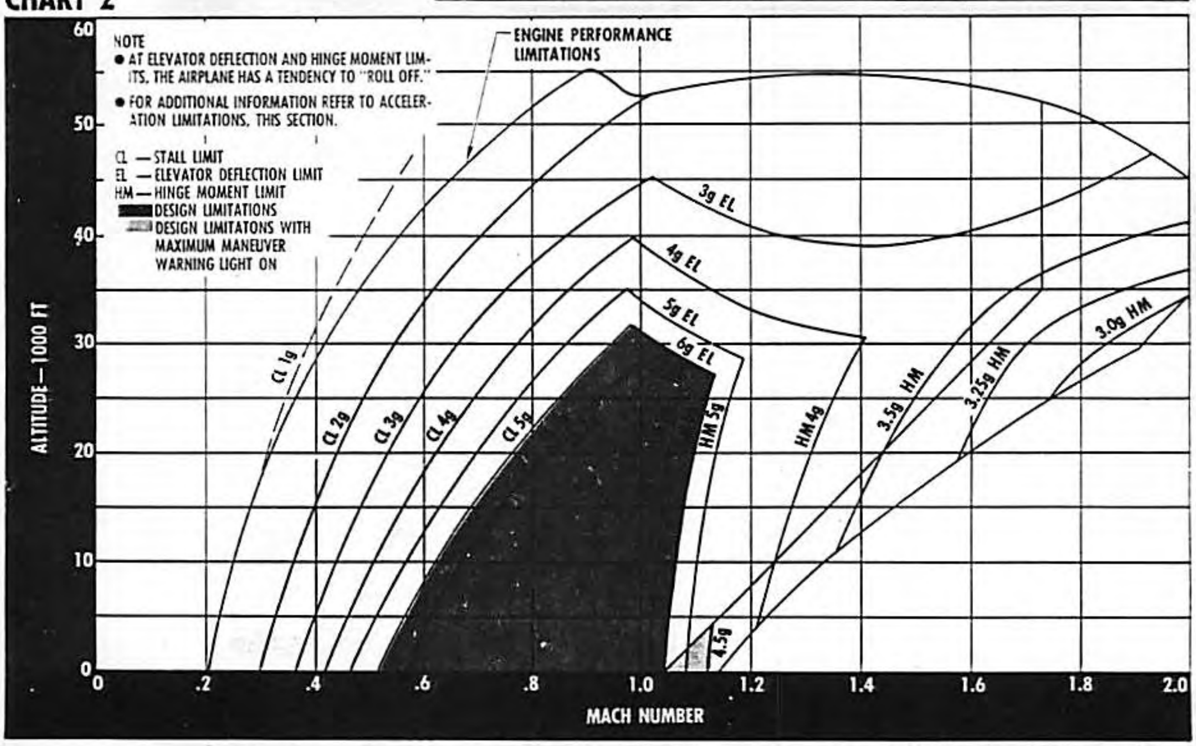


CHART 2



48093-2A

flight characteristics

Section VI

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NOTE

The flight characteristics information in this section is based on flight test data.

INTRODUCTION

The airplane has a conventional turbojet airplane response to change of thrust; however, the change of thrust in the operating range can be relatively large for a small throttle movement due to engine bleed valve operation. Fuel transfer maintains the center of gravity within allowable limits, depending on flight conditions, to provide optimum high-speed performance. The use of elevons requires no unusual flying techniques as stick movement for a desired maneuver is the same as it would be if conventional elevator and ailerons were employed. All flight control surfaces are hydraulically actuated to permit accurate control of the airplane at airspeeds which would otherwise impose such airloads that control surface movement would be impossible. An artificial feel system is provided to simulate stick forces felt in conventional type control systems. Control response is good at all speeds; however, a "snaking" motion appears in the transonic speed range. This snaking or oscillating motion is damped out by the yaw damper system. In addition to the yaw damper (which incorporates a turn coordination system) a pitch damper is installed to minimize pitch oscillations.

LEVEL FLIGHT CHARACTERISTICS

LOW SPEED

Control effectiveness at low speeds is good when operating at, or near, the recommended minimum speeds or at normal landing and approach speeds. Flight at airspeeds in the approach, landing, and go-around speed ranges is characterized by a relatively high angle of attack. Because of a high degree of control effectiveness during low-speed flight, it is possible to develop a critical angle of attack without being aware of it. A critical angle of attack may lead to excessive sink rate or result in operation in an area of directional instability. Once either of these conditions (high sink rate or directional instability) develops at low altitude, it may be impossible to recover with full thrust applied before contact with the ground. Refer to STALLS, this Section, for additional information. See figure 6-1 for minimum speed capabilities, and figure 6-2 for low-speed characteristics. Although the design of the airplane provides for good forward visibility during all normal flight, the wide range of the obtainable angle of attack (angle of attack in excess of about 20°) at low speed makes it possible to obstruct forward visibility along the flight path with the nose of the airplane.

MEDIUM SPEED

Medium speed flight characteristics are conventional; however, dampers should be utilized. The speed at which dampers are most beneficial is in the high subsonic region.

WARNING

- High-speed, low-altitude flight without dampers can cause control problems and should be avoided except where required to accomplish the assigned mission.
- Flight during rapid deceleration through the transonic region with high positive load factors can result in stall and/or post-stall gyrations. This condition results from a large increase in elevator effectiveness when decelerating through the transonic region (1.1 Mach to 0.9 Mach).

The airplane is dynamically stable at all speeds with the dampers off; however, lateral-directional oscillations (dutch roll) may occur when flying through the transonic range. Quite often these oscillations give the impression of being divergent; however, this is not the case as the oscillations will damp with release of the controls.

HIGH SPEED

When supersonic flight is anticipated, the pitch and yaw damper system should be engaged to aid in maintaining coordinated flight. As speed increases in the supersonic speed range, up elevator trim is required. See figure 6-1 for maximum speed capabilities in level flight. When using maximum thrust at low altitude, it is possible to exceed temporary or design speed limitations (refer to Section V). At high supersonic speeds the airplane may encounter stall-buzz. Refer to STALL-BUZZ, Section VII.

MANEUVERING FLIGHT CHARACTERISTICS

Stick forces per load factor (maneuvering stick forces) are very satisfactory throughout the speed range of the airplane. There is a tendency toward a slight increase in stick force per g at supersonic speed and as altitude increases. In supersonic flight, maneuvering capability is limited by elevator deflection at high altitudes and by the elevator hinge moment at low altitudes. When these limits are reached the airplane may have a tendency to roll-off because no further elevon movement is available for lateral (aileron) control. If the airplane starts to roll-off when approaching maximum "g" conditions, use rudder to counteract the roll-off. Pitch and yaw dampers provide adequate damping throughout the flight envelope and smooth out any oscillations, including those at high subsonic and transonic speeds. A turn coordinator is also incorporated into the yaw damper system to aid in coordinating maneuvers. The turn coordinator corrects the tendency of the airplane to yaw away from the turn (adverse yaw) at low speeds, while at high speeds it corrects the tendency of the airplane to yaw into the turn (favorable yaw). This is done in a scheduled manner so that during maneuvering the turn coordinator applies rudder in the proper direction to minimize side slip. During high angle of attack maneuvers smooth rudder application should be used as the primary control for directional change. The aileron must be held in the neutral position and back pressure applied to generate the rate of turn or roll required. Back trim should be used to minimize inadvertent addition of aileron. Any application of aileron will reduce roll or turn performance and may cause entry into a control loss condition. (At high angles of attack small aileron deflections produce large increments of adverse yaw although the amount necessary to induce a spin depends on many variables including airplane rigging, Mach number, angle of attack, and gross weight.) During turns in which heavy buffet is encountered, the airplane will ordinarily tend to roll out of turn, over the top. When this roll occurs, do not oppose it with aileron but keep the ailerons neutral, relax G load, and maintain directional control with rudder. Maneuvering characteristics with external tanks installed are essentially the same as the clean airplane. Roll rates for both rudder and aileron rolls are reduced slightly below that obtained with the clean airplane. It will be noticed that there is a slight increased pitch up tendency when decelerating through the transonic range (Mach 1.1 to 0.9). During high load factor maneuvers through the transonic range, this increased pitch up tendency should be anticipated and back stick pressure should be reduced to prevent exceeding positive load factor limits. There is also a slight increase in rudder sensitivity during low airspeed (below 200

KCAS) maneuvers. Flight with a single external tank aboard requires no additional restrictions. However, there is insufficient lateral trim available at supersonic speeds and the pilot must hold some lateral stick force. Supersonic roll performance away from the single tank is seriously degraded and should be considered prior to engaging air combat maneuvers.

STALLS, POST-STALL GYRATIONS, AND SPINS

For the purpose of this discussion, the term "stall" is defined as that condition wherein control is lost about any airplane axis. Airframe buffet is the first warning of a potential stall whether from straight and level flight or an accelerated turn. Buffet onset occurs at about 10° angle of attack and at an airspeed dependent upon weight and load factor as shown in Section V of the Flight Manual. The buffet intensity increases as the angle of attack is allowed to increase by either decreasing airspeed or increasing load factor. As angle of attack approaches 18°, the buffet intensity starts to decrease and deterioration of the lateral directional stability begins to occur coupled with lateral directional oscillations. Just prior to the stall, loss of aileron and rudder effectiveness occurs with a tendency for the airplane to roll out of a turn. The complete stall is characterized by a sudden increase in angle of attack above 28°.

In wings-level stall approaches, airframe buffet is first encountered at 170 to 175 KCAS. Buffet intensity increases with decreasing airspeed and reaches a maximum at 145 to 150 KCAS. The intensity then decreases with decreasing airspeed until, in most cases, it has dissipated at around 130 KCAS. Lateral directional instability is encountered at about 115 KCAS and usually persists until the stall. Aileron control ineffectiveness followed by a loss of rudder effectiveness occurs about 0 to 10 KCAS above the stall. At the stall (approximately 105 KCAS), angle of attack increases abruptly. Neutral aileron at the stall causes the airplane to wallow, lose altitude, and turn slowly in the direction of rudder application. Angles of attack will vary from approximately 50 to 70° with a sink rate of 10,000 to 15,000 fpm.

In accelerated entry stalls using wind up turns of from 1.5 to 2.5 g, airframe buffet is first encountered about 90 to 100 KCAS above the stall and increases with decreasing airspeed. About 10 to 15 KCAS before the stall, short period lateral-directional oscillations will occur which last until the stall. Just before the stall is reached, the airplane has a tendency to roll out of the turn. The complete stall is again characterized by an abrupt increase in angle of attack accompanied by a sudden roll off away from the direction of turn caused by adverse yaw.

Adverse yaw results from a combination of aileron application and rolling motion and occurs in a direction opposite the roll (i.e., when attempting to correct for a right-wing-low condition while flying near a stall, moving the stick to the left for a left-hand rolling motion will cause the nose to yaw to the right). At extreme angles of attack, adverse yaw can be severe enough to result in a spin or post-stall gyrations. During critical phases of flight (such as immediately before touchdown and during go-around), if approaching a stall or a suspected critical angle of attack, it is especially important to use rudder as the primary control for lateral and directional corrections. This action precludes the addition of adverse yaw into an already unsafe flight condition.

WARNING

Intentional stalls are prohibited due to the possibility of encountering post-stall gyrations.

Abrupt elevator inputs should be avoided while maneuvering at low airspeeds. A large elevator input will cause a rapid pitch change that will in turn reduce the amount of stall buffet warning. The onset of the stall buffet warning changes with changing airspeed but the range is approximately 10° angle of attack wide. For example, while maneuvering at 30,000 feet altitude and 220 KCAS with little or no buffet at 10° angle of attack and 8° up elevator, a rapid increase to 20° up elevator will increase the angle of attack to the full stalled condition in less than one second. The angle of attack increases through the 10° stall buffet range in less than 1/2 second giving little buffet warning of the approach to the stall. The airplane can fly into the area of buffet decrease, lateral control oscillations or even loss of control (if aileron is deflected) before corrective action can be initiated.

WARNING

Avoid large abrupt elevator inputs when maneuvering at low airspeed due to the possibility of encountering post-stall gyrations or spin without buffet warning.

Any control input at airspeeds below 200 KCAS should be gradual and no more than that required to achieve maximum performance. Avoid complete reliance on heavy buffet as an indication of approach to a stall. The amplitude and frequency of buffet varies with airspeed; therefore, buffet at high airspeeds is much more obvious than the heaviest buffet attainable at low airspeeds. At low airspeed, any increase in angle of attack above moderate buffet will decrease buffet intensity and lead to lateral control oscillations and loss of control.

A post stall gyration is defined as an erratic oscillatory motion of the airplane which occurs when the complete stall is reached with sideslip angle of at least 10°. During this maneuver, control surface deflections have no apparent effect upon airplane motion and the airplane will oscillate rapidly in roll, pitch, and yaw. It is not until the control surfaces are neutralized and the airplane is unloaded that the airplane will begin to recover.

Entry into post-stall gyration will most probably occur during turns with a high angle of attack where buffet is encountered and pressed to the point that the airplane begins to roll out of the turn. Application of more aileron into the turn at this point will cause sufficient adverse yaw to force the airplane to rapidly roll away from the turn and into a post-stall gyration or spin.

If a post-stall gyration is entered, neutralizing the aileron and unloading the airplane is the only way to recover. Any attempt by the pilot to initiate corrective control would be much too late and would only tend to aggravate the condition. There are large changes in pitch, bank, and heading. However, the heading oscillates, unlike the unidirectional rotation characteristics of the spin. When accompanied by high yaw rates, the lateral oscillations could very well give the pilot the first impression of a spin. If spin recovery aileron were applied under these conditions, the airplane would be forced into a spin in the direction opposite to the applied aileron. Application of stick forward of neutral may cause the airplane to go into an inverted spin; however, the rudder will be effective and the airplane may be rolled upright with rudder.

WARNING

If a post-stall gyration is encountered below **10,000** feet above the terrain—**EJECT**.

The airplane will spin only against the ailerons i.e., right-wing-down aileron produces a left spin and vice versa. As pro-spin controls are added at the stall—rudder with, aileron against, and up elevator —an abrupt pitch-up occurs followed by a turn and pitch down motion developing into the spin. The airplane will spin with the nose 20° to 40° below the horizon. Spin rate will be between 4 and 15 rpm with altitude loss per turn of 2000 feet at the higher rate and 5000 feet at the lower rate. With a four turn spin, altitude loss from entry to pullout can be 20,000 to 30,000 feet.

The airplane will not spin with only rudder input. During extremely high angle of attack maneuvering, directional control can be maintained with rudder alone. Any aileron input will probably force the airplane into a spin or post-stall gyration. During a full stall with aileron neutral the airplane will turn slowly in the direction of rudder application.

WARNING

Intentional spins are prohibited due to excessive loss of altitude and the possibility of not being able to effect a recovery.

During deceleration through the transonic speed range, the level flight trim change is moderate. However, if this deceleration occurs while in a hard turn or dive pullout, the large amount of up elevator necessary to turn at supersonic speeds must be reduced through the transonic range or the airplane will rapidly "dig in." Elevator effectiveness about triples; and if deceleration is rapid enough, the increased angle of attack will force the airplane into a fully stalled condition. Therefore, be prepared for this "dig in" effect and relax back stick pressure to maintain optimum turn performance.

Recovery from buffet or lateral instability is accomplished by relaxing back stick pressure and smoothly flying the airplane out of the pre-stall indication. Increasing the angle of attack beyond lateral control oscillations is not recommended under any circumstances due to the probability of encountering post-stall gyration and spin.
At the first sign of a control loss, neutralize the ailerons and unload the airplane to approximately zero "G". If the airplane does not recover:

1. CONTROL STICK–CENTER FORWARD OF NEUTRAL.
   Hold the above control stick position until recovery is evident or a spin developes.

WARNING

Do not attempt to oppose roll oscillations that occur during a post-stall gyration. A small aileron deflection may force the airplane into a spin in a direction opposite to control stick movement.

Do not apply full forward stick. Full forward stick may cause the airplane to roll inverted into a negative stall or an inverted spin. Either of these conditions can be recognized by a sustained negative load factor. If at any time during a recovery a negative G (less than zero G) condition occurs, neutralize the controls. If required, roll upright using rudder only. Any aileron input at this point may force the airplane back into a control loss condition.

NOTE

Rudder inputs under negative G conditions will result in a roll opposite to the direction of rudder application.

2. EMERGENCY DIRECT MANUAL BUTTON–DEPRESS.

If the above does not produce immediate signs of recovery and control loss continues:

3. THROTTLE–IDLE.
   Retarding the throttle to IDLE will reduce the possibility of engine overtemp and compressor stall.

If a spin developes:

WARNING

Oscillations in roll and yaw associated with a post-stall gyration can create the sensation of being in a spin. A spin is characterized by a constant rotation in one direction about the yaw axis. Definitely establish the direction of spin rotation by reference to the turn needle. When high yaw rates and lateral oscillations exist, it may not be possible to determine the direction of rotation visually by any means other than the turn needle.

4. HOLD FULL AILERON IN THE DIRECTION OF SPIN ROTATION MAINTAINING FORWARD STICK UNTIL THE SPIN IS BROKEN OR EJECTION ALTITUDE IS REACHED.

WARNING

After applying recovery controls, four or more turns may be required to effect recovery. If spin rotation is not stopped by 10,000 feet above the terrain–EJECT.

NOTE

Termination of the spin is characterized by an abrupt nose down pitch change to the near vertical attitude and airspeed increasing above 140 KCAS.

When the spin is broken:

5. NEUTRALIZE AILERONS—MAINTAIN DIVE ATTITUDE UNTIL AIRSPEED REACHES 140 KCAS.

NOTE

- A rolling motion may accompany the recovery and may appear to be a continuation of spin rotation. As airspeed builds, the rolling motion will cease after ailerons are neutralized.
- After recovery from a spin during which excessive exhaust gas temperatures may have been encountered, the airplane should be recovered using minimum thrust to maintain flight. If the airplane has been fully stalled or spun, a write-up is required on the AFTO Form 781 so the engine can be inspected for possible damage.

FLIGHT CHARACTERISTICS WITH AFT CG

The handling characteristics of this airplane with aft cg (30.5 to 35% MAC) obtained by the fuel transfer system are excellent in the supersonic region. (Refer to CG FUEL TRANSFER SYSTEM, Section I.) The fuel transfer signal is applied by the air data computer at approximately Mach 1.2. After application of the transfer signal, some delay may occur, dependent upon which tanks are pressurized and the available air pressure.

NOTE

Forward transfer will also occur when descending through 13,000 (±500) feet.

If deceleration is extremely rapid, as might be the case when thrust is reduced to idle at Mach 1.2, the engine bleed air pressure will be insufficient to pump the transferred fuel forward before the airplane is subsonic. Stick forces will lighten when flying subsonic with most of the transferred fuel remaining aft; i.e. there is a tendency to decrease longitudinal stability and, with maximum aft cg (35% MAC) in the subsonic region, a tendency toward neutral longitudinal stability. The situation is compounded during load factor maneuvers at idle rpm initiated near the transfer Mach region. The airplane can decelerate from Mach 1.2 to .95 in ten seconds or less during these latter maneuvers and fuel transfer time will exceed 20 seconds. The net result of flying subsonic with an aft cg is (1) a possibility of overshooting the intended load factor and (2) if in the region of maximum g capability, a possibility of going beyond design limits. The above can be more pronounced due to the inherent transonic trim change. It is therefore recommended:

1. Load factor maneuvers should be avoided in the fuel transfer Mach range when flying with an aft cg.
2. Do not reduce engine thrust to idle until sufficient time has elapsed to insure completion of forward fuel transfer (approximately 20 seconds).

ARTIFICIAL FEEL SYSTEM INOPERATIVE

Internal failure of the elevator feel force regulator normally directs unregulated ram air into the elevator feel-force cylinder which will produce higher stick forces but the airplane remains controllable at restricted speeds. Clogging of the small intake will normally cause the rudder pedal forces to be low, being about the same as the pedal forces encountered on the ground. The abnormally low rudder pedal forces will require caution when operating at high indicated airspeeds to prevent overcontrolling, but will probably appear normal during approach and landing. However, it is possible for the intake to clog at such a time as to trap high "q" pressure, resulting in high rudder pedal forces for approach and landing. Elevator stick forces will be normal or low in most cases of "q" intake clogging. With the small intake clogged, elevator stick force will be normal below .88 Mach, and high above that range. With both "q" intakes clogged, elevator stick force will be low, with the extreme low force being the same as that experienced while on the ground. The low elevator stick force will feel approximately normal during approach and landing.

FLIGHT WITH DAMPERS OFF

This airplane can be safely flown throughout the speed envelope with the pitch and yaw dampers off. The airplane is dynamically stable at all speeds; however, lateral-directional oscillations (dutch roll) may occur when flying through the transonic and lower supersonic speed range. Quite often these oscillations give the impression of being divergent; however, this is not the case as the oscillations will damp with release of controls.

NOTE

A light, relaxed grip on the stick is much less likely to produce induced oscillations than a firm, rigid grip.

SPEED BRAKES

The hydraulically operated speed brakes may be used to slow the airplane at all speeds. When the speed brakes are extended, a slight nose-up tendency occurs. When the speed brakes are retracted, a nose-down tendency occurs.

DIVES

The airplane is capable of attaining high airspeeds and/or indicated Mach numbers during dives at steep angles. The Dive Recovery Chart (figure 6-3) demonstrates altitude lost during these conditions. To achieve the results predicted, it is necessary to have both hydraulic systems operating, speed brakes out, idle thrust, and to have entered the pullout altitude with the g-load already established. Failure of one hydraulic system reduces the airplane's dive recovery performance, but adequate control remains to perform normal dive recovery and rolling maneuvers. The g-load which can be attained at a given altitude and Mach number during the pullout without exceeding heavy buffet, maximum elevator deflection, or hinge moment limits is shown in the Load Factor Capability Chart (figure 6-4). This chart is based on a representative airplane gross weight and center of gravity location.

NOTE

Refer to Section V for operating limitations.

The Recommended Maximum Dive Angles Charts (figure 6-5) indicate ground clearance at a constant Mach and maximum g pullout for various altitudes, airspeeds, and dive angles. The curves indicate the recommended maximum dive angles to insure ground clearance without exceeding the design limit of the airplane. The Altitude Lost in Maximum Available Load Factor Recovery charts (figure 6-6) show altitude lost after entering an altitude with established maximum available load factor, Mach number, and dive angle. Charts are shown for idle, full military, and maximum thrust settings. To achieve the results predicted, it is necessary to have both hydraulic systems operating.

FLIGHT WITH EXTERNAL WING TANKS

General flight characteristics with external wing tanks installed are basically the same as for the clean airplane. There is a notable difference in elevator required for level flight during low-altitude, trasonic and supersonic operation. This difference is attributed to a large positive pitching moment (nose-up tendency) caused by the external tanks in this area of flight. In the transonic and low supersonic speed ranges (approximately Mach 0.95 to 1.2) below approximately 10,000 feet altitude, large forward stick displacement is required to maintain level flight. At altitudes below 7,000 feet, insufficient down elevator is available to maintain level flight. Flight tests of external tank jettisoning have shown that the tanks separate cleanly without striking the airplane structure when jettisoned in a level flight attitude at a constant airspeed. If only one tank separates from the airplane at any fuel load, the resulting roll can be easily controlled with aileron and rudder. Tanks will clear the airplane when jettisoned in a climb, dive, or accelerated flight if they are either full or empty. However, if tanks are partially full and are jettisoned in a climb, dive, or accelerated flight, they will probably strike the airplane. This results from fuel collecting in one end of the tank causing an undesirable tank rotation upon release. Therefore, partially full tanks should be jettisoned in level flight after airspeed has been stabilized for 10 to 15 seconds when time and conditions permit. Full or empty tanks may be jettisoned at any speed without appreciable change in aircraft attitude except in the transonic and low supersonic speed range (0.95 to 1.2 Mach number) below 25,000 feet. Tank jettison in this region will result in momentary negative load factors approaching or exceeding negative load factor limits. The maximum load factor of this post jettison oscillation occurs 0.6 to 0.9 seconds after ejection forces are felt. The resultant oscillations are self-dampening in three to four cycles and an attempt to lead or correct this initial tendency could result in violent pilot induced oscillations. Therefore, do not jettison the tanks in the 0.95 to 1.2 Mach number range below 25,000 feet.

ENGINE VIBRATION

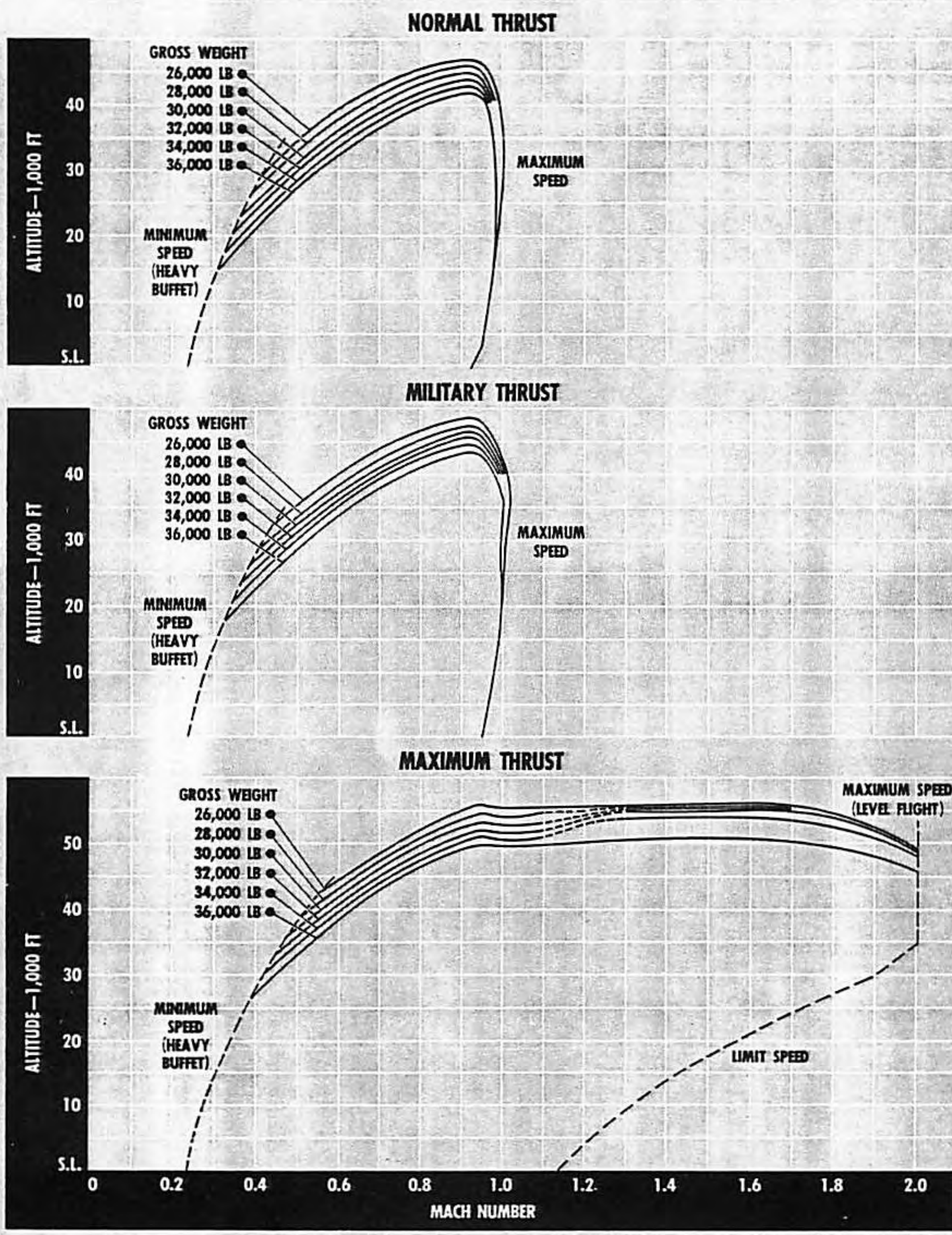
The engine has a vibration characteristic that is considered to be normal and does not jeopardize the structural integrity of the airplane. Engine vibration in subsonic and nonafterburner flight is more noticeable than at supersonic speed. Use of afterburner dampens out or reduces normal engine vibration. Coincident with engine vibration, a noise which can be described as a "moan" or low frequency oscillation, may be experienced. The noise is a normal condition and should cause no concern to the pilot.

minimum and maximum

MODEL: F-106A
DATE: 1 SEPTEMBER 1961
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE
1.0g LEVEL FLIGHT

ENGINE: J75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



48101-1

Figure 6-1 (Sheet 1 of 4)

speed capabilities

MODEL: F-106A
DATE: 21 FEBRUARY 1967
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO–360 GAL EXTERNAL TANKS
STANDARD ATMOSPHERE
1.0g LEVEL FLIGHT

