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FLIGHT MANUAL
F-102A
USAF SERIES AIRCRAFT

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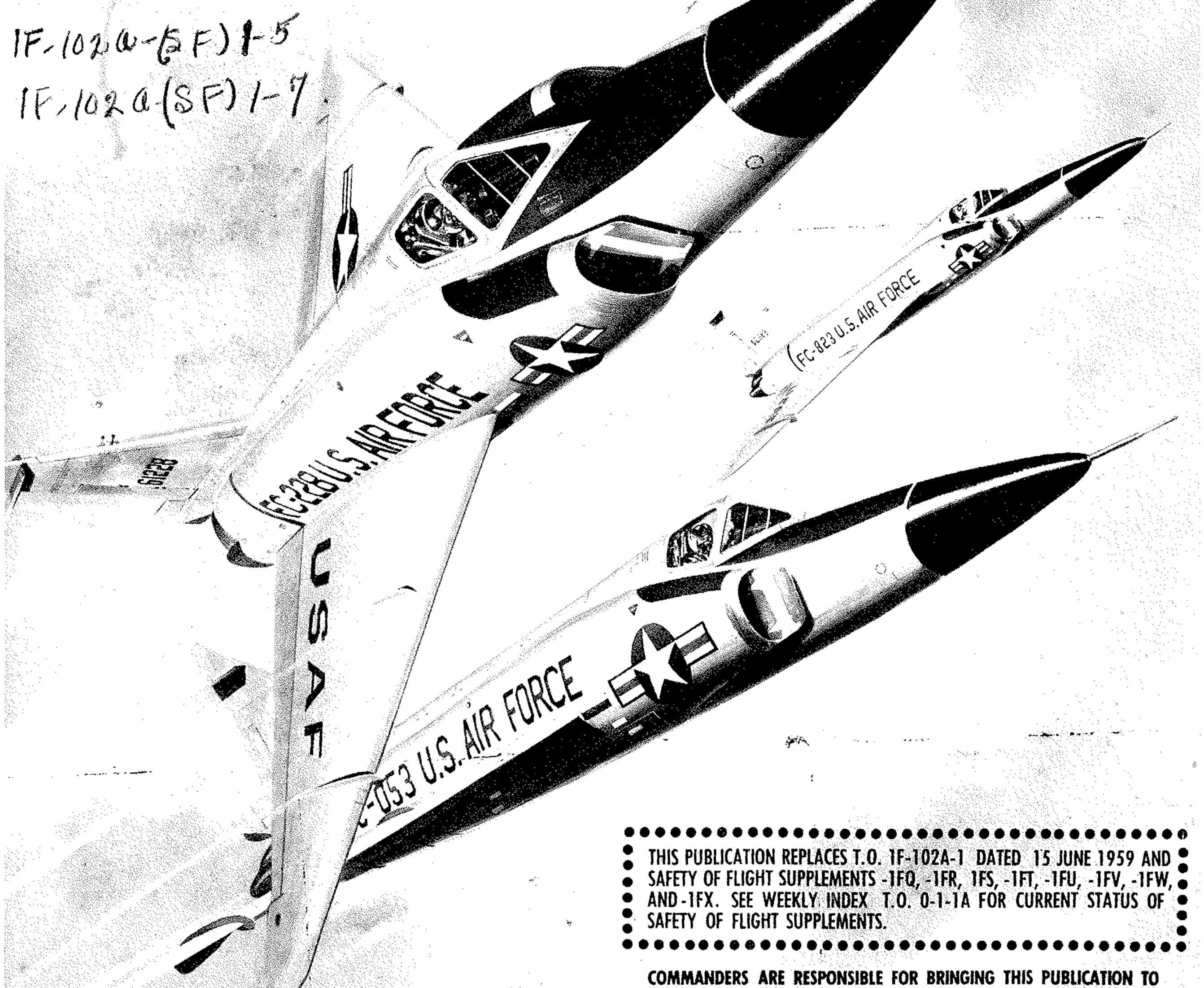

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21800

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15 SEPTEMBER 1960

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TOTAL NUMBER OF PAGES IN THIS PUBLICATION IS 334 CONSISTING OF THE FOLLOWING:

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T.O. 1F-102A-(CL)1-1

15 September 1960

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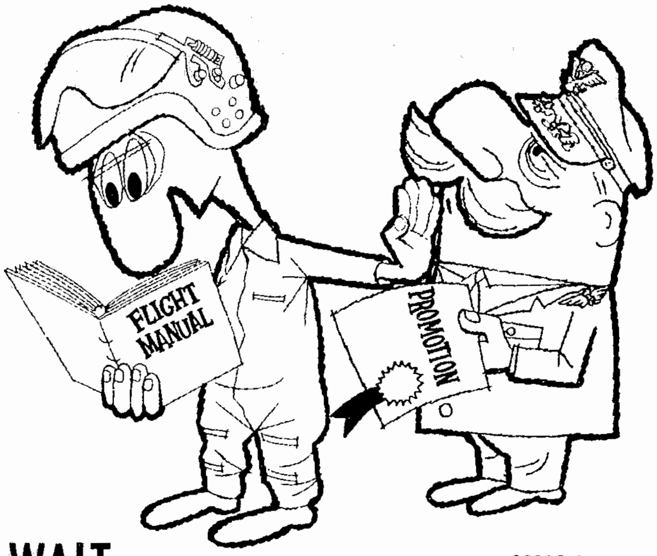

21802-1

WAIT...

THIS INFORMATION IS **IMPORTANT!!**

SCOPE

This manual contains all the information necessary for safe and efficient operation of the F-102A. These instructions do not teach basic flight principles, but are designed to provide you with a general knowledge of the aircraft, its flight characteristics, and specific normal and emergency operating procedures. Your flying experience is recognized, and elementary instructions have been avoided.

SOUND JUDGMENT

The instructions in this manual are designed to provide for the needs of a pilot inexperienced in the operation of this aircraft. This book 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 contained herein.

PERMISSIBLE OPERATIONS

The Flight Manual takes a "positive approach" and normally tells you only what you can do. Any unusual operation or configuration (such as asymmetrical loading) is prohibited unless specifically covered in the Flight Manual. Clearance must be obtained from SAAMA before any questionable operation is attempted which is not specifically covered in the Flight Manual.

STANDARDIZATION

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

ARRANGEMENT

The manual has been divided into ten fairly independent sections, each with its own table of contents. The objective of this subdivision is to make it easy both to read the book straight through when it is first received and thereafter to use it as a reference manual. The independence of these sections also makes it possible for the user to rearrange the book to satisfy his personal taste and requirements. The first three sections cover the minimum information required to safely get the aircraft into the air and back down again. Before flying any new aircraft these three sections must be read thoroughly and fully understood. Section IV covers all equipment not essential to flight but which permits the aircraft to perform special functions. Sections V and VI are obvious. Section VII covers lengthy discussions on any technique or theory of operation which may be applicable to the particular aircraft in question. The experienced pilot will probably be aware of the information in this section but he should check it for any possible new information. The contents of the remaining sections are fairly obvious.

YOUR RESPONSIBILITY

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

PERSONAL COPIES, TABS AND BINDERS

In accordance with the provisions of AFR5-13, each pilot is entitled to have a personal copy of the Flight Manual. Flexible, loose leaf tabs and binders have been provided to hold your personal copy of the Flight Manual. These good-looking, simulated leather binders will make it much easier for you to revise your manual as well as to keep it in good shape. These tabs and binders are secured through your local materiel staff and contracting officers.

HOW TO GET COPIES

If you want to be sure of getting your manuals on time, order them before you need them. Early ordering will assure that enough copies are printed to cover your requirements. Technical Order 00-5-2 explains how to order Flight Manuals, classified supplements thereto, and Safety of Flight Supplements so that you automatically will get all original issues, changes, and revisions. Basically, all you have to do is order the required quantities in the Publication Requirements Table (T.O. 0-3-1). Talk to your Senior Materiel Staff Officer—it is his job to fulfill your Technical Order requests. Make sure to establish some system that will rapidly get the books and Safety of Flight Supplements to the pilots once they are received on the base.

SAFETY OF FLIGHT SUPPLEMENTS

Safety of Flight Supplements are used to get information to you in a hurry. Safety of Flight Supplements use the same number as your Flight Manual, except for the addition of a suffix letter. Supplements covering loss of life will get to you in 48 hours; those concerning serious damage to equipment will make it in 10 days. You can determine the status of Safety of Flight Supplements by referring to the Weekly Supplemental Index (T.O. 0-1-1A). This is the only way you can determine whether a supplement has been rescinded. The title page of the Flight Manual and title block of each Safety of Flight Supplement should also be checked to determine the effect that these publications may have on existing Safety of Flight Supplements. It is critically important that you remain constantly aware of the status of all supplements—you must comply with all existing supplements but there is no point in restricting the operation of your aircraft by complying with a supplement that has been replaced or rescinded. Technical Order 00-5-1 covers some additional information regarding these supplements.

CHECK LIST

The Flight Manual now contains only amplified check lists. The abbreviated check lists have been issued as separate cardboard technical orders. For the T. O. number and date of the latest check list applicable to this manual, see the back of the title page. Order your check list as you would any technical order. Line items in the Flight Manual and applicable cardboard check lists are identical as pertains to arrangement and item number. The cardboard check list is designed for use with binders having plastic envelopes into which the individual cards are placed. You will be advised via your command headquarters when the binders are available for distribution through normal Air Force supply channels.

WARNINGS, CAUTIONS, AND NOTES

For your information, the following definitions will apply to the "Warnings," "Cautions," and "Notes" found throughout the manual:

WARNING

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

CAUTION

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

Note

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

MB-8 FLIGHT COMPUTER

The MB-8 Flight Computer for this aircraft is presently available. This computer is designed to provide pilots of single and twin jet engine aircraft with compact cruise control data which will aid in preparation of flight plans, inflight operation, and emergency inflight planning and operation. The computer is a five-disc, metal and plastic circular computer with a canvas carrying case. Three of the discs can be used with any aircraft and are referred to as "standard discs." The remaining discs contain data only for this aircraft and are described as "data discs." The standard discs and carrying cases are carried in Class 05-A and are available through normal supply channels. The data discs are distributed automatically to all bases having this aircraft. New or revised discs are issued each time the performance data in the Flight Manual is revised. The performance data in the computer and the manual is always kept current and consistent. If you have not yet received your computer, see your Base Operations Officer or T.O. 5F5-1-1. Reference should also be made to T.O. 5F5-1-1 for information on the operation of the computer.

COMMENTS AND QUESTIONS

Comments and questions regarding any phase of the Flight Manual program are invited and should be forwarded through your Command Headquarters to Commander SAAMA, Kelly Air Force Base, Texas. Attn: SANEF.

21802-2

"the F-102A delta dagger"

21803

SECTION I

description

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THE AIRPLANE

The F-102A is a single-place, supersonic, all-weather interceptor built by CONVAIR, A Division of General Dynamics Corporation. The airplane is equipped with a radar fire control system and is powered by a J57-P-23 axial-flow turbojet engine with afterburner. The airplane is best characterized by the large area, 60-degree delta wing and the absence of a conventional empennage. Later airplanes* are equipped with a modified wing (Case XX wing) which produces greater lift and increases performance. The Case XX wing may be distinguished from the wing on earlier airplanes (Case X wing) by the droop at the wing-tip. The delta wing is equipped with "elevons" which provide combination aileron and elevator action from conventional cockpit controls. All control surfaces are hydraulically actuated and incorporate an artificial feel system. The airplane is equipped with a pressurized cockpit which contains an ejection seat. Conventional tricycle landing gear is utilized for takeoff and landing. The aft fuselage mounted speed brakes also serve as compartment doors for a drag chute. The six integral wing tanks are serviced by a single-point pressure refueling system and fuel usage is sequenced automatically to maintain desirable center of gravity.

DIMENSIONS

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

Wing span .... 38 feet, 1.6 inches

Length (including pitot boom) ... 68 feet, 1.8 inches

Height (to tip of vertical stabilizer—
AF 54-1390, -1398, -1401, 55-3357
and subsequent, and airplanes
modified by TCTO 1F-102A-538) 21 feet, 2.5 inches

**AF 55-3385, 56-1317 & on.*

Height (to tip of vertical stabilizer—
all other airplanes) . . . . 18 feet, 2.0 inches

Tread . . . . 14 feet, 2.25 inches

Refer to Section II for minimum turning radius and ground clearance.

GROSS WEIGHTS

Gross weights range from approximately 28,150 pounds to approximately 31,276 pounds, according to various mission loading conditions. The figures below are averages, and are provided to show average gross weights at different configurations:

	Clean	With External Tanks
Empty Weight*	19,903	20,234
Usable Fuel	7,053	9,848
Armament	1,194	1,194
Totals	28,150	31,276

Note

The figures above are averages and are not to be used in flight planning. For further information refer to the Handbook of Weight and Balance Data, T.O. 1-1B-40, and to Weight Limitations, Section V of this manual.

ENGINE

Thrust is supplied by a Pratt and Whitney J57-P-23 or J57-P-23A engine with afterburner (figure 1-5). These two engines are identical except for the type of afterburner igniter. Approximate standard sea level static thrust ratings for either engine is as follows:

MAXIMUM (with afterburning) ....16,000 pounds

MILITARY (without afterburning) ....10,200 pounds

The J57 engine is an axial-flow gas turbine, commonly known as a "two spool" engine because it has two rotors revolving on concentric shafts. Each rotor assembly includes a compressor and turbine. One distinctive characteristic of this type of engine is that the ratio of the turbine discharge pressure to compressor inlet pressure is a better indication of thrust than is rpm. One percent variation in rpm results in approximately five percent variation in thrust at the higher thrust settings; but one percent variation in turbine discharge pressure results in only approximately one and one-half percent variation in thrust. The nine-stage, low-pressure compressor is mounted on a solid drive shaft with a two-stage turbine. The seven-stage, high-pressure compressor is mounted on a hollow drive shaft, rotating around the first with a single-stage turbine. Each rotor set revolves independently of the other. A compressor air bleed system is used to direct part of the low-pressure, compressor air overboard at low engine rpm to aid in fast engine acceleration. The air bleed system is actuated automatically and is controlled by a governor driven by the low-pressure rotor. The engine combustion section includes eight interconnected combustion chambers (burners) arranged in an annular configuration. The main engine accessory section, driven by the high-pressure rotor, provides reduction gearing and mounting pads for all the engine-driven accessories.

Includes 713 pounds clean or 752 pounds with tanks for the weight of the pilot, survival equipment, trapped fuel and oil, etc.

ENGINE FUEL CONTROL SYSTEM

Fuel flow requirements are established by the pilot's throttle movement, and fuel flow to the engine is delivered and regulated by the engine fuel control system (figure 1-6). The system includes the engine-driven fuel pump unit, the fuel control unit, the fuel pressurizing and dump valve, 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 1-6) supplies the fuel pressure required by the engine and afterburner systems. The unit consists of three pumps. A centrifugal pump draws in fuel and forces it to two gear-type pumps. One gear-type pump is the engine stage fuel pump; the other is the afterburner stage fuel pump. The engine fuel pump furnishes fuel to the fuel control unit which regulates fuel flow to the engine combustion chambers. The afterburner fuel pump furnishes 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 afterburner shutoff valve in the engine-driven fuel pump is closed and the output of the afterburner fuel pump is routed back to the discharge stream from the centrifugal pump. A warning light in the cockpit illuminates if the engine fuel pump fails, and a transfer valve in the engine-driven fuel pump unit automatically opens to allow fuel from the output side of the afterburner pump to flow to the fuel control unit. During this condition, fuel is supplied to both the engine and afterburner systems, if the throttle is in AFTERBURNER range. If the engine stage fuel pump fails during conditions of afterburner operation (i.e., takeoff), some additional thrust will be obtained if afterburner operation is continued. However, if the engine is in a non-afterburning condition at the time of engine stage fuel pump failure, afterburner operation should not be attempted. If the afterburner pump fails, the engine fuel pump cannot supply fuel to the afterburner.

Fuel Control Unit

The fuel control unit (figure 1-6) regulates fuel flow to the engine 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 metering valve. The computer, in addition to throttle position requirements, senses changes in flight conditions and regulates fuel flow to insure optimum engine operation. 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. Excess fuel not required by the engine is routed back to the engine-driven fuel pump unit by the main bypass valve. 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. There are no provisions for automatic transfer to the emergency system in event of failure of the normal system. 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 control system will, however, compensate for altitude variations up to approximately 30,000 feet. (At higher altitudes the throttle must be successively retarded to maintain a constant rpm.)

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.

Excess fuel not required by the engine when operating on the emergency system is routed back to the engine fuel pump unit by the emergency bypass valve. When the throttle is placed in OFF, a mechanically controlled cutoff valve in the fuel control unit shuts off all fuel flow to the combustion chambers.

Fuel Control Takeoff Locks

As fuel flow control is not transferred automatically to the emergency system if the normal system fails, mechanical locks are included, in early airplanes, in the fuel control unit to maintain adequate fuel flow for takeoff and initial climb. These locks are controlled by placing the throttle to TAKEOFF position. To avoid feeding too much fuel at high altitude, it is necessary to retard the throttle lever below TAKEOFF position before reaching 7000 feet altitude.

Note

Since normal thrust variations are not obtainable when the takeoff locks are engaged, TAKEOFF position of the throttle is not normally suitable for formation takeoff operations.

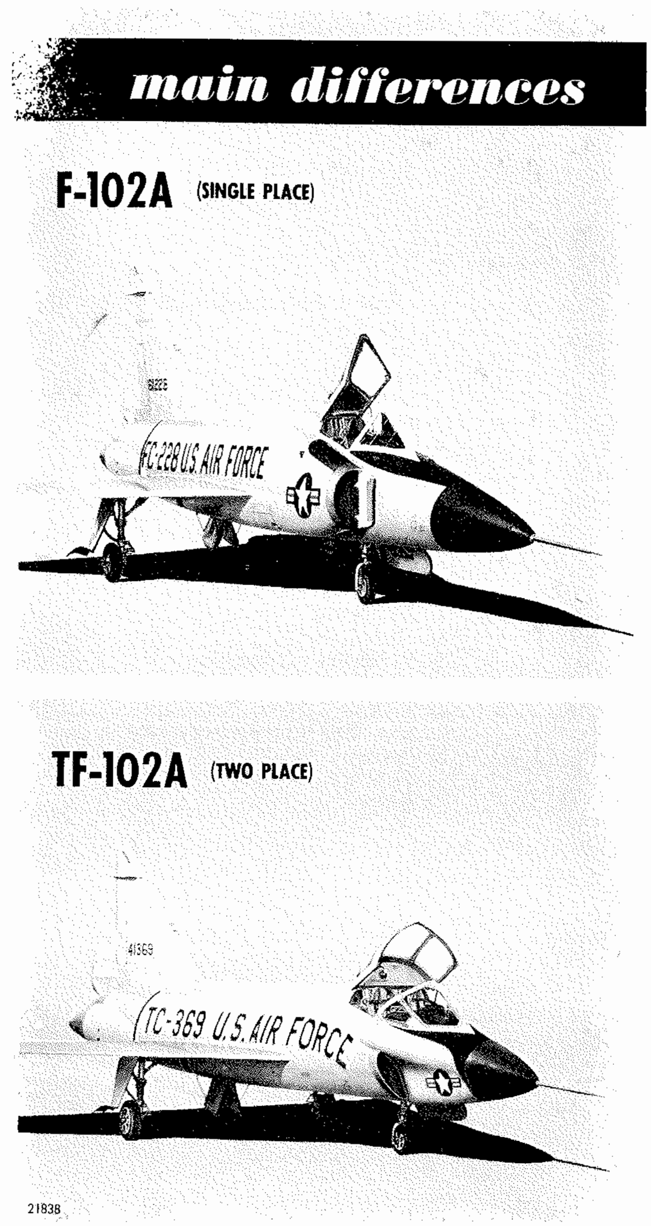


Figure 1-1

On later airplanes, the fuel control takeoff locks are removed to preclude the possibility of critical engine overspeed or temperature conditions when the throttle is inadvertently left in TAKEOFF while climbing above 7000 feet.

Note

A placard is installed on the throttle quadrant on all airplanes which do not incorporate the takeoff locks.

Fuel Pressurizing and Dump Valve

The fuel pressurization and dump valve (figure 1-6) is located in the fuel control system between the fuel valve and the combustion chambers. The unit controls flow to the primary and secondary dual injector nozzles in the

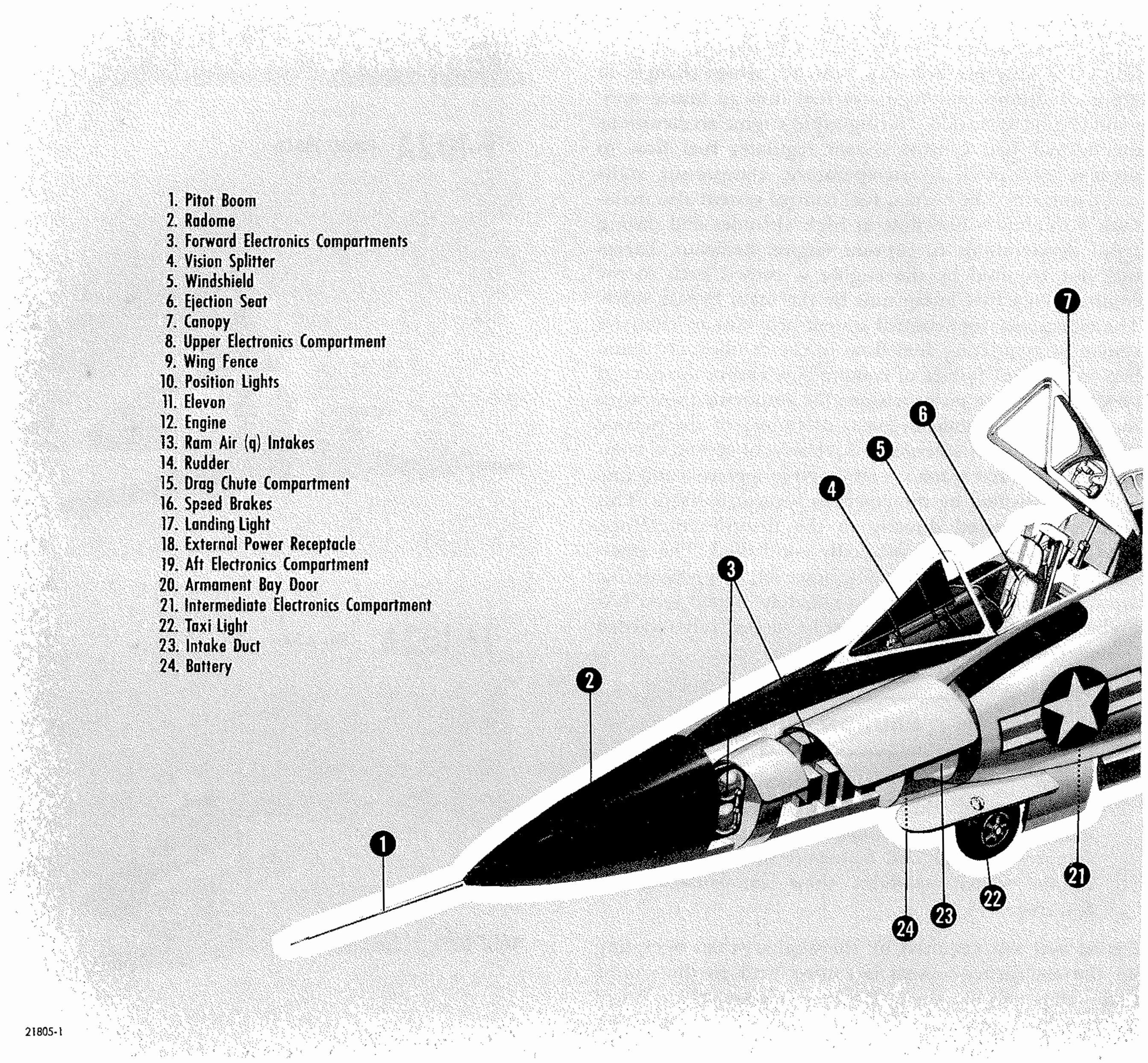


Figure 1-2

engine combustion chambers. To facilitate starting, fuel at relatively low pressure is directed through the primary manifold. 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. When the engine is stopped, the cutoff valve in the fuel control unit is closed by throttle movement, and the dump valve in the pressurizing and dump valve unit opens to permit residual fuel to drain overboard.

THROTTLE

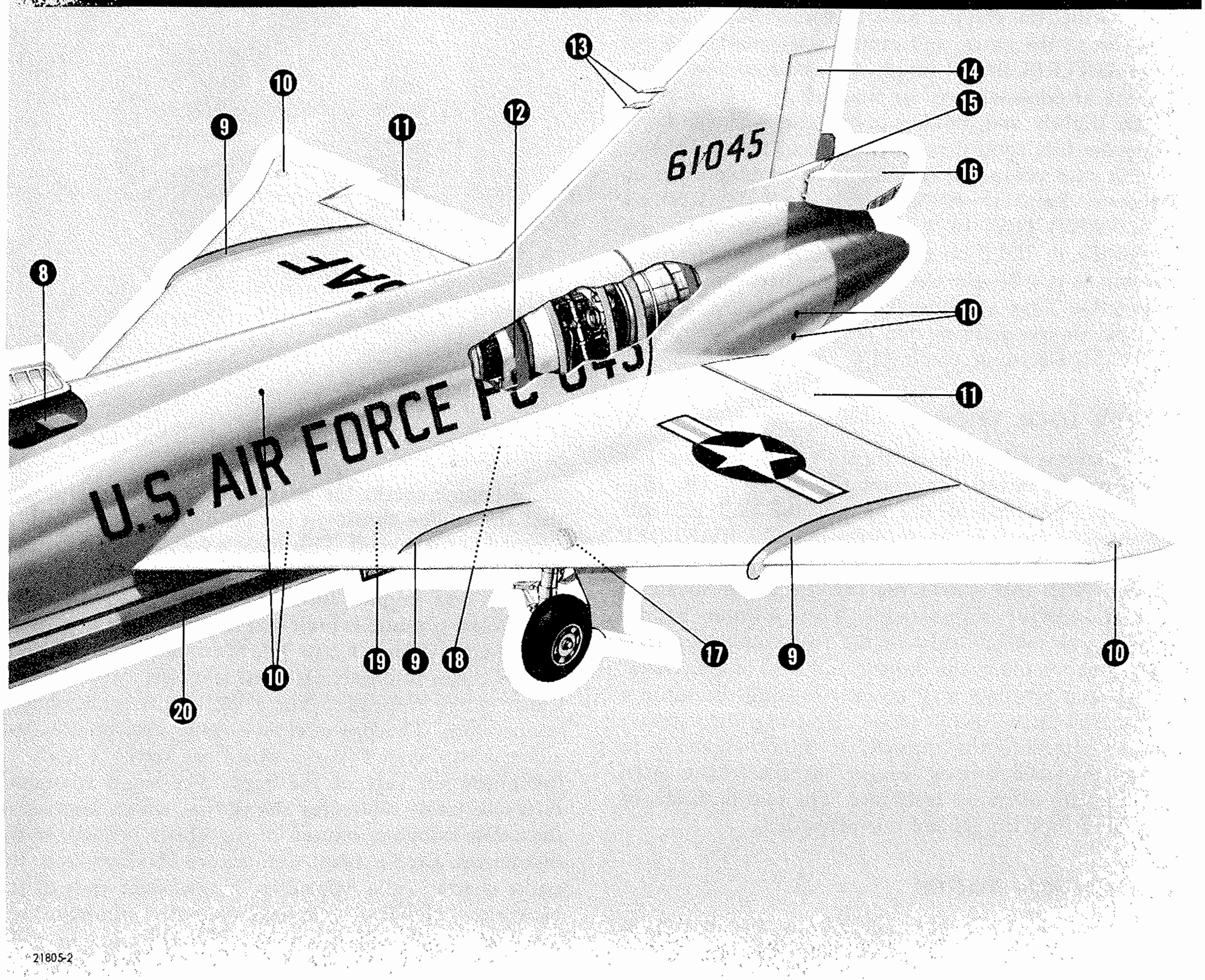
Engine thrust is controlled by the throttle (figure 1-7) which is located on the left-hand console. The throttle is mechanically linked to the fuel control unit and controls the normal and emergency fuel system and afterburner operation.

Note

The throttle is actually a thrust control lever as its function is to set up a thrust condition for which the fuel control meters fuel to the engine with automatic compensation (under normal fuel control) for engine rpm, ambient air conditions and compressor discharge pressure.

The throttle grip contains the speed brakes switch, ignition button, microphone button and takeoff lock trigger (some airplanes). An automatic anti-creep spring assembly is installed in the throttle quadrant to prevent throttle

airplane general arrangement



creep in fore and aft directions, thus eliminating the need for a manual friction lock. Positioning the throttle outboard from OFF to START arms the starter. When the ignition button is depressed (with throttle in START position) the combustion starter is actuated by air and air-motors the engine to approximately six percent rpm. With the ignition button depressed, moving the throttle inboard to OFF energizes the starter and ignition circuit. At the forward IDLE stop, the throttle must be moved inboard to clear the stop. This places the throttle in the IDLE position of the normal range. The throttle may be advanced to the FULL MIL POWER position and after depressing the takeoff lock trigger, to the TAKEOFF position which engages the takeoff locks in the fuel control unit. On airplanes not equipped with fuel control takeoff locks, the throttle should also be advanced to TAKEOFF position after depressing the takeoff lock trigger. Placing the throttle outboard of the mechanical lock prevents inadvertently moving the throttle out of AFTERBURNER when retracting the landing gear. To disengage the mechanical locks, the throttle is moved aft of TAKEOFF position without depressing the trigger as the FULL MIL POWER stop offers no resistance when retarding the throttle. On other airplanes which are not equipped with a takeoff lock trigger, there is no TAKEOFF position on the quadrant. On these airplanes, placing the throttle in FULL MIL POWER and moving it outboard will lock the throttle outboard in AFTERBURNER range until the throttle is retarded 5° aft of the stop. At any point along the normal thrust range forward of the afterburner rear stop, the throttle may be moved outboard to the AFTERBURNER range. A microswitch in the throttle quadrant fires the afterburner when the throttle is moved into this range. A detent mechanism in the quadrant separates the two arcs of travel, so that the throttle must be forced from one side of its slot to the other, and cannot accidentally slip out of the AFTERBURNER range. At maximum power setting, the afterburner may be shut off by retarding the throttle slightly and moving it inboard, without disengaging the fuel control mechanical locks. Throttle position can also cause the landing gear warning light to illuminate. Refer to LANDING GEAR WARNING LIGHT AND TEST BUTTON, this Section. Retarding the throttle to IDLE on early airplanes also activates a jet pump to provide for electronics compartment ground cooling. On later airplanes,* retarding the throttle to 72% rpm or less activates the jet pump. (Refer to Section IV for electronic cooling.)

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TAKEOFF LOCK TRIGGER

A two-position takeoff lock trigger (some airplanes) is located on the throttle quadrant (figure 1-7). The trigger is spring-loaded in the up position and prevents inadvertently moving the throttle into the TAKEOFF thrust range. When depressed, the trigger mechanically lowers a latch that allows the throttle to be advanced to the TAKEOFF thrust range. This position, besides engaging the takeoff locks in the fuel control unit on some airplanes, places the throttle outboard of a mechanical lock that prevents inadvertently moving the throttle out of AFTERBURNER when retracting the landing gear. In retarding the throttle, it is not necessary to depress the takeoff lock trigger, as the FULL MIL POWER stop offers no resistance. The two positions of the takeoff lock trigger are not placarded.

FUEL CONTROL SWITCH

The guarded fuel control switch (figure 1-7) is located on the throttle quadrant, to the left of the throttle, and is used to select either normal or emergency fuel control system. The switch is placarded "Fuel Controls" and has two positions, NORMAL and EMERGENCY. When in NORMAL position the emergency shuttle valve in the fuel control unit is electrically positioned to permit normal fuel control system operation. When in EMERGENCY position, the emergency shuttle valve is electrically positioned to permit emergency fuel control operation. A warning light illuminates when the emergency fuel control switch is in EMERGENCY position. Power is taken from the 28-volt dc essential bus.

ENGINE PRESSURE RATIO GAGE

The engine pressure ratio gage (28, figure 1-4; figure 1-8) is used 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. This gage is located on the instrument panel and compares pitot pressure (from pitot-static system) and engine turbine discharge pressure. This differential pressure is indicated in pressure ratio ranging in increments from 1.2 to 3.4. Maximum and minimum takeoff thrust limits are obtained from the Takeoff Check Chart in the Appendix. These limits vary with ambient temperature. The pressure ratio gage incorporates two adjustable reference marks which are set by a knob on the lower left side of the gage. The small triangular reference mark references the setting which appears in the cruise window, located in the upper portion of the instrument. Cruise thrust settings are obtained from the cruise charts in the Appendix. The extreme ends of the elongated reference mark reference the minimum and maximum takeoff limits and the notched portion of the reference mark represents the trim limits which are utilized for stabilized engine operation. When using the pressure ratio gage to check thrust prior to takeoff, a thrust overshoot may be noted when the throttle is advanced to FULL MIL POWER from IDLE on a cold engine. This thrust overshoot will gradually diminish to the specified value within approximately five minutes. This condition is considered to be normal. Power to the engine pressure ratio gage is supplied from the 200/115-volt, 400-cycle, ac essential bus.

TACHOMETER

The tachometer (15, figure 1-4; figure 1-8) indicates percentages of high-pressure rotor rpm, based on the figure 9976 rpm as 100%. The rpm at which full military thrust is obtained varies for each engine. Therefore, the rpm percentage indicated by the tachometer is not necessarily a direct indication of percentage of military thrust

*AF 55-3371 & on.

cockpit general arrangement (typical)

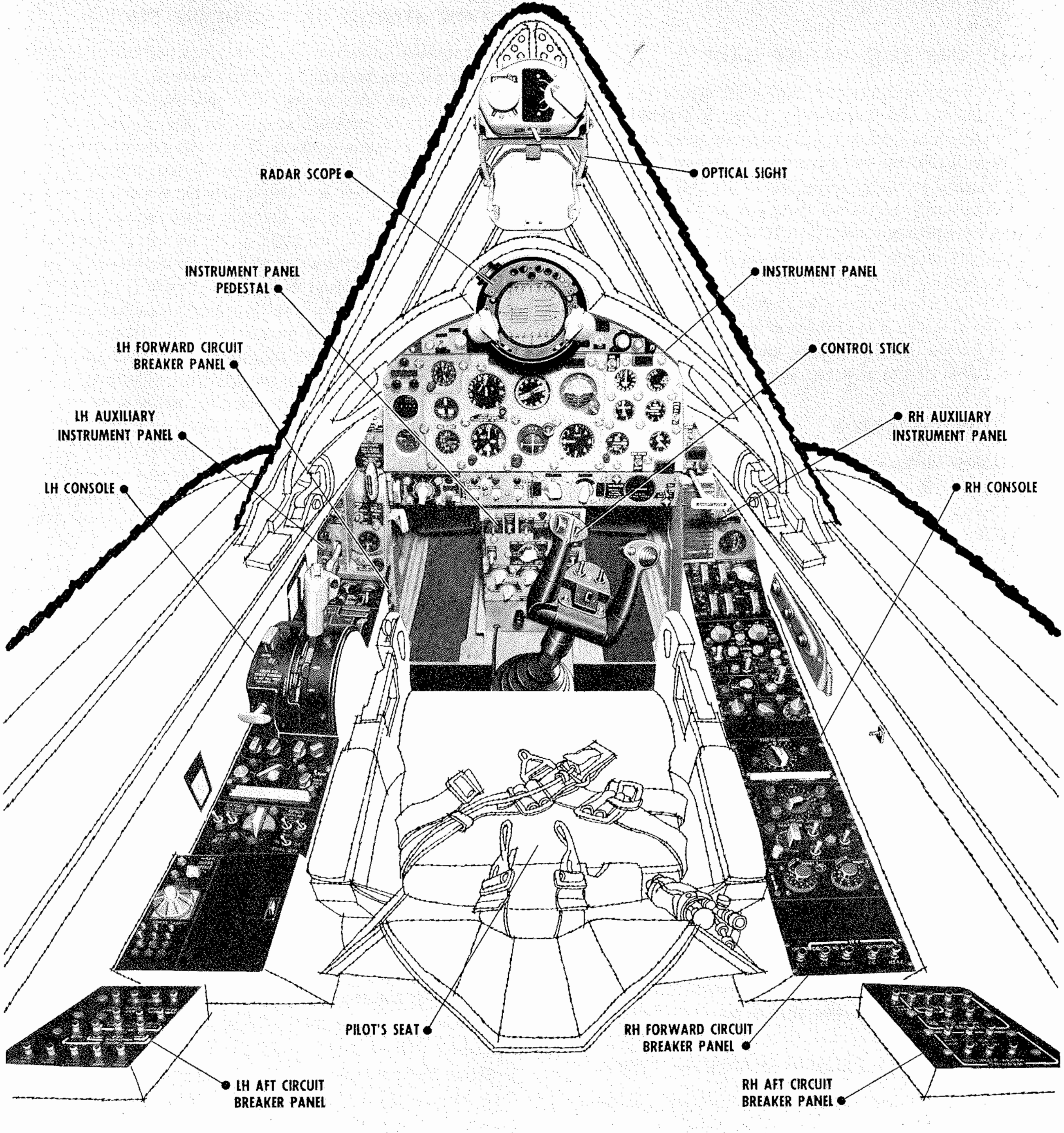


Figure 1-3

rpm. The military power rpm for each engine is placarded on the engine data plate. The tachometer is a synchronous motor which responds to the speed of an engine-driven generator and therefore operates independently of the airplane electrical system. The main pointer is calibrated up to 100% rpm and the sub-pointer makes one complete revolution for each 10% change in engine rpm. By using the sub-pointer, up to 110% rpm can be read.

EXHAUST GAS TEMPERATURE GAGE

The exhaust gas temperature gage (17, figure 1-4; figure 1-8), located on the instrument panel, is a remote indicating instrument which registers in degrees centigrade the turbine discharge temperature measured by thermocouples located in the tailpipe aft of the last stage of the turbine. On some airplanes the face of the indicator is calibrated to indicate from 0° to 1000°C (32° to 1832°F) in increments of 100°C up to 500°C, in 20°C from 500° to 800°C, and in 100°C from 800° to 1000°C. On other airplanes, the indicator is calibrated in increments of 20°C from 0° to 1000°C. The indicator is actuated by voltages produced by the thermocouples, and therefore the system is independent of the airplane electrical system.

FUEL FLOW INDICATOR

The fuel flow indicator (18, figure 1-4; figure 1-8) registers the rate of flow (consumption) of fuel to the engine in pounds per hour. Indications shown are measured by a transmitter located in the fuel line downstream from the fuel control unit (figure 1-6). All fuel entering the engine passes through the transmitter. The indicator is calibrated to show the rate of flow from 0 to 12,000 pounds per hour. The indicator dial is graduated in 100-pound increments up to 3000 pounds and in 1000-pound increments from 3000 to 12,000 pounds. The fuel flow indicator does not indicate fuel flow to the afterburner system. The indicator power is supplied from the 26-volt, 400-cycle ac essential bus.

OIL PRESSURE-LOW WARNING LIGHT

The oil pressure-low warning light (5, figure 1-26), located on the warning light panel, illuminates and displays "OIL PRESS" when the engine oil pressure falls below 36 (±2) psi and goes out when the pressure rises above 40 psi. The light is controlled by a pressure switch in the oil line. The oil pressure-low warning system receives power from the 28-volt dc essential bus.

ENGINE FUEL PUMP FAILURE WARNING LIGHT

The engine fuel pump failure warning light (8, figure 1-26), located on the warning light panel, illuminates and displays "ENG FUEL PUMP" if the engine stage of the engine-driven fuel pump unit fails. Power is supplied from the 28-volt dc essential bus.

EMERGENCY FUEL CONTROL WARNING LIGHT

The emergency fuel control warning light (4, figure 1-26), located in the warning light panel, illuminates and displays "EMER FUEL ON" when the fuel control switch is moved from NORMAL to EMERGENCY position. The light receives power from the 28-volt dc essential bus.

ENGINE STARTER AND IGNITION SYSTEM

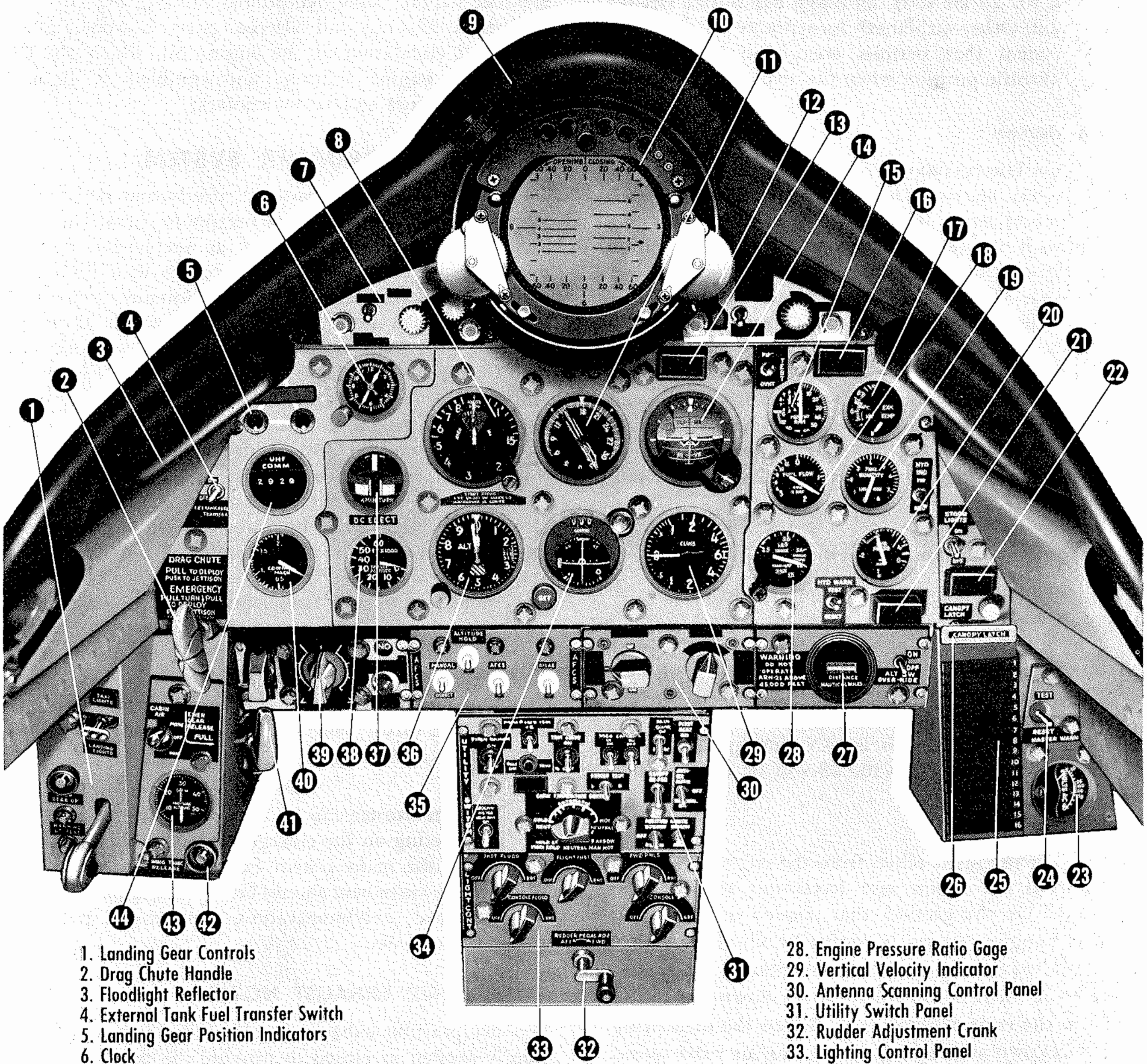
The combustion starter is a self contained unit consisting of a small gas turbine, a drive assembly, and electrical control valves. Air under 3000 psi pressure is supplied from either a ground source or the airplane's high-pressure pneumatic system. When the starter is supplied with air pressure from a ground source, a high-pressure air hose is attached to the pneumatic system filler valve on the forward side of the left main wheel well. The pressure is reduced to 300 psi by an air control valve, through which air flows to the starter combustion chamber and the starter fuel flask. Fuel supply is tapped from the engine fuel supply line, at a point downstream from the left-hand fuel boost pumps. The electrically operated fuel and air valves and the starter ignition system are controlled 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 28-volt dc essential bus.

Note

The air supply, under normal conditions, for the combustion starter will be from a ground cart. In the event that a ground cart is not available, starting air for one start may be taken from the airplane high-pressure pneumatic system supply flasks by placing the manual shutoff valve in the left main landing gear wheel well to the open position placarded AIRCRAFT. The number of starts available will depend upon the mission requirements. For a tactical armament firing mission, air will normally be available for one start. For a mission not requiring armament firing, air will be available for two starts with each start using about 900 psi air pressure.

Ignition is accomplished by the use of high tension transformers, two igniter plugs located in combustion chambers No. 4 and 5, and the ignition button which is electrically connected with the starter control circuit. The engine 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. Since the ignition and starter circuits are electrically

instrument panel (typical)



1. Landing Gear Controls
2. Drag Chute Handle
3. Floodlight Reflector
4. External Tank Fuel Transfer Switch
5. Landing Gear Position Indicators
6. Clock
7. Left-Hand Scope Control Panel
8. Machmeter—Airspeed Indicator
9. Glare Shield
10. Radar Scope
11. Radio Magnetic Indicator
12. Right-Hand Scope Control Panel
13. Engine Fire and Overheat Warning Light
14. Attitude Indicator
15. Tachometer
16. Master Warning Light
17. Exhaust Gas Temperature Gage
18. Fuel Flow Indicator
19. Fuel Quantity Gage
20. Hydraulic Pressure Gage
21. Hydraulic Pressure Low Warning Light
22. Canopy Unlocked Warning Light
23. AC Voltmeter
24. Warning Light Panel Test and Reset Switch
25. Warning Light Panel
26. Canopy Latch Handle
27. Tacan Range Indicator Panel
28. Engine Pressure Ratio Gage
29. Vertical Velocity Indicator
30. Antenna Scanning Control Panel
31. Utility Switch Panel
32. Rudder Adjustment Crank
33. Lighting Control Panel
34. Course Indicator
35. AFCS Control Panel
36. Altimeter
37. Turn-and-Slip Indicator
38. ⚠1 Target Altitude Indicator
39. Armament Control Panel
40. ⚠1 Command Mach Indicator
41. Landing Gear Emergency Extension Handle
42. Wing Tank Release Button
43. Cabin Pressure Altitude Gage
44. Remote Indicator (UHF)

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⚠1 SOME AIRPLANES—SEE APPLICABLE TEXT

Figure 1-4

interconnected, the throttle must be moved outboard to START in order to arm the ignition circuit. The ignition button must be depressed to the START position and held depressed while the throttle is moved inboard to energize the igniters. On some airplanes, the same action is required for an air start, although the starter circuits are bypassed. Other airplanes* incorporate an all-points ignition system that permits energizing the igniters from any throttle position when the airplane is airborne.

Ignition Button

The ignition button (figure 1-7) is used to energize the igniters during starting and is located on the throttle. During ground starts, the throttle must be used in conjunction with the ignition button to arm the ignition circuit and control the starter (refer to THROTTLE, this Section). The ignition button serves the same basic function when making an air start; however, on airplanes with all-points ignition*, it is not necessary to arm the ignition circuit by placing the throttle to START. With the all-points system, the ignition circuit is armed and the button is "hot" whenever the weight of the airplane is off the left main landing gear. The ignition button receives power from the 28-volt dc essential bus.

Engine Ignition Disconnect Switch

The guarded engine ignition disconnect switch is located in the left main wheel well. The switch is used to interrupt power between the ignition power circuit breaker and the engine igniters. The engine can then be "motored" without ignition in the combustion chambers. The two-position switch is placarded "Eng Ign Arm" with ON and OFF positions. The switch is ON during flight. The circuit receives power from the 28-volt dc essential bus.

ENGINE COOLING

In flight, cooling air is taken from the engine air intake ducts to cool the engine and accessories and the electronic bays. On the ground, that portion of the engine within the engine shroud is cooled by bleed air ducted from the low-pressure compressor section of the engine. The bleed air valve is controlled by the down-and-locked switch on the main landing gear. When the main landing gear is down and locked, the cooling air valve opens. When the landing gear is unlocked for retraction, the switch recloses the valve. During ground operation, and at speeds up to 150 knots, the flow in the outer annulus between the shroud and fuselage skin is reversed due to the existence of a vacuum, or low pressure area within the engine air inlet duct. Air enters the fuselage at the tail cone and flows forward over the engine shroud, through the engine accessory section, then flows into the engine air inlet duct via the air-oil cooler duct and the intake duct scroll. Reverse airflow also occurs at the aft electronic bay cooling air duct, whereby air is drawn in from the landing gear wheel well and exits into the engine air intake duct. The cooling air for reverse flow of the ac and dc generator ducts is drawn from the outside of the fuselage. This air is controlled by a differential pressure flapper valve. Additional cooling, especially of the engine bearings and adjacent parts, is accomplished by forced circulation of the engine oil, which absorbs heat from engine bearings and conducts it to heat exchangers (fuel and air-oil coolers).

ENGINE AFTERBURNER SYSTEM

The engine is equipped with an afterburner which augments engine thrust to obtain maximum thrust (approximately 50% additional thrust at sea level). Engine operation with maximum thrust results in a high fuel consumption rate; therefore, afterburner operations should be used only when this additional thrust is required. Operation of the afterburner is controlled by the throttle (see figure 1-6). When the throttle is placed outboard to AFTERBURNER, the afterburner shutoff valve (in the engine-driven fuel pump unit) is opened by power from the 28-volt dc essential bus. Fuel is routed to the afterburner metering valve which regulates afterburner fuel in relation to compressor discharge pressure (which is governed by flight conditions and engine speed). Metered fuel is then supplied to the afterburner igniter and afterburner fuel spray bars. At the same time, unmetered afterburner fuel actuates the exhaust nozzle control to pneumatically open the nozzle during afterburner operation. When engine thrust is varied, afterburner thrust will also vary. Thrust variation will range from maximum thrust available to approximately 50% afterburning thrust when the throttle is retarded in the AFTERBURNER range.

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 at the end of the tail pipe is opened or closed to provide the proper exhaust area for either normal or afterburner engine operation. The nozzle flaps are moved to the full open position during afterburner operation and returned to the minimum nozzle opening area when the afterburner is not in use. Positioning of the nozzle flaps is accomplished automatically by means of the exhaust nozzle control unit. No emergency override control is included.

AFTERBURNER IGNITER

When the afterburner system is actuated, fuel from the afterburner metering valve is directed to the igniter unit. (See figure 1-6.) This metered fuel flow actuates the

**Airplanes modified by TCTO 1F-102-746.*

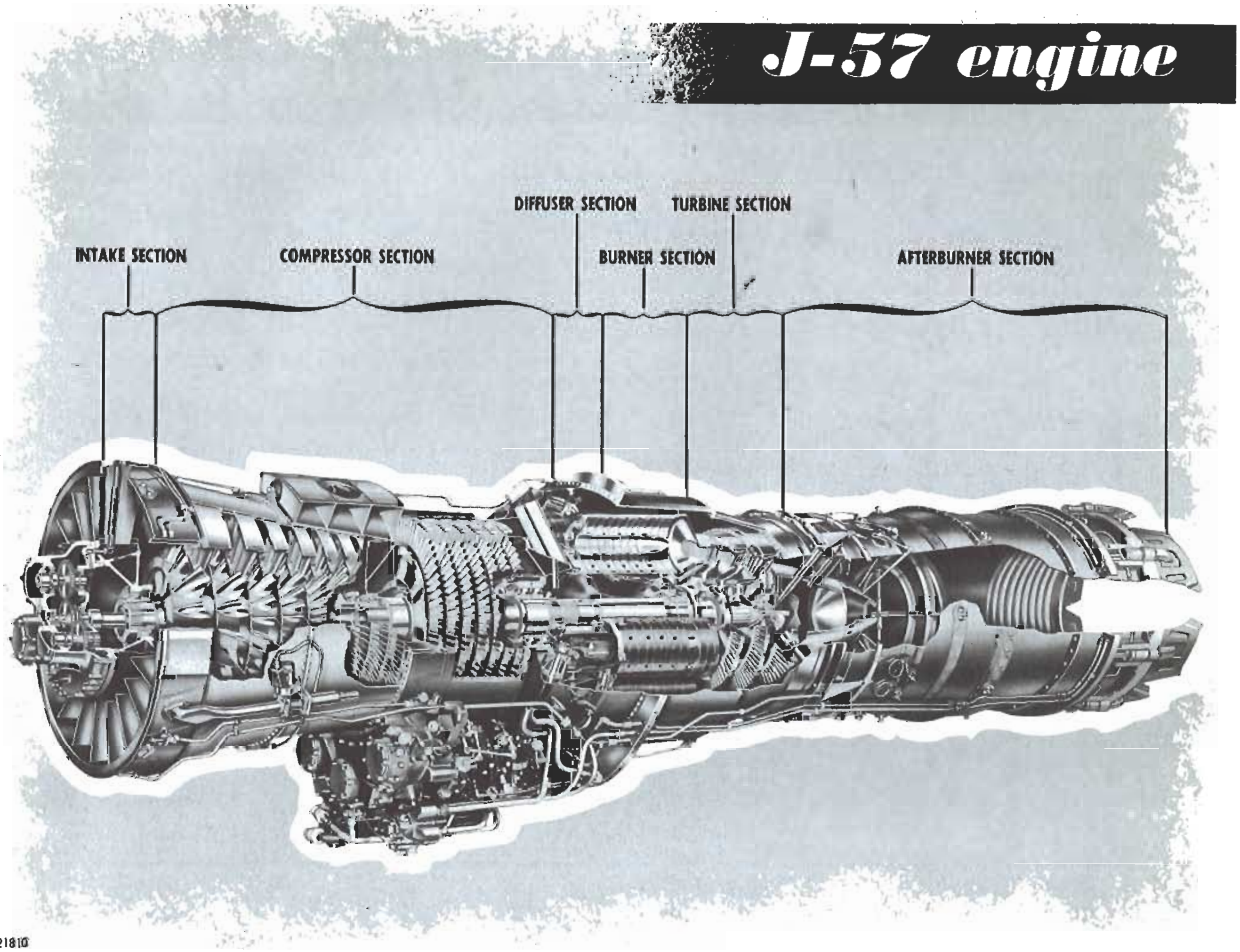

Figure 1-5

igniter unit, which momentarily injects a small amount of fuel into one of the engine combustion chambers, thereby creating a local excessively rich fuel-air mixture. The excess fuel forms a longer 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 igniter is actuated only when full pressure is built up within the afterburner manifold, so that fuel is available at the spray bars when the igniter introduces fuel to the burner for ignition. If afterburner light-up is not accomplished within two seconds at sea level (five seconds at altitude) after the throttle is moved outboard to the AFTERBURNER range, the throttle should again be moved inboard, and then after two to five seconds it should be moved to AFTERBURNER range to recycle the igniter. An afterburner recirculating igniter is installed that improves internal cooling, thereby preventing "coking" and increasing reliability of the system.

AFTERBURNER EXHAUST NOZZLE CONTROL UNIT

Opening and closing the afterburner exhaust nozzle is controlled by afterburner fuel pressure through the exhaust nozzle control unit (figure 1-6). When the throttle is moved outboard to AFTERBURNER, the electrically operated shutoff valve in the fuel pump unit is opened (by power from the 28-volt dc essential bus), permitting unmetered fuel from the afterburner fuel pump to enter the afterburner exhaust nozzle control unit. This fuel pressure moves a valve within the control unit, which directs engine bleed air to the nozzle actuating cylinders to open the nozzle flaps. Moving the throttle inboard from AFTERBURNER range closes the shutoff valve, so that fuel pressure is no longer supplied to the nozzle control unit. The valve in the control unit is then positioned to route engine bleed air to close the nozzle flaps for normal engine operation.

engine fuel control system

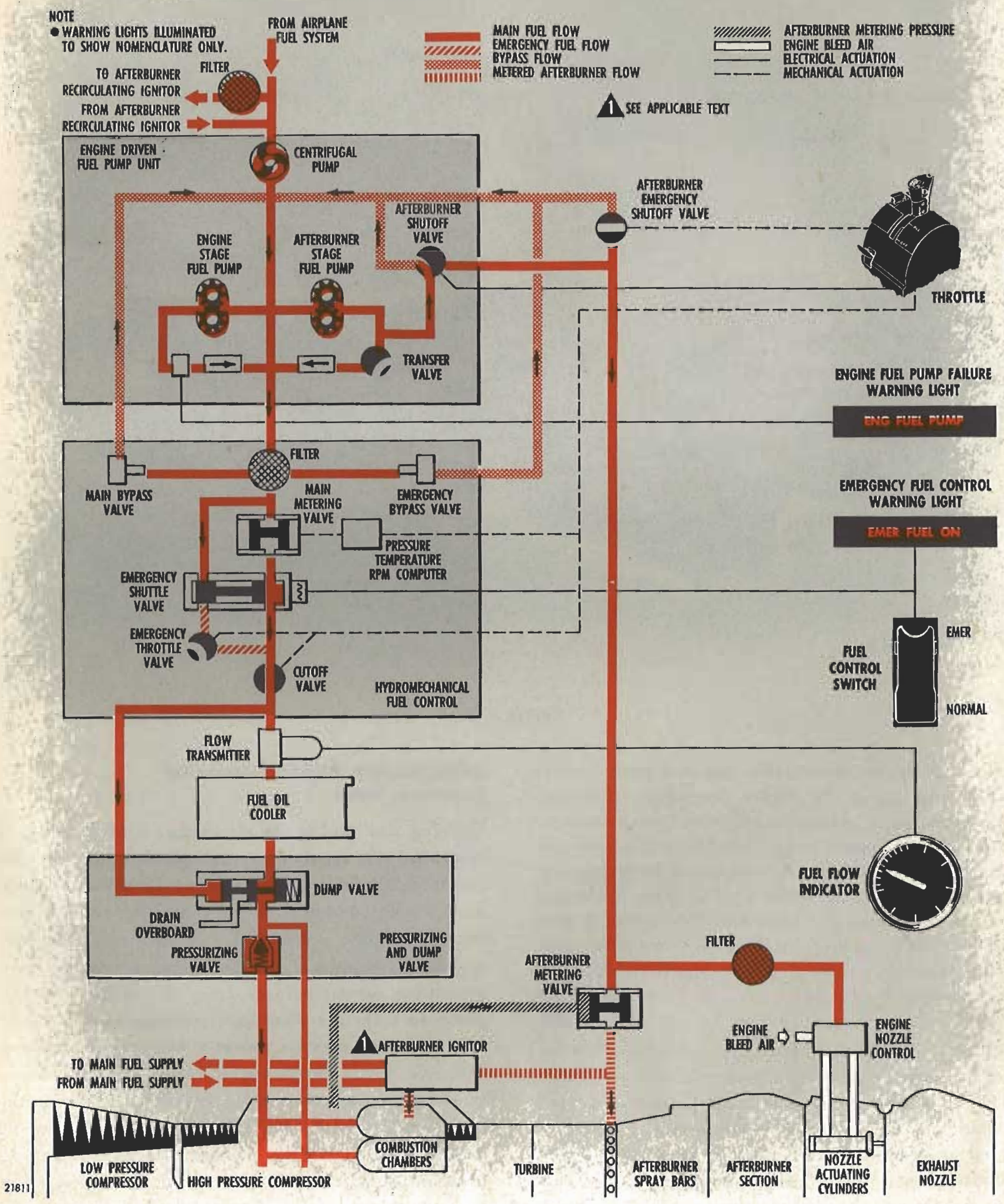

Figure 1-6

throttle

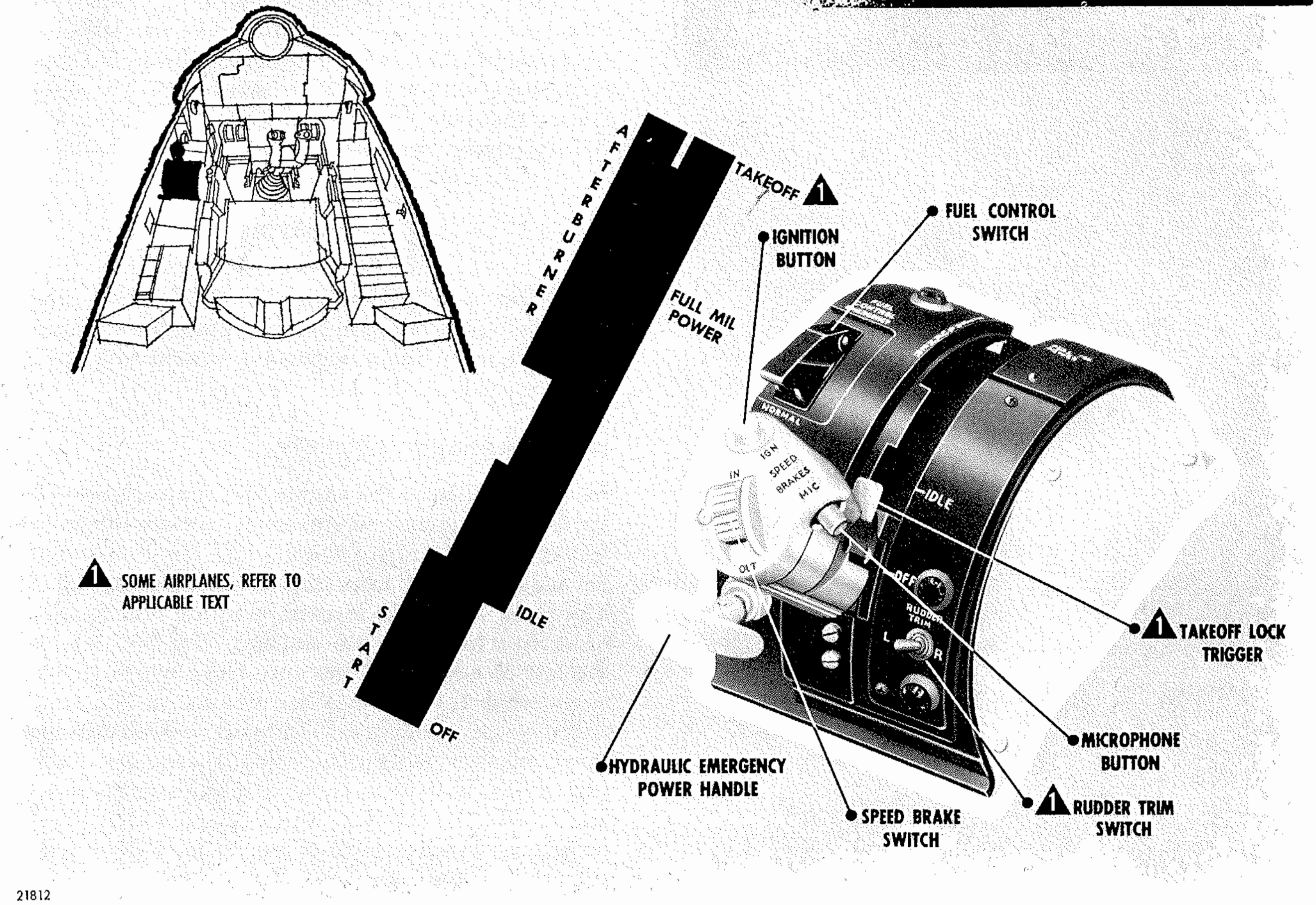

Figure 1-7

AFTERBURNER EMERGENCY SHUTOFF

The afterburner system is shut off mechanically in the event afterburner normal electrical control fails. When the throttle is moved out of the AFTERBURNER range and then retarded below approximately 90% rpm (throttle should be retarded smartly through this range) mechanical linkage will open the afterburner emergency bypass valve (see figure 1-6). All fuel being supplied to the afterburner system will then be returned to the engine fuel pump unit. This terminates afterburning operation and closes the exhaust nozzle. Whenever the throttle is advanced through the AFTERBURNER cut-off range, the bypass valve will close and again permit afterburning operation. If the afterburner shuts off normally, the operation of the emergency shutoff bypass valve will not affect normal operation.

OIL SUPPLY SYSTEM

The engine oil system supplies the engine with oil for lubrication and cooling and also supplies oil to serve as both a lubricating and an actuating fluid in the ac generator constant-speed drive unit. The engine oil tank has a capacity of 5.5 U. S. gallons plus an expansion space of 1.6 U. S. gallons. Maximum engine oil consumption is four pints per hour. Oil from the engine oil tank is taken into a gear-type pump, from which it is discharged under pressure to the engine gears, bearings, and also to the constant-speed drive, engine-mounted gear box. Scavenged oil is picked up by six gear-type pumps and routed through an air-oil cooler and to a fuel-oil cooler thermostatic valve. If the oil has been cooled sufficiently it is then routed to the oil tank. If it is not cooled sufficiently the oil is routed through a fuel-oil cooler and then returned to the oil tank. The oil tank contains a de-aerator which separates entrained air from the oil. The fuselage mounted constant-speed drive transmission oil supply is taken from the bottom of the engine oil tank. On some airplanes the oil is gravity-fed to the transmission. On other airplanes the oil reaches the transmission through a combination boost pump and negative g recirculating valve. During negative g conditions, the

engine instruments (typical)

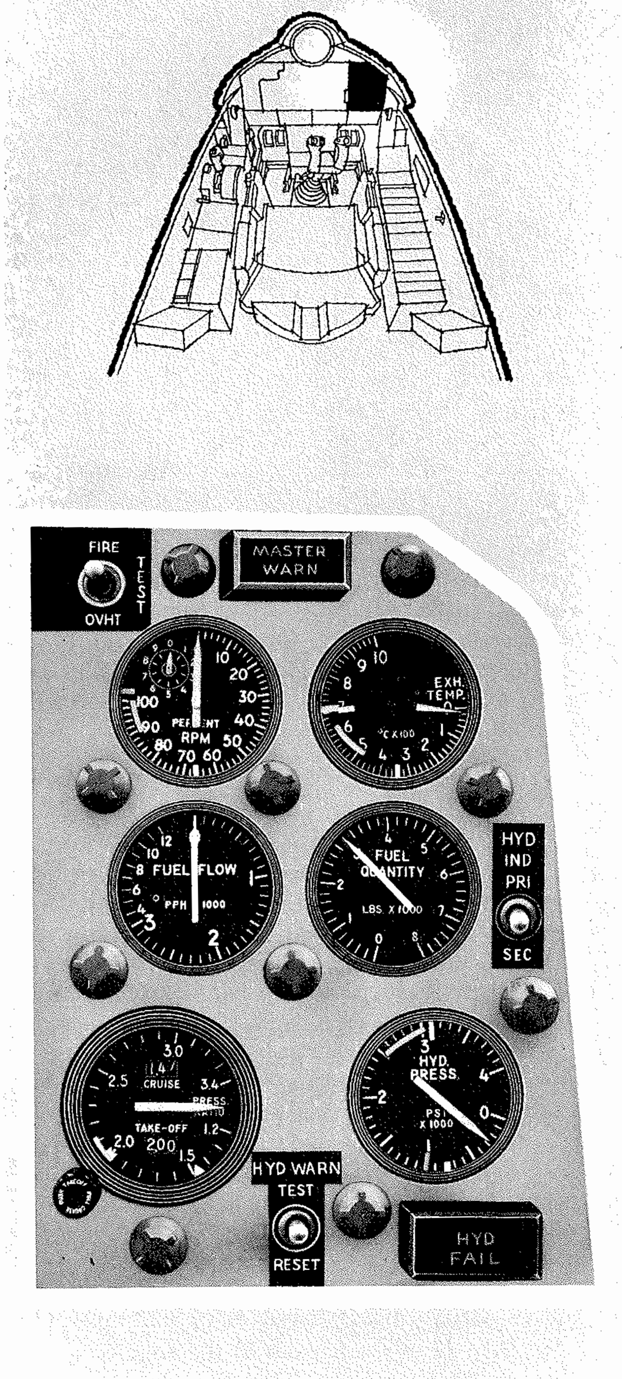

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Figure 1-8

boost pump, recirculating valve, and an accumulator maintain a steady supply of oil to the constant-speed drive transmission by diverting the scavenge oil to the inlet of the transmission and recirculating it until g conditions return to normal. On some airplanes* a standpipe oil system has been installed. The standpipe reduces the possibility of complete loss of engine oil if a constant-speed drive failure results in loss of oil. The standpipe system provides a seven-quart emergency reserve of oil for the engine. See figure 1-33, for oil specifications.

CAUTION

Refer to PROHIBITED MANEUVERS, Section V, for negative g flight limits for airplanes with and without negative g recirculating valve installed.

FUEL SUPPLY SYSTEM

Fuel is supplied to the engine fuel control system by the airplane fuel system which consists of six integral tanks located in the wings (figure 1-9). The fuel supply can be augmented by installing droppable external tanks. The three fuel tanks in each wing are numbered in the order in which they are emptied. The No. 3 tanks are the last to empty and the only tanks which feed to the engine. No. 1, No. 2, and No. 3 tanks are located respectively in the aft-outboard, forward, and aft-inboard sections of each wing. The wing spars, ribs, and skin embody the individual tanks, and joints within the tanks are sealed by a bonding agent. Access to the individual tanks is obtained through access doors on the wing lower surface. Fuel is transferred from tank to tank within each wing by pressurization. This pressurization is provided by engine compressor bleed air and also serves to prevent excessive vaporization at high altitudes. Air pressure enters each No. 1 tank through a pressure regulator. As fuel is drawn from the No. 3 tank, the pressure differential between tanks forces fuel from the No. 1 tank to the No. 2 tank which, in turn, forces fuel into the No. 3 tank. Relief valves are provided to automatically protect the tanks against excessive pressure and vacuum. During a dive, the fuel transfer lines to the No. 3 tanks might be uncovered, disrupting fuel transfer and allowing air to enter these tanks while fuel remains in the No. 2 tanks. As level flight is resumed, float valves automatically vent the No. 3 tanks to atmosphere, restoring the pressure differential and normal fuel transfer. Fuel is supplied from the No. 3 tanks through fuel shutoff valves and a fuel flow equalizer to the engine fuel control system (figure 1-6). Two electrically driven boost pumps in each No. 3 tank provide additional system pressure. However, if the boost pumps are inoperative, the engine-driven fuel pump, aided by tank pressurization, will supply fuel through a dual check valve inlet bellmouth located on the output side of the boost pumps. When operating without boost

**Airplanes modified by TCTO 1F-102-694 and 2J-J57-591.*

pumps, care is required to prevent unporting the inlet bellmouths which will let air enter the engine fuel control system and could cause flameout.

SINGLE-POINT REFUELING

The internal fuel tanks can be serviced only by a single-point refueling adapter located in the aft side of the left main wheel well. During refueling operations fuel is routed to each No. 3 tank through a flow limiter-pressure regulator and refueling shutoff valve. When each No. 3 tank has filled, fuel passes into and fills each No. 2 tank and then the No. 1 tanks. As soon as each No. 1 tank fills, a high-level float valve will automatically close the refueling shutoff valve by hydraulic action and stop fuel flow input. External electrical power is not required during refueling. Inflight refueling provisions are not provided on this airplane.

Note

Fuel pressure of 55 to 60 psi should be provided from the fuel truck during refueling to obtain satisfactory operation of the refueling system.

Fuel tank capacities are given in figure 1-11 and fuel specifications in figure 1-33.

EXTERNAL TANKS

Droppable external fuel tanks can be installed on some airplanes* to augment the internal fuel supply. Installation provisions are made on the lower surface of each wing for two standard 230-gallon tanks which can be jettisoned by a ballistic charge. The fuel in these tanks is transferred into the No. 1 tanks by engine bleed air pressure which is regulated at a higher pressure than that of the normal fuel system pressurization. On some airplanes, once the engine has started, fuel transfer from the external tanks commences automatically. On later airplanes**, fuel from the external tanks may be controlled by an external fuel transfer switch (4, figure 1-4; figure 1-10). High-level floats within the No. 1 tanks will close shutoff valves in the fuel transfer lines between the external tanks and the No. 1 tanks by hydraulic action. This prevents the higher fuel pressure within the external tanks from entering the internal fuel system. A low-level float switch (power from the dc nonessential bus) in each external tank will close a shutoff valve in the pneumatic pressure line, preventing excessive pneumatic pressure from entering the internal fuel system, when fuel is exhausted from the external tanks. Vacuum relief valves in the pneumatic pressure lines will relieve negative pressures in the external tank during high rates of descent. 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.

**AF 53-1791, -1794, 56-1045 & on, & airplanes modified by TCTO 1F-102-534.*

***AF 53-1791, 1794, 56-1045 & on, & airplanes modified by TCTO 1F-102-677.*

Note

In the event of dc generator failure, loss of power to the low-level float switches in the external tanks would cause fuel transfer from these tanks to stop.

The external tanks are jettisoned by depressing the external tank jettison button. A ballistic charge is used to unlock and separate the tanks and pylons from the wings. Refer to Section V for operating limitations with external tanks installed.

FUEL BOOST PUMPS

Two electrically driven, submerged dual impeller boost pumps are installed in each No. 3 fuel tank (figure 1-9) to supply fuel under pressure to the engine fuel control system. Operation of these pumps is controlled by switches located on the fuel control panel. Each pump contains two suction-feed lines for fuel pickup at both top and bottom of the tank. Also, the pumps are diagonally located in the fuel tank to insure fuel supply regardless of flight attitude when there is usable fuel remaining in each No. 3 wing tank. If the boost pumps are inoperative, a dual-check valve inlet, located in the output fuel line, will permit the engine-driven pump to supply fuel to the engine by suction feed. These impeller boost pumps will not pump air. As long as any of the four pumps is operating in fuel, an uninterrupted flow of fuel is assured and entry of air into the engine fuel system is precluded. The pumps are powered by the 200/115-volt, 400-cycle nonessential ac bus and are protected by fuses. Control relays receive power from the 28-volt dc essential bus.

Note

Failure of boost pumps within one wing fuel system will establish a bypass condition of the fuel flow equalizer. Asymmetrical fuel usage should be corrected by shutting off boost pumps within the opposite wing.

FUEL SHUTOFF VALVES

Two electric, motor-driven, sliding-gate shutoff valves are used to cut off fuel supply to the engine from the fuel tanks. These valves, located one in each No. 3 tank outlet fuel line, can be controlled from the fuel control panel on the left console. On some airplanes, the valves can be controlled individually or simultaneously by use of the fuel selector switch. On other airplanes* a separate fuel shutoff valve switch is provided for each valve. The fuel shutoff valves are normally open throughout a flight and are powered by the 28-volt dc essential bus.

Note

The valves require three to five seconds to rotate fully closed. If the master switch is turned OFF during rotation, the sliding gate shutoff valves remain partially open.

**Airplanes modified by TCTO 1F-102-690.*

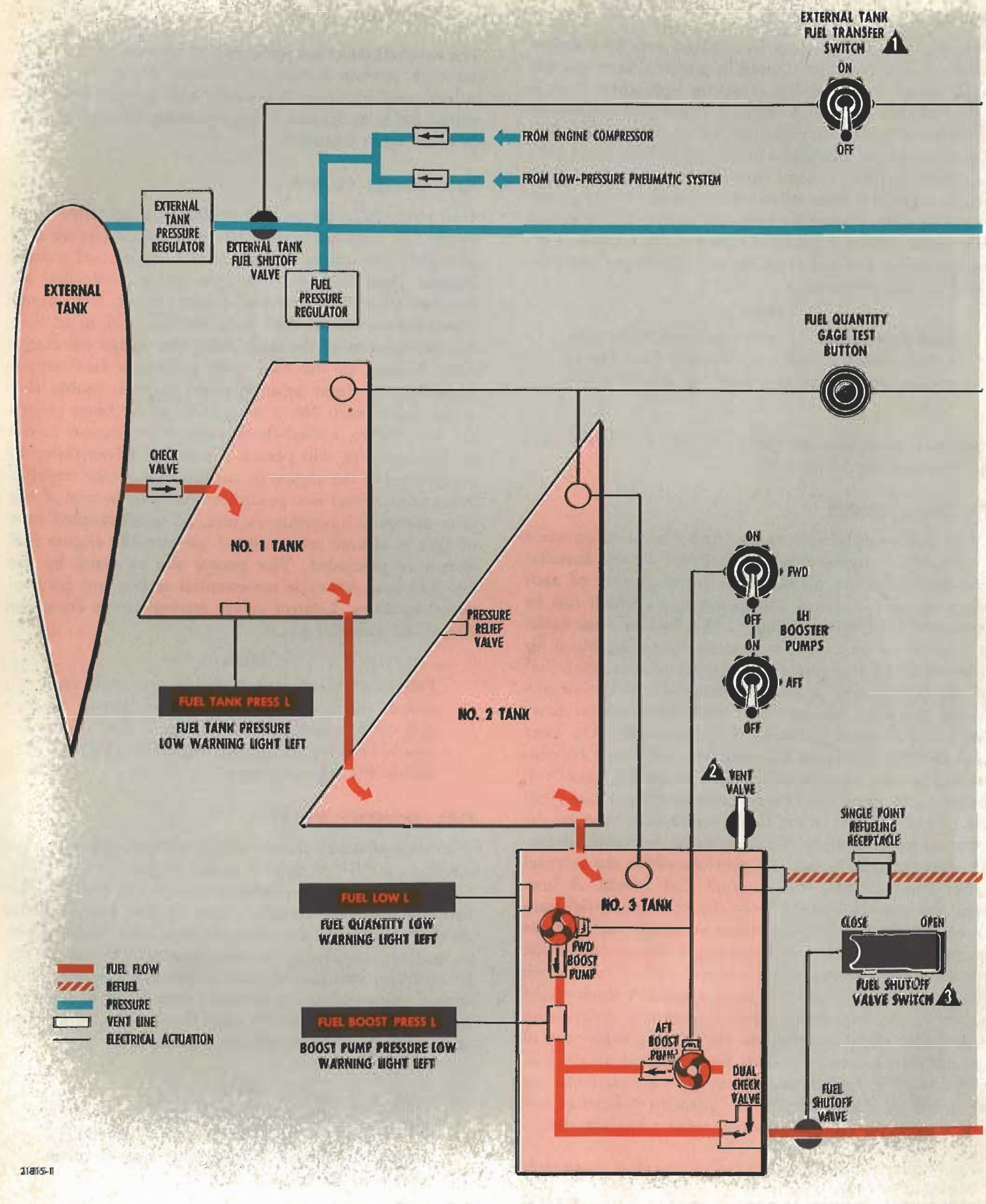
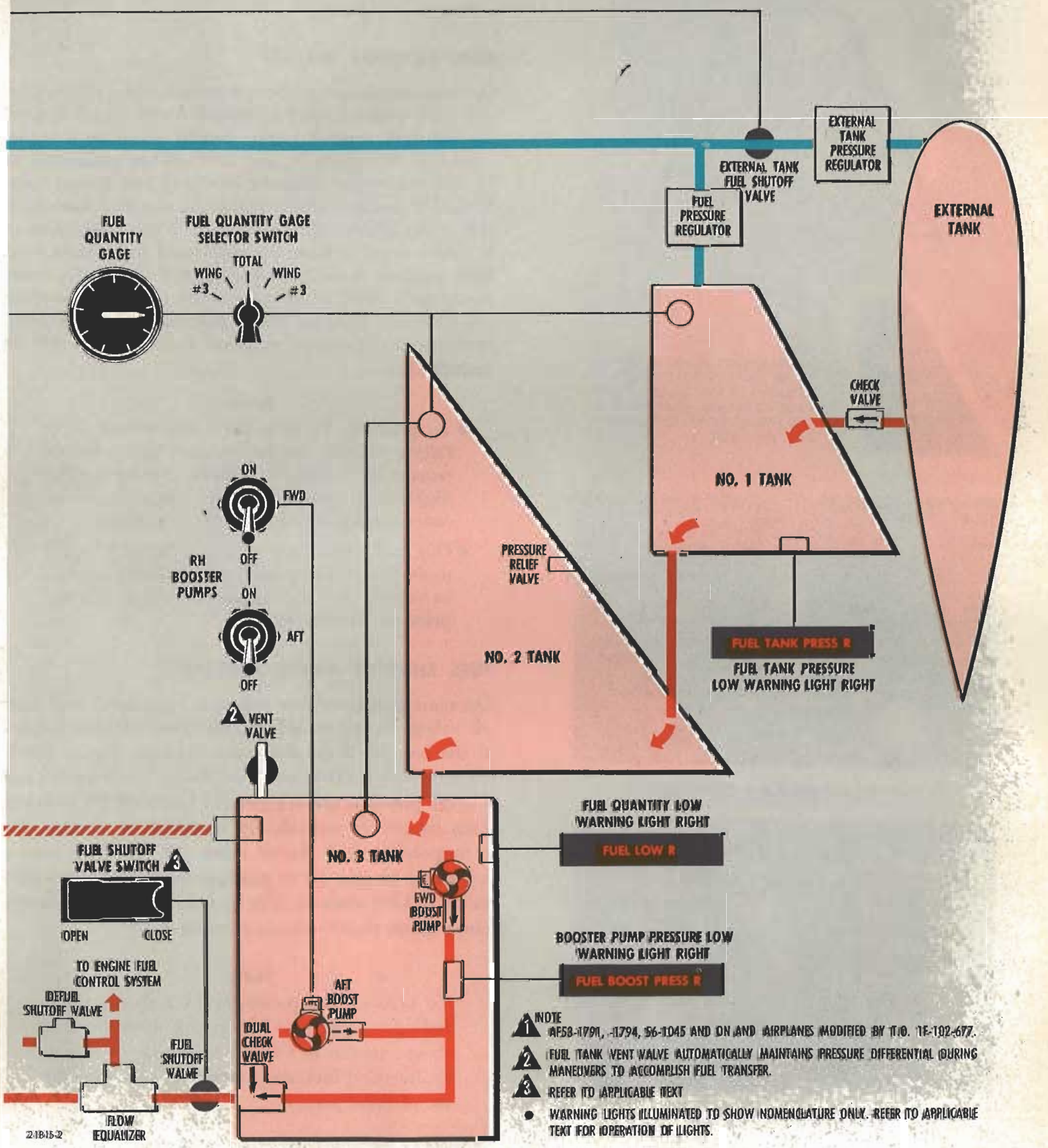

EXTERNAL TANK FUEL TRANSFER SWITCH 1
ON
OFF
FROM ENGINE COMPRESSOR
FROM LOW-PRESSURE PNEUMATIC SYSTEM
EXTERNAL TANK PRESSURE REGULATOR
EXTERNAL TANK FUEL SHUTOFF VALVE
FUEL PRESSURE REGULATOR
EXTERNAL TANK
FUEL QUANTITY GAGE TEST BUTTON
CHECK VALVE
NO. 1 TANK
PRESSURE RELIEF VALVE
FUEL TANK PRESS L
FUEL TANK PRESSURE LOW WARNING LIGHT LEFT
NO. 2 TANK
ON
FWD
OFF
ON
LH BOOSTER PUMPS
AFT
OFF
2 VENT VALVE
SINGLE POINT REFUELING RECEPTACLE
NO. 3 TANK
FUEL LOW L
FUEL QUANTITY LOW WARNING LIGHT LEFT
FWD BOOST PUMP
CLOSE
OPEN
FUEL SHUTOFF VALVE SWITCH 3
FUEL FLOW
REFUEL
PRESSURE
VENT LINE
ELECTRICAL ACTUATION
FUEL BOOST PRESS L
BOOST PUMP PRESSURE LOW WARNING LIGHT LEFT
AFT BOOST PUMP
DUAL CHECK VALVE
FUEL SHUTOFF VALVE
21815-11

Figure 1-9

fuel supply system



NOTE

1. AF58-1791, -1794, 56-1045 AND ON AND AIRPLANES MODIFIED BY T.O. 1F-102-677.
2. FUEL TANK VENT VALVE AUTOMATICALLY MAINTAINS PRESSURE DIFFERENTIAL DURING MANEUVERS TO ACCOMPLISH FUEL TRANSFER.
3. REFER TO APPLICABLE TEXT

● WARNING LIGHTS ILLUMINATED TO SHOW NOMENCLATURE ONLY. REFER TO APPLICABLE TEXT FOR OPERATION OF LIGHTS.

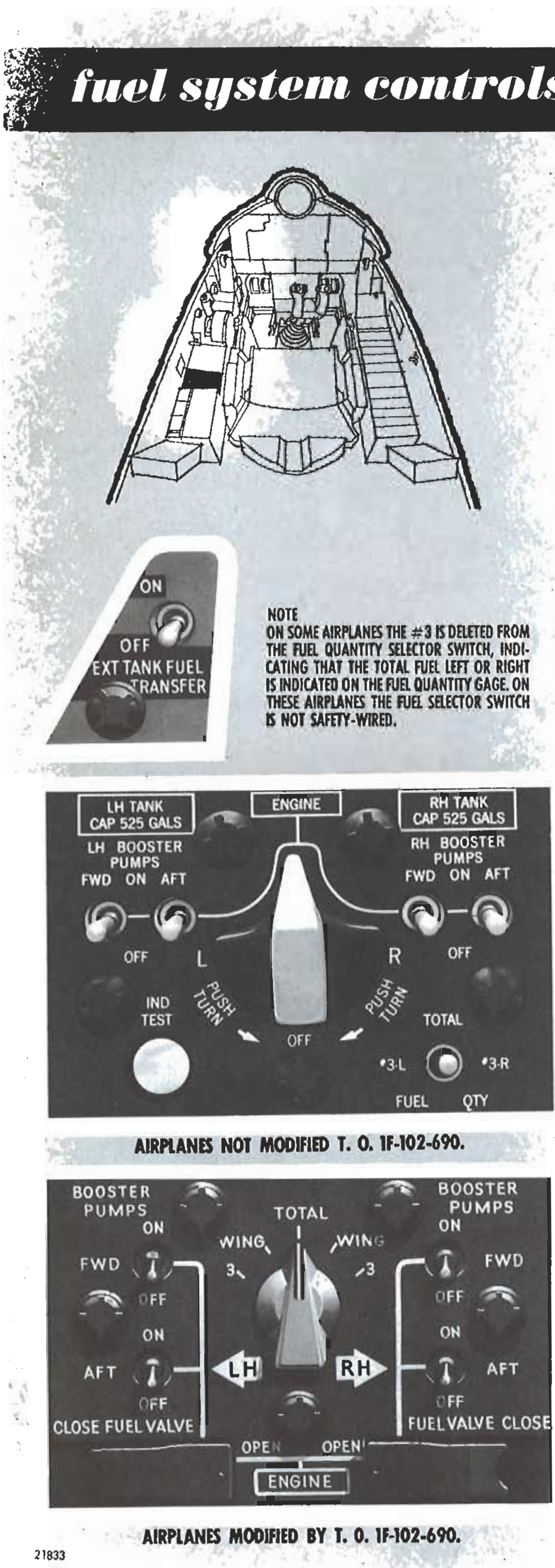


Figure 1-10

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 the No. 3 tank and the engine fuel control unit. A bypass condition is automatically established to insure fuel supply in the event of boost pump failure within one wing or malfunction of the flow equalizer.

FUEL SELECTOR SWITCH

On some airplanes* a four-position selector switch (figure 1-10) controls the fuel shutoff valves and is located on the fuel control panel. Switch positions are placarded L, ENGINE, R, and OFF. When positioned to L, fuel is supplied from the left-hand fuel system only. ENGINE position allows fuel to be supplied from both left- and right-hand fuel systems. When positioned to R, fuel is supplied from the right-hand fuel system only. OFF position closes both fuel shutoff valves. On some airplanes** the fuel selector switch is safety-wired to the ENGINE position and is placarded with operating instructions. Power is supplied from the 28-volt dc essential bus.

Note

- To preclude the possibility of a shutoff valve failing to open, the fuel selector switch should remain on ENGINE position during normal flight. The symmetrical fuel usage should be controlled by use of boost pump switches.
- Due to the proximity of the fuel selector switch to the ARC-34 command radio selector control, be certain that the desired control is selected prior to movement.

FUEL SHUTOFF VALVE SWITCHES

On some airplanes†, the left- and right-hand fuel shutoff valves are controlled by individual switches located at the bottom of the fuel control panel (figure 1-10). These switches, placarded "Fuel Valve," have OPEN and CLOSE positions and are guarded in the OPEN position. Each switch is individually operated to control the corresponding fuel shutoff valve. All fuel flow from a wing may be shut off by placing the appropriate switch in the CLOSE position. The fuel shutoff switches receive power from the 28-volt dc essential bus.

Note

To preclude the possibility of a shutoff valve failing to open, the fuel shutoff valve switches should remain at OPEN during normal flight. Symmetrical fuel usage should be controlled by use of boost pumps.

**Airplanes not modified by TCTO 1F-102-690.*

***In accordance with TCTO 1F-102-686.*

†*Airplanes modified by TCTO 1F-102-690.*

FUEL BOOST PUMP SWITCHES

Four two-position fuel boost pump switches (figure 1-10) are located on the fuel control panel to provide individual control of the fuel boost pumps. The pairs of switches are placarded "LH Boost Pumps" and "RH Boost Pumps" with ON and OFF positions which control the boost pumps accordingly. Individual switches are placarded "Fwd" and "Aft" to identify the specific pump within each fuel tank. On some airplanes*, fuel boost pumps switches are wired in series with the fuel shutoff valves, and the valves must be open for the pumps to operate. The control circuit receives power from the 28-volt dc essential bus.

EXTERNAL TANKS JETTISON BUTTON

The ring-guarded external tanks jettison button (figure 1-24) 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 pressed electrically fired ballistic charges unlock and separate the pylons and tanks from the wings. The jettison circuits receives power from the 28-volt dc essential bus.

CAUTION

The external tank jettison button should not be depressed while the landing gear is extended except during an emergency. The landing gear fairing doors will be damaged as the external tanks are released.

EXTERNAL TANK FUEL TRANSFER SWITCH

An external tank fuel transfer switch** (4, figure 1-4; figure 1-10) is installed on some airplanes to control fuel transfer from the external tanks. This switch is located on the left-hand side of the instrument panel and is placarded "Ext Tank Fuel Transfer" with positions ON and OFF. Placing the switch in the ON position opens a solenoid-operated valve in each external tank pressurization line. When fuel in the external tanks is expended pressurization to these tanks is shut off automatically. The switch is powered by the 28-volt dc nonessential bus.

Note

Refer to Section V for limitations on external tank use without the external tank fuel transfer switch installed.

EXTERNAL TANK JETTISON WARNING LIGHTS AND SAFETY PIN

Some airplanes** are equipped with an external tank ground safety switch and a ground safety switch warning light panel, both of which are located in the left side of the main wheel well. The ground safety switch is mounted on the side of the panel and consists of a receptacle into which a safety pin is inserted. The safety pin has a red streamer and should be installed during all ground operations. Two red, press-to-test lights are mounted on the panel, one placarded LH and the other RH. If either or both lights are illuminated, it is an indication that either or both tank jettison circuits are energized.

WARNING

Do not remove safety lock pin if red light is on.

Prior to removing the safety pin the lights should be press-tested for satisfactory operation. The warning lights are powered by the 28-volt dc essential bus.

FUEL QUANTITY GAGE SELECTOR SWITCH

The fuel quantity gage selector switch (figure 1-10), located on the fuel control panel, connects the fuel quantity gage to various tank quantity indicating circuits. On some airplanes, the switch is a three-position toggle switch with L, R, and TOTAL positions, permitting the indicator to read total internal fuel in both wings or total internal fuel in either wing. On some airplanes the L and R positions are replaced by #3L and #3R. Selecting these positions provides an indication of fuel quantity in the No. 3 fuel tanks. On still other airplanes†, the fuel quantity gage selector switch has a TOTAL position as well as a WING and #3 position for both left and right wings. Thus, it is possible to determine total internal fuel, total fuel in each wing, and total fuel in each No. 3 tank. There are no provisions for indicating fuel quantity in the external tanks. Power to the fuel quantity gage selector switch is supplied by the 28-volt dc essential bus.

FUEL QUANTITY GAGE AND GAGE TEST BUTTON

The fuel quantity gage (19, figure 1-4; figure 1-8) is located on the instrument panel and indicates internal fuel quantity in pounds. Each fuel tank is equipped with two tank probes to measure the quantity of fuel. The fuel quantity indicating system is a capacitance-type and compensates for changes in fuel density. The indicator dial reads from 0 to 8000 pounds and is marked in 200-pound increments. The fuel quantity gage indication depends upon the position of the fuel quantity gage selector switch.

Note

External tank fuel quantity is not indicated on the fuel quantity gage; therefore when external tanks are used, a decrease is not indicated until external fuel is exhausted.

**AF 56-1206 & on, & airplanes modified by TCTO 1F-102-651.*
***AF 53-1791, -1794, 56-1045 & on, & airplanes modified by TCTO 1F-102-677.*

†Airplanes modified by TCTO 1F-102-690.

FUEL BOOST PUMP SWITCHES

Four two-position fuel boost pump switches (figure 1-10) are located on the fuel control panel to provide individual control of the fuel boost pumps. The pairs of switches are placarded "LH Boost Pumps" and "RH Boost Pumps" with ON and OFF positions which control the boost pumps accordingly. Individual switches are placarded "Fwd" and "Aft" to identify the specific pump within each fuel tank. On some airplanes*, fuel boost pumps switches are wired in series with the fuel shutoff valves, and the valves must be open for the pumps to operate. The control circuit receives power from the 28-volt dc essential bus.

EXTERNAL TANKS JETTISON BUTTON

The ring-guarded external tanks jettison button (figure 1-24) 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 pressed electrically fired ballistic charges unlock and separate the pylons and tanks from the wings. The jettison circuits receives power from the 28-volt dc essential bus.

CAUTION

The external tank jettison button should not be depressed while the landing gear is extended except during an emergency. The landing gear fairing doors will be damaged as the external tanks are released.

EXTERNAL TANK FUEL TRANSFER SWITCH

An external tank fuel transfer switch** (4, figure 1-4; figure 1-10) is installed on some airplanes to control fuel transfer from the external tanks. This switch is located on the left-hand side of the instrument panel and is placarded "Ext Tank Fuel Transfer" with positions ON and OFF. Placing the switch in the ON position opens a solenoid-operated valve in each external tank pressurization line. When fuel in the external tanks is expended pressurization to these tanks is shut off automatically. The switch is powered by the 28-volt dc nonessential bus.

Note

Refer to Section V for limitations on external tank use without the external tank fuel transfer switch installed.

EXTERNAL TANK JETTISON WARNING LIGHTS AND SAFETY PIN

Some airplanes** are equipped with an external tank ground safety switch and a ground safety switch warning light panel, both of which are located in the left side of the main wheel well. The ground safety switch is mounted on the side of the panel and consists of a receptacle into which a safety pin is inserted. The safety pin has a red streamer and should be installed during all ground operations. Two red, press-to-test lights are mounted on the panel, one placarded LH and the other RH. If either or both lights are illuminated, it is an indication that either or both tank jettison circuits are energized.

WARNING

Do not remove safety lock pin if red light is on.

Prior to removing the safety pin the lights should be press-tested for satisfactory operation. The warning lights are powered by the 28-volt dc essential bus.

FUEL QUANTITY GAGE SELECTOR SWITCH

The fuel quantity gage selector switch (figure 1-10), located on the fuel control panel, connects the fuel quantity gage to various tank quantity indicating circuits. On some airplanes, the switch is a three-position toggle switch with L, R, and TOTAL positions, permitting the indicator to read total internal fuel in both wings or total internal fuel in either wing. On some airplanes the L and R positions are replaced by #3L and #3R. Selecting these positions provides an indication of fuel quantity in the No. 3 fuel tanks. On still other airplanes†, the fuel quantity gage selector switch has a TOTAL position as well as a WING and #3 position for both left and right wings. Thus, it is possible to determine total internal fuel, total fuel in each wing, and total fuel in each No. 3 tank. There are no provisions for indicating fuel quantity in the external tanks. Power to the fuel quantity gage selector switch is supplied by the 28-volt dc essential bus.

FUEL QUANTITY GAGE AND GAGE TEST BUTTON

The fuel quantity gage (19, figure 1-4; figure 1-8) is located on the instrument panel and indicates internal fuel quantity in pounds. Each fuel tank is equipped with two tank probes to measure the quantity of fuel. The fuel quantity indicating system is a capacitance-type and compensates for changes in fuel density. The indicator dial reads from 0 to 8000 pounds and is marked in 200-pound increments. The fuel quantity gage indication depends upon the position of the fuel quantity gage selector switch.

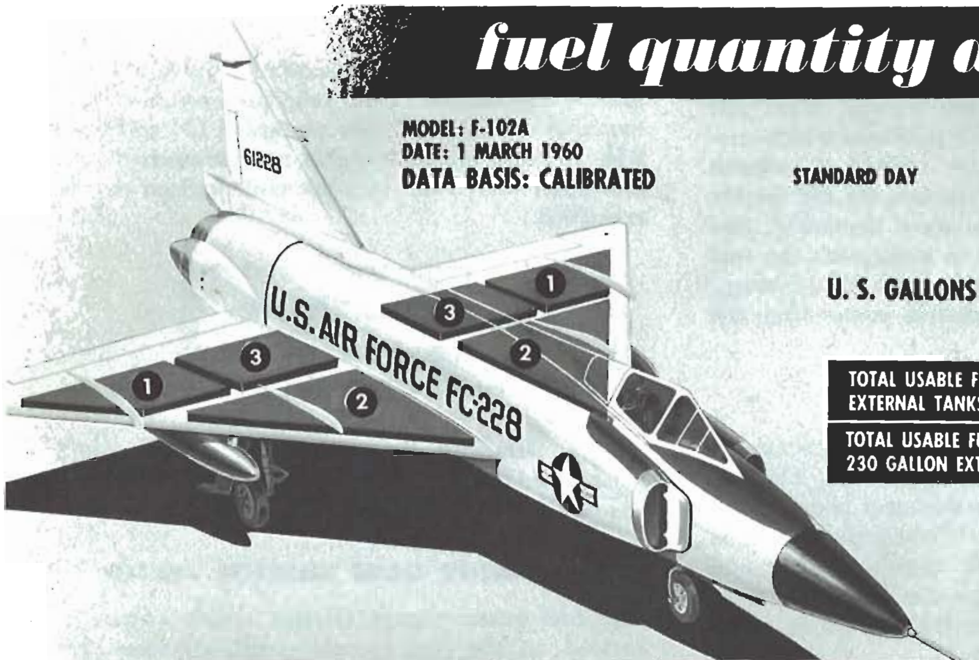
Note

External tank fuel quantity is not indicated on the fuel quantity gage; therefore when external tanks are used, a decrease is not indicated until external fuel is exhausted.

*AF 56-1206 & on, & airplanes modified by TCTO 1F-102-651.
**AF 53-1791, -1794, 56-1045 & on, & airplanes modified by TCTO 1F-102-677.

†Airplanes modified by TCTO 1F-102-690.

fuel quantity data table



MODEL: F-102A
DATE: 1 MARCH 1960
DATA BASIS: CALIBRATED

STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

U. S. GALLONS AND POUNDS

	POUNDS	GALLONS
TOTAL USABLE FUEL WITHOUT EXTERNAL TANKS	7053	1085
TOTAL USABLE FUEL WITH TWO 230 GALLON EXTERNAL TANKS	9848	1515

TANKS	NUMBER	USABLE FUEL		FULLY SERVICED	
		POUNDS	GALLONS	POUNDS	GALLONS
RIGHT WING	TANK NO. 1	920.0	141.5	3595.5	563
	TANK NO. 2	1631.5	251.0		
	TANK NO. 3	975.0	150.0		
LEFT WING	TANK NO. 1	920.0	141.5	3595.5	563
	TANK NO. 2	1631.5	251.0		
	TANK NO. 3	975.0	150.0		
EXTERNAL TANKS	LEFT HAND	1397.5	215.0	1417.0	218
	RIGHT HAND	1397.5	215.0	1417.0	218

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Figure 1-11

On some airplanes, gage operation can be checked by a test button (figure 1-10) placarded "Ind Test" on the fuel control 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. On other airplanes*, the only test of the indicator is to select the various positions of the selector switch and observe the indications. The indicating system receives power from the 115-volt essential bus and the 28-volt dc essential bus.

Airplanes modified by TCTO 1F-102-690.

FUEL QUANTITY-LOW WARNING LIGHTS

Two fuel quantity-low warning lights (9 and 10, figure 1-26), 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 88 gallons (570 pounds). The fuel quantity-low warning circuits receive power from the 28-volt dc essential bus.

FUEL BOOST PRESSURE-LOW WARNING LIGHTS

Two fuel boost pressure-low warning lights (13 & 14, figure 1-26), 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 or on some airplanes* if a fuel shutoff valve is not in fully open position. The appropriate light will remain illuminated until the boost pump pressure exceeds 12 psi if the illumination was due to boost pump failure. The warning light will not illuminate if only one boost pump fails. The fuel boost pressure-low warning circuits receive power from the 28-volt dc essential bus.

FUEL TANK PRESSURE-LOW WARNING LIGHTS

Two fuel tank pressure-low warning lights (11 & 12 figure 1-26), 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 0.5 psi. The appropriate light will remain illuminated until tank pressurization exceeds 1.5 psi. The fuel tank pressure-low warning circuits receive power from the 28-volt dc essential bus.

Note

Fuel tank pressure-low warning lights may illuminate temporarily during rapid descent from high altitude or negative g maneuvers. The lights should extinguish immediately after resuming level flight.

ELECTRICAL POWER SUPPLY SYSTEM

The airplane is equipped with direct-current and alternating current electrical power systems. The dc system is powered by an engine-driven generator and a battery. The ac system is powered by a three-phase main ac generator driven by an engine-driven, constant-speed drive unit. Two ac transformers are used to reduce the ac voltages. A hydraulically driven emergency ac generator serves as a standby power source. All electrical power can be shut off by use of the master switch, located on the electrical control panel. Both dc and ac power systems can be connected to an external power source for ground operations through external power receptacles.

DC ELECTRICAL POWER DISTRIBUTION

The 28-volt dc power supply system (figure 1-12) is powered by an engine-driven, 200-ampere generator and a 24 ampere-hour battery. Direct current is distributed through an essential and nonessential bus and, on some airplanes**, an emergency bus. The dc generator and battery are connected to the essential bus which in turn is connected to the nonessential bus by the nonessential bus tie relay. In the event of dc generator failure the nonessential bus is disconnected from the battery by the bus tie relay. The battery then becomes the source of power for the essential bus. The emergency dc bus normally receives power from the dc essential bus through an emergency dc bus changeover relay. This relay is energized by power from the dc nonessential bus. In the event of dc generator failure, the changeover relay automatically connects the emergency bus to a transformer-rectifier bus powered by the ac essential bus (see figure 1-13), thereby reducing the load on the battery. If ac power also fails, the emergency dc bus can be reconnected to the dc essential bus by operation of the battery switch to the TR FAIL position. The only power indication for the dc system is the dc power failure warning light.

AC ELECTRICAL POWER DISTRIBUTION

The ac power supply system (figure 1-13) is powered by a constant-speed, 30-kva generator. Alternating current is distributed through four bus networks, and by the use of two transformers, supplies three separate voltage systems. These systems consist of 200/115, 3-phase, 400-cycle essential and nonessential busses; a 115-volt, 3-phase 400-cycle essential bus; and a 26-volt, single phase, 400-cycle essential bus. In addition, some airplanes** have a transformer-rectifier connected to the 115-volt, 3-phase essential bus to provide emergency 28-volt dc power to the ac control circuits and for certain other systems (see figure 1-12). In the event of main ac generator failure selecting the emergency generator disconnects the nonessential bus and energizes and connects the hydraulically driven emergency ac generator to the essential buses. There is no provision for automatic switchover from the normal to the emergency ac generator.

ELECTRICALLY OPERATED EQUIPMENT

See figures 1-12 and 1-13 for complete reference to electrically operated equipment.

EXTERNAL POWER RECEPTACLES

The dc and ac power systems can be connected to an external power source for ground operations through main external power receptacles located on the forward side of the left main wheel well. These receptacles are protected by a spring-loaded dust cover.

Note

The dc essential bus must be energized to actuate the ac external power relay when external ac power is connected to the airplane unless the power cart contains a dc energizing circuit for the relay.

A dc generator test receptacle is located above the main external power receptacles for maintenance purposes. An auxiliary dc power receptacle is located on the aft side of the aft electronics bay to facilitate landing gear maintenance retraction tests. An outlet test receptacle is located on the nose wheel well switch panel, for electronics test purposes.

AF 56-1206 & on, & airplanes modified by TCTO 1F-102-651.

***Airplanes modified by TCTO 1F-102-727.*

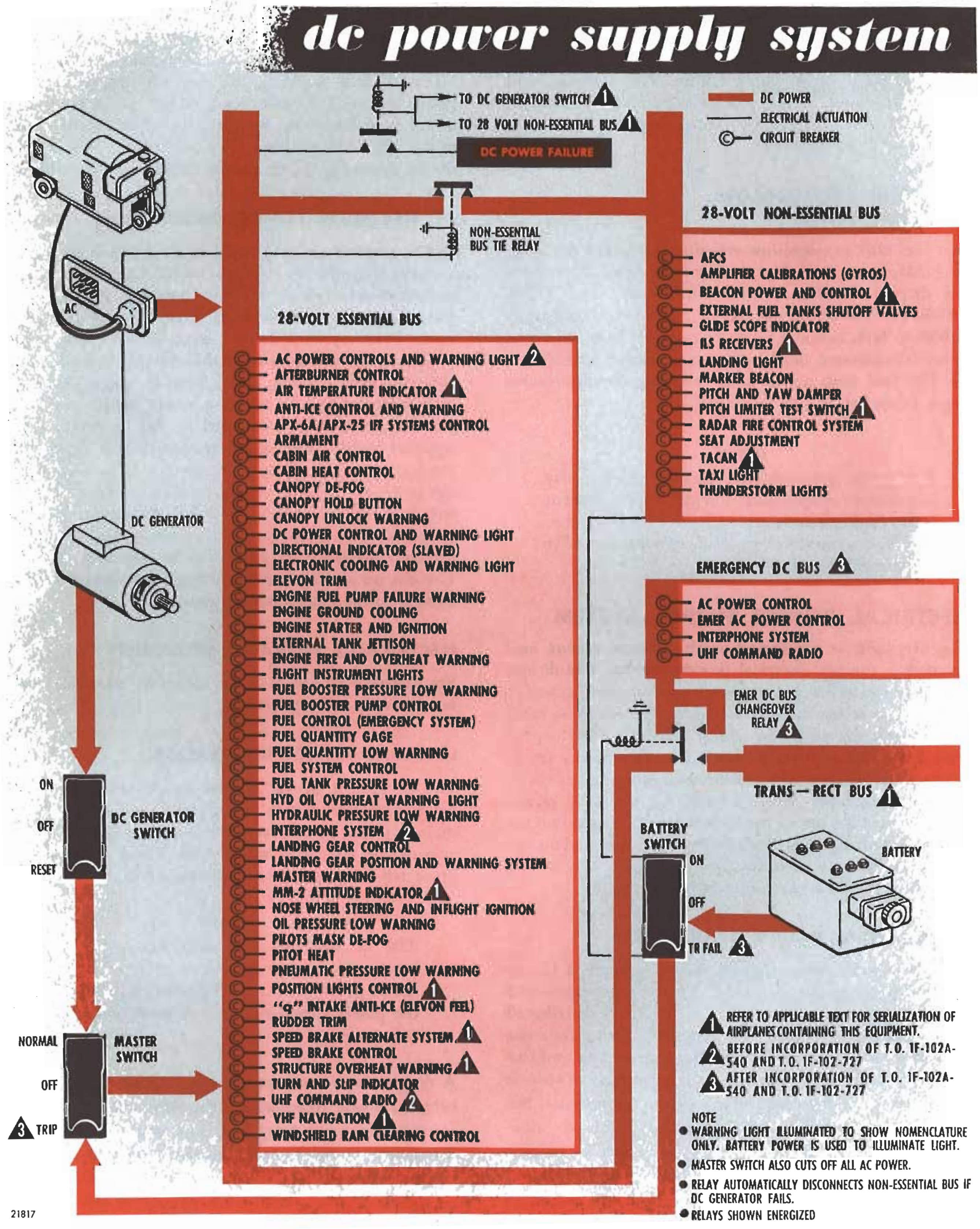
dc power supply system
TO DC GENERATOR SWITCH 1
TO 28 VOLT NON-ESSENTIAL BUS 1
DC POWER FAILURE
DC POWER
ELECTRICAL ACTUATION
CIRCUIT BREAKER
NON-ESSENTIAL BUS TIE RELAY
28-VOLT NON-ESSENTIAL BUS
AFCS
AMPLIFIER CALIBRATIONS (GYROS)
BEACON POWER AND CONTROL 1
EXTERNAL FUEL TANKS SHUTOFF VALVES
GLIDE SCOPE INDICATOR
ILS RECEIVERS 1
LANDING LIGHT
MARKER BEACON
PITCH AND YAW DAMPER
PITCH LIMITER TEST SWITCH 1
RADAR FIRE CONTROL SYSTEM
SEAT ADJUSTMENT
TACAN 1
TAXI LIGHT
THUNDERSTORM LIGHTS
AC
28-VOLT ESSENTIAL BUS
AC POWER CONTROLS AND WARNING LIGHT 2
AFTERBURNER CONTROL
AIR TEMPERATURE INDICATOR 1
ANTI-ICE CONTROL AND WARNING
APX-6A/APX-25 IFF SYSTEMS CONTROL
ARMAMENT
CABIN AIR CONTROL
CABIN HEAT CONTROL
CANOPY DE-FOG
CANOPY HOLD BUTTON
CANOPY UNLOCK WARNING
DC POWER CONTROL AND WARNING LIGHT
DIRECTIONAL INDICATOR (SLAVED)
ELECTRONIC COOLING AND WARNING LIGHT
ELEVON TRIM
ENGINE FUEL PUMP FAILURE WARNING
ENGINE GROUND COOLING
ENGINE STARTER AND IGNITION
EXTERNAL TANK JETTISON
ENGINE FIRE AND OVERHEAT WARNING
FLIGHT INSTRUMENT LIGHTS
FUEL BOOSTER PRESSURE LOW WARNING
FUEL BOOSTER PUMP CONTROL
FUEL CONTROL (EMERGENCY SYSTEM)
FUEL QUANTITY GAGE
FUEL QUANTITY LOW WARNING
FUEL SYSTEM CONTROL
FUEL TANK PRESSURE LOW WARNING
HYD OIL OVERHEAT WARNING LIGHT
HYDRAULIC PRESSURE LOW WARNING
INTERPHONE SYSTEM 2
LANDING GEAR CONTROL
LANDING GEAR POSITION AND WARNING SYSTEM
MASTER WARNING
MM-2 ATTITUDE INDICATOR 1
NOSE WHEEL STEERING AND INFLIGHT IGNITION
OIL PRESSURE LOW WARNING
PILOTS MASK DE-FOG
PITOT HEAT
PNEUMATIC PRESSURE LOW WARNING
POSITION LIGHTS CONTROL 1
"q" INTAKE ANTI-ICE (ELEVON FEEL)
RUDDER TRIM
SPEED BRAKE ALTERNATE SYSTEM 1
SPEED BRAKE CONTROL
STRUCTURE OVERHEAT WARNING 1
TURN AND SLIP INDICATOR
UHF COMMAND RADIO 2
VHF NAVIGATION 1
WINDSHIELD RAIN CLEARING CONTROL
DC GENERATOR
EMERGENCY DC BUS 3
AC POWER CONTROL
EMER AC POWER CONTROL
INTERPHONE SYSTEM
UHF COMMAND RADIO
EMER DC BUS CHANGEOVER RELAY 3
TRANS — RECT BUS 1
ON
OFF
RESET
DC GENERATOR SWITCH
BATTERY SWITCH
ON
OFF
TR FAIL 3
BATTERY
NORMAL
OFF
3 TRIP
MASTER SWITCH
1 REFER TO APPLICABLE TEXT FOR SERIALIZATION OF AIRPLANES CONTAINING THIS EQUIPMENT.
2 BEFORE INCORPORATION OF T.O. 1F-102A-540 AND T.O. 1F-102-727
3 AFTER INCORPORATION OF T.O. 1F-102A-540 AND T.O. 1F-102-727
NOTE
• WARNING LIGHT ILLUMINATED TO SHOW NOMENCLATURE ONLY. BATTERY POWER IS USED TO ILLUMINATE LIGHT.
• MASTER SWITCH ALSO CUTS OFF ALL AC POWER.
• RELAY AUTOMATICALLY DISCONNECTS NON-ESSENTIAL BUS IF DC GENERATOR FAILS.
• RELAYS SHOWN ENERGIZED
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Figure 1-12

CIRCUIT BREAKERS AND FUSES

Individual circuit protection is provided by thermal-type circuit breakers throughout the airplane, with the exception of the pitot heat control circuit. These circuit breakers are called "trip-free" and manually holding the circuit breaker in the depressed position will not complete the circuit if it remains faulty or overloaded, thereby reducing fire hazard. The pitot heat control circuit is provided with a switch-type circuit breaker which can be held in the ON position, in case of emergency. The circuit breakers which are accessible to the pilot are on the circuit breaker panels (figure 1-15) located at the aft end of the left and right consoles and at the forward side of the left console. Fuses are used in the boost pump motor circuit, and spare fuses for this circuit are located adjacent to the boost pump relays on the aft side of each main wheel well.

MASTER SWITCH

The guarded master switch (figure 1-14), located on the electrical power control panel, is used to shut off all generator and battery power during emergency conditions. The switch is placarded "Master" and has NORMAL and OFF positions and, on some airplanes*, a TRIP position. In the NORMAL position the switch connects the battery and the generators to the airplane buses if the battery and generator switches are ON. The TRIP position is a momentary position spring-loaded to OFF, and provides battery power for tripping the generator control panels before dropping out the battery relay. The master switch receives power for disconnecting the generators from the ac generator exciter circuit on some airplanes, and from the dc essential bus on other airplanes*.

CAUTION

The master switch should be used in cases of emergency only. When in OFF position, generator disconnect relays are energized (on some airplanes**) which would drain the battery if left in this position.

BATTERY SWITCH

The guarded battery switch (figure 1-14), located on the electrical power control panel, is used to disconnect the battery from the airplane electrical system. The switch is placarded "Bat" and has ON and OFF positions, which control the circuit accordingly. On some airplanes*, the battery switch also has a TR FAIL position which energizes the emergency dc bus changeover relay directly from the battery. The TR FAIL position is used to connect the emergency dc bus to the dc essential bus (battery) when the dc generator and the transformer-rectifier have failed or when the ac, dc, and emergency ac generators have failed.

**Airplanes modified by TCTO 1F-102-727.*
***AF 53-1791 thru -1818.*

CAUTION

Placing the battery switch in the TR FAIL position at any other time may result in complete electrical power failure after battery power is depleted.

The battery control circuit receives power directly from the battery.

DC GENERATOR SWITCH

The guarded dc generator switch (figure 1-14), located on the electrical power control panel, is used to control dc generator operation. The switch is placarded "DC Gen" and has three positions, ON, OFF, and RESET. When the generator switch is ON, generator output is supplied to the dc electrical system. Placing the generator switch to OFF disconnects the generator from the essential bus which causes the nonessential bus tie relay to disconnect the nonessential bus from the battery. If a malfunction cuts out the generator or the master switch has been placed to OFF, then to NORMAL, the generator switch should be held momentarily at RESET then returned to ON to restore normal generator operation. The dc generator control circuit receives power from the 28-volt dc essential bus.

AC GENERATOR SWITCH

The guarded ac generator switch (figure 1-14), located on the electrical power control panel, is used to control ac generator operation. The switch is placarded "AC Gen" and has three positions, ON, OFF, and RESET. When the generator switch is ON, generator output is supplied to the ac electrical system. In the event of main ac generator failure, the ac generator switch should be turned to OFF to preclude the possibility of a fire hazard from a faulty generator. The emergency ac generator will energize directly from the ac bus switch. Placing the switch to OFF disconnects the generator from the buses. If a malfunction cuts out the generator or the master switch has been placed to OFF, then to NORMAL, the generator switch should be held momentarily at RESET, then returned to ON to restore normal generator operation. The ac generator control circuit receives power from the 28-volt dc essential bus or, on some airplanes*, from the emergency dc bus.

AC BUS SWITCH

The guarded ac bus switch (figure 1-14), located on the electrical power control panel, is used to energize the hydraulically driven emergency ac generator. The switch is placarded "AC Bus." On some airplanes the switch has two positions, NOR and EMER. Placing the switch to NOR connects the main ac generator to the buses, and

**Airplanes modified by TCTO 1F-102-727.*

ac power supply system

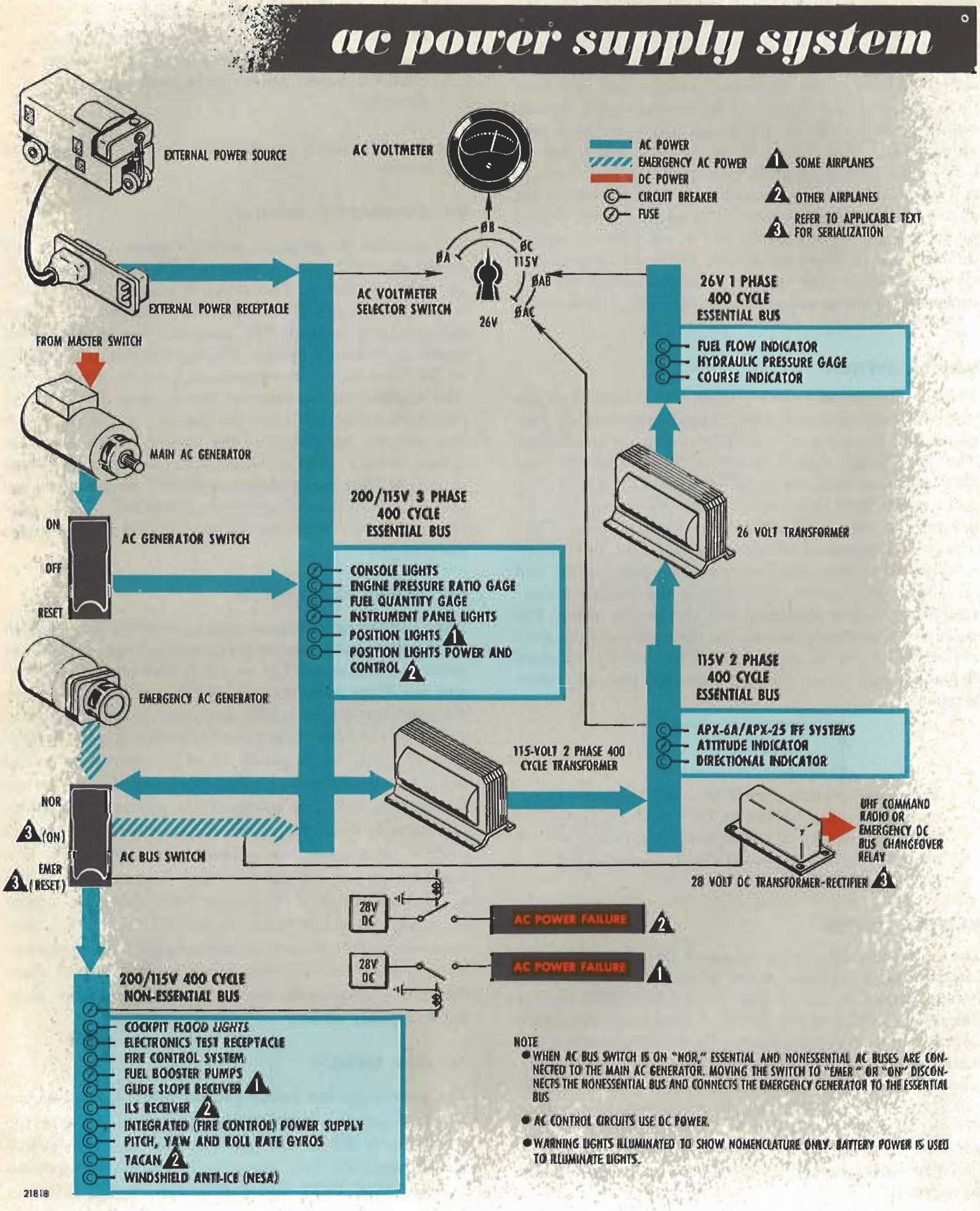

Figure 1-13

when placed to EMER the emergency ac generator is energized, connected to the essential bus and the main ac generator and nonessential bus is disconnected from the essential bus. On these airplanes, control circuit power is supplied from the 28-volt dc essential bus through the main ac generator switch. On other airplanes*, the ac bus switch is placarded "Emer Gen" and has three positions, NOR, ON, and RESET. In the NOR position, the main ac generator is connected to the ac buses. Actuation of the switch to the spring-loaded RESET position completes a circuit to the ac emergency disconnect relay and the emergency generator shutoff valve. The completed circuit starts the emergency ac generator, ties it to the ac essential bus, and disconnects the main ac generator. When released, the ac bus switch will return to the ON position which ties the emergency disconnect relay and emergency shutoff valve to the emergency dc bus. On these airplanes, the ac bus switch receives power through the emergency dc bus and, in the RESET position, directly from the dc essential bus.