ENGINE: J75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

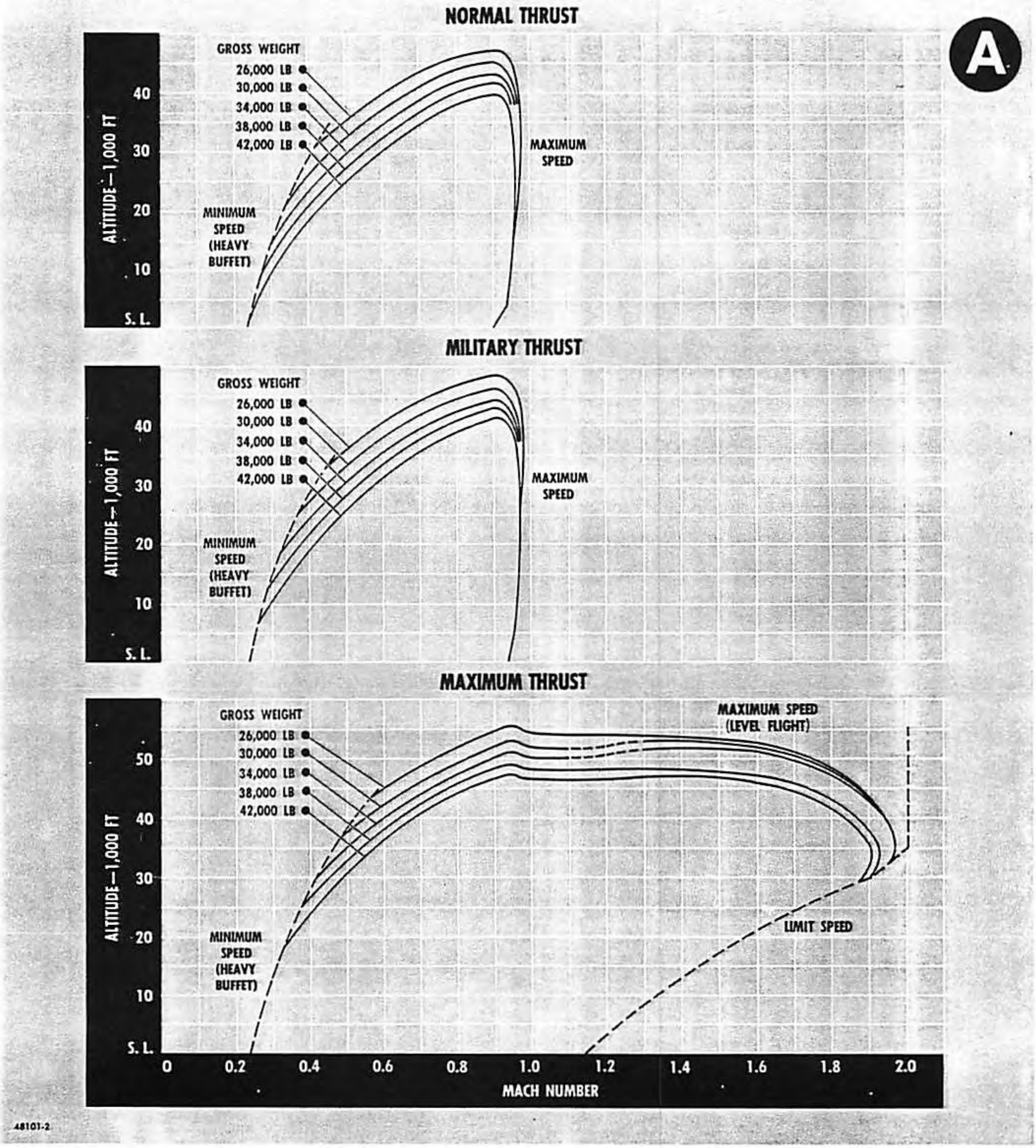


Figure 6-1 (Sheet 2 of 4)

minimum and maximum

MODEL: F-106B
DATE: 1 SEPTEMBER 1961
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE
1.0g LEVEL FLIGHT

ENGINE: J75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

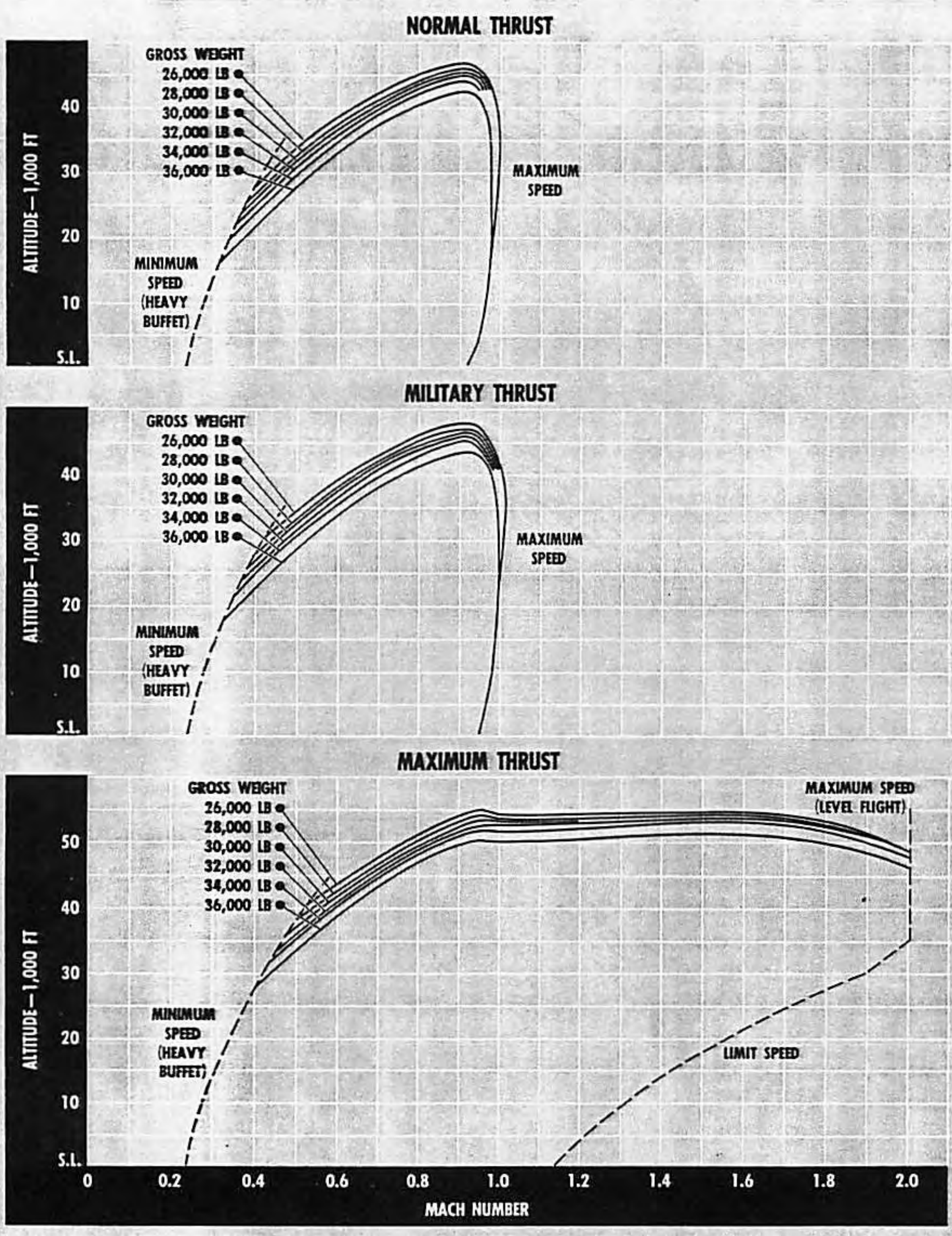


Figure 6-1 (Sheet 3 of 4)

speed capabilities

MODEL: F-106B
DATE: 21 FEBRUARY 1967
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO—360 GAL EXTERNAL TANKS
STANDARD ATMOSPHERE
1.0g LEVEL FLIGHT

ENGINE: J75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

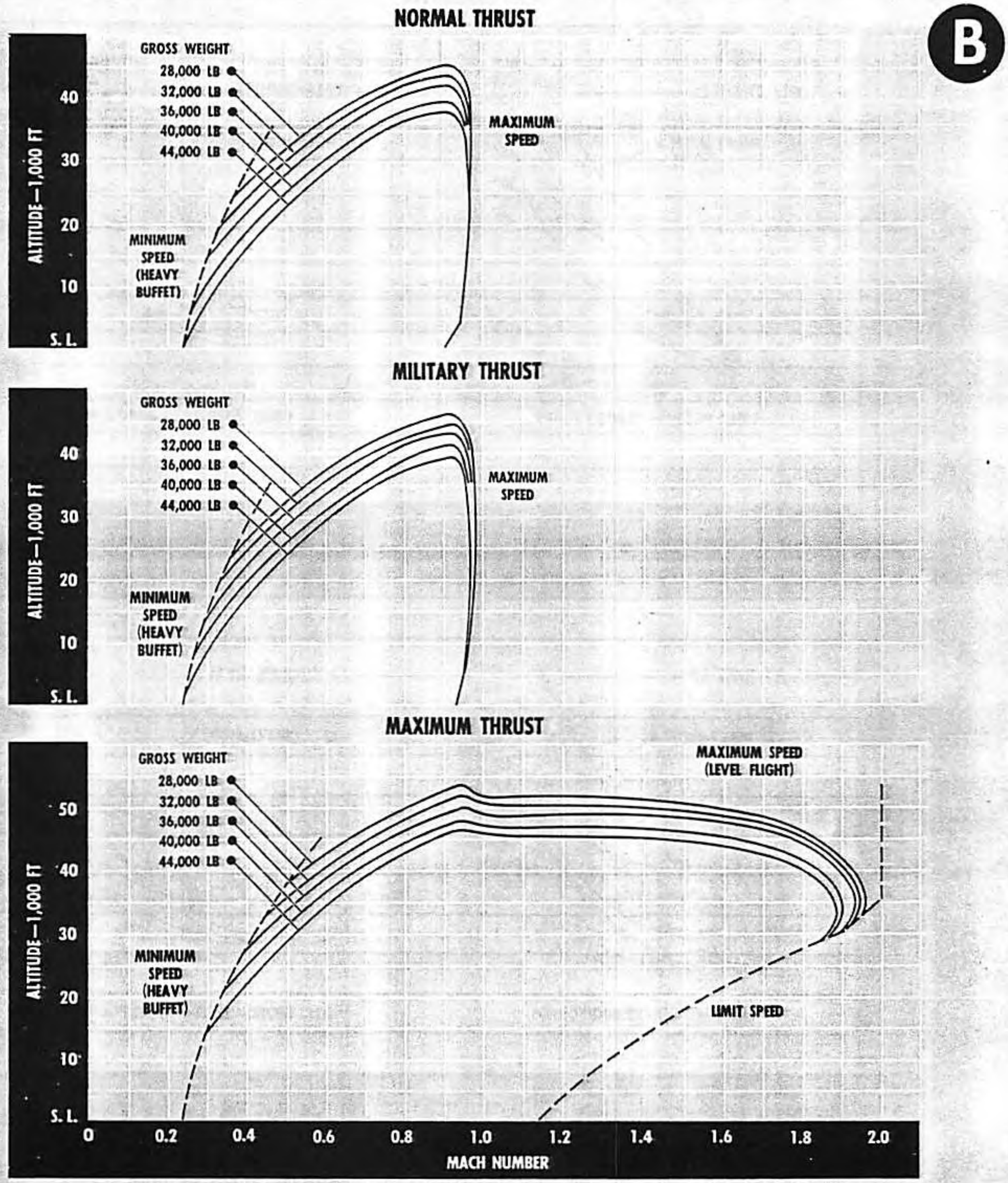


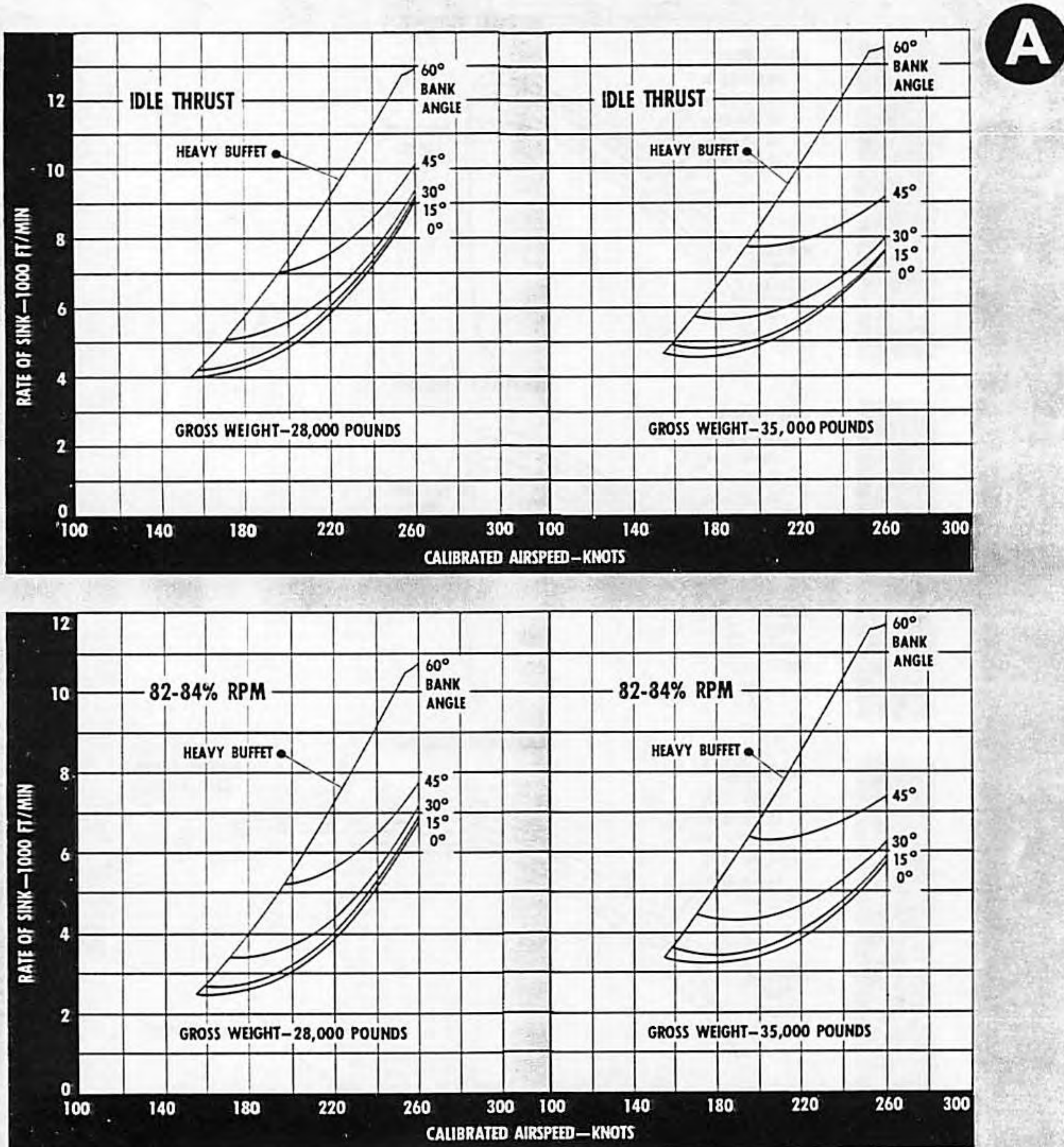
Figure 6-1 (Sheet 4 of 4)

low speed

MODEL F-106A
DATE: 1 SEPTEMBER 1961
DATA BASIS: FLIGHT TEST

CONFIGURATION: GEAR DOWN SPEED BRAKES OPEN • SEA LEVEL
STANDARD ATMOSPHERE

ENGINE: J75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE
INCREASE SINK RATES 10 FEET PER MINUTE FOR EACH DEGREE CENTIGRADE INCREASE FROM STANDARD AND 100 FEET PER MINUTE FOR EACH 1000 FEET OF ALTITUDE.

48093-1

Figure 6-2

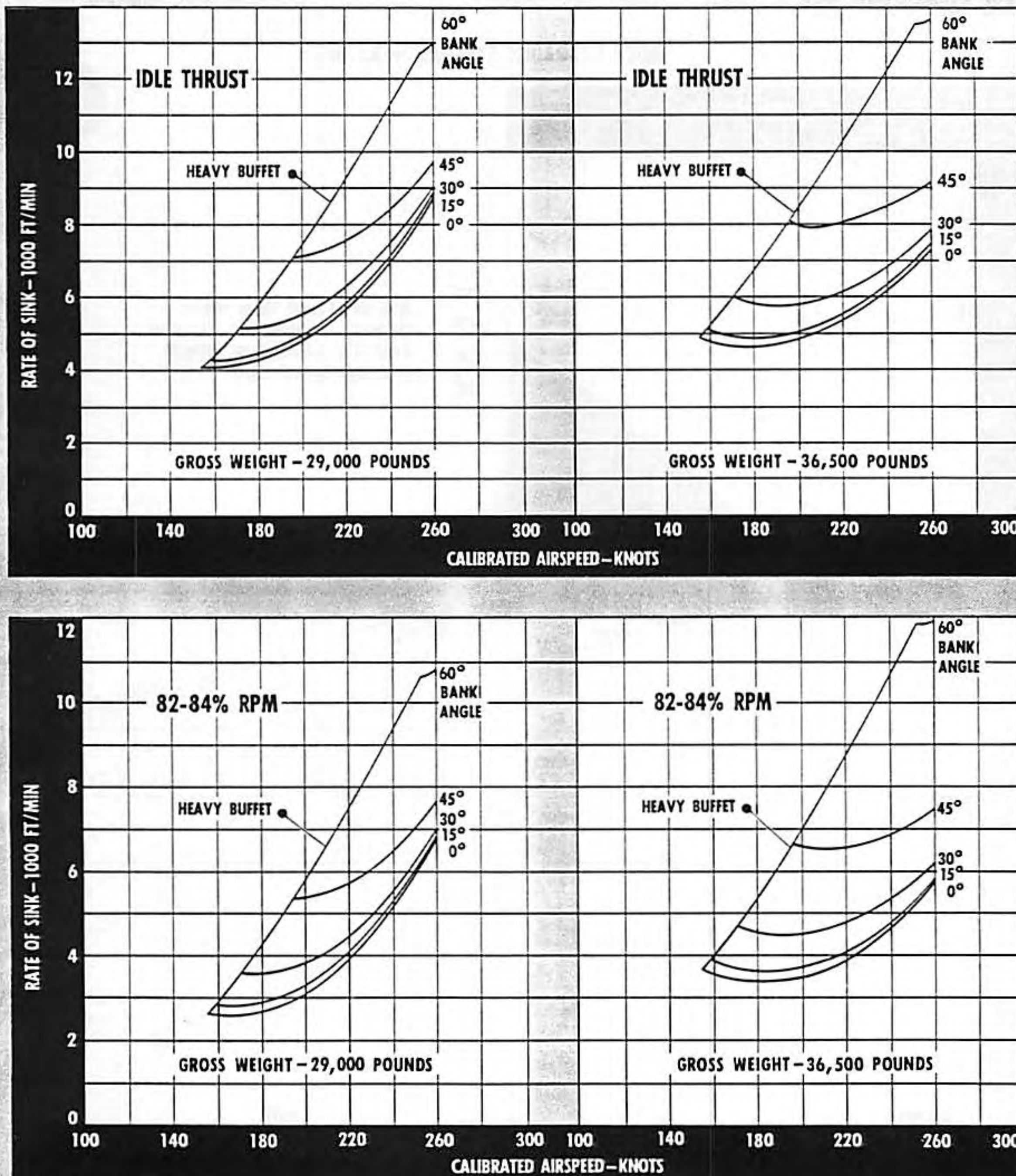
characteristics

MODEL F-106B
DATE: 1 SEPTEMBER 1961
DATA BASIS: FLIGHT TEST

CONFIGURATION: GEAR DOWN, SPEED BRAKES OPEN • SEA LEVEL STANDARD ATMOSPHERE

ENGINE: J75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

B



NOTE
INCREASE SINK RATES 10 FEET PER MINUTE FOR EACH DEGREE CENTIGRADE INCREASE FROM STANDARD AND 100 FEET PER MINUTE FOR EACH 1000 FEET OF ALTITUDE.

48091-2

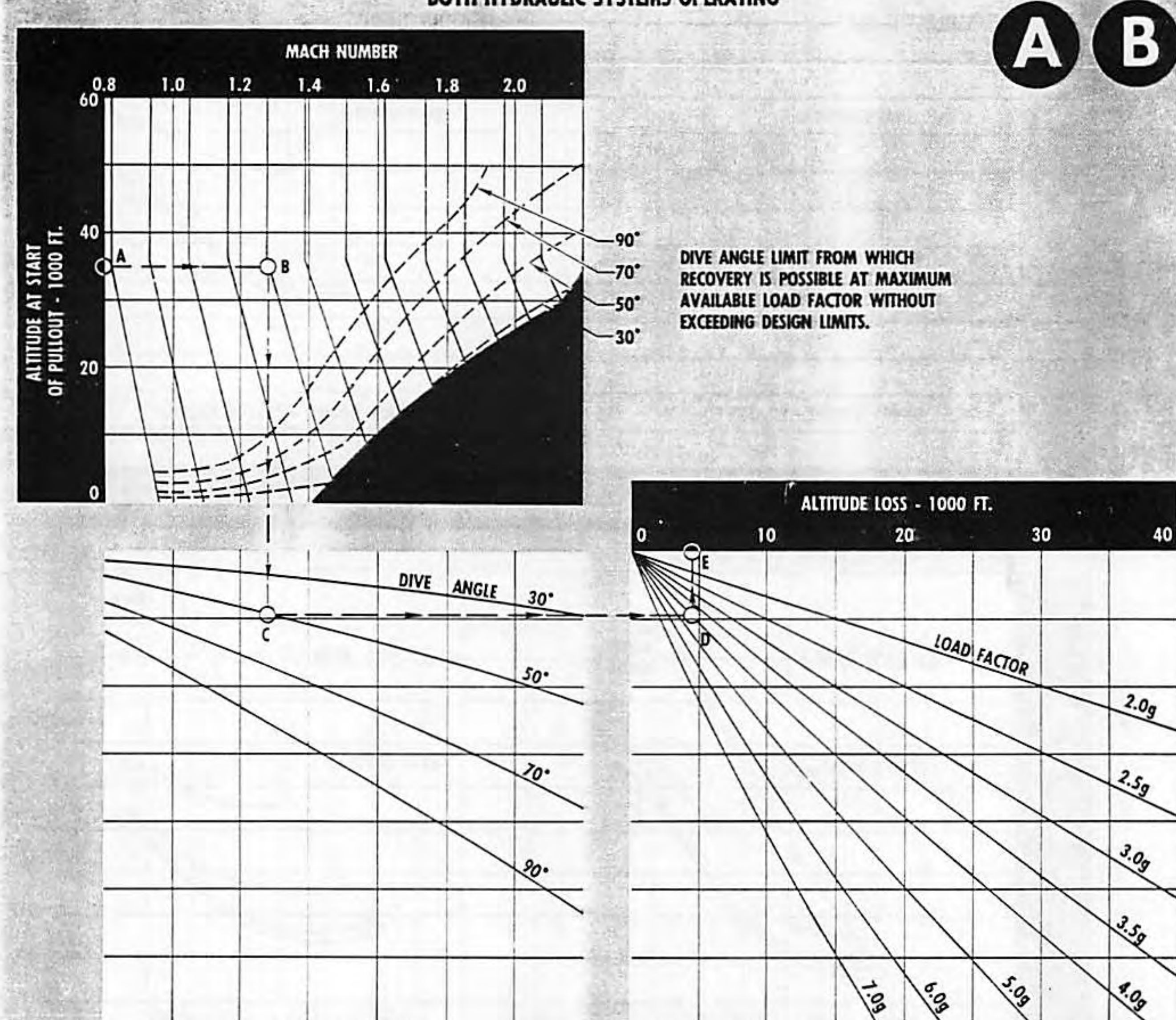
dive recovery chart

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE
IDLE THRUST

MODEL: F-106 A/B
DATE: 21 JUNE 1968
DATA BASIS: FLIGHT TEST

ENGINE: J75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

BOTH HYDRAULIC SYSTEMS OPERATING



EXAMPLE:

A - ALTITUDE AT START OF RECOVERY (35,000 FEET)
B - MACH NO. AT START OF RECOVERY (1.2)
C - DIVE ANGLE AT START OF RECOVERY (50°)
D - LOAD FACTOR DURING RECOVERY (4.0g)
E - ALTITUDE LOSS (4500 FEET)

NOTE:
- REFER TO SECTION V FOR OPERATING LIMITATIONS.
- FOR LOAD FACTOR COMPATIBILITY, REFER TO FIGURE 6-4.
- FOR RECOMMENDED MAXIMUM DIVE ANGLES, REFER TO FIGURE 6-5.

48102

Figure 6-3

load factor capabilities

MODEL: F-106 A/B
DATE: 21 JUNE 1968
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE

GROSS WEIGHT: 34,000 LB
CG: 27 PER CENT MAC

ENGINE: J75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

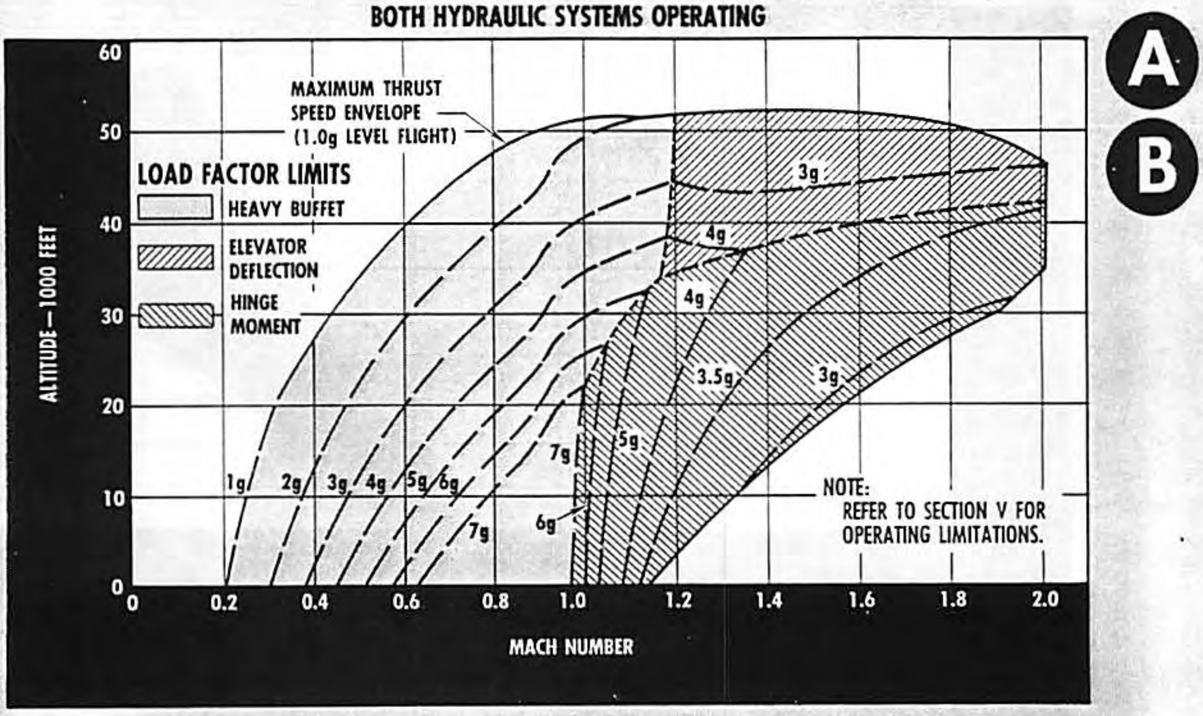


Figure 6-4

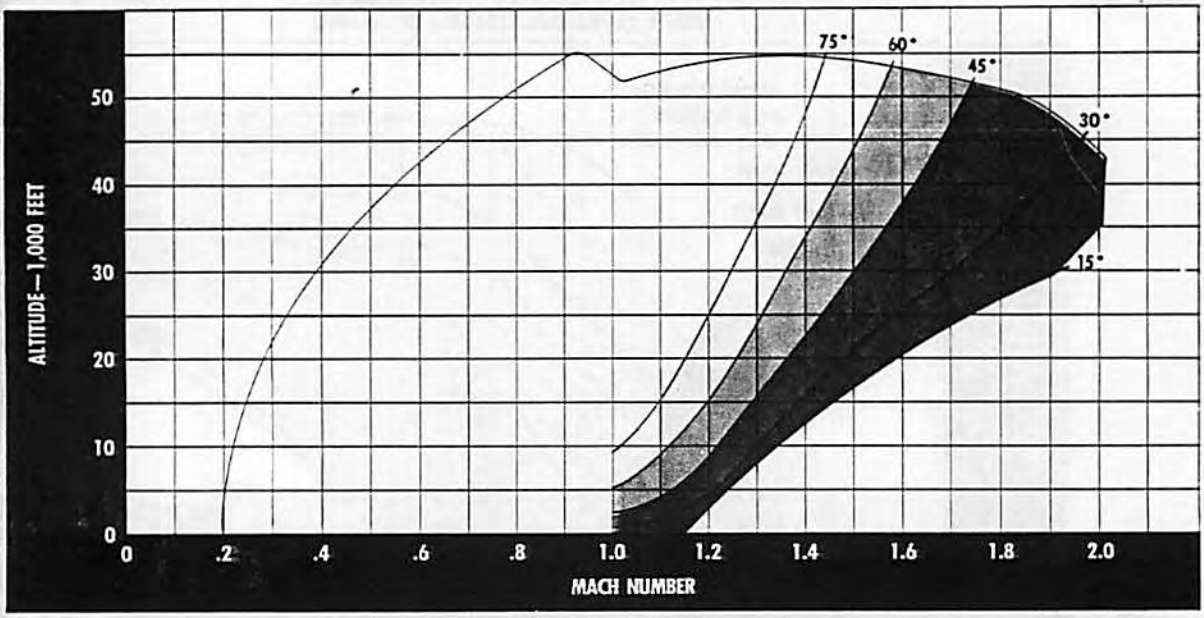
recommended maximum

MODEL: F-106A
DATE: 1 SEPTEMBER 1961
DATA BASIS: FLIGHT TEST

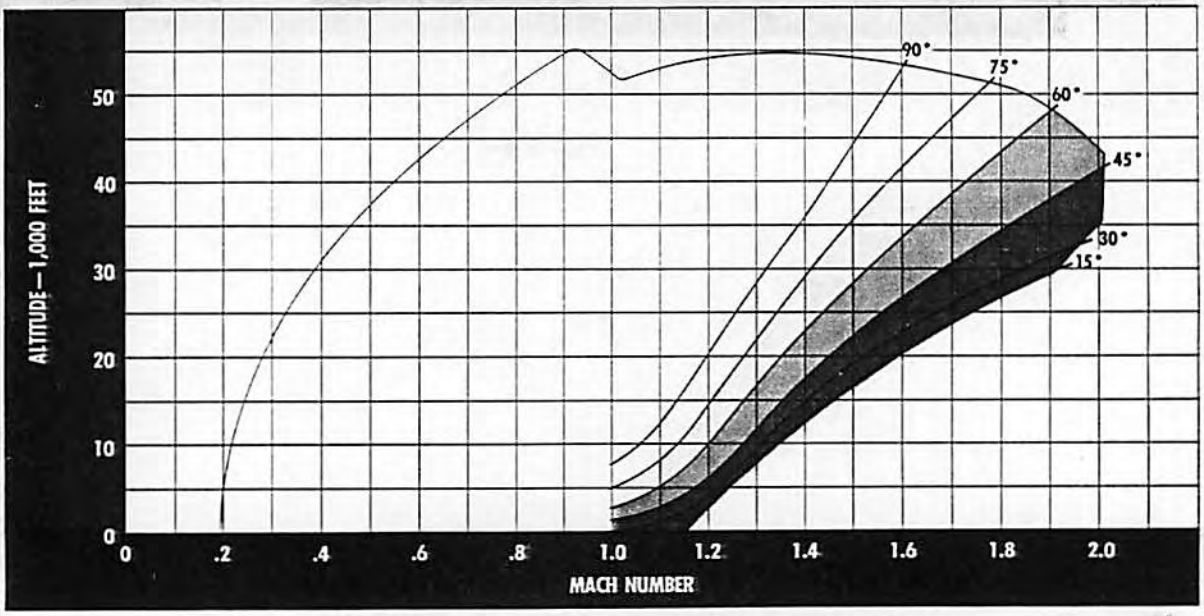
GROSS WEIGHT 29,776 LBS
STANDARD ATMOSPHERE

ENGINE: J-75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

SINGLE HYDRAULIC SYSTEM OPERATING NO FUEL TRANSFER (C.G. 26%)



BOTH HYDRAULIC SYSTEMS OPERATING NO FUEL TRANSFER (C.G. 26%)



EXAMPLE

- AT 35,000 FEET AND 1.5 MACH NUMBER WITH A FORWARD C.G. (26% MAC) AND WITH ONLY A SINGLE HYDRAULIC SYSTEM OPERATING, A DIVE ANGLE SHOULD BE NO GREATER THAN 50°.
- THE PULLOUT LOAD FACTOR IS BASED UPON THE AVERAGE MAXIMUM AVAILABLE (DUE TO HINGE MOMENT LIMITING) BETWEEN INITIATION-ALTITUDE AND FINAL ALTITUDE.

48174-1

Figure 6-5 (Sheet 1 of 3)

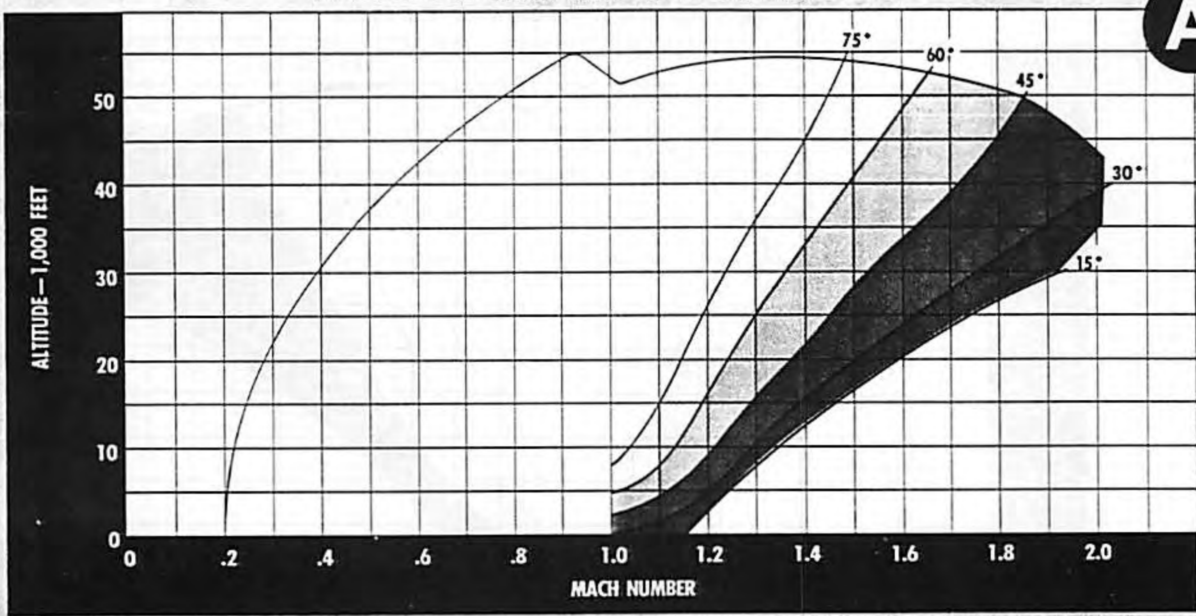
dive angles

MODEL: F-106A
DATE: 1 SEPTEMBER 1961
DATA BASIS: FLIGHT TEST

GROSS WEIGHT 29,776 LBS
STANDARD ATMOSPHERE

ENGINE: J75-17
FUEL GRADE JP-4
FUEL DENSITY: 6.5 LB/GAL

SINGLE HYDRAULIC SYSTEM OPERATING FUEL TRANSFER (C.G. 30.5%)



BOTH HYDRAULIC SYSTEMS OPERATING FUEL TRANSFER (C.G. 30.5%)

NOTE
- THESE CURVES INDICATE THE RECOMMENDED MAXIMUM DIVE ANGLE TO INSURE AGAINST EXCEEDING V_L, OR BELOW $M = 1.15$, TO ASSURE GROUND CLEARANCE
- DURING THE RECOVERY. THEY ARE CONSERVATIVE IN THAT A CONSTANT MACH NUMBER RECOVERY IS ASSUMED.

48374-2

Figure 6-5 (Sheet 2 of 3)

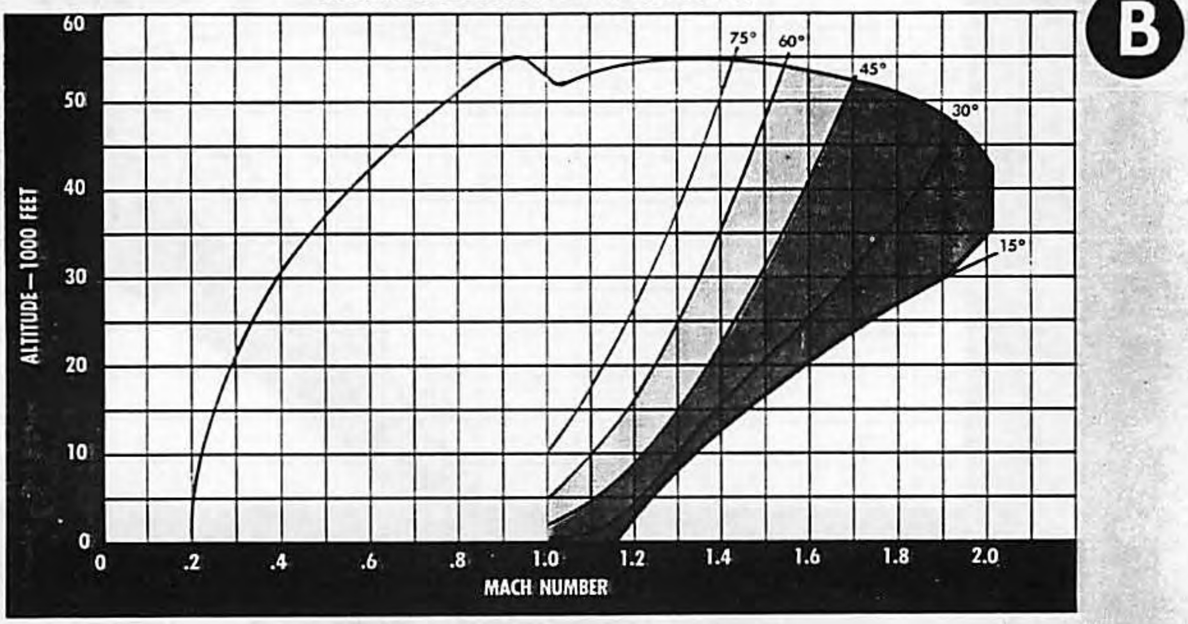
recommended maximum dive angles

MODEL: F-106B
DATE: 1 SEPTEMBER 1961
DATA BASIS: FLIGHT TEST

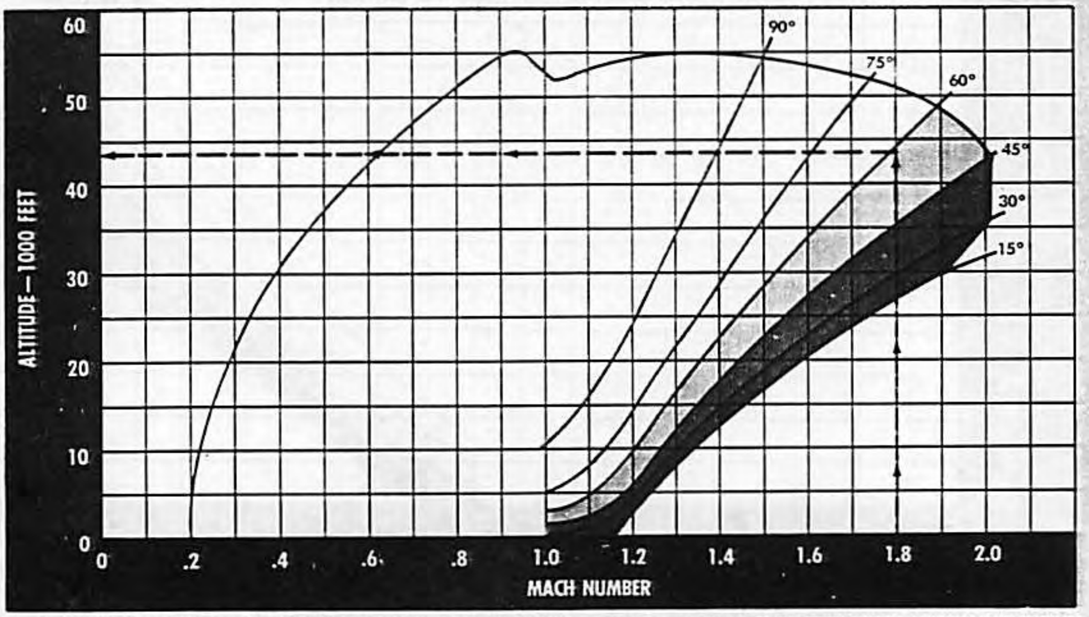
GROSS WEIGHT 31,576 LBS
STANDARD ATMOSPHERE

ENGINE: J-75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

SINGLE HYDRAULIC SYSTEM OPERATING (C.G. 26.8%)



BOTH HYDRAULIC SYSTEMS OPERATING (C.G. 26.8%)



EXAMPLE:

- AT 43,000 AND 1.8 MACH NUMBER AND BOTH HYDRAULIC SYSTEMS OPERATING, A DIVE ANGLE SHOULD BE NO GREATER THAN 60°.
- THE PULLOUT LOAD FACTOR IS BASED UPON THE AVERAGE MAXIMUM AVAILABLE (DUE TO HINGE MOMENT LIMITING) BETWEEN STARTING ALTITUDE AND FINAL ALTITUDE.
- THESE CURVES INDICATE THE RECOMMENDED MAXIMUM DIVE ANGLE TO INSURE AGAINST EXCEEDING V_L, OR BELOW M = 1.15 TO ASSURE GROUND CLEARANCE.
- DURING THE RECOVERY, THEY ARE CONSERVATIVE IN THAT A CONSTANT MACH NUMBER RECOVERY IS ASSUMED.

48174-3

Figure 6-5 (Sheet 3 of 3)

altitude lost in maximum available load factor dive recovery

MODEL: F-106 A/B
DATE: 21 JUNE 1968
DATA BASIS: FLIGHT TEST

IDLE THRUST

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE

GROSS WEIGHT: 34,000 LB
CG: 27 PER CENT MAC

ENGINE: J75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

BOTH HYDRAULIC SYSTEMS OPERATING

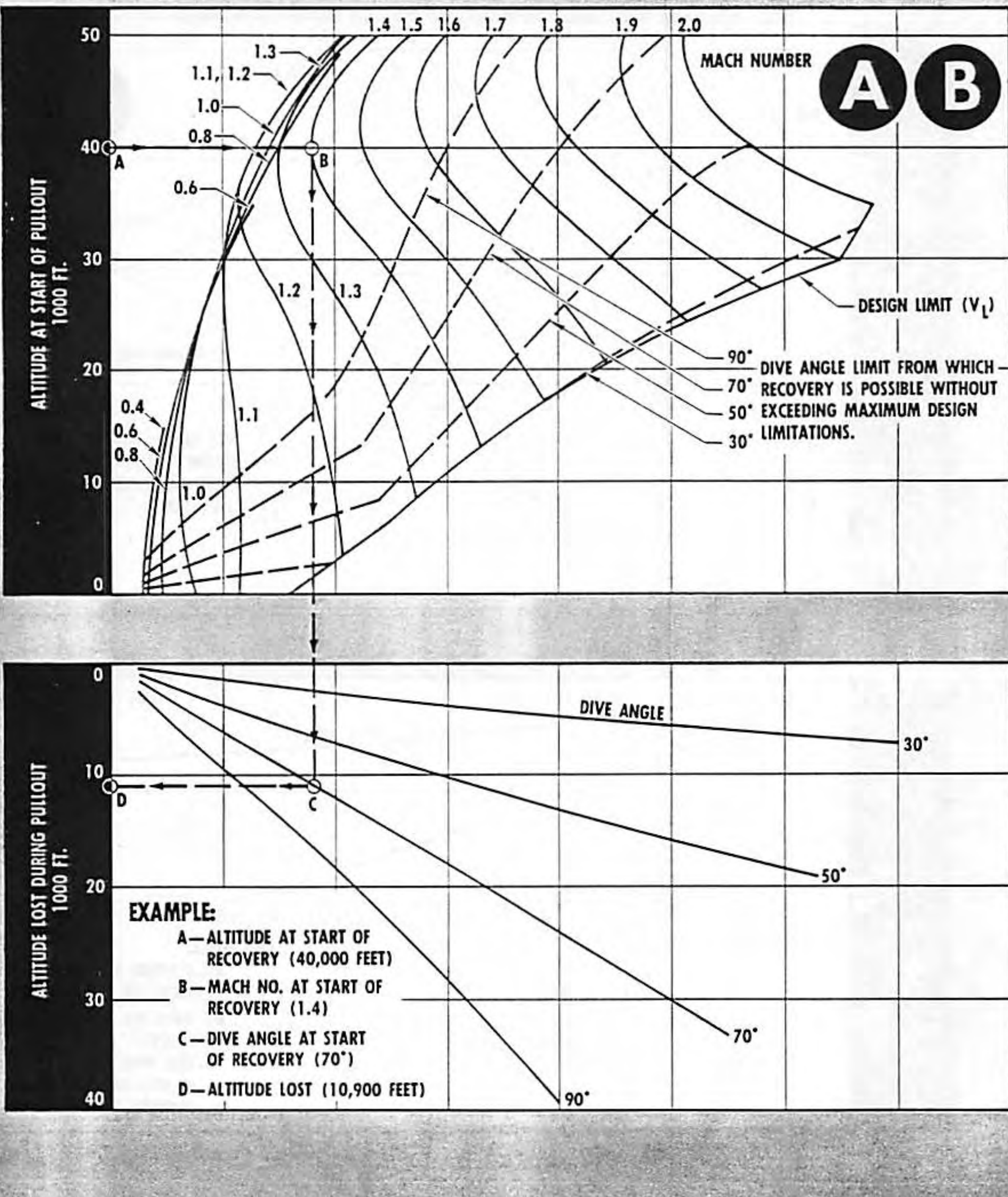


Figure 6-6 (Sheet 1 of 3)

altitude lost in maximum available load factor dive recovery

MILITARY THRUST

MODEL: F-106 A/B
DATE: 21 JUNE 1968
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE

GROSS WEIGHT: 34,000 LB
CG: 27 PER CENT MAC

ENGINE: J75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

BOTH HYDRAULIC SYSTEMS OPERATING

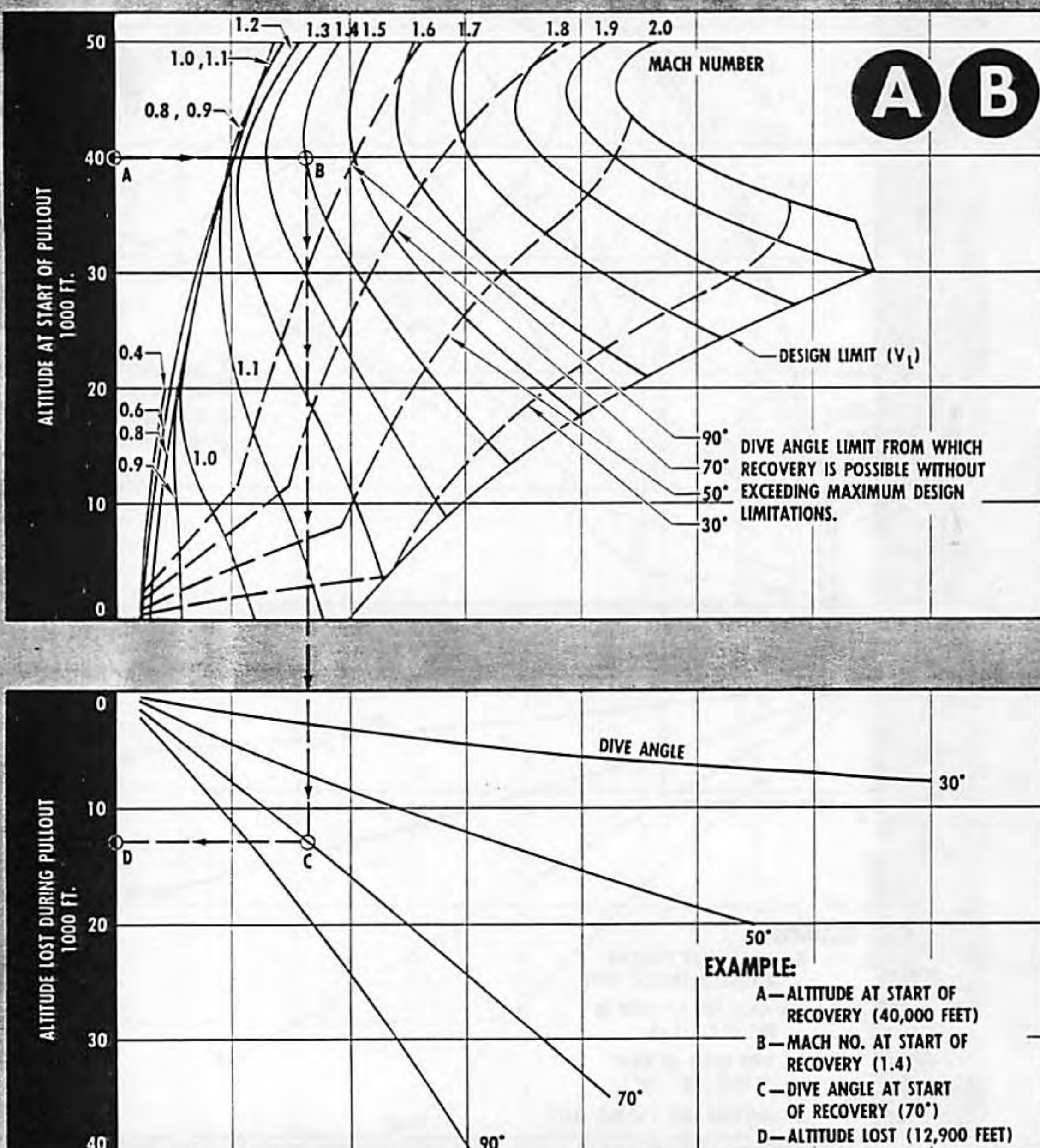


Figure 6-6 (Sheet 2 of 3)

altitude lost in maximum available load factor dive recovery

MAXIMUM THRUST

MODEL: F-106 A/B
DATE: 21 JUNE 1968
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD ATMOSPHERE

GROSS WEIGHT: 34,000 LB
CG: 27 PER CENT MAC

ENGINE: J75-17
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

BOTH HYDRAULIC SYSTEMS OPERATING

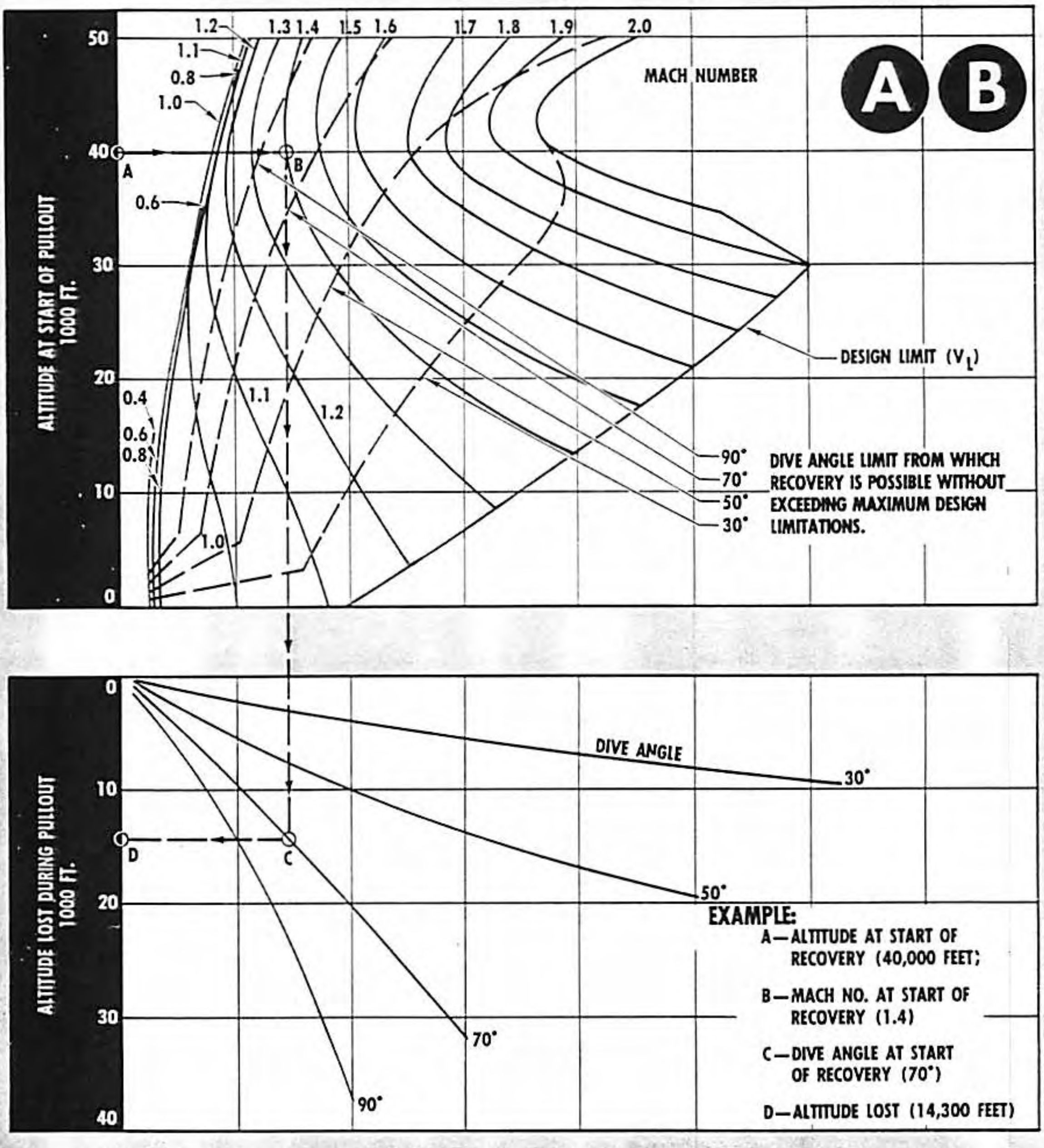


Figure 6-6 (Sheet 3 of 3)

systems operation

Section VII

TABLE OF CONTENTS

SMOKE FROM TAILPIPE AFTER SHUTDOWN

Normally the pressurizing and dump valve in the fuel control system will prevent the accumulation of fuel in the engine after shutdown. However, if for any reason fuel or oil should collect in the turbine housing during or immediately after shutdown, residual heat will vaporize the liquid and create a potential hazard. The situation is indicated by smoke or vapor exiting from the tailpipe. The engine should be cleared by the procedure given in Section II under STARTING ENGINE. All personnel should remain clear of the tailpipe for several minutes after engine shutdown and at all times when smoke or vapor is issuing from the nozzle.

USE OF WHEEL BRAKES

To minimize brake wear, the following precautions shall be observed insofar as practicable:

1. In order that brakes can be used as little and as lightly as possible, take full advantage of the length of the runway, utilizing aerodynamic braking to stop the airplane.
2. Do not apply brakes immediately after touchdown or at any time when there is considerable lift on the wings, to prevent skidding the tires. Heavy brake pressure will lock the wheels more easily if brakes are applied immediately after touchdown than if the same pressure is applied after the full weight of the airplane is on the tires. A wheel once locked in this manner immediately after touchdown will not become unlocked as load increases, as long as brake pressure is maintained. Brakes can stop the wheel from turning, but stopping the airplane is dependent on the frictional force between the tires and the runway. As the load on the tires increases, the frictional force increases, giving better braking action. During a skid, the frictional force is reduced, thus requiring more distance to stop.
3. If maximum braking is required after touchdown on a dry surface, lift should first be decreased as much as possible by dropping the nose before applying brakes. With the nose wheel on the ground, the wings provide negative lift which forces the wheels down against the runway. This procedure will improve braking action by increasing

the frictional force between the tires and the runway.

4. For minimum length landing rolls, a single, smooth application of the brakes with constantly increasing pedal pressure will result in optimum braking.
5. A minimum of 15 minutes should elapse between landings where the landing gear remains extended in the slipstream, 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 or cross-wind taxiing operation, or if a series of landings is performed.
6. At the first indication of brake malfunction, or if brakes are suspected to be in an overheated condition after excessive use, the airplane should be maneuvered off the active runway and stopped. The airplane should not be taxied into a crowded parking area. Overheated wheels and brakes shall be cooled before the airplane is subsequently towed or taxied. Peak temperatures occur in the wheel and brake assembly from 5 to 15 minutes after a maximum braking operation. In extreme cases, heat buildup can cause the wheel and tire to fail with explosive force or be destroyed by fire if proper cooling is not effected. Taxiing at low speeds to obtain air cooling of overheated brakes will not reduce temperatures adequately and can actually cause additional heat buildup.
7. The brakes should not be dragged when taxiing, and should be used as little as possible for turning the airplane on the ground.

COMPRESSOR STALL

A breakdown of compressor airflow is known as a compressor stall. Stalls may be encountered at any speed. Compressor stalls may occur as individual loud reports or as a series of loud reports in rapid order. Compressor stalls are accompanied by airframe vibration and in some cases, emission of vapor from the engine air intake duct.

NOTE

The engine can sustain numerous rapid order type compressor stalls without engine damage.

Compressor stalls may be recognized by loss of thrust reflected through engine instruments, rapid reduction or fluctuation of engine pressure ratio at a constant throttle position, or failure of rpm to increase during acceleration. A compressor stall may be accompanied by a rise in exhaust gas temperature. The possibility of encountering compressor stall is increased during high-altitude operation as the thinner air does not conform as easily to smooth airflow through the compressor section as does the heavier air at low altitude. This results in a more easily disturbed compressor airflow pattern. If a compressor stall is experienced, it should be noted on Form 781. Refer to STALL-BUZZ CORRECTIVE PROCEDURE, this Section.

STALL-BUZZ

The term "stall-buzz" refers to engine compressor stall combined with inlet duct pressure fluctuations which may occur at speeds greater than Mach 1.25. The engine compressor stalls are characterized by a series of loud metallic banging noises. The inlet duct pressure fluctuations result in a low pitched, rumbling, buzzing sound which is heard between individual compressor stall surges. Although loud and startling, this pnenomenon (stall-buzz) does not jeopardize the structural integrity of the airplane as demonstrated by flight tests under all conditions of stall-buzz. Along with the loud noise, airplane yaw will occur because of asymmetric duct pressure distribution (i.e., one duct swallowing a shock wave while the other duct is rejecting a shock wave). The steady yaw angle may reach two or three degrees and can induce a low roll rate. Lateral load factors associated with the yaw condition will tend to move the pilot to an awkward position in the cockpit. Flight tests have shown that a small amount of aileron opposing the roll will counteract both the roll and yaw. The yawing motion creates no structural problem. At speeds above approximately Mach 1.25, stall-buzz can be caused by rapid or improper throttle movement, variable ramp malfunction (resulting from secondary hydraulic system failure, air data computer failure, ac or dc nonessential power failure, ramp controller or actuator failure), and fuel control malfunction. Stall-buzz is most commonly encountered when shutting down afterburner near limit speed at high altitude. It can also be encountered when throttling excessively in afterburner or during afterburner relight.

THROTTLE MANAGEMENT TO AVOID STALL-BUZZ

1. Below Mach 1.25, no restriction on throttle movement.
2. Between Mach 1.25 and 1.7, no restriction on throttle movement in the afterburner range. Use slow throttle movement when retarding throttle aft of the minimum afterburner detent.

NOTE

Slow throttle movement is defined as that rate required with any jet engine when operating on the emergency fuel system.

3. Between Mach 1.7 and limit speed, use slow throttle movement between maximum and minimum afterburner.

NOTE

As altitude, Mach number, and angle of attack limits are approached, stall-buzz is increasingly likely to occur.

STALL-BUZZ CORRECTIVE PROCEDURE

1. When in afterburner, shut down afterburner.
2. When coming out of afterburner, maintain present throttle setting.
3. When maneuvering, reduce load factor and accomplish either step 1 or 2 above, as applicable.

NOTE

- If stall-buzz is caused by rapid or improper throttle movement, the stall-buzz will tend to subside within 2 to 5 seconds and before appreciable deceleration occurs.
- If stall-buzz is caused by variable ramp malfunction or by fuel control malfunction, stall-buzz will tend to continue until the airplane has decelerated to approximately Mach 1.3, which may require up to 10 to 12 seconds.
- Afterburner relight should never be attempted until stall-buzz subsides and the engine stabilizes.
- If stall-buzz is experienced in flight, it should be noted on Form 781.