AC VOLTMETER AND SELECTOR SWITCH

The ac voltmeter (figure 1-14), located above the electrical power control panel, provides a means of determining ac generator output as selected by the voltmeter selector switch. The six-position voltmeter selector switch (figure 1-14), located on the electrical power control panel, provides a means of connecting the voltmeter to the essential ac bus circuits as follows: Phases A, B, and C of the main ac generator output (approximately 115 volts); phase AB and AC of the 115-volt transformer; and the single phase output of the 26-volt transformer.

DC POWER FAILURE WARNING LIGHT

The dc power failure warning light (3, figure 1-26), located on the warning light panel, illuminates and displays "DC POWER FAILURE" if the dc generator or the dc power disconnect relay fails. On some airplanes** the warning light will come on when the battery switch is ON or when the external power unit is connected for starting and will remain on until external power is disconnected after the engine is running. On other airplanes the light will remain out when power (external or airplane power) is applied to the buses. The dc power failure warning circuit receives power from the 28-volt dc essential bus (battery power).

Note

DC power is used to operate the electrical control panel which contains the master switch. In the event of both dc generator and battery failure, this panel will become inoperative on on some airplanes and ac power will not be available. On other airplanes*, dc power will be provided by the transformer-rectifier for the control of the ac system.

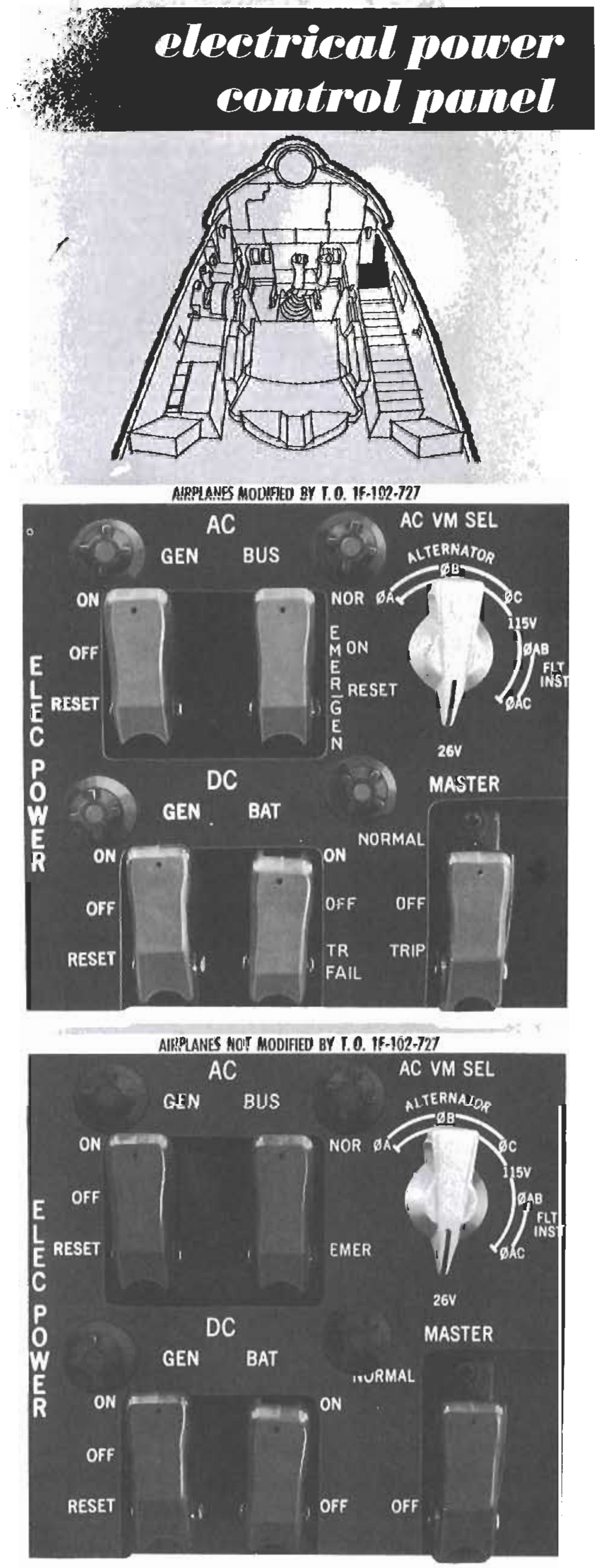

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Figure 1-14

*Airplanes modified by TCTO 1F-102-727.
**AF 56-1275 thru -1316, -1332 & on.

circuit breaker panels (typical)

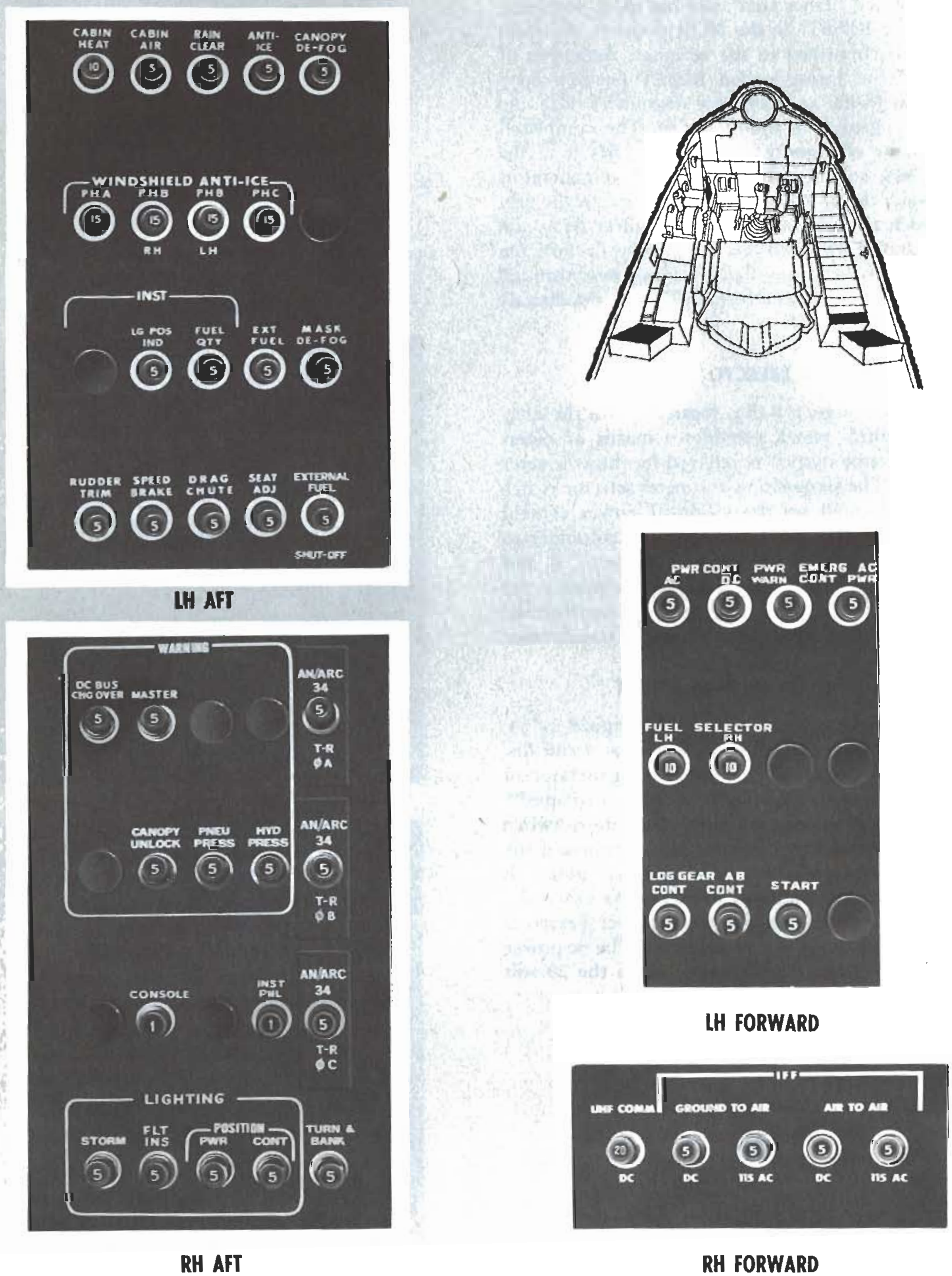


Figure 1-15

AC POWER FAILURE WARNING LIGHT

The ac power failure warning light (2, figure 1-26), located on the warning light panel, illuminates and displays "AC POWER FAILURE" if the main ac generator or the ac power disconnect relay fails. On some airplanes* the warning light will come on when the battery switch is ON or the external power unit is connected for starting and will remain on until external power is disconnected after the engine is running. On other airplanes the light will remain out when power (external or airplane power) is applied to the buses. The ac power failure warning circuit receives power from the 28-volt dc essential bus.

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 major operating components of the airplane. 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 event of an emergency. Pressure in either of these systems can be read on a single hydraulic pressure gage by the use of a hydraulic pressure gage selector switch. A flashing red light is provided to warn of a single system failure, or steady illumination of the same light indicates failure of both systems. On some airplanes**, a hydraulic fluid overheat warning light is also provided to warn of overheated hyraulic fluid in either the primary or secondary hydraulic systems. Illumination of the light will not differentiate between the primary and secondary hydraulic systems. See figure 1-33 for hydraulic fluid specification.

PRIMARY HYDRAULIC SYSTEM

The primary hydraulic system supplies power for operation of the flight controls only. The system is completely independent of the secondary system and consists primarily of a reservoir, a 3000 psi variable volume engine-driven pump, accumulator, supply lines, and thermal-pressure relief valves for system protection. On some airplanes† hydraulic fluid is cooled by heat exchanger coils in the right No. 3 fuel tank. The system contains conventional filters with bypass features. The reservoir has a capacity of 252 cu. in. or 1.09 gallons and is pressurized by the low-pressure pneumatic system. The piston-type accumulator contains a pressure gage (not accessible in flight) for ground checking the preload pressure.

SECONDARY HYDRAULIC SYSTEM

The secondary hydraulic system supplies power for operation of the flight controls (parallels the primary system action). The system also supplies power for operation of the landing gear and doors, nose wheel steering, speed brakes and the emergency ac generator. The secondary hydraulic system consists primarily of a reservoir, a 3000 psi variable volume engine-driven pump, accumulator, supply lines, and thermal-pressure relief valves for system protection. On some airplanes† hydraulic fluid is cooled by heat exchanger coils in the left No. 3 fuel tank. The system contains conventional filters with bypass features. The reservoir has a capacity of 420 cu. in. or 1.82 gallons and is pressurized by the low-pressure pneumatic system. The piston-type accumulator contains a pressure gage (not accessible in flight) for ground checking the preload pressure.

**AF 56-1275 thru -1316, -1332 & on.*
***Airplanes modified by TCTO 1F-102-847.*
†Airplanes modifed by TCTO 1F-102-778.

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 pulling a 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. When only the primary hydraulic system has failed, the RAT should be extended just after turn on final approach as a precautionary measure. Once extended the RAT cannot be retracted in flight. This emergency system will supply sufficient power for limited maneuvering and a safe approach and landing. A RAT door test hook is installed on some airplanes 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.

RAT (HYDRAULIC EMERGENCY POWER) HANDLE

The RAT handle (figure 1-7), located in the outboard side of the throttle quadrant, is used to energize the emergency hydraulic system. Pulling the handle (approximately two inches) mechanically selects the pneumatic pressure to extend the RAT into the airstream. To insure satisfactory extension, the handle must be extended to full travel and held for a minimum of four seconds.

HYDRAULIC PRESSURE GAGE AND SELECTOR SWITCH

The hydraulic pressure gage (20, figure 1-4; figure 1-8) is located on the instrument panel (on early airplanes this gage is located on the right-hand console) and indicates either the primary or secondary hydraulic system pressure in increments from 0 to 4000 psi. The indicator receives system pressure information from either the

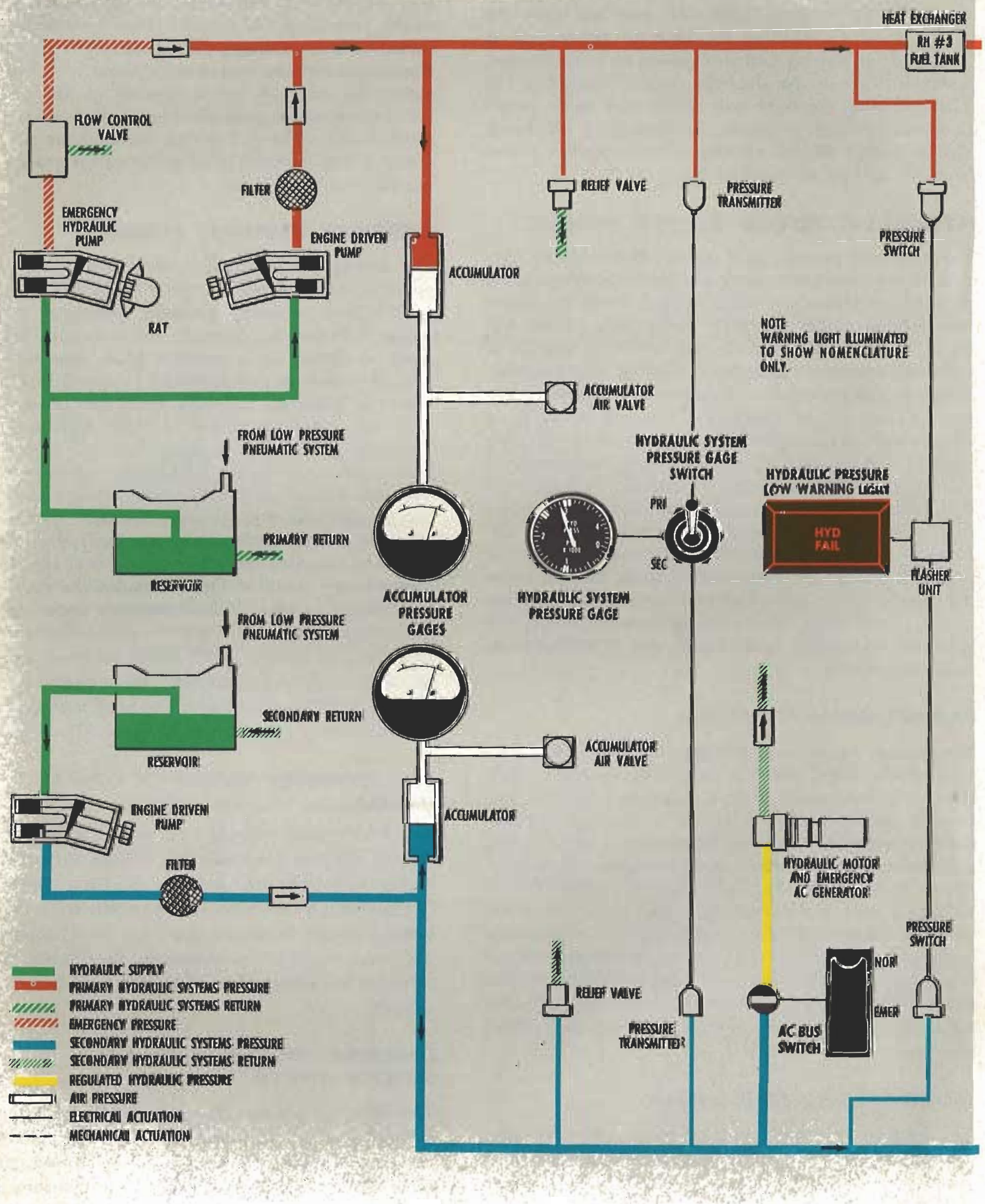
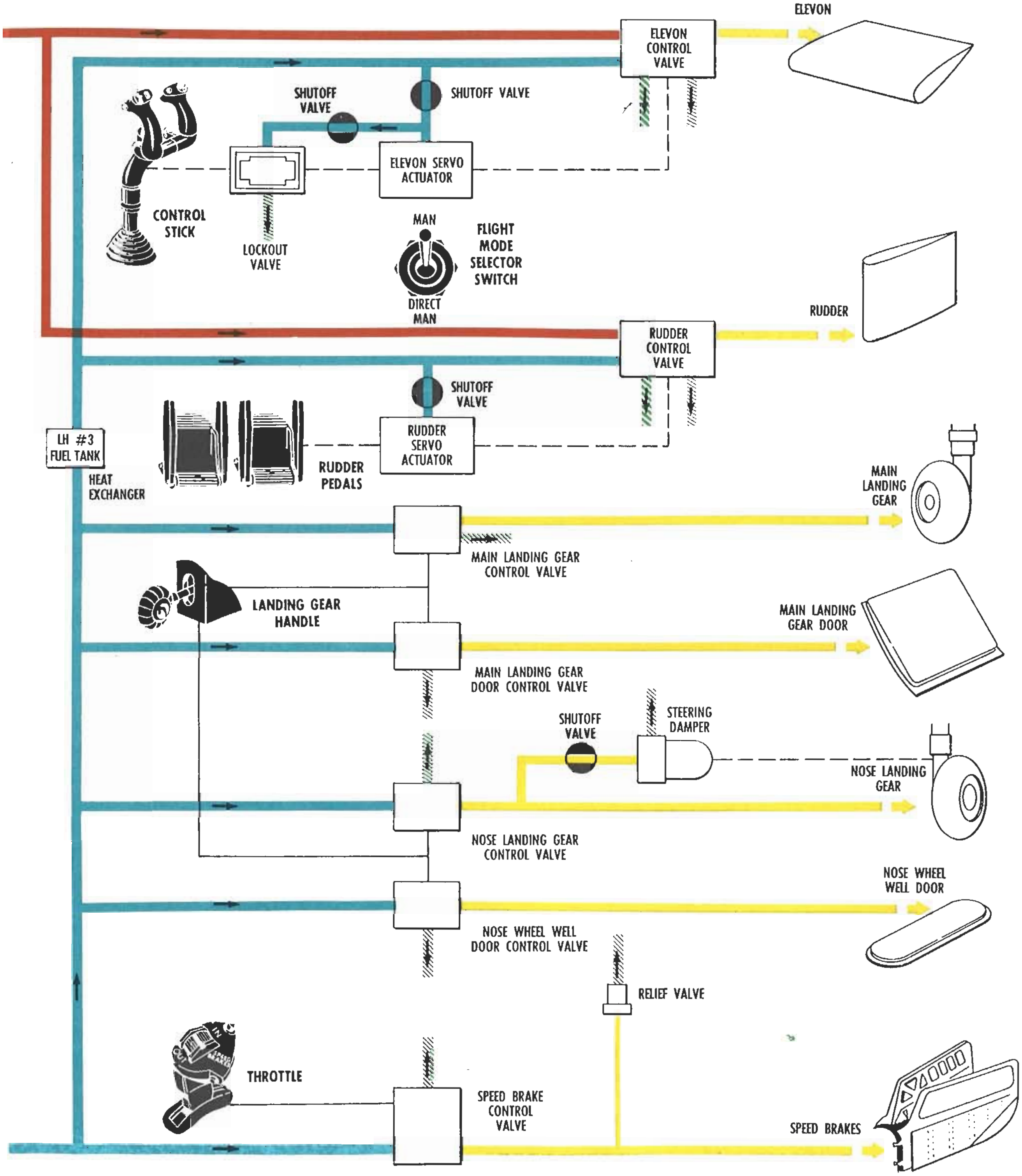

HEAT EXCHANGER
RH #3 FUEL TANK
FLOW CONTROL VALVE
FILTER
EMERGENCY HYDRAULIC PUMP
ENGINE DRIVEN PUMP
RAT
ACCUMULATOR
RELIEF VALVE
PRESSURE TRANSMITTER
PRESSURE SWITCH
NOTE
WARNING LIGHT ILLUMINATED TO SHOW NOMENCLATURE ONLY.
ACCUMULATOR AIR VALVE
FROM LOW PRESSURE PNEUMATIC SYSTEM
HYDRAULIC SYSTEM PRESSURE GAGE SWITCH
PRI
SEC
HYDRAULIC PRESSURE LOW WARNING LIGHT
HYD FAIL
PRIMARY RETURN
RESERVOIR
FLASHER UNIT
ACCUMULATOR PRESSURE GAGES
HYDRAULIC SYSTEM PRESSURE GAGE
FROM LOW PRESSURE PNEUMATIC SYSTEM
SECONDARY RETURN
RESERVOIR
ACCUMULATOR AIR VALVE
ENGINE DRIVEN PUMP
ACCUMULATOR
FILTER
HYDRAULIC MOTOR AND EMERGENCY AC GENERATOR
PRESSURE SWITCH
NOR
EMER
RELIEF VALVE
PRESSURE TRANSMITTER
AC BUS SWITCH
HYDRAULIC SUPPLY
PRIMARY HYDRAULIC SYSTEMS PRESSURE
PRIMARY HYDRAULIC SYSTEMS RETURN
EMERGENCY PRESSURE
SECONDARY HYDRAULIC SYSTEMS PRESSURE
SECONDARY HYDRAULIC SYSTEMS RETURN
REGULATED HYDRAULIC PRESSURE
AIR PRESSURE
ELECTRICAL ACTUATION
MECHANICAL ACTUATION
23B21-11

Figure 1-16

hydraulic power supply system



21821-2

primary or secondary transmitter, depending upon the position of the selector switch adjacent to the indicator. This two-position hydraulic pressure gage selector switch enables the pilot to select either PRI or SEC systems. The indicating and selection systems are powered by the 26-volt ac essential bus.

HYDRAULIC PRESSURE-LOW WARNING LIGHT AND TEST-RESET SWITCH

The red hydraulic pressure-low warning light (21, figure 1-4; figure 1-8), located on the instrument panel, illuminates (flashing) and displays "HYD FAIL" if either the primary or secondary hydraulic system pressure is below approximately 800 psi. Pressure loss of 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 both systems are in excess of approximately 1000 psi the light will go off. The three-position hydraulic pressure-low warning light test and reset switch (figure 1-8) has TEST and RESET positions and is spring-loaded to the center (OFF) position. If either hydraulic system has failed and caused a flashing light, placing the switch to RESET will turn the light off. RESET position will not turn the light off if both hydraulic systems have failed. Placing the switch to TEST position will illuminate the light in a flashing condition if the pressure in both systems is above approximately 1000 psi. The warning light is automatically dimmed when the instrument panel lights are on if the thunderstorm lights are off. Power is supplied by the 28-volt dc essential bus.

HYDRAULIC FLUID OVERHEAT WARNING LIGHT

The hydraulic fluid overheat warning light* (1, figure 1-26) located on the warning light panel, illuminates and displays "HYD OIL HOT" when hydraulic fluid in either the primary or secondary hydraulic system reaches a temperature of 225°(−5 +10°)F. The hydraulic fluid overheat warning light receives power from the dc essential bus.

Note

A hydraulic system overheat warning indication will not differentiate between the primary and secondary hydraulic systems.

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.

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. This engine bleed air is limited to 5.5% of total engine airflow but is normally much less than 1%. The bleed air varies in temperature and pressure up to approximately 800°F and 225 psi under extreme operational and climatic conditions. A portion of this air is passed through a refrigeration unit that includes an air-to-air heat exchanger and 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 suit; to supply operating pressure to the variable air pressure regulator in the elevator feel system; and to supply warm air for anti-icing, rain-clearing and defogging.

HIGH-PRESSURE PNEUMATIC SYSTEM

The high-pressure pneumatic system (figure 1-17) supplies pneumatic pressure for actuation of various system components. Air under pressure is stored in four spherical fiberglas flasks and in the main landing gear drag braces. The air storage flasks are mounted in the left and right armament bays. The system is serviced from an external source through a filler valve located on the forward side of the right main wheel well on some airplanes and the left main wheel well on other airplanes. System pressure is indicated on a pressure gage, located adjacent to the filler valve, and when fully serviced the system contains 3000 psi of compressed air. Pressure regulators and relief valves protect the system from excessive pressure. The pressure in the main landing gear drag braces is isolated by check valves to ensure adequate brake system pressure if the high-pressure pneumatic supply is depleted in flight. On some airplanes* a check valve isolates the priority air flask to provide full operating pressure for emergency landing gear extension, ram air turbine extension, and drag chute deployment. On other airplanes** a priority valve is installed between the priority and nonpriority air flasks in lieu of the check valve. The priority valve enables priority air to augment nonpriority air in the operation of all system components until pressure drops below 1400 psi in the priority air flask. Below 1400 psi, pressure in the priority flask is available for only ram air turbine extension, drag chute deployment, and landing gear extension. The pressure gage, located adjacent to the filler valve, will not indicate pressure through the entire system unless the system is fully charged. The pressure indicated at lower pressures in various air flasks will be dependent upon the various system configurations (see figure 1-17). For servicing requirements, see Servicing Diagram, figure 1-33.

PNEUMATICALLY OPERATED EQUIPMENT

See figures 4-1 and 1-17 for complete reference to pneumatically operated equipment.

**Airplanes modified by TCTO 1F-102-847.*

**AF 53-1791 thru 56-1518, unless modified by TCTO 1F-102-655.*
***AF 57-770 & on, & airplanes modified by TCTO 1F-102-655.*

PNEUMATIC PRESSURE-LOW WARNING LIGHT

The pneumatic pressure-low warning light (6, figure 1-26), located on the warning light panel, illuminates to display "PNEU PRESS." Operation of the warning light is dependent upon configuration of the high-pressure pneumatic system (see figure 1-17). On all airplanes, illumination of the warning light will indicate that pneumatic pressure in the priority air flask has dropped below 1500 psi. On those airplanes using a check valve to isolate the priority air flask*, at the moment of illumination of the warning light, enough high-pressure air remains in the priority flask for emergency landing gear extension, RAT extension, and drag chute deployment, and that pressure in the nonpriority flasks is less than 1500 psi. The check valve is designated to maintain full priority flask pressure and as long as there is 1500 psi, or more, in the nonpriority flasks, there will be no illumination of the warning light. When there is 1500 psi or more pressure in the priority flask, there will be no illumination of the warning light regardless of the pressure in the nonpriority flasks, and there is, therefore, no direct indication of the condition of the nonpriority air pressure. On other airplanes**, which use the priority valve to isolate the priority air flask, an equalizing flow between the priority and nonpriority flasks is permitted until pressure in the priority flask drops to 1400 psi, indicating that at the moment of warning light illumination, there is enough pressure in the priority flask for emergency operation of the landing gear, extension of the RAT, and deployment of the drag chute, and that 1400 psi remains in the system. When the light does not illuminate, in these airplanes, it will be an indication of satisfactory pressure in both the priority and the nonpriority flasks.

FLIGHT CONTROL SYSTEM

The flight control system (figures 1-18 and 1-19) provides desirable control of the airplane throughout the speed range. The delta wing configuration utilizes elevons instead of aileron and elevator control surfaces. The system incorporates standard stick and rudder pedals with conventional control action in response to stick movement. The elevons when moved coincidentally act as elevators and differentially as ailerons. This is accomplished by a mixer assembly which consists of a bellcrank assembly which rotates to provide aileron action and moves fore and aft to provide elevator action. To induce both aileron and elevator force, the elevons are large and extend almost the full span of the wing. Each elevon consists of an inboard and outboard panel which function as one unit permitting free surface movement unimpaired by normal in-flight wing deflections. Pitch and yaw damper systems are installed to provide a stable platform for armament firing and to stabilize high speed flight. Turn coordination is also furnished through the yaw damper system if the yaw damper is engaged. Turn coordination is supplied primarily for fire control system attacks and is sufficient for bank angles up to 35° and roll rates up to about 30° per second. Both the elevons and rudder are actuated by two complete, independent, simultaneously operating hydraulic systems. The ram air turbine can also supply pressure to one of these systems in event of emergencies. Movement of the stick and rudder pedals mechanically position 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 opposed 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 surface and is indicated by the no load cockpit control position. The design of the flight control system eliminates the necessity for surface gust locks except during storm conditions.

AF 53-1791 thru 56-1518, unless modified by TCTO 1F-102-655.
**AF 57-770 & on, & airplanes modified by TCTO 1F-102-655.*

ARTIFICIAL FEEL SYSTEM

As the hydraulic portion of the control system is irreversible, the pilot does not have an indication of existing aerodynamic forces 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. At approximately ¾ aileron deflection (5°), on airplanes which have the enlarged vertical fin, an additional 10 pounds resistance to aileron movement is imposed by added spring tension and must be overcome to obtain full aileron. 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 an elevator intelligence unit which loads the correct amount of ram air pressure into the elevator feel force cylinder. The control surfaces are most effective in the transonic speed range. Therefore, the feel force cylinder pressure is highest in this area. The feel system applies a force which results in approximately a constant stick force per g throughout the flight envelope. The elevator intelligence unit is operated by low-pressure pneumatic system pressure and controls the ram air pressure from the large impact tube on the vertical fin. 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 high-pressure pneumatic system pressure. This pressure is metered by a variable air pressure regulator controlled by ram air from the small

high-pressure pneumatic

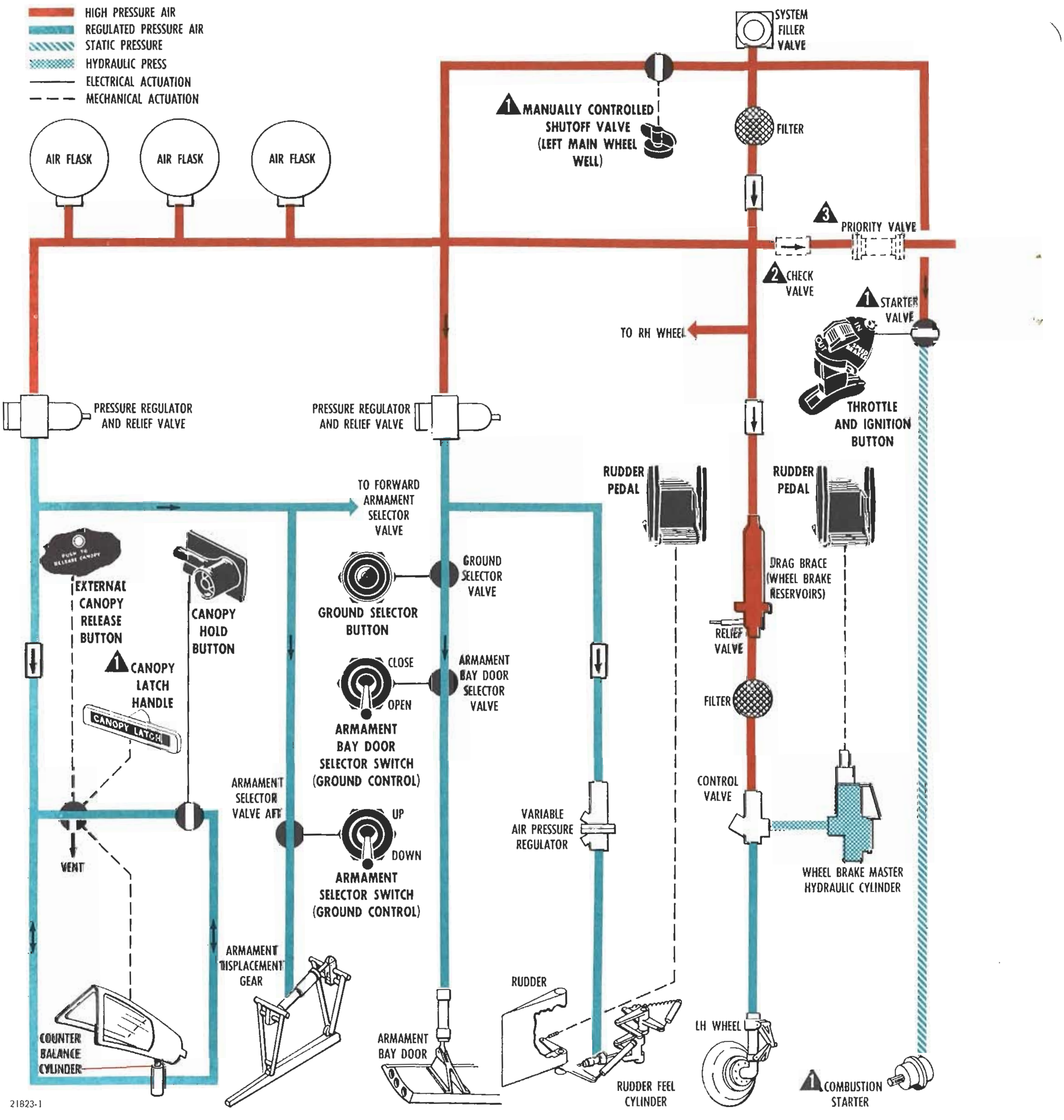
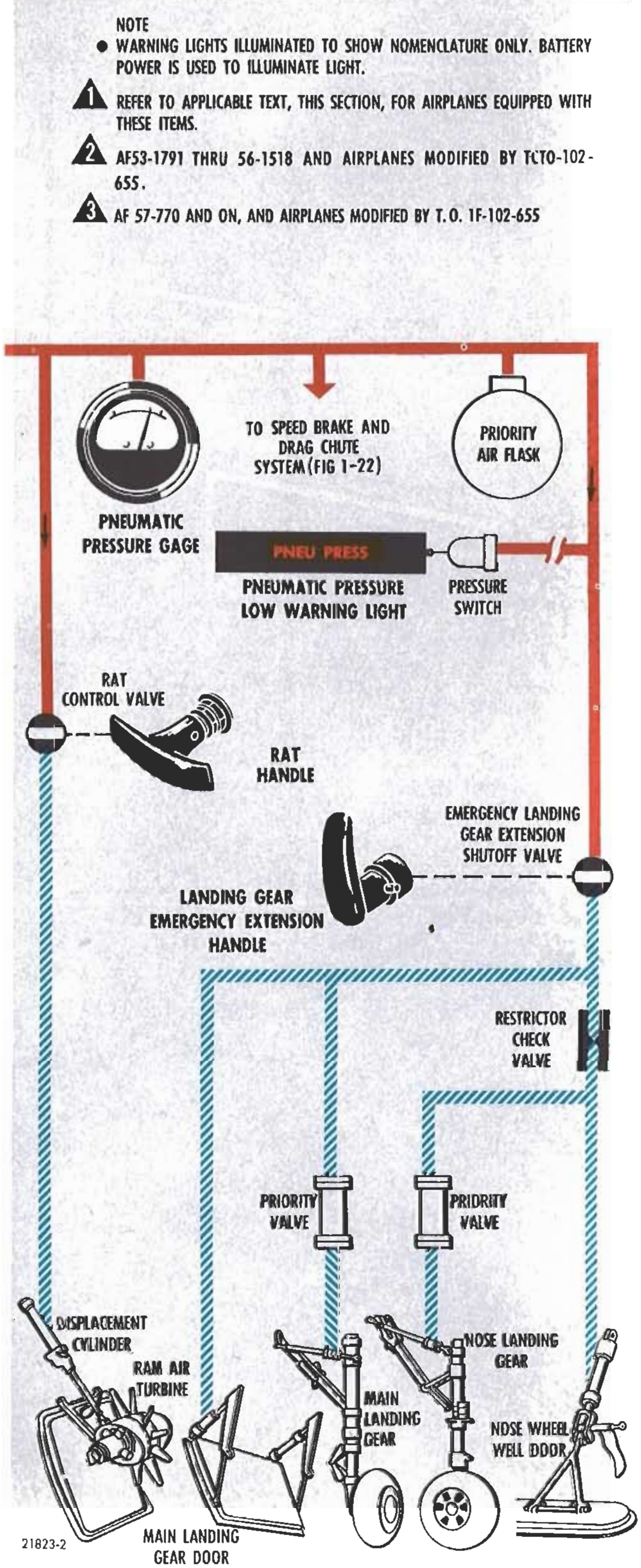


Figure 1-17

system



impact tube on the vertical fin. Refer to Section V for operating limitations.

TRIM SYSTEM

Trim is accomplished by deflecting the control surfaces with electrical trim actuators, installed between the feel mechanism and the mechanical control, which change the neutral (no load) position of the respective systems. Since the mechanical control system friction is less than the feel system forces, the 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 utility switch panel or, on some airplanes, on the lower right-hand side of the throttle quadrant. Takeoff trim can be attained automatically by depressing a button, on the utility switch panel, which will reposition aileron, elevator, and rudder trim to a preset position. An indicator light illuminates when the proper takeoff trim positions are obtained. The trim system is powered from the 28-volt dc essential and nonessential busses and the 115-volt ac nonessential bus.

CAUTION

Do not operate the trim system (including takeoff trim) unless hydraulic pressure is being supplied to the flight control system, to prevent damage to the trim actuators.

PITCH AND YAW DAMPER SYSTEM

Pitch and yaw damper systems are installed to damp out short period oscillations of the airplane and to provide automatic turn coordination. The hydraulically actuated damper systems are electrically controlled. When the damper system is energized, secondary hydraulic pressure is supplied to the damper servo valve portion of the hydraulic elevon package (HEP valve) on some airplanes* or, on early airplanes, to the servo actuator (extendible link), and damper signals are imposed on pilot inputs to control surface positions. These elevons and rudder deflections are not felt at the pilot's controls. When the oscillations have been corrected, the control surfaces return to their original position. Rate gyros sense the direction and velocity of the oscillations and apply signals to the control surfaces to dampen the pitch or yaw oscillations. Aileron motion of the elevons and roll rate is measured and electrically controls the rudder servo actuator to provide automatic turn coordination when the yaw damper system is engaged. Such coordination is optimum only in level flight and deteriorates with variation in load factor. Turn coordination signals to the rudder are modified by an airspeed compensator to insure proper coordination at all airspeeds. Turn coordination is supplied primarily for fire control system attacks and is sufficient for bank angles up to

**AF 56-973 & on.*

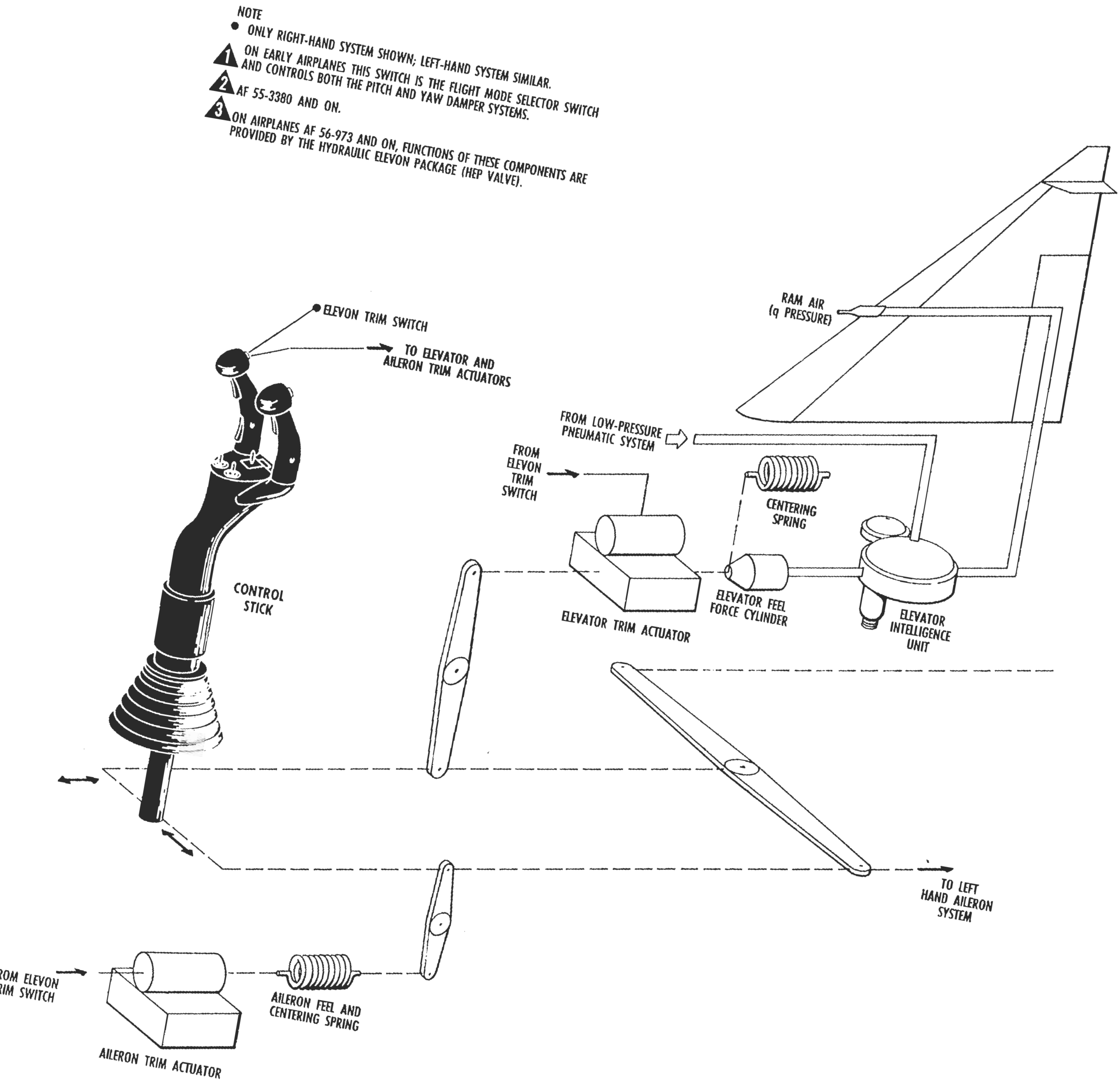
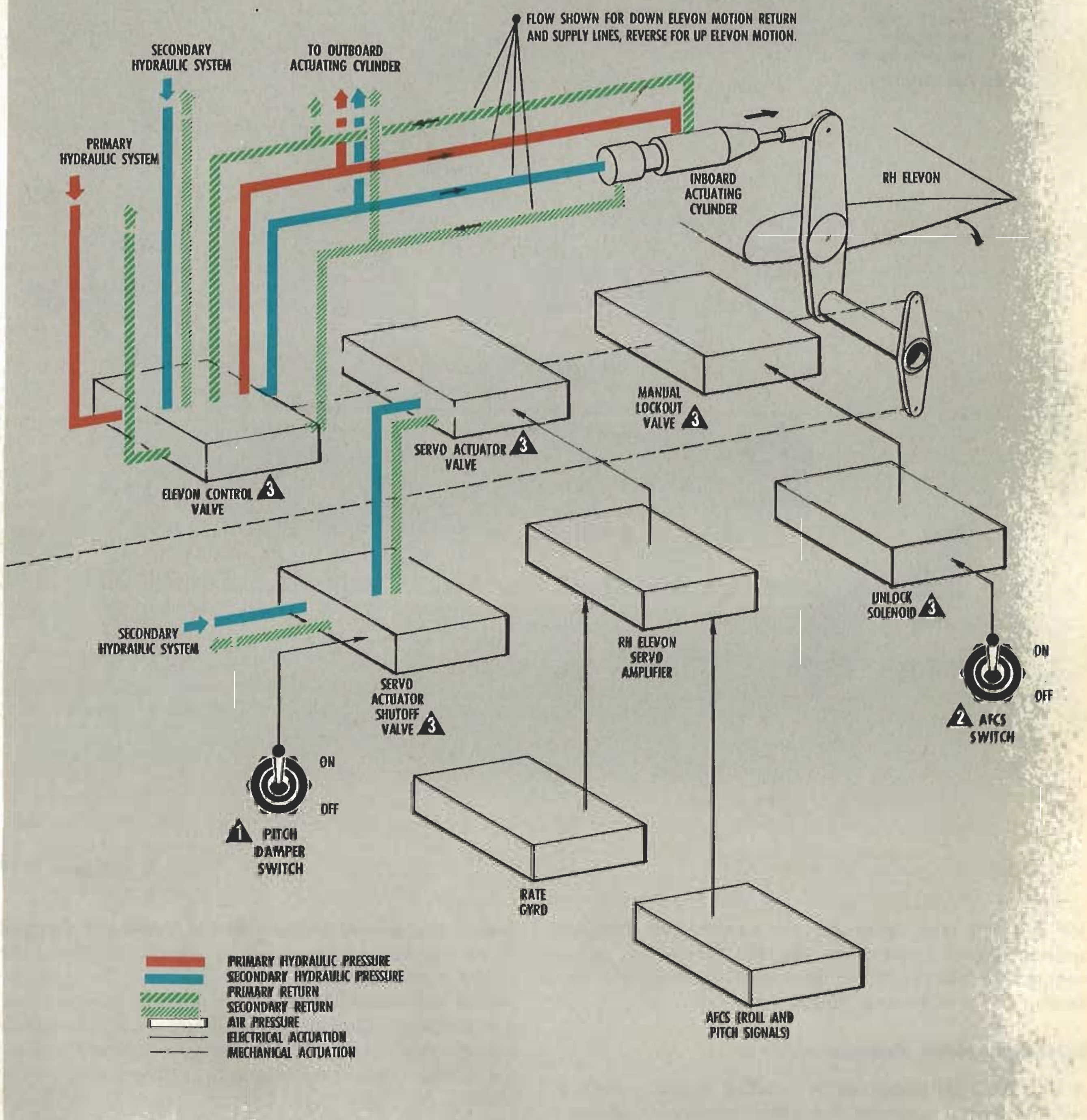

NOTE
● ONLY RIGHT-HAND SYSTEM SHOWN; LEFT-HAND SYSTEM SIMILAR.
1 ON EARLY AIRPLANES THIS SWITCH IS THE FLIGHT MODE SELECTOR SWITCH AND CONTROLS BOTH THE PITCH AND YAW DAMPER SYSTEMS.
2 AF 55-3380 AND ON.
3 ON AIRPLANES AF 56-973 AND ON, FUNCTIONS OF THESE COMPONENTS ARE PROVIDED BY THE HYDRAULIC ELEVON PACKAGE (HEP VALVE).
ELEVON TRIM SWITCH
TO ELEVATOR AND AILERON TRIM ACTUATORS
RAM AIR (q PRESSURE)
FROM LOW-PRESSURE PNEUMATIC SYSTEM
FROM ELEVON TRIM SWITCH
CENTERING SPRING
CONTROL STICK
ELEVATOR TRIM ACTUATOR
ELEVATOR FEEL FORCE CYLINDER
ELEVATOR INTELLIGENCE UNIT
TO LEFT HAND AILERON SYSTEM
ROM ELEVON
RIM SWITCH
AILERON FEEL AND CENTERING SPRING
AILERON TRIM ACTUATOR

Figure 1-18

elevon control system



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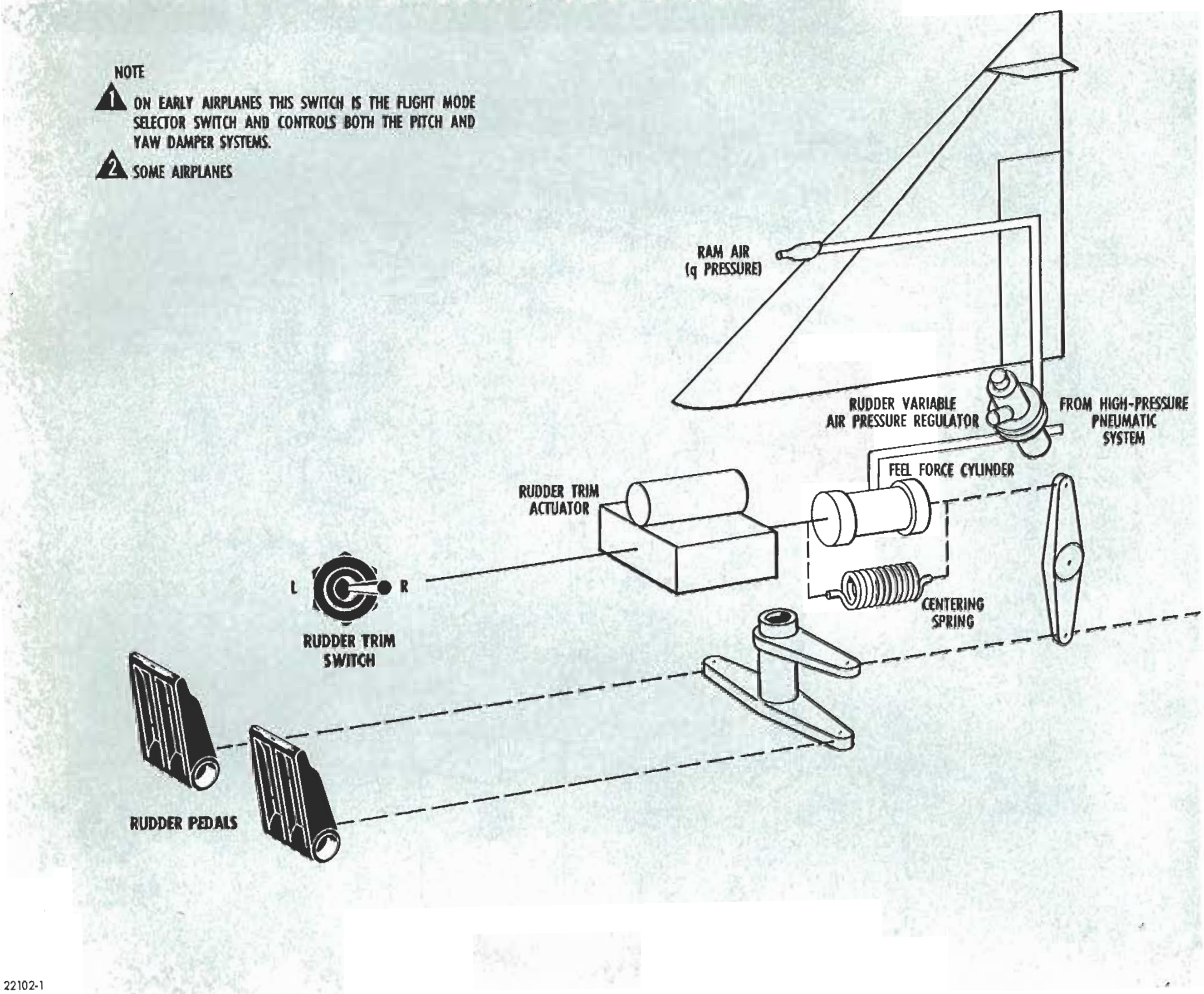


Figure 1-19

35° and roll rates up to 30° per second. The damper system receives power from the 200/115-volt ac nonessential bus and the 28-volt dc nonessential bus. Refer to Section V for operating limitations.

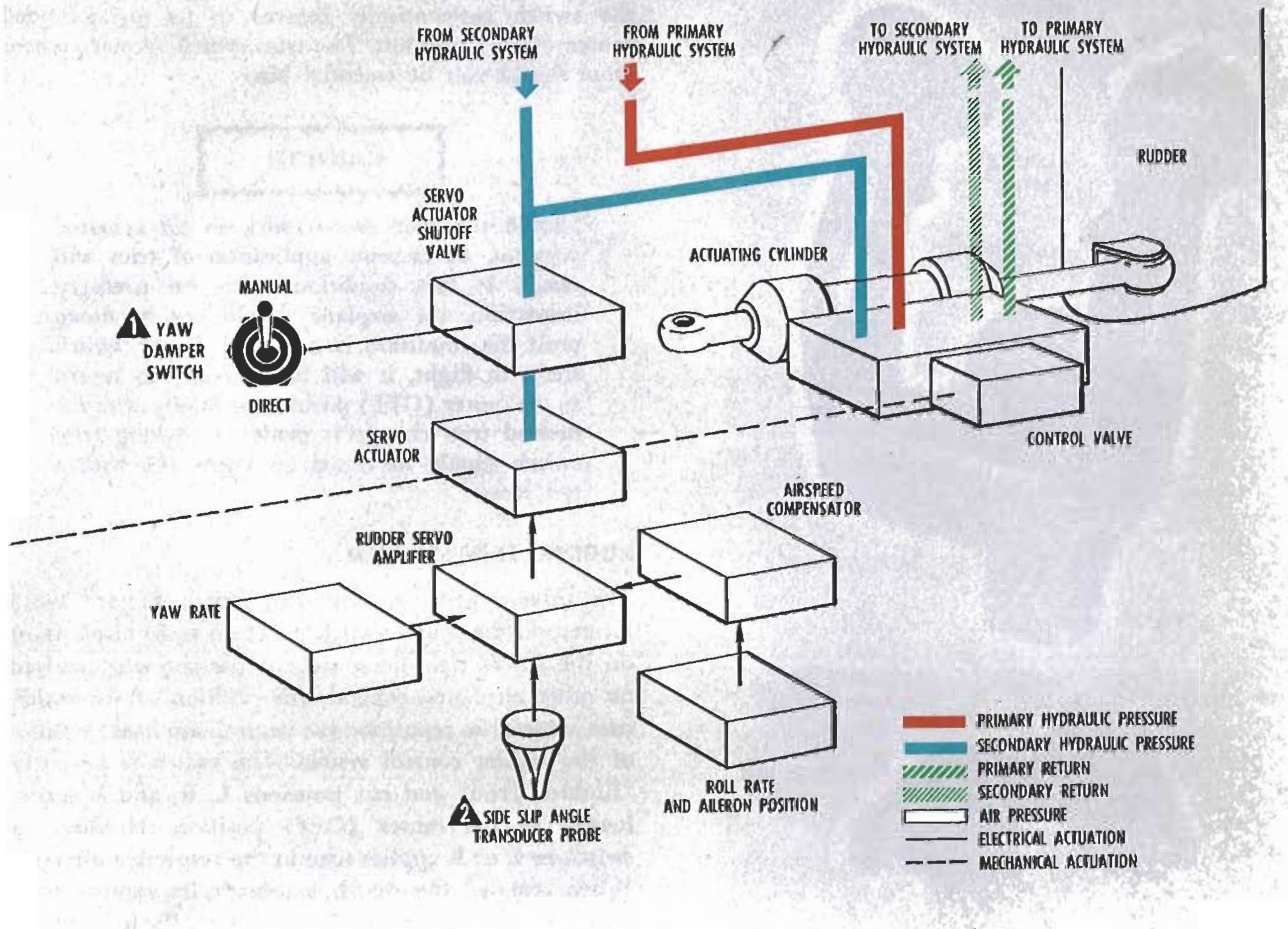
SIDESLIP ANGLE TRANSDUCER

A sideslip angle transducer is installed on early, small-tail airplanes to augment the static directional stability by applying corrective rudder against sideslip. The system incorporates a small detector probe which extends down from the fuselage centerline just forward of the nose wheel well. The detector probe contains two vertical slots which are approximately 90° apart and each positioned approximately 45° from the relative wind during flight. Any sideslip induced by the airplane is detected by this probe and the resultant signals are fed to the rudder servo actuator which initiates corrective action by rudder deflections. The detector probe is free to rotate and will align itself with the relative wind, well below takeoff speed. There are no separate cockpit controls for the sideslip angle transducer as the system is engaged by placing the flight mode selector switch in the MAN position. The sideslip angle transducer receives 200/115-volt ac nonessential and 28-volt dc power through the pitch and yaw damper system.

CONTROL STICK

The control stick (figure 1-20) controls the position of the elevon hydraulic control valves which direct primary

rudder control system



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and secondary system hydraulic pressure to the actuating cylinders to displace the elevon control surfaces. Followup linkages then reposition the control valves and enable control surface deflection to be in proportion to stick movement. Conventional stick movements are used to obtain aileron and elevator action of the elevons. The control stick grips are composed of two grips mounted on a common base. The right-hand grip is the primary grip and incorporates a combination microphone and nose wheel steering button, elevon trim switch, emergency damper disconnect button, armament trigger and momentary interrupt trigger. The left-hand grip, when unlocked, serves as the radar antenna hand control and incorporates controls for the fire control system. Refer to FIRE CONTROL SYSTEM, Section IV.

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. Followup linkage repositions the control valve which maintains rudder deflection in proportion to pedal movement. The rudder pedals can be simultaneously adjusted fore and aft by an adjustment crank mounted between the pedals. 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 button is depressed. Refer to WHEEL BRAKE SYSTEM and NOSE WHEEL STEERING SYSTEM, this Section.

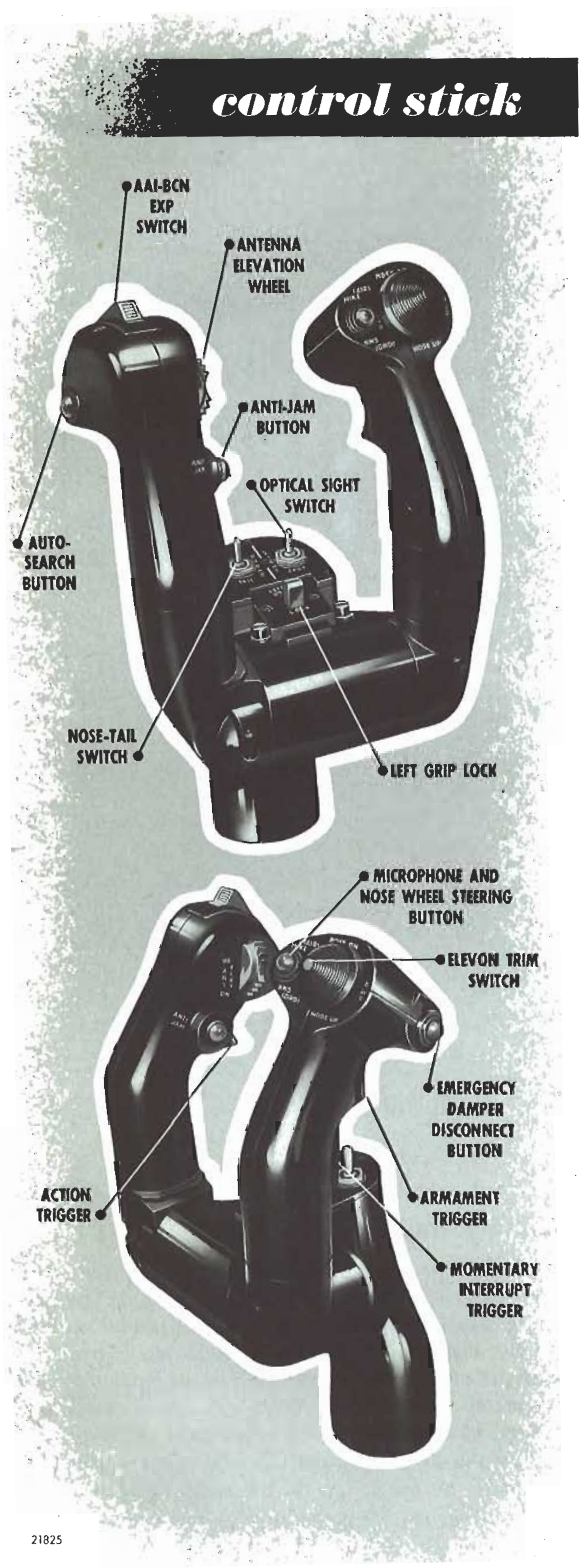

Figure 1-20

ELEVON TRIM SWITCH

Lateral and longitudinal trim is controlled by a five-position elevon trim switch (figure 1-20), located on the right-hand control stick grip. The switch has NOSE UP, NOSE DOWN, LWD, RWD and is spring-loaded to the center (OFF) position. Holding the switch in the direction of desired trim powers the ailerons or elevator trim actuator to reposition the neutral (no load) position of the respective flight control system. When released the switch automatically returns to its spring-loaded center (OFF) position. The trim switch receives power from the 28-volt dc essential bus.

CAUTION

Should the trim switch stick in an actuated position, an extreme application of trim will result. 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 with a red cross.

RUDDER TRIM SWITCH

The three-position rudder trim switch (figure 1-21), located on the utility switch panel on some airplanes or on the lower right-hand side of the throttle quadrant on other airplanes, controls the position of the rudder trim actuator to reposition the neutral (no load) position of the rudder control system. The switch is placarded "Rudder Trim" and has positions L, R, and is spring-loaded to the center (OFF) position. Holding the switch to L or R applies trim in the respective direction. When released the switch automatically returns to its spring-loaded center (OFF) position. The trim switch receives power from the 28-volt dc essential bus.