CAUSES OF VARIABLE RAMP FAILURE

System failures which can affect variable ramp operation are: (1) secondary hydraulic system; (2) air data computer: (3) loss of ac or dc nonessential power; and (4) failure of the variable ramp components, such as the screw actuator. The most immediate indications of variable ramp failure, as caused by any one of the above, are compressor stall with associated inlet duct pressure fluctuations (stall-buzz), possible yaw, and loss of acceleration. Refer to STALL-BUZZ CORRECTIVE PROCEDURE, this Section.

SECONDARY HYDRAULIC SYSTEM FAILURE

Failure of the secondary hydraulic system will cause the ramps to remain in the position which exists at the time of failure. Fuel should be monitored to determine the range capabilities. Retract the ramps in accordance with VARIABLE RAMP EMERGENCY RETRACTION, Section III.

VARIABLE RAMP CONTROLLER UNIT FAILURE

Faulty variable ramp controller output signals may cause premature extension, a fixed position, or premature retraction of the ramps. If the ramps fail at an angle greater than the position required to maintain the scheduled inlet duct Mach, the airflow through the engine inlet ducts will become supercritical. This supercritical or highspeed airflow results in low performance (i.e., change of acceleration rate, decrease of engine pressure ratio, and possible compressor stall if allowed to continue), and may prevent the airplane from accelerating. Noise created by the turbulent airflow through the intake will normally accompany a supercritical intake condition. If the ramps are at an angle less than required, stall-buzz will result above approximately Mach 1.5. In either case, monitor fuel flow for range considerations and retract the ramps in accordance with VARIABLE RAMP EMERGENCY RETRACTION, Section III; if necessary.

AIR DATA COMPUTER FAILURE

Failure of the air data computer above Mach 1.25 will cause variable ramps to schedule improperly, resulting in stall-buzz, or a high duct Mach number and a resultant loss of performance at high speeds. Air data computer failure may be indicated by an "OFF" flag appearing in either the Mach indicator or the airspeed-Mach indicator or illumination of "CADC FAIL" warning light. Unless the ramps are retracted in accordance with VARIABLE RAMP EMERGENCY RETRACTION, Section III, the ramps will continue attempting to schedule at subsonic speeds where the ramps would normally be retracted. With this type of fail failure, sufficient thrust will be available to accomplish a safe go-around, even though the variable ramps are maintaining the highest programmed inlet duct Mach.

RAMP MECHANISM FAILURE

If any part of the ramp mechanism fails while the ramps are extended, use of the emergency system may not retract either or both ramps. The vari-ramp warning light will remain on at speeds below Mach 1.20. Care should be exercised during the landing phase because power available decreases as airspeed (ram effect) decreases; follow flameout approach procedure (figure 3-2). Airspeed should remain above 200 KCAS until starting flare.

NOTE

Should a failure of the crossover flex drive shaft occur, the left ramp may be extended without variable ramp warning light indication.

CG TRANSFER SYSTEM FAILURE Ⓐ

The cg transfer system should be monitored for proper operation by checking fuel quantity in the fuselage tank. If F tank fuel fails to transfer aft during supersonic flight above 13,500 feet, place the cg control switch in the FWD position to prevent inadvertent aft transfer of the fuel, and to ensure proper sequencing of fuel. If the fuel quantity-low warning lights illuminate too soon with fuel transferred aft, place the cg control switch in the FWD position and slow to a subsonic speed. If the low-level warning lights do not extinguish within one minute, only the fuel in the No. 3 tanks will be available. Land as soon as practicable.

AUTOMATIC FLIGHT CONTROL SYSTEM

NOTE

Refer to OTHER OPERATING LIMITATIONS, Section V, for limitations concerning AFCS operation.

The automatic flight control system provides pitch and yaw stability augmentation, automatic turn coordination, pilot assist (automatic pitch or altitude hold and heading or bank hold), and automatic steering for attack, navigation, or instrument landing approach. The system has five modes of operation: direct manual, yaw damper, pitch damper, assist, and automatic. In direct manual mode, no automatic flight control features are available. Automatic flight control modes are engaged, by using the flight mode selector switch, in steps as follows: first, yaw damper mode (yaw damping and turn coordination only); next, pitch damper mode is added and yaw damper mode remains engaged; then, because the damper system must be engaged before assist or automatic modes can operate, the assist mode can be engaged, and finally, the system can be placed in automatic mode. When the system is disengaged these steps are reversed. In yaw damper mode the automatic flight control system provides yaw damping and turn coordination. In the pitch damper mode the system provides pitch damping in addition to yaw damping and turn coordination. In the assist mode, electrical signals from the MA-1 aircraft and weapon control system are used to maintain airplane pitch attitude or altitude and hold either heading or bank angle. In the automatic mode, electrical signals from the MA-1 system are used to steer the airplane on the desired attack, navigation, or instrument approach and landing course. To prevent application of excessive loads, a pitch g limiting system is provided to automatically return the system to the yaw damper mode if a malfunction occurs while the system is in the assist or automatic mode.

NOTE

- When in ASSIST or AUTO mode, the automatic pilot function of AFCS can be overridden by control stick movement. However, if rapid, abrupt control stick movements are made to counteract any undesired airplane attitude, the flight mode selector switch may step down to PITCH DAMPER.
- The ASSIST and AUTO modes are automatically disengaged during the seat ejection cycle. The stick will return to the neutral position, thus eliminating interference with the pilot and seat.

Pitch G Limiter

The pitch g limiter prevents the automatic flight control system from subjecting the airplane to excessive pitch g forces when the system is in assist or automatic modes. The limiter is mechanized so that the system reverts to yaw damper mode when the limiter trips and the yaw damper is left engaged to provide adequate damping. The pitch g test switch may be used for an inflight check. Power is supplied to the pitch g limiter from the 115-volt ac and dc nonessential buses.

Pitch G Limit Test Switch. A pitch g limit test switch (26, figure 1-10) located on the left console is placarded "Pitch G Limit Test" has positions +G, −G, and a spring-loaded center position. On Ⓑ airplanes the pitch g limit test switch is located in the forward cockpit only. When the automatic flight control system is engaged, placing the switch to either the +G or −G position will actuate the g limiter, thus disengaging holding solenoids to return the system to the yaw mode. When the switch is held in the +G position, a force is simulated to actuate the limit system. In the −G position, a simulated force actuates the limit system. The pitch g limit test switch is operated by the 28-volt dc nonessential bus.

Damper Modes

In the damper modes, yaw mode and pitch mode, the damper system provides stability by damping out short period oscillations of the airplane. The damper system is hydraulically actuated and electrically controlled. Electrical signals from rate gyros, which sense the direction and velocity of airplane oscillations in yaw, pitch, and roll, operate hydraulic control valves which in turn regulate primary and secondary hydraulic system flow to the surface actuator valves. The surface actuator valves then move the control surfaces to stop the pitch or yaw oscillation of the airplane. When the oscillations have been stopped, the control surfaces return to their original position. These damping movements of the controls are superimposed on the pilot's control action and are not felt at the control stick. In the yaw damper mode, the system damps yaw oscillations and in addition actuates a turn coordinator which automatically maintains the airplane in coordinated flight. Aileron motion of the elevons and airplane roll rate are electrically measured and then modified as a function of altitude and airspeed by the air data computer. The sum of yaw rate, roll rate and aileron position is used to electrically control the hydraulic control valve which in turn controls the rudder surface actuator. The actuator then moves the rudder surface to maintain coordinated flight. In the pitch damper mode, the system continues to provide yaw damping and automatic turn coordination and also damps the pitch oscillations. The damper system receives power from the airplane ac and dc nonessential buses.

Assist Mode

The assist mode relieves the pilot of routine steering tasks by performing conventional autopilot functions such as: (1) pitch attitude or altitude hold and (2) heading or bank attitude hold. Primary inputs to AFCS in this mode are from the stable platform unit of the MA-1. When the flight mode selector switch is in the ASSIST position and if the airplane's wings are within 5° of level flight with the heading hold engaged at the time of momentary interrupt trigger release, signals from the MA-1 system maintain the heading. When the flight mode selector switch is in the ASSIST position and if the bank angle is between 6° and 60°, or heading hold is not engaged, when the momentary interrupt trigger is released, the bank angle will be maintained. Large changes of pitch attitude, bank attitude, or heading may be made by using the momentary interrupt trigger. Small changes of pitch attitude, and bank attitude (if prevailing bank angle exceeds 5°) may be made by using the elevon trim button to "beep" in signals to the AFCS. When beep trimming in bank attitude, if the trim button is released when the wings are within 5° of level flight and the heading hold switch is ON, the airplane will return to level flight (heading hold). If the trim button is released when the bank angle exceeds 5° or if the heading hold switch is OFF, the prevailing bank angle at the time of release will be maintained. When the trim button is used, the pitch and roll references will change at a programmed rate as long as the button is depressed and saturation is not reached. When engaging assist mode, if the system does not function properly, monitor circuits prevent engaging and return the system to pitch damper mode.

Automatic Modes

Data Link Modes. In data link modes, the computer supplies command signals to the automatic flight control system and to the cockpit indicators to provide automatic steering of the airplane in response to data link signals. Data link signals can be used to provide either maximum range performance or minimum time performance of the airplane.

Approach Mode (Automatic ILS). In this mode, the airplane is automatically flown throughout an ILS approach. Throttle control, landing gear actuation, and flareout and landing are accomplished manually. Throughout the approach, the course indicator and the approach horizon (some airplanes), or the HSI and ADI (other airplanes), should be monitored for proper operation of the system. The automatic approach consists of two phases: constant altitude and glide-slope. The constant altitude phase extends from automatic ILS engagement to glide-slope entry. The glide-slope phase begins with glide-slope entry and extends to termination of the automatic approach. Selecting the proper localizer frequency prior to initial automatic ILS engagement insures that localizer course deviation (as determined from the localizer beam signal) and the heading error signal (difference between the airplane compass heading and the selected runway heading) combine to provide horizontal steering signals for localizer beam entry, bracketing, and flying the center of the localizer beam. Engagement may be made within the localizer engage area, which is a circle of four miles radius centered 15 miles from the runway, with the following airplane heading limits:

a. If the approach is made from the right of the runway, the engage heading should be between 60° to the right and 120° to the left of the runway heading.

b. If the approach is made from the left of the runway, the engage heading should be between 60° to left and 120° to right of the runway heading.

Altitude at engagement should be 1500 feet above runway altitude as automatic ILS steering sensitivity during glide-slope descent is reduced as a function of sensed barometric pressure. Glide-slope entry is indicated by airplane pitch change in response to the glide-slope signal. Thereafter, glide-slope deviation signals are used for vertical steering in place of pressure altitude signals which served to maintain altitude during the constant altitude phase. At minimum approach altitude the momentary interrupt trigger should be held depressed and the landing completed manually.

CAUTION

- Flare and landing must be accomplished visually as the landing gear will not withstand the impact at 170 KCAS along the 3° glide slope.
- In automatic ILS modes, the flight mode selector switch will engage and remain engaged without a valid ILS signal. The localizer and glide-slope flags will be displayed and the airplane will turn to the heading set in the course readout window of HSI.

If there are interruptions after glide-slope interception, the flight mode selector switch will drop to ASSIST, and prevailing attitude will be maintained. If the signals are restored, Automatic ILS may be re-engaged but this should be done on a re-penetration for optimum performance.

CAUTION

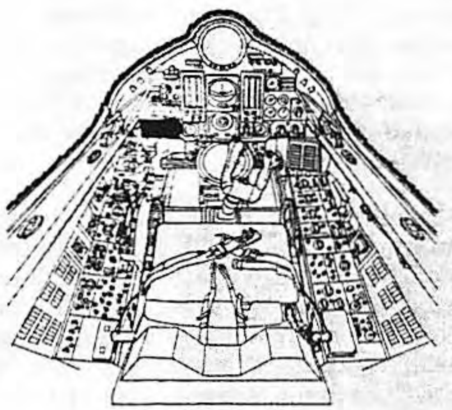
Automatic ILS should never be engaged at speeds above 300 KCAS or above 10,000 feet altitude.

Automatic Navigation Mode. In the automatic navigation mode, steering signals are supplied to the automatic flight control system to automatically fly the airplane to the selected homing point. The throttle must be manually controlled. If the automatic navigation mode becomes invalid (navigation display on scope disappears) the flight mode selector switch will revert to ASSIST. The flight mode selector switch also will revert to assist when reaching the homing point.

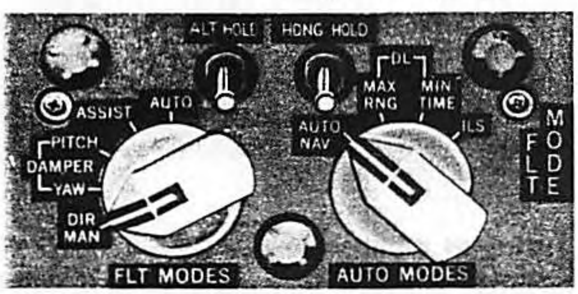
Flight Mode Selector Switch

A flight mode selector switch is located on the flight modes panel (figure 4-23). The five-position rotary switch is spring-loaded to the DIR MAN position. A holding solenoid retains the switch in the selected mode. In the other mode positions, YAW, PITCH, ASSIST, and AUTO, power from the selected mode circuit actuates the holding solenoid and holds the switch in the selected position. If power for the selected mode is lost, the holding solenoid will be deenergized and the switch will rotate toward the DIR MAN position until it reaches a position at which the mode engage circuits are operative and will supply power to energize the holding solenoid. At this point the solenoid will be energized, and the switch will be held in the operative position. For example: if the switch is in AUTO, and power from the automatic mode is lost, the switch will rotate first to the ASSIST position. If power is available for the assist mode, the switch will stop at ASSIST. If power is not available, the switch will move to the PITCH position. If power is available for pitch mode operation, the switch will stop at PITCH. If power is not available for pitch mode operation, the switch will rotate to YAW, and if power is not available for yaw mode operation, the switch will go to DIR MAN position. With the switch in DIR MAN position, no automatic flight control system functions are available and airplane control is direct from the stick through the hydraulic actuators to the control surfaces. When the switch is in the YAW position, the airplane flight control system operates in yaw damper mode. The airplane flight control system operates in pitch damper mode when the switch is placed in the PITCH position. With a climb commanded and the interceptor within 0.07 mach of the MA-1 climb schedule, the aircraft will be in mach hold. When within 500 feet or 15 seconds of the command altitude, altitude hold is entered. With AUTO NAV selected, if a descent is commanded and the aircraft mach is within 0.07 of the descent schedule, and aircraft will enter mach hold. If MAX RNG or MIN TIME is selected and a descent is commanded, the aircraft will enter descent attitude regardless of aircraft mach. Moving the switch to ASSIST position puts the airplanes flight control system in assist mode. When the switch is in AUTO position, the system operates in the automatic mode selected on the automatic modes

flight modes panels (typical)



AIRPLANES WITH CONVENTIONAL INSTRUMENT DISPLAY



AIRPLANES WITH INTEGRATED FLIGHT INSTRUMENT SYSTEM

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Figure 7-1

switch. If the automatic flight control system is disengaged by the pitch g limiter, the switch will rotate to the YAW position. The switch receives power from the 28-volt dc nonessential bus.

Altitude Hold Switch

On some airplanes, an altitude hold switch (figure 4-23) is located on the flight modes panel and has ALT HOLD and OFF positions. When the flight mode selector switch is in the ASSIST position and the altitude hold switch is in the ALT HOLD position, the AFCS will maintain the existing airplane altitude. If the airplane enters the transonic speed range (Mach 0.96 to 1.05) after altitude hold has been selected, the flight mode automatically changes from altitude hold to pilot assist. When the airplane leaves the transonic speed range the system automatically re-engages and maintains altitude hold at the re-engagement altitude. The altitude hold switch will revert to OFF whenever the airplane is trimmed in pitch attitude or any time the flight mode selector switch is moved from the ASSIST position.

Heading Hold Switch

On some airplanes, a heading hold switch (figure 4-23) is located on the flight modes panel and has HDNG HOLD and OFF positions. With the flight mode selector switch in ASSIST position and the heading hold switch in HDNG HOLD position, airplane heading is maintained within ±1°. Bank angle is maintained when the HDNG HOLD position is selected when in a bank angle of more than 5°. With the flight mode selector switch in ASSIST position and the heading hold switch in OFF position, airplane heading hold is not assured within ±5° and bank angle may be varied within ±60°.

Manual Mode Trigger (Momentary Interrupt Trigger)

A manual mode trigger (momentary interrupt trigger) is located on the control stick (figure 1-18). If the flight mode selector switch is in either assist or automatic mode, pressing the trigger will place the system in pitch damper mode. The system will remain in pitch damper mode while the trigger is held depressed. When the trigger is released, the system will return to the previously established mode, assist or automatic. The trigger interrupts power to assist mode components in the engage circuit but allows the flight mode selector switch to stay in assist or auto. (Power is maintained for pitch damper mode.)

NOTE Ⓑ

Regardless of which cockpit has control of AFCS, the manual mode trigger will operate in either cockpit.

On airplanes with air refueling capability, the manual mode trigger (Ⓑ front cockpit only) serves as a manual disconnect switch when the air refuel switch is ON.

Emergency Direct Manual Button (Emergency Damper Disconnect Button)

An emergency direct manual button (emergency damper disconnect button) is located on the control stick (figure 1-18). The automatic flight control system is automatically disengaged, including the damper modes, and the flight mode selector switch moves to the DIR MAN position when the button is depressed. The button interrupts 28-volt electrical power to the automatic flight control system engage circuitry.

NOTE Ⓑ

Regardless of which cockpit has control of the AFCS, the EDM button will operate in either cockpit and will place the AFCS in direct manual mode.

Flight Mode Failure Warning Light

The flight mode failure warning light (23, figure 1-30), located on the master warning light panel, illuminates and displays "FLIGHT MODE FAILURE" whenever the selected flight mode is inoperative. The light may be extinguished by depressing the manual mode trigger. When the selected flight mode fails, the flight mode selector switch automatically "steps back" to the next operative flight mode, and the flight mode failure warning light illuminates to indicate that the initially selected flight mode is inoperative and the switch has "stepped back." Although there has been no flight mode failure, the light will illuminate when power is first applied to the airplane and should be extinguished by momentarily depressing the manual mode trigger. Power is supplied by the dc essential bus.

AFCS Control Transfer Button, Monitor Light, and Indicator Ⓑ

The AFCS control transfer button, monitor light, and indicator are located on the AFCS transfer panel (figure 4-15) on the forward and aft consoles. The buttons are placarded "AFCS & ILS AUTO NAV Transfer." The monitor light is enclosed in the switch housing. Transfer of control of the flight mode selector switch, the automatic mode selector switch, the instrument landing and approach equipment, and the automatic navigation equipment is affected by this switch; i.e., the cockpit with control would receive valid TACAN ILS information to the flight director whereas the other would not. Before control can be transferred, the forward and aft flight mode selector switches and automatic mode selector switches must be at the same setting. The indicator placarded "Will Transfer" will read "NO" if these switches are not at the same positions. If the switches are at the same positions, the indicator will read "YES" or "OK," and AFCS control can be transferred forward or aft by depressing the transfer button.

The monitor light in the transfer button will illuminate in the cockpit which has control. On some airplanes*, it is necessary to hold the flight mode selector switch in the selected position in the cockpit to which control is being transferred when the transfer button is actuated. When transfer is effected, the flight mode selector switch in the cockpit from which control is being transferred will revert to DIR MAN position, and the flight mode failure warning lights will illuminate. To extinguish the lights, it is necessary to momentarily depress the momentary interrupt trigger in both cockpits. On other airplanes** the above condition does not exist and transfer may be accomplished by actuating the transfer button when the will transfer indicator reads "YES," or "OK." Power is supplied from the MA-1 electrical power supply system.

Automatic Flight Control System Preflight Check

After starting engine, the following checks may be performed prior to AFCS flights:

1. MA-1 power switch — RADAR STBY.
2. Flight mode selector switch—YAW then PITCH.
   Check dampers, note any transients.
3. Manual mode trigger — Depress and hold.
   Flight mode failure warning light should extinguish after the trigger is depressed.
4. Flight mode selector switch — ASSIST.
5. Manual mode trigger — Release.
   Check that no objectionable stick movement occurs when the trigger is released.
6. Longitudinal trim — Check.
   Check longitudinal trim by trimming NOSE UP and NOSE DOWN, checking

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that control stick follows trim button displacement. After trimming NOSE UP to obtain approximately three inches of stick displacement, depress the manual mode trigger; there should be little or no stick movement. Release momentary interrupt trigger and trim NOSE DOWN to obtain approximately three inches of stick displacement and again depress manual mode trigger; there should be little or no stick displacement. Release manual mode trigger.

7. Lateral trim — Check.
8. Manual mode trigger — Depress, manual flight checked.
   Depress manual mode trigger and check that manual flight control is available, then release the trigger.
9. Pitch g limit test switch — (±) G: note that flight mode selector switch moves to YAW. Return flight mode selector switch to ASSIST.
10. Altitude hold check (if altitude hold switch is installed).
    a. Altitude hold switch — ALT HOLD.
    b. Elevon trim switch — NOSE UP.
       Altitude hold switch should revert to OFF. Release elevon trim switch.
    c. Altitude hold switch — ALT HOLD.
       Control stick should immediately center and remain stationary without oscillations or drift from center position.
    d. Elevon trim switch — NOSE DOWN.
       Altitude hold switch should revert to OFF. Release elevon trim switch.
    e. Altitude hold switch — ALT HOLD.
       Control stick should immediately center and remain stationary without oscillations or drift from center position.
    f. Flight mode selector switch — PITCH DAMPER.
       Altitude hold switch should revert to OFF. Return flight mode selector switch to ASSIST.
11. Heading hold check (if heading hold switch installed).
    a. Heading hold switch — OFF.
    b. Elevon trim button—Trim ailerons approximately 1/3 side stick movement.
       Control stick should remain stationary without oscillations or drift toward the center stick position.
    c. Heading hold switch — HDNG HOLD.
       Control stick should immediately center and remain stationary without oscillations or drift from the center position.
12. Emergency direct manual button — Depress.
    Depress the emergency direct manual button and check that the flight mode selector switch moves to DIR MAN.
13. Takeoff trim — Set.

Operation of Automatic Flight Control System Engaging Procedure. Use the following procedure to engage AFCS:

1. MA-1 power switch — RADAR STBY or ON.
2. Flight mode selector switch — PITCH.
3. Trim for desired flight attitude.
4. Manual mode trigger — Depress.
5. Flight mode selector switch — ASSIST.
6. Manual mode trigger — Release.

AFCS is operating in the assist mode, maintaining pitch attitude and heading, or roll attitude depending on roll angle existing when the manual mode trigger was released. If a specific heading is desired instead of bank attitude, the wings should be within 5° of level flight. To make small corrections while in the assist mode, the elevon trim button is used as a trim control to feed in signals to vary the airplane attitude. Large corrections should be made by depressing the manual mode trigger to give the pilot manual control for faster, more positive corrections. For automatic steering, select the automatic mode desired, depress the manual mode trigger, place the flight mode selector switch in the AUTO position, and then release the manual mode trigger.

NOTE (B)

- On some airplanes*, due to possible intermittent malfunction of the flight mode selector switch and failure of the flight mode failure warning light to indicate switch position, a visual check of the switch should be made to ascertain that the switch is in the proper position after a change in position has been made. The mode switch may (1) step back to a mode lower than that selected, (2) "stick" and fail to step back when the manual mode trigger is depressed, or (3) step back as far as the DIR MAN position when the manual mode trigger is depressed.

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- On some airplanes** the automatic mode selector switches in both cockpits must be in the same DL position. Do not move the automatic mode selector switch in the noncontrolling cockpit from a DL position to an ILS position while the automatic mode selector switch in the controlling cockpit is in a DL position. If this were done, data link information would be erased from the digital computer. This could result in a mission abort unless data link information were retransmitted.

Transfer Procedure Ⓑ. To transfer AFCS control from one cockpit to the other, use the following procedure:

1. Flight mode selector switch — Same position in both cockpits.
2. Automatic mode selector switches — Same position in both cockpits.
3. Will transfer indicator — "YES" or "OK" (both cockpits).
4. Flight mode selector switches — Hold in place (both pilots) (some airplanes).

NOTE Ⓑ

On some airplanes both flight mode selector switches must be held while the AFCS control transfer button is depressed. Otherwise, both flight mode selector switches will rotate to the DIR MAN position.

5. Control Stick — Maintain grip until AFCS transfer is complete (pilot in "transferred from" cockpit).
6. AFCS control transfer button — Depress (pilot in "transferred to" cockpit).
7. Manual mode trigger — Depress momentarily (both pilots).

NOTE Ⓑ

- During AFCS control transfer in other than direct manual mode, the flight modes failure warning light will illuminate in both cockpits. To extinguish the light in both cockpits, the manual mode trigger is momentarily depressed.
- During direct manual AFCS control transfer, the flight modes failure warning light will not illuminate.

Disengage Procedure. Momentary disengaging is available by depressing the manual mode trigger. If it is not desired to reengage pilot assist immediately, either of the following actions will disengage the system:

1. Flight mode selector switch — PITCH or YAW.
2. Emergency direct manual button — Depress (disengages pitch and yaw damper systems).

Automatic Mode Selector Switch (Conventional Instrument Display)

An automatic mode selector switch, placarded "Auto Modes," is located on the flight modes panel (figure 4-23). The switch positions are: AUTO NAV, DL MAX RNG, DL MIN TIME, and ILS. With the switch in AUTO NAV, the autonomous navigation system is operative, and if the flight mode selector switch is in AUTO, the automatic navigation system will supply steering signals to the automatic flight control system when the computer has valid TACAN or DR capabilities. The DL MAX RNG position is selected for close control or modified close control data link operation. If, by SAGE command, the dominant mode is CC, the selection of either MAX RNG or MIN TIME yields the same result: the guidance calculations are performed by SAGE. However, if the dominant mode is MCC, the position of the switch governs the profile and tactics commanded by the MA-1 in attacking the target. The tactics associated with the MAX RNG selection are the same as CC. The DL MIN TIME (data link minimum time) position is selected for close control or modified close control data link operation. If the mode is CC, this selection will achieve the same result as a MAX RNG selection. When MCC is the dominant mode, the selection of MIN TIME will cause the MA-1 computer to utilize the CUTOFF tactic used by SAGE with Profile I selection. With the switch in ILS, the instrument landing and approach system will supply steering signals to the automatic flight control system if the flight mode selector switch is in AUTO position.

NOTE

- When the automatic mode selector switch is placed to ILS the computer memory circuit is cleared of all data link and wind.
- During ILS approach continue to monitor the HSI course deviation indicator.
- Ⓑ If ILS is selected in either cockpit during data link operation, data link will not be displayed.

***AF 57-2516 thru -2564.*

Display/Automatic Mode Selector Switch (Integrated Instrument Display)

A display/automatic mode selector switch, placarded "Disp/Auto Mode," is located on the flight modes panel (figure 4-23). The switch positions are: MAN NAV, AUTO NAV, DL MAX RNG, DL MIN TIME, ILS, and ILS APCH. With the switch at MAN NAV, autonomous navigation information cannot be utilized, and the command functions of the AMI, HSI, and AVVI must be set manually, if desired.

NOTE

When MAN NAV is selected on the display/automatic modes switch, the AUTO position cannot be selected on the flight mode selector switch.

With the switch in AUTO NAV, the autonomous navigation system is operative, and if the flight mode selector switch is in AUTO, the automatic navigation system will supply steering signals to the automatic flight control system when the computer has valid TACAN or DR capabilities. With the switch in ILS, the instrument landing and approach system will supply steering signals to the automatic flight control system if the flight mode selector switch is in AUTO position.

The DL MAX RNG position is selected for close control or modified close control data link operation. If, by SAGE command, the dominant mode is CC, the selection of either MAX RNG or MIN TIME yields the same result: the guidance calculators are performed by SAGE. However, if the dominant mode is MCC, the position of the switch governs the profile and tactics commanded by the MA-1 in attacking the target. The tactics associated with the MAX RNG selection are the same as CC.

The DL MIN TIME (data link minimum time) position is selected for close control or modified close control data link operation. If the mode is CC, this selection will achieve the same result as a MAX RNG selection. When MCC is the dominant mode, the selection of MIN TIME will cause the MA-1 computer to utilize the CUTOFF tactic used by SAGE with Profile 1 selection.

NOTE

- When the display/automatic mode selector switch is placed to ILS the computer memory circuit is cleared of all data link and wind.
- During ILS approach continue to monitor the HSI course deviation indicator.
- Ⓑ IF ILS or ILS APCH is selected in either cockpit during data link operation, data link will not be displayed.

CAUTION

ILS localizer course and command altitude must not be set until after selection of ILS display mode.

With the switch in ILS APCH, the pitch steering bar of the ADI is operative and will be displayed.

ARMAMENT

NOTE

Refer to Confidential Supplement, T.O. 1F-106A-1A, and Aircrew Special Weapon Delivery Manual, T.O. 1F-106A-29, for armament launching provisions, AIM missile description, AIR-2A rocket description, armament controls, weapon effects, armament limitations, jettison ballistics, and kill probabilities.

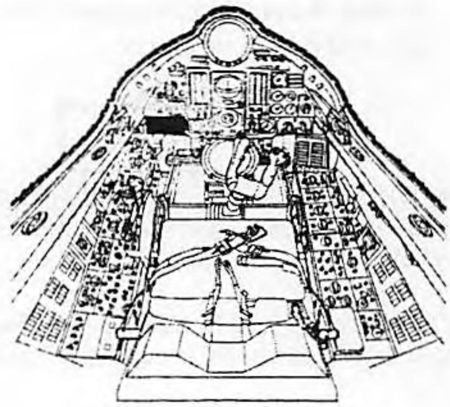
AIRCRAFT AND WEAPON CONTROL SYSTEM (MA-1)

NOTE

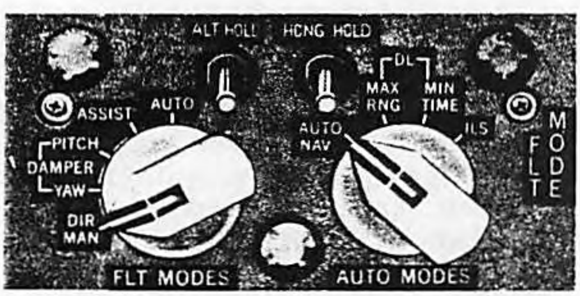
This manual covers two aircraft and weapon control systems, the MA-1 in the F-106A airplane and the AN/ASQ-25 in the F-106B airplane. The text distinguishes between the two configurations only where necessary and will refer to all switches, controls, etc., in both the F-106A and F-106B aircraft weapon and control system as the "MA-1."

For additional information on the MA-1 system refer to Confidential Supplement, T.O. 1F-106A-1A.

flight modes panels (typical)



AIRPLANES WITH CONVENTIONAL INSTRUMENT DISPLAY



AIRPLANES WITH INTEGRATED FLIGHT INSTRUMENT SYSTEM

48072

Figure 7-2

STABLE COORDINATE REFERENCE GROUP

The Stable Coordinate reference group, provides the MA-1 system with the vertical and directional reference required for navigation and flight control of the interceptor. The stable reference group consist of a platform, 3 gyros, an accelerometer and a system of 3 gimbals, the intermost of which holds the platform. The platform can precess 360° about the roll and azimuth axis and ± 85° about the pitch axis.

NOTE

Maneuvers in excess of plus or minus 85° pitch attitude will cause the stable platform to tumble, resulting in loss of MA-1 system navigation and attack capabilities.

Computer Subsystem

Operation of the MA-1 aircraft and weapon control interceptor system is based on use of a digital computer as a control element. The inflight function of the MA-1 digital computer is to solve the navigation and control problems which arise during a tactical mission. In solving problems, the computer follows a group of pre-recorded directions called a program. The total computer program consists of three separate programs: navigation, attack, and test. The navigation program furnishes command signals to fly the interceptor in the proper geometry for the navigation mode selected. The attack program provides final attack positioning and armament timing. It functions with DROT, ATOT, IR full action switch, a manual pursuit selection, or VIS IDENT within one and one quarter miles of target. The test program is entered by switch settings in the cockpit or the computer bay.

WARNING

If computer malfunction is suspected, monitor target range for accurate time-to-go.

In order to measure interceptor roll, pitch, and heading angles accurately, the vertical axis of the stable platform must be aligned with the local vertical and its heading must be established relative to computer grid north. Thereafter, the alignment is corrected by the computer for any changes in the direction of the local vertical and in the orientation of navigational coordinate system due to earth rotation and the movement of the interceptor over the earth's surface. When there is an interceptor attitude change, the change is simultaneously measured and converted into a useable electrical signal and distributed to the MA-1 system. Information thus obtained is used in stabilizing the radar antenna during search: provides stabilized radar displays; provides roll, pitch, and heading signals for AFCS operation; and allows fire control computations to be made in reference to a stabilized base.

Auto Nav System

NOTE

> The "auto nav" system refers to autonomous navigation. When fully automatic navigation is referenced, the term "automatic navigation" will be spelled out.

For auto nav, the digital computer computes the course to fly to a preselected homing point. These homing points are stored in the digital computer and are selected manually. The digital computer determines a course on a maximum range basis and supplies steering signals to the tactical situation display, the command altitude indicator, the Mach indicator, and on some airplanes, to the HSI, AMI, and AVVI. The computer grid coordinate system is based on a "computer origin" point. The stable platform is the only source of aircraft heading in this grid system. The automatic modes switch must be in the AUTO NAV position for operation of the auto nav equipment. Valid navigation begins upon reception of TACAN or when initial position is inserted. If the airplane is under automatic flight control, it is automatically flown on the prescribed course. The auto nav system will continue to dead reckon after the TACAN annunciator displays "OFF"; however, accuracy is degraded. The auto-nav equipment receives power from the MA-1 electrical power supply system.

NOTE

> If the grid reference of the stable platform is not correctly aligned by the pilot, the computer cannot solve navigation problems correctly.

Auto-Nav Homing Point Selector Switch. An auto-nav homing point selector switch (figure 4-17) is located on the right console. The switch has 20 positions, corresponding to the number of homing point coordinates which may be stored in the digital computer. The MA-1 computer is programmed for as many as 20 homing points of which homing point T will always be the position of the selected TACAN station. The MA-1 computer positions the TSD target symbol over the selected homing point. The homing point selector switch receives power from the MA-1 electrical power supply system.

The following information is given for each homing point selection: Homing point "A" - displays target marker circle and target elevation on MMST. Target mach on command mach marker and command mach readout window will be displayed for 9.6 seconds with DISP/AUTO MODE switch in MAX RNG or MIN TIME and with valid data link.

Homing point "T" - displays free air temperature in target altitude readout window (also the target altitude marker for integrated instruments) for temperatures below zero degrees centigrade.

Homing point "U" - indicates tactics selected on target altitude readout window (also target altitude marker for integrated instruments) with MAX RNG or MIN TIME selected on DISP/AUTO MODE switch. The following tactics and altitudes will be displayed:

50K-P (Pursuit)
45K-M (Manual)
40K-B (Beam)
35K-S (Stern)
30K-C (Cut-off)
0-NC (Non-committed)

AUTO-NAV Homing Point Profiles

Normal cruise for the interceptor in the AUTO-NAV mode is 0.92 mach at 40,000 feet (0.88 mach at 37,000 feet with tanks). Four AUTO-NAV homing point profiles are possible. Examples of these four possible situations and the interceptor response to each are as follows:

a. Normal profile—interceptor is above or below normal cruise altitude but distance to homing point is far enough to permit subsonic climb or descent to normal cruise altitude and cruise for ten nautical miles or more.

b. Intermediate profile—interceptor is below normal cruise altitude but so close to the homing point that ten nautical miles cruise, after climb ing to normal cruise altitude would be impossible. An "Intermediate" cruise altitude which will permit ten nautical miles cruise will be computed.

c. Close-in profile—interceptor, regardless of present altitude above or below homing point altitude, is so close to the homing point that no cruise altitude above homing point altitude would permit a ten nautical mile cruise: the homing point altitude would be immediately commanded. The climb or descent, as appropriate, to homing point altitude may still be in progress when homing point is reached.

d. Homing point T profile—selecting homing point T assigns the selected TACAN station to become the homing point position. The target bug will be positioned at TSD center. Command heading will be the wind-corrected heading to track directly to the station and command altitude will be present interceptor altitude until AUTO AFCS is engaged at which time command altitude will be frozen at AUTO AFCS engagement altitude. Command Mach will be descent Mach for the selected command altitude. When AUTO AFCS is engaged the interceptor will turn the shortest direction to track directly to the TACAN station and will maintain altitude. (The CDI will continue to show displacement from the selected TACAN radial). Heading correction will cease 25 seconds prior to arrival over the TACAN station. AFCS drops to ASSIST upon arrival at the station. No provisions are made for automatic tracking outbound from the TACAN station. If present TACAN is selected, the program will continue to AUTO navigate to the TACAN station if TACAN bearing and/or range becomes invalid. However, if manual TACAN is selected, the AFCS will drop to ASSIST 25 seconds after loss of TACAN bearing and/or range. The operation of this program utilizes existing auto-navigation logic and, therefore, requires that the stable table be properly erected. Target altitude will display outside air temperature

Operation of Auto Nav Equipment

1. MA-1 power switch — Radar STBY or ON and power annunciator displaying "OK."
2. Without previous initial positioning check TACAN annunciator for "OK" or "DR" indication.
3. Automatic mode selector switch — AUTO NAV.
4. Auto-nav homing point selector switch — Set (as desired).
5. For automatic flight, flight mode selector switch — AUTO.

DATA LINK

The cockpit data link equipment consists of the data link converter-receiver control panel, computer mode annunciator and data link antenna switch.

DATA LINK Converter-Receiver Control Panel

The DATA LINK converter-receiver control panel. (figure 7-3) controls the receiver frequency and aircraft address used for TDDL. Functions of the controls on the DATA LINK converter-receiver control panel are discussed in the following paragraphs.

a. CHANNEL Selection Controls.
The CHANNEL selection controls, consisting of a thumbwheel selector drum and INCR (increase) and DECR (decrease) pushbuttons, are used to select one of 26 preset channels. The drum may be positioned quickly by the thumbwheel or in steps of one channel by the INCR or DECR pushbuttons until the desired channel number appears in the

data link converter-receiver control panel

7-914-172A

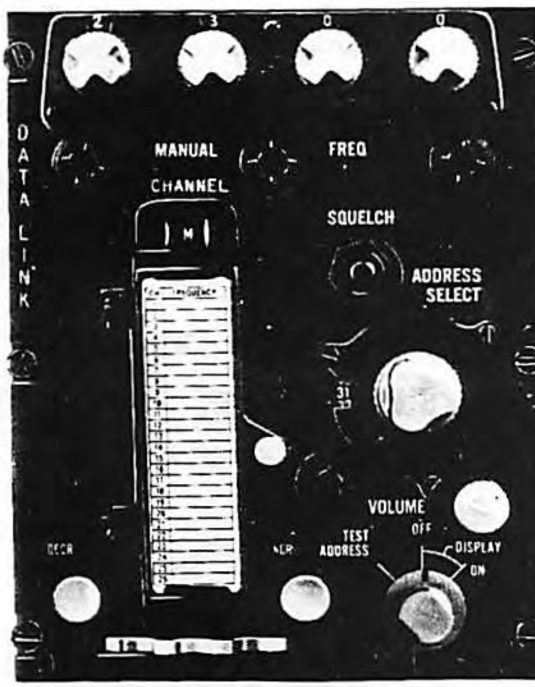


Figure 7-3

CHANNEL window. When the CHANNEL selection controls are positioned so that M appears in the CHANNEL window, the MANUAL FREQ controls may be used to manually select any one of a possible 1750 frequencies.

b. MANUAL FREQ Controls.
The MANUAL FREQ controls are used in conjunction with the backup voice reception capability of the data link receiver. The CHANNEL selection controls must be set so that the CHANNEL window indicates M before the MANUAL FREQ controls are operative. At this time, windows above the MANUAL FREQ controls open and each MANUAL FREQ knob is rotated until the desired digit appears in the window above the knob. The voice backup capability is not available if the MA-1 power is off or if the MA-1 power switch is set to EMER.

c. ADDRESS SELECT Control.
The ADDRESS SELECT control selects the desired aircraft address for data link. The control is rotated until the desired number appears in the window to the left of the knob. Capability is provided for 32 possible aircraft addresses.

d. TEST ADDRESS/DISPLAY OFF/DISPLAY ON Control.
The TEST ADDRESS/DISPLAY OFF/DISPLAY ON control determines the source of the aircraft address supplied to the data link converter. When this switch is set to DISPLAY OFF or DISPLAY ON, the time-division data link converter will accept only messages containing the aircraft address set up on the ADDRESS SELECT switch. When this switch is set to TEST ADDRESS, the data link converter will accept messages containing a predetermined test address.

e. VOLUME Control.
The VOLUME control permits adjustment of the audio level of data link voice signal in the headset. This control operates in conjunction with the VOLUME control on the COMM control panel.

f. SQUELCH Button.
The SQUELCH button, when pressed and held down, disables the noise-limiting circuits in the data link voice receiver. The operation of the receiver may be checked by increasing the setting of the VOLUME

control and pressing the SQUELCH button. If the receiver is operating, a noise will be heard in the headset.

COMPUTER MODE ANNUNCIATOR

The computer mode annunciator, (4, figure 1-9) may display any one of three indications (DL, DR, or "barber pole") to show the currency of data link. DL is set in the annunciator when CC is primary. DR is set when MCC is primary. When the messages are older than one minute the barber pole is displayed. When the barber pole is displayed the data link indications are dead reckoned.

data link antenna switch

7-914-335

Figure 7-4

NOTE

When entering an attack mode a barber pole is displayed and commands are frozen until a Close Control Message is received. Receipt of Modified Close Control Messages does not change the displays.

DATA LINK ANT Switch

The DATA LINK ANT switch, (figure 4-16D) controls the directional capability of the data link antenna. The switch is located on the right console aft of the canopy switch.

The DATA LINK ANT switch should be set to the position which gives the best reception of data link during a particular phase of flight. Setting th switch to the TAIL ONLY position decreases the reception sensitivity of the data link receiver in the forward direction.

auto nav homing point selector

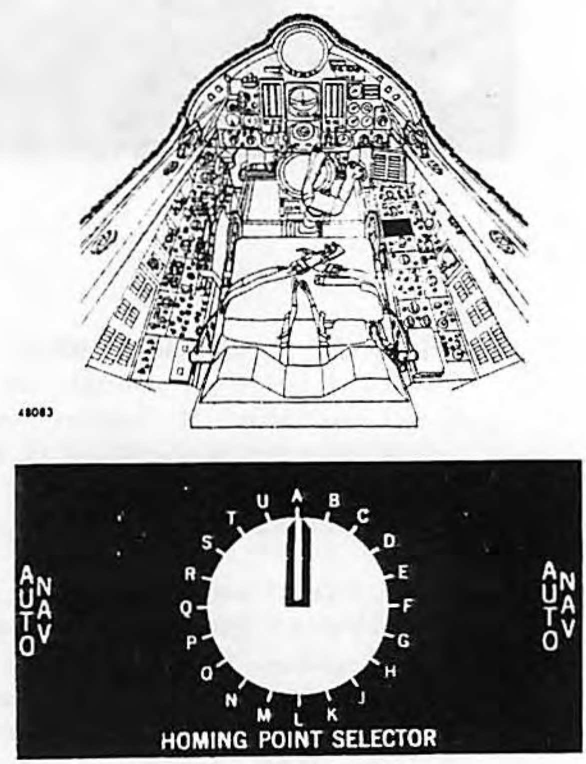


Figure 7-5

MODE SWITCHING FEATURES AND RESULTANT DISPLAYS

The instrument displays presented by the integrated flight instrument system depend upon settings of various cockpit switches. The primary determinant of the operating mode is the display/automatic mode selector switch. Dependent upon the setting of this switch, the instruments will display navigation, data link, or ILS indications and commands.

NOTE

> Switching of the flight mode selector switch from one position to another will not affect the basic display as presented thru the display/automatic mode selector switch.

Supporting the display/automatic mode selector switch in distribution of signals to the various instrument functions are the heading selector switch and the bearing selector switch. Figure 7-8 shows the AMI and AVVI presentation as affected by the positions of the display/automatic mode selector switch. Figure 7-9 shows the ADI presentation as affected by the automatic mode selector switch and the heading selector switch. Figure 7-10 shows the HSI presentation as affected by the positions of the display/automatic mode selector switch, heading selector switch, and the bearing selector switch.

NOTE

- If the display/automatic mode selector switch is moved from any one major mode (NAV, DL, or ILS) to another major mode, both the heading selector switch and the bearing selector switch will revert to NORMAL and NORM respectively. These switches will not revert if merely switching within the major mode; i.e., MAN NAV TO AUTO NAV (or vice versa), except that when switching from ILS to ILS APCH (or vice versa) the heading selector switch will revert to NORMAL. It is always possible to again select the MANUAL position if desired.
- Whenever the display/automatic mode selector switch is set to an ILS mode, the computer memory circuit is cleared of all data link information.
- During MCC Data Link Navigation, any repositioning of the Disp/Auto mode selector switch causes the computer to reselect the appropriate tactics and profile for the current intercept conditions.

Once the various switching features are established, as described in the previous paragraph and in figures 7-8, 7-9, and 7-10, the integrated flight instrument system can be applied to actual mission functions. The basic modes which can be selected are navigation (TACAN), data link, ILS and ADF. All other modes, such as takeoff, climb-out, penetration, and GCA, can be considered as submodes of the basic modes listed above. The basic modes, including ADF, are presented in the following paragraphs.

NAVIGATION MODE

In the navigation mode, inputs to the integrated flight instruments are determined by settings of the display/automatic mode selector switch, the heading selector switch, and the bearing selector switch.

Manual Navigation

In the manual navigation mode, the HSI is the primary reference instrument. All of the command indicators are visible on the cockpit instruments and can be set to any desired positions, since command signals are not present.

NOTE

Whenever the display/automatic mode selector switch is placed to the MAN NAV position, it is not possible to select AUTO on the flight mode selector switch.

1. MA-1 power switch—RADAR STBY or ON.
2. Flight mode selector switch — As desired.
3. Display/automatic mode selector switch — MAN NAV.
4. Heading selector switch — As desired.

airspeed - mach indicator and altitude - vertical velocity indicator presentations

		DISP/AUTO MODE SELECTOR SWITCH			
		MAN NAV	AUTO NAV	DL MAX RNG OR DL MIN TIME	ILS OR ILS APCH
AIRSPEED-MACH INDICATOR	COMMAND MACH	MANUALLY SET	INDICATES COMMAND MACH FOR CLIMB, CRUISE, OR DESCENT SCHEDULES	COMMAND MACH FROM DATA LINK STATION OR DIGITAL COMPUTER	NOT USED
AIRSPEED-MACH INDICATOR	COMMAND AIRSPEED (KNOTS)	SET MANUALLY FOR DESIRED PURPOSE (OR SIDE DETENT) SO COMMAND AIRSPEED MARKER SLAVES TO LUBBER LINE AND CAS READS IN AIRSPEED READOUT WINDOW.			
ALTITUDE-VERTICAL VELOCITY INDICATOR (AVVI)	COMMAND ALTITUDE	MANUALLY SET	COMMAND ALTITUDE FOR CLIMB, CRUISE, OR DESCENT SCHEDULE	COMMAND ALTITUDE FROM DATA LINK STATION OR DIGITAL COMPUTER	MANUALLY SET
ALTITUDE-VERTICAL VELOCITY INDICATOR (AVVI)	TARGET ALTITUDE	NOT USED	HOMING POINT ALTITUDE	TARGET ALTITUDE	NOT USED

48204

Figure 7-8

attitude director indicator presentations

	DISP/AUTO MODE SELECTOR SWITCH							
	MAN NAV		AUTO NAV		DL MAX RNG OR DL MIN TIME		ILS OR ILS APCH	
	HEADING SELECTOR SWITCH*				HEADING SELECTOR SWITCH*		HEADING SELECTOR SWITCH*	
	MAN	NORMAL	MAN	NORMAL	MAN	NORMAL	MAN	NORMAL
BANK STEERING BAR***	FLYING SELECTED HEADING	OUT OF VIEW	** FLYING SELECTED HEADING	** COMMAND HEADING	** FLYING SELECTED HEADING	** COMMAND HEADING	FLYING SELECTED HEADING	LOCALIZER COURSE
PITCH STEERING BAR***	OUT OF VIEW EXCEPT WHEN DISP/AUTO MODE SELECTOR SWITCH IS IN ILS APCH (OR IN ILS AFTER REACHING GLIDE SLOPE WHEN MAKING AN AUTOMATIC APPROACH), AND HEADING SELECTOR SWITCH IS IN NORMAL.							
ILS GLIDE SLOPE INDICATOR	OUT OF VIEW						INDICATES POSITION RELATIVE TO GLIDE SLOPE	
ILS LOCALIZER OFF FLAG	OUT OF VIEW						IF OUT OF VIEW, INDICATES VALID LOCALIZER RECEIVER INFORMATION	
GLIDE SLOPE WARNING FLAG	OUT OF VIEW						IF OUT OF VIEW, INDICATES VALID GLIDE SLOPE RECEIVER INFORMATION	
POWER OFF WARNING FLAG	OUT OF VIEW (EXCEPT IN CASE OF COMPLETE AC ESSENTIAL BUS POWER FAILURE).							

* IF THE FLIGHT MODE SELECTOR SWITCH IS PLACED TO AUTO, THE HEADING SELECTOR SWITCH WILL REVERT TO NORMAL.

** IF THE FLIGHT MODE SELECTOR SWITCH IS PLACED TO AUTO, THE BANK STEERING BAR WILL DISAPPEAR.

*** ADI PITCH STEERING AND BANK STEERING BARS ARE ON AC NONESSENTIAL BUS (SOME AIRPLANES) OR THE ATG BUS (OTHER AIRPLANES).

48205A

Figure 7-9

NOTE

The ADI bank steering bar is removed from the display unless the heading selector switch is placed to MANUAL. The bar will then give steering corrections to headings selected by the heading selector knob.

5. Heading marker — Set, if desired.
6. Course arrow — Set to desired TACAN radial.
7. Bearing selector switch — NORM or TACAN.
8. TSD scale selector switch — As desired.
9. TSD mode selector switch — MAN.
10. TSD command heading pointer — As desired.
11. TACAN channel selector — Set to desired channel.

The following will be displayed:

NOTE

No automatic command functions will be displayed on the AMI and AVVI.

ADI

Bank steering bar — Operative if heading selector switch is in MANUAL.

horizontal situation indicator presentations

	MAN NAV		AUTO NAV		DL MAX RNG OR DL MIN TIME		ILS OR ILS APCH	
	MAN	NORMAL	MAN	NORMAL	MAN	NORMAL	MAN	NORMAL
NORM	NAV MAN		NAV MAN	NAV	DL MAN	DL	ILS MAN	ILS
	TAC				TGT		NONE	
TAC	NAV MAN		NAV MAN	NAV	DL MAN	DL	ILS MAN	ILS
	TAC							
ADF	NAV MAN		NAV MAN	NAV	DL MAN	DL	ILS MAN	ILS
	UHF OR DL							
	MANUALLY SET		MAN SET	COM HEADING	MANUALLY SET	COMMAND HEADING	MANUALLY SET	SERVOS TO AIRPLANE HEADING
NORM	TO SELECTED TACAN STATION				BEARING TO TARGET		SERVOS TO AIRPLANE HEADING	
TAC	TO SELECTED TACAN STATION							
ADF	TO UHF OR DL ADF SOURCE							
	MANUALLY SET FOR COURSE TO SELECTED TACAN STATION				TARGET HEADING * *		MANUALLY SET TO INBOUND LOCALIZER HEADING	
	ANGULAR DEVIATION FROM SELECTED TACAN COURSE				LATERAL DISTANCE FROM INTERCEPTOR TO TARGET TRACK * *		DEVIATION FROM LOCALIZER	
	TO-FROM TACAN STATION				TARGET DIRECTION * *		OUT OF VIEW	
NORM	DISTANCE TO TACAN STATION				DISTANCE TO TARGET * *		MASKED	
TAC	DISTANCE TO TACAN STATION							
ADF	MASKED							

** ONLY DISPLAYED IF MCC IS VALID IN THE COMPUTER.

* IF THE FLIGHT MODE SELECTOR SWITCH IS PLACED TO AUTO, THE HEADING SELECTOR SWITCH WILL REVERT TO NORMAL.

48206

Figure 7-10

MA-1 climb and descent schedules—auto nav

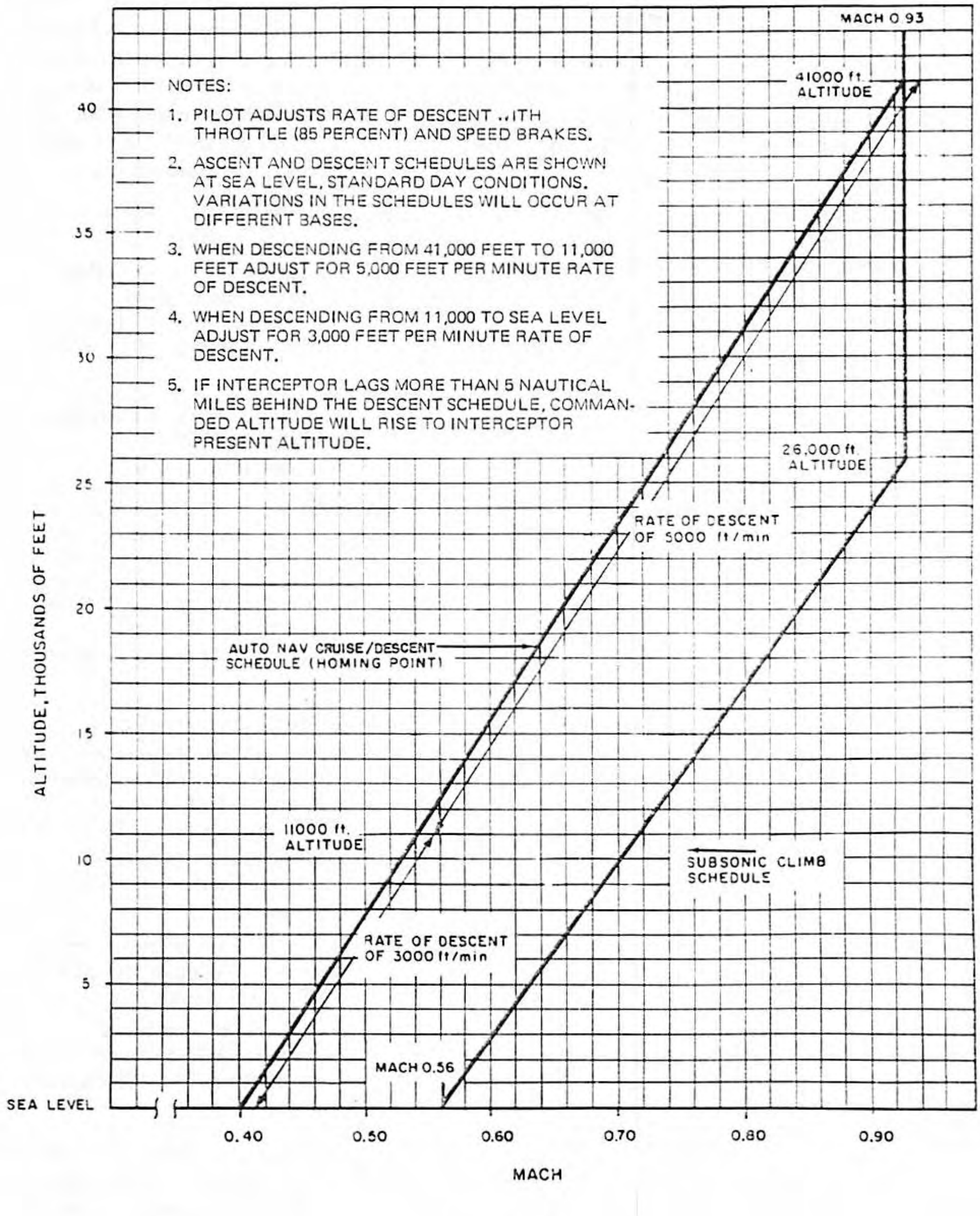


Figure 7-11

HSI

a. Bearing pointer—Points to TACAN station bearing.
b. "To-From" indicator—Indicates that selected TACAN course will take airplane to or from TACAN station.
c. Range indicator—Distance to TACAN station.
d. Illumination of left-hand "NAV-MAN" lights and right-hand "TAC" light.
e. Course deviation indicator—Angular deviation from selected TACAN course.

TSD

a. Interceptor symbol will display airplane position.
b. Heading cursor — Airplane heading.

Auto Nav

1. MA-1 power switch—RADAR STBY or ON.
2. Flight mode selector switch — As desired.
3. Display/automatic mode selector switch — AUTO NAV.
4. Heading selector switch — NORMAL.
5. Course selector knob — Desired course.
6. Bearing selector switch — NORM.
7. TSD scale selector switch — As desired.
8. TSD mode selector switch — AUTO.
9. TACAN channel—As desired.
10. Homing point selector switch—As desired.

The following will be displayed:

RADAR SCOPE

a. Time-to-go circle — Represents time-to-go to homing point.
b. Steering line – Indicates steering tc homing point.

AMI

a. Command Mach positioned automatically.
b. Command airspeed marker — As desired.

ADI

a. Bank steering bar — Out of view.

AVVI

a. Command altitude marker — Command altitude.
b. Target altitude — Homing point altitude.

HSI

a. Mode indicator light—"NAV."
b. Bearing pointer indicator light—"TAC."
c. Heading marker — Command heading to homing point.
d. Bearing pointer—Bearing to TACAN station.
e. Range indicator — Distance to TACAN station.
f. Course arrow and course deviation indicator — Angular deviation from selected TACAN course.
g. "To-From" indicator—Indicates that selected TACAN course will take airplane to or from TACAN station.

TSD

a. TACAN annunciator—"OK."
b. Interceptor symbol and heading—Geographical position & heading.
c. Target symbol — Located over homing point on proper heading.

d. Command-heading pointer — Command heading.

By following the above indications the airplane will be directed to the desired homing point.

DATA LINK MODE

Data link is the interceptor's primary source of command and navigation information from the SAGE ground environment. There are two modes of control in data link navigation: close control and modified close control.

Close Control. The primary mode of data link control is close control. In this mode the SAGE computer performs the calculations required. The data link commands are then automatically sent from the SAGE transmitter site to the aircraft. The equipment aboard the aircraft checks the information for validity and proper address, then sends it to the MA-1 computer, which in turn sends command to the AFCS and to cockpit displays. When the pilot sets in the grid reference correction prior to takeoff, he compensates for variation as well as the convergence angle at the takeoff base. When an interceptor using close control is paired with a target, modified close control (MCC) messages also are normally sent. The role of the MCC information in this case is to provide the pilot with extensive target information in the form of cockpit displays and to act as a backup in the event of a loss of data link reception.

Modified Close Control. When the dominant mode of control is MCC, SAGE sends target information. The MA-1 computer uses this information to calculate the tactics required and sends the necessary speed, altitude and heading commands to the cockpit displays and to the AFCS. Two types of MCC messages can be transmitted by SAGE: MCC-Region Origin (MCC TACAN) and MCC-interceptor (MCC-NO TACAN). In MCC-Region Origin, target X and Y position is referenced with respect to the SAGE division origin. In MCC-Interceptor, target x and y position is referenced with respect to the interceptor. The automatically selected MCC mode is MCC-Region Origin. The MCC-Interceptor mode must be requested by the pilot. When the mode of control is CC and the MCC backup messages are normally MCC-Interceptor.

When the interceptor receives a CC message for CAP/RTB, MCC information will not be displayed.

ILS MODES

NOTE

Refer to Section IX for ILS procedures.

The ILS mode can be flown in an automatic or manual flight mode.