TAKEOFF TRIM BUTTON AND INDICATOR LIGHT

The takeoff trim button (figure 1-21), located on the utility switch panel, when depressed, trims all control surfaces to the proper position for takeoff. The takeoff trim position of the aileron action and rudder is neutral, and the elevator action is 5° nose up. While the button is held depressed, a green indicator light illuminates when the trim system reaches the proper position for takeoff. This light displays "TAKEOFF TRIMMED" when illuminated and is automatically dimmed when the instrument panel lights are on if the thunderstorm lights are off. The takeoff trim circuit is deenergized when the nose landing gear is retracted and receives power from the 28-volt dc nonessential bus.

TRIM SERVO SWITCH

Note

If the trim servo switch is installed, it is deactivated.

FLIGHT MODE SELECTOR SWITCH

On some airplanes, a two-position flight mode selector switch* is used to engage the pitch and yaw damper system and the sideslip angle transducer (if installed). The switch, located on the utility switch panel, is placarded "Flight Mode" and has MAN and DIRECT MAN positions. The switch is spring-loaded to DIRECT MAN and utilizes a solenoid to hold the switch in MAN position. Damper system components are energized whenever power is on the airplane (nonessential buses), and placing the switch to MAN position supplies hydraulic pressure to the damper system servo (extendible link) actuators which engage the system. The MAN position also energizes the sideslip angle transducer, which is a part of the yaw damper system. The damper system will automatically disengage (switch returns to DIRECT MAN position) when the emergency damper disconnect button (on the control stick grip) is depressed, or the switch may be manually placed to DIRECT MAN position to disengage the system. The flight mode selector switch receives power from the 28-volt dc nonessential bus.

YAW DAMPER SWITCH

A yaw damper switch**, located in the lower left-hand corner of the AFCS panel (figure 4-16), is used to engage the yaw damper system. On early airplanes, the switch has a spring-loaded DIRECT (off) position to disconnect the yaw damper and a solenoid held MANUAL (on) position which engages the damper. On later airplanes, the switch positions are YAW DAMPER and OFF. Damper system components are energized when the nonessential buses are supplied with power and activated when the switch is placed in the MANUAL or YAW DAMPER position. Hydraulic pressure is supplied to the rudder damper system servo actuators to damp yaw oscillations and provide turn coordination. The yaw damper system will automatically disengage (the switch will return to DIRECT or off position) when the emergency damper disconnect button is depressed or the switch may be manually placed to DIRECT or off position to disengage the system. The yaw damper switch receives power from the 28-volt dc nonessential bus.

PITCH DAMPER SWITCH

A pitch damper switch** (figure 1-21), located on the utility switch panel, is used to engage the pitch damper system. The switch is placarded "Pitch Damper" and has

*AF 53-1791 thru 55-3379 unless modified by TCTO 1F-102A-546.

**AF 55-3380 & on, & airplanes modified by TCTO 1F-102-546.

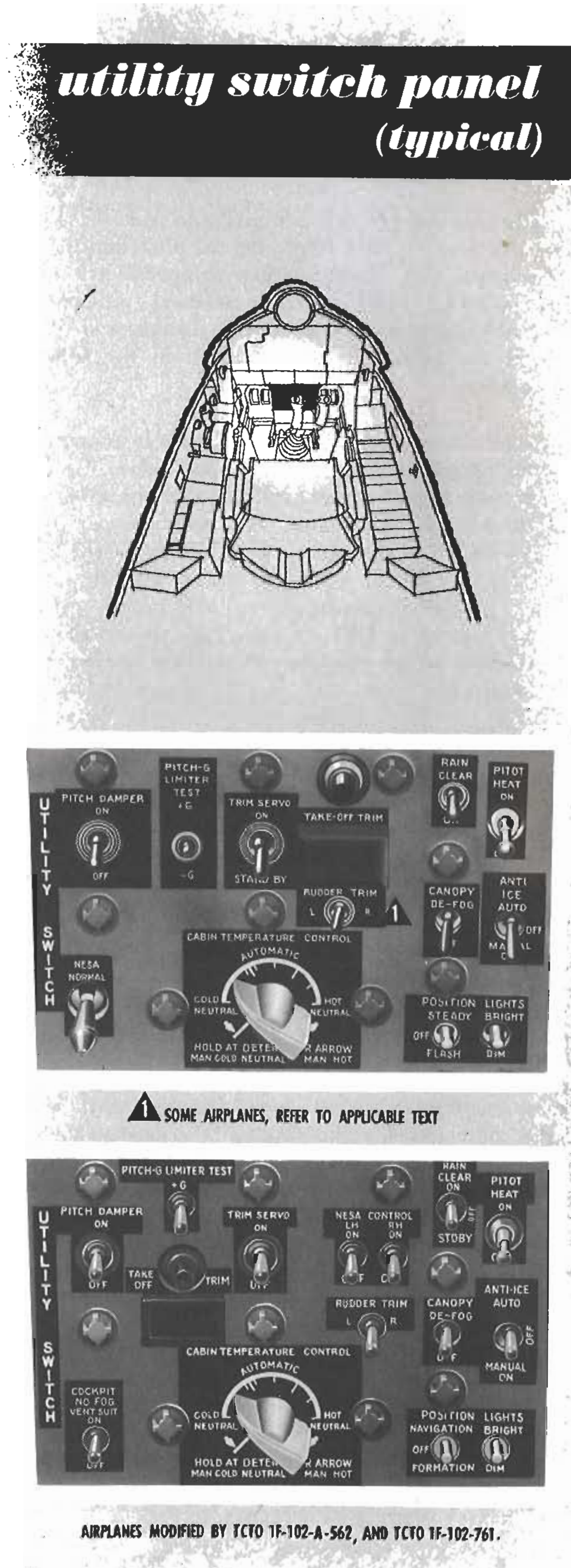

Figure 1-21

a spring-loaded OFF position and a solenoid-held ON position. Pitch damper system components are energized when power is on the nonessential buses. Placing the switch in the ON position supplies hydraulic pressure to the elevon damper system servo actuators to activate the system.

Note

The yaw damper switch must be in MANUAL (on) position prior to placing the pitch damper switch to ON. When the yaw damper switch is in DIRECT (off), power is removed from the holding solenoid of the pitch damper switch and the switch will not engage in the ON position.

When the automatic flight control system is in operation and the preset pitch g limits are exceeded, the pitch damper switch will automatically go to the OFF position, disengaging the system. The pitch damper system will also automatically disengage (the switch will return to the OFF position) when the emergency damper disconnect button is depressed or the switch may be manually placed to OFF to disengage the system. The pitch damper switch receives power from the 28-volt dc nonessential bus.

EMERGENCY DAMPER DISCONNECT BUTTON

The emergency damper disconnect button placarded "Emer Man" (figure 1-20), located on the control stick right-hand grip, is used to simultaneously disengage the damper systems and AFCS. When the button is depressed, the circuit is broken to the holding solenoids for the flight mode selector and AFCS switches which automatically return to DIRECT MAN and OFF positions, respectively.

MOMENTARY INTERRUPT (MANUAL MODE) TRIGGER

The momentary interrupt trigger (figure 1-20), located on the control stick right-hand grip, is used to momentarily disengage the AFCS. This trigger should be used to disengage the AFCS to make large attitude or heading changes. Releasing the trigger will re-engage the AFCS to hold the attitude or heading prevailing at the time of release. The trigger when depressed interrupts power to a phase of the AFCS.

SPEED BRAKES SYSTEM

Two hydraulically operated, electrically controlled speed brakes (figure 1-22) are located above the tail cone and can be used to slow the airplane at all speeds. 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. When extending the speed brakes, a slight nose-up trim change occurs, and when retracting the speed brakes a nose-down trim change occurs. Secondary hydraulic system pressure is supplied through a dc electrically operated selector valve to actuate a pair of hydraulic cylinders for each brake. The speed brakes are synchronized to give equal angular operation and require approximately 2 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 incorporated and is to be used in the event of secondary hydraulic system failure to allow the drag chute to deploy. The emergency system bypasses the speed brakes switch on the throttle and furnishes high-pressure pneumatic air to the speed brakes actuating cylinders when the drag chute handle is pulled out (full travel), then rotated 90° right and pulled again.

CAUTION

- Emergency speed brakes extension should be used only in event of secondary hydraulic system failure. Use of the system when the secondary hydraulic system is pressurized and operating will not aid drag chute deployment and will result in pumping hydraulic fluid through a vent into the vertical fin structure on some airplanes. This could result in hydraulic fluid draining onto the tailpipe.
- Emergency speed brakes extension is designed for use only on the landing roll (or aborted takeoff roll) when drag chute deployment is desired. Inflight use of this system will result in loss of the drag chute when above 160 KIAS.

The speed brakes cannot be opened or closed without dc power, either by the normal or emergency systems.

SPEED BRAKES GROUND SAFETY LOCKS

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

SPEED BRAKES SWITCH

The three-position speed brakes switch (figure 1-7), located on top of the throttle, is used to control speed brake operation. The switch is placarded "Speed Brakes" and has fixed positions of IN, OUT, and a center (neutral) position which controls the selector valve accordingly. The center (neutral) position is indicated by a white 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. The speed brakes switch receives power from the 28-volt dc essential bus.

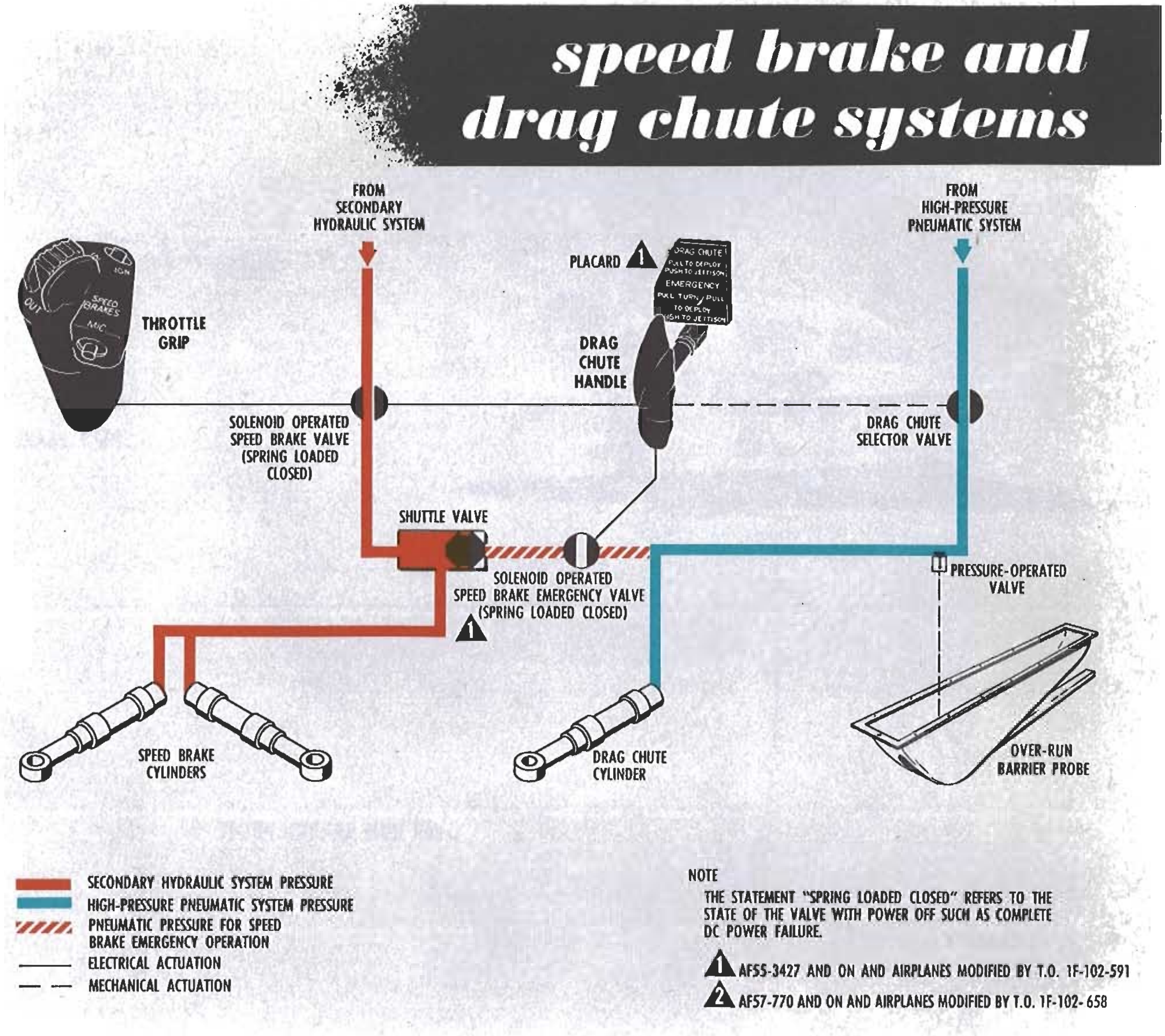

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Figure 1-22

CAUTION

Damage can result to the speed brakes actuators or their electrical conduits if the speed brakes are closed after the drag chute has been jettisoned, or if the doors are closed with drag chute not installed.

LANDING GEAR SYSTEM

The tricycle landing gear and wheel well doors 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 and the normal landing gear handle to permit emergency gear retraction while on the ground or while airborne. Normal gear extension, retraction, and emergency retraction are actuated by secondary hydraulic system pressure. In the event of electrical or hydraulic system failure the gear can be extended by

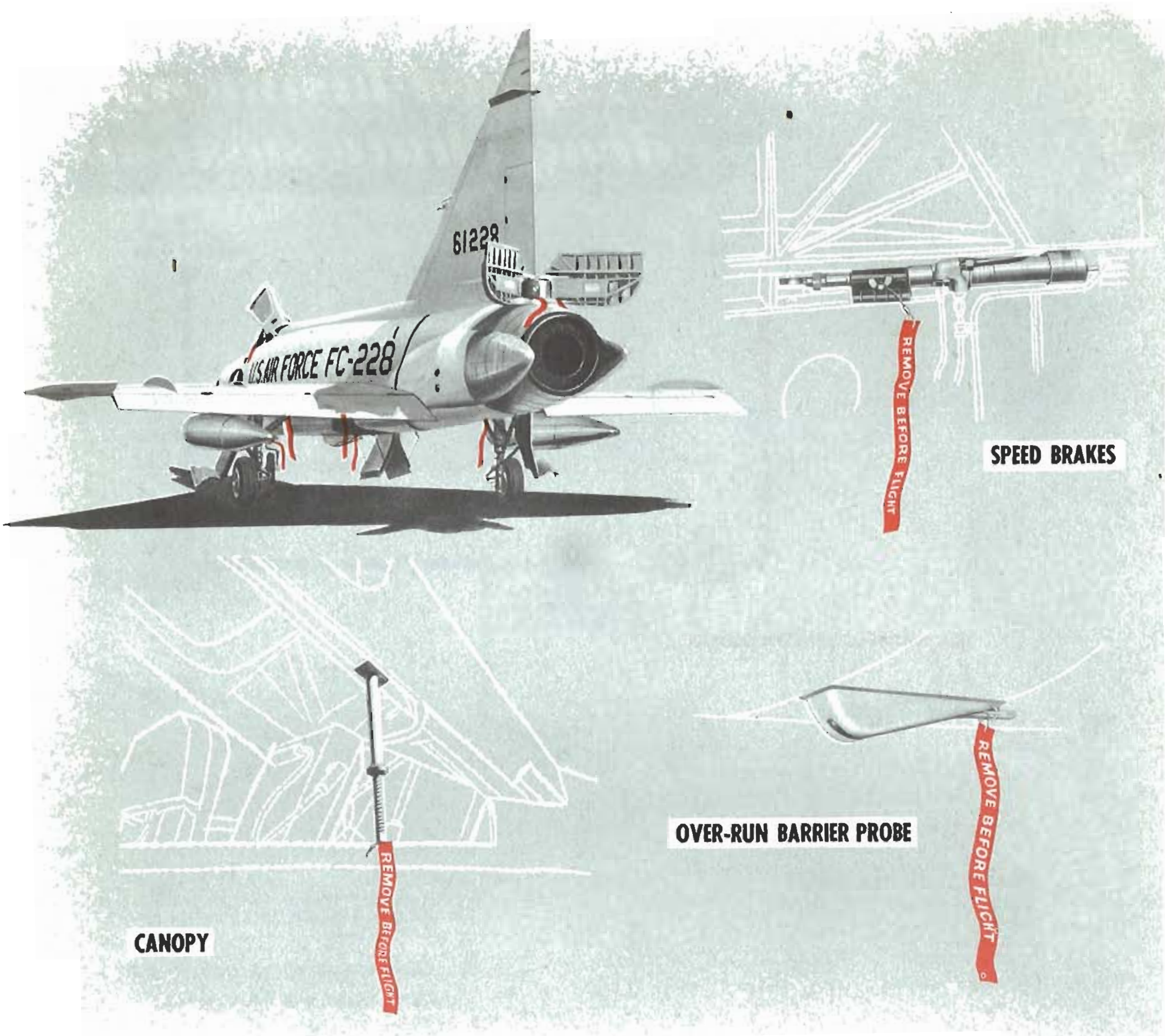


Figure 1-23

high-pressure pneumatic system pressure which is routed to the normal hydraulic actuating cylinders. During normal operation, gear extension and retraction time is approximately four to six seconds. A hydraulic steering unit is built into the nose gear assembly to provide nose wheel steering and also serve as a conventional shimmy damper. The main wheels are equipped with pneumatically operated multiple-disc type brakes. The main gear drag braces serve as pneumatic pressure reservoirs for the wheel brake system. Later airplanes* are equipped with a main landing gear which, when extended, is tilted forward at a slight angle. Therefore, while the airplane is on the ground the main gear is farther forward in relation to the airplane cg, which improves landing characteristics.

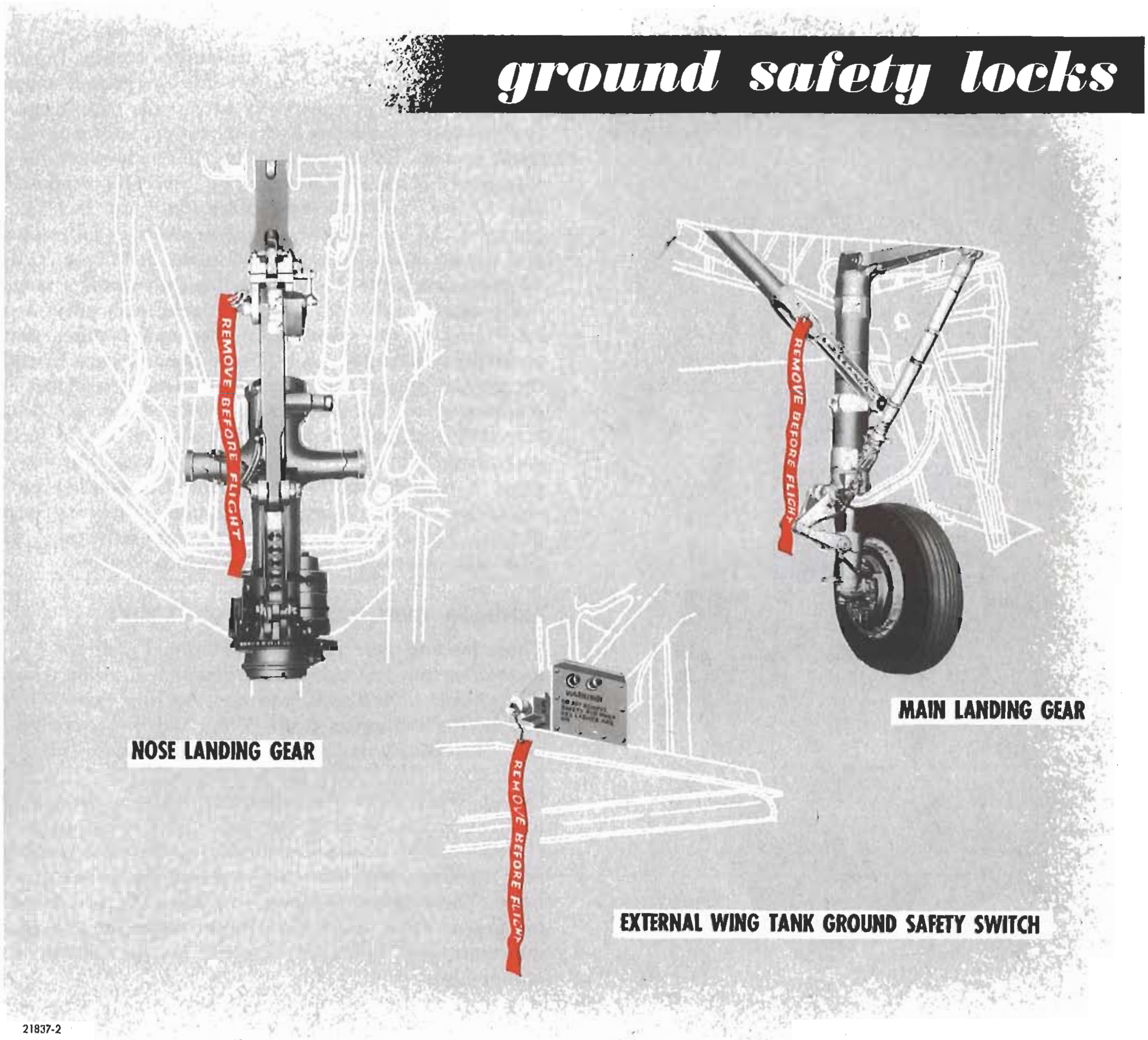
LANDING GEAR GROUND SAFETY LOCKS

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 landing gear handle (figure 1-24), located on the left-hand auxiliary instrument panel, electrically controls normal operation of the gear and wheel well door hydraulic selector valves. When the airplane is airborne, moving the handle to the UP position energizes the hydraulic selector valves to apply secondary hydraulic

**AF 53-1812 & on.*



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.

When the landing gear handle is moved to the 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 extended. The knob on the landing gear handle is wheel-shaped and contains the landing gear warning light. The landing gear circuit is powered by the 28-volt dc essential bus.

LANDING GEAR EMERGENCY-UP BUTTON

The landing gear emergency-up button is located on the left-hand auxiliary instrument panel (figure 1-24). The button is placarded "Emer Gear Up" and can be used to retract the landing gear when the airplane is in the air or moving on the ground. On the ground, depressing the button bypasses the ground safety switches and the gear will retract if the normal landing gear handle is in the UP position and the airplane is moving. When airborne, depressing the button will allow the landing gear control circuit to bypass the landing gear handle and

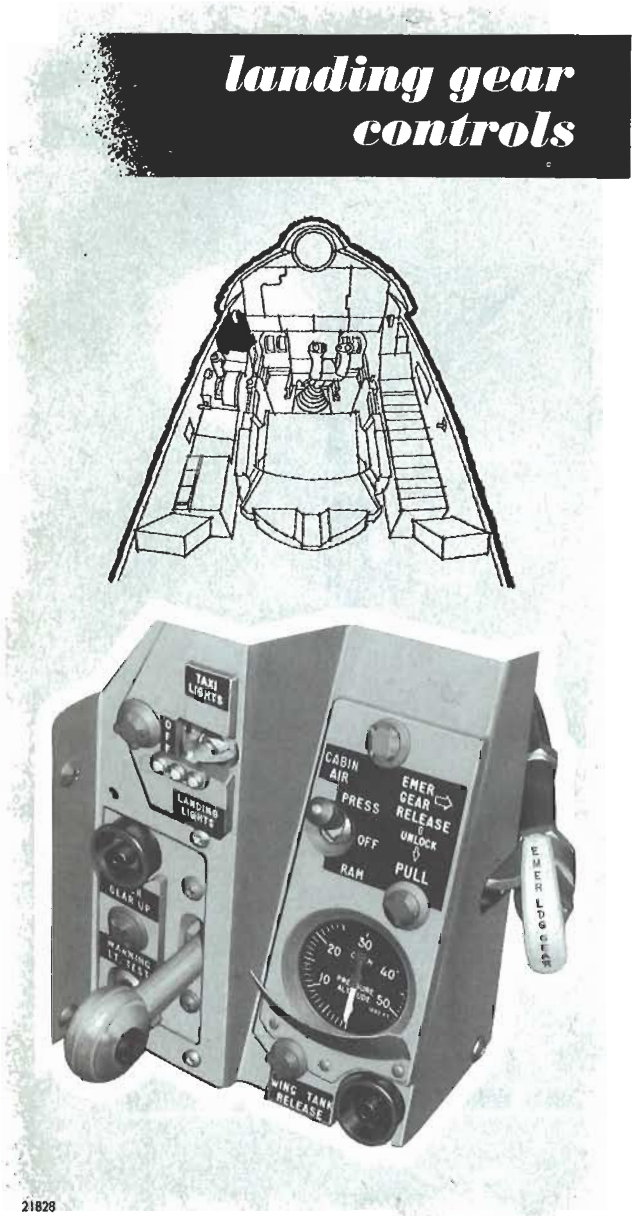


Figure 1-24

retract the gear with the normal control handle in the UP or DOWN position. To insure positive operation after take-off, the gear emergency-up button must be held depressed until the gear completely retracts. If the gear is retracted with the button while airborne with the gear handle in the DOWN position, it will be necessary to cycle the gear handle to the UP position and return to DOWN position to extend the gear. After emergency gear extension has been accomplished, the emergency-up button will be inoperative until the landing gear emergency extension handle is pushed in. In the event of ground safety switch malfunction, the emergency-up button may be used to retract the gear when airborne. The emergency-up button receives power from the 28-volt dc essential bus.

LANDING GEAR EMERGENCY EXTENSION HANDLE

The landing gear emergency extension handle (figure 1-24), suspended below the left side of the instrument panel, is used to pneumatically extend the landing gear in the event of secondary hydraulic system failure or electrical system failure or malfunction. The handle is placarded "Emer Gear Release-Pull" on early airplanes* and "Emer Gear Release-Unlock-Pull" on later airplanes**. An arrow indicates the necessity to pull down and out on later airplanes. Pulling the handle out fully (approximately two inches) mechanically opens a pneumatic shutoff valve to supply high-pressure pneumatic system air to the wheel well door and landing gear actuating cylinders which will open the doors and extend the gear. A spring clip is installed on early airplanes at the base of the handle to lock the handle in the fully extended position. On later airplanes the handle locks automatically in the fully extended position. The emergency extension handle will extend the landing gear regardless of the position of the normal landing gear handle. There are no provisions for retracting the gear after extension by the emergency systems.

LANDING GEAR POSITION INDICATORS

Three landing gear position indicators (5, figure 1-4), located on the left side of the instrument panel, show the position of the main and nose landing gear. When one of the indicators reads "UP," the respective gear and door are up and locked. Each indicator displays a symbolized wheel when the respective gear is down and locked. When there is no electrical power or the gear is in an unlocked position, the indicator displays parallel red and yellow stripes. On some airplanes† the landing gear position indicators are replaced by three green lights. These lights illuminate only when the corresponding gear is down and locked. Power to the landing gear indicators and lights is supplied by the 28-volt dc essential bus.

LANDING GEAR WARNING LIGHT AND TEST BUTTON

A red warning light, located within the wheel-shaped knob on the landing gear handle (figure 1-24) will illuminate at any time the landing gear is not in the position selected by the landing gear handle. The light will also illuminate when the landing gear is not down and locked at an altitude of 13,500 (±1000) feet climbing or 9500 (±1000) feet descending, if the throttle is retarded below FULL MIL POWER position, and the airspeed is less than 250 (±14) KIAS on some airplanes and 210 (±10) KIAS on other airplanes‡. The warning light is automatically dimmed when the instrument panel lights are on if the thunderstorm lights are off. On airplanes that have the three green landing gear indicator lights†, a

*AF 53-1791 thru 56-972.
**AF 56-973 & on.
†Airplanes modified by TCTO 1F-102-728.
‡AF 57-770 & on.

red gear unsafe warning bar light has been added. The bar light has a press-to-dim feature and when dimmed, the landing gear handle warning light will also dim. The warning bar light is wired into the circuit with the landing gear control handle warning light and both will illuminate under the above conditions. On all airplanes the gear unsafe warning light(s) can be checked by depressing the test button located on the landing gear control panel. An audible warning signal is also provided to give an audible signal in the radio headset at any time the landing gear warning lights illuminate. The light(s) receive power from the 28-volt dc essential bus.

CAUTION

On airplanes that have the red gear unsafe warning bar light, do not depress the press-to-test button on the landing gear control panel and the red warning bar light at the same time. This may result in damage to the audible signal generator.

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 a button on the control stick grip. The hydraulically powered nose wheel steering unit will position the nose wheel within approximately 50° each side of center, when the airplane is on the ground.

Note

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

A mechanically operated valve 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 or has turned in excess of 50° from center the nose wheel is free to swivel. The steering unit also serves as a conventional shimmy damper up to 50° either side of center.

NOSE WHEEL STEERING UNIT GROUND LOCK PIN

The nose wheel steering unit ground lock pin 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 WHEEL STEERING BUTTON

The nose wheel steering button (figure 1-20), located on the control stick right-hand grip, engages the nose wheel steering system. The button is placarded "Mic (Air)-Nws (Gnd)." When the button is depressed a solenoid operated valve allows secondary hydraulic system pressure to engage the steering unit which is then controlled by the rudder pedals. If the rudder pedals are displaced and the nose wheel is centered (as during cross-wind landing roll) it is necessary to neutralize the rudder pedals to obtain control of the steering unit. Likewise, if the nose wheel is not centered when the button is depressed, the pedals must be positioned in relationship to the nose wheel before steering will engage. On some airplanes, once engaged, nose wheel steering is available on the ground as long as the button is held depressed and the wheel does not exceed 50° from center in either direction. On other airplanes* a relay is installed enabling the pilot to actuate the nose wheel steering by momentarily depressing the button. This eliminates the requirement of the pilot holding the button depressed to keep the system engaged. Depressing the button again will subsequently disengage the system.

Note

The nose wheel steering button functions as a secondary microphone control when the weight of the airplane is off the nose gear, or the nose wheel exceeds 50° from center in either direction.

The nose wheel steering button receives power from the 28-volt dc essential bus.

WHEEL BRAKE SYSTEM

The multiple-disc type, pneumatically operated brakes are installed on the inboard side of the main wheels. Conventional toe action on the rudder pedals individually applies independent hydraulic pressure to control the position of spring-loaded metering valves which, in turn, control pneumatic pressure to actuate the brakes. The main landing gear drag braces serve as pneumatic reservoirs for the brake system and are connected to the high-pressure pneumatic system through check valves which maintain braking pressure in the event high-pressure 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. However, pneumatic brake system pressure should always be equal to or greater than high-pressure pneumatic system pressure.

**AF 56-1430 & on, & airplanes modified by TCTO 1F-102A-557.*

Relief valves are installed to protect the brake system against excessive pressure. No emergency, antiskid, or parking brake system is provided.

DRAG CHUTE AND OVERRUN BARRIER PROBE SYSTEM

A drag chute is provided to reduce landing roll distance and is designed 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. The drag chute is deployed and jettisoned by a drag chute control handle which supplies penumatic pressure to operate the chute deployment mechanism. Protective strips are also added to the chute risers to prevent heat from aft section of the airplane from melting the risers. The chute pack is secured to the airplane to prevent deployment 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.

Note

- If the speed brakes are closed and dc essential bus power is available, secondary hydraulic system pressure, or in the event of secondary hydraulic system pressure failure, high-pressure pneumatic system pressure will open the brakes for drag chute deployment.
- Pulling the drag chute handle will open the speed brakes regardless of the speed brakes switch position unless the jettison pin is dislodged or a malfunction exists in the speed brakes circuit. If speed brakes are open, dc power is not required for drag chute deployment.

The chute mechanism incorporates a shear pin to prevent structural damage if the chute is deployed above approximately 160 KIAS. On some airplanes an extendible overrun barrier probe is installed and will extend simultaneously with the drag chute activation.

DRAG CHUTE HANDLE

The drag chute handle (2, figure 1-4), located to the left 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 1.5 inches) actuates a switch that electrically controls the speed brakes selector valve which applies secondary hydraulic system pressure to open the speed brakes. Simultaneously, the handle mechanically operates a selector valve to supply high-pressure pneumatic system pressure to the chute deployment mechanism securing the chute risers and pulling the ripcord pins which deploys the chute. The drag chute handle controls high-pressure pneumatic system air to both the speed brakes and the drag chute and is used in the event of secondary hydraulic system failure to operate the speed brakes for drag chute deployment. Pulling the drag chute handle fully out, rotating 90° to the right, then out again, supplies 28-volt dc power to a solenoid operated pneumatic selector valve which supplies high-pressure pneumatic air to the speed brakes cylinders. 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 fully in releases pneumatic pressure to the deployment mechanism which jettisons the chute. 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.
- Do not jettison the drag chute unless the speed brakes switch is in the neutral position. This will prevent venting hydraulic fluid from the secondary hydraulic system if inadvertent emergency speed brakes extension had occurred during drag chute deployment.
- Damage can result to the speed brakes actuators or their electrical conduits if the speed brakes doors are closed after the drag chute has been jettisoned, or if the doors are closed with drag chute not installed. This happens when the lower drag chute restraining strap falls down so that the bolt and nut on the strap are in the direct path of the protruding bolts on the speed brakes electrical conduits.

On airplanes equipped with overrun barrier probe*, pulling the drag chute handle will mechanically extend the overrun barrier probe at the same time the drag chute is activated.

PITOT-STATIC SYSTEM

The pitot-static system (figure 1-25) supplies pitot pressure to the airspeed indicator, the pressure switch of the landing gear warning system, the fire control system, AFCS, and the engine pressure ratio gage. Static pressure is applied to the airspeed indicator, the vertical velocity indicator, AFCS, and the altimeter. The pitot-static tube is mounted on the end of the nose boom. Ram air (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. All pitot tubes are anti-iced by electrical power from the 28-volt dc essential bus.

**AF 57-770 & on, & airplanes modified by TCTO 1F-102-658.*

INSTRUMENTS

Note

This paragraph covers only those instruments which cannot be considered to be parts of complete systems, such as fuel system, engine, etc.

MM-2 ATTITUDE INDICATOR

The airplane is equipped with the MM-2 remote attitude indicator to give visual indication of the flight attitude of the airplane in pitch and roll. This indicator is a remote indicating instrument having the gyroscopic control unit located on the centerline of the airplane at the aft end of the upper electronics compartment with the indicating phase located on the instrument. The system is powered from the 115-volt, 3 phase, 400 cycle essential bus and the 28-volt dc essential bus. Erection of the gyro requires approximately 2½ minutes after application of power and can be observed by disappearance of the "OFF" power failure flag visible through the cover glass of the indicator. The "OFF" flag will appear in case of complete ac or dc power failure. However, a slight reduction in ac or dc power, or failure of certain electrical or mechanical components within the system, will not cause the "OFF" flag to appear, even though the system is not operating properly.

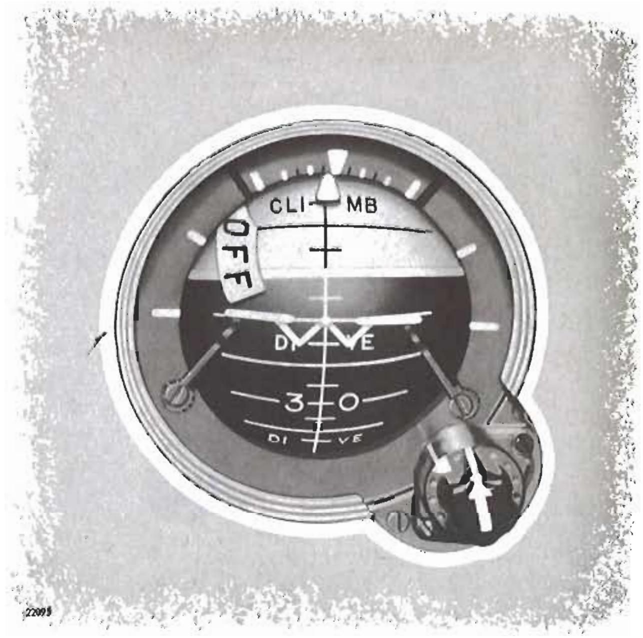
WARNING

- During flight, it is possible that a malfunction of the attitude indicator might be determined only by checking it with the other flight instruments and with the artificial horizon on the radar scope.
- If the "OFF" flag requires longer than 2½ minutes to retract or any oscillations are noted on the indicator after the "OFF" flag retracts, a possibility of a malfunction exists. Either of the above is cause for rejection of the indicator and should be noted on Form 781.

The instrument is operative through 360° of roll, 82° of climb, and 82° of dive, and it is not likely to tumble even during extreme maneuvers. However, should the gyro tumble, it will require approximately 15 minutes to erect. Indicator error is less than ½° in level flight, and, up to a turn rate of 40° per minute, the indication error compares to that of a conventional gyro. In turns of 40° or more per minute a compensating mechanism in the instrument limits turn error indication to 2°

WARNING

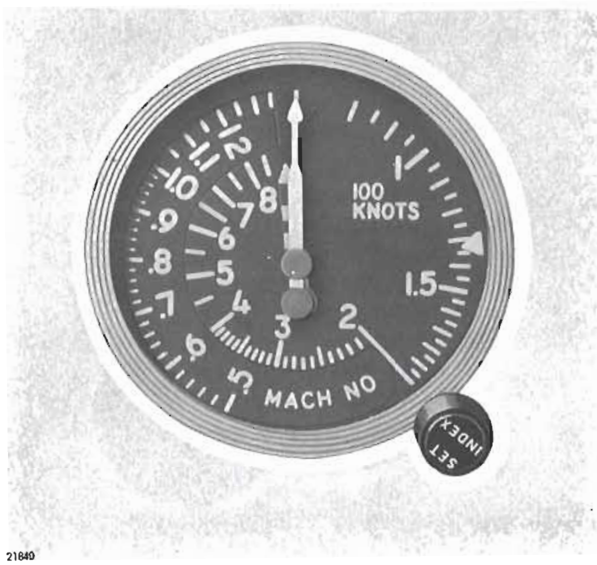
A slight amount of pitch error in the indication of the MM-2 attitude indicator will result from accelerations or decelerations. It will appear as a slight climb indication after a forward acceleration and as a slight dive indication after deceleration when the airplane is flying straight and level. This error will be most noticeable at the time the airplane breaks ground during the takeoff run. At this time, a climb indication error of approximately 1½ bar widths will normally be noticed; however, the exact amount of error will depend upon the acceleration and elapsed time of each individual takeoff. The erection system will automatically remove the error after the acceleration ceases.



The indicator does not have a manual caging handle. When power is turned off, a snubber automatically grips the gimbal and keeps it from tumbling. When power is turned on, the snubber is released after a 15-second time delay. As level flight pitch attitude of the airplane varies with different loadings and speeds, a pitch trim knob is provided on the indicator to center the horizon bar after the airplane has been trimmed for level flight.

MACHMETER-AIRSPEED INDICATOR

The Machmeter airspeed indicator (8, figure 1-4) will indicate airspeeds up to 850 knots and up to Mach 2.2. Early airplanes are equipped with an ME-2 indicator and later airplanes are equipped with an ME-4 indicator. The instrument contains a dual-pointed needle which points to a movable Mach scale that rotates with altitude changes to show the Mach number that is equivalent to indicated airspeed for the particular flight altitude. For example, at sea level, the Mach 1.0 scale might be opposite the 650 knot graduation of the IAS scale. A climb to 40,000 feet would cause the Mach scale to rotate so that Mach 1.0 would be opposite the 310 knot scale. A knurled knob, located at the lower right side of the



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instrument, allows setting of a movable index marker (this marker moves along the perimeter of the dial) to reference a desired speed. Mach number is read from the airspeed pointer on the Mach number dial (inside the cut-out on the ME-4 indicator and on the outer left-hand edge on the ME-2 indicator). The indicators use impact and static pressures from the pitot-static system. The red and black limiting pointer is not used on this airplane and has been set to the full limit of its upward travel. Early airplanes* have an airspeed correction card in a holder, located on the left side of the cockpit, above the console and forward of the throttle quadrant.

AF 53-1791 thru 55-3357.

22099

ACCELEROMETER

A three-pointer accelerometer located at the lower left corner on the main instrument panel, on some airplanes, shows positive and negative g-loads. In addition to the conventional indicating pointer, there are two recording pointers (one for positive g-loads and one for negative g-loads) which follow the indicating pointer to its maximum attained travel. The recording pointers remain at the maximum travel position reached by the indicating pointer, thus providing a record of maximum g-loads encountered. To return the recording pointers to the normal (one g) position, it is necessary to press the knob on the lower left corner of the instrument ring.

CAUTION

Approach g limits slowly as difference in location of accelerometer and airplane center of gravity can introduce a lag as much as one g when making rapid changes in attitude.

ALTIMETER

A conventional altimeter (36, figure 1-4) is installed for use in determining pressure altitude of the airplane above sea level. Some airplanes are equipped with an MB-2 altimeter which is conventional in indication except for warning hash-marks in the lower portion of the instrument. The hash-marks are covered above 16,000 feet and become visible when descending below 16,000 feet. Other airplanes are equipped with the MA-1 altimeter which is similar to the MB-2 except for greater accuracy. Later airplanes* have an altimeter calibration card in a holder, located on the left side of the cockpit, above the console and forward of the throttle quadrant.

WARNING

The barometric setting knob can be turned so that the barometric disc will rotate through 360°. If the correct altimeter setting is then established, the altimeter will indicate 10,000 feet in error.

TURN-AND-SLIP INDICATOR

A conventional turn-and-slip indicator (37, figure 1-4) located on the instrument panel is rated for a four-minute turn (1½° per second). On some airplanes** the turn-and-slip indicator is mounted perpendicular to the longitudinal axis of the airplane and presents correct turn indications.

**AF 55-3358 & on.*
***Airplanes modified by TCTO 1F-102-759.*

WARNING

On other airplanes the turn-and-slip indicator presents erroneous turn indications due to mounting the instrument flush with the inclined instrument panel. This mounting results in inclination of the longitudinal axis of the turn-and-slip indicator with respect to the roll axis of the airplane, which causes false turn indications. For example, if the instrument were mounted in the floor of the cockpit, it would no longer indicate rate of turn but would indicate rate of roll (due to gyroscopic action this would be indicated by a left needle deflection when rolling to the right, or vice versa.) The inclined installation combines both roll and turn indications to the instrument which cause false indications. Since a turn is induced by first rolling into the turn, the turn needle will indicate roll (in the opposite direction), and once the turn is established the needle will reverse its movement and indicates a turn in the proper direction. The faster the rate of the roll the greater the erroneous indication in the opposite direction. In the event the attitude indicator and the radar scope reference are inoperative, and the turn-and-slip indicator is used as the primary flight instrument, excessive rate of roll should be avoided when establishing the desired bank angle.

The turn-and-slip indicator is powered by the 28-volt dc essential bus.

STANDBY COMPASS

A conventional magnetic compass, suspended from the canopy, is furnished for navigation purposes in the event of failure of navigation equipment or electrical system failure. Illumination of a light within the compass case is controlled by a switch on the right side of the cockpit. A standby compass correction card and holder are located on the left side of the cockpit, above the console and aft of the throttle quadrant.

OUTSIDE AIR TEMPERATURE GAGE

An outside air temperature gage is installed on early airplanes* and is located on the right-hand console. The indicator has a range from −50° to +50°C. It is an electrical resistance type instrument and measures ambient temperature. The temperature sensing unit is a resistance bulb located in the engine inlet duct. The temperature indicating system also receives power from the 28-volt dc essential bus.

AF 53-1791 thru 54-1383.

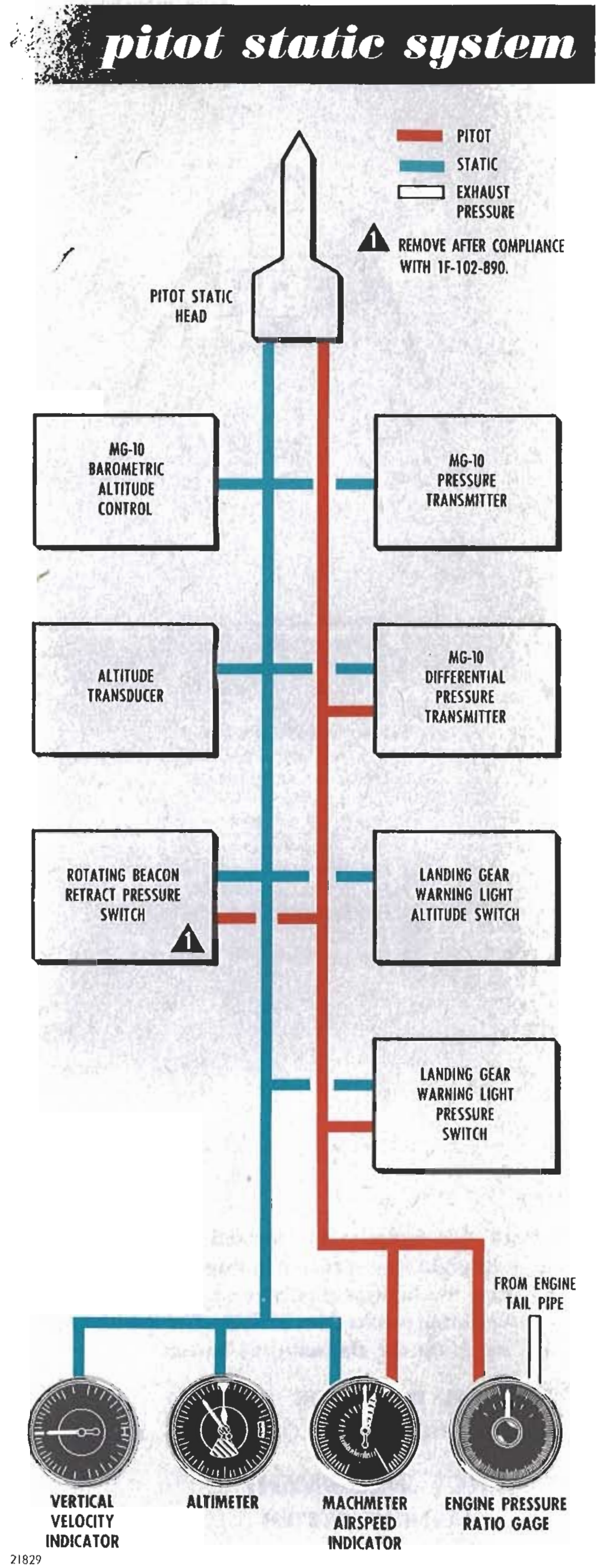

Figure 1-25

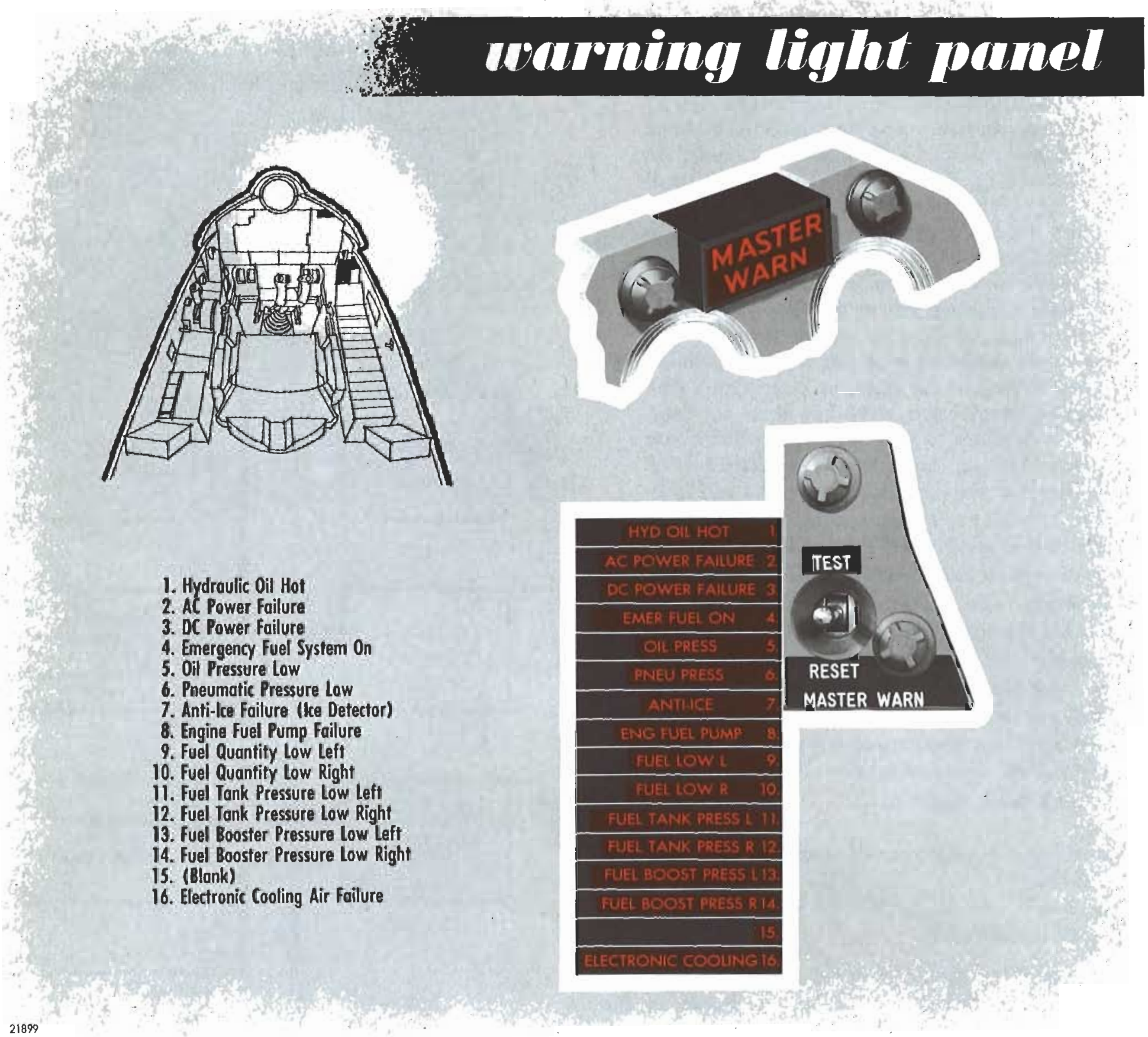

Figure 1-26

CLOCK

The clock (6, figure 1-4), located on the instrument panel, is an eight-day spring-winding 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.

DIRECTIONAL INDICATOR (SLAVED)

Refer to NAVIGATION EQUIPMENT, Section IV.

EMERGENCY EQUIPMENT

MASTER WARNING SYSTEM

A warning light panel (figure 1-26), located at the forward end of the right-hand console has 16 individual amber warning lights which indicate malfunctions or failures of various systems and equipment. Illumination of any individual light also illuminates an amber master warning light on the main instrument panel, within the pilot's normal line of vision. Once illuminated, the master warning light can be extinguished with a master warning test and reset switch. 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 flight instrument lights are on, if the thunderstorm 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, or fire and overheat warning light.

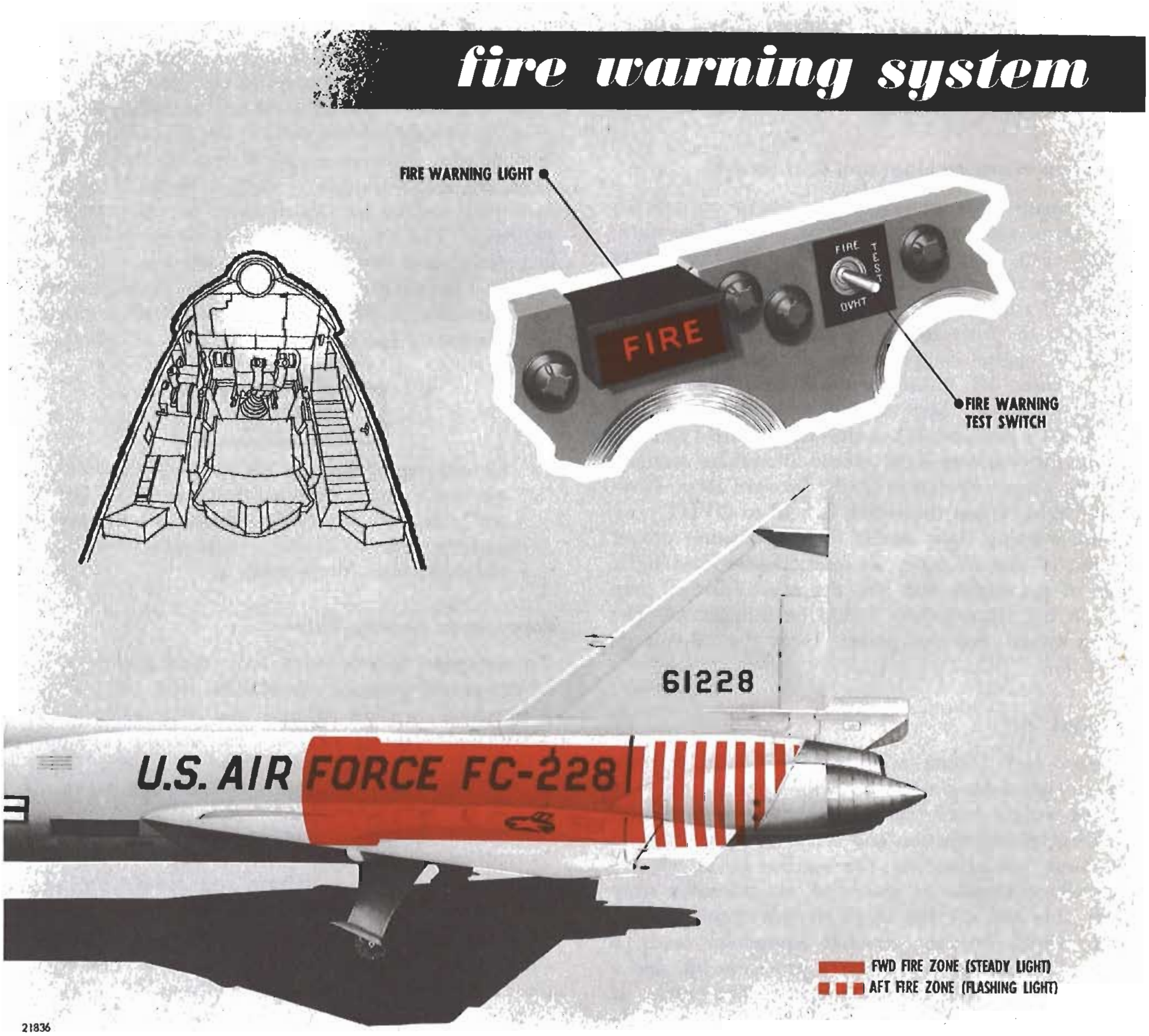

Figure 1-27

Master Warning Test and Reset Switch

A three-position master warning test and reset switch (figure 1-26), placarded "Master Warn," is located at the right of the master warning light panel. It has positions TEST and RESET, and is spring-loaded to a center (OFF) position. When held in the TEST position all lights in the master warning system should illuminate.

Note

Placing the switch to TEST is a functional check of lights only; not of complete warning circuits.

In the event of a system malfunction, both individual and master warning lights will illuminate. The RESET position is used to extinguish and reset the master warning light only so that it will relight again in case of a subsequent malfunction. The master warning system receives power from the 28-volt dc essential bus.

ENGINE FIRE WARNING SYSTEM

A fire warning system (figure 1-27) is installed to detect and indicate fire and overheat conditions in the forward or aft engine compartments. The forward engine compartment (which includes the compressor and accessory sections) and the aft compartment (which includes the combustion chambers and the turbine) have separate detector loops and detector units. The detector loops are of resistance-type coaxial construction. A hot spot anywhere along the length of the loop completes the circuit (between a center conductor and the outside tube) which the detector unit senses. A fire burning through the loop would not affect its operation. Both forward and aft detector networks are electrically connected to a warning light in the cockpit.

Engine Fire Warning Light and Test Switch

An abnormally high temperature in the forward or aft engine compartments is indicated by the red fire warning light (13, figure 1-4), located on the instrument panel. When illuminated the light displays "FIRE." To identify the zone of the fire, the light will illuminate steadily for the forward engine compartment and flash for the aft engine compartment. A three-position test switch (figure 1-8), located at the right of the warning light, is placarded "Test" and is spring-loaded to the center (OFF) position. When the switch is held to FIRE position, the warning light should illuminate steadily, indicating proper operation of the forward loop, detector, and light. When the switch is held to OVHT position, the warning light should flash, indicating proper operation of the aft loop, detector, flasher, and light. Failure of the flasher unit will not affect forward loop operation but the aft loop would be inoperative. The warning system receives power from the 28-volt dc essential bus.

SURVIVAL KIT

A survival kit* (figure 1-28) is furnished with some airplanes and is designed to fit in the ejection seat. The kit, fitted with a rubber cushion which snaps to the kit, is of fiberglas construction and serves as a seat cushion with a back-type parachute. The survival kit attaches to the parachute harness by means of an adjustable strap on each side and consists of an oxygen regulator, two bailout oxygen bottles, personal equipment leads, a one-man life raft (if required), a provision kit, and a reflector on back of the survival kit lid. The bundle of personal equipment leads is inserted into a receptacle in the right rear corner of the kit. The receptacle contains connections for oxygen, partial pressure suit, mask defog, communications leads, and the green knob for bailout bottle manual actuation. The oxygen regulator in the kit controls all oxygen used by the pilot (breathing and partial pressure suit) both in the event of ejection and during normal flight and ground operation. An oxygen system press-to-test button is located on the front panel of the kit (refer to LIQUID OXYGEN SYSTEM, Section IV). In event of ejection, oxygen from the bailout bottles provides breathing oxygen and pressure suit oxygen for a minimum of 12 minutes. The bailout bottles utilize one pressure gage which is visible through a small window in the rear portion of the kit. Pressure in the bottles should read 1800 psi and should be checked prior to each flight. A bailout bottle reducer lowers the pressure to 40-60 psi and delivers this pressure to the regulator. The bailout bottles are actuated automatically as the ejection seat leaves the airplane during ejection, or may be manually actuated during flight whenever the bailout bottles are required. Manual actuation of the bottles is accomplished by pulling the round green knob attached to a cable in the personal equipment lead bundle. Manual actuation may be desired at any time the ship's oxygen system is depleted or is not supplying oxygen for breathing or partial pressure suit operation. The life raft is inflated by an automatically actuated carbon dioxide bottle mounted on the inside of the kit. The provision kit is a waterproof packet strapped in the bottom of the survival kit and should contain the items necessary for survival in the area of operation.

CAUTION

Do not step or stand on the survival kit as damage to the case or personal leads can result. Dirt and grease can also be introduced into the oxygen regulator, rendering it inoperative and possibly creating a fire hazard.

Emergency Airway Valve

An emergency airway valve, located on the reverse side of the partial pressure suit bladder lead (figure 1-28), is provided on some survival kits. This valve is being deactivated** in all airplanes and should be capped.

Survival Kit Emergency Release Handle

The yellow emergency release handle (figure 1-28) is located on the right side of the survival kit and is placarded "Emergency Release" on some airplanes or "Kit Release" on other airplanes. The handle is hinged at the rear, and when raised following ejection and chute deployment, will release the kit. The handle also releases the kit for quick ground egress. When fully raised, the handle will separate from the kit. Raising the handle causes the following to occur simultaneously after ejection and parachute deployment:

- Disconnects kit from personal equipment lead bundle.
- Lower portion of kit drops but remains attached to parachute harness by lanyard.
- Actuation of carbon dioxide bottle for life raft inflation.

For quick ground egress, raising the handle will cause the following to occur:

- Disconnects kit from personal equipment lead bundle.
- Releases parachute attaching wedges, completely separating the pilot from the survival kit.

*AF 56-1275 thru -1316, -1332 & on, & airplanes modified by TCTO 1F-102-642.