NOTE

- If the automatic feature is engaged, bank angle is limited to 30° until glide-slope interception; after interception, bank angle is limited to 15°.
- If the glide-slope signals should fail, the AFCS will revert to ASSIST and any subsequent AUTO selection will revert to ASSIST.
- During the automatic approach it is not necessary to place the display/automatic mode selector switch to ILS APCH upon interception of the glide-slope. Switching is automatic within the system and the switch will remain in the ILS position. The glide-slope indicator will display glide-slope deviation throughout this mode. The bank steering bar is not utilized for an automatic approach.

When flying the ILS mode for a manual localizer intercept and approach, the ADI pitch steering bar is initially out of view. The display/automatic mode selector switch must be placed to ILS APCH to activate the pitch steering bar. The ILS APCH mode should not be selected until stabilized on localizer.

1. MA-1 power switch—WARM, RADAR STBY, ON, or RADIO SILENCE.
2. ILS channel selector switch—Set to desired channel.
3. ILS volume control knob—Identify station and adjust.
4. Localizer warning flag—Not visible.

NOTE

- Appearance of the localizer warning flag at the bottom center of the ADI indicates that localizer information is invalid on both the ADI and the HSI.
- Disappearance of the localizer warning flag in the ADI indicates only that the localizer receiver is receiving a valid signal. In order for valid ILS localizer information to be displayed in the cockpit, continue with the following procedures:

5. Display/automatic mode selector switch—ILS.

NOTE

If ILS APCH position is selected on the display/automatic mode switch when flying a manual ILS approach, bank steering signals will be limited to 15°. If selected prior to being established on the localizer it may result in inability to stabilize on the localizer.

6. Course selector knob—Set to published localizer course (also valid for back-course ILS approach).

   Set the published localizer course in the course selector window of the HSI. The course deviation indicator will then present correct magnitude and direction of deviation from the selected ILS course, and the bank steering bar on the ADI will present accurate steering information.

WARNING

ADI and HSI steering information will be erroneous unless proper ILS localizer course is set into the course selector window. The course selector window should be checked after selecting ILS to ensure the selected ILS final course remains in the window. It is possible for the reading to shift from the selected value.

7. Flight mode selector switch—As desired.
8. Heading selector switch—NORMAL.
9. Bearing selector switch—NORM or TAC.
10. TSD scale selector switch—As desired.
11. TSD mode selector switch—As desired.
12. AVVI barometric pressure knob—Set.
13. Command altitude slewing switch—Place marker as desired.
14. Command airspeed slewing switch—As desired.

The following will be displayed:

ADI

a. Glide-slope indicator—Deflected to top of scale.
b. Pitch steering bar—Out of view.
c. Bank steering bar—Indicates steering required to intercept and follow localizer.

WARNING

It is possible that a malfunction of the ADI might be determined only by checking it with the turn-and-slip indicator and the AVVI. Also, cross-check with artificial horizon on radar scope.

HSI

a. Mode indicator light—"ILS."
b. Command heading marker—Slaved to airplane heading.
c. Bearing pointer—Slaved to airplane heading or pointing to TACAN station.
d. Course deviation indicator—Angular deviation from selected heading, or from localizer course.

NOTE

The HSI "To-From" indicator will be out of view and the range indicator will be masked unless the bearing selector switch is in TAC.

15. Display/automatic mode selector switch—In manual flight, switch to ILS APCH when established on the localizer course (CDI centered).

The following will be displayed:

ADI

a. Glide-slope indicator — Centered.

b. Pitch steering bar — Centered.

c. Bank steering bar — Centered.

HSI (Unchanged).

WARNING

When attempting to intercept the outbound course of an ILS after a missed approach, if the airplane heading is not within 90° of the previous inbound course, the bank steering bar will indicate that the "on course" is on the opposite side of the airplane from the actual "on course" position. Therefore, monitor and use the HSI course deviation indicator as the HSI presents the true situation.

NOTE

If the approach is automatic, when the runway is in clear view or when published minimums are reached, take over manually by depressing the momentary interrupt trigger.

AUTOMATIC DIRECTION FINDING (ADF)

The ADF feature can be used to determine a bearing to a ground or airborne transmitter in the event that TACAN is jammed or if such operation is directed otherwise. UHF signals can be received from a ground or airborne source, through the communications receiver, when the bearing selector switch is placed to the UHF-ADF position. If the bearing selector switch is placed to DL-ADF, the signals are received through the data link receiver. When the ADF antenna is at the null position, the correct bearing of the transmitting station is displayed through the HSI bearing indicator (airplanes with integrated flight instrument system), and the TSD command heading pointer (airplanes with conventional instrument display).

NOTE

For the most accurate DF steering information, the airplane should be in a straight and level flight attitude.

UHF — ADF

1. Flight mode selector switch—As desired.
2. Display/automatic mode selector switch—As desired.
3. Heading selector switch—NORMAL.
4. Bearing selector switch—UHF-ADF.
5. UHF channel selector — Set to desired channel.

The following will be displayed:

HSI

a. Mode indicator light(s) — Correspond to display/automatic mode selector switch setting.

b. Bearing pointer indicator light—"UHF."

c. Bearing pointer — Points to transmitting station (ground or airborne).

TSD

a. Interceptor heading cursor—Indicates magnetic heading of interceptor.

DL — ADF

1. Flight mode selector switch—As desired.
2. Display/automatic mode selector switch—As desired.
3. Heading selector switch—NORMAL.
4. Bearing selector switch—DL-ADF.
5. Data link channel selector — Set to desired channel.

The following will be displayed:

HSI

a. Mode indicator light(s) — Correspond to display/automatic mode selector switch setting.

b. Right light—"DL."

c. Bearing pointer—Points to transmitting ground station.

TSD (Same as UHF-ADF).

Tactical Situation Display (TSD)

The TSD (figure 7-12) is a map display instrument which presents tactical and navigation data pictorially during the various mission phases. On airplanes with the conventional instrument display an automatic direction finder switch position has been incorporated. The face of the indicator displays a map centered on the selected TACAN station. A delta-wing symbol representing the interceptor, and an X-bisected-by-an-arrow symbol representing an enemy target or a homing point, move over the map. A grid of three parallel lines engraved on the screen indicates command heading and a cursor at the edge of the display indicates interceptor heading. Functions of controls and indicators on the TSD are discussed in the following paragraphs.

Map Selection. When the preset selector switch on the TACAN control panel is set to the desired TACAN station, a map of the area surrounding that station appears on the TSD. The selected map is the stationary reference, and it appears with magnetic north at the top. The aircraft and target symbols move in relation to the map. The TSD stores film maps for each of the 23 preset TACAN stations. The scale and the information on the film maps is similar to that on the standard foldout flight charts. For additional information on the maps presented in conjunction with preset TACAN selection, refer to RADIUS MILE SELECTOR SWITCH, this Section. When a TACAN Station is selected by the manual method, area maps of the stations selected are not available. Rather, a choice of three maps (depending on the position of the radius mile selector switch) is available as follows:

a. The 50-mile map contains a homing index and a calibrated test map.

b. The 200-mile map is a graphic index showing geographical locations of the available preset TACAN stations.

c. The 400-mile map is a calibrated test map which includes a tabular index listing the available preset TACAN stations and map charts available for each station.

Interceptor Symbol. The delta-wing interceptor symbol, projected on the screen, is about one inch long and has a dashed range line extending forward to indicate the no-wind flight path. After takeoff, when the interceptor exceeds 0.25 Mach and before initial reception of TACAN, the interceptor symbol is positioned by the digital computer, based on the computer's dead reckoning from initial position. When reliable information is being received from a TACAN station, the interceptor symbol is positioned directly from the TACAN equipment and serves as a direct readout of TACAN bearing and slant range. Whenever TACAN is lost and the computer is dead reckoning interceptor position, the symbol is positioned by direct inputs from the computer. Correction for altitude is made, assuming that the TACAN station is at sea level. The heading of the symbol always shows actual interceptor heading and always agrees with the heading cursor. Dead reckoning moves the interceptor symbol along its ground course on the TSD map. The dashed line along the longitudinal axis of the interceptor symbol may be used for approximate range determination. The first line segment from the interceptor is one-half inch long as is the first space between segments. All succeeding line segments and the succeeding spaces between them are one inch in length. On all of the map scales, one inch equals 1/8 of the total range of the chart. For example, on the 50-mile radius scale, one inch represents 1/8 of 100 miles or 12.5 miles. On the 200-mile radius scale, the first line segment (one-half inch long) from the interceptor represents 25 miles, since one inch represents 1/8 of 400 miles or 50 miles.

TARGET Symbol. The position and heading of the target are represented by a symbol in the form of an X bisected by an arrow. In the AUTO NAV mode, the target symbol indicates the position and assigned heading of the selected homing point. If the homing point is not located on the map selected, the display can be seen by turning to a map which does contain the homing point location. In data link modes, the symbol indicates target position and heading from MCC messages of target position and velocity. Target position is dead-reckoning when MCC messages cease. The target heading displayed is the estimated air-mass heading of the target, assuming that the wind value at the target is the same as that at the interceptor. In data link modes, the appearance of the target symbol is the best indication that valid MCC messages have been received. The symbol remains on the TSD until the pilot puts the MA-1 into the attack mode or otherwise clears the computer. If the desired heading to the homing point is greater than 90° from present interceptor heading, the heading of the target symbol will show a maximum of 90° off present interceptor heading. As the interceptor turns toward the homing point, the target symbol will turn with the interceptor, remaining 90° off interceptor heading, until the difference between interceptor heading and desired heading-to-homing-point becomes less than 90°. At this time the heading of the symbol will stabilize on the computed desired heading-to-homing-point for the remainder of the interceptor's turn.

Interceptor Heading Cursor. The interceptor heading cursor at the edge of the fixed compass rose indicates the magnetic heading of the interceptor, and may be used as a quick-reference check to determine angular difference between present interceptor heading and the command (data link) or desired (AUTO NAV) heading, as indicated by the command heading indicator (three parallel lines on TSD face).

Command Heading Indicator. The three parallel lines engraved on the TSD screen indicate command heading from data link inputs, or desired heading-to-homing-point during AUTO NAV mode. The action of these lines during AUTO NAV is the same as that of the target symbol heading discussed above when the desired heading-to-homing-point is initially greater than 90° off present interceptor heading. The mode selector switch on the right-hand TSD control panel controls the operation of the command heading indicator. (Refer to the mode selector switch discussion this Section.) DL command headings are referenced to both the stable platform and the J-4 compass before being displayed on the TSD. Any compass errors, therefore, will be reflected in the position of the command heading indicator on the TSD. (The steering line and command heading markers on the HSI will not reflect J-4 compass errors.)

Radius Miles Selector Switch. The radius miles selector switch (figure 7-12) is located on the tactical situation display. The switch is placarded "Radius Miles" and has 50-mile, 200-mile, and 400-mile positions. The switch is used to select the desired scale of the display map. A discussion of the three maps available in conjunction with manual TACAN selection is included under MAP SELECTION, this Section. When TACAN selection is by the preset method, three maps (50-, 200-, and 400-mile scales) are available for each station and contain the following information:

a. The 50-mile scale is a terminal map which includes such information as the length and direction of runways, dangerous and prohibited areas, emergency safe altitude, approach lights, approach markers, ILS and tower communication channels, published letdown plates or AFIO's, stable platform correction angles, and related data.

b. The 200-mile scale includes information concerning holding points, recovery and departure procedures, and related data.

c. The 400-mile scale is a general tactical and cross-country navigation map which shows locations of air bases, navigational aids, and other pertinent data.

Power is supplied from the MA-1 electrical power supply system.

NOTE

In the data link mode of operation the target and interceptor bugs must both be on the TSD for an accurate range display on the HSI.

TSD Manual Heading Knob. The TSD manual heading knob (figure 7-12) is located on the TSD. The knob is placarded "Man. Course" and operates only when the TSD mode selector switch is in the MAN position. Manually turning the knob rotates the grid lines on the display screen. The TSD manual heading knob receives power from the MA-1 electrical power supply.

TSD Mode Selector Switch (Airplanes With Conventional Instrument Display). The TSD mode selector switch (figure 7-12) is located in the upper right corner on the TSD. The switch is placarded "Mode" and has four positions: ADF CMD, ADF DL, AUTO and MAN. With the switch in the ADF CMD position, the grid lines are positioned to indicate command heading to the UHF radio transmitter to which the command radio is tuned. With the switch in the ADF DL position, the grid lines are positioned to indicate command heading to the GCI transmitter to which the data link equipment is tuned. When the switch is in AUTO position the command heading displayed is determined by selection of a TACAN station (or associated homing point), a data link transmitter, or ADF (data link on UHF) fix. When flying an ILS with the AUTO position selected, the grid lines and command heading pointer are slaved to airplane heading. When the switch is in MAN position, the grid is manually positioned by turning the TSD manual heading knob. The TSD mode selector switch receives power from the MA-1 electrical power supply system.

TSD Mode Selector Switch (Airplanes With Integrated Flight Instrument System). The TSD mode selector switch (figure 7-12) is located in the upper right corner on the TSD. The switch is placarded "Mode" and has positions AUTO and MAN. When the switch is in the MAN position, the grid and command heading pointer are manually positioned by turning the TSD manual heading knob. When the switch is in the AUTO position, the command heading pointer presents command heading derived from data link or the digital computer. The TSD mode selector switch receives power from the MA-1 electrical power supply system.

Light Intensity Rheostat and Off Switch. The light intensity rheostat (figure 7-12) is located on the TSD, has positions OFF and BRT, and controls the intensity of the display screen lighting accordingly. The switch normally controls power from the MA-1 electrical power supply system; however, when the MA-1 power switch is in EMER position, power supply to the switch is from the airplane ac essential bus.

CAUTION

Place OFF/BRT control to first detent for a minimum of 30 seconds to prevent TSD lamp burnout.

NOTE

On airplanes with the conventional instrument display, the loss of MA-1 electrical power to the light intensity rheostat causes automatic switching to the airplane nonessential bus when the MA-1 switch is in any position other than OFF.

Red Filter Switch. The red filter switch (figure 7-12) is located on the TSD. The switch is placarded "Red Filter" and has two positions, OUT and IN, which manually control the red filter.

Lamp Selection Switch. The lamp selection switch (figure 7-12) manually controls the selection of either of the two lamps available for projecting the map on the TSD screen. The switch is placarded "Lamp" and has positions 1 and 2 which select map projecting lamp one or two as indicated. The switch is located on the TSD.

tactical situation display
(typical)

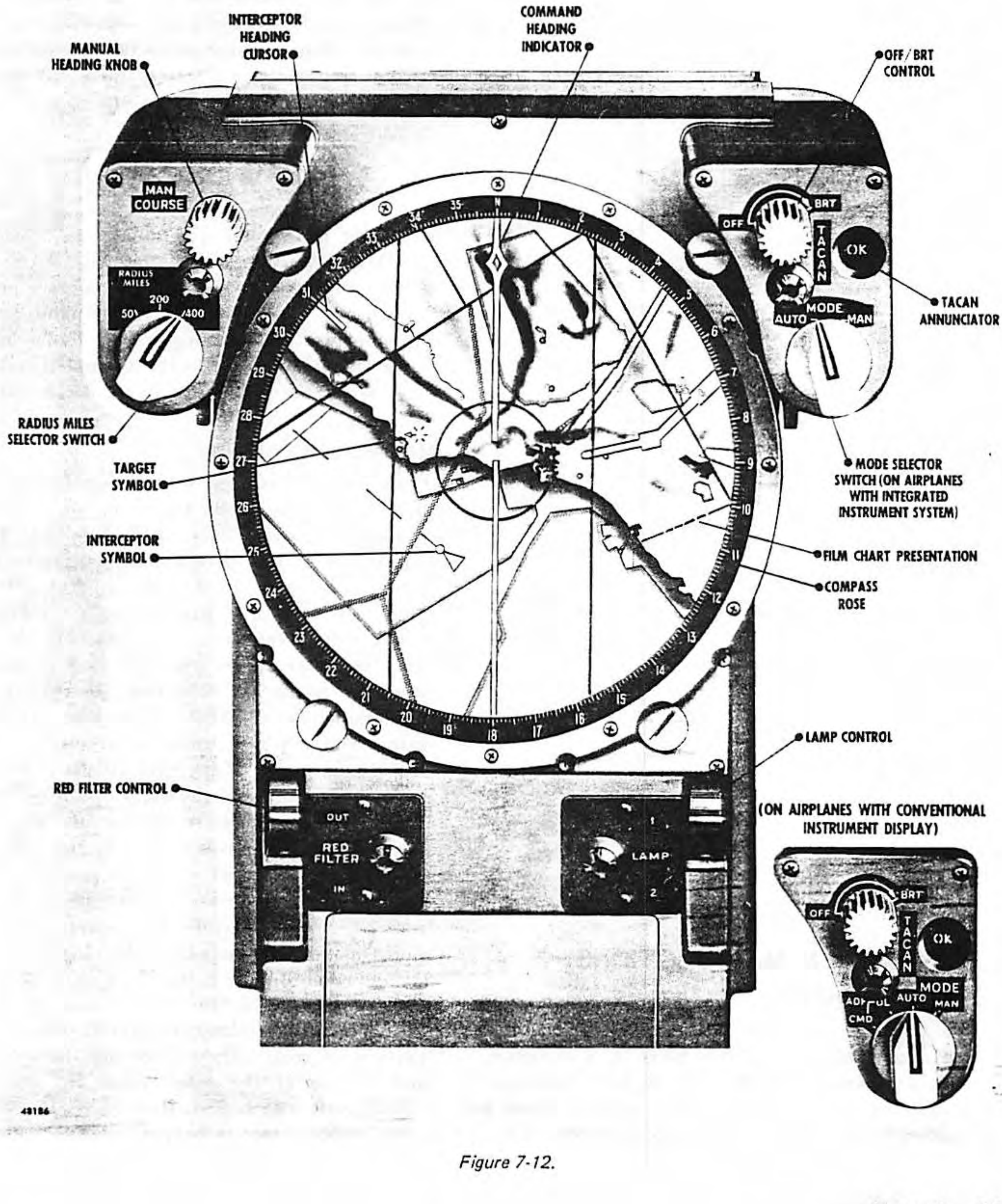


Figure 7-12.

MA-1 SYSTEM MALFUNCTIONS

INTRODUCTION

The pilot has a definite responsibility for helping to keep equipment malfunctions to a minimum and for completing a mission when malfunctions occur. All too often a system will not show a malfunction on the ground in the same way that it does in the air; sometimes the malfunction will appear only when the aircraft is airborne.

The pilot is the only one with the system while airborne; the pilot is the only one to see it malfunction; the pilot is the only one who can describe the malfunction and the circumstances under which it happened. The following is devoted to a discussion of some of the failures that may be encountered. In most cases, there are specific things to look for to tell the ground crew.

POWER

WARNING

Never change the position of the MA-1 POWER switch unless the ARM-SAFE switch is in SAFE and the armament selector is in VIS IDENT. Failure to observe this warning may result in irreversible damage to the armament and sets up the possibility of an accidental firing.

MA-1 POWER FAILURE

A transient overvoltage or undervoltage will cause the MA-1 system to revert from an "on" condition to either a "warm" or "off" condition. If the transient is of less than 2 seconds duration, the system will revert to a warm condition; if the transient is of greater than 2 seconds duration, the system will revert to an off condition. In either case, the scope range scale light will go out. To determine which has happened, observe the TSD for illumination. The TSD will be illuminated if the system is in a WARM condition. If system is in an OFF condition the TSD light will be out. If the system had reverted to the warm condition, set the MA-1 POWER switch to WARM and the range scale light will now come on. The MA-1 POWER switch should then be set to ON and the system will time in to an on condition (as indicated by the radar scope becoming operational) in approximately 90 seconds. If the range scale light did not come on when the MA-1 POWER switch was set to WARM, the system was in an off condition. This condition can result from either a transient of greater than 2 seconds duration or a fault in one of the sub-systems. Set the MA-1 POWER switch to EMER, then to RADAR STBY. If the system times in properly, (as indicated by the TSD becoming operational) set the MA-1 POWER switch to ON. Should the system time in properly in RADAR STBY and then remain in an on condition when the MA-1 POWER switch is set to ON, continue with the mission. If the system fails to remain in a "standby" condition, set the MA-1 POWER switch to EMER, then to WARM. If the system will not maintain an on condition, but will stay in either a standby or warm condition, the MA-1 system has full navigation and communication capabilities, provided that the fault causing power failure does not lie in these subsystems.

NOTE

> If at any time the MA-1 power dumps to OFF, stable table erection may be necessary.

If it is impossible to maintain the system in either a warm, standby, on, or "radio silence" condition, set the MA-1 POWER switch to EMER. Setting the MA-1 POWER switch to EMER provides additional cooling for the communications equipment.

When debriefed, the pilot should tell the ground crew:

a. The circumstances under which power failed-cycling armament, pulling positive or negative g's, switching functions, transmitting on UHF, etc.

b. Whether power was recycled, the results of recycling, and also the time required for system to dump (5 or 50 seconds).

c. Whether there was any scope flickering or fluctuations on any indicators prior to the power loss.

MA-1 COMMUNICATON LOSS

The communications equipment should automatically transfer to another power supply bus in the event of an MA-1 power failure. If a communications failure occurs, a fault exists in either the power transfer circuitry or in the communications equipment itself. If MA-1 power is regained by recycling and the communications equipment is then operative, a fault in the power transfer circuitry is indicated. Should the communications equipment still be inoperative after MA-1 power is regained, the failure may be due to arcing in the UHF transmitter. If this is the case, communications may be regained by descending to an altitude of 20,000 feet or less. If communications can not be regained or if it is suspected that a fault in the communications equipment is causing MA-1 power loss, set the OFF-MAIN-BOTH switch to OFF and recycle MA-1 power.

NOTE

> Under these conditions, the data link receiver may be available as a backup receiver if MA-1 power switch is in WARM and power is available.

The pilot should inform the ground crew:

a. The circumstances under which communications failed.

b. The effects of descending to a lower altitude and recycling.

c. The effect of setting the OFF-MAIN-BOTH switch to OFF and recycling MA-1 power.

FAULT ISOLATION

In general, if power fluctuations exist which are degrading the system to the point where it is useless, try changing the electrical load by the following methods:

a. Set the FLT MODE switch to PITCH DAMP.

b. Set the IFF CONT switch to STBY.

c. Set the MA-1 POWER switch to RADAR STBY.

d. Set the MA-1 POWER switch to RADIO SILENCE.

If none of the above is effective, set the POWER switch to EMER and then back to ON. The pilot should inform the ground crew:

a. The circumstances under which the fluctuations occurred.

b. The effects of the measures used to isolate the fault.

c. Whether the fluctuations were the same at different altitudes.

AUTOMATIC POWER SUPPLY TEST

The computer provides automatic monitoring of power system voltage and noise levels. A power test failure is indicated by lighting of the PTF lamp on the radar scope which will appear on the photographic recorder film. The lamp should stay on for 15 seconds or the duration of the failure whichever is greater.

STABLE PLATFORM DRIFT

Roll and/or pitch displacement of the radar scope artificial horizon may occur during taxi turns, following an abrupt maneuver, or subsequent to periods of aircraft acceleration, particularly where turns are involved. The horizon should return to the radar scope centerline shortly after the maneuver is completed. Should this return to centerline be excessively long or if constant roll or pitch drift is observed, radar search capability and the AFCS may be adversely affected. The radar antenna and IR seeker head horizontal reference is generated from the stable platform. When the stable platform drifts, the search pattern will drift in a similar manner. If the drift is excessive, the search pattern will not cover the entire region in which targets are expected. Since the AFCS pitch, roll, and heading reference are predicated on proper stable platform stabilization, all AFCS modes (navigation, VIS IDENT, and ILS) will be affected. When such drift becomes excessive, stable platform re-erection in the air may be necessary

CAUTION

When the radar artificial horizon is continuously drifting and will not stabilize with re-erection of the stable platform, don't use the AFCS.

RADAR

During the search portion of a mission, there are various display malfunctions that may occur. These malfunctions may vary from distortion of a single display to complete loss of all displays. The following paragraphs detail these malfunctions and give any corrective action that may be possible.

TRANSMITTER MALFUNCTIONS

The quality of the system radar scope display is dependent on proper operation of several subsystems. MA-1 power regulates receiver-transmitter operation, and the cooling and pressurizing function must be operating properly in order to have an optimum display. Radar scope presentations less than optimum frequency occur particularly at high altitudes and after prolonged operation at low altitude under conditions of high ambient temperature. If such a condition occurs as a result of the MA-1 transmitter switching to reduced power, the RP light on the radar scope will light. Under these conditions the following steps should be taken:

a. Recycle the MA-1 POWER switch through RADAR STBY.

b. If the condition persists, observe whether normal video is restored during descent and report the occurence to MA-1 maintenance personnel.

It can be determined if the transmitting circuits are malfunctioning by switching to RADAR STBY. To assist maintenance, note if RP light is on prior to standby. In this position the synchronizer is connected directly to the range trace generator. If the synchronizer, range trace generator, and the indicator are working, the B-sweep should appear; then it is known that the trouble is in the transmitter. If the transmitter has failed because of a temporary overload, normal operation can be restored by setting the MA-1 POWER switch to RADAR STBY for several seconds then back to ON.

RECEIVER MALFUNCTION

The pilot may notice that the video is intermittent or that there is no video at all. That is, the B-sweep appears to be satisfactory and is searching, but targets are intermittent or no target appears at all.

In the case where targets are intermittent, the AFC is not tuning consistently to the transmitter frequency. As a result, targets appear during the short period when the receiver and the transmitter are at the proper frequency. It may help to select a different tuning rate. If this procecure does not remedy the malfunction, it is impossible to complete a radar attack. The pilot must then rely on IR or visual contact to complete an attack. If the pilot notices that the video is intermittent as it crosses the scope, the trouble is in the transmitter or AFC functions. Turning the power switch to RADAR STBY and then back to ON may remedy the situation in some cases. If operating in the paramp mode, select another mode.

RADAR DISPLAY

Part of Display Distorted or Missing

The seriousness of the malfunction when part of the display is distorted or missing depends on which part of the display is affected. The artificial horizon may be distorted, half missing, or completely missing, without greatly affecting system operation. It is more serious if the time-to-go-to-offset circle or the steering dot are distorted or missing.

If the artificial horizon should disappear or become distorted to where it is unusable, the pilot can refer to the ADI and obtain approximately the same information. If the antenna elevation marker should become unusable, the pilot can approximate the antenna's position by noting the ANT ELEV controls' position. Distortion or loss of the reference circle should not prevent the pilot from completing his mission. The B-sweep, RGM, and target displays may be distorted without seriously affecting system operation. Distortion of the RGM or target displays may require some additional manipulating in order to achieve lockon.

Whenever the steering dot and/or time-to-go-to-offset circle becomes unusable due to distortion or loss, the pilot should ensure that the AFCS is engaged (FLT MODE switch to AUTO). If only the time-to-go-to-offset circle is unusable, the pilot can still determine when he has reached offset by observing as to when the target elevation marker becomes operational. The pilot may also be able to obtain voice control from the ground to replace the information lost as a result of one or both of the above displays failing.

Erase Malfunctions

The types of malfunctions of the erase function which should be noted and relayed to maintenance personnel during post flight are as follows:

a. The erase sweep lead on the B-sweep is too large, too small, or non-symmetrical. It is critical that the erase sweep not be too close to the B-sweep for it could degrade detection.

b. The erase sweep is too short, doesn't cover the search raster or become a dot.

c. The raster covered in raster erase does not include the entire tube.

All of the above are malfunctions or misadjustments and should be easy to duplicate on the ground.

Search Display Shrinkage

This search malfunction is characterized by the B-sweep covering less than its normal area on the scope as shown in (figure 7-12). The antenna is still covering its normal search area, but the scope display is compressed. This condition may be alleviated by selecting a different position of the AZ SCAN switch or by engaging the manual search or super-search submodes. If the situation cannot be corrected but lockon can be acquired, a normal attack can be accomplished.

Figure 7-12

Stationary B-Sweep

Loss of pressurization in the radar waveguide assembly will cause the radar transmitter to be disabled. If this should occur, the B-sweep will be stationary at the center of the scope, no target will be displayed but the B-sweep can be moved in manual search. The only way to alleviate the problem is to descend to a lower altitude.

Antenna Servo Disabled

A failure of the antenna servo is characterized by the appearance of bands across the entire display. These are caused by the antenna's failure to sweep across its search area while the B-sweep continues to move. Thus, a target return from the stationary antenna is spread across the face of the scope by the moving B-sweep. If this has happened, the radar cannot be used. Before aborting a radar attack, however, make sure that the bands are not caused by clutter. Elevate the antenna to see if an aerial target can be obtained, or depress the antenna to get a different ground clutter return. If the bands remain, the antenna drive is disabled and a successful radar attack cannot be accomplished.

Artificial Horizon Drift

This can be a serious malfunction during the search phase of a mission because it is usually an indication of stable platform drift. The radar search scan uses the same horizontal references as the artificial horizon. When the stable platform drifts, the search pattern drifts in a similar manner. If the drift is excessive, the search pattern will not cover the entire region in which the targets are expected. If the stable platform has drifted, follow the procedure for stable platform alignment in flight. Improper grid reference setting may cause slight precessions.

WARNING

When the radar artificial horizon is obviously drifting, not just aligning itself after an abrupt maneuver, don't use AFCS. The AFCS uses the same vertical references as the artificial horizon that appears on the scope.

Disappearance of B-Sweep off One Side of Scope

In the event the B-sweep should disappear off one side of the scope, obtaining lockon becomes impossible. Since the cause of this malfunction is usually due to shorting as a result of excessive moisture in one of the coaxial cables feeding the deflection circuitry; there is nothing the pilot can do to regain the B-sweep. He may try IR, however since the deflection circuits for RADAR and IR are common, the chances are that IR will also be missing its sweep. If this is true, the pilot will have to rely on visual contact to complete an attack. A selection of ILS may return the B-sweep to its normal position on the range.

RADAR LOCKON

There are various malfunctions that may occur to that circuitry directly concerned with lockon. The following paragraphs give the indications that result from these malfunctions and any corrective action that may be possible.

NOTE

If the system will not automatically range track a target, it is possible to productively fire the radar missiles in a pursuit mode by manually positioning and holding the radar range gate on the target until launch.