**In accordance with TCTO 1F-102-678.

survival kit

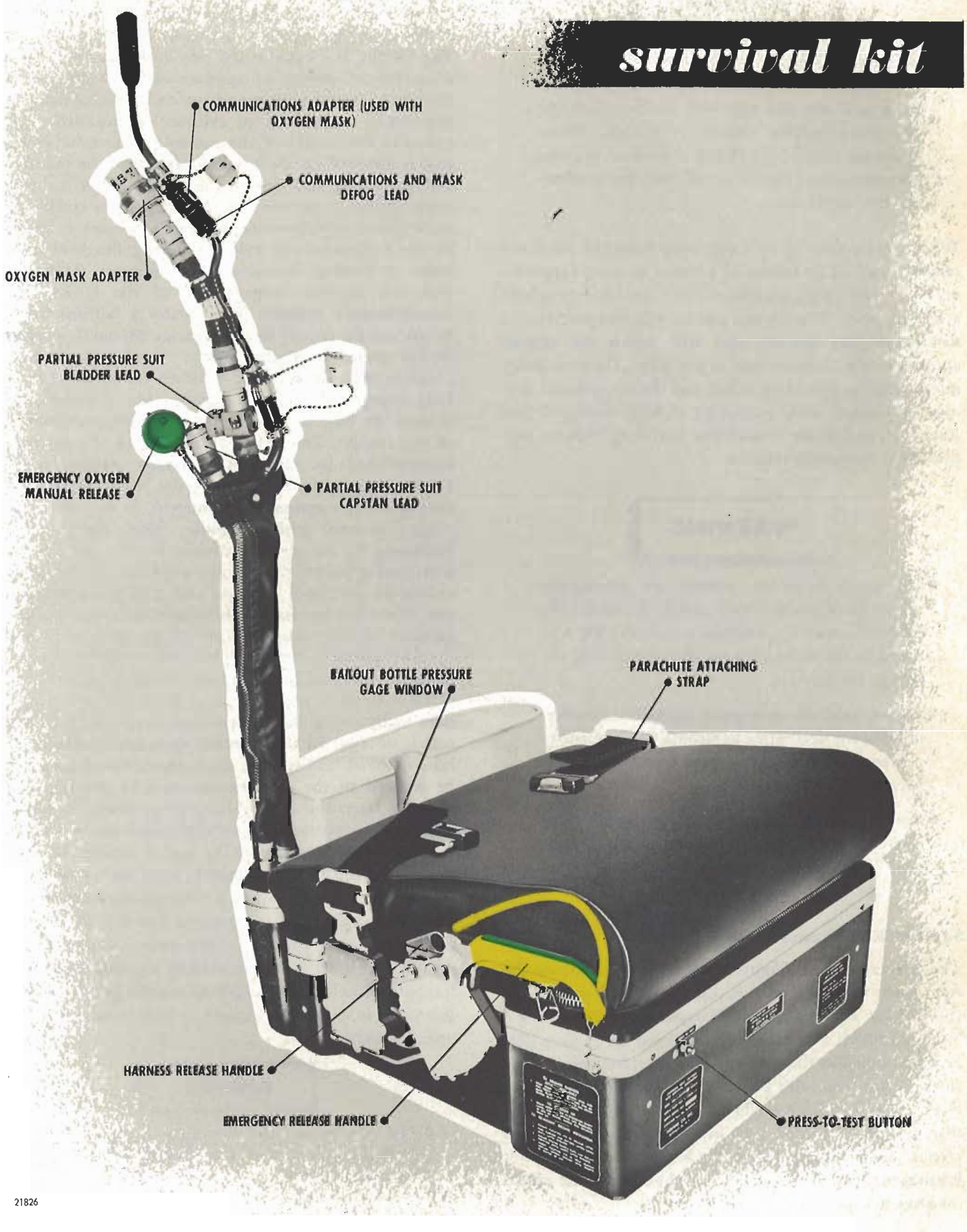

Figure 1-28

WARNING

On survival kits — Part No. 8-09353-7 — the 25-foot lanyard will remain attached to the parachute and the life raft will inflate when the emergency release handle is pulled. When equipped with the 8-09353-7 survival kit, the parachute should be removed prior to abandoning the airplane.

When released the kit will fall away from the pilot and remain attached by means of a nylon lanyard (approximately 25 feet long) attached to the right-hand parachute attaching strap. The life raft and kit will remain attached throughout the descent and will strike the surface approximately 25 feet ahead of the pilot. The emergency release handle should be raised and the kit released during the descent after parachute is fully deployed and stabilized, and a safe altitude for breathing without supplemental oxygen is reached.

WARNING

- On some airplanes, pulling the emergency release handle will always inflate the rubber life raft (if installed), whether or not the kit is in the seat. On other airplanes*, the raft will not inflate in the seat.
- Do not raise the emergency release handle during descent until after parachute deployment to prevent the kit or the lanyard from fouling the parachute.
- Do not raise the 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.

Survival Kit Harness Release Lever

The harness release (figure 1-28) is a lever aft of the emergency release handle on the right-hand side of the survival kit. It is placarded "Harness Release" and has a hole into which a finger may be inserted for raising. The lever is hinged at the rear and, when raised, releases the survival kit from the parachute attaching straps. This lever is designed for use when an emergency escape other than ejection is desired, such as escape from the airplane after a crash landing. When the lever is raised the pilot is released from all connections on the survival kit except the personal leads.

**Airplanes modified by TCTO 1F-102-642 or TCTO 1F-102-679.*

CANOPY

The metal-reinforced plexiglas canopy (7, figure 1-2) is a clamshell type which is manually opened and closed. The canopy is hinged at the rear and opens to provide a maximum opening of approximately 40 degrees. To facilitate manual raising and lowering of the canopy, a pneumatic counterbalance cylinder is installed which equalizes the weight of the canopy. An exterior canopy grip is installed on the forward left side of the canopy to aid in opening and closing the canopy when the pneumatic system is not serviced. A clamping or cinch-down force (from high-pressure pneumatic system) is applied by the counterbalance cylinder during the final movement of closing the canopy to compress the canopy seal and facilitate engagement of the latches. The counterbalance cylinder incorporates a ballistic charge to jettison the canopy in emergencies. Manually operated latches secure the canopy in the closed position and a warning light illuminates whenever the latches are not fully engaged. An inflatable canopy seal is incorporated around the base of the canopy to permit pressurization of the cockpit. Emergency jettisoning of the canopy is accomplished by raising the canopy jettison handle (located on the forward end of the left-hand armrest), raising either ejection seat handgrip, or by pulling the canopy external jettison handle. When the canopy is jettisoned by the above methods, it arms the ejection seat arming initiator. On some airplanes, if the canopy cannot be jettisoned in flight and it is manually raised and allowed to be removed by windblast, the seat arming initiator becomes armed. However, on other airplanes*, the seat arming initiator will not arm when the canopy is manually released in flight, and ejection from the airplane will not be possible. On early airplanes** to facilitate taxiing with the canopy open, a canopy hold-open rod is secured to the forward right-hand edge of the canopy. With the canopy open, the hold-open rod can be secured to the airplane structure to prevent small airloads from shifting the canopy position. The hold-open rod is secured by a spring-loaded sleeve which is raised to fasten or unfasten the rod. A spring clip, along the base of the canopy, is used to stow the rod when not in use. On later airplanes† a solenoid-operated valve is installed in the pneumatic pressure line that connects the upper and lower portions of the canopy counterbalance cylinder. This valve is controlled by a pushbutton switch located on the internal left-hand canopy grip, and when the valve is closed the canopy is locked in any desired open position.

CAUTION

- The canopy is not designed to be opened in flight as it would be completely removed by the wind blast.

**AF 55-3427 thru 56-1429 unless modified by TCTO 1F-102A-565.*
***AF 53-1791 thru 54-1400.*
†AF 54-1401 & on, & airplanes modified by TCTO 1F-102A-571.

- If the canopy is open during taxi operations, observe canopy limit speeds. The canopy hold-open rod must be used on airplanes which do not have a canopy hold button. Keep hands away from canopy sill unless the canopy support tool is in place.

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 pushed fully in. The initial pull of the canopy latch handle dumps pressure from the canopy seal.

CANOPY GROUND SUPPORT TOOL

A removable canopy ground support tool (figure 1-23) is provided for use during ground operations. The tool fits between the canopy and canopy sill on the left side of the airplane.

WARNING

The canopy ground support tool 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 tool is not in position, special care should be taken to keep the area between the canopy and the canopy sill clear.

CANOPY COUNTERBALANCE CYLINDER

To facilitate manual raising and lowering of the canopy a pneumatic cylinder is installed and connects to the canopy forward of the canopy hinge line. The cylinder (in counterbalance condition) has air pressure on each side of a piston which, due to larger area on the lower side, applies an upward force that equalizes the weight of the canopy. The counterbalance cylinder is pressurized by the high-pressure pneumatic system through a regulator which reduces the operating pressure to approximately 1500 psi. During the final movement of closing the canopy a valve is mechanically opened which automatically relieves pressure from the lower side of the piston. Then the existing pressure on the upper side of the piston applies a downward force (cinch-down force) on the canopy which compresses the canopy seal and enables the canopy latches to be manually engaged.

WARNING

The final movement (one to two inches) of closing the canopy is rapid and is cinched down with considerable force; therfore, care should be taken that the area beneath the canopy is clear to prevent personal injury or damage to equipment.

To open the canopy it is necessary to push the canopy latch handle fully in and return to fully out position or depress the canopy external release button which will mechanically open a valve allowing high pressure (at a regulated value) to enter the lower side of the counterbalance cylinder restoring the counterbalance condition, and permitting the canopy to be raised manually. On later airplanes* a solenoid-operated valve is installed in the counterbalance air line between the upper and lower sides of the counterbalance cylinder to lock the canopy in any desired open position for taxi operations. This valve is spring-loaded to the open position; therefore, whenever power is being supplied to the 28-volt dc essential bus and the canopy is opened, the valve will close forming an air lock within the upper and lower portions of the counterbalance cylinder which will lock the canopy in its existing position. To facilitate desired movement of the canopy, a canopy hold button is installed on the internal left-hand canopy grip. This button, when depressed, will open the circuit to the solenoid-operated valve allowing the canopy to be manually raised or lowered. In addition, to facilitate complete closure and initial opening of the canopy from the outside when 28-volt dc power is connected, a limit switch is installed to disconnect dc power to the canopy hold button which permits free canopy movement for approximately six inches from the fully closed position. Depletion of high-pressure pneumatic pressure during flight will prevent restoring complete counterbalance action; however, the canopy can be raised manually with partial aid of counterbalance action.

CANOPY EXTERIOR RELEASE BUTTON

When the canopy is manually closed (on a parked airplane) the counterbalance cylinder applies the cinch-down force to seal the cockpit from the elements. To gain access again, it is necessary to restore counterbalance action before the canopy can be raised. This is accomplished by an exterior release button (figure 2-2) placarded "Push to Release Canopy" located on the outside of the left-hand intake duct near the base of the canopy. Pushing the button mechanically controls a pneumatic valve which supplies pneumatic pressure to the counterbalance cylinder thus relieving the cinch-down force. The canopy can then be raised manually.

**AF 54-1401 & on.*

- If the canopy is open during taxi operations, observe canopy limit speeds. The canopy hold-open rod must be used on airplanes which do not have a canopy hold button. Keep hands away from canopy sill unless the canopy support tool is in place.

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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 pushed fully in. The initial pull of the canopy latch handle dumps pressure from the canopy seal.

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WARNING

The canopy ground support tool 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 tool is not in position, special care should be taken to keep the area between the canopy and the canopy sill clear.

CANOPY COUNTERBALANCE CYLINDER

To facilitate manual raising and lowering of the canopy a pneumatic cylinder is installed and connects to the canopy forward of the canopy hinge line. The cylinder (in counterbalance condition) has air pressure on each side of a piston which, due to larger area on the lower side, applies an upward force that equalizes the weight of the canopy. The counterbalance cylinder is pressurized by the high-pressure pneumatic system through a regulator which reduces the operating pressure to approximately 1500 psi. During the final movement of closing the canopy a valve is mechanically opened which automatically relieves pressure from the lower side of the piston. Then the existing pressure on the upper side of the piston applies a downward force (cinch-down force) on the canopy which compresses the canopy seal and enables the canopy latches to be manually engaged.

WARNING

The final movement (one to two inches) of closing the canopy is rapid and is cinched down with considerable force; therfore, care should be taken that the area beneath the canopy is clear to prevent personal injury or damage to equipment.

To open the canopy it is necessary to push the canopy latch handle fully in and return to fully out position or depress the canopy external release button which will mechanically open a valve allowing high pressure (at a regulated value) to enter the lower side of the counterbalance cylinder restoring the counterbalance condition, and permitting the canopy to be raised manually. On later airplanes* a solenoid-operated valve is installed in the counterbalance air line between the upper and lower sides of the counterbalance cylinder to lock the canopy in any desired open position for taxi operations. This valve is spring-loaded to the open position; therefore, whenever power is being supplied to the 28-volt dc essential bus and the canopy is opened, the valve will close forming an air lock within the upper and lower portions of the counterbalance cylinder which will lock the canopy in its existing position. To facilitate desired movement of the canopy, a canopy hold button is installed on the internal left-hand canopy grip. This button, when depressed, will open the circuit to the solenoid-operated valve allowing the canopy to be manually raised or lowered. In addition, to facilitate complete closure and initial opening of the canopy from the outside when 28-volt dc power is connected, a limit switch is installed to disconnect dc power to the canopy hold button which permits free canopy movement for approximately six inches from the fully closed position. Depletion of high-pressure pneumatic pressure during flight will prevent restoring complete counterbalance action; however, the canopy can be raised manually with partial aid of counterbalance action.

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**AF 54-1401 & on.*

CANOPY LATCH HANDLE

The T-shaped canopy latch handle (26, figure 1-4), located on the right side of the instrument panel, is used to mechanically engage and release the canopy latches. When the canopy is completely closed, pushing the handle fully in will engage the canopy latches and allow inflation of the canopy seal. A warning light above the handle will go out when the latches are fully engaged. Initial movement of the handle from the fully in position causes the canopy unlocked warning light to illuminate. Subsequent aft movement of the handle unlatches the canopy latch hooks and, when the handle is fully out (approximately eight inches), relieves canopy seal pressure and places the pneumatic control valve in the counterbalance condition. The canopy can then be raised manually.

CANOPY HOLD BUTTON

The canopy hold button (figure 2-2) located on the internal left-hand canopy grip, is spring-loaded to the out position which supplies power to the solenoid-operated valve, located on the canopy counterbalance cylinder. Whenever power is being supplied to the 28-volt dc essential bus and canopy movement is desired, it is necessary to depress this button permitting free flow of pneumatic pressure within the counterbalance cylinder, and allowing the canopy to be manually raised or lowered. The canopy hold button receives power from the 28-volt dc essential bus whenever the canopy is opened in excess of approximately six inches.

CANOPY JETTISON HANDLE

The yellow canopy jettison handle (figure 1-29) may be used to jettison the canopy by a ballistic charge. The handle is located on the forward end of the left armrest and incorporates a safety latch which is released by pressing a button located on the outboard end of the handle. Raising the handle will unlatch and jettison the canopy with a ballistic charge. This system functions through the same mechanisms as used by the ejection seat handgrips and should be used when it is desired to jettison the canopy without exposing the handgrip triggers.

EJECTION SEAT HANDGRIPS

The canopy can be jettisoned by raising either the right or left ejection seat handgrips. Refer to EJECTION SEAT, this Section.

CANOPY EXTERNAL RELEASE HANDLE

In the event of rescue, if the canopy fails to jettison or if the presence of fuel fumes makes jettisoning inadvisable, the canopy may be unlatched and manually raised. The external canopy release handle is located below the right-hand windshield and is covered by a small access door (figure 3-6). When the access door is opened, the spring-loaded handle pops out. Pushing the handle aft releases canopy cinchdown pressure and unlatches the canopy which may then be raised manually.

CANOPY EXTERNAL JETTISON HANDLE

During emergency conditions, the canopy can be jettisoned from outside the airplane for emergency rescue by the external jettison handle (figure 3-6). The handle is located in a small compartment on the left side of fuselage forward of the wing intersection. Removing the access door exposes the handle which has approximately six feet of excess cable, allowing the operator to stand at a safe distance from the airplane before jettisoning the canopy. Pulling the cable outboard six feet, then applying a steady pull of approximately 30 pounds, initiates the sequence by mechanically pin-firing the canopy initiator (figure 1-30) which gas-fires the thruster. This releases the canopy latches and cinchdown pressure and gas-fires the canopy remover initiator which in turn fires the ballistic cartridge located at the base of the canopy remover cylinder. The expanding gases from the ballistic cartridge open the canopy, then jettison it. The canopy should travel up and aft and will probably strike the tail. The trailing wire will fire the seat arming initiator and arm the seat ejection system, preparing it for operation. Pulling the external jettison handle actuates the same mechanisms as does the canopy jettison handle or the ejection seat handgrips.

WARNING

The external canopy jettison handle should not be pulled except for emergency reasons. The ground safety lock pin which is installed in the ejection seat right handgrip linkage prevents jettisoning of the canopy from the cockpit but will not prevent jettisoning if the external handle is pulled.

CANOPY UNLOCKED WARNING LIGHT

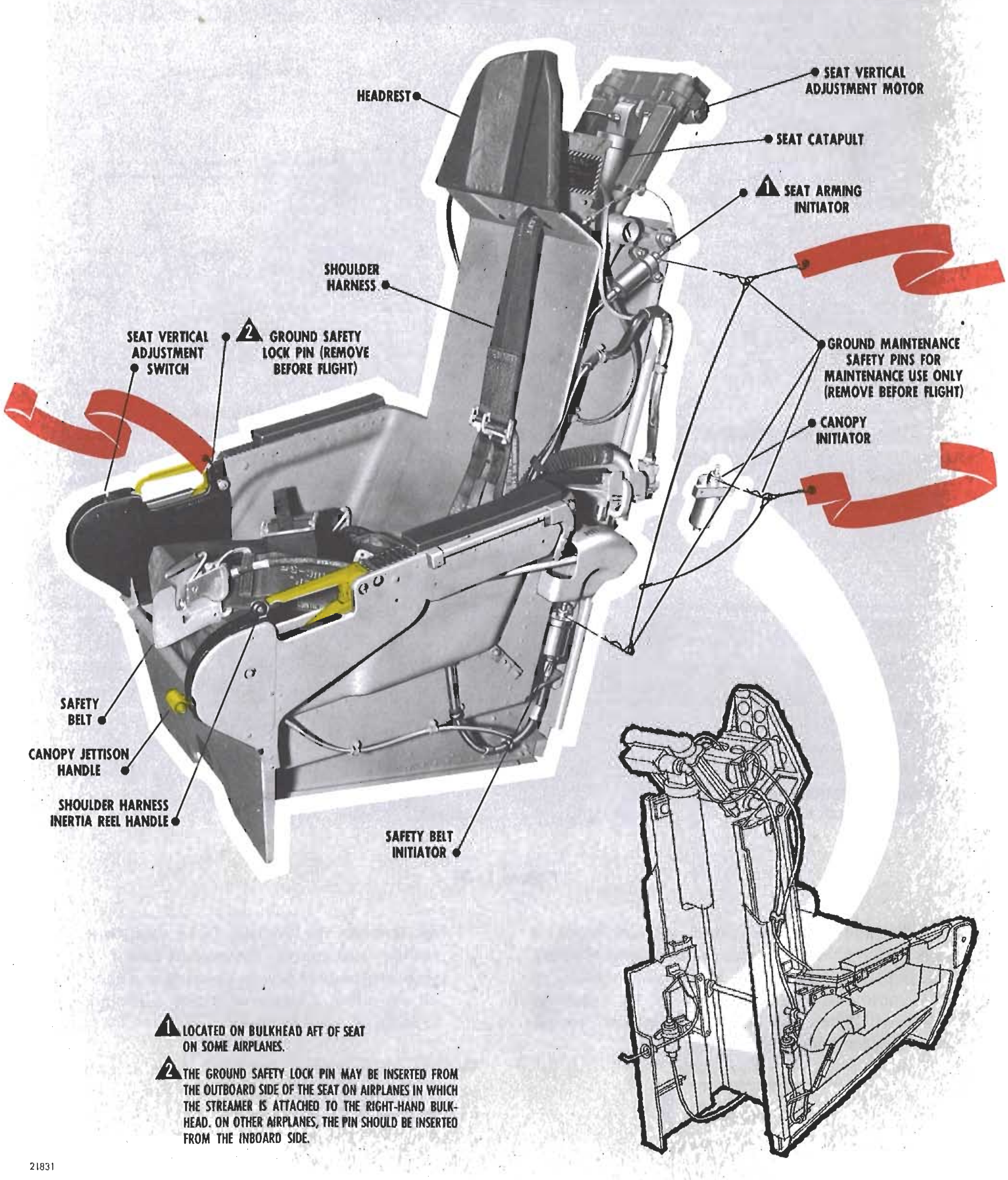
The red canopy unlocked warning light (22, figure 1-4), located above the canopy latch handle, illuminates and displays "CANOPY UNLOCKED" when both canopy latches are not fully engaged. The warning light receives power from the 28-volt dc essential bus.

EJECTION SEAT

The ejection seat (figure 1-29) permits bailout at high speeds and any flight attitude. A catapult fired by a ballistic charge supplies the necessary force to eject the seat and pilot upward from the airplane. Some airplanes are equipped with the M-3 ballistic catapult and other airplanes* are equipped with the MK-1 rocket catapult. The seat has an automatic-opening safety belt and accommodates a back-type parachute. A seat cushion is furnished; however, a one-man life raft or a survival kit may be used instead of the seat cushion. A rigid seat style survival and oxygen kit container is used on some

**AF 56-1513 & on.*

ejection seat (typical)



1 LOCATED ON BULKHEAD AFT OF SEAT ON SOME AIRPLANES.

2 THE GROUND SAFETY LOCK PIN MAY BE INSERTED FROM THE OUTBOARD SIDE OF THE SEAT ON AIRPLANES IN WHICH THE STREAMER IS ATTACHED TO THE RIGHT-HAND BULKHEAD. ON OTHER AIRPLANES, THE PIN SHOULD BE INSERTED FROM THE INBOARD SIDE.

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Figure 1-29

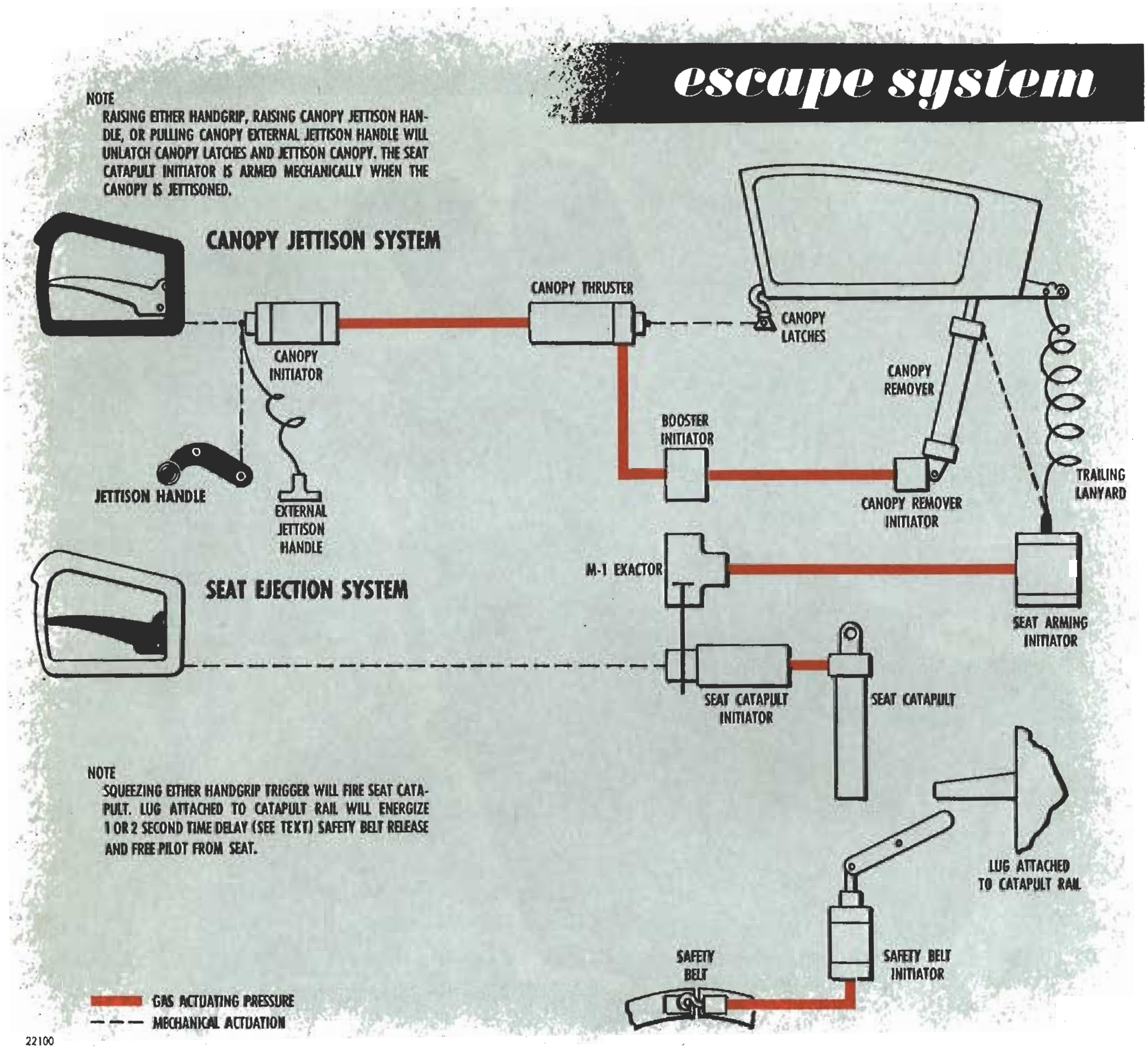


Figure 1-30

airplanes. No additional parachute support block is required as the survival kit container has its own support. On other airplanes, either the MC-2 seat cushion, or the MD-1 contoured seat style survival kit container should be used. When the MC-2 seat cushion is used, a parachute support block is required.

WARNING

Do not use the A-5 cushion, or any similar sponge rubber cushion, when equipped with a one-man life raft or survival kit. If ejection becomes necessary, serious spinal injuries can result when the ejection force compresses the cushion and enables the seat to gain considerable momentum before exerting a direct force on the pilot. Chance of injury during forced landing is also increased.

Vertical adjustment of the seat can be accomplished by an electric actuator. The shoulder harness inertia reel, located on the back of the seat, locks automatically when a rapid pull (equivalent to two to three g's deceleration) force is exerted on the harness assembly. The inertia reel is equipped with a manual control. On airplanes not equipped with a survival kit, the pilot's mask defog, headset and microphone leads, oxygen and anti-g suit hoses are attached to a disconnect unit on the lower left side of the seat and disconnect from the airplane automatically when the seat is ejected. On airplanes equipped with the survival kit, the pilot's mask defog, headset, microphone leads, and oxygen hoses are attached to the right-rear of the survival kit. The anti-g suit and vent are attached to a disconnect unit on the lower left side of the seat. All connections automatically disconnect from the airplane when the seat is ejected. Elbow guards on the armrests are folded down out of the way during normal flight but are raised automatically to a protective position when the seat handgrips are raised prior to ejection. The ejection sequence is so designed that the seat cannot be ejected until the canopy has been jettisoned.

WARNING

Ground maintenance safety pins (figure 1-29) are inserted in the canopy initiator, the seat arming initiator and the safety belt initiator during maintenance operations. A ground safety lock pin is also inserted in the right-hand seat ejection grip. On some airplanes a fitting is provided at the right-hand side of the seat on the bulkhead for stowing the ejection seat safety lock pin. If any of the pins are left in place, canopy jettisoning, seat ejection and/or automatic opening of the safety belt is prevented.

AUTOMATIC-OPENING SAFETY BELT

An automatic-opening safety belt (figure 1-31) is provided to extend the maximum and minimum safe altitudes for seat ejection. In a high-altitude bailout, the automatic belt, combined with the automatic parachute, delays parachute deployment until a safe altitude is reached. In a low-altitude bailout, the automatic system reduces the altitude required for safe ejection by reducing the time required for separation from the seat and parachute deployment. 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. Manual opening of the belt prior to ejection precludes actuation of the automatic timing and release mechanism on the parachute and permits immediate separation from the seat upon contact with the airstream. If immediate separation occurs, the pilot is subjected to far greater deceleration forces and the wind shock could open the parachute pack prematurely. Therefore, a delay of one second is incorporated into the automatic belt release to allow a more satisfactory deceleration and provide wind blast protection for the parachute. Release of the automatic-opening belt is accomplished either by manual operation or by gas pressure from an automatically controlled initiator. The initiator supplies approximately 1500 psi gas pressure through a high-pressure hose and actuates a piston inside the belt release,

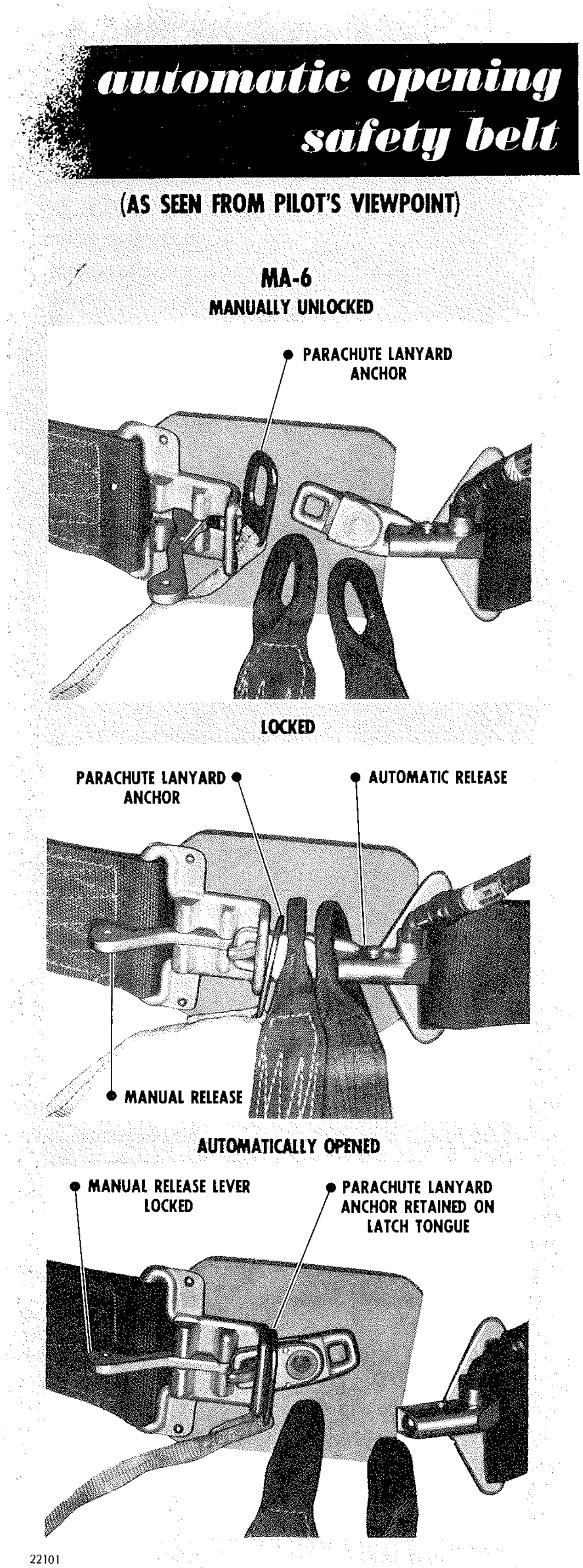


Figure 1-31

zero delay lanyard hook attachment

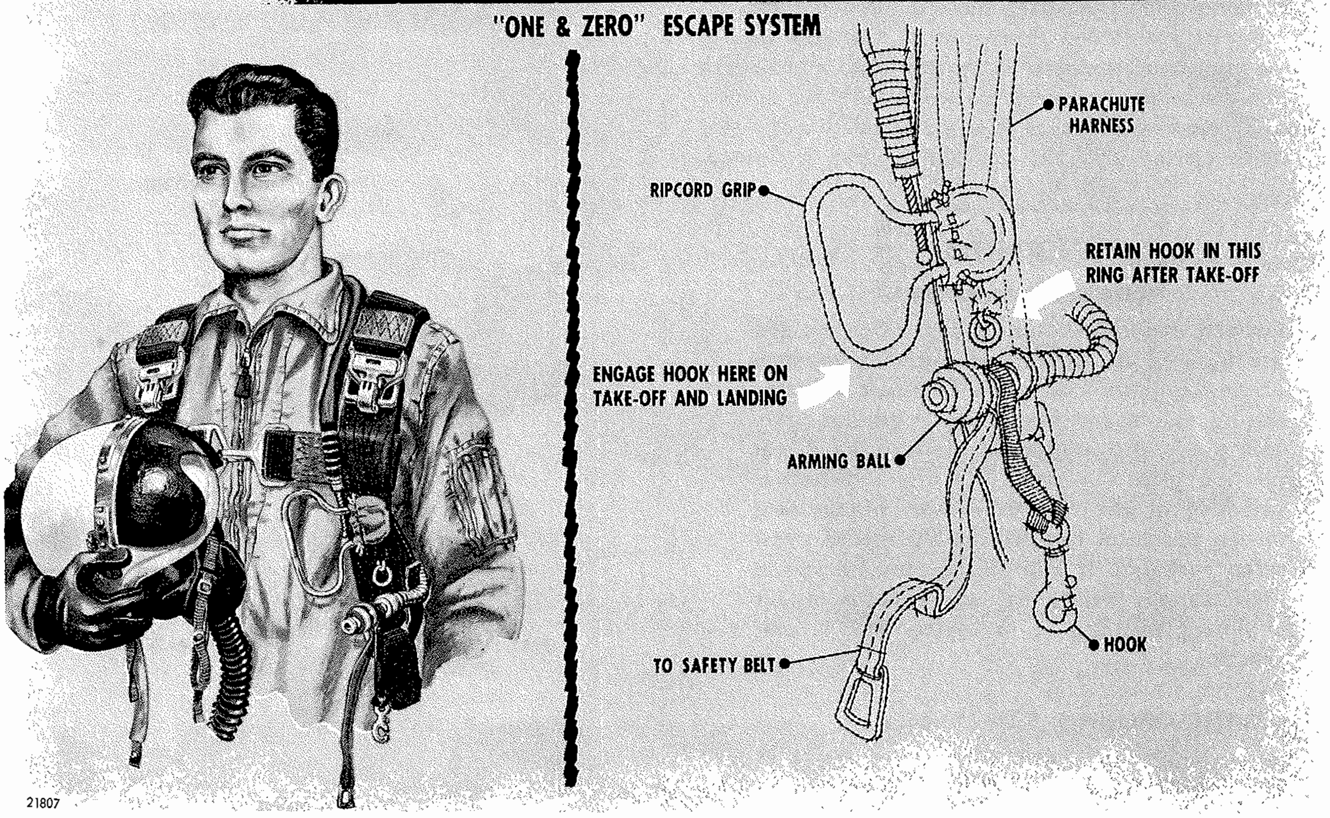


Figure 1-32

separating the belt at the latch. The release incorporates a means of attaching a lanyard which connects the belt with the static line to the timer of the automatic parachute. The MA-6 type belt connects the parachute release mechanism by inserting the safety belt tongue through a metal lanyard anchor. To close the MA-6, the shoulder harness loops must be placed on the safety belt tongue before the lanyard anchor. When the initiator is fired, the portion of the lap belt tongue holding the anchor is retained in the locked position and the portion holding the shoulder harness loops separates to allow separation from the seat and actuation of the parachute release. After the parachute release has been actuated by any of the automatic belts, a preset delay of two seconds must elapse, then chute deployment begins if ejection was at a safe altitude. In the case of high altitude ejection, the aneroid action of the parachute release will delay deployment until a preselected safe altitude is reached by free falling, then deployment begins after a two-second delay. Automatic operation of the safety belt systems can be overridden at any time by manual operation.

WARNING

If the automatic-opening safety belt is opened manually, the automatic parachute release will not be actuated and the parachute ripcord must be pulled manually.

"ONE-AND-ZERO" ESCAPE SYSTEM

A system incorporating a one-second safety belt delay and a zero-second parachute delay ("one-and-zero" system) is provided to improve low-altitude ejection seat escape capability. This system makes use of a detachable lanyard that connects the parachute timer knob to the parachute ripcord. (A sketch of the hook, figure 1-32, depicting the "hooked" and "unhooked" conditions is provided for illustrative purposes only.) At very low altitudes and airspeeds, this lanyard must be connected, thus providing parachute actuation immediately after separation from the ejection seat. At other altitudes and airspeeds, the lanyard must be disconnected from the ripcord, thus allowing the parachute timer to actuate the parachute below the critical parachute opening speed and below the parachute timer altitude setting. A ring attached to the parachute harness is provided for stowage of the lanyard hook when it is not connected to the parachute ripcord.

Note

See figure 3-3, Section III for the Zero Delay Lanyard Engagement Requirements Chart.

EJECTION SEAT HANDGRIPS

Raising either seat handgrip (figure 1-29) will lock the shoulder harness, raise the elbow guards, and jettison the canopy. The linkage is such that either handgrip may be raised without raising the other.

WARNING

- The canopy jettison and seat ejection system is safetied by a ground safety lock pin inserted through the right handgrip linkage. This pin must be removed before flight and replaced after flight by the pilot. Be certain the canopy jettison handle is in the detent position with the release button out when the pin is inserted to prevent accidentally jettisoning the canopy.
- The ground safety lock pin does not safety the canopy jettison system if the external canopy jettison handle is pulled.

When either handgrip is raised to the fully up position (handgrips will lock in up position exposing the seat catapult triggers) mechanical linkage will lock the shoulder harness, raise the elbow guards and fire an initiator unit. The expanding gases produced are routed to a thruster unit which also fires to disengage the canopy latches; and these gases which are bypassed fire an additional initiator unit. The resulting expanding gases are then routed to energize the canopy remover which will fire and jettison the canopy. A canopy trailing wire attached to the canopy fires an initiator unit and the expanding gases produced are routed to remove a pin, thus arming the seat catapult initiator.

SEAT CATAPULT TRIGGERS

The seat catapult triggers (figure 1-29) are located within the ejection seat handgrips, and are accessible only when the handgrips are in the fully up position. Squeezing either trigger fires an initiator. The expanding gases produced are routed to the seat catapult which fires the catapult, ejecting the seat.

Note

The seat ejection system is dependent upon the canopy jettison system. If raising the handgrips does not fire the canopy, the seat catapult initiator will not be armed; therefore, it cannot be fired by squeezing the triggers.

SEAT VERTICAL ADJUSTMENT SWITCH

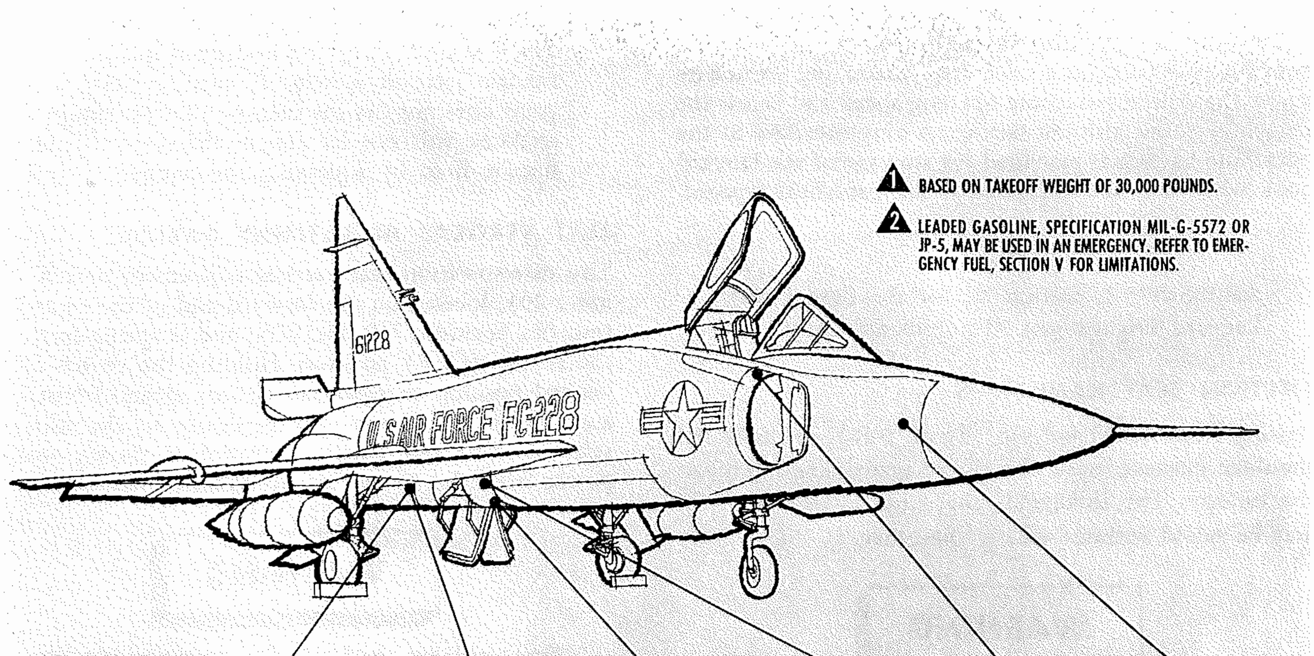
The three-position seat vertical adjustment switch (figure 1-29), located on the forward end of the right armrest, has positions UP, DOWN, and is spring-loaded to the center (OFF) position. Holding the switch to the desired position energizes an electric motor-driven actuator which moves the seat vertically in the direction selected. The seat vertical adjustment system receives power from the 28-volt dc nonessential bus.

WARNING

On airplanes without the survival kit, the oxygen lead may separate from the quick-disconnect fitting when the seat is adjusted to the fully up position, and must be checked after seat adjustment.

SHOULDER-HARNESS INERTIA REEL HANDLE

Manual control of the shoulder-harness inertia reel is provided by the shoulder-harness inertia reel handle (figure 1-29), located on the forward end of the left handgrip. The handle is placarded with an arrow pointing forward to the LOCKED position, and has overcenter detents which restrain it from slipping out of either LOCKED or aft (UNLOCKED) position. When the handle is in the UNLOCKED 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 LOCKED and then returned to the UNLOCKED position. When the handle is in the LOCKED position, the reel harness cable is manually locked so that the pilot is prevented from bending forward. The LOCKED position is used only when a crash landing is anticipated and provides an added safety precaution over and above that of the automatic reel lock. If the harness is automatically or manually locked while the pilot is leaning forward, this 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



NAME	Fuel Flask (Combustion Starter)	Constant Speed Drive Accumulator	Hydraulic System Reservoirs	Hydraulic System Accumulators (Filler Valves)	Survival Kit Emergency Oxygen Bottles	Radome Anti-Ice
REPLENISHING AGENT	Engine Fuel	Dry Air Or Nitrogen	Red Hydraulic Fluid	Dry Nitrogen Or Clean Dry Air	Gaseous Oxygen	Water (40%) Glycol (60%) (By Volume)
SPECIFICATION	Refer To Combustion Start Second Attempt, Section VII		Mil-H-5606		BB-0-925 Grade A Type-I	Mil-A-8243A
CAPACITY		Placarded	Fill To 1/4 Inch Below Full Mark	750 Psi At 21°C (70°F)	1800 Psi	2 U.S. Gal.
LOCATION	Right Engine Access Compartment	Right Engine Access Compartment	RAT Compartment Right Side	RAT Compartment Right Side	Survival Kit Assembly	Forward Electronics Compartment Right Side

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Figure 1-33

against the seat, he may not be able to unlock the harness without first loosening it slightly by using the adjustment buckles.

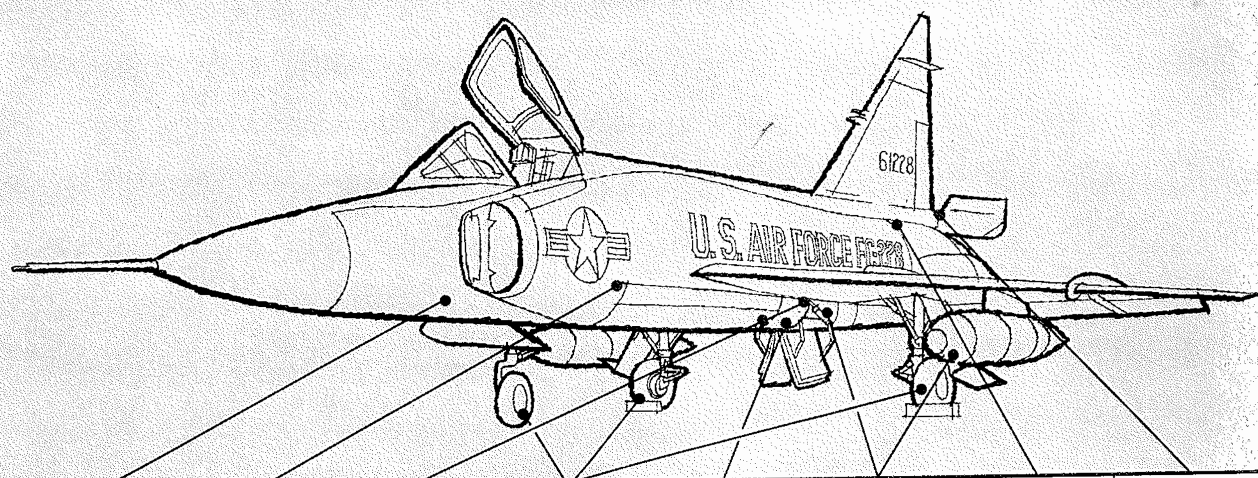
Note

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 one-half inch or less of cable movement. Pulling the harness in any direction, up, down, or sideways will lock the reel.

AUXILIARY EQUIPMENT

Information concerning the following auxiliary equipment is supplied in Section IV: Air-Conditioning and Pressurization System, Defogging System, Anti-Icing Systems, Communications and Associated Electronic Equipment, Lighting Equipment, Liquid Oxygen System, Automatic Flight Control System, Navigation Equipment, Armament Equipment, Fire Control System, Miscellaneous Equipment.

servicing diagram



Battery	Oxygen System (Liquid)	High-Pressure Pneumatic System	Tires	External Power	Fuel Tanks	Engine Oil Tank	Drag Chute
Distilled Water	Oxygen	Pneumatic Pressure	Pneumatic Pressure	115/200V 400 Cps 3 Ph AC 28V DC	Fuel	Oil	Drag Chute Pack
	BB-0-925 Grade A Type II	Ma-1 Cart	Nose—175 Psi Main—195 Psi ▲1		Grade JP 4 Mil-J-5624 ▲2	Mil-L-7808	Pack In Accordance With T.O. 14D 1-3-93
	4-4½ Liters	3000 Psi			1126 U.S. Gal. (Int.) 436 U.S. Gal. (Ext.)	5.5 U.S. Gal.	
Nose Wheel Well	Fuselage Forward Left Side			Main Wheel Well Left Side	Main Wheel Well Aft Outboard (External) Left Side (Internal)	Upper Fuselage Left Side	Above Tail Cone

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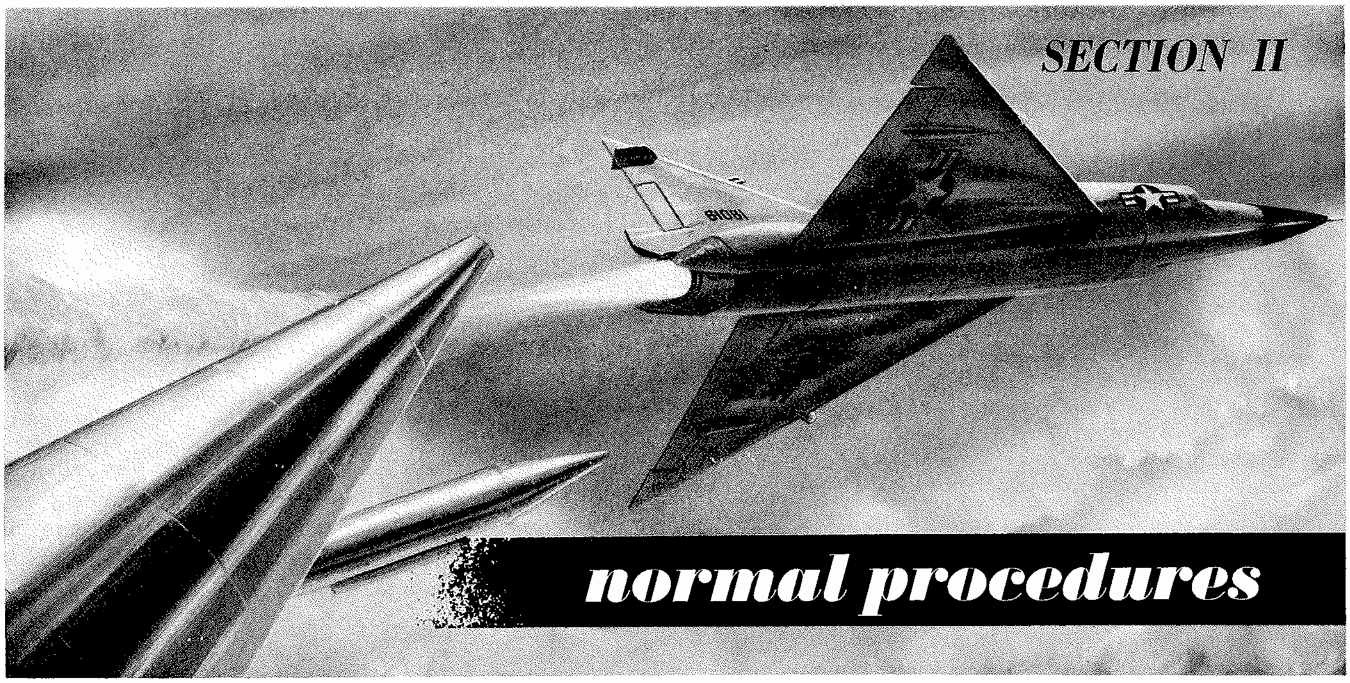

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PREPARATION FOR FLIGHT

The normal procedures in this Section have been presented to include items to place the airplane in an alert status and cocked for a scramble. In order to reduce the resulting bulk of the pilot's check list, the INTERIOR INSPECTION (ALL FLIGHTS) is arranged to include the alert cocking check list items so as not to sacrifice clarity or standardization. The airplane when cocked is ready for STARTING ENGINE (SCRAMBLE) with either external power or battery. However, external power is required to place the airplane in alert status and cocked for STARTING ENGINE (SCRAMBLE). All other phases of the check list remain unchanged.

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 proposed mission.

TAKEOFF AND LANDING DATA CARD

Complete Takeoff and Landing Data Cards in the Pilot's Abbreviated Check List, T.O. 1F-102A-(CL)1-1, by following instructions in the Appendix.

WEIGHT AND BALANCE

Refer to Section V for WEIGHT LIMITATIONS. For detailed loading information, refer to the Handbook of Weight and Balance Data, T.O. 1-1B-40. Before each flight check the following:

1. Takeoff and anticipated landing gross weights.
2. Weight and balance clearance (Form 365F).

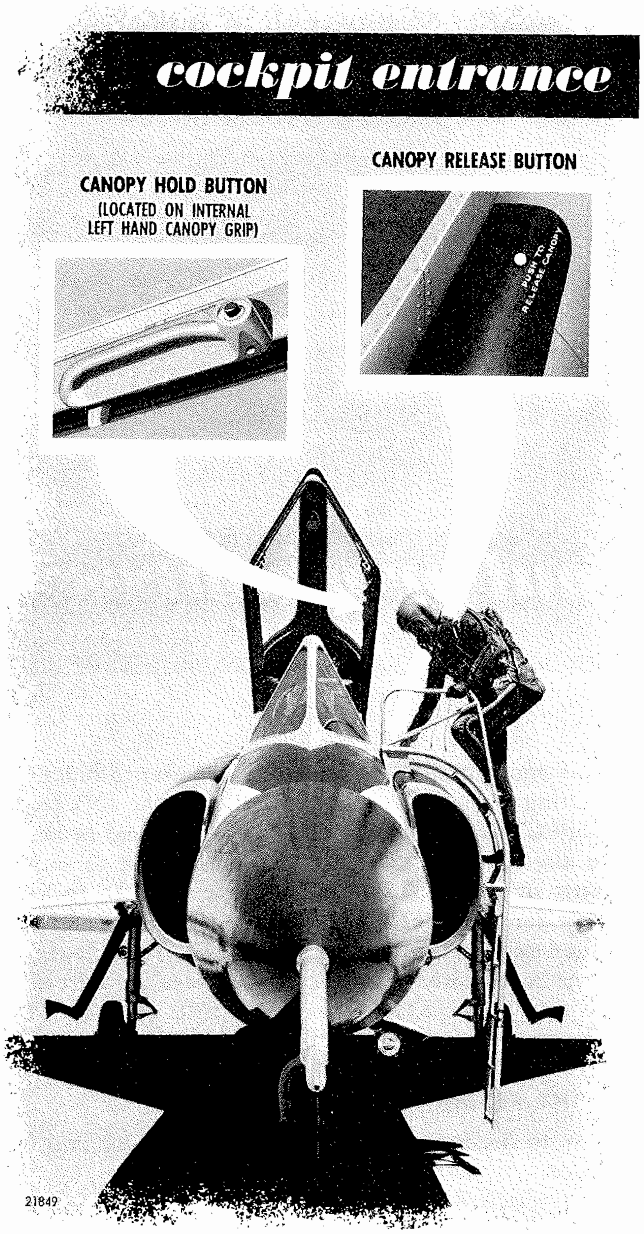


Figure 2-1

PREFLIGHT CHECK

ENTRANCE

Cockpit entry is accomplished from the left side of the airplane (figure 2-1) by use of a ladder. To open the canopy, press the external canopy release button and manually raise the canopy. On later airplanes, the canopy should be raised prior to connecting external power. However, if external power is applied before opening, the canopy can be manually raised approximately six inches. It will then be necessary to reach inside and depress the canopy hold button to completely raise the canopy.

BEFORE EXTERIOR INSPECTION

WARNING

Do not store any luggage, papers, personal effects, etc., in any of the electronic compartments at any time.

1. Canopy support tool — In place.
2. Form 781 — Check.
   Check Form 781 for engineering status.
3. Servicing — Check.
   Make certain the airplane has been serviced with fuel, hydraulic fluid, oxygen, air, armament and any special equipment required to complete the proposed mission. For detailed replenishing requirements see figure 1-33.
4. FLIPS — Check.
   Check that flight information publications required for the mission, are available.
5. Windshield — Check.
   Check that windshield delamination (marred windshield glass) does not extend more than two inches from the outer edge of the glass panels or one inch from sectional tension straps. If delamination exceeds these limits, the windshield should be considered unsatisfactory for flight.
6. Upper electronics compartment doors — Secured.
7. Canopy trailing wire — Coiled (if installed).
   Check canopy trailing wire for proper coiling.

WARNING

If canopy trailing wire is not properly coiled, ejecting the canopy may not arm the ejection seat, preventing seat ejection.

8. Throttle — OFF.
9. Survival kit — Check (if installed).
   a. Check pressure in emergency bailout bottle—1800 psi.
   b. Survival kit lid—Fully locked.
   c. Green knob assembly—Intact.
   d. Personal equipment bundle connection—Check.
10. Ejection seat safety pin—Installed, streamer visible.
11. Armament safety switch — SAFE.
12. Armament selector switch — SNAKE.
13. Igniter control switch—TRAINING (safety-wired).
14. Radar master switch — OFF.

exterior inspection

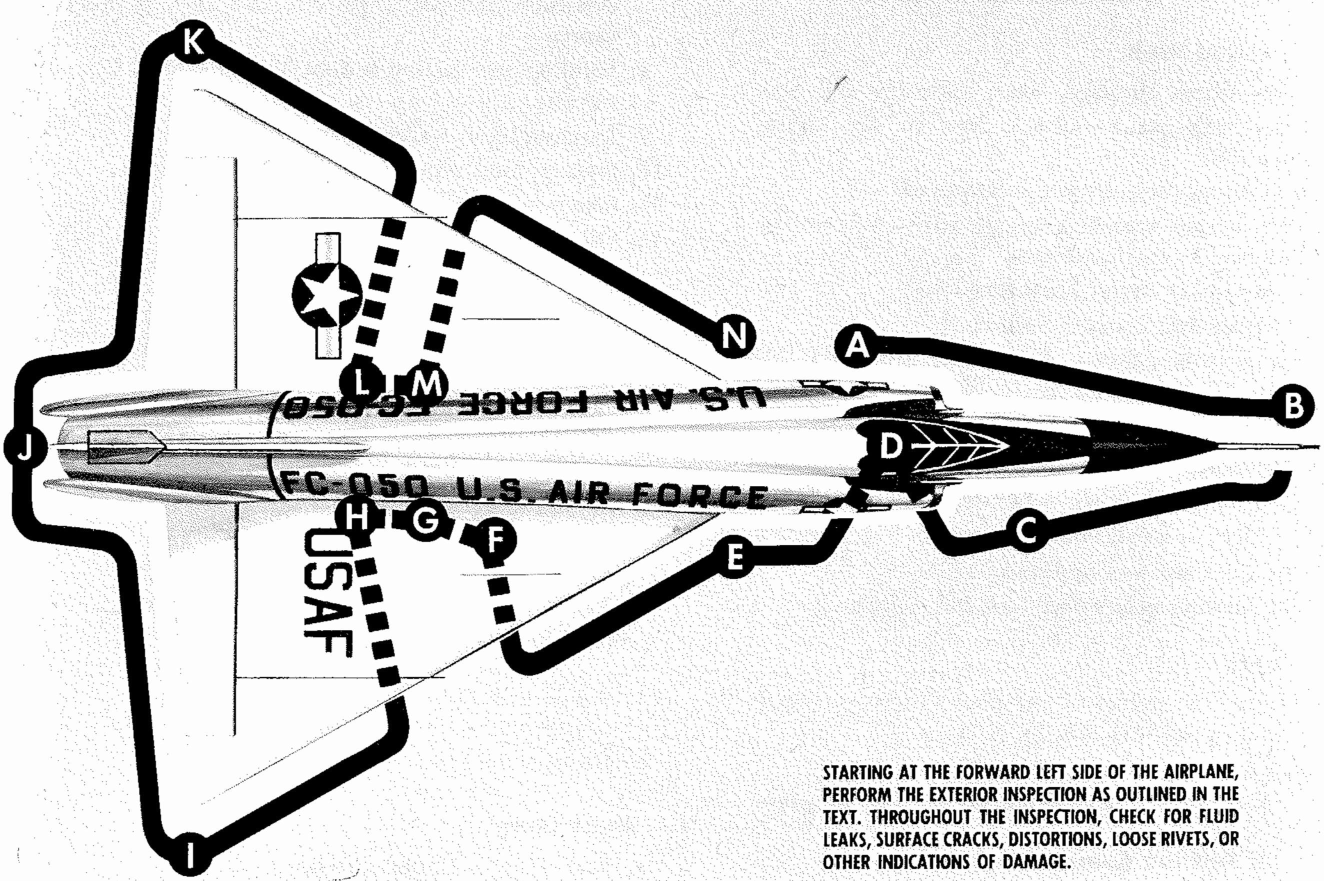


Figure 2-2

15. Canopy jettison handle — DETENT position, release button OUT.
16. RAT handle — Fully in.
17. Canopy ground maintenance safety pin—Removed.
18. Safety belt ground safety pin – Removed.

    Check that the M-12 safety belt initiator safety pin has been removed. This pin is provided for use during maintenance and, if installed, maintenance personnel should be consulted regarding the status of the ejection system before occupying the ejection seat.
19. Battery switch — OFF.
20. Master switch — NORMAL.
21. Flashlight — Check (if required).

EXTERIOR INSPECTION

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

A. Forward Left Side

1. Intake duct — Condition & no loose articles.
2. Boundary layer duct — Clear.
3. Forward electronic bay doors — Secure.
4. Rocket jump angle vane — Condition.

B. Nose

1. Static ports — Clear.
2. Radome — Condition & security.
3. Glycol ring — Undamaged & general condition.

4. Mast and pitot tube — Security, condition, & cover removed.
5. Sideslip angle transducer probe — Free & clear*.

C. Forward Right Side

1. Forward electronic bay doors — Secure.
2. Boundary layer duct — Clear.
3. Intake duct — Condition & no loose articles.