Low Lockon Sensitivity

When this malfunction occurs, difficulty will be experienced in locking on to targets when they are clearly visible. Also, lockon may break easily. However, a successful radar attack may still be made. Try locking on by placing the range gate just under the target, releasing the action switch, and letting the target fly into the range gate. As long as the radar stays locked-on, the system has full capability. After returning to base, report the difficulty in locking on to clearly visible targets.

Apparent Loss of Lockon

On rare occasions, the attack display will disappear temporarily but the system will remain locked-on. When the attack display disappears, check to see if the B-sweep and range gate stay on the target. If they do the system is still locked-on. The attack display will probably reappear as the interceptor closes on the target. If the target echo and range gate or B-sweep definitely separate, lockon has been lost and must be regained. If the attack display disappeared temporarily but the system remained locked-on, the technician should be informed of this and also whether the target was weak.

Loss of Target in Hand Control

It is possible that the target may disappear when the action switch is pressed to the second detent. When this happens, the source of trouble will usually be in the PFN (Pulse Forming Network). It may help to set the RANGE switch to 40SP, set the POWER switch to RDR STBY and then to ON.

If this procedure does not correct the malfunction, it is impossible to complete a radar attack. The pilot must then rely on IR or MTWS to complete an attack.

Lockon Transfer to Altitude Line

Lockon transfer to the altitude line is possible while tracking targets near a strong altitude line. This condition is more pronounced in low overtake situations where range rate is not adequate to allow the range gate to pass through the altitude line prior to transfer of lockon.

Part of Attack Display Distorted or Missing

Distortion of the artificial horizon, reference circle, Z-marker, or range rate gap can take place without seriously affecting system operation. It is more serious if the time-to-go-to-fire circle and/or the steering dot and reference circle are distorted or missing. If these symbols are lost, the system performance is degraded but firing is still possible. Radar attacks can be flown in AUTO with full attack capability with complete loss of attack displays. If necessary, the target range can be obtained from the B-sweep. If AUTO is not available, the attack can be completed by centering the B-sweep in azimuth and the elevation marker in elevation.

WARNING

The pullout signal may not appear when the armament is fired, and collision warning may not be available.

Erratic Motion of Steering Dot

This attack malfunction is characterized by erratic motion of the steering dot. In a sensitive display, he dot is moving somewhat erratically but by steering to center the average position of the dot, a successful attack can be made. With a very "hot" dot, the dot is moving so erratically that the excursions are far out of the steering circle, and a successful radar attack cannot be accomplished. CADJ may be effective in damping a hot dot.

IR

IR SEARCH DISPLAY

The IR search display is subject to a variety of malfunctions that result in loss or distortion of the display. The following paragraphs enumerate these malfunctions and give any corrective action that may be possible.

Search Pattern Drift

The IR seeker head uses the stable platform as a horizontal reference during search. If the stable platform drifts, the search pattern will also drift. Visually checking the rear of the seeker head may reveal search pattern malfunctions. A platform drift does not affect the C-scan display since it is scope oriented. This drift could result in a search pattern which does not cover the area intended.

Part of Search Display Distorted

Because of the mechanical construction of the IR seeker head, it is necessary that the azimuth and elevation drive motors operate differentially during search. A fault in the interconnecting circuitry could cause the search pattern to be rotated to some degree about the longitudinal axis of the aircraft and the scope search dispaly to be similarly rotated. If not present to an excessive degree, this fault would not seriously degrade the system's ability to detect a target.

No Apparent IR Targets

The IR search trace as displayed on the scope will normally exhibit small deflections due to clutter and system noise. A very smooth trace is an indication of a malfunction in the IR subsystem or an improperly adjusted IR THRESHOLD VIDEO control. If the search trace shown no deflections, set the IR THRESHOLD VIDEO control to its extreme clockwise position. The appearance of IR targets on the trace is indicative of a properly operating system. The control should then be adjusted so that the trace deflections are barely visible. If the adjustment of the control was ineffective, engage the manual search mode by pressing the action switch to the second detent and, using the left hand control and the ANT ELEV control, seek a target such as a cloud, the sun, ground reflections, or if possible another interceptor. If a target indication can be seen on the scope, the system may function properly. The pilot should then follow the procedure for realigning the platform. Should the above methods fail to produce a deflection of the scope trace, there may be a fault in the cooling system for the IR detector cell.

Complete loss of the search pattern or the expanded sweep may be due to a fault in the display circuitry. If this is the case, it is still possible to accomplish lockon by use of the aural tone. To determine if the IR receiver is functioning properly, press the action switch to the second detent to engage the manual search mode. Try to find a target such as a cloud ground reflections or if possible another interceptor. If the aural tone is heard while manually sweeping across the target, the IR receiver is operational. A sharp increase in frequency of the aural tone will indicate the detection of a target. Depress the action switch to the first detent to engage the supersearch submode immediately upon hearing the target. In this mode the seeker head is scanning in aximuth a total of 33-degrees centered on a line determined by the lateral position of the left hand control. To achieve lockon, it will be necessary to hunt for the target in azimuth only. As soon as the aural tone is heard, carefully adjust the ANT ELEV control for a tone of the highest pitch possible. When this has been accomplished, depress the action switch to the second detent, adjust the left hand control laterally for maximum deflection, and release the action switch. A relatively constant aural tone and the appearance of the attack displays will indicate lockon.

IR LOCKON

There are various malfunctions that may occur to that circuitry directly concerned with lockon. The following paragraphs detail these malfunctions and give any corrective action that may be possible.

Lockon Difficulties

The pilot may experience difficulty in locking on to an IR target due to the IR reference generators being out of calibration or to an erroneous drive signal in the seeker head stabilization circuitry.

If IR lockon cannot be accomplished after spotlighting a target and releasing the action switch, there is usually nothing the pilot can do to remedy the difficulty. However, the pilot should make several observations in order to assist the maintenance technicians in rectifying the malfunction. If lighting conditions permit, observe the movement of the seeker head when the action switch is released. If the seeker head jumps in a particular direction, inform the ground crew of that fact. If adequate illumination of the seeker head is not available, the pilot may be able to tell the nature of the difficulty from scope indications.

When the action switch is released, RDR SLVD selected, observe the expanded C-scan. If the C-scan jumps in elevation, it is an indication that the seeker head is doing the same thing. If the system locks-on momentarily, but an erroneous azimuth drive signal exists, the point source display peaks will move closer together and then disappear. If an erroneous elevation drive signal exists, one peak grows shorter and disappears.

Transfer of Lockon

If the target being tracked should pass between a source of infrared radiation and the interceptor, the seeker head may then transfer its lockon to the unwanted target. The unwanted target could be the sun, a sunlit cloud, or in the case of low altitude missions, reflections from water or from terrain. During track, the closing rate will normally cause the pitch of the aural tone to steadily increase. If lockon should transfer to an unwanted target, there will be a change in the aural tone and possibly an excursion of the steering dot. Such a transfer might be determined by an examination of the expanded sweep. Press the RDR IR/EXP switch and compare the trace with that obtained at original lockon. If it is determined that transfer of lockon has occurred, return the system to search and lockon to the proper target. It may sometimes be necessary to change altitude so that the unwanted target will no longer be in a direct line with the interceptor and the desired target. Transfer of lockon can also result in loss of lockon. During low altitude missions over water, the seeker head may lockon to a reflection from a wave or white cap. The wave then disappears or is quickly passed by and lockon is lost. A similar situation can exist over terrain, particularly over desert terrain. The seeker head transfers-lockon to a new heat source. As soon as the interceptor has passed the false target, lockon is lost. In both instances, it will be necessary to return to search and once again lockon to the desired target.

Transfer of lockon is not a malfunction of the IR subsystem, but is described here since it may result in loss of lockon which could be misinterpreted as a fault in the system.

No Radar Target in RDR SLVD Submode

This malfunction can result from antenna-seeker head misalignment. Switch to RDR SCAN and observe whether or not a radar target appears. If a target does not appear, the fault may be due to a vertical antenna-seeker head misalignment. To determine this, the system would have to be returned to search and the antenna moved in elevation by rotating the ANT ELEV control.

There are also certain transmitter problems that could cause the above malfunction. The problem may be an overload within the transmitter. It may be possible to cure this problem by setting the POWER switch to RDR STBY and back to ON. This however, should be done only after observing the warning given in the POWER recycle paragraph. This means setting the armament selector switch to VIS IDENT which would break IR lockon. If none of the above actions remedy the situation, there is little that the pilot can do except make any observations that may assist the maintenance personnel in locating the fault.

IR Attack Displays Distorted or Missing

It may still be possible to determine the validity of an IR target even though the ability to recall the expanded C-display is lost. If radar is still functioning and manages to burn-through any jamming that is present, the pilot can, by observing the radar display, compare its range and elevation with where he believes the IR target to be. If the comparison is favorable he can assume that the IR target is valid. If the range and FIR markers should become inoperative or missing, the pilot will have to try and obtain his ranging information from the radar subsystem. If this is possible, he can launch his armament when the target passes the firing bar.

If radar is completely jammed or inoperative the optical sight or angle ranging can be used to determine when he is to launch his armament. If the steering dot becomes inoperative or missing, the mission can still be completed by setting the system to RDR SLVD and using the B-sweep to determine azimuth position and the expanded C-scan to determine elevation relative to the target. When any or all of the above malfunctions occur, there is little corrective action that the pilot can take except to make those observations that he feels will assist the maintenance personnel in locating the fault.

SECTION VIII—CREW DUTIES

Not Applicable to this Airplane

all weather operation

Section IX

TABLE OF CONTENTS

NOTE

Except for some repetition necessary for emphasis, clarity, or continuity of thought, this section contains only those procedures that differ or are in addition to the normal operating instructions covered in Section II and Section IV.

INSTRUMENT FLIGHT PROCEDURES

For instrument approach purposes, this aircraft is classified as Category E.

These procedures and techniques pertain primarily to instrument flight conditions and are in addition to normal procedures. Fuel requirements for completion of instrument penetration, approach, and possible diversion to an alternate are much greater than for VFR flight and must be included in preflight planning.

INSTRUMENT TAKEOFF

Complete the normal TAXI and BEFORE TAKEOFF checks as prescribed in Section II, and check pitot heat ON. After aligning the airplane visually with runway centerline, check heading indicator and magnetic compass against runway heading. Set attitude indicator at 5° nose-down with thrust at IDLE. This setting will provide a more accurate indication as it will tend to offset the pitch error resulting from takeoff acceleration.

CAUTION

To insure that nose wheel steering is engaged for takeoff, the steering should be used to line up and should not be disengaged until the rudder becomes effective on the takeoff roll.

Instrument takeoffs may be made at either military or maximum thrust.

NOTE

Afterburner is recommended to shorten the takeoff roll in conditions of low visibility.

Maintain directional control with nose wheel steering, then disengage steering when rudder becomes effective.

NOTE

Nose wheel steering may be used throughout the takeoff roll; however, directional control becomes increasingly sensitive above 75 KCAS.

The heading indicator is primarily for directional control but reference should be made to runway centerline and runway lights if possible. At 140 KCAS apply back pressure and rotate the airplane to a 10° nose-up attitude on the attitude indicator. Airplane will fly off the ground with the pitch change. Raise landing gear when a positive climb indication is noted on both the altimeter and vertical velocity indicator. Maintain a positive climb indication until the climb schedule is reached.

INSTRUMENT CLIMB

An instrument climb is accomplished in the same manner as a VFR climb by maintaining a positive climb indication until intercepting the climb schedule.

NOTE

During an instrument climb in afterburner, do not exceed a 30° nose-up indication on the attitude indicator to preclude a low airspeed, high angle of attack condition.

HOLDING

Holding procedures are based on airspace requirements, ease of handling, and fuel consumption and are found in FLIP II. Approximate thrust requirements to maintain 250 KCAS for holding altitudes are presented below. An increase of 1 to 2% rpm is necessary to maintain 250 KCAS during turns at 30,000 feet and 40,000 feet. A 30° angle of bank is used.

ALTITUDE FEET	KCAS	RPM %	FUEL FLOW LB/HR
20,000	250	85	2800
30,000	250	87	2800
40,000	250	89	2800

DESCENT

The optimum configuration for descents in instrument flight conditions is speed brakes extended, 275 KCAS and 85% rpm.

NOTE

Idle power descents may be accomplished; however, compressor stalls may occur and compressor bleed air may be insufficient for anti-icing.

Radar Recovery

Radar recovery can be accomplished with a minimum amount of time and fuel required. When a PAR or ILS approach is required, the descent from the inbound cruising altitude should be started at a sufficient distance to permit a straight-in approach at the recommended airspeed. A distance of three to four miles should be allowed for deceleration and changing to approach configuration before reaching the final approach fix.

Jet Penetration

Typical penetrations are shown in figures 9-1 and 9-2. At the initial approach fix, reduce thrust to 85% rpm, lower the nose to approximately 10° nose-down indication and establish 275 KCAS. Extend speed brake when reaching 275 KCAS. Start penetration turn as published and maintain 30° bank angle. Begin level-off 1000 feet above minimum altitude. Maintain 250 KCAS (82 - 84% rpm) inbound.

INSTRUMENT APPROACHES

Do not decrease airspeed below 200 KCAS until turn onto final approach is completed. Use speed brakes to change altitude and for descent to minimum altitude. Recommended final approach airspeeds are for 2000 pounds fuel remaining.

TACAN

Establish final approach airspeed with gear extended prior to final approach fix. Thrust required for level flight is approximately 88% rpm. Upon reaching the final approach fix, extend speed brakes and descend to minimum altitude.

PAR

Maintain 250 KCAS on downwind base and dogleg. Thrust required is approximately 83%. After turn onto final approach, extend landing gear and establish final approach airspeed. As glide-path is intercepted, extend speed brakes to establish desired rate of descent. See figure 9-3 for airspeeds, configurations and thrust settings.

Tacan—Circling Approach

With the recommended base leg airspeed (landing gear down and speed brakes closed) proceed from the TACAN final approach fix and/or designated point to begin a letdown to the circling approach altitude. Maintain recommended base leg airspeed (landing gear down and speed brakes closed) during maneuvering in the circling approach until descent from the circling approach altitude for landing or missed approach is initiated. When this descent is initiated, open the speed brakes and reduce airspeed to final approach airspeed. See figure 9-4 for one type of circling approach.

NOTE

AFM 51-37 depicts various methods to maneuver for circling approaches.

ILS

Refer to typical ASR-PAR pattern, figure 9-3, for airspeed, configuration and thrust setting. Fly outbound for a sufficient length of time so that final approach configuration can be established prior to intercepting glide-slope.

Missed Approach

Advance throttle to military thrust position, retract speed brakes, and establish a 10° nose-up indication on the attitude indicator. When a definite climb is indicated on the altimeter and vertical velocity indicator, retract landing gear. Execute missed approach procedure. If another approach is to be executed, reduce power to maintain 250 KCAS as missed approach altitude is reached.

ICE AND RAIN

WARNING

Avoid flight in areas of heavy icing. If unable to avoid these areas, climb or descend through them as rapidly as practicable. When flight is conducted in heavy icing conditions, expect erratic airspeed indications and ice build-up in engine inlets and on the canopy. Place the surface and engine anti-ice switch in the MAN ON position.

When the anti-icing systems are in operation, the airplane may be flown safely under icing conditions. No wing or vertical fin surface anti-icing systems are required as there is sufficient thrust available to overcome the increased drag from surface ice buildups. Surface icing will reduce range as increased thrust is required to maintain desired flying speed. The stability and control of the airplane will not be noticeably affected with surface ice buildups. The most probable free air icing temperatures vary from −4°C (+25°F) at sea level to −24°C (−12°F) at 20,000 feet. Above 20,000 feet, due to the inability of the air to contain moisture, the amount of icing is negligible. The mission of the airplane is so designed that most phases of a typical mission will be performed at altitudes above icing levels. The phase most susceptible to ice and most critical to operation is the instrument approach. If icing conditions are known to exist at instrument approach altitudes, the most expeditious means of recovery (normally the GCI penetration to final approach with a straight-in PAR or ILS) should be used to minimize the surface buildup. If icing is encountered unexpectedly and is allowed to build up, more thrust will be required to maintain desired speeds and rates of descent during the instrument approach. Flight under icing conditions with the engine and intake duct anti-icing systems inoperative could result in two forms of engine damage. Ice buildup on the engine inlet guide vanes may result in a restricted flow of inlet air, causing loss of thrust and possible compressor stalls. The possibility of this occurring is reduced by the absence of inlet screens and the relatively clean unrestricted intake. Light to medium ice buildup in the engine inlet will not cause an appreciable rise in exhaust gas temperature as it does on some other engine models, so thrust loss and compressor stalls will probably be the first engine icing indications. In the event of compressor stalls, depress the ignition button to alleviate the possibility of a flameout. Place the surface and engine anti-icing switch in the MAN ON position. In the event of inlet ice formation with the anti-icing system inoperative, the airplane should be flown out of the icing area as soon as possible, preferably with a reduced thrust setting. The second form of engine damage could result from intake duct ice breaking loose and being drawn into the compressor section of the engine, resulting in compressor section failure. Because of the possible damage that may result due to engine ice, the anti-icing system should be operable and the surface and engine anti-icing switch in the automatic position at all times when flight under icing conditions is anticipated.

NOTE

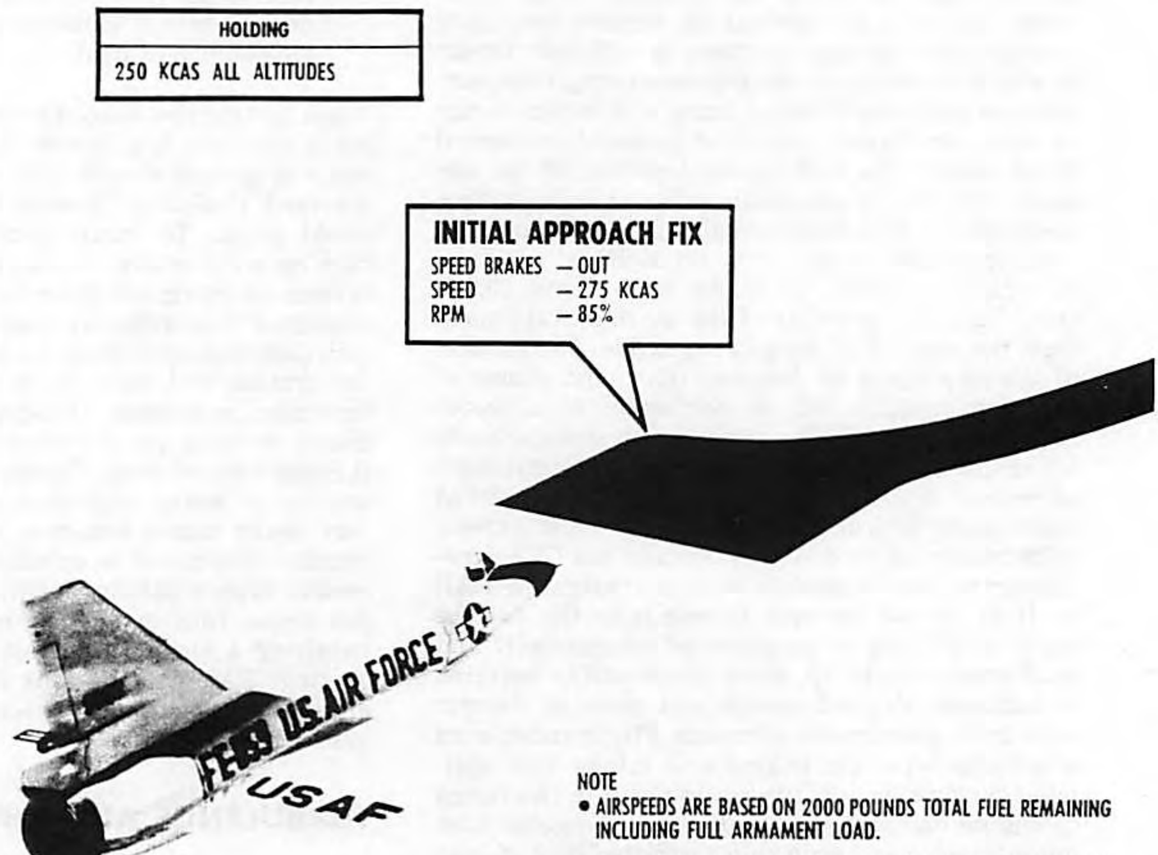
Engine operation below approximately 85% rpm may not supply sufficient heat to keep the engine compressor inlet guide vanes clear of ice under severe icing conditions. When descending in icing conditions, thrust should be increased to provide sufficient heat.

When instrument takeoffs or approaches are to be made and rain is anticipated, the windshield rain removal system should be in operation to increase forward visibility through the left-hand windshield panel. To insure proper operation of the rain removal system, it is recommended that the system be energized prior to flight whenever it is suspected that moisture may have collected in the rain clearing ducts since the last flight. Energizing the system will blow from the ducts any water that may be present. If reported rain intensity is heavy or less, good vision should be obtained through cleared area. Cleared area will be slightly smaller in heavy rain than in moderate rain. If very heavy rain is reported, some visibility will be retained but it will be substantially impaired. Successive flights through rain at supersonic speeds can cause rain erosion to occur to the radome, requiring a visual inspection of the radome after landing. Refer to Sections II and IV for procedures and operation of anti-icing and rain removal systems.

TURBULENCE AND THUNDERSTORMS

1. Accomplish subsonic thunderstorm penetration at an airspeed of 275-325 KCAS, not to exceed optimum cruise Mach number, unless profile of an intercept mission demands otherwise.
2. If the profile of an intercept mission so demands, penetrate thunderstorms at any supersonic Mach number, up to and including limit Mach number. Large g-loads experienced will not exceed design limit load factor at any speed.
3. Engine performance and airplane control are satisfactory during both subsonic and supersonic thunderstorm penetrations.

typical penetration (straight in)

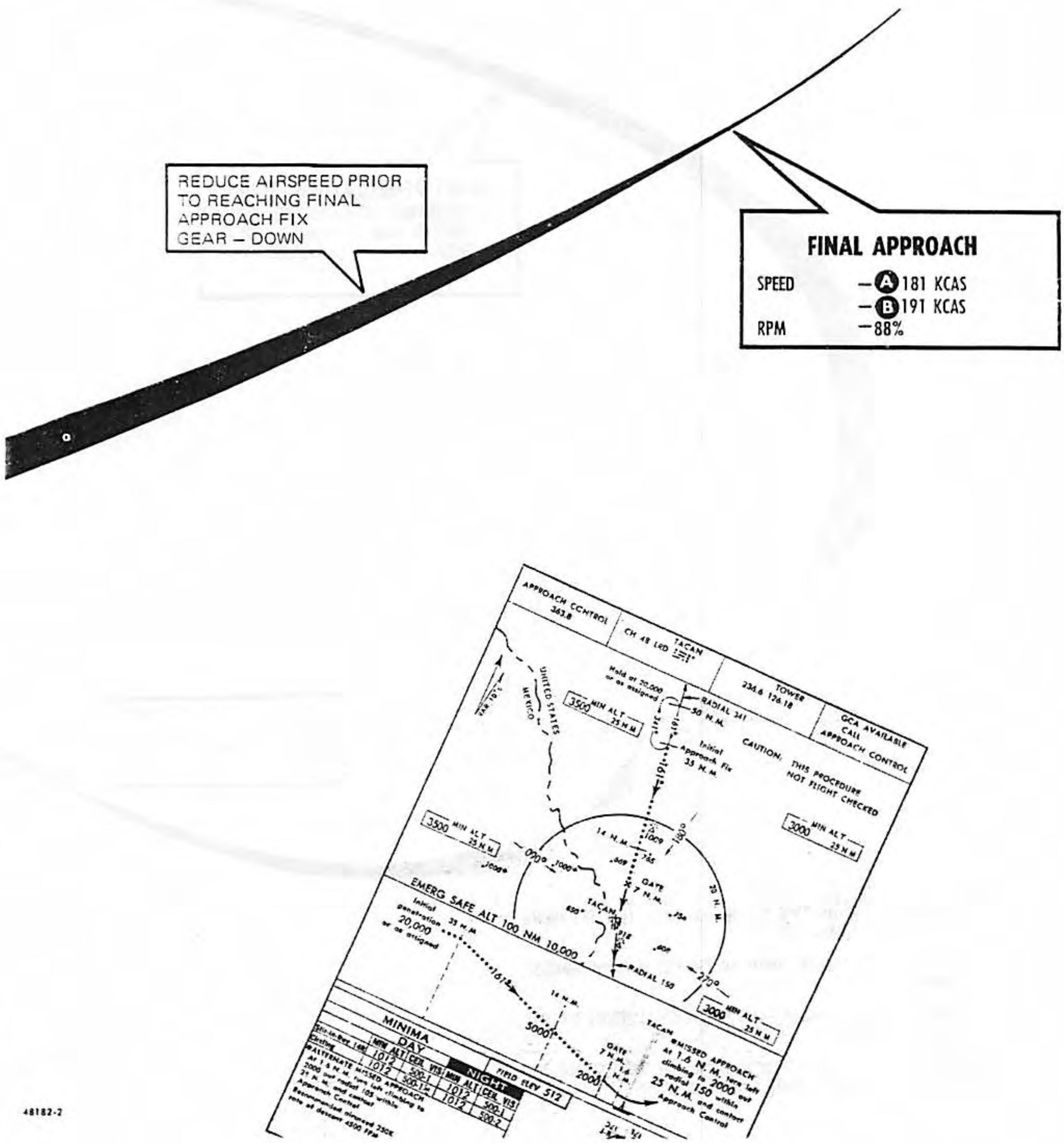


NOTE

- AIRSPEEDS ARE BASED ON 2000 POUNDS TOTAL FUEL REMAINING INCLUDING FULL ARMAMENT LOAD.
- FUEL REQUIRED FOR DESCENT FROM 40,000 FEET IS APPROXIMATELY 210 POUNDS.
- REFER TO "TERMINAL FLIGHT INFORMATION (HIGH ALTITUDE) - JET" FOR PUBLISHED PENETRATION.
- IF WEATHER IS TOO LOW FOR TACAN APPROACH, REQUEST PAR, ILS, OR PROCEED TO ALTERNATE WHILE STILL AT ALTITUDE.