D. Nose Wheel Well

1. Tire — Check slippage, wear, cuts, and inflation.
2. Nose gear strut — Check five to six inches extension.
3. Steer damper scissors pin — Removed.
4. Nose wheel steering hinge pin — Check for absence of lateral movement.
5. Left-hand aft circuit breakers — In.
6. Left-hand dust cover — Secure.
7. Nose gear door lock — Remove (if installed).
8. Radar ground cooling door — Secure.
9. Battery — Secure.
10. Right-hand forward circuit breakers — In.
11. Right-hand dust cover — Secure.
12. Taxi light — Condition and security.
13. Nose door seal — Condition.
14. Nose landing gear safety lock pin — Removed.

E. Right Side

1. Intermediate electronic bay door — Secure (punch marks aligned with stripe).
2. Armament bay doors — Secured.
3. Position lights — Condition & security of glass.

F. Ram Air Turbine Compartment

1. RAT door — Check for free play.
2. Primary & secondary hydraulic fluid levels — Check.
3. Primary & secondary hydraulic accumulator pressure — 750 psi.
4. Caps on reservoirs, safety pins in and clipped (if applicable).
5. RAT rotor — Turns freely.
6. RAT — Condition of blades.
7. RAT door — Close.
   Close the RAT door by a gradual increase in force until click is heard (40-60 lbs).
8. RAT door test hook—Pull (approximately 80 lbs).
   Pull the RAT door test hook to check the RAT door for security.
9. RAT door test hook — Remove.

G. Right Main Wheel Well

1. Circuit breakers — In.
2. Emergency ac generator — Check.
3. Ground static wire — Check for security.
4. Chocks — In place.
5. Tire — Check slippage, wear, cuts, inflation, & wheel tie bolt torque marks.
6. Brake & hydraulic lines — Check.
7. Gear strut extension — Check five to six inches extension between scissors torque arm, center to center.
8. Landing gear fairing & door —Condition & security.
9. Landing light — Condition & security.
10. Gear ground safety lock pin — Removed.
11. Combustion starter exhaust port — Clear.
12. Oil overboard vent — Clear.

H. Engine Access Compartment

1. Constant-speed drive accumulator pressure — Check.
2. Hydraulic & fuel leaks — Check.
3. Bellcrank cable and drum — Check visually.
4. Elevator intelligence unit — Pin flush with case.

Note

Refer to OTHER OPERATIONAL LIMITATIONS, Section V, for limitations if pin is not flush with case.

5. Starter and duct installation — Free of contamination.
6. General condition — Check.

I. Right Wing

1. Fuel drain — Unobstructed.
2. External tank — Check (if installed).
   Visually check fuel level. If quantity is questionable, use dip stick. Cap secured, ground safety lock pin removed (if installed).
3. Wing leading edge & tip — Condition.
4. Position lights — Condition & security of glass.
5. Trailing edge & elevon — Condition.

J. Tail Section

1. Speed brakes — Condition.
2. Drag chute — Stowage & pin.
3. Rudder — Condition.
4. Position lights — Condition & security of glass.
5. Tailpipe & exhaust nozzles — Check.

CAUTION

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

**Some airplanes.*

K. Left Wing

1. Trailing edge & elevon — Condition.
2. Position lights — Condition & security.
3. Engine oil cap access door — Secure.
4. Wing tip & leading edge — Condition.
5. External tank — Check fuel level, cap secured, pin removed.

   Visually check fuel level. If quantity is questionable, use dip stick. Cap secured, ground safety lock pin removed (if installed).
6. Fuel drain — Clear.

L. Left Main Wheel Well

1. Chocks — In place.
2. Landing gear fairing & door — Condition & security.
3. Landing light — Condition & security.
4. Ground static wire — Check.
5. Tire–Check slippage, wear, cuts, inflation, & wheel tie bolt torque marks.
6. Brakes & hydraulic lines — Check.
7. Gear strut extension — Check five to six inches extension between scissors torque arm, center to center.
8. Ground safety switch — Check.
9. Armament control panel — Check.
10. Fuel filler cap — Secure.
11. Engine ignition disconnect switch — ON.
12. Combustion starter manual air valve — GROUND (if external air source is available).
13. Circuit breakers — IN.
14. High-pressure pneumatic system pressure — 3000 psi, filler cap secure.
15. Gear ground safety lock pin — Removed.
16. External tank jettison warning lights — Check (if installed).

    Apply dc power and check that external tank jettison warning lights are not illuminated, then press-to-test lights for satisfactory operation. Disconnect dc power.
17. External tank safety pin — Removed (if installed).

    If warning lights are not illuminated and are press-tested satisfactorily remove safety pin.
18. Overrun barrier probe* — Physically check in fully up and locked position.

WARNING

To prevent inadvertent actuation of the overrun barrier probe, do not remove the ground safety lock pin unless barrier probe is in the fully up and locked position.

19. Overrun barrier probe ground safety lock pin — Removed (if installed).*

M. Left Side

1. Aft electronic bay door — Secure.
2. Armament bay doors — Secure.
3. Emergency canopy jettison lanyard — Stowed & secure.
4. Position lights — Condition & security of glass.
5. Ram air intakes — Covers removed & condition.
6. Liquid oxygen build-up & vent handle — BUILD-UP.
7. Oxygen filler access door — Secured.

INTERIOR CHECK (ALL FLIGHTS)

Note

Each of the following checks should be performed when an external power source is available for interior inspection. Items marked with the symbol ▲ preceding the step cannot be performed if making the interior check with battery power prior to battery start. These items (▲) should be checked after battery start. (Refer to INTERIOR CHECK AFTER BATTERY START, this Section.) The alert cocking and the scramble start as presented in this check list are predicated on the following: The preflight and cocking procedures must be performed with external power available. The scramble start can be accomplished on either external or battery power.

General

1. Survival kit — Attach (if installed).

   Attach survival kit parachute attaching straps to parachute harness if survival kit is installed. Pull straps snug. Pull fastener end of straps. If slippage occurs, the attachment straps are installed backwards.
2. Safety belt & shoulder harness — Secure.
   a. Do not accomplish when cocking airplane.

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.

**AF 57-770 & on, & airplanes modified by TCTO 1F-102-658.*

Note

The safety belt must be fastened snugly to prevent the possibility of the survival kit oxygen quick-disconnect from separating from the oxygen connection on the seat during negative g maneuvers.

3. Shoulder harness handle — UNLOCKED.
   Check for freedom of movement of the shoulder harness. If the harness has been locked by a rapid pull it is necessary to cycle the shoulder harness inertia reel handle to LOCKED, then to UNLOCKED.
4. Zero delay lanyard hook — Attach (if installed).
   If "one & zero" escape system is installed, attach lanyard hook to ripcord.
5. Emergency bailout bottle hose — Connect.
6. Personal equipment leads — Connect.
7. Personal equipment lead bundle — Strapped to parachute harness (if survival kit is installed).
8. ▲ External power — Connected (if available).
   When external power is connected on some airplanes*, the dc power failure warning light and the ac power failure warning light will illuminate and will remain on until the engine is started and the external power unit is disconnected.
9. ▲ AC voltage — 105-125 volts.
   To determine that all three ac phases of the external power source are available for booster pump operation, place the voltmeter selector switch to PHASE A, B, and C and note 105-125 volts indication on each phase.

CAUTION

Do not operate the fuel boost pumps if the voltmeter readings are not as specified.

10. Battery switch — ON (if no external power source is available).
11. Seat & rudder pedals — Adjust.
    Adjust seat vertical position by use of the vertical adjustment switch and rudder pedals by crank.

WARNING

On airplanes without the survival kit, the oxygen lead may separate at the quick-disconnect fitting when the seat is adjusted to the fully up position and must be checked after seat adjustment.

Left-Hand Console

1. LH aft circuit breakers — In.
2. Spare lamps — Check.
3. Anti-g suit valve — Set.
4. Mask defog — Set.
   If partial pressure suit is worn, set mask defog as desired.
5. ▲ Fuel quantity — Check.
   Select each position on the fuel quantity selector switch and check fuel quantity gage for correct readings.
6. ▲ Fuel quantity gage test button — TEST (some airplanes).
7. Fuel selector switch – OFF (unless safety-wired†) if external power is available, or ENGINE if battery start is anticipated (some airplanes).

   Fuel shutoff valve switches — CLOSE if external power is available, or OPEN if battery start is anticipated (other airplanes).
8. ▲ Fuel system — Check.
   a. Fuel boost pump pressure-low warning lights — On.
   b. Fuel boost pump — fuel shutoff valve electrical interlock — Satisfactory operation** (do not accomplish this step if the fuel selector switch is safety-wired†).
      (1) To determine that fuel boost pump — fuel shutoff valve electrical interlock is operating satisfactorily, place the left-hand fuel boost pump switches ON and note that left-hand boost pump pressure-low warning light remains on.

CAUTION

If the boost pump pressure-low warning light does not remain on, it is an indication of a malfunction in the electrical interlock, since the pumps should not operate unless the shutoff valves are fully open. If a malfunction exists, the cause should be established and corrected before proceeding further.

      (2) Repeat step (1) for right-hand boost pumps.
   c. Fuel boost pump switches — OFF.
   d. Fuel selector switch — ENGINE (some airplanes).
      Fuel shutoff valve switches — OPEN (other airplanes).
   e. Fuel boost pumps — Check.

**AF 56-1275 thru -1316, -1332 & on.*
***AF 56-1206 & on, & airplanes modified by TCTO 1F-102-651.*
†In accordance with TCTO 1F-102-686.

(1) Observe that both boost pump pressure-low warning lights are illuminated.

(2) Left-hand forward boost pump switch ON and check that left-hand boost pump pressure-low warning light extinguishes within five seconds.

CAUTION

The warning light will normally extinguish within two seconds. However, if the light is still illuminated after five seconds, the boost pump switch should be turned OFF and the cause established and corrected before attempting to operate the other pumps.

(3) Left-hand forward boost pump switch — OFF.

(4) Repeat steps (1) through (3) for the left-hand aft, right-hand aft, and right-hand forward pumps.

f. Left-hand aft boost pump switch — ON.

At the completion of the above checks, the left-hand aft boost pump should be turned ON. Do not turn on the other boost pumps until the airplane generator is on the line and three-phase ac has been checked.

▲ 9. UHF radio — ON (if needed).

10. Speed brakes switch — NEUTRAL.

a. Speed brakes switch – IN (for cocking).

11. Fuel control switch — NORMAL (guard closed).

12. Oxygen system — Check as outlined in Section IV. (Refer to OXYGEN SYSTEM, Section IV, for additional information and detailed checks on the liquid oxygen systems.)

Left-Hand Auxiliary Panel

1. Landing gear handle — Down.
2. Landing gear warning light(s) — Check.

Check that the landing gear handle light and on some airplanes,* a red bar warning light is out. Depress landing gear warning test button and lights should illuminate. On airplanes with the red bar warning light, the audible signal should be heard in the radio headset.

CAUTION

Do not depress the landing gear warning light test button and the red bar warning light at the same time as damage may result to the audible signal generator.

Airplanes modified by TCTO 1F-102-728.

3. Landing gear position indicators — Wheels or green lights.

▲ 4. Landing & taxi lights — Check (if required).

5. LH forward circuit breakers — In.
6. Cabin air switch — PRESS.
7. Landing gear emergency extension handle — In & secure.
8. Cabin altimeter — Check.
9. Drag chute handle — In & secure.
10. External tank fuel transfer switch — OFF (if installed).

Instrument Panel

1. Directional indicator (slaved) slaving switch — NORMAL (if installed).
2. Clock — Set.
3. Flight instrument–Check & Set.

a. Directional indicator (slaved) (stabilizing).

b. Attitude indicator–Check as follows:

(1) Power warning flag for retraction within specified time limit.

(2) Horizon line for proper attitude and freedom from oscillation (energized).

(3) Horizon line for response to trim knob.

WARNING

If the "OFF" flag requires longer than 2½ minutes to retract or any oscillations are noted on the indicator after the "OFF" flag retracts, possibility of a malfunction exists. Either of the above is cause for rejection of the indicator and should be noted on Form 781.

c. Vertical velocity indicator, Machmeter-airspeed indicator, and turn-and-slip indicator indicating proper static conditions.

d. Altimeter and cockpit pressure altitude gage readings correspond to pressure conditions.

WARNING

The barometic setting knob can be turned so that the barometric disc will rotate through 360°. If the correct altimeter setting is then established, the altimeter will indicate 10,000 feet in error.

4. Engine fire warning test switch — FIRE, then OVHT.

Place engine fire test switch to FIRE and check fire warning light on steadily. Place switch to OVHT and check for flashing light.

5. Engine instrument — Check.
   Check tachometer, exhaust gas temperature gage, fuel flow indicator, and pressure ratio gage all reading zero.
6. Hydraulic pressure-low warning light — On.

Right-hand Auxiliary Panel

▲ 1. Thunderstorm lights — Checked, then OFF (if not required).
2. Canopy unlocked warning light – On.
3. Master warning system — Test.
   Place master warning light test switch to TEST and check that all lights on the warning light panel and the master warning light illuminate.

Right-hand Console

1. AC generator switch — ON; guard closed.
2. AC bus switch — NOR; guard closed.
3. DC generator switch — ON.
▲ 4. Navigation receiver — ON and check.
▲ 5. J-4 compass — Check.
   a. Latitude — Set.
   b. Function selector switch DG.
   c. Precess compass card approximately 45°.

CAUTION

Do not operate "set" switch continuously for more than 30 seconds to avoid overheating the slew motor.

   d. Function selector switch MAG, heading checked.
▲ 6. IFF — STBY; check mode code.
   a. IFF — NORMAL (for cocking).
7. Standby compass light — Check (if required).
8. RH forward and aft circuit breakers — In.

Utility Switch Panel

1. Flight mode selector switch — DIRECT MAN*.
2. Pitch damper switch — OFF**.
   Check pitch damper OFF and observe that pitch damper, AFCS, altitude hold, and AILAS switches are in the OFF positions.
3. Nesa switch — NORMAL (some airplanes); both Nesa switches — ON (other airplanes).
4. Cabin temperature control knob — AUTOMATIC.
5. Canopy defog switch — Set.
6. Windshield rain clear switch — OFF (some airplanes); STBY (other airplanes).

*AF 53-1791 thru 55-3379 unless modified by TCTO 7F-102A-546.
**AF 55-3380 & on, & airplanes modified by TCTO 1F-102A-546.

7. Cockpit no-fog/vent suit switch — OFF (if installed).
▲ 8. Pitot heat switch – Check, then OFF.
   a. Pitot heat switch — ON (for cocking).
      Have crew chief check pitot tube for heating.
9. Anti-ice switch – OFF.

Lighting Control Panel

▲ 1. Position lights — Check (if required).
   Test position light switches for FLASH, STEADY, DIM, and BRIGHT (some airplanes) or FORMATION and NAVIGATION (other airplanes).

Note

The anticollision lights will illuminate when the position light switch is placed in NAVIGATION 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 and increases maintenance problems, and during ground emergencies the operating light could confuse rescue operations since emergency ground vehicles use a similar light.

▲ 2. Cockpit lights — Check (if required).
   Check operation of cockpit lights and set rheostats as desired (OFF or ON).
3. Left-hand aft boost pump switch – OFF (for cocking).
4. UHF radio – OFF (for cocking).
5. External power — Disconnected (for cocking).

BEFORE STARTING ENGINE

An outside air source is normally utilized for starting; however, the start may be accomplished by using the airplane high-pressure pneumatic system by opening the combustion starter manual air valve in the main wheel well. An external electrical power source should be connected for starting to conserve battery power.

Note

- If it is necessary to start and no external ac power source is available, a start may be accomplished by turning the boost pumps off and following the normal start procedure. There will be no indication of fuel flow, hydraulic pressure, or fuel quantity. Starting without ac power may be harmful to the engine-driven fuel pump because of cavitation and the resulting lack of lubrication.
- Using the battery for starting will cause a heavy drain of battery power.

CAUTION

- If continuous ground operation at low rpm (below 80%) is required, it will be necessary to pressurize the hydraulic reservoirs to 50 (±5) psi before starting the engine in order to prevent cavitation of the hydraulic pumps.
- Before starting engine, determine that wheels are firmly chocked and hold wheel brakes on.
- Before starting engine, insure that no loose items of personal equipment or airplane forms are on instrument hood and all loose items on operator are secured. This is required to prevent ingestion of loose items into the engine as a result of engine operation when the canopy is open.

STARTING ENGINE

Note

- Each of the following steps should be performed when starting engine with an external power source. Items marked with the symbol ▲ preceding the step cannot be performed if starting with battery.
- For engine starting restrictions at remote bases where a compressor air source is not available refer to HIGH PRESSURE PNEUMATIC CAPABILITY, Section VII.
- When making a battery start using the combustion starter, if the start is aborted for any reason and a second start is desired, it will be necessary to recharge the starter fuel flask prior to the second start. The fuel flask will automatically charge if external power is supplied and the left-hand aft fuel boost pump is turned on. However, if no external power is available, refer to COMBUSTION START—SECOND ATTEMPT, Section VII.

Note

It is recommended, if practical, to preheat the starter compartment when ambient air temperature is at 0°C or below. This will thaw out any possible frozen condensation which may have accumulated in the starter overspeed switch mechanism since last engine shutdown.

CAUTION

Before starting engine, determine that wheels are firmly chocked, and hold wheel brakes on.

1. Starting air — Connected.

   Signal crew chief to connect compressed air for starting.

Note

If no external compressed air source is available, the combustion starter manual air valve should be placed in the AIRCRAFT position immediately prior to starting to provide starting air.

2. Danger areas — Clear.

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 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.

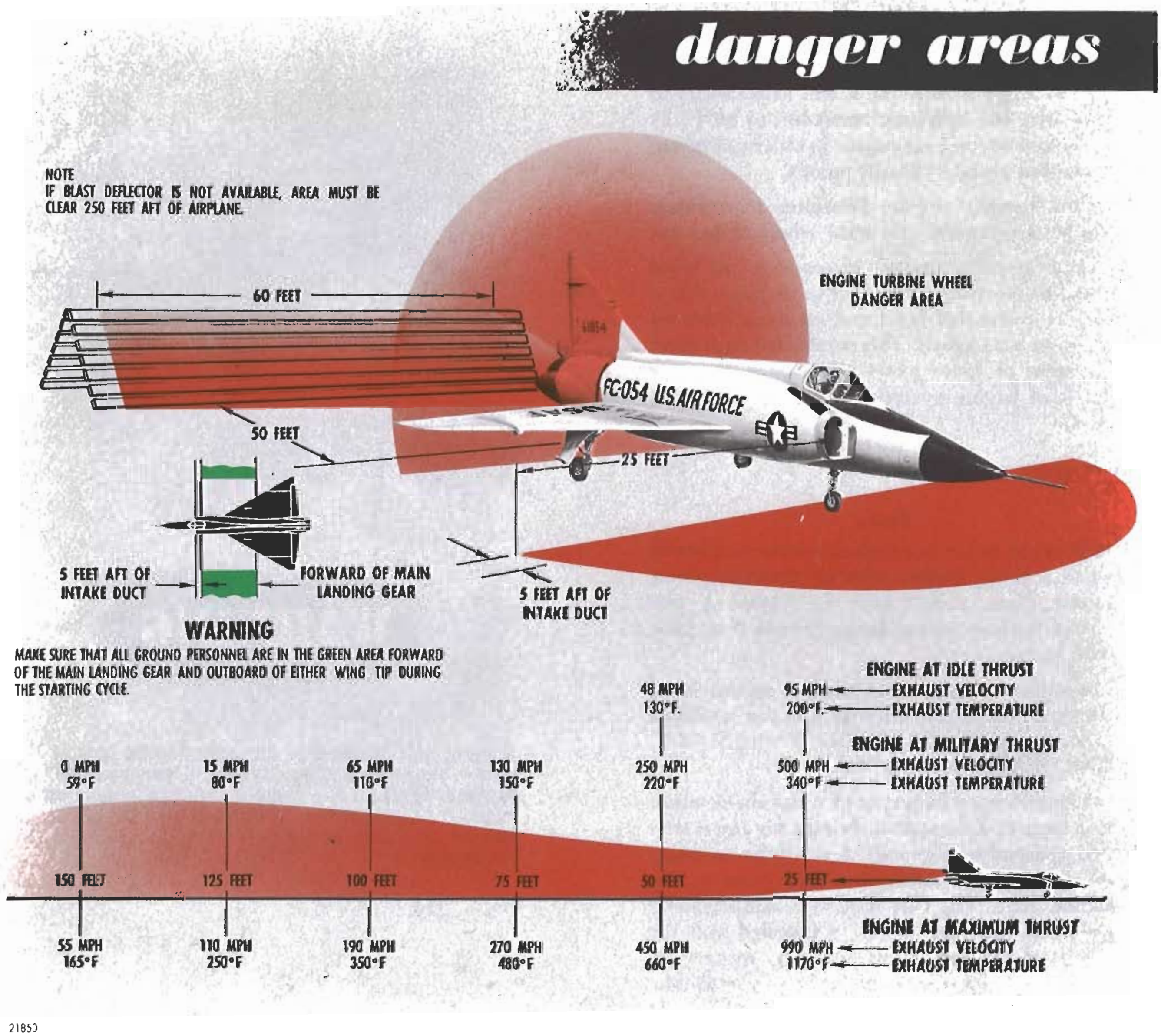


Figure 2-3

3. Throttle — START.
   Move throttle outboard to START position.
4. Ignition button — Depress.
   Hold throttle outboard and ignition button depressed until an indication of rpm is evident on the tachometer or until positive indication of hydraulic pressure. This time should not exceed six seconds, as holding for a longer period may cause condensation to form on the starter igniter plugs and thus reduce the possibility of ignition. The engine is being air-motored only at this point. For air motoring restrictions at remote bases where a compressed air source is not available, refer to HIGH-PRESSURE PNEUMATIC CAPABILITY, Section VII.

CAUTION

Once depressed the ignition button must be held depressed until the starting sequence is complete to avoid closing the starter shutoff valve and terminating ignition. 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. Do not attempt another start until fuel drainage has ceased and engine is visually checked for trapped fuel.

5. RPM — Positive rpm or hydraulic pressure indication.

WARNING

Release ignition button immediately after starter light-off if an rpm reading is not evident on the tachometer. Do not move the throttle inboard to the OFF position with the ignition button depressed. This could result in disintegration of the combustion starter. No rpm reading indicates the starter failed to engage the engine. A maximum of two attempts should be made, but if still unsuccessful, the operation should be discontinued until the cause has been established and corrected.

6. Throttle — OFF, then IDLE.

   As soon as a definite rpm indication is noted on the tachometer, move throttle inboard to the OFF position, then to IDLE.

WARNING

If starter ignition is not heard almost immediately or if a rapid increase in engine rpm is not noted, the start should be aborted immediately in accordance with the above procedure.

▲ 7. Fuel flow — Check indication.

8. Exhaust gas temperature gage — Shows increase.

   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.

9. Oil pressure-low warning light — Out (approximately 30% rpm; at least by idle rpm).

10. Ignition button — Release at 33% rpm.

CAUTION

- If the engine does not light up when 33% rpm has been reached, or within 20 seconds after throttle is advanced to IDLE, the start should be aborted. If engine rpm of less than 25% is obtained, this indicates possible starter malfunction. Abort start and have malfunction corrected prior to attempting another start. The starter may be stopped at any time by releasing the ignition button located on the throttle. Shut down the engine by using the UNSUCCESSFUL START procedure, this Section.
- Ignition operation is limited to three minutes of continuous operation to prevent overheating and subsequent damage to the ignition unit.

11. Exhaust gas temperature — Stabilized.

    Check that exhaust gas temperature rises normally and is within limits.

CAUTION

A possible hot start can be anticipated by observing a rapid increase to 500°C and temperature is still rising. When a hot start occurs, record on Form 781 the maximum exhaust temperature reached and the duration the exhaust temperature exceeded engine operating limits shown in figure 5-2.

12. Idle rpm — 55 to 65% — Check.

    Check that engine accelerates to idle rpm (approximately 55 to 65% rpm).

Note

Starter will automatically cut out at approximately 35% rpm by means of a centrifugal switch or when the ignition button is released.

13. Engine fuel pump failure warning light — Off.

CAUTION

- If the engine fuel pump failure light comes on, shut down the engine. The light will indicate that the engine stage of the fuel pump has failed and the afterburner stage of the fuel pump is supplying fuel to the engine fuel system. Investigate cause of light indication.
- If airplane pneumatic system air was used in starting, the combustion starter manual air valve must be returned to GROUND position after start is completed to prevent loss of pneumatic pressure through leakage and to prevent actuation and possible disintegration of the starter in the event that an air start becomes necessary.

▲14. LH aft fuel boost pump switch — OFF.

    Fuel boost pumps should be turned OFF prior to disconnecting external power and should remain off until all phases of the airplane's ac voltage have checked satisfactorily.

15. UHF radio — OFF.

▲16. Compressed air and external power — Disconnected.

17. Throttle — 80%.

Note

Throttle must be advanced to 80% momentarily to insure that the hydraulic reservoirs are pressurized to 50 (±5) psi as required to prevent pump cavitation. As an alternative, pressure from an external source can be used to pressurize reservoirs before starting engine. If the engine must be operated at low rpm for an extended period of time, the throttle should momentarily be advanced to 80% at approximately five-minute intervals.

UNSUCCESSFUL START

1. Throttle — OFF.

   Throttle should be placed in OFF position to shut off fuel flow to the engine and reduce the possibility of fire.

2. Check for fire.

   Check instruments and have crew chief check for visible evidence of fire.

Note

Investigate cause of the unsuccessful start. Do not attempt another start until cause has been determined and corrected. If it is determined another start can be made, wait 30 seconds for fuel drainage before attempting another start. Have engine visually checked for trapped fuel. Check for zero rpm before re-starting.

HUNG START OR SLOW START

A hung start is indicated by failure of rpm to increase after lightup with exhaust temperature remaining within limits. A slow start is similar to a hung start and is evidenced by a slow but continuous acceleration of rpm to idle after lightup. When a hung start or slow start is experienced, proceed as follows:

1. Tachometer–Between 25 and 55% rpm.
2. Fuel control switch–EMERGENCY.

CAUTION

Do not place fuel control switch to EMERGENCY if engine rpm is below 25%. This could introduce an excess of fuel into the engine and cause exhaust gas temperature to exceed limits. It may be necessary to advance the throttle slightly past IDLE while in EMERGENCY to obtain IDLE rpm of 55 to 65%.

3. Fuel control switch — NORMAL at 55 to 65% rpm.

CLEARING ENGINE

1. Compressed air — Connected.

   If no external compressed air source is available, the combustion starter manual air valve should be in AIRCRAFT position to provide air for clearing.

2. Engine ignition disconnect switch — OFF.

   Have crew chief place the engine ignition disconnect switch in the left-hand main wheel well to OFF.

3. Throttle — OFF.
4. Fuel selector switch — ENGINE (some airplanes).
   Fuel shutoff valve switches — OPEN (other airplanes).

   Both fuel valves should be open unless internal engine fire is suspected. This permits lubrication of the fuel control unit.

5. Throttle — START.
6. Ignition button — Depress and hold.
7. RPM — Check positive indication.

WARNING

Release ignition button immediately if an rpm reading is not evident on the tachometer. Do not move the throttle inboard to the OFF position with the ignition button depressed. This could result in disintegration of the combustion starter. No rpm reading indicates the starter failed to engage the engine. A maximum of two attempts should be made, but if still unsuccessful, the operation should be discontinued until the cause has been established and corrected.

8. Throttle — OFF; insure that engine clears.

   Position throttle inboard to OFF while holding the ignition button depressed to energize the starter. Hold ignition button depressed until combustion starter has completed its cycle.

CAUTION

The following cooling periods must be observed to prevent damage to the starter from overheating:

a. A second start may be attempted any time after returning the throttle to OFF for approximately 15 seconds.

b. A second start within a 15-minute period is considered a double start. A double start shall be followed by a one-hour cooling period before the next start.

c. A second start between 15 and 40 minutes from the first shall be followed by a 40-minute cooling period before the next start is attempted.

d. Any start after a prescribed cooling period or 40 minutes after a first start shall be considered as a first start.

9. Ignition button — Release.

   After engine clears, release ignition button. Have crew chief place the combustion starter manual air valve to GROUND if airplane compressed air was required for clearing.

ENGINE GROUND OPERATION

Engine warmup is not required under normal conditions. After the engine stabilizes at idle, it may be operated at full thrust; however, cooling limitations (ground operation) must not be exceeded.

CAUTION

- Insufficient tire traction requires the use of an airplane restraining bridle when using afterburner on the ground.
- If the throttle is inadvertently retarded to OFF, a flameout occurs immediately. Do not reopen throttle, as relight is impossible and resultant flow of unburned fuel into the engine creates a fire hazard.

BEFORE TAXIING

ELECTRICAL POWER SUPPLY SYSTEM

DC System

1. DC generator output — Check.

   After external power has been disconnected, check that dc operated equipment or lights are operating from the dc generator and that dc power failure light is out. The dc generator should be operating when engine speed is above approximately 30% rpm.

2. Battery switch — ON.
3. DC generator switch — OFF.
4. Battery output — Check.

   Check that dc operated equipment or lights are operating and that dc power failure light is illuminated.

5. DC generator switch — ON.

AC System

1. AC voltage (all phases) — 105-125 volts.

   To check operation of main ac generator, the ac bus switch should be in NOR position. All positions of the ac voltmeter selector switch should give the same voltmeter readings in 105-125-volt range except that the 26v position should read above 0 but not over 50 volts.

2. AC generator switch–OFF.
3. AC bus switch–EMER (some airplanes); RESET, then ON (other airplanes); 103-140 volts.

   Check operation of emergency ac generator. All positions of the ac voltmeter selector switch should give voltmeter readings of 103-140 volts except that the 26v position should read above 0 but not over 50 volts.

4. AC bus switch–NOR.
5. AC generator switch–RESET, then ON.
6. Transformer-rectifier–Check (some airplanes*).

   Determine that the transformer-rectifier is operative and providing dc power to the emergency dc bus as follows:

**Airplanes modified by TCTO 1F-102-727.*

a. AC generator switch — OFF.

Placing the ac generator switch to OFF, prevents premature energizing of the emergency dc bus by the transformer-rectifier.

b. DC generator switch — OFF.

With the dc generator OFF, the dc nonessential bus and the emergency dc bus are disconnected from the dc essential bus. Check that the battery is functioning.

c. AC bus switch – EMER (check for 0 volts).

With the ac bus switch in EMER, (do not hit the RESET position) check for 0 volts. If ac power is available, a malfunction is indicated.

d. AC bus switch – RESET, then EMER.

All positions of the ac voltmeter selector switch should give voltmeter readings of 103-140 volts except that the 26v position should read above 0 but not over 50 volts.

Note

Emergency ac control is powered by the emergency dc bus but is reset by dc power from the essential dc bus. Once reset, the emergency ac generator continues to power the emergency dc bus through the transformer-rectifier, and the emergency dc bus becomes self sustaining. Failure to reset indicates a malfunction.

e. AC bus switch — NOR (voltage should be 0).

f. Battery switch —TR FAIL.

The TR FAIL position reconnects the emergency dc bus to the dc essential bus.

g. AC generator switch—RESET, then ON. Check 105-125 volts.

Note

AC power control and reset are powered by the emergency dc bus. If the TR FAIL position of the battery switch does not reconnect the emergency dc bus to the dc essential bus, the main ac generator will not reset.

h. Battery switch — ON.

i. DC generator switch – RESET then ON.

7. Fuel boost pump switches—ON (if external power source was used for start).

   Fuel boost pumps should be turned ON, one at a time allowing five seconds for each pump to start, when a successful check of all phases of ac voltage has been completed after starting with external power. If battery start was made, boost pumps should not be turned ON until after completion of INTERIOR CHECK AFTER BATTERY START.

8. Radar master switch—STBY.

INTERIOR CHECK AFTER BATTERY START

Except for the following items, the interior check should have been completed prior to starting engine. Starting at the left-hand console, perform the following checks:

1. Fuel quantity — Check.

   Select each position on the fuel quantity gage selector switch and check fuel quantity gage for correct indications.

2. Fuel quantity gage test button — TEST.

   Depress fuel quantity gage test button; check gage for decrease.

3. Fuel boost pump pressure-low warning lights — ON.

4. Fuel boost pump — Fuel shutoff valve electrical interlock — Satisfactory operation* (do not accomplish this step if the fuel selector switch is safety-wired**).

   a. Fuel selector switch — R (some airplanes).

   Rotate the fuel selector switch to R in order to close the left-hand fuel shutoff valve.

   Left-hand fuel shutoff valve switch — CLOSE (other airplanes).

   b. Left-hand fuel boost pump switches — ON; left-hand boost pressure-low warning light remains on.

CAUTION

If the boost pressure-low warning light does not remain on, it is an indication of a malfunction in the electrical interlock, since the pumps should not operate unless the shutoff valve is fully open. If a malfunction exists, the cause should be established and corrected before proceeding further.

   c. Fuel selector switch — L (some airplanes).

   Rotate the fuel selector switch through ENGINE to L position in order to close the right-hand fuel shutoff valve.

   Left-hand fuel shutoff valve switch — OPEN (other airplanes).

   Right-hand fuel shutoff valve switch — CLOSE (other airplanes).

   d. Repeat step "b" above for the right-hand boost pumps and pressure-low warning light.

5. Fuel boost pump switches — OFF.

6. Fuel selector switch — ENGINE (some airplanes). Fuel shutoff valve switches — OPEN (other airplanes).

**AF 56-1206 & on, & airplanes modified by TCTO 1F-102-651.*
***In accordance with TCTO 1F-102-686.*

7. Fuel boost pumps — Check.
   a. Observe that both boost pressure-low warning lights are illuminated.
   b. Left-hand forward boost pump switch ON and check that left-hand boost pressure-low warning light extinguishes within five seconds.

CAUTION

The warning light will normally extinguish within two seconds. However, if the light is still illuminated after five seconds, the boost pump switch should be turned OFF and the cause established and corrected before attempting to operate other pumps.

   c. Left-hand forward boost pump switch — OFF.
   d. Repeat steps "a" through "c" for the left-hand aft, right-hand forward and right-hand aft boost pumps.
8. Boost pump switches — ON.
   The boost pump switches should be turned on one at a time allowing approximately five seconds for each pump to start.
9. UHF radio — ON.
10. Landing & taxi lights — Climatic.
11. Thunderstorm lights — Climatic.
12. Navigation receiver — ON and check.
13. J-4 compass — Check.
   a. Latitude set.
   b. Function selector switch — DG.
   c. Precess compass card approximately 45°.

CAUTION

Do not operate "set" switch continuously for more than 30 seconds to avoid overheating the slew motor.

   d. Function selector switch — MAG, heading checked.
14. IFF — STBY; check mode code.
15. Pitot heat switch — Check, then OFF.
16. Position lights — Climatic.
   Test position light switches for FLASH, STEADY, DIM, and BRIGHT (some airplanes) or FORMATION and NAVIGATION (other airplanes).

Note

The anticollision lights will illuminate when the position lights switch is placed in NAVIGATION 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 and increases maintenance problems, and during ground emergencies the operating light could confuse rescue operations since emergency ground vehicles use a similar light.

17. Cockpit lights — Climatic.
   Check operation of cockpit lights and set rheostats as desired (OFF or ON).

HYDRAULIC AND FLIGHT CONTROL SYSTEMS CHECK

CAUTION

Rapid and abrupt movement of the control stick should be avoided because of the possibility of damage to system components.

1. Throttle — IDLE.
2. Speed brakes switch — IN, then center (neutral) position.
   Retract speed brakes and check with crew chief for proper speed brake operation. Secondary hydraulic system pressure may drop momentarily but must return to normal within two seconds.
3. Pitch and yaw dampers — Engage.
4. Flight control surface movement — Check.
   Check rudder surface movement. Then, with the stick in the fully aft position, check elevon surfaces for proper movement (aileron as well as elevator position).

Note

During the above check there may be jerky rudder pedal motions caused by the turn coordinator. This is a normal condition.

5. Hydraulic system recovery (primary and secondary) — Check.
   With the hydraulic pressure selector switch in the PRI position, check primary hydraulic system pressure for 2950 (±100) psi. Check operation of the primary hydraulic system by moving the control stick from any corner to the diagonally opposite corner in approximately three seconds. If system pressure drops, the hydraulic pressure must return to system pressure within two seconds, or less, after control stick movement has stopped. Then place the hydraulic pressure selector switch in the SEC position and repeat the above procedures for a check on the secondary hydraulic system.
6. Momentary interrupt trigger — Depress.
   Note that AFCS disengages.

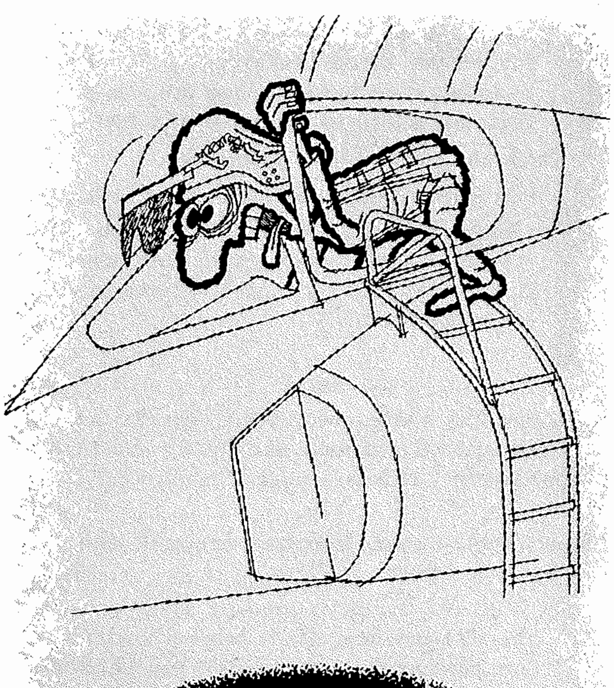
7. Emergency damper disconnect button — Depress.
   Check that pitch and yaw dampers disengage.
8. Trim — Check and set for takeoff trim.
   Trim nose-down, right wing down, and right rudder; then depress takeoff trim button. Control surfaces should return to neutral. Takeoff trim light should illuminate.

GENERAL

In addition to the above checks, observe the following instructions before taxiing:

CAUTION

If the canopy is open during taxi operations, observe canopy limit speeds. The canopy hold-open rod must be used on airplanes which do not have a canopy hold button. Keep hands away from canopy sill unless the canopy support tool is in place.



WARNING

THE FINAL MOVEMENT (1 TO 2 INCHES) OF CLOSING THE CANOPY IS RAPID AND IT IS CINCHED DOWN WITH CONSIDERABLE FORCE; THEREFORE, CARE SHOULD BE TAKEN THAT THE AREA BENEATH THE CANOPY IS CLEAR TO PREVENT PERSONAL INJURY OR DAMAGE TO EQUIPMENT.

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1. Ejection seat safety pin — Remove.
   Remove ground safety lock pin from right arm rest and stow. If pin is not visible, ascertain that the pin streamer has not dropped down beside the seat or become disconnected from the pin.
2. Canopy support tool — Remove.
3. Canopy — Close and lock.
4. Canopy warning light — Out.
   Check canopy warning light out indicating latches are fully engaged.
5. IFF master switch — As desired.
6. Radar master switch — As desired.
7. Anti-ice switch — AUTOMATIC.
8. Radio call — Accomplished.
   Obtain taxi-takeoff instructions and set altimeter.
9. Engine pressure ratio gage — Set.
   Check outside air temperature and set engine pressure ratio gage to indicate takeoff limits.

TAKEOFF CHECK TABLE — J57-P-23 ENGINE

TEMP		PRESSURE RATIO SETTING
°F	°C	
32	0	2.30
37	2.5	2.28
41	5	2.26
45	7.5	2.24
50	10	2.22
54	12.5	2.20
59	15	2.18
63	17.5	2.17
68	20	2.15
72	22.5	2.13
77	25	2.11
81	27.5	2.09
86	30	2.07
90	32.5	2.06
95	35	2.04
99	37.5	2.02
104	40	2.00
108	42.5	1.98
113	45	1.96

minimum turning radius and ground clearances

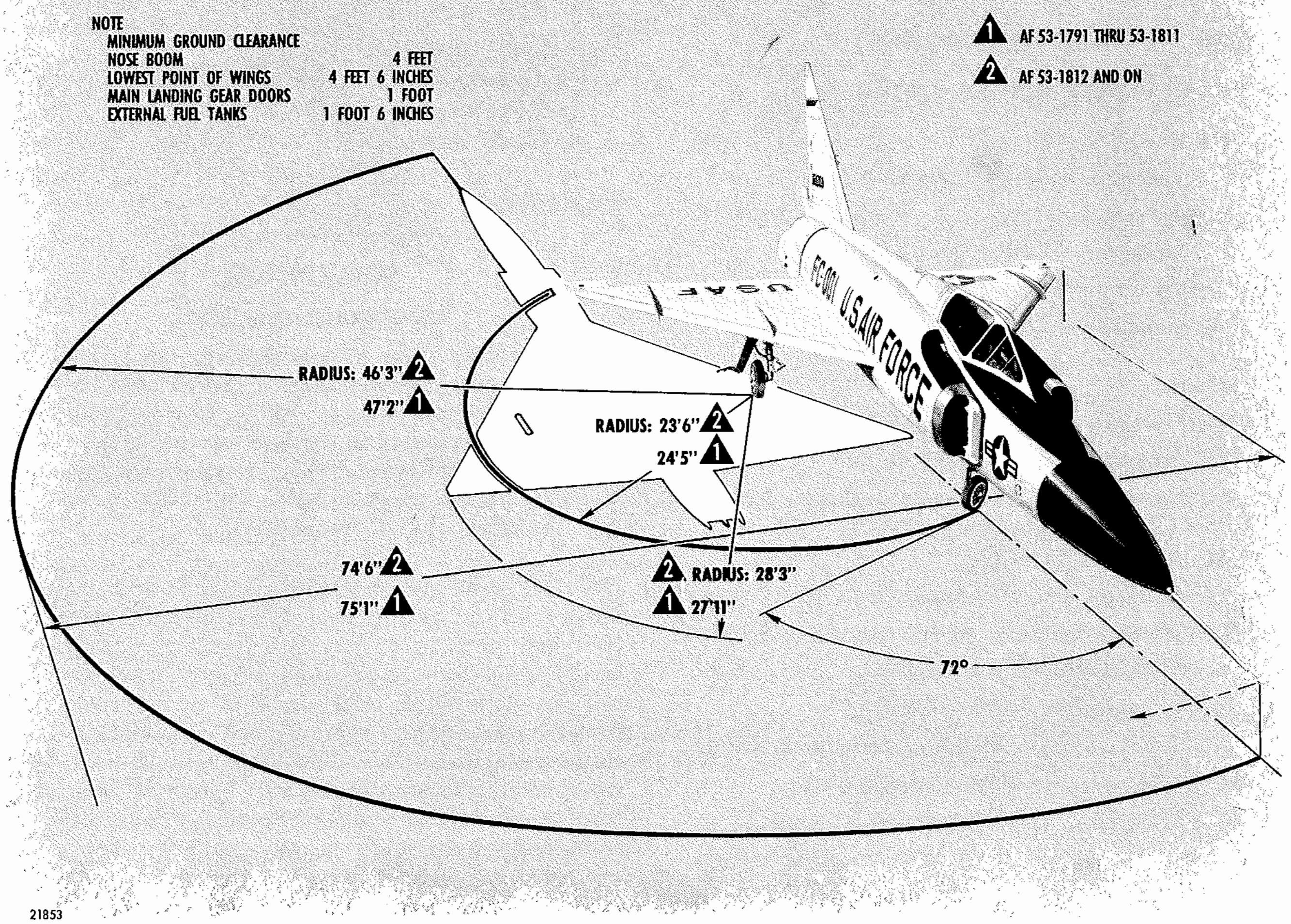

Figure 2-4

10. Armament reset switch — Actuated by crew chief (airplanes with missiles aboard).

    Have crew chief actuate the armament reset switch and obtain illumination of the armament reset light to insure that the armament system is ready for operation and that snubbing pressure is being applied to the missile bay area.

11. Chocks — Removed.

    Signal crew chief to remove wheel chocks.

STARTING ENGINE (SCRAMBLE)

1. Starting air — Connected.
2. Danger areas — Clear.
3. External power — Connected or Battery switch — ON.
4. Throttle — START.
5. Ignition button — Depress.
6. RPM — Positive indication.
7. Throttle — OFF, then IDLE.
8. ▲ Fuel flow — Check indication.
9. Exhaust gas temperature gage — Shows increase.
10. Oil pressure-low warning light — Off (approximately 30% rpm; at least by idle rpm).
11. Ignition button — Release at 33% rpm.
12. All personal equipment — Attach.
13. Canopy support tool — Remove.

14. Exhaust temperature — Stabilized.
15. Idle rpm — 55 to 65% rpm.
16. Engine fuel pump failure warning light — Off.
▲17. LH aft fuel boost pump — OFF.
18. UHF radio — OFF.
19. Compressed air and external power — Disconnected.
20. Throttle — 80%.

BEFORE TAXIING (SCRAMBLE)

ELECTRICAL POWER SUPPLY SYSTEM

DC System

1. DC generator output — Check.
2. Battery switch — ON.
3. DC generator — OFF.
4. Battery output — Check.
5. DC generator — ON.

AC System

1. AC voltage (all phases) — 105-125 volts.
2. AC generator switch — OFF.
3. AC bus switch — EMER (some airplanes); RESET, then ON (other airplanes); 103-140 volts.
4. AC bus switch — NOR.
5. AC generator switch — Reset then ON.
6. Transformer-rectifier — Check (some airplanes*).
   a. AC generator switch — OFF.
   b. DC generator switch — OFF.
   c. AC bus switch — EMER. (check for 0 volts).
   d. AC bus switch – RESET then EMER.
   e. AC bus switch — NOR (voltage should be 0).
   f. Battery switch —TR FAIL.
   g. AC generator switch — RESET then ON. Check 105-125 volts.
   h. Battery switch — ON.
   i. DC generator switch — RESET then ON.
7. Fuel boost pump switches — ON (if external power source was used for start).
8. Radar master switch —WARM.

INTERIOR CHECK AFTER BATTERY START (SCRAMBLE)

1. Fuel boost pumps — Check, then ON.
2. UHF radio — ON.

HYDRAULIC POWER SUPPLY SYSTEM

1. Throttle — IDLE.
2. Flight controls — Check.
3. Hydraulic system recovery — Check.
4. Trim — Check and set for takeoff.

GENERAL (SCRAMBLE)

1. Ejection seat safety pin — Remove.
2. Canopy — Close and lock.
3. Canopy warning light — Out.
4. Radar master switch — As desired.
5. Radio call — Accomplished.
6. Engine pressure ratio — Set.
7. Armament reset switch — Actuated by crew chief (airplanes with missiles aboard).
8. Chocks — Removed.

TAXIING

WARNING

If airplane is to be operated on the ground under possible conditions of carbon monoxide contamination, such as taxiing directly behind another operating jet airplane, or during operation with tail into wind, use oxygen with regulator diluter lever at 100% OXYGEN (some airplanes) or ON (other airplanes).

The taxiing and ground handling qualities of the airplane are normal and require no special techniques. There may be a slight delay in the braking between the pedal release and braking action release. Idle thrust is sufficient for all taxi operations under most conditions on concrete except initial movement from standstill; fuel consumption at idle thrust setting is approximately 840 pounds per hour. Nose wheel steering is not difficult but slightly higher rudder forces are noticeable which, while tending to keep the airplane aligned down the runway, cause the airplane to straighten more quickly than first expected during recovery from a turn. Initial reaction to this effect results in a "snaking" motion of short duration.

Note

Beyond 50 degrees left or right from center, or when airplane weight is off the gear, nose wheel steering is inoperative and the steering button will be keying the microphone.

Visibility during taxiing is very good except that vision to the rear is blocked through an arc of approximately 60 degrees; the wing tip and approximately one foot of the elevon can be seen. The following checks should be accomplished during taxi:

1. Nose wheel steering and brakes — Check.

   Engage nose wheel steering, commence taxiing and maintain directional control by operating rudder pedals as required. While rolling straight, release nose wheel steering and check

**Airplanes modfied by TCTO 1F-102-727.*

the brakes individually, noting that nose wheel steering disengages. See figure 2-4 for Minimum Turning Radius.

CAUTION

Excessive pressure on the rudder pedals with nose wheel steering engaged and the airplane not in motion may cause damage to steering mechanism.

2. Flight instruments — Check & set.
3. Navigation equipment — Check.

   Check operation of navigation equipment and directional indicator (slaved) for indicator changes during taxiing.

BEFORE TAKEOFF

PREFLIGHT AIRPLANE CHECK

1. Warning lights — Out.
2. Takeoff trim — Set.
3. Flight controls — Checked & free.

   Check rudder surface movement. Then, with the stick in the fully aft position, check elevon surfaces for proper movement (aileron as well as elevator position).
4. Safety belt — Fitted snugly.
5. Zero delay lanyard hook–Attached (if installed).
6. Shoulder harness inertia reel handle — UNLOCKED.
7. Cockpit no-fog vent suit switch – ON (if installed).

   Cabin temperature control knob – AUTOMATIC, midway or hotter (airplanes without cockpit no-fog vent suit switch).

WARNING

Excessive moisture condensation may occur through the cabin pressurization system. This condensation may become so dense when operating in conditions of high dew point temperature as to make it impossible to read the instrument panel presentation. On airplanes that do not have a cockpit no-fog vent suit switch, the temperature control knob should be placed to the hottest level possible that permits comfort and still prevents fog formation.

8. Pitot heat – As required.

Taxi into takeoff position, center nose wheel, and hold wheel brakes.

CAUTION

In order 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.

PREFLIGHT ENGINE CHECK

Fuel Control Emergency System Check

1. Throttle — IDLE.
2. Fuel control switch — EMER.
3. Emergency fuel control warning light — On.
4. Fuel flow 750 pph minimum; 1050 pph maximum.

   Check fuel through the emergency system with the engine at idle rpm.
5. Fuel control switch — NORMAL (guard closed).
6. Emergency fuel control warning light — Out.

Thrust Check

When ready to roll accomplish the following checks:

1. Throttle — TAKEOFF (some airplanes) or FULL MIL POWER (other airplanes).

   If a takeoff lock trigger is installed, advance throttle to TAKEOFF. On airplanes not equipped with a takeoff lock trigger, advance to FULL MIL POWER.
2. Engine instruments — Check.

   Check tachometer, exhaust gas temperature gage, fuel flow indicator, and engine pressure ratio gage for normal operating limits.

Note

Engine rpm will vary for individual engines for military thrust. The rpm for takeoff should be in relation to the rated rpm as specified on the engine data plate. Engine rpm will be less when obtaining military thrust with temperatures below standard.

TAKEOFF

WARNING

If excessive moisture condensation occurs during takeoff, place the cabin air switch to the RAM position. Return to the PRESS position after becoming airborne and above approximately 3000 feet.

NORMAL TAKEOFF

A typical takeoff is illustrated in figure 2-5. Refer to Appendix for takeoff charts showing distances required at varying gross weights, temperatures, and field elevations. Use the following procedures for normal takeoff:

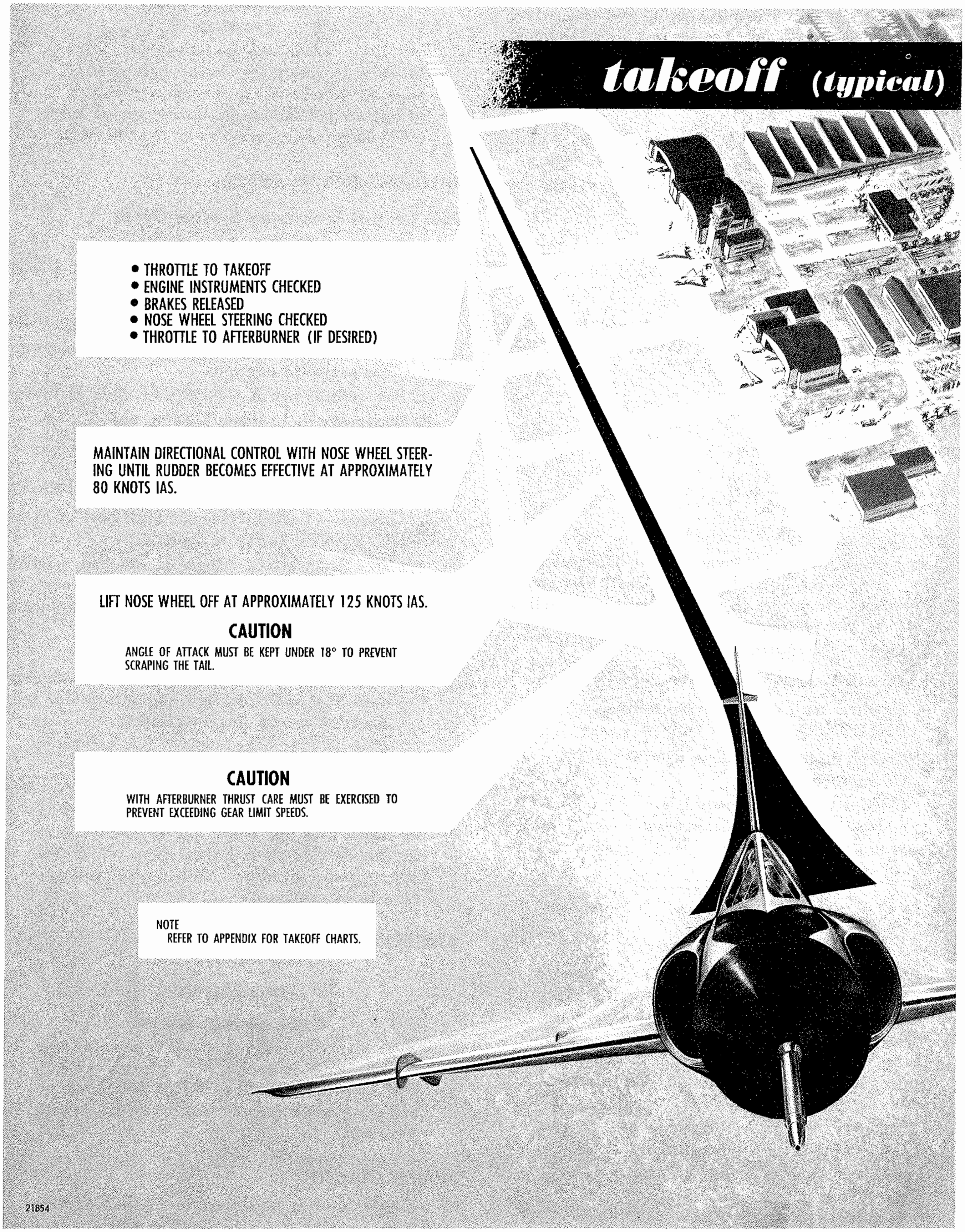
takeoff (typical)
• THROTTLE TO TAKEOFF
• ENGINE INSTRUMENTS CHECKED
• BRAKES RELEASED
• NOSE WHEEL STEERING CHECKED
• THROTTLE TO AFTERBURNER (IF DESIRED)
MAINTAIN DIRECTIONAL CONTROL WITH NOSE WHEEL STEERING UNTIL RUDDER BECOMES EFFECTIVE AT APPROXIMATELY 80 KNOTS IAS.
LIFT NOSE WHEEL OFF AT APPROXIMATELY 125 KNOTS IAS.
CAUTION
ANGLE OF ATTACK MUST BE KEPT UNDER 18° TO PREVENT SCRAPING THE TAIL.
CAUTION
WITH AFTERBURNER THRUST CARE MUST BE EXERCISED TO PREVENT EXCEEDING GEAR LIMIT SPEEDS.
NOTE
REFER TO APPENDIX FOR TAKEOFF CHARTS.
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Figure 2-5

1. Throttle — TAKEOFF.
2. Engine instruments — Check.
   Check tachometer, exhaust gas temperature gage, fuel flow indicator, and engine pressure ratio gage for normal operating limits.
3. Brakes — Release.
   Release brakes and establish a straight takeoff roll.
4. Nose wheel steering — Check.
   Move rudder pedals until it is ascertained that nose wheel steering is engaged.

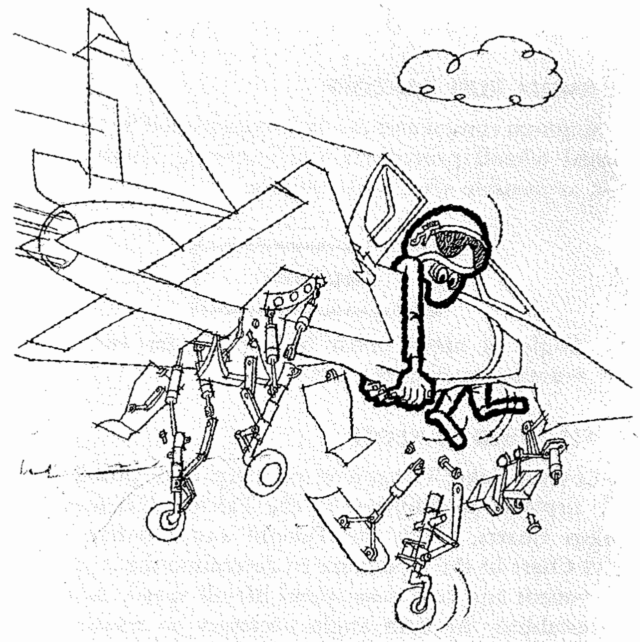
CAUTION

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

5. Throttle — AFTERBURNER (if desired).

WARNING

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

TO PREVENT AIRLOADS FROM INFLICTING STRUCTURAL DAMAGE, THE LANDING GEAR SHOULD BE UP AND LOCKED BEFORE LIMIT SPEED IS REACHED.

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Maintain directional control with nose wheel steering until the rudder becomes effective at approximately 80 KIAS. Beginning at 125 KIAS, raise the nose to the horizon with maximum thrust, or slightly below the horizon with military thrust, and allow the airplane to fly off the ground. Lower angles of attack will result in higher speed, longer run takeoffs. Higher angles of attack will result in lower airspeeds, and lower rates of climb immediately following takeoff. Do not prematurely raise the nose during takeoff as increased angle of attack at low speeds will result in excessive ground roll during takeoff.

CAUTION

- Angle of attack must be kept under 18° to prevent scraping the tail.
- With afterburner thrust, care must be exercised to prevent exceeding gear limit speeds before landing gear is fully retracted.

If EGT is increasing, without engine acceleration, so as to exceed allowable limits, reduce thrust to maintain temperature within limits. Monitor tachometer. Any engine speed in excess of maximum rpm limits should be noted on Form 781.

Note

In maximum thrust takeoffs, the altimeter may lag or even indicate a decrease in altitude just before breaking ground and during the initial climbout. This altimeter error is the result of disturbed pressure ahead of the airplane due to acceleration and high angle of attack. The altimeter will indicate correctly after the airplane reaches approximately 300 feet of altitude.

MILITARY THRUST TAKEOFF

If a takeoff is to be made without 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. 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 18° to prevent scraping the tail.

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 to determine the cross-wind component and best nose wheel lift-off speed. After lift-off, establish the crab angle necessary to maintain the desired flight path.

AFTER TAKEOFF — CLIMB

Note

There are no provisions for automatic transfer from normal to emergency fuel flow within the fuel control unit.

When airplane is definitely airborne:

1. Landing gear handle — Up.

Note

Exercise care not to knock the throttle aft and out of AFTERBURNER position when retracting the landing gear.

2. Gear checked — Up.

   Check that landing gear position indicators indicate gear up and landing gear warning light off.

3. EGT — Monitor.

WARNING

If EGT exceeds the limits shown in figure 5-2, reduce thrust and increase airspeed. Abort the mission and land as soon as practicable. Make an entry on Form 781 indicating time and temperature peaks of overtemperature operation.

4. Takeoff locks — Release before 7000 feet (if installed).

   To release the takeoff locks in the fuel control unit, retard the throttle aft of TAKEOFF position before reaching an altitude of 7000 feet, flight conditions permitting.

Note

On later airplanes, without the fuel control takeoff locks incorporated, it is not necessary to retard the throttle out of TAKEOFF when climbing above 7000 feet.

5. Oxygen — Check.

   If 100% OXYGEN was used for takeoff and partial pressure suit is not worn, return oxygen regulator diluter lever to NORMAL OXYGEN, unless carbon monoxide contamination is suspected. If such is the case, continue use of 100% oxygen as long as considered necessary.

WARNING

On airplanes not equipped with a survival kit, oxygen regulator diluter lever must be returned to NORMAL OXYGEN as soon as possible to prevent premature depletion of the oxygen supply unless it is determined that the supply is adequate for the duration of the flight.

6. Zero delay lanyard hook — Detach and stow (if installed).

   If "one & zero" escape system is installed, detach zero delay lanyard hook from ripcord after reaching 2000 feet above terrain in accordance with limitations established in the Zero Delay Lanyard Engagement Requirements Chart (figure 3-3) and stow on parachute harness. For minimum ejection altitudes, see the Emergency Minimum Ejection Altitude Table, figure 3-4.

7. Damper systems — Engage.

WARNING

Initial engagement of the pitch damper should be avoided at less than 5000 feet above the terrain. This altitude will provide adequate ground clearance in the event of any phasing errors that could occur during the initial engagement.

8. External tank fuel transfer switch — ON, when internal fuel quantity indicates 5500 pounds or less.

9. Cockpit no-fog vent suit switch — OFF (if installed).

   Place cockpit no-fog vent suit switch to OFF to return temperature control to the automatic temperature controller.

10. IFF — Checked.
    If positive operation of the normal mode of IFF has not been established during departure with an air traffic facility, a check should be made with such a facility as soon after takeoff as flight conditions will permit. This check must be made prior to entering a radar advisory area. If IFF is inoperative consult the appropriate navigation publications.
11. Altimeter — Reset to 29.92 (above 23,500 feet).

CLIMB

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

CRUISE

Note

- During operation in high humidity conditions (ground dew point 20°C or higher) operate canopy defog continuously at altitude in order to insure fog-free canopy during descent. For less severe humidity conditions, canopy defog need not be turned on until just before descent.
- If EGT is increasing, without engine acceleration, so as to exceed allowable limits, reduce thrust to remain within limits.

Refer to Appendix for recommended cruise procedures.

FLIGHT CHARACTERISTICS

Refer to Section VI for flight characteristics of the airplane.

DESCENT

Note

Refer to Section V for limitations applicable to descent. Refer to the Appendix for the recommended descent speeds, time required fuel consumption, etc.

1. Canopy defog switch–CANOPY DEFOG.
2. Cockpit no-fog vent suit switch–ON (if installed). Cabin temperature control knob–AUTOMATIC, midway or hotter (airplanes without cockpit no-fog vent suit switch).

WARNING

On airplanes that do not have a cockpit no-fog vent suit switch, the cabin temperature control knob should be placed in the hot range prior to entering the traffic pattern to prevent cockpit fog during landing or go-around.

3. IFF — Checked.
   Check the IFF within one hour prior to estimated time of landing.
4. Altimeter — Reset to point of descent setting (passing through 24,000 feet).

BEFORE LANDING

1. Zero delay lanyard hook — Attach (if installed).
   If "one & zero" escape system is installed, attach zero delay lanyard hook to ripcord prior to reaching 2000 feet above terrain in accordance with limitations established in the Zero Delay Lanyard Engagement Requirements Chart (figure 3-3) and stow on parachute harness. For minimum ejection altitudes, see the Emergency Minimum Ejection Altitude Table, figure 3-4.

Note

After landing it is not necessary to disconnect the zero delay lanyard, since it connects the ripcord to the timer knob and is not attached to the safety belt. The parachute may be removed from the airplane with the lanyard in the hooked-up condition.

2. Fuel quantity — Check.
3. Boost pump switches — ON.
4. Fuel selector switch — Check.
5. Hydraulic pressures — Check.
6. Radar master switch — STBY.
7. Armament safety switch — SAFE (guard closed).
8. Armament selector switch — SNAKE.
9. Shoulder harness inertia reel handle — UNLOCKED.
10. Entering traffic pattern.
    a. Airspeed — 300-325 KIAS.
    b. Speed brakes — As desired.

Note

In the event of a speed brakes system malfunction, it is possible for one brake to extend while the other remains retracted, resulting in violent yawing. Consequently, it is desirable to open the speed brakes sufficiently early in the pattern to permit immediate retraction and recovery should this condition occur.

    c. RPM 80 to 85%.
       Reduce thrust to decelerate to safe landing gear extension speed.

Note

At no point in the pattern should rpm be less than 75% to assure sufficient engine response in the event of a go-around.

d. Altitude — 1500 feet above field elevation.

11. Downwind.

a. Altitude — 1500 feet above field elevation.

b. Airspeed — 220 KIAS.

12. Before turning base.

a. Landing gear handle — Down; gear checked down; warning light out.

Note

- Refer to Section V for landing gear maximum extension speed and tire ground speed limits.
- Above 200 KIAS with landing gear extended a high frequency buffet occurs. This is normal and is due to airflow in wheel well areas.

b. Altitude — 1500 feet above field elevation.

c. Airspeed — 220 KIAS.

d. RPM — 85-90%.

13. Base (descending).

RECOMMENDED SPEEDS

GROSS WEIGHT	(KIAS)			
	BASE	APPROACH	PRIOR TO FLARE	TOUCH-DOWN
20,000	173	164	159	130
22,000	182	172	167	137
24,000	191	180	175	143
26,000	199	188	182	149
28,000	206	195	189	155
30,000	214	202	196	160

a. Airspeed — 185 KIAS minimum.
Establish descent during turn on base.

14. Final approach.

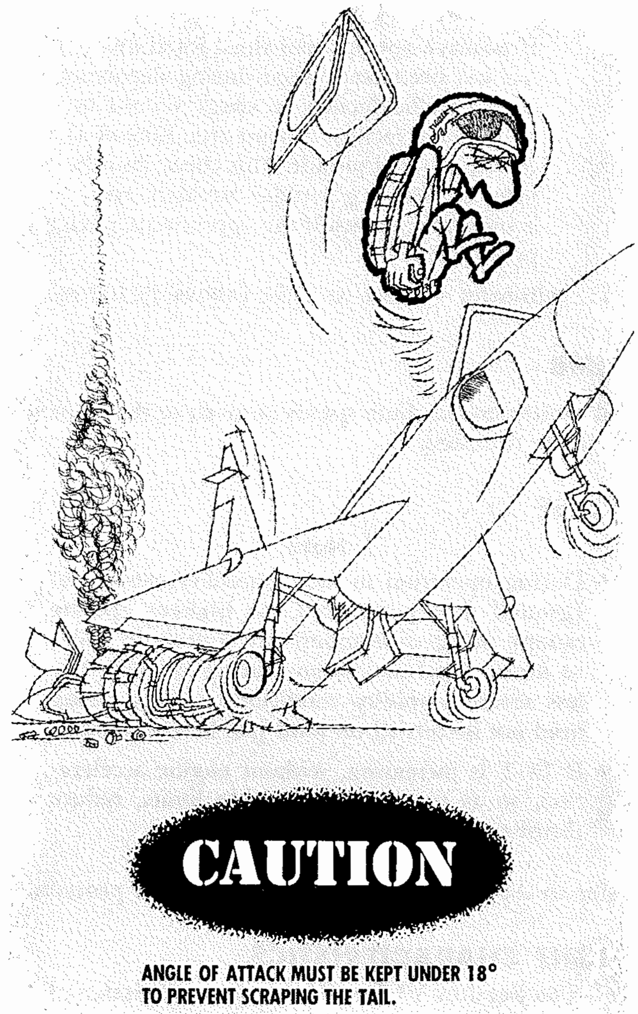
a. Airspeed — 175 KIAS minimum.

CAUTION

If airspeed becomes excessively low, a high sink rate may develop, resulting in a hard landing. See figure 6-1, Minimum Speeds, for information concerning rate of sink vs. airspeed for other than standard conditions. Normally, 85% rpm will be required on final approach to maintain sufficient airspeed.

15. Prior to flare.

a. Airspeed — 170 KIAS minimum.



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LANDING

NORMAL LANDING

A typical landing pattern and recommended procedures are shown in figure 2-6. Refer to Appendix for recommended approach and touchdown speeds at varying gross weights. The landing should be accomplished in a wings-level attitude with the airplane aligned with the runway and an angle of attack approximately 15° (nose on the horizon). Typical landing procedures are as follows:

1. Flare out.

a. Throttle — IDLE.
Reduce thrust to idle during flareout.

WARNING

If engine fails to decelerate to idle rpm (approximately 55 to 65% rpm) when throttle is reduced to IDLE, place the fuel control switch to EMERGENCY to reduce landing roll.

2. Touchdown.
   a. Airspeed — 140 KIAS minimum.
   b. Drag chute — Deploy.

      Drag chute may be deployed immediately after touchdown. There is a slight nose-down pitch when the drag chute is deployed. The drag chute will deploy slowly in some cases if the nose is held off. For faster deployment, the nose may be lowered.
   c. Lower nose wheel to runway.

The nose should be lowered to the runway at approximately 105 KIAS on early airplanes*. On later airplanes** sufficient elevator control will remain at 90 KIAS, and the nose should be lowered at this speed. Wheel braking should be applied as necessary.

Note

- Full length of the runway should be used during landing to reduce brake wear. If conditions permit, delay the use of wheel brakes until the airplane has slowed to 90 KIAS or less to ensure against skidding the tires.
- There may be a slight delay between pedal release and release of braking action.

Directional Control During Landing Roll

Good 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. Rudder, wheel brakes, and nose wheel steering should normally be used for directional control during the landing roll. Nose wheel steering is sensitive at high speed and should not be used until the airplane has been slowed to 80 KIAS or below.

CROSS-WIND LANDING

Before Touchdown

The traffic pattern for a cross-wind landing should be normal, making proper allowances for strength and direction of the cross-wind. 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 wet or dry runway landing, maintain cross-wind correction to touchdown 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.

After Touchdown

Accomplish touchdown on upwind side of runway. The drag chute should be deployed at touchdown. Drag chute deployment at touchdown tends to counteract the downwind weather vaning tendency of the airplane.

CAUTION

Be prepared to jettison chute if excessive turning into the wind occurs as the airplane slows down.

Prior to nose wheel touchdown, directional control should be accomplished by use of rudder and aileron.

Note

After nose wheel touchdown, directional control can be maintained by using rudder, aileron, brakes, and nose wheel steering. Avoid use of nose wheel steering above 80 KIAS due to sensitivity. The use of aileron to "steer" (left aileron to turn left) is effective. Rudder or coordinated control is preferable in high cross-winds, particularly when the nose wheel is on the ground.

Refer to the Takeoff and Landing Cross-Wind Chart in the Appendix to determine cross-wind components and recommended nose wheel touchdown speed for a particular cross-wind condition.

LANDING ON SLIPPERY RUNWAYS

A normal landing pattern should be flown when landing on a slippery runway, planning the touchdown to allow maximum distance for the landing roll. The drag chute should be deployed immediately after touchdown.

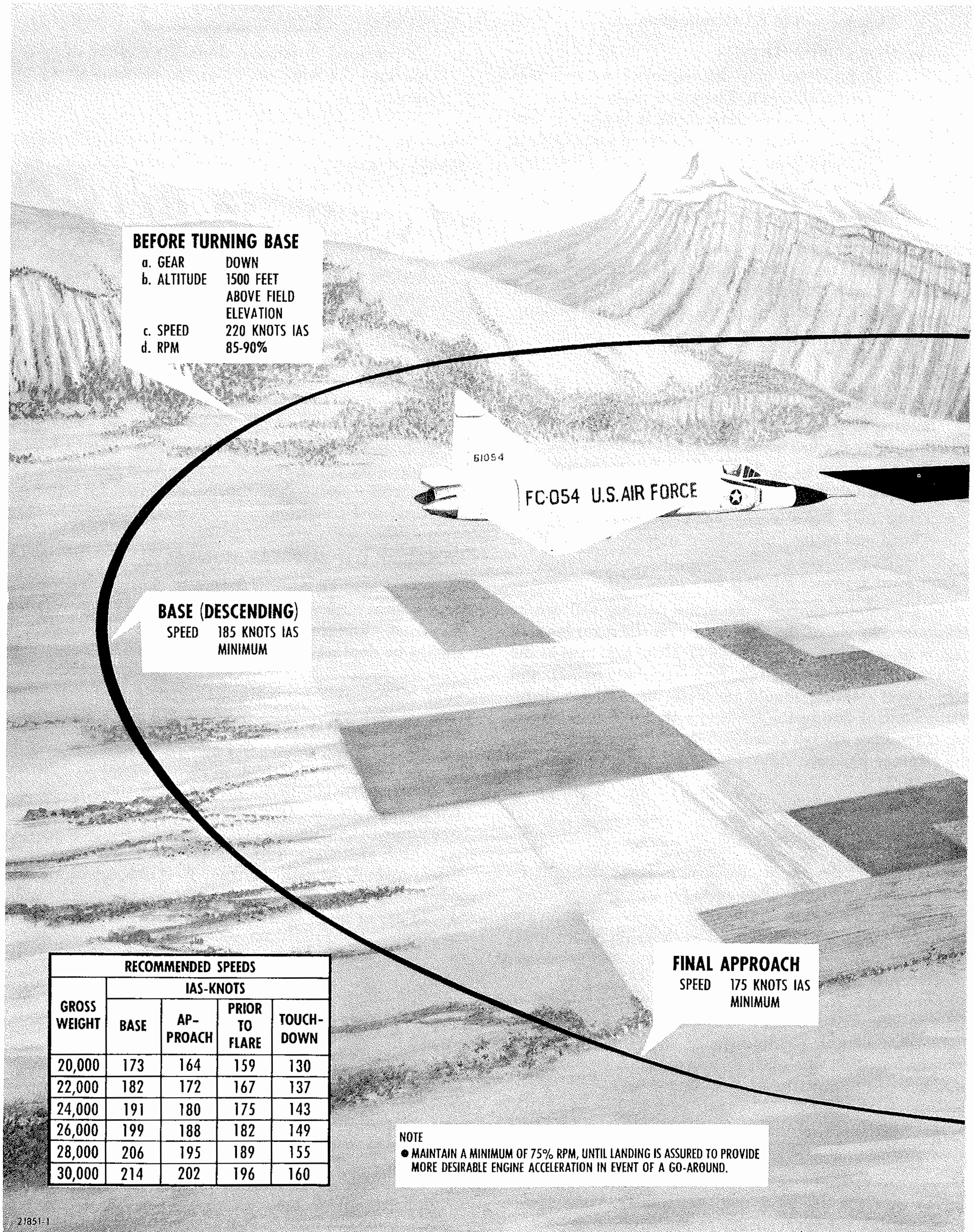
Wet Runways

A normal landing pattern should be used for landing on wet runways. Excessive touchdown speed should be avoided. Establish the final approach speed for the airplane gross weight in accordance with the landing distance charts in the Appendix. Deploy the drag chute at touchdown. During the initial ground roll, maintain a nose-high attitude and use rudder and aileron for directional control. At approximately 90 KIAS, lower the nose wheel to the runway and commence intermittent braking. Avoid use of nose wheel steering above 80 KIAS, due to sensitivity.

CAUTION

Wheel brakes are relatively ineffective until the lift of the wing has dissipated. During the initial ground roll on a wet runway, it is virtually impossible to determine when a wheel has stopped rotating. Therefore, the best assurance against blowing a tire is intermittent braking, with equal pressure applied to both brakes. If the airplane starts to yaw, release both brakes until directional control is regained, then resume intermittent braking. For additional information, refer to USE OF WHEEL BRAKES, Section VII.

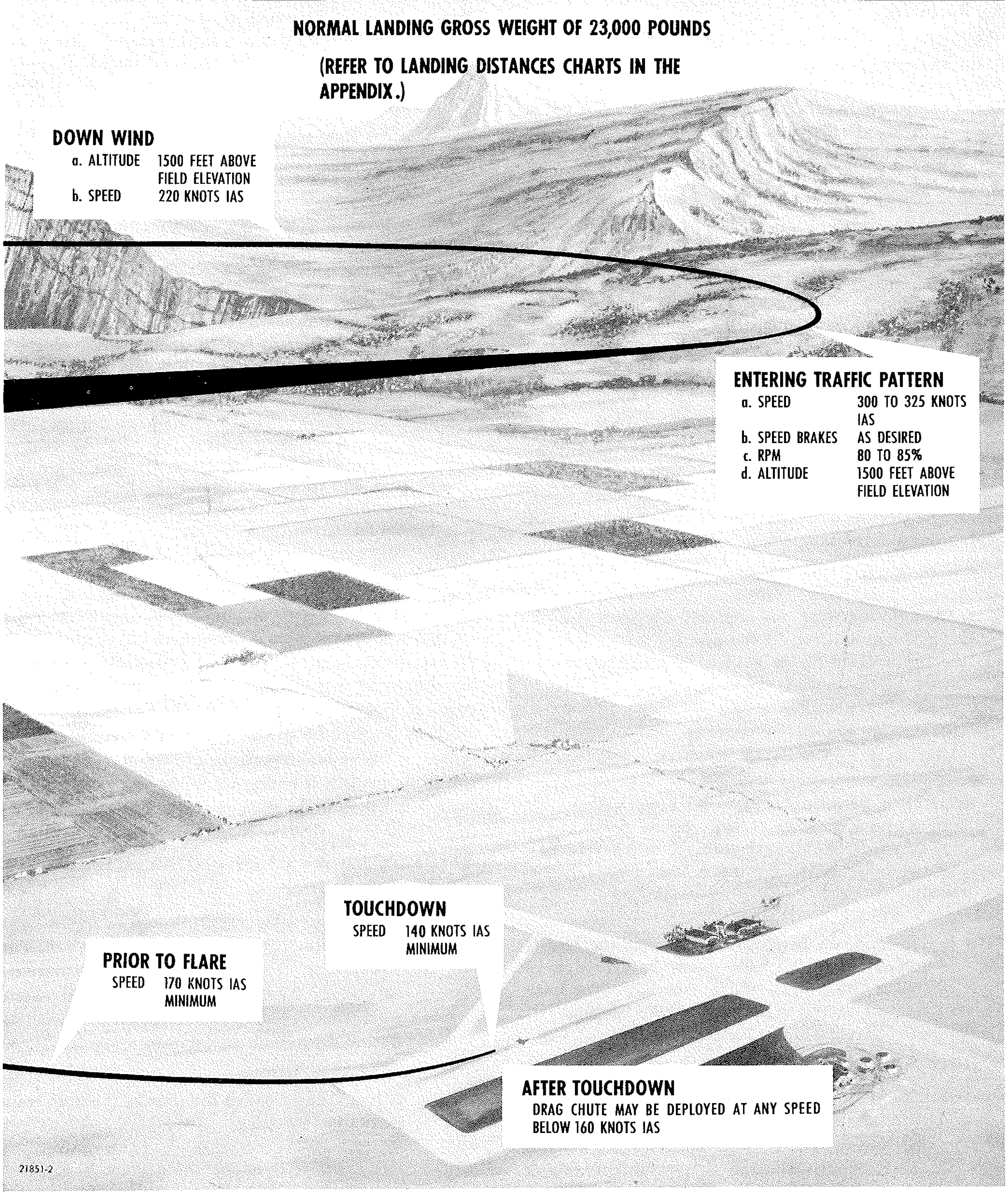
*AF 53-1791 thru 53-1811.
**AF 53-1812 & on.



RECOMMENDED SPEEDS				
GROSS WEIGHT	IAS-KNOTS			
	BASE	AP-PROACH	PRIOR TO FLARE	TOUCH-DOWN
20,000	173	164	159	130
22,000	182	172	167	137
24,000	191	180	175	143
26,000	199	188	182	149
28,000	206	195	189	155
30,000	214	202	196	160

Figure 2-6

normal landing pattern (typical)



If the drag chute fails to deploy, hold the nose up as long as possible to achieve maximum aerodynamic braking. Without the chute the landing roll will be greatly increased.

Icy Runways

The technique for landing this airplane on an icy runway is the same as for a wet runway. However, the reduced friction between the tires and the runway makes directional control more difficult and results in further reduction in brake effectiveness. Consequently it is of the utmost importance to touch down at the minimum speed for the particular airplane gross weight. Use of the drag chute is essential and ice grip tires are recommended.

HEAVY-WEIGHT LANDING

A heavy-weight landing is accomplished by using the same procedures as a normal landing; however, additional power may be required on final approach. Refer to the Appendix for prior to flare and touchdown speeds at varying gross weights.

MINIMUM-RUN LANDING

Use normal approach patterns and speeds. Touch down with the maximum angle-of-attack which will not damage the tail cone. Retard throttle to OFF at touch down, deploy drag chute, and immediately lower the nose and apply maximum braking.

Note

Although the drag chute is not designed for inflight use, it is considered practical to deploy the chute while in the landing attitude at the instant before touchdown.

CAUTION

Braking is permissible above 90 KIAS. However, extreme caution is required to prevent blowouts from skidding the tires, since it is very difficult to feel when a wheel is locked at high speeds.

After a minimum-run landing, or any time the brakes are used excessively, the airplane should be parked away from congested areas until the brakes and tires have cooled. Refer to USE OF WHEEL BRAKES, Section VII.

TOUCH-AND-GO LANDING

Touch-and-go landings are not recommended in this airplane. The usual hazards in takeoff and landing are compounded during touch-and-go landings due to the rapid actions required while rolling at high speeds and flying at low altitude. In addition, tire temperatures may become excessive after a series of touch-and-go landings.

GO-AROUND

A go-around may be made from any point in the approach. See figure 2-7 for typical go-around procedures. The descent attitude is approximately the same as the climb attitude, and the difference between climb and descent is thrust. Inasmuch as relatively high engine rpm is maintained during descent, thrust is almost instantaneously available when the throttle is advanced. No appreciable trim change is experienced when applying power or retracting landing gear. Some yaw may be experienced due to uneven speed brakes retraction at speeds below 180 knots.

Note

- If a go-around is attempted after unsuccessful or successful deployment of drag chute, push drag chute handle in, retract speed brakes, and advance throttle using afterburner as required.
- If drag chute deployment was unsuccessful, the speed brakes should not be opened until after touchdown on subsequent landing.

CAUTION

Be prepared to counteract yawing in the event a speed brakes malfunction causes unequal speed brakes retraction.

AFTER LANDING

Use extreme care when applying brakes immediately after touchdown, or at any time when there is considerable lift on the wings, to prevent skidding the tires and causing flat spots and possible blowout. If maximum braking is required after touchdown, 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 brakes are applied immediately after touchdown than if the same pressure is applied after the full weight of the airplane is on the wheels.

1. Pitch and yaw dampers — Disengage.
2. Speed brakes switch — Neutral.

   The speed brakes switch should be in the "neutral" position prior to drag chute jettison. This will prevent venting hydraulic fluid from the secondary hydraulic system if inadvertent emergency speed brakes extension had occurred during drag chute deployment.
3. Drag chute — Jettison.

   After taxiing clear of the runway the drag chute should be jettisoned.

Note

Drag chute should be jettisoned at slowest speed necessary to maintain parachute inflation.

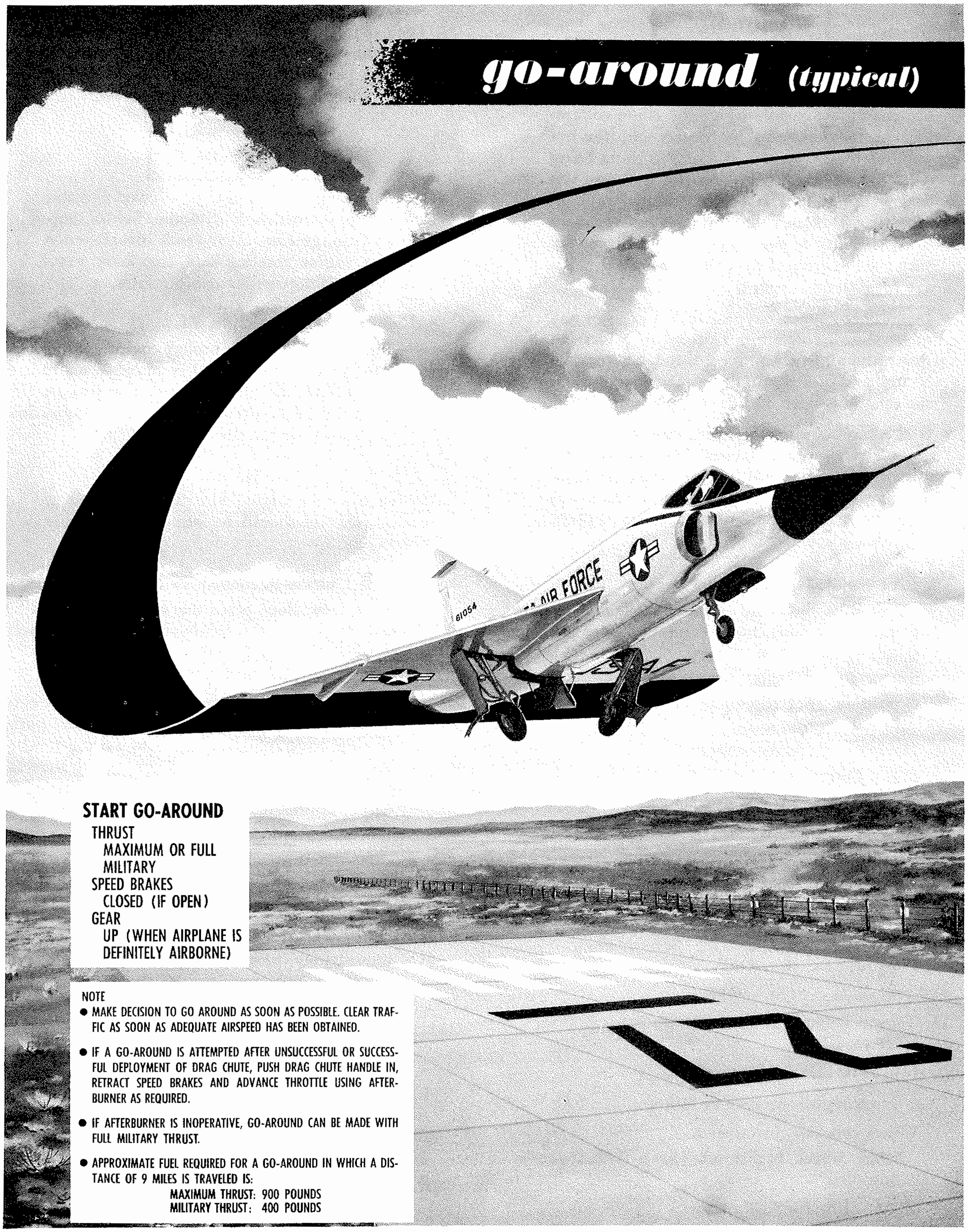

go-around (typical)
START GO-AROUND
THRUST
MAXIMUM OR FULL MILITARY
SPEED BRAKES
CLOSED (IF OPEN)
GEAR
UP (WHEN AIRPLANE IS DEFINITELY AIRBORNE)
NOTE
● MAKE DECISION TO GO AROUND AS SOON AS POSSIBLE. CLEAR TRAFFIC AS SOON AS ADEQUATE AIRSPEED HAS BEEN OBTAINED.
● IF A GO-AROUND IS ATTEMPTED AFTER UNSUCCESSFUL OR SUCCESSFUL DEPLOYMENT OF DRAG CHUTE, PUSH DRAG CHUTE HANDLE IN, RETRACT SPEED BRAKES AND ADVANCE THROTTLE USING AFTERBURNER AS REQUIRED.
● IF AFTERBURNER IS INOPERATIVE, GO-AROUND CAN BE MADE WITH FULL MILITARY THRUST.
● APPROXIMATE FUEL REQUIRED FOR A GO-AROUND IN WHICH A DISTANCE OF 9 MILES IS TRAVELED IS:
MAXIMUM THRUST: 900 POUNDS
MILITARY THRUST: 400 POUNDS

Figure 2-7

CAUTION

- The drag chute should be jettisoned before taxiing downwind in winds exceeding 15 knots because of the possibility of the chute collapsing and risers burning by contact with the hot areas of the exhaust nozzle or the chute being damaged from dragging on the ground. The drag chute should also be jettisoned in high wind or rain if the chute collapses because there is a possibility of the risers burning by contact with the hot areas of the exhaust nozzle or damage from dragging on the ground.
- Insure that helmet, oxygen mask, gloves or any other loose items are not placed in a location that could allow them to be dislodged from the cockpit and sucked into the engine intakes when the canopy is open and the engine running.
- If the canopy is open during taxi operations, observe canopy limit speeds. The canopy hold-open rod must be used on airplanes which do not have a canopy hold button. Keep hands away from canopy sill unless canopy support tool is in place.

4. RAT handle — Pull.

   The RAT should be extended while taxiing in to prevent possible injury to ground personnel.
5. Speed brakes switch — OUT.

   The speed brakes should be in the extended position prior to engine shutdown to facilitate turnaround.
6. Navigation receiver — OFF.
7. IFF — OFF.
8. Nesa switches — OFF.
9. Anti-ice switch — OFF.
10. Windshield rain clear switch — OFF.
11. Cockpit no-fog vent suit switch — OFF (if installed).
12. Canopy defog switch — OFF.
13. Pitot heat switch — OFF.
14. Anticollision light — OFF.
15. Takeoff trim — Set.

ENGINE SHUTDOWN

To shut down engine, proceed as follows:

1. Wheel chocks — Installed.

   Hold wheel brakes on; have wheel chocks installed.
2. RPM — Idle for five minutes.

   Operate engine at idle rpm for five minutes to stabilize engine temperature (taxi time at idle may be included).

Note

The five-minute stabilization period applies only when the engine has been operated above 85% rpm for periods exceeding one minute during the five-minute period prior to shutdown.

CAUTION

If engine temperatures are not allowed to stabilize for approximately five minutes before shutdown, damage can result from interference between engine rotating and stationary parts which have differences in cooling rates.

3. Compressed air — Connected.

   Signal crew chief to connect compressed air for shutdown. If no external compressed air source is available, the combustion starter manual air-valve should be placed in AIRCRAFT position to provide air for clearing.

Note

In the event of engine fire during shutdown, compressed air should be provided to motor the engine.

4. Engine ignition disconnect switch — OFF.

   Have crew chief place the engine ignition disnect switch in the left-hand main wheel well to OFF.
5. Hydraulic systems check – Two-second recovery.

   With the engine operating at idle, check operation of each hydraulic system by smoothly positioning stick fully forward and fully aft. The hydraulic system pressure gage should show a pressure drop and then return to system pressure within approximately two seconds.
6. Radar master switch — OFF.
7. Boost pump switches — OFF.
8. UHF radio — OFF.
9. DC and ac generator switches — OFF.
10. RPM — 70% or above for 30 seconds.

    Immediately prior to shutdown, operate the engine at 70% rpm for 30 seconds to assure complete scavenging of oil system.
11. Throttle — OFF.

Note

Check that engine decelerates freely by listening for any excessive engine noises during shutdown. Leave fuel selector switch to ENGINE (fuel shutoff valve switch to OPEN on some airplanes).

12. RPM zero, fuel selector switch — OFF (some airplanes) if not safety-wired*.

**In accordance with TCTO 1F-102-686.*

Fuel shutoff valve switches — CLOSE (other airplanes).

Allow engine to stop prior to shutting off fuel to provide lubrication for the fuel control unit.

13. Canopy support tool — In place.

WARNING

With low pneumatic air pressure, canopy will drop when dc power is removed if canopy support tool is not in place.

14. Battery switch — OFF.
15. Move elevons to remove hydraulic system pressure.

BEFORE LEAVING AIRPLANE

Before leaving the airplane check the following:

1. Ejection seat safety pin — Installed, streamer visible.

   Insert the seat safety pin from the inboard side** of the right arm rest with the streamer visible.
2. Oxygen supply switch — OFF (if equipped with survival kit) before removing mask or faceplate.
3. Personal equipment leads — Disconnect.

CAUTION

If wearing an automatic opening parachute that has a key attached to the aneroid arming lanyard, make sure key does not foul when leaving cockpit, to prevent chute from being opened inadvertently.

4. Survival kit — Detach from parachute (if installed).
5. Form 781 — Completed.

   Enter 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, and long taxi runs at high speeds, etc.

6. Chocks, landing gear safety pins, and external tank safety switch pin (Part #SE 1100) — Install.

***Outboard side on AF 56-1275 & on & earlier airplanes modified by TCTO 1F-102-751.*

STRANGE FIELD TURN-AROUND PROCEDURES

The following procedures supplement the Normal Abbreviated Check List, providing servicing instructions in check list form for use where ground crew personnel are not familiar with the F-102A airplane. This check list should be used by the pilot only after a thorough briefing by maintenance personnel on all aspects of servicing this airplane. Servicing fluid specifications are contained in the Servicing Diagram, Section I.

Note

These procedures may be removed from the Flight Manual for convenient use as they will not appear in the cardboard check list, T.O. 1F-102A-(CL)1-1.

When required to direct or accomplish servicing of the airplane at a strange field, proceed as follows:

ENGINE OIL TANK SERVICE

Note

The engine oil tank should be filled as soon as possible after engine shutdown (not to exceed five minutes) or oil will drain from the tank into the accesory gear case in sufficient quantities to prevent an accurate oil quantity check.

The oil filler cap is located under the access panel on the top left side of the fuselage just forward of the vertical stabilizer.

1. Check oil level with clean dipstick.
2. Service with oil.
3. Replace and check filler cap for proper installation.

ARMAMENT DOOR OPENING

CAUTION

All armament doors must be fully closed before operation.

1. Check cockpit for the following:
   a. Battery switch — OFF.
   b. Armament selector switch — SNAKE.
   c. Armament safety switch — SAFE (guard closed).
   d. Igniter control switch — Training (safety-wired).

WARNING

Warn personnel and clear armament bay area.

2. Connect 28-volt dc external power or turn battery switch ON.
3. Pneumatic pressure-low warning light — Out (if the warning light is illuminated, have the high-pressure pneumatic system serviced).
4. Armament reset circuit breaker — In (do not push power circuit breaker in).
5. Armament door switch — CLOSE, then release (do not actuate launcher reset switch).
6. Armament door reset switch — RESET and hold. Do not actuate launcher reset switch.

Note

If air is heard entering the door cylinders and the green reset light illuminates, the system is reset and armament door operation can be continued.

7. Armament door reset switch — Release.
8. Armament door switch — OPEN (doors should open).
9. Armament reset circuit breaker — Out (air exhaust should be loud).
10. External power — Disconnect (or) battery switch — OFF.
11. Check that armament doors can be moved with little resistance.
12. Armament bay door safety locks — Installed (if available).

ARMAMENT DOOR CLOSING

To close armament doors, check for launchers up and locked and armament doors fully open. Repeat steps 1 through 3 above, then proceed as follows:

1. Armament bay door safety locks — Removed (if installed).
2. Armament door switch — OPEN, then release (do not actuate the launcher reset switch).
3. Armament door reset switch – RESET and hold.

Note

If air is heard entering the door cylinders and the green reset light illuminates, the system is reset and door operation can be continued.

4. Armament door reset switch – Release.
5. Armament door switch — CLOSE (doors should close).
6. Armament reset circuit breaker — Out (air exhaust should be loud).
7. External power — Remove (or) battery switch — OFF.

BARRIER PROBE OPERATION

To release the barrier probe when in the down position, depress the latch just inside the forward probe housing, raise probe to up position until it catches, then check security.

PRESSURIZING CONSTANT-SPEED DRIVE ACCUMULATORS

The pressurization valve is located in the right-hand engine access compartment by the constant-speed drive gage.

1. Remove valve cover.
2. Connect low-pressure air source.
3. Loosen hex nut approximately one-half turn.
4. Pressurize until gage reads in the green.
5. Tighten hex nut.
6. Disconnect air source.
7. Replace valve cover and check for leaks.

LANDING GEAR STRUT SERVICING

Lowering Strut

1. Remove valve cap cover and loosen valve one-half turn.
2. Depress valve stem, releasing small amount of air at a time, then rock the wing briefly. Check for proper strut extension.

Extending Strut

1. Remove valve cap cover and connect air source.
2. Loosen valve one-half turn and pressurize to proper extension.
3. Tighten valve and remove air source. Replace valve cap, then check for leaks.

DRAINING OVERFULL HYDRAULIC RESERVOIR

Primary Reservoir

The drain line is located inside rear bulkhead of right-hand armament bay.

Note

Open armament doors using procedures previously outlined.

1. Depressurize reservoir by depressing button on top of reservoir cap.
2. Remove drain line cap cover.
3. Hold button on top of reservoir depressed or remove cap to allow excess fluid to drain.
4. Drain until fluid level is 1/4 inch below full mark.
5. Release button. Replace reservoir cap.
6. Replace drain line cap cover and check for leaks.
7. Pressurize reservoir and recheck fluid level.

Note

If fluid level decreases more than 1/4 inch, excessive air is in the system.

8. Close armament doors as previously outlined.

Secondary Reservoir

The drain line is located on the forward side of the right-hand wheel well. The procedure for draining the secondary reservoir is the same as for the primary except that the fluid level should not decrease more than 1/8 inch when the reservoir is pressurized.

SERVICING HYDRAULIC RESERVOIR AND ACCUMULATORS

The reservoir and accumulator air charge fittings are located in the RAT compartment.

1. Operate control stick to bleed system pressure.
2. Extend the RAT.
3. Primary and secondary accumulator air gages — 750 (±25) psi. If air pressure is low, charge with dry air or nitrogen to 750 (±25) psi and check for leaks.

Note

To pressurize accumulators, remove cap from fitting, connect air source, loosen fitting one-half turn, then apply pressure. Tighten fitting prior to removing air source.

4. Check hydraulic reservoir sight gages for correct fluid level.
5. If fluid level is at refill, release air pressure by depressing the button on top of the reservoir cap. Remove reservoir cap and service with hydraulic fluid to 1/4 inch below full mark.
6. Replace reservoir cap.
7. Pressurize reservoirs with low-pressure air source. The quick-disconnect is located at the top center of the RAT compartment.

Note

The airplane is unacceptable for flight if the fluid level in the primary reservoir decreases over 1/4 inch or the secondary decreases over 1/8 inch after pressurization.

REFUELING (SINGLE-POINT ONLY)

Note

If refueling source with single-point adapter is not available and the airplane is equipped with external tanks, the airplane may be fully serviced by filling the external tanks, then starting the engine and transferring fuel into the internal wing tank system. Repeat this procedure until the airplane is fully serviced.

Refuel the airplane with JP-4. Refer to Section V for emergency fuel and limitations. The filler adapter is located on the rear bulkhead of the left wheel well. Remove the red dust cap from refueling adapter and attach fuel hose nozzle. Shutoff valve on nozzle shall be in off position. Start pump on fuel truck and adjust fuel pressure to 55-60 psi. Open shutoff valve on fuel hose nozzle and maintain 55 to 60 psi. Check for normal exhaust of air from left or right fuel tank vents at the beginning and during refueling.

CAUTION

If air does not flow from each wing vent, shut off fuel flow and investigate the trouble.

A normal discharge of fuel from refueling adapter drain will occur after refueling hose is removed. Tank should be allowed to completely drain before filler cap is installed. Check fuel filler cap installed with arrows aligned.

Note

Check fuel service against remaining fuel to insure full load.

HIGH-PRESSURE PNEUMATIC SYSTEM SERVICING

The high-pressure pneumatic system filler valve and pressure gage are located in the left main wheel well. Use high-pressure compressor (MC-1 or like equipment) which will produce 3200 psi. Service to 3000 psi. All precautions must be exercised to prevent contamination of the high-pressure pneumatic system.

CAUTION

Insure that combustion starter manual air valve in the left main wheel well is in the GROUND supply position. If high-pressure air is not available, air-motoring of starter is to be minimized.

WEATHER PROCEDURES

Canopy

The canopy should be left open (ground support tool installed) when exposed to direct rays of the sun at temperatures above 85°F. If the ground support tool is not available, leave the canopy cracked.

RADOME ANTI-ICING TANK SERVICE

The radome anti-icing tank is located in the right forward electronics compartment.

1. Release air pressure by depressing the release valve on filler cap.
2. Fill tank with anti-ice fluid (40% water — 60% glycol).

DRAG CHUTE INSTALLATION

1. Speed brakes — OPEN.
2. Speed brakes ground safety locks — Installed.
3. Inspect drag chute compartment for metal tears, cleanliness, screws flush and tight.

4. Square drag chute pack (risers should be 26 inches long).
5. Check riser gathering strap fully against "D" ring and check to see safety pin is properly aligned in pilot chute.
6. Place speed brake safety pin in position.
7. Place "D" ring in jaws.
8. Drag chute handle — Out. This will hold "D" ring securely in the jaws during installation).
9. Lay risers on bottom of drag chute compartment and install chute on top of risers. (Hold pressure against jaws to keep risers from jamming or batting up against "D" ring.)
10. Remove ground safety aft pilot chute pin.
11. Place top retainer strap over cone.
12. Place bottom retainer strap over cone.
13. Drag chute handle – In.
14. Install ball of ripcord pin in cable guide. Assure ball is in race and fingers on top of ball. Install ripcord pin in pilot chute cone.

Note

Do not pull pilot chute inner safety pin.

15. Drag chute handle – Out. If ripcord pin comes out (normal operation), repeat steps 13 and 14.

Note

If ripcord pin does not come out on operational check, a cocked "D" ring or malfunction exists and installation should not be continued until malfunction is corrected.

16. Pull pilot chute safety pin and do not touch drag chute thereafter.

COMBUSTION START — SECOND ATTEMPT

The following procedure should be followed to manually replenish the starter fuel flask. Gain entrance to the fuel flask through the right-hand engine access door.

1. Remove bleed line cap cover from starter flask bleed line.
2. Apply approximately 10 psi of air to fuel flask bleed line.
3. Listen for bottoming of the plunger in starter flask.
4. Remove air source and allow fuel to gravity feed from fuel tanks to fill starter fuel flask. Use a suitable container to collect fuel drainage from fuel flask.
5. Install bleed line cap cover.
6. Check for leaks.

Note

The Normal Abbreviated Check List is now contained in T.O. 1F-102A-(CL)1-1.

TABLE OF CONTENTS

Note

Critical actions, which must be performed immediately and instinctively if the emergency is not to be aggravated or damage is to be avoided, are listed in bold face capital letters.

ENGINE FAILURE

Generally flameouts occur from improper fuel scheduling. Failure of components outside of the engine proper can often be detected by reference to the engine instruments or warning lights before a flameout occurs. Air starts should be attempted to restore engine thrust unless mechanical failure occurs within the engine. Should an engine mechanical failure occur, the fuel supply should be shut off by retarding the throttle to OFF, placing the fuel selector switch to OFF (fuel shutoff valve switches to CLOSE on some airplanes), and placing boost pump switches to OFF. Pilot's discretion should determine the action to be taken following an engine failure. Refer to EJECTION VS. FORCED LANDING, this Section.

Note

The airplane has normal flight characteristics with a dead engine except above speeds of approximately 220 KIAS. Above these airspeeds, the interruption of airflow through the engine inlet ducts creates a duct rumble which causes a very startling buffet and heavy vibration within the airframe. The severity of this rumble increases proportionally with IAS and is only slightly perceptible at best glide speed of 220 KIAS.

COMPRESSOR STALL

A breakdown of compressor airflow is known as a compressor stall. The compressor stall is induced by a change in the pattern of airflow being fed to the engine. The intensity of the stall is determined by the magnitude of the airflow pattern change and may affect a number, or all of the compressor blades. Stalls may be encountered at any speed. Retarding or advancing the throttle too rapidly can cause either deceleration or acceleration stalls, respectively. Without throttle movement, rapid changes of flight condition which distort the inlet duct airflow pattern can induce compressor stall under certain atmospheric conditions. Compressor stalls may be experienced during operation at high altitudes in areas of heavy ice crystal concentration such as found in or around thunderstorms.

Note

Refer to TURBULENCE AND THUNDERSTORMS, Section IX, for applicable procedures.

The ice crystals entering the inlet duct go into the engine with the air where they are heated during the compression process and become effectively, ingested water. The ingested ice crystals reduce the compressor stall margin, and compressor stall or flameout follows. These 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. RPM drops to 80% or below, fuel flow drops to 800 pph and temporary loss of EPR occurs when a series of stalls are experienced. Fuel flow and rpm will fluctuate when individual stalls are encountered.

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 operations 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. There are several things that can be done to avoid compressor stall or to reduce its intensity. Erratic and abrupt throttle movements should be avoided. Rapid throttle advances during periods of high distortion of the air entering the air inlet duct, such as at low airspeeds, can cause acceleration stalls. Coordinated flying increases the efficiency of the compressor inlet air duct. Airspeeds should be maintained above the acceptable minimum. Following a single loud explosion, immediately check for signs of fire (trailing smoke or flames and fire warning system indications) to determine whether a compressor stall or engine failure has occurred. When compressor stalls occur, or when operating in heavy ice crystal concentrations, proceed as follows:

1. Ignition button — Depress and hold (airplanes with all-points ignition.) Move throttle inboard if in AFTERBURNER.

CAUTION

Continuous use of the ignition system in excess of ten minutes, or consecutive continuous usages without a ten-minute interim cooling period may result in damage to the ignition system which will render it inoperable for restarts. Energizing the ignition system will not prevent compressor stall, but it will aid in preventing flameout.

2. Check engine instruments for engine mechanical failure and overtemperature.
3. Do not retard throttle unless engine failure or overtemperature is evident.

Note

If overtemperature condition exists, retard throttle slowly to prevent exceeding EGT limits. During operation above 40,000 feet, avoid retarding throttle below 85% rpm.

4. Maintain normal coordinated straight and level flight attitudes until engine stall condition is relieved.

Note

Maintain unaccelerated flight until engine stall condition is relieved, if possible. If not, maintain altitude and heading until airspeed drops to 220 KIAS then establish glide speed of 220 KIAS and maintain heading. Do not open speed brakes.

5. If an overtemperature persists, fuel control switch to EMERGENCY.
6. **IF EGT EXCEEDS MAXIMUM ACCELERATION LIMIT, THROTTLE TO OFF.**

   Retard throttle to OFF if above procedures do not keep EGT with maximum acceleration limits (refer to ENGINE OPERATING LIMITS, Section V).
7. Restart engine.

   After a stabilized condition is obtained, restart the engine. Refer to ENGINE AIR START, this Section, for restarting procedures.

Note

Record on Form 781, any compressor stall and indicate duration and peak temperatures of any overtemperature operation.

Off-idle compressor stalls (sometimes referred to as "choo-choo" or acceleration stalls) may be experienced between idle and approximately 80% rpm while accelerating during engine ground runup. If a stall occurs as the throttle is being advanced in this range, momentarily stop or slow throttle advancement as required to correct the stalled conditions. Off-idle stall or "choo-choo" is of no consequence unless it is severe to the degree that more than 15 seconds are required for the engine to accelerate from 65% to FULL MIL POWER. If more than 15 seconds are required, the engine should be shut down and the cause determined before flight.

ENGINE FAILURE DURING TAKEOFF

Engine Failure During Takeoff Run (Before Airborne)

If engine failure occurs before the airplane is airborne, abort takeoff; refer to ABORT, this Section.

Engine Failure During Takeoff (With Airplane Airborne)

If engine failure occurs during takeoff but after becoming airborne, an immediate relight may be possible on airplanes with all points ignition*. If this is unsuccessful, the proper course of action depends on altitude at time of failure, automatic escape equipment capabilities (see figure 3-4), and terrain features of the available landing area. If the decision is to land, proceed as follows:

1. **THROTTLE — OFF.**
2. RAT handle — Pull & hold for four seconds.

   Pull the RAT handle to displace the ram air turbine. Hold the handle in the fully extended position for four seconds to insure a satisfactory extension of the turbine.
3. **LANDING GEAR HANDLE — DOWN.**
4. External tanks jettison button — Depress (if required).

   Depress the external tanks jettison button to jettison the external tanks if tanks are installed and contain fuel. If external tanks are empty, it is recommended that they be retained to cushion the impact.
5. **CANOPY — JETTISON (IF NECESSARY).**

   Raise the canopy jettison handle on the left-hand armrest if it is necessary to jettison the canopy.

WARNING

- If canopy is to be jettisoned, make sure it is jettisoned before touchdown. Otherwise, sparks from the canopy remover may cause a fire if fuel is spilled in the vicinity of the airplane, or the canopy may become jammed.
- If the canopy is jettisoned below 175 KIAS while airborne it may strike the fin but probably will not result in loss of control.

6. Shoulder harness inertia reel handle — LOCKED.

CAUTION

The pilot is prevented from bending forward when the shoulder harness is locked; therefore, all switches not readily accessible should be "cut" before moving the shoulder harness inertia reel handle to LOCKED position.

7. Fuel selector switch — OFF (some airplanes).

   Fuel shutoff valve switches — CLOSE (other airplanes).
8. **DRAG CHUTE HANDLE — PULL (UPON CONTACT).**
9. Master switch — OFF (some airplanes); TRIP, then OFF (other airplanes).
10. **ABANDON AIRPLANE AS SOON AS POSSIBLE.**

WARNING

Because of the height of the cockpit above the ground (eight feet), care should be exerted when abandoning the airplane without a ladder to prevent bodily injury.

ENGINE FAILURE DURING FLIGHT

(Air Start Not Probable)

Note

If engine failure occurs but there is no indication of mechanical failure within the engine, an air start may be attempted. Refer to ENGINE AIR START, this Section.

When an air start is not probable, use the following procedure:

1. Throttle—OFF.

Airplanes modified by TCTO 1F-102-746.

2. Airspeed 220 KIAS (gear up, speed brakes closed).
   Establish a glide speed of 220 KIAS with gear up and speed brakes closed.
3. AC bus switch—EMER (some airplanes); RESET, then ON (other airplanes).
   Engine windmilling rpm is insufficient to provide normal ac generator operation. The emergency ac generator should be energized to supply power for flight and engine instruments.

Note

A "frozen" engine will result in complete loss of ac power.

4. RAT handle—Pull (if necessary) and hold for four seconds.
   Pull the RAT handle to displace the RAT if immediate emergency flight control hydraulic pressure is necessary. The handle should be held in the fully extended position for four seconds to insure a satisfactory extension of the ram air turbine.

WARNING

- If engine failure results in a frozen engine, it will be necessary to immediately displace the RAT to obtain hydraulic pressure for flight control operation.
- If airspeed is above RAT maximum extension speed and engine is frozen, do not extend the speed brakes to slow the airplane. Speed brakes 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.

Note

If engine failure results in a windmilling engine, there will normally be sufficient hydraulic pressure available for flight control operation, unless airspeed is low, without extending the RAT. Rapid movement of the flight controls should be avoided. Once extended the RAT cannot be retracted during flight.

5. Fuel control switch—EMER (if required).
   If fuel control failure is suspected, place fuel control to EMER and check that emergency fuel control warning light illuminates.
6. Fuel selector switch—ENGINE (some airplanes).
   Fuel shutoff valve switches—OPEN (other airplanes).
7. Nonessential electrical equipment—OFF.
   Turn nonessential electrical equipment off to reduce battery load.

CAUTION

At engine speeds below approximately 40% rpm, dc generator output is not available and the battery becomes the only source of power to the dc essential bus. Usable battery power is available for approximately 5 to 20 minutes. On airplanes equipped with an emergency dc bus*, the transformer-rectifier will automatically be connected to the emergency dc bus when dc generator output is lost. The ac system is then self-sustaining and will not be affected by battery power depletion.

Note

Attempt an air start if there is no indication of mechanical failure within the engine. Refer to ENGINE AIR START, this Section.

ENGINE FAILURE DURING FLIGHT AT LOW ALTITUDE

If the engine fails during flight at extremely low altitude, and sufficient airspeed is 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, establishing forced landing pattern, ejection, etc.).

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. If the decision is to continue attempting air starts, the climb should be terminated before the airspeed drops below best glide speed in order that engine windmilling rpm will not drop below the minimum required for air start.

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 external tanks, consideration must be given to such factors as sufficient airspeed to

**Airplanes modified by TCTO 1F-102-727.*

allow time for pilot reaction and an unpopulated area where the tanks will fall. In any event, the decision to jettison or retain external loads must be made by the pilot on the basis of his evaluation of the above factors and conditions existing at the time of the emergency.

ENGINE AIR START

Engine air starts are accomplished by utilizing the same basic sequence as a normal ground start but using windmilling effect rather than the starter to motor the engine. An air start should be attempted as soon after flameout as possible regardless of airspeed. If this immediate attempt is unsuccessful, there are certain conditions that should be established. Without starter operation, airspeed must be controlled to obtain optimum rpm for an air start. The best engine windmilling speed for obtaining an engine air start is between 15 and 30% rpm. The best glide speed of 220 KIAS will provide this rpm between sea level and approximately 35,000 feet. Above this altitude, at 220 KIAS, the minimum stabilized rpm will increase by approximately one percent per 1000 feet. Due to excessive altitude loss at other than best glide speed, it is recommended that 220 knots be used for all air starts, regardless of altitude. The probability of a relight at 220 KIAS is increased below approximately 35,000 feet. A windmilling engine will not deliver ac power directly through the main ac generator, but will produce and maintain sufficient hydraulic pressure to drive the emergency ac generator. In the event of a flameout condition, it is necessary to switch to the emergency ac generator to maintain power for flight and engine instruments and under night conditions, instrument panel and console lights. See figure 3-1 for engine air start procedure.

Unsuccessful Air Start

1. Throttle — OFF.

   Retard the throttle to OFF if during an air start any one of the following conditions prevail:

   a. Light up does not occur within 20 seconds after throttle has been advanced to IDLE if there was positive fuel flow indication.

   Note

   If source of flameout is due to a temporary interruption of fuel flow from the fuel tanks, restart may take up to four minutes. This condition can be detected by absence of fuel flow indication.

   b. Engine fails to accelerate to idle within approximately 45 seconds after lightup.

2. Attempt another air start.

If the engine cannot be restarted on the normal or emergency fuel systems, prepare to make a forced landing or abandon the airplane. Refer to applicable procedures, this Section.

MAXIMUM GLIDE

Maximum glide distances with a windmilling engine are obtained with an airspeed of 220 KIAS. See figure 3-2 for additional information.

FIRE

A steady illumination of the fire warning light is an indication of a fire in the forward engine compartment. A flashing fire warning light is an indication of a fire in the aft engine compartment. The hot section is enclosed in an engine-mounted fireproof cannular shroud and there is no lateral firewall; thus, a flashing light could be due to a fire between the engine shroud and the airplane fuselage structure and not merely an overheat of the engine and/or airplane structure. Therefore, a warning from one compartment can be considered no less serious than a warning from the other. A fire warning may be the result of hot gases leaking from the engine or fire fed by fuel leaking from the engine and thus is a function of engine thrust. Following a fire warning in flight, engine thrust should be immediately reduced below the operating point of the afterburner mechanical shutoff (approximately 80% thrust) and then eased off until the fire warning light extinguishes or the minimum thrust necessary to maintain safe ejection altitude is reached. In flight the throttle should be stop-cocked only after a positive verification of fire has been made. Following a fire warning on the ground, however, where power is not required, the engine should be stop-cocked immediately.

ENGINE FIRE DURING STARTING

Fire Warning Light Illuminated During Start

If the engine fire warning light illuminates or there is visible evidence of fire during starting or ground operations proceed as follows:

1. **THROTTLE — OFF.**
2. Fuel selector switch — OFF (some airplanes).

   If the switch is safety-wired and has been rotated or safety wire broken, it must be re-established that both valves are open before the next flight and before safety wire is replaced.

   Fuel shutoff valve switches — CLOSE (other airplanes).
3. Fire fighting equipment — Summon.
4. Master switch — OFF (some airplanes); TRIP, then OFF (other airplanes).
5. Abandon the airplane.

Excessive Exhaust Gas Temperature During Start

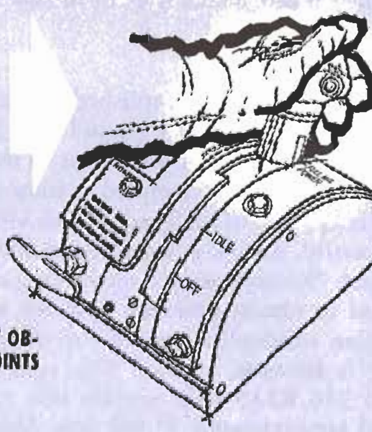
1. Starting air — Connected.

   Signal crew chief to connect a supply of compressed air to the high-pressure pneumatic ground connection for motoring.

IMMEDIATE AIRSTART

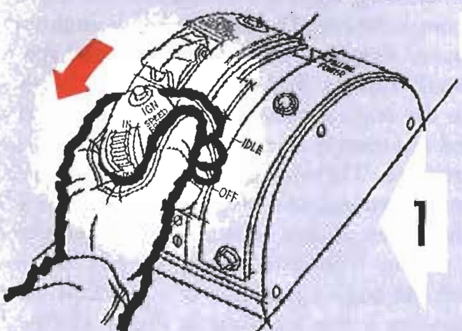
IGNITION BUTTON—DEPRESS (AIRPLANES WITH ALL POINTS IGNITION).

ATTEMPT IMMEDIATE AIR START BY DEPRESSING AND HOLDING THE IGNITION BUTTON. MOVE THROTTLE INBOARD (IF IN AFTERBURNER). SLOWLY RETARD TO IDLE AS RPM DROPS.



RESTARTING

(IMMEDIATE AIRSTART NOT POSSIBLE)



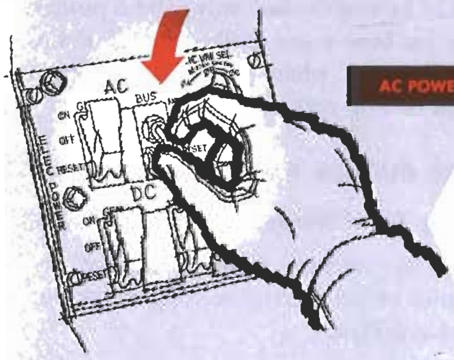
1

THROTTLE—OFF (IMMEDIATE RELIGHT NOT OBTAINED AND AIRPLANES WITHOUT ALL POINTS IGNITION).

AIRSPEED 220 KIAS

DUE TO EXCESSIVE ALTITUDE LOSS AT OTHER THAN BEST GLIDE SPEED, 220 KNOTS IAS IS RECOMMENDED FOR ALL AIR STARTS. THE PROBABILITY OF A RELIGHT AT 220 KNOTS IAS INCREASES BELOW APPROXIMATELY 35,000 FEET.

2



AC POWER FAILURE

3

AC BUS SWITCH—EMER (SOME AIRPLANES); RESET, THEN ON (OTHER AIRPLANES).

ENGINE WINDMILLING RPM IS INSUFFICIENT TO PROVIDE NORMAL AC GENERATOR OPERATION. THE EMERGENCY AC GENERATOR SHOULD BE ENERGIZED TO SUPPLY AC POWER FOR FLIGHT AND ENGINE INSTRUMENTS.

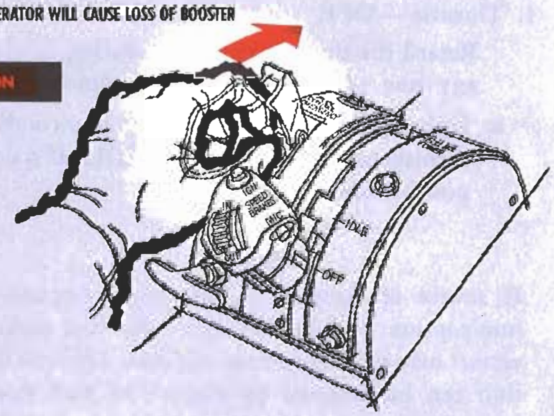
NOTE
LOSS OF MAIN AC GENERATOR WILL CAUSE LOSS OF BOOSTER PUMP OPERATION.

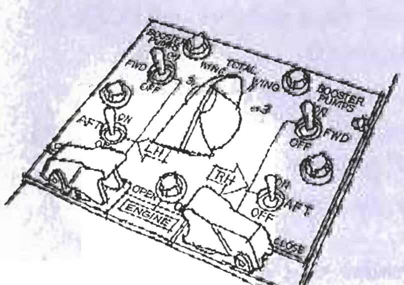
EMER FUEL ON

FUEL CONTROL SWITCH—EMERGENCY (IF REQUIRED)

IF FUEL CONTROL FAILURE IS SUSPECTED, PLACE FUEL CONTROL SWITCH TO EMERGENCY AND CHECK THAT EMERGENCY FUEL CONTROL WARNING LIGHT ILLUMINATES.

4





5

FUEL CONTROL PANEL—CHECK

FUEL SELECTOR SWITCH—ENGINE (SOME AIRPLANES).

FUEL VALVE SHUTOFF SWITCHES—OPEN (OTHER AIRPLANES).