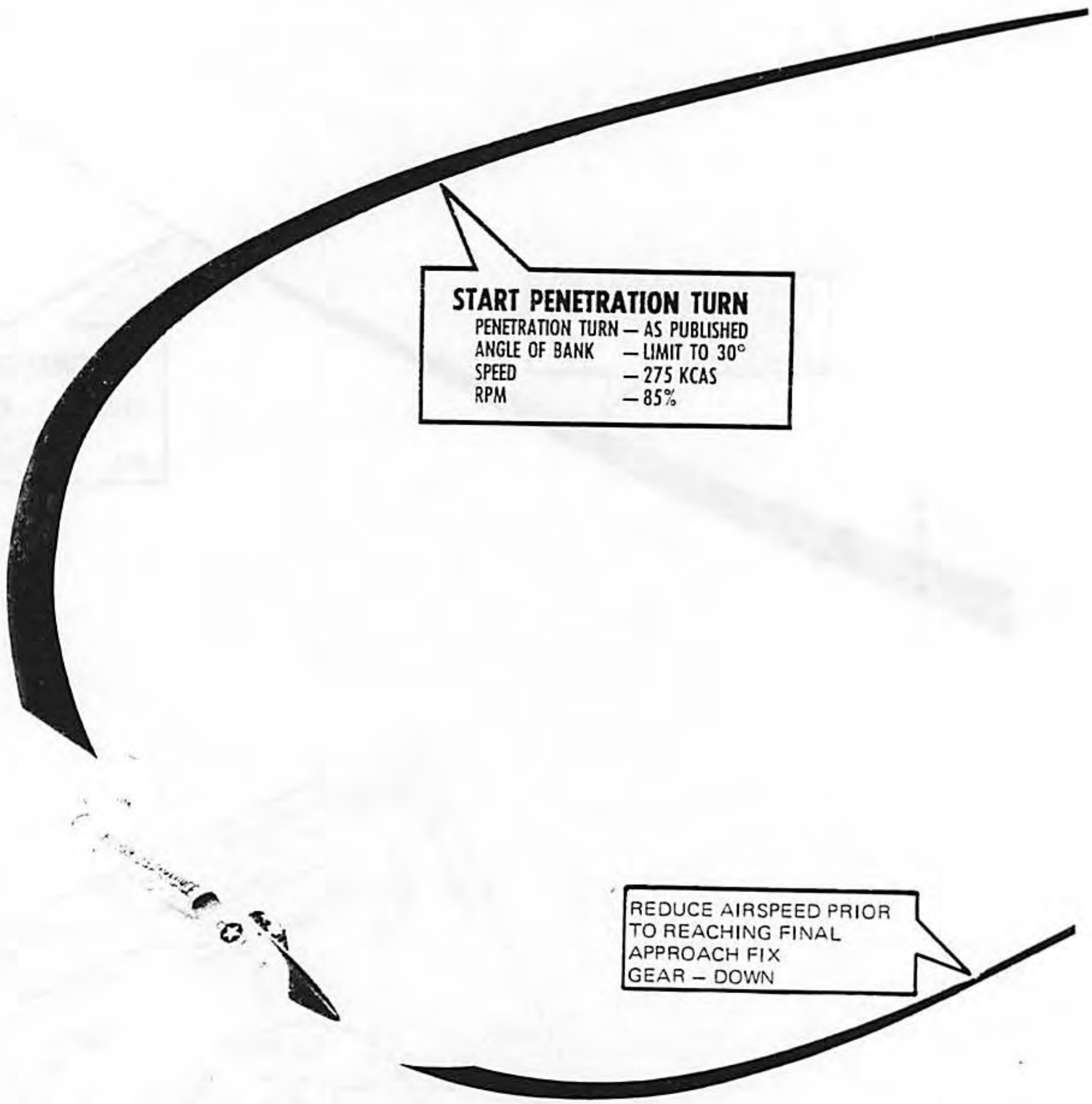
48182 · A

Figure 9-1


REDUCE AIRSPEED PRIOR
TO REACHING FINAL
APPROACH FIX
GEAR – DOWN
FINAL APPROACH
SPEED –A 181 KCAS
–B 191 KCAS
RPM –88%
APPROACH CONTROL
TACAN
TOWER
GCA AVAILABLE
CALL
APPROACH CONTROL
CAUTION: THIS PROCEDURE NOT FLIGHT CHECKED
UNITED STATES
MEXICO
EMERG SAFE ALT 100 NM 10,000
MINIMA
DAY
NIGHT
FIELD ELEV 512

48182-2

typical penetration (tear drop)

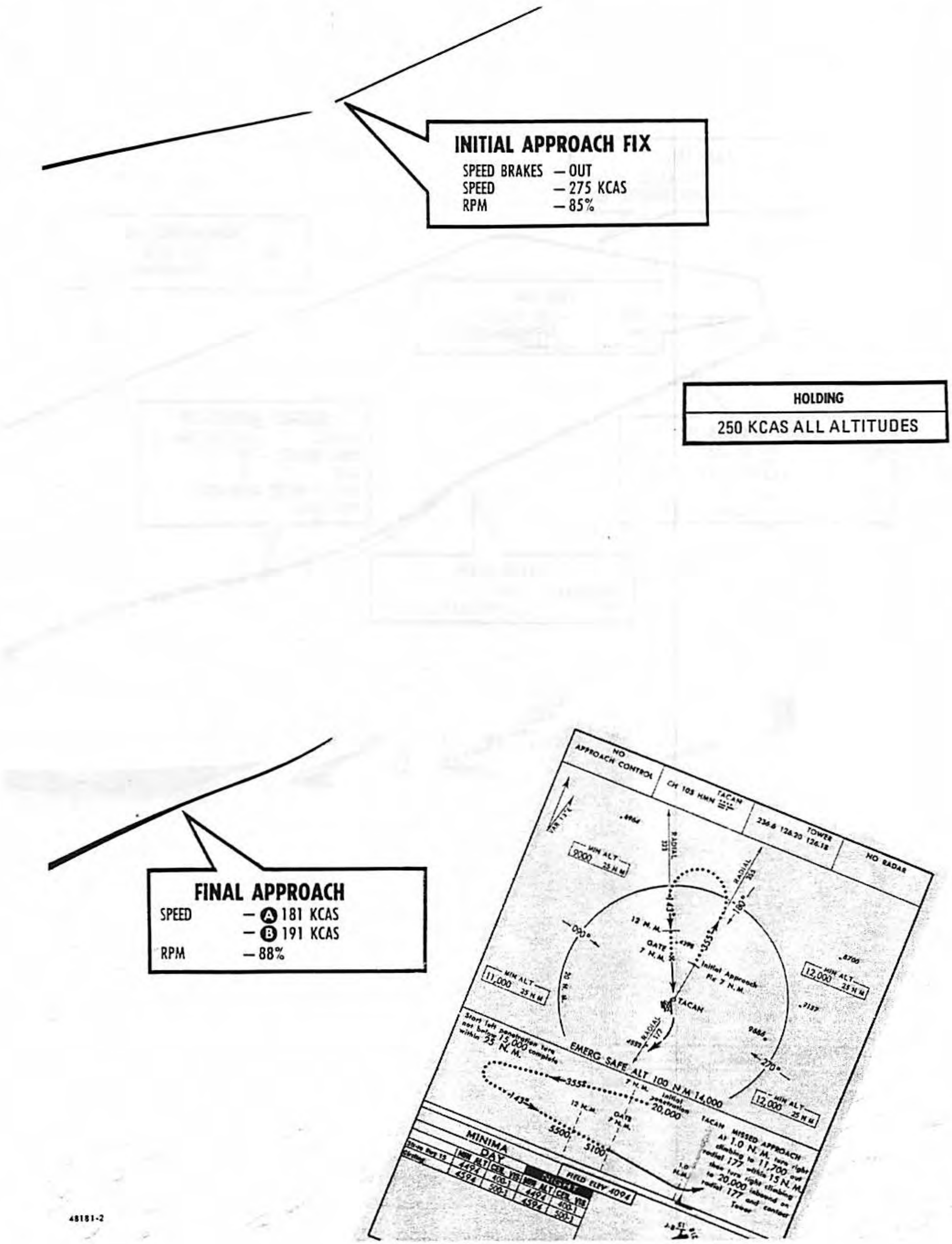


NOTE

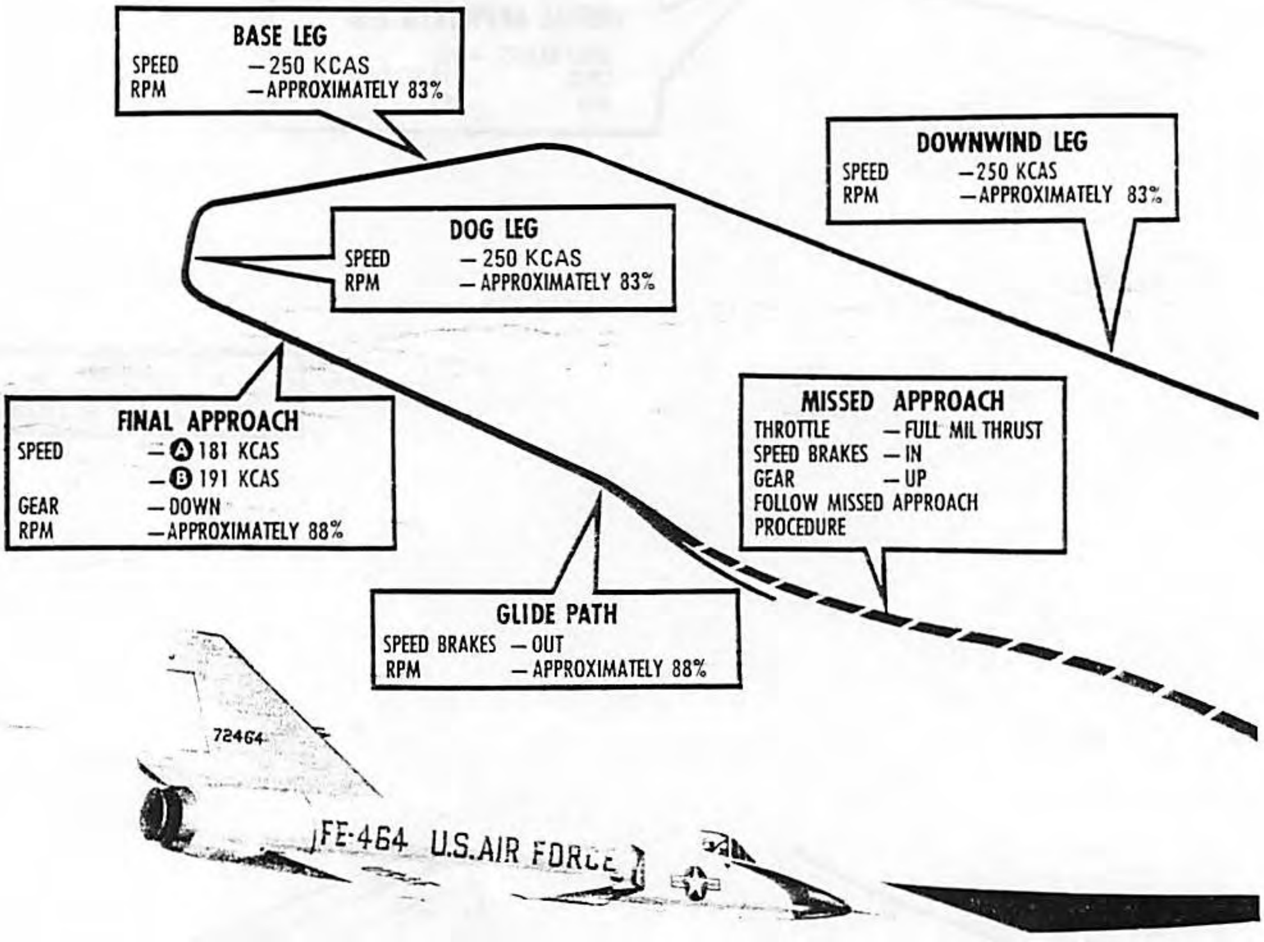
- AIRSPEEDS ARE BASED ON 2000 POUNDS OF TOTAL FUEL REMAINING INCLUDING FULL ARMAMENT LOAD.
- FUEL REQUIRED FOR DESCENT FROM 40,000 FEET IS APPROXIMATELY 210 POUNDS.
- REFER TO "TERMINAL FLIGHT INFORMATION (HIGH ALTITUDE) JET" FOR PUBLISHED PENETRATION.
- IF WEATHER IS TOO LOW FOR TACAN APPROACH, REQUEST PAR, ILS, OR PROCEED TO ALTERNATE WHILE STILL AT ALTITUDE.

48181-1A

Figure 9-2


INITIAL APPROACH FIX
SPEED BRAKES — OUT
SPEED — 275 KCAS
RPM — 85%
HOLDING
250 KCAS ALL ALTITUDES
FINAL APPROACH
SPEED — Ⓐ 181 KCAS
— Ⓑ 191 KCAS
RPM — 88%
NO APPROACH CONTROL
TACAN
CH 105 HMN
TOWER
236.6 126.20 126.18
NO RADAR
MIN ALT 9000 25 N M
MIN ALT 11,000 25 N M
MIN ALT 12,000 25 N M
MIN ALT 12,000 25 N M
12 N. M.
GATE 7 N. M.
Initial Approach Fix 7 N. M.
TACAN
EMERG SAFE ALT 100 N M 14,000
Start left penetration turn not below 15,000 complete within 25 N. M.
20,000
5500
5100
MINIMA
DAY
FIELD ELEV 4094
TACAN MISSED APPROACH At 1.0 N. M. turn right climbing to 11,700 out radial 177 within 15 N. M. then turn right climbing to 20,000 inbound on radial 177 and contact Tower

48181-2

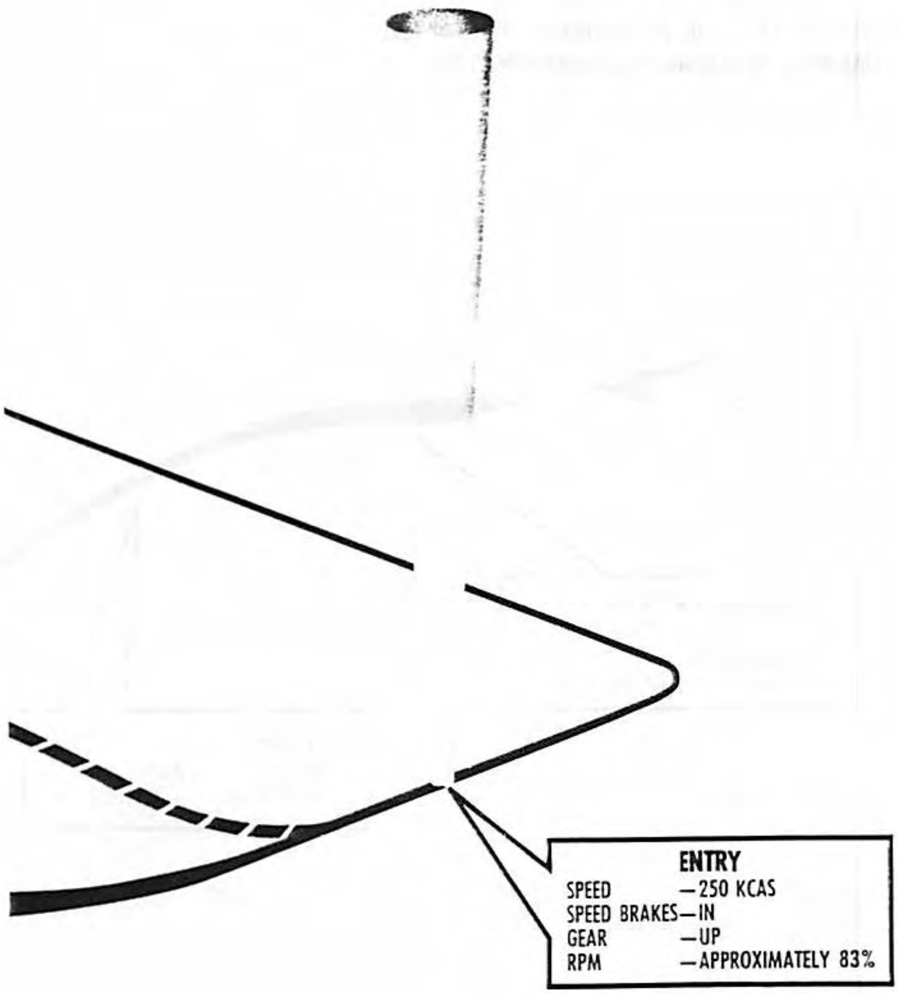


TYPE APPROACH	FUEL	TIME
NORMAL PATTERN	500 LB	7 MIN.
MISSED APPROACH	700 LB	9 MIN.

48180-1

Figure 9-3

asr and par pattern
(typical)



NOTE
- IF FUEL IS CRITICALLY LOW, REQUEST AN EMERGENCY PAR PATTERN.
- AIRSPEEDS ARE BASED ON 2000 POUNDS TOTAL FUEL REMAINING INCLUDING FULL ARMAMENT LOAD.

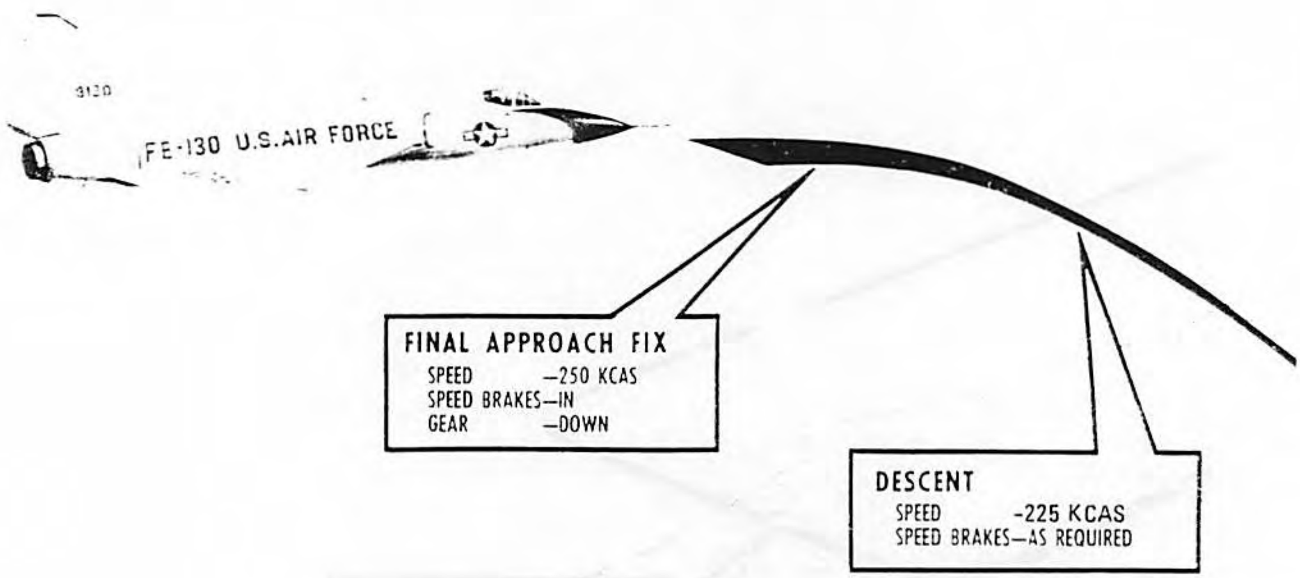
CAUTION

FLARE AND LANDING MUST BE ACCOMPLISHED VISUALLY AS THE LANDING GEAR WILL NOT WITHSTAND THE IMPACT AT 174 KCAS ALONG THE 3° GLIDE SLOPE.

48180-2 A

circling approach

AIRSPEEDS BASED ON 2000 POUNDS OF TOTAL FUEL
REMAINING INCLUDING FULL ARMAMENT LOAD.



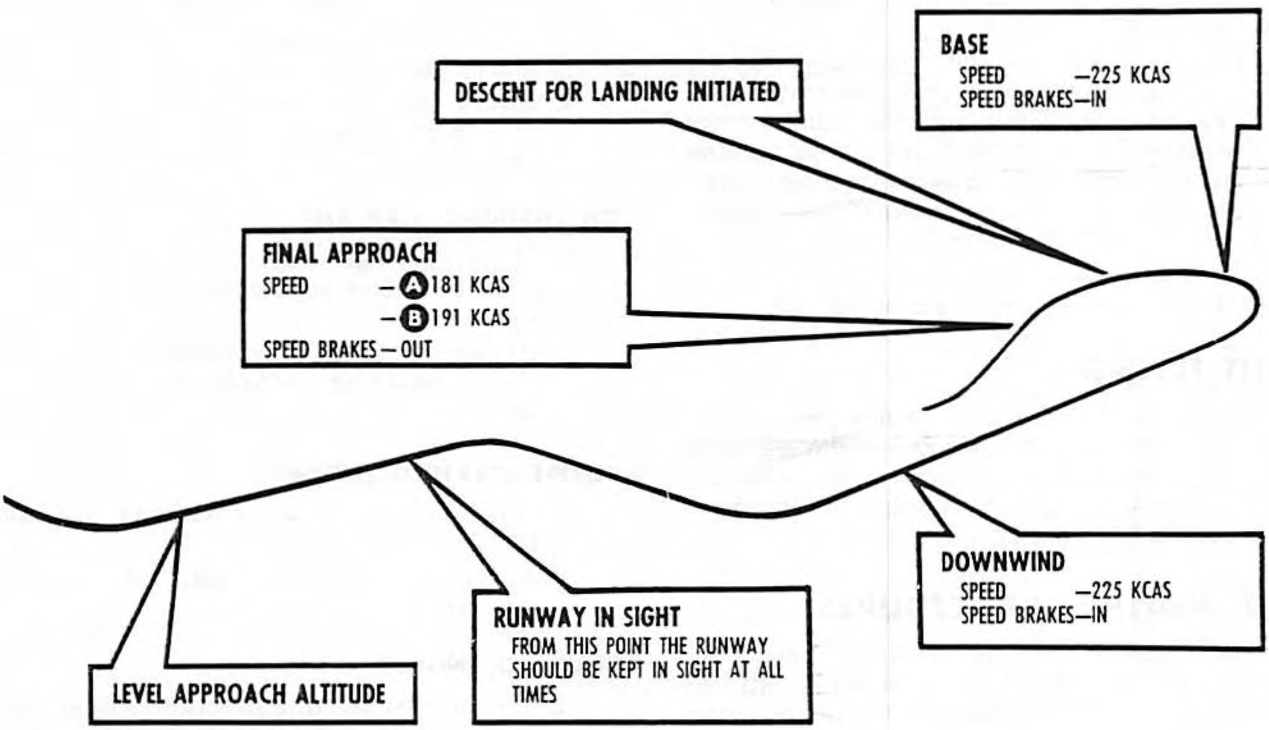
NOTE

THIS IS ONLY ONE OF THE VARIOUS METHODS OF PERFORMING A CIRCLING APPROACH. THE FINAL SELECTION OF THE PATTERN REQUIRED TO MANEUVER THE AIRPLANE TO A SAFE LANDING RESTS WITH THE PILOT.

CAUTION

A CIRCLING APPROACH IS FLOWN AT A LOWER ALTITUDE THAN VFR OVERHEAD PATTERNS. THERE IS A TENDENCY TO FLY DOWNWIND TOO CLOSE TO THE RUNWAY DUE TO THE PILOT'S SHALLOWER LOOK ANGLE TO THE RUNWAY. THIS MUST BE AVOIDED TO PREVENT A STEEP ANGLE OF BANK WHEN TURNING FROM DOWNWIND TO FINAL.

Figure 9-4


DESCENT FOR LANDING INITIATED
BASE
SPEED —225 KCAS
SPEED BRAKES—IN
FINAL APPROACH
SPEED — A 181 KCAS
— B 191 KCAS
SPEED BRAKES — OUT
DOWNWIND
SPEED —225 KCAS
SPEED BRAKES—IN
RUNWAY IN SIGHT
FROM THIS POINT THE RUNWAY SHOULD BE KEPT IN SIGHT AT ALL TIMES
LEVEL APPROACH ALTITUDE

48603-2

4. Be prepared for large roll and pitch rates during a turbulence penetration at any flight speed. The largest disturbances will be in roll.
5. Airplane reaction to turbulence varies between subsonic and supersonic penetrations. The majority of the pilot's efforts are required in combating roll disturbances during subsonic penetrations and pitch disturbances during supersonic penetrations.
6. Airspeed indications may be lost because of pitot icing, and the windshield may ice over during thunderstorm penetrations. If descent immediately follows thunderstorm penetration, windshield and canopy fogging can be expected. If extreme fogging occurs, the pilot will have to wipe the canopy/windshield to maintain visibility.
7. Hail can cause extensive damage to the radome and leading edges of airfoils.

NIGHT FLYING

Night flights in this airplane do not present any special problems or techniques except during landing. When lowering the nose, it will be necessary to switch from landing to taxi lights to illuminate the area ahead of the airplane.

COLD WEATHER PROCEDURES

To insure satisfactory cold weather operation, utilize the normal operating procedures outlined in Section II in conjunction with the following additions. It should be expected that due to adverse conditions imposed by the presence of snow and ice and the requirement to wear heavy Arctic clothing, the time required for preflight and postflight inspections will be increased over normal operating conditions.

BEFORE ENTERING THE AIRPLANE

1. Perform exterior inspections as outlined in Section II.
2. Check the entire airplane for freedom from snow and ice. Loose snow should be removed from the windshield and canopy with a broom, but do not attempt to chip or scrape ice since it can be successfully removed by the windshield and canopy anti-icing and defog system once power is applied to the airplane.

WARNING

Due to increased drag with accumulated snow and ice, takeoff distances and climbout performance can be seriously affected. The roughness and distribution of the ice and snow could vary stall speeds and characteristics to an extremely dangerous degree. Loss of an engine shortly after takeoff is a serious enough problem without the added, and avoidable, hazard of snow and ice on the wings. In view of the unpredictable and unsafe effects of such a practice, the ice and snow must be removed before flight is attempted.

3. Check angle of attack vane for freedom of movement.
4. Insure that the drag chute compartment is free from ice and/or moisture which might restrict deployment of chute.

ON ENTERING AIRPLANE

1. Make a normal cockpit check for switch and circuit breaker positions as outlined in Section II.
2. As soon as power is available, turn on windshield anti-icing, anti-fog and canopy anti-fog switches.

BEFORE STARTING ENGINE

1. Make normal checks as outlined in Section II.
2. Actuate brake pedals until normal brake pressure is available.

STARTING ENGINE

1. Use normal starting procedures as outlined in Section II. When a battery start is anticipated below freezing temperatures, ascertain that the battery has been preheated.
2. If airplane has been cold soaked to a temperature of −35°C (−31°F), operate the engine at idle rpm for at least two minutes to preclude damage to oil and hydraulic systems.
3. If temperature is between +32°F and −32°F, operate engine at idle rpm for at least one minute.
4. If the oil pressure-low warning light remains illuminated after start, the engine should be operated at idle until the light extinguishes. Advancing the throttle above idle shortly after the light extinguishes may cause the light to again illuminate. Operation of the engine at idle for approximately two minutes after the light extinguishes will allow advancing the throttle to maximum thrust without light illumination. Electrical power supply system check should be delayed until after the light extinguishes to minimize the load on the accessory drive shaft.

WARMUP AND GROUND CHECKS

Special attention should be paid to operational checks on all ice protection and defogging equipment. Refer to Section IV for anti-icing systems operation.

1. Keep wheel chocks in place until ready to taxi.
2. Slowly cycle elevon controls several times and actuate speed brakes. Have the crew chief check for leaks.
3. Check all instruments, radar, communication and navigation equipment, etc., for proper operation.
4. Set cabin temperature control knob for a comfortable level in AUTOMATIC.

NOTE

- At low temperatures, cockpit heat will be inadequate at idle rpm even with cockpit temperature control knob set to MANUAL HOT.
- If cold wind blast in the face is annoying, place cockpit air selector handle to direct airflow out of aft vertical outlets.

TAXIING INSTRUCTIONS

WARNING

Make sure all instruments have warmed up sufficiently to insure proper operation. Check for sluggish instruments during taxiing.

1. At all temperatures, excess thrust is available at idle rpm, and at lower temperatures this condition is further intensified, resulting in dangerously high taxi speeds unless controlled by brakes or the idle thrust control system.

NOTE

If bottoming of a landing gear strut occurs during taxiing, stop airplane and notify ground crew.

2. Taxi at slow speed when taxiing over rough, snow-packed surfaces.
3. Increase spacing when taxiing behind other aircraft or in vicinity of parked aircraft, and allow more room than on a cleared surface to bring the airplane to a stop.
4. After cold soaking, an increase of rpm up to 80% may be necessary to start taxiing. This is due to flat spots on the tires, stiff grease, etc. Once rolling, idle rpm is more than sufficient.
5. Successful taxiing can be accomplished in snow 6 inches deep with only a slight increase in thrust.
6. For all normal taxiing, nose wheel steering will be adequate. If minimum radius turns are required, disengage nose wheel steering and use rapid brake applications on the inboard wheel, being careful not to slide the tire. Approximately 75% rpm will be required. This will result in a pivot turn.

CAUTION

If nose wheel steering is inoperative and the airplane is stopped on ice or slippery packed snow, it will be difficult to make a minimum radius turn without first obtaining some forward motion.

BEFORE TAKEOFF

If other airplanes are in takeoff position, taking the runway behind them should be avoided if possible. Flying debris in the form of ice, snow, or ice fog from other jet engines can considerably reduce visibility prior to takeoff or during takeoff roll.

1. Make normal before takeoff check as outlined in Section II.
2. If one wheel starts to slip on engine runup, release brakes and make normal takeoff or retard throttle and stop the airplane.

WARNING

Under conditions of high relative humidity, excess moisture through the air-conditioning system could cause fog condensation so dense that the instrument panel is not visible. In event this occurs, place the cabin air switch to RAM.

TAKEOFF

1. During low temperature operation, engine performance is considerably improved over normal temperature operation. Because of this improved performance, takeoff roll will be reduced and initial climb attitude will be steeper than normal. Oil pressure may reach 80 psi for a short duration on takeoff and transient yaw may be experienced at speeds low as 50 KCAS. Afterburner takeoffs may produce an uncomfortably steep initial climb angle which is disconcerting when operating

in adverse weather. Unless there is an operational requirement for an afterburner climb, a military climb is recommended when operating in cold weather.

2. Nose wheel steering is somewhat sensitive and overcontrolling is possible at higher speeds. Disengage nose wheel steering at 75 KCAS if not needed to counteract transient yaw.
3. Due to the increased acceleration, double check for complete gear-up indication before exceeding gear speed limit.

LANDING

1. Use normal landing technique as outlined in Section II.
2. When operating at temperatures below −10°F be prepared for landing in ice fog. This phenomenon is characterized by good visibility straight down; however, the visibility will be zero at any angle off the vertical. Depth of the ice fog will usually be less than 100 feet. When this condition exists and no alternate is available, ascertain that the approach and runway lights are at full intensity. Use normal PAR procedures and once in the fog reduce rate of descent to 300 to 400 fpm and hold that attitude until touchdown. Approach and runway lights should be visible on both sides.
3. Plan each landing with the possibility of a drag chute failure in mind.
4. If strong cross winds are encountered on landing, use normal cross-wind landing technique and be prepared to jettison the drag chute if airplane direction cannot be controlled by nose wheel steering and brakes.

CAUTION

When landing on runways that have patches of dry surface, increased vigilance will be required to avoid tire skidding or locking the wheel, which could result in tire blowout.

WARNING

When the pilot wears heavy Arctic clothing, the stick travel is restricted.

DESERT AND HOT WEATHER PROCEDURES

In general, hot weather and desert procedures differ from normal procedures mainly in that added precautions must be taken to protect the airplane from damage due to high temperature and dust. Particular care should be taken to prevent the entrance of sand into the various airplane parts and systems (engine, fuel system, pitot static system, etc.). All filters should be checked more frequently than under normal conditions. Units incorporating plastic or rubber parts should be protected as much as possible from wind-blown sand and excessive temperatures. Tires should be checked frequently for signs of blistering, etc.

BEFORE ENTERING THE AIRPLANE

Check exposed portions of the shock strut pistons for dust and sand, and have them cleared if necessary. Check intake ducts for accumulations of dust or sand. Make sure crew chief has had all filters cleaned and that the airplane has been thoroughly inspected for fuel or hydraulic leaks caused by the swelling of packing or expanding of fittings. Inspect area behind the airplane to make sure sand and dust will not be blown onto personnel or equipment during starting operations. Check inflation of shock struts and tires, which may have become overinflated from the heat.

ON ENTERING AIRPLANE

Check the cockpit for excessive accumulation of dust or sand. Check instruments and controls for moisture from high humidity, and ground heat them if necessary to dry them. Complete as much of preflight cockpit check as possible before starting the engine to avoid prolonged ground running.

BEFORE TAKEOFF

The air-conditioning system should be turned ON before takeoff. If, under humid climatic conditions, fog forms in the cockpit, adjust the cabin temperature control knob toward HOT until the fog disappears.

TAKEOFF

WARNING

Excessive moisture condensation may occur through the cabin pressurization system. This condensation may become so dense when operating under conditions of high dew point temperature that it may be impossible to read the instrument panel presentation. In the event this occurs, place the cabin air selector switch to RAM.

CAUTION

It is imperative that takeoff be made at recommended speeds. Refer to the Appendix for takeoff distances required at varying gross weights, temperatures and field elevations. When outside air temperature is high, more than usual takeoff distance will be required to obtain takeoff speed.

DESCENT

Check that the windshield anti-ice and canopy anti-fog system is on at least 4 minutes before any rapid descent from altitude to prevent fogging and frosting of the windshield and canopy.

APPROACH

Maintain the recommended indicated airspeeds for approach and touchdown. Because of high outside air temperatures, the true airspeed will be higher than normal, and longer landing roll will result.

LANDING

Avoid heavy braking during the landing roll. Small increments of braking with the drag chute deployed will stop the airplane in a reasonably short distance without excessive tire wear. Heavy braking may cause brake grabbing and tire failure.

All data on page A1-1 thru A11-22, including figure A1-1 thru A11-22 deleted and related current data published in T.O. 1F-106A-1-1.