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Figure 3-1

engine air start procedures

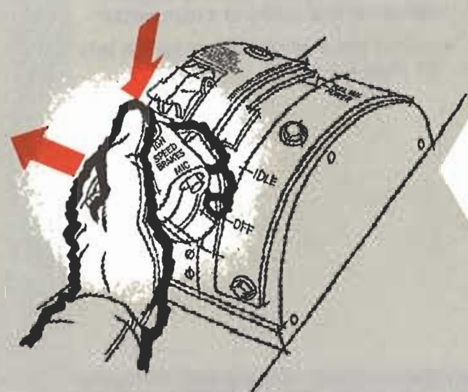
RPM 15% TO 30%

AIRSTARTS SHOULD BE MADE WITH ENGINE WINDMILLING BETWEEN 15 AND 30 PERCENT RPM. AN AIRSPEED OF 220 KNOTS IAS WILL PROVIDE THIS RPM BETWEEN SEA LEVEL AND APPROXIMATELY 35,000 FEET. AS ALTITUDE INCREASES, RPM WILL INCREASE.

6

WARNING

IF HYDRAULIC PRESSURE FOR FLIGHT CONTROL OPERATION BECOMES MARGINAL AT LOW ENGINE RPM, EXTEND THE RAT



7

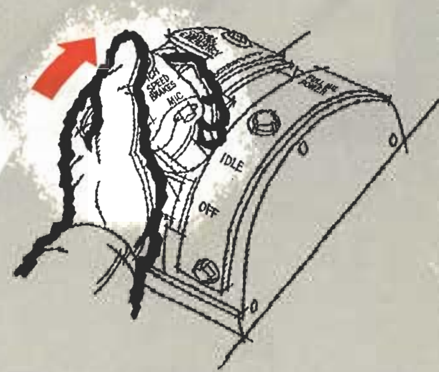
DEPRESS AND HOLD IGNITION BUTTON; THROTTLE OUTBOARD TO START. (AIRPLANES WITHOUT ALL POINTS IGNITION.)

NOTE

IT IS NECESSARY TO MOVE THE THROTTLE FULL OUTBOARD TO START POSITION TO ARM THE IGNITION CIRCUIT ON SOME AIRPLANES.

THROTTLE—IDLE

8



9

RPM 60% TO 80%, THEN RELEASE IGNITION BUTTON

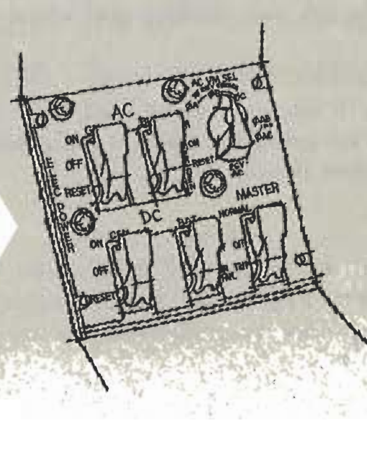
AFTER RESTART

1. DC GENERATOR SWITCH—RESET THEN ON
2. AC BUS SWITCH—NORMAL
3. BOOSTER PUMP SWITCHES—OFF

NOTE

TURN BOOSTER PUMPS OFF TO REDUCE LOAD ON AC GENERATOR.

4. AC GENERATOR SWITCH—RESET THEN ON
5. BOOSTER PUMP SWITCHES—ON (ONE AT A TIME)



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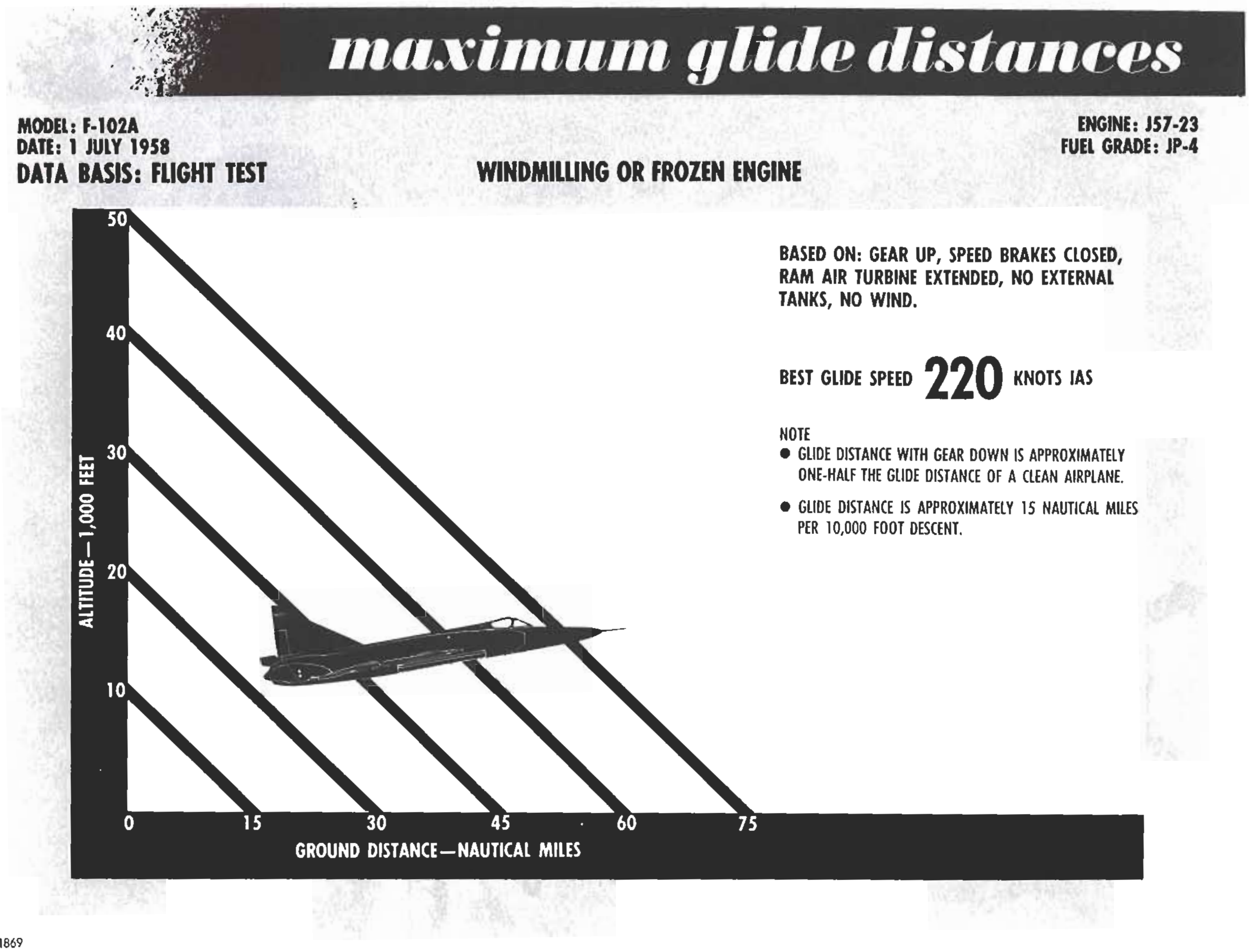

Figure 3-2

Note

In the event that a ground cart is not available, air for motoring the engine may be taken from the airplane high-pressure pneumatic system supply flasks by opening the manual shutoff valve in the left main landing gear wheel well.

2. Engine ignition disconnect switch — OFF.

   While in the main wheel well, crew chief lifts the switch guard and actuates the engine ignition disconnect switch.

Note

With the switch in the OFF position, the combustion starter may be fired but the engine will not light up since the engine ignition has been disconnected.

3. **THROTTLE — OFF.**
4. Boost pump switches — OFF.
5. Fuel selector switch — OFF (some airplanes).

   If the switch is safety-wired and has been rotated or safety wire broken, it must be re-established that both valves are open before the next flight and before safety wire is replaced.

   Fuel shutoff valve switches — CLOSE (other airplanes).
6. Throttle — START.

   Move throttle outboard to START to air motor the starter. Hold this position until rpm indication is evident on the tachometer.
7. Ignition button — Depress.
8. Throttle — OFF.

   As soon as definite rpm indication is noted, hold ignition button and move throttle inboard to the OFF position. This will fire the combustion starter and aid in clearing the engine.

9. Ignition button — Release.

   Release the ignition button after the engine clears or if the fire cannot be extinguished by clearing.

10. Abandon the airplane.

ENGINE FIRE DURING TAKEOFF

The exact procedure to follow for a fire warning during takeoff depends on the condition of the emergency. Airspeed, altitude, length of runway and overrun available, location of populated areas, etc., have to be considered before the required action is taken. The following procedures are recommended:

1. If runway and overrun area permit, abort takeoff; refer to ABORT, this Section.
2. If committed to takeoff:
   a. External tanks jettison button — Depress (if required).

      Jettison the external tanks if tanks are installed and contain fuel and drop area is unpopulated.

   b. **MAINTAIN MAXIMUM THRUST UNTIL SAFE EJECTION ALTITUDE IS REACHED.**

      Adjust thrust to maintain safe ejection altitude.

   c. Check for fire.

      Check for positive indications of fire, such as abnormal engine instrument readings, smoke in cockpit, trailing smoke or flame, or report from ground or another airplane.

   d. **IF FIRE EXISTS — EJECT.**

   e. If fire cannot be confirmed, make decision to land or eject.

Note

See figure 3-4 for emergency minimum ejection altitudes.

ENGINE FIRE IN FLIGHT

1. **REDUCE THRUST.**

   Reduce thrust to minimum necessary to maintain safe ejection altitude.

CAUTION

At high altitude, compressor stall may occur as a result of large thrust reductions. Exhaust gas temperature may rise as a result of compressor stall and should not be taken as a positive indication of fire.

2. If warning light extinguishes:
   a. Check warning light.
   b. Continue flight at minimum safe thrust.
   c. Land as soon as practicable.
3. If warning light remains on — Check for fire.

   Determine whether a fire actually exists by a report from another airplane or the ground, fumes, heat, cockpit smoke, trailing smoke following a turn, abnormal airplane responses, or abnormal engine instrument readings.

4. Fire not evident by check:
   a. Continue flight at minimum safe thrust.
   b. Land as soon as possible.
5. Fire evident:
   a. Throttle — OFF.
   b. Fuel selector switch — OFF (some airplanes). Fuel shutoff valve switches — CLOSE (other airplanes).
   c. Master switch — OFF (some airplanes); TRIP, then OFF (other airplanes).

Note

The fire warning system is deactivated by turning off the master switch and the fire warning light will go out.

6. If fire ceases — Eject or make forced landing.
7. **IF FIRE CONTINUES — EJECT; REFER TO EJECTION PROCEDURES, THIS SECTION.**

ENGINE FIRE AFTER SHUTDOWN

If engine fire is suspected after engine shutdown on the ground, use the following procedure for clearing the engine:

1. Starting air — Connected.

   Signal crew chief to connect a supply of compressed air to the high-pressure pneumatic ground connection for motoring.

Note

In the event that a ground cart is not available, air for motoring the engine may be taken from the airplane high-pressure pneumatic system supply flasks by placing the manual shutoff valve in the left main landing gear wheel well to the AIRCRAFT position.

2. Engine ignition disconnect switch — OFF.

   While in the main wheel well, crew chief lifts the switch guard and actuates the engine ignition disconnect switch.

Note

With the switch in the OFF position, the combustion starter may be fired but the engine will not light up since the engine ignition has been disconnected.

3. **THROTTLE — CHECK OFF.**
4. Boost pump switches – Check OFF.
5. Fuel selector switch — OFF (some airplanes).
   Fuel shutoff valve switches — CLOSE (other airplanes).
6. Ignition button – Depress and hold.
7. Throttle — START.
   Move throttle outboard to START to air motor the starter. Hold this position until rpm indication is evident on the tachometer.
8. Throttle — OFF.
   As soon as definite rpm indication is noted, hold ignition button and move the throttle inboard to the OFF position. This will fire the combustion starter and aid in clearing the engine.
9. Fire fighting equipment — Summon (if necessary).
10. Ignition button — Release.
    Release the ignition button after the engine clears or if the fire cannot be extinguished by clearing.
11. Abandon the airplane (if fire continues).

ELECTRICAL FIRE

There is no system to warn of electrical fire in this airplane. Circuit breakers protect most of the circuits and tend to prevent electrical fire. If an electrical fire occurs, however, and its source cannot be readily determined visually, attempt to isolate and eliminate the fire as follows:

1. AC and dc generator switches–OFF.
   Turn the generator switches OFF to eliminate electrical power to the nonessential buses.
2. Emergency AC generator–ON.
   Turn the emergency ac generator on to energize the essential buses and the emergency dc bus.
3. Master switch–OFF (if fire or smoke persists).
4. Land as soon as practicable if fire subsides.

Note

It will be necessary to turn the master switch on long enough to facilitate speed brakes operation if landing is to be made and drag chute operation is desired.

5. **IF FIRE CONTINUES AND BECOMES SEVERE — EJECT.**

SMOKE, FUMES, OR FOG ELIMINATION

Should smoke, fumes, or fog enter the cockpit, proceed as follows:

1. Oxygen regulator diluter lever — 100% OXYGEN.
2. Oxygen regulator emergency toggle lever pushed either way from center.
3. **CABIN AIR SWITCH — RAM (BELOW 25,000 FEET IF POSSIBLE).**

EJECTION

Note

For considerations affecting the decision to eject, refer to EJECTION VS FORCED LANDING, this Section.

Every emergency in which ejection is considered will have its particular set of circumstances, involving such factors as airplane speed, attitude and control, as well as altitude. However, a decision should be made before takeoff as to what action is to be taken in the event of an emergency, particularly at low altitude. The ejection seat should be used to abandon the airplane in flight. The airplane should be slowed as much as possible if at high airspeeds and wings should be level. In any low-altitude ejection (below 2000 feet) the possibility of success can be materially improved by "zooming" the airplane and ejecting while the nose of the airplane is above the horizon (wings level) and the airspeed is above 120 knots. The zoom will exchange airspeed for altitude, thus providing maximum terrain clearance at time of ejection as well as reducing airspeed within safe limits for ejection. Ejecting while the nose of the airplane is above the horizon results in a more nearly vertical trajectory of the seat and crew member, thus providing more altitude and time for seat separation and parachute deployment. At low altitudes, a minimum airspeed of 120 KIAS is recommended to assure rapid deployment of the chute.

WARNING

- When the airplane is in a descending attitude and cannot be leveled, ejection should not be delayed as this will reduce the possibility of a successful ejection.
- Under level flight conditions, ejection should be accomplished above 2000 feet whenever possible.
- Under spin or dive conditions, ejection should be accomplished above 10,000 feet.
- Eject at the lowest practical airspeed above 120 KIAS (lowest practical would be that speed below which level flight cannot be maintained).

If possible, ejection should be made at a speed between 120 and 525 KIAS since relatively minor forces are exerted on the body. Betwen 525 and 600 KIAS, appreciable forces will be exerted on the body. Above 600 KIAS, ejection is extremely hazardous since excessive forces are exerted on the body. The structural limits of

zero delay lanyard engagement requirements

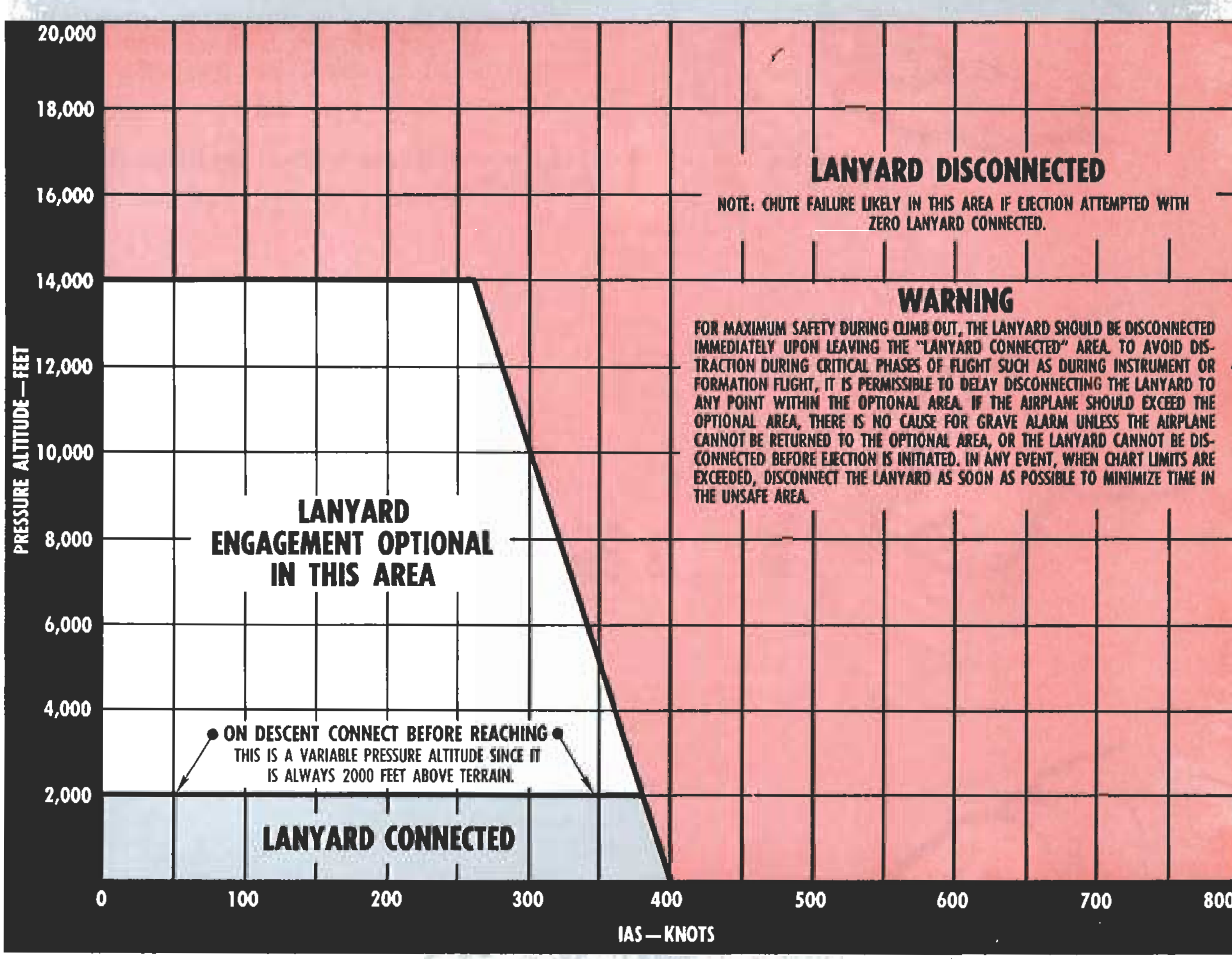

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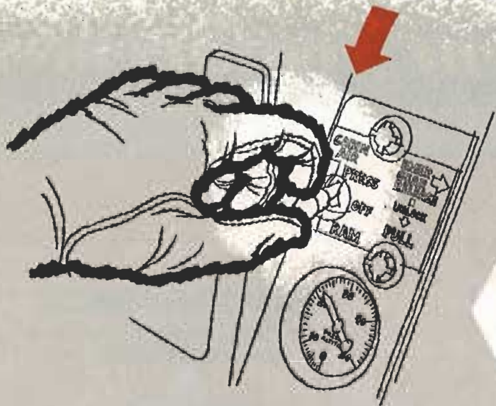
Figure 3-3

the seat may be exceeded at speeds in excess of 650 KIAS. There is no need to quote maximum airspeeds for ejection. If the airplane is controllable, airspeed will be reduced to as near 120 KIAS as practicable, which eliminates any high-speed problem. If the airplane is not controllable, ejection must be accomplished at whatever airspeed exists at the time, since ejection offers the only opportunity for survival. Speeds and altitudes in which the zero delay lanyard is connected or disconnected are outlined in the Zero Delay Lanyard Requirements Chart, figure 3-3. The chart is self explanatory in that it is divided into three positive areas; lanyard connected area, lanyard engagement optional area, and lanyard disconnected area. The pressure altitude feet scale indicates pressure altitude above sea level.

CAUTION

- The lanyard connected area is a variable pressure altitude area the top of which is always 2000 feet above the surrounding terrain.
- With the zero delay lanyard connected, the maximum speed for ejection must be as indicated on the Zero Delay Lanyard Engagement Requirements Chart, figure 3-3, to avoid parachute failure.

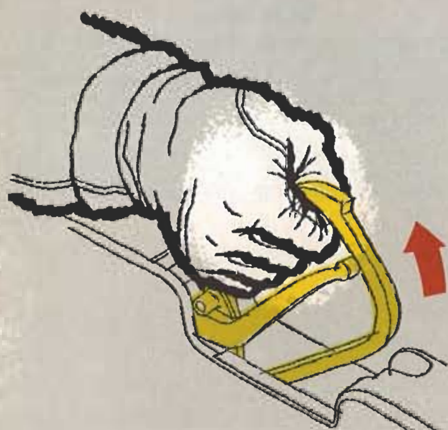
The Emergency Minimum Ejection Altitudes Table is presented on figure 3-4 and covers all possible combinations of seats, belts, and parachutes. The figures given



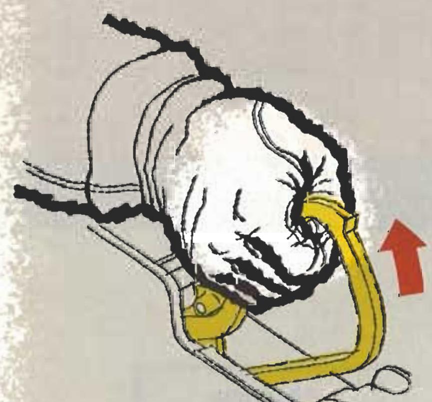
NOTE

- DO NOT MANUALLY OPEN THE AUTOMATIC OPENING SAFETY BELT PRIOR TO EJECTION AT ANY ALTITUDE.
- AFTER A 1 SECOND TIME DELAY THE SAFETY BELT WILL AUTOMATICALLY OPEN, THE FORCE REQUIRED TO SEPARATE THE PILOT FROM THE SEAT IS SUFFICIENT TO BREAK ALL PERSONAL LEAD CONNECTIONS.

1 TO PREVENT EXPLOSIVE DECOMPRESSION AT HIGH ALTITUDE, PLACE CABIN AIR SWITCH TO RAM.



2 PLACE ARMS ON ARMRESTS AND PULL EITHER OR BOTH HANDGRIPS TO JETTISON CANOPY. (SHOULDER HARNESS AUTOMATICALLY LOCKS WHEN HANDGRIPS ARE RAISED.)



3 SQUEEZE EITHER OR BOTH TRIGGERS TO EJECT SEAT.

EMERGENCY MINIMUM ALTITUDES FOR EJECTION (LEVEL FLIGHT)

	2 SECOND PARACHUTE		1 SECOND PARACHUTE		0 SECOND PARACHUTE	
	(F-1A TIMER)		(F-1B TIMER)		(LANYARD TO "D" RING)	
	B-5 PACK	C-9 CANOPY	B-5 PACK	C-9 CANOPY	B-5 PACK	C-9 CANOPY
AUTOMATIC SAFETY BELT WITH 1 SECOND (M12) INITIATOR	300		125		0	

WARNING

- THESE ARE EMERGENCY MINIMUMS. EJECTION SHOULD BE STARTED ABOVE 2000 FEET IF POSSIBLE.
- AT LOW ALTITUDES, A MINIMUM AIRSPEED OF 120 KIAS IS RECOMMENDED TO ASSURE RAPID DEPLOYMENT OF THE CHUTE.

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Figure 3-4

ejection procedures

BEFORE EJECTION, IF TIME AND CONDITIONS PERMIT

1. THROTTLE—OFF.
2. STOW ALL LOOSE EQUIPMENT.
3. IFF TO EMERGENCY.
4. ACTUATE BAILOUT OXYGEN BOTTLE. (SURVIVAL KIT OXYGEN WILL BE FURNISHED AUTOMATICALLY UPON EJECTION.)
5. TIGHTEN CHIN STRAP OF HELMET AND LOWER VISOR.
6. SIT ERECT, BRACE ARMS IN ARMRESTS, HOLD UPPER ARMS AND ELBOWS TIGHTLY AGAINST BODY, HEAD BACK HARD AGAINST HEADREST WITH CHIN TUCKED IN.

WARNING

- UNDER LEVEL FLIGHT CONDITIONS, EJECTION SHOULD BE ACCOMPLISHED ABOVE 2000 FEET WHENEVER POSSIBLE.
- UNDER SPIN OR DIVE CONDITIONS, EJECTION SHOULD BE ACCOMPLISHED ABOVE 10,000 FEET.
- EJECT AT THE LOWEST PRACTICAL AIRSPEED ABOVE 120 KIAS (LOWEST PRACTICAL WOULD BE THAT SPEED BELOW WHICH LEVEL FLIGHT CANNOT BE MAINTAINED).
- DO NOT MANUALLY OPEN THE AUTOMATIC OPENING SAFETY BELT PRIOR TO EJECTION.
- IF POSSIBLE, PRIOR TO EJECTION, THE PILOT SHOULD ATTEMPT TO TURN THE AIRPLANE TOWARD AN AREA WHERE INJURY OR DAMAGE TO PERSONS OR PROPERTY ON THE GROUND OR WATER IS LEAST LIKELY TO OCCUR.
- WHEN EJECTING AT LOW ALTITUDES, (BELOW 2000 FEET), PULL THE NOSE OF THE AIRPLANE ABOVE THE HORIZON, IF AT ALL POSSIBLE, AND USE EXCESS SPEED TO GAIN ALTITUDE.

IF CANOPY FAILS TO JETTISON, RELEASE CANOPY AS FOLLOWS

1. CANOPY JETTISON HANDLE—RAISE.

IF CANOPY IS STILL ON

1. MASTER SWITCH—OFF (SOME AIRPLANES); TRIP, THEN OFF (OTHER AIRPLANES).
2. CANOPY LATCH HANDLE—PULL FULL OUT.
3. PUSH CANOPY INTO AIRSTREAM.
4. MASTER SWITCH—ON AT PILOT'S DISCRETION.

WARNING

IF THE CANOPY CATAPULT FAILS TO FIRE, THE EJECTION SEAT WILL NOT EJECT ON AIRPLANES AF 55-3427 THRU 56-1429 UNLESS MODIFIED BY T.O. 1F-102A-565.

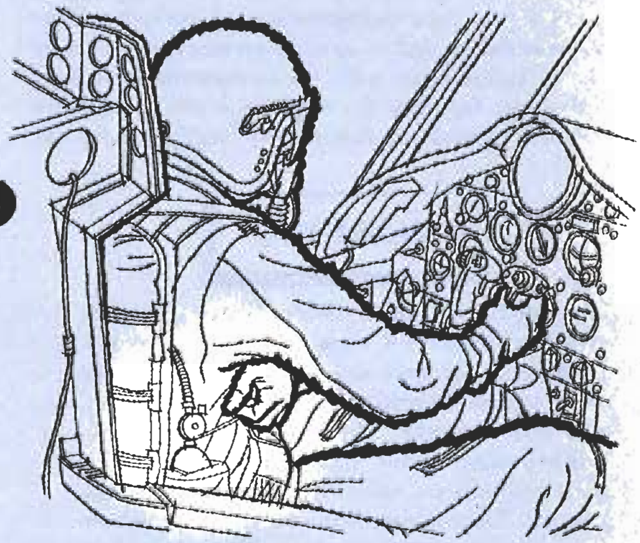
AFTER EJECTION

1. AFTER EJECTION, ATTEMPT TO "BEAT" THE AUTOMATIC SAFETY BELT, THEN IMMEDIATELY PUSH AWAY FROM SEAT WITH A POSITIVE ACTION.
2. IF AUTOMATIC SAFETY BELT FAILS AND SAFETY BELT IS RELEASED MANUALLY, PULL PARACHUTE ARMING LANYARD IF ABOVE 14,000 FEET.
3. MANUALLY PULL RIPCORD HANDLE IMMEDIATELY FOLLOWING SEPARATION FROM SEAT FOR ALL EJECTIONS BELOW 14,000 FEET.
4. IF WEARING A SURVIVAL KIT, AFTER PARACHUTE HAS OPENED AND STABILIZED, RAISE SURVIVAL KIT EMERGENCY RELEASE HANDLE. THIS WILL SUSPEND LIFE RAFT AND SURVIVAL KIT BELOW PILOT AND PREVENT INJURY ON LANDING.

WARNING

- DO NOT RAISE EMERGENCY RELEASE HANDLE UNTIL AFTER PARACHUTE DEPLOYMENT TO PREVENT THE KIT OR LANYARD FROM FOULING THE PARACHUTE.
- 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.

4



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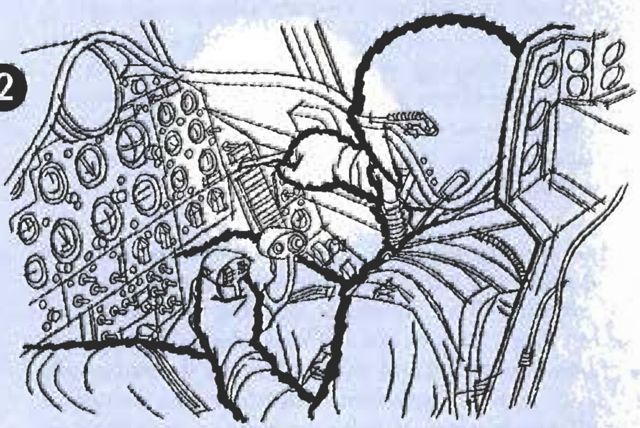

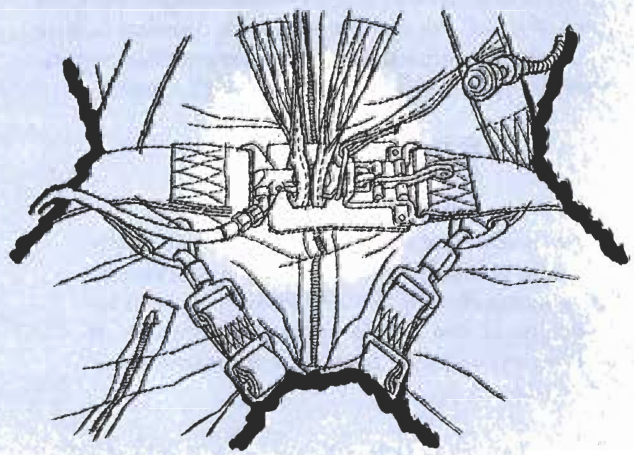

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are for level flight attitudes and are amply safe for climbs but inclined to be too optimistic for descending flight attitude. In order to use the chart, the style of automatic parachute and type of canopy, pack, and automatic release must first be determined. These are defined in T.O. 14D1-1-1. Once a minimum altitude has been determined for a particular configuration of equipment, the decision whether or not to eject in an emergency should be made in conjunction with the circumstances at hand and not by the fact that the airplane is above or below the minimum altitude as determined from these figures.

WARNING

Emergency minimum ejection altitudes presented on figure 3-4 were determined through extensive flight tests and are based on distance above terrain on initiation of seat ejection (i.e., time seat is fired). These figures do not provide any safety factor for such matters as equipment malfunction, delays in separating from the seat, etc. These figures are quoted only to show the minimum altitude that must be achieved in the event of such low altitude emergencies as fire on takeoff. They shall not be used as the basis for delaying ejection when above 2000 feet since accident statistics show a progressive decrease in successful ejections as altitude decreases below 2000 feet. Therefore, whenever possible, eject above 2000 feet.

See figure 3-4 for ejection procedures. On ejection, the seat and pilot will have a component of thrust provided by the airplane. The automatic opening belt and parachute are timed to provide the most desirable sequence under these conditions and will achieve the desired results faster than manual operation. Therefore, the automatic belt shall not be opened prior to ejection regardless of altitude because of several serious disadvantages, the most important of which are that the automatic opening feature of the parachute is eliminated, and crew member separation from the seat may be too rapid at high speeds.

CAUTION

Immediately after ejection, attempt to manually open the seat belt. This is strictly a precautionary measure in case the belt fails to open automatically. If the belt is operating normally, it will be impossible to "beat" the automatic opening action.

As soon as the safety belt releases, a determined effort must be made to separate from the seat to obtain full parachute deployment at maximum terrain clearance. THIS IS EXTREMELY IMPORTANT FOR LOW-ALTITUDE EJECTIONS.

WARNING

- If the seat belt is opened manually, the automatic feature of the parachute is eliminated. Therefore, under these circumstances, the parachute arming lanyard must be pulled if above 14,000 feet or the ripcord handle must be pulled if below 14,000 feet.
- Manually pull the parachute ripcord handle immediately following seat separation for all ejections below 14,000 feet. This is strictly a precautionary measure since the parachute should deploy automatically.
- Positive seat separation must be achieved prior to pulling the parachute ripcord handle to preclude parachute entanglement with the ejection seat.

Release survival kit after parachute is fully deployed and stabilized, and a safe altitude for breathing without supplemental oxygen is reached.

Note

Normally, the kit may be released as soon as the parachute stabilizes since the automatic opening device will not deploy the chute until a safe breathing altitude is reached. However, if for any reason the parachute deploys at a high altitude, do not release the survival kit until reaching a safe altitude for breathing without supplemental oxygen because releasing the kit cuts off the emergency oxygen supply.

TAKEOFF AND LANDING EMERGENCIES

ABORT

An aborted takeoff, regardless of cause, should be accomplished with landing gear extended. Use the following procedures if a takeoff is aborted prior to becoming airborne.

1. Throttle — OFF.
2. Drag chute handle — Pull.
3. External tanks jettison button — Depress (if required).

   If it is apparent that the airplane cannot be stopped on the runway, the external tanks should be jettisoned immediately.
4. Brakes — As required.
5. Fuel selector switch — OFF (some airplanes). Fuel shutoff valve switches — CLOSE (other airplanes).
6. Master switch — OFF (some airplanes); TRIP, then OFF (other airplanes).
7. Shoulder harness inertia reel handle — LOCKED.

CAUTION

The pilot is prevented from bending forward when the shoulder harness is locked; therefore, all switches not readily accessible should be "cut" before locking the shoulder harness inertia reel.

8. Canopy — Jettison (if necessary).

   Raise the canopy jettison handle on the left-hand armrest if it is necessary to jettison the canopy.

WARNING

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, or the canopy may become jammed.

9. Abandon the airplane as soon as possible.

WARNING

The cockpit side rails are approximately eight feet above the ground. Exercise care when abandoning the airplane to prevent bodily injury.

FLAT TIRE DURING TAKEOFF

If a main landing gear tire is blown on takeoff, and sufficient runway is available, abort the takeoff (refer to ABORT, this Section). Deploy the drag chute and use differential braking and nose wheel steering to maintain directional control. As speed decreases, if vibration or shimmy increases, lock the brake on the wheel with the blown tire. If the nose wheel tire has blown out, the brakes should be used for primary directional control and the drag chute should be deployed to reduce landing roll and nose wheel shimmy. Full up elevator should be used to relieve as much pressure as possible on the nose wheel.

WARNING

With a blown tire, directional control is more difficult and braking efficiency is greatly reduced at high gross weights.

If an abort cannot be accomplished because of insufficient runway, continue the takeoff.

WARNING

Do not raise the gear until it has been determined visually from the ground or another airplane that no fire exists, and that further damage will not be incurred by raising the gear.

Landing gross weight should be reduced, if possible, and landing accomplished as directed in FLAT TIRE LANDING, this Section.

EJECTION VS. FORCED LANDING

Normally, ejection is the best course of action with a windmilling or frozen engine, or failure of both the primary and the secondary hydraulic systems. However, 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 that would provide a ready made decision applicable to all emergencies of

this nature. The basic conditions listed below, combined with the pilot's analysis of the condition of the airplane, type of emergency, and his 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. If the decision is made to eject, prior to ejection, if possible, the pilot should attempt to turn the airplane toward an area where injury or damage to persons or 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 by pilots who have satisfactorily completed simulated flameout approaches in this airplane.

b. Flameout landings should be attempted only on a prepared or designated suitable surface.

c. Approaches to the runway should be clear and should not present a problem during flameout approach.

WARNING

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, the pilot should attempt to turn the airplane toward an area where injury or damage to persons or property on the ground or water is least likely to occur.

d. 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.

WARNING

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.

e. Flameout landings should be attempted only when either a satisfactory "high key" or "low key" position can be achieved.

f. If at any time during the flameout approach, conditions do not appear ideal for successful completion of the landing, ejection should be accomplished. Eject no later than the "low key" altitude.

FORCED LANDING

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 to the pilot 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 a wheels-up landing. Whenever the terrain is unknown or unsuited for forced landings, or whenever the landing gear cannot be extended, consideration should be given to the use of the ejection seat. This recommendation is made because of the increasing incidence of vertebral injuries (especially spinal compression fractures) to pilots during forced landings of other high-performance aircraft. If a crash landing is to be made, the canopy should be jettisoned just before touchdown to preclude jamming in the event of fuselage buckling. Whenever the canopy is jettisoned in the normal landing speed range during a landing emergency, no noticeable change in flight characteristics should be experienced and wind blast in the cockpit should be mild. At speeds lower than 175 KIAS the canopy may strike the tail. However actual emergency experiences indicate that airplane control is not affected.

Note

- Salvo firing is the only method of jettisoning the armament load as no mechanical release is provided.
- If it is desired to salvo armament prior to making a forced landing or ditching, the area must be clear for sufficient distance to allow the armament propellant to burn out and the armament to fall to earth in the clear area.
- The helmet visor should be lowered during any emergency landing. The visor provides eye protection from impact or flying objects and from wind blast after the canopy is jettisoned.

The handling characteristics of the airplane during a flameout landing are satisfactory. Immediately after flameout, jettison external tanks and establish and maintain best glide speed of 220 KIAS until sure of reaching the field. Turn off all electrical equipment not essential for flight. The hydraulic systems operate satisfactorily, maintaining adequate pressure and capacity with windmilling engine.

Note

If a frozen engine has resulted, the secondary hydraulic system will not supply pressure to operate the following systems:

- Emergency ac generator
- Speed brakes
- Landing gear retraction

If sufficient hydraulic pressure is available with a windmilling engine the RAT is not extended at the time of flameout, it should be extended at 2500 feet in the forced landing pattern. Gear extension should be accomplished when entering the 10,000-foot high key point.

Note

- In planning the landing when the gear is to be lowered by the emergency system, one minute should be allowed for gear extension time on some airplanes.* On other airplanes** emergency extention should not require more than ten seconds.
- Nose wheel steering is inoperative when gear is extended by emergency system.

If secondary hydraulic pressure is available and the gear is extended using normal extension, the main gear will extend and lock immediately and nose gear will lock in approximately 15 seconds. During the glide approaching pattern with the gear up, rate of descent is approximately 3000 feet per minute. After the gear is extended, rate of descent will range from about 4500 feet per minute in wings-level flight to about 6000 feet per minute in a 45-degree bank. Because of this relatively high rate of descent, the flare should be started approximately 200 feet above the ground. Windmilling engine rpm at touchdown speed of 160 knots is 10 to 12 percent. Brake effectiveness does not noticeably decrease during the landing roll even with excessive use of brakes. At seven to eight percent rpm and 90 to 100 knots during the ground roll, hydraulic capacity becomes too low to recover from control requirements and the stick becomes sluggish and freezes. For the purpose of making simulated forced landings, it has been determined that a power setting of 76 percent rpm with speed brakes fully extended will simulate flameout conditions. Figure 3-5 contains recommended procedures and techniques for a forced landing on suitable terrain.

BELLY LANDING

If forced to make a gear-up landing, proceed as follows:

CAUTION

If conditions permit, salvo the armament.

1. External tanks jettison button — Depress (if required).

   Depress the external tanks jettison button to jettison the external tanks if tanks are installed and contain fuel. If external tanks are empty, it is recommended that they be retained to cushion the impact unless landing is to be made on an unprepared surface.

*AF 53-1791 thru 56-1518 unless modified by TCTO 1F-102-655.
**AF 57-770 & on, & airplanes modified by TCTO 1F-102-655.

2. Make normal approach.
3. **LANDING GEAR HANDLE — DOWN.**
4. Shoulder harness inertia reel handle — LOCKED.

CAUTION

The pilot is prevented from bending forward when the shoulder harness is locked; therefore, all switches not readily accessible must be "cut" before moving the shoulder harness inertia handle to LOCKED position.

5. Immediately before touchdown:
   a. Speed brakes switch — OUT.

      Open speed brakes to allow drag chute deployment.
   b. Throttle — OFF.
   c. Fuel selector switch — OFF (some airplanes). Fuel shutoff valve switches — CLOSE (other airplanes).
   d. Canopy jettison handle — Raise.

      Raise canopy jettison handle on the left armrest to jettison the canopy.

WARNING

Make sure canopy is jettisoned before touchdown. Otherwise, sparks from the canopy remover may cause a fire if fuel is spilled in the vicinity of the airplane, or the canopy may become jammed.

6. Normal landing attitude for touchdown.
7. Drag chute handle — Pull.
8. Master switch — OFF (some airplanes); TRIP, then OFF (other airplanes).
9. Abandon the airplane.

   When the airplane stops, abandon immediately.

ANY ONE GEAR UP OR UNLOCKED

If landing is to be made with partial gear extension, proceed as follows:

Note

If time and conditions permit, request runway be foamed as soon as possible to assist directional control and decrease fire hazard.

1. External tanks jettison button — Depress (if required).

   Depress the external tanks jettison button to jettison the external tanks if tanks are installed and contain fuel. If external tanks are empty, it

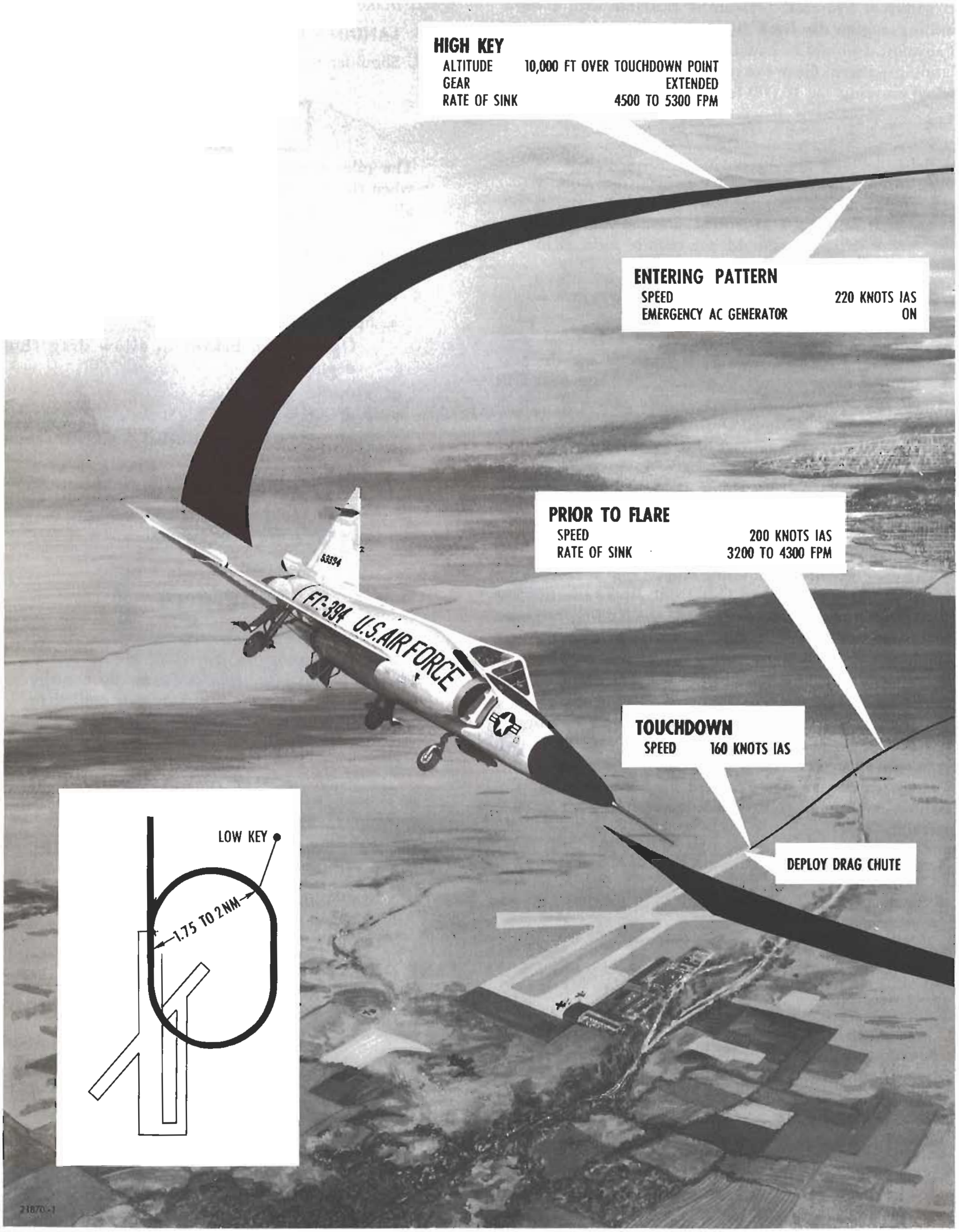
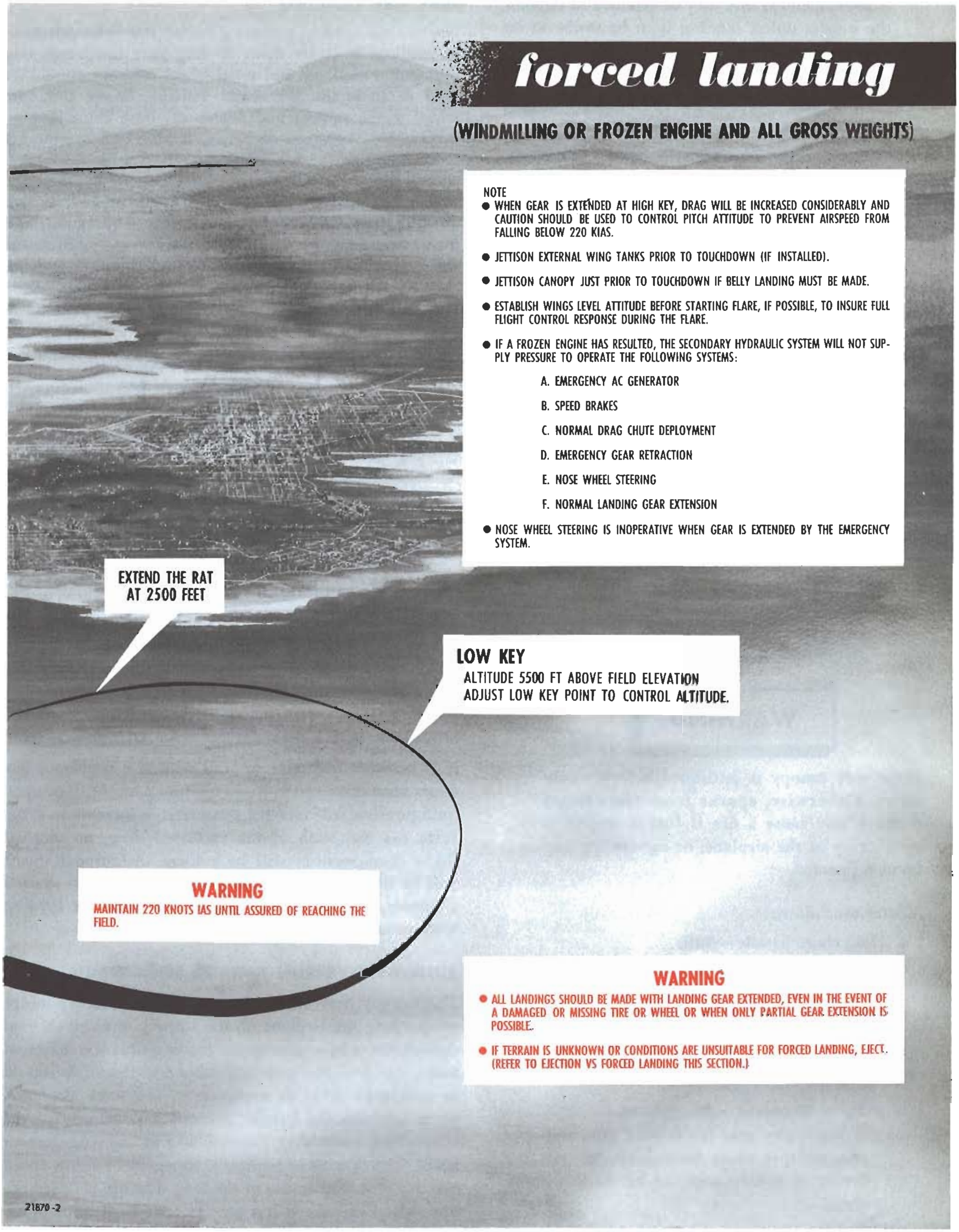

HIGH KEY
ALTITUDE 10,000 FT OVER TOUCHDOWN POINT
GEAR EXTENDED
RATE OF SINK 4500 TO 5300 FPM
ENTERING PATTERN
SPEED 220 KNOTS IAS
EMERGENCY AC GENERATOR ON
PRIOR TO FLARE
SPEED 200 KNOTS IAS
RATE OF SINK 3200 TO 4300 FPM
TOUCHDOWN
SPEED 160 KNOTS IAS
DEPLOY DRAG CHUTE
LOW KEY
1.75 TO 2 NM
FC-394 U.S. AIR FORCE

Figure 3-5


forced landing
(WINDMILLING OR FROZEN ENGINE AND ALL GROSS WEIGHTS)
NOTE
● 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 220 KIAS.
● JETTISON EXTERNAL WING TANKS PRIOR TO TOUCHDOWN (IF INSTALLED).
● JETTISON CANOPY JUST PRIOR TO TOUCHDOWN IF BELLY LANDING MUST BE MADE.
● ESTABLISH WINGS LEVEL ATTITUDE BEFORE STARTING FLARE, IF POSSIBLE, TO INSURE FULL FLIGHT CONTROL RESPONSE DURING THE FLARE.
● IF A FROZEN ENGINE HAS RESULTED, THE SECONDARY HYDRAULIC SYSTEM WILL NOT SUPPLY PRESSURE TO OPERATE THE FOLLOWING SYSTEMS:
A. EMERGENCY AC GENERATOR
B. SPEED BRAKES
C. NORMAL DRAG CHUTE DEPLOYMENT
D. EMERGENCY GEAR RETRACTION
E. NOSE WHEEL STEERING
F. NORMAL LANDING GEAR EXTENSION
● NOSE WHEEL STEERING IS INOPERATIVE WHEN GEAR IS EXTENDED BY THE EMERGENCY SYSTEM.
EXTEND THE RAT AT 2500 FEET
LOW KEY
ALTITUDE 5500 FT ABOVE FIELD ELEVATION
ADJUST LOW KEY POINT TO CONTROL ALTITUDE.
WARNING
MAINTAIN 220 KNOTS IAS UNTIL ASSURED OF REACHING THE FIELD.
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. (REFER TO EJECTION VS FORCED LANDING THIS SECTION.)
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is recommended that they be retained to cushion the impact unless landing is to be made on an unprepared surface.

2. Plan normal landing.

   Plan normal approach and touchdown and provide maximum distance for ground roll.

3. Shoulder harness inertia reel handle — LOCKED.

CAUTION

The pilot is prevented from bending forward when the shoulder harness is locked; therefore, all switches not readily accessible should be "cut" before moving the shoulder harness inertia reel handle to LOCKED position.

4. Immediately before touchdown:
   a. Speed brakes switch – OUT.

      Open the speed brakes to allow drag chute deployment.
   b. Throttle — OFF.
   c. Fuel selector switch — OFF (some airplanes). Fuel shutoff valve switches — CLOSE (other airplanes).
   d. Canopy jettison handle – Raise.

      Raise canopy jettison handle on the left armrest to jettison the canopy.

WARNING

Make sure canopy is jettisoned before touchdown. Otherwise, sparks from the canopy remover may cause a fire if fuel is spilled in the vicinity of the airplane, or the canopy may become jammed.

5. After touchdown:
   a. Drag chute handle – Pull.
   b. Master switch – OFF (some airplanes); TRIP, then OFF (other airplanes).
   c. Hold faulty gear off the ground.

      Hold the up or unlocked gear off the ground; then before flight controls become ineffective, ease the faulty gear down.
   d. Braking technique – As required.

      If the faulty gear is the nose gear and does not collapse upon touchdown, do not use brakes if a safe stop can be made without them.
6. Abandon the airplane.

   When the airplane stops, abandon immediately.

FLAT TIRE LANDING

Use normal landing pattern if blown tire is known prior to landing. If either main landing gear tire blows out, differential braking and nose wheel steering should be used to maintain directional control. Lower the nose wheel to the runway and deploy the drag chute as soon as possible. As speed decreases, if vibration or shimmy increases, lock the brake on the wheel with the blown tire. If the nose wheel tire blows out on landing, the brakes should be used for primary directional control and the drag chute deployed to reduce landing roll and nose wheel shimmy. Because of the extreme shimmy of the nose wheel with a blown tire, nose wheel steering may be uncontrollable. When a nose wheel tire blows out, trim full up elevator to relieve as much pressure as possible on the nose wheel.

CAUTION

- If it is known before landing that a main landing gear tire is flat, the airplane should be landed on the left side of the runway when the right main landing gear tire is flat or on the right side of the runway when the left main landing gear tire is flat. The airplane will veer in the direction of the flat tire and complete directional control may not be possible.
- When it is known before landing that the nose wheel tire is flat, lower the nose wheel to the runway at approximately 110 knots to permit making a controlled nose wheel touchdown.

LANDING WITH ARMAMENT BAY DOORS OPEN

If it becomes necessary to land with the armament bay doors open due to an in-flight malfunction, the door-open configuration will create a considerable increase in drag. With the exception of the increased drag, no unusual flight characteristics will be evident. Additional thrust will be required around the pattern to maintain desired airspeeds. Do not increase pattern airspeeds for landing with armament doors open.

RUNWAY OVERRUN BARRIER ENGAGEMENT

The runway overrun barrier provides an effective means of stopping the airplane on the runway after an aborted takeoff run or in an emergency landing situation. External tanks may foul the arresting cable and therefore should be jettisoned prior to engagement. On some airplanes, when entering the barrier, there is a possibility of the nose wheel overriding the webbing at any engagement speed. The chances of successful engagement at low speed are also reduced because of the long distance between the nose wheel and the main gear. On later airplanes a probe has been added on the nose landing gear and a deflector over the taxi light to insure that the nose gear engages the webbing. Also, some airplanes* have an extendible barrier probe mounted on the centerline of the airplane just aft of the missile bay doors. This probe is extended simultaneously with the actuation of the drag chute and serves as a guide to insure that the arresting cable engages the main landing gear. However, the probability of engagement increases as speed increases. Below 60 knots ground speed the airplane cannot be expected to engage the barrier and between 60 and 70 knots ground speed engagement is marginal. Above 70 knots ground speed the airplane can be expected to engage the barrier if the webbing is not overridden. During successful engagement, structural failure of the gear is not likely, and the deceleration forces are negligible. The airplane should enter the barrier at a 90 degree angle, although some deviation will not preclude successful engagement. Directional control of the airplane is normally not difficult to maintain if both main landing gear are engaged. Just prior to engagement, the throttle should be OFF and nose wheel steering engaged.

CAUTION

Above 80 knots ground speed, nose wheel steering becomes sensitive and directional control is very difficult to maintain.

If only one main landing gear is engaged, the airplane will veer in the direction of the engaged gear. This tendency increases as the engagement speed increases. Directional control should be maintained with nose wheel steering and differential braking. Brakes should not be applied as the barrier is entered because braking will lower the nose of the airplane and the pitot boom will possibly engage the webbing.

EMERGENCY ENTRANCE

The procedure to be used by the rescue personnel when assisting a disabled pilot from the airplane following a crash landing is contained in figure 3-6.

DITCHING

Do not attempt to ditch this airplane except as a last resort. Ditching should be attempted only when altitude is insufficient for a successful bailout or in the event the ejection seat fails to function. (refer to EJECTION, this Section.) This recommendation is made because of the danger of vertebral injury (especially spinal compression fracture) to the pilot during a water landing. The helmet visor should be lowered prior to ditching. The visor provides eye protection from impact or flying objects and from wind blast after the canopy is jettisoned. If ditching is unavoidable, proceed as follows:

1. Follow radio distress procedure.
2. External fuel tanks button — Depress.
3. RAT handle — Pull (if necessary) & hold for four seconds.

   Extend the RAT by pulling the RAT handle fully out. Hold the handle in the fully extended position for four seconds to insure a satisfactory extension of the turbine.
4. Oxygen diluter lever — 100% OXYGEN.
5. **LANDING GEAR HANDLE — UP.**
6. Shoulder harness inertia reel handle — LOCKED.
7. Master switch — OFF (some airplanes); TRIP, then OFF (other airplanes).
8. Canopy jettison handle — Raise.

   Raise canopy jettison handle on the left armrest to jettison the canopy just before touchdown.
9. Normal approach and touchdown.

   Land parallel to swells unless the wind is in excess of 25 knots, in which case it is recommended to land into the wind, touching down on the falling side of the wave, if possible. Maintain nose-high attitude and touch down as slowly as possible with a small rate of descent. As the nose of the airplane settles into the water, it may tend to flip over due to the intake ducts acting as water scoops.
10. Throttle — OFF at touchdown.
11. Abandon the airplane as soon as forward motion stops. Upon contact with the water, the airplane will usually bounce, during which time the sensation of being stopped may be experienced. Do not unfasten the seat belt until deceleration forces have stopped, water spray is noticeable, and water begins to enter the cockpit. When leaving the airplane retain the survival kit if possible (if survival kit is installed).

Note

The diluter-demand type oxygen regulator is a suitable underwater breathing device when the regulator is set at 100% OXYGEN. If for some reason an immediate escape is not possible this equipment may be used for underwater breathing provided all connections are tight. The bailout bottle can be used under water only on airplanes equipped with a survival kit. On these airplanes, pulling the "green knob" will supply oxygen for approximately 12 minutes.

AFTERBURNER FAILURE

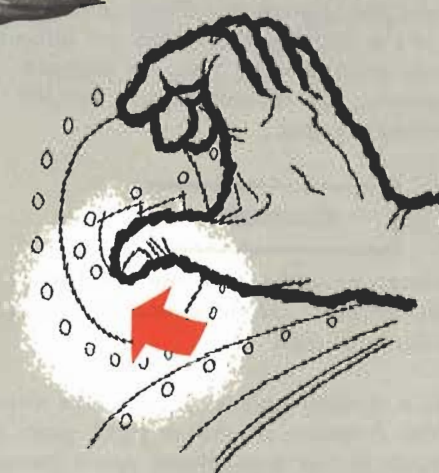
AFTERBURNER FAILURE DURING TAKEOFF

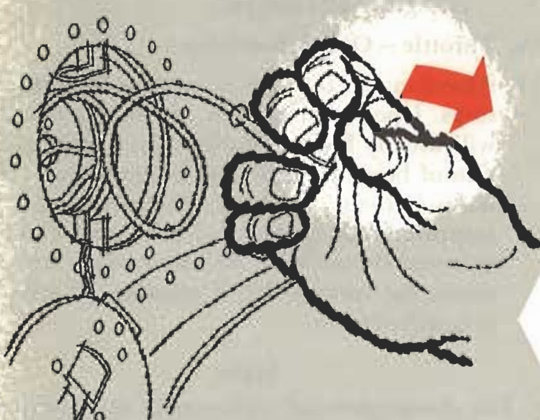
Afterburner failure may be noted by a sudden loss of acceleration.

1. If sufficient runway remains to make a safe stop or insufficient runway available to continue takeoff at military thrust, abort takeoff; refer to ABORT, this Section.

**AF 57-770 & on, & airplanes modified by TCTO 1F-102-658.*

REMOVE ACCESS DOOR. 1





2

GRASP "T" HANDLE AND PULL OUTBOARD APPROXIMATELY 6 FEET TO JETTISON CANOPY.

NOTE

CANOPY WILL BE JETTISONED BY SAME SYSTEM AS INSTALLED FOR CANOPY JETTISON BY PILOT. CANOPY WILL TRAVEL UP AND AFT AND WILL PROBABLY STRIKE THE TAIL. RESCUE PERSONNEL SHOULD WATCH PATH OF CANOPY TO AVOID BEING HIT.

WARNING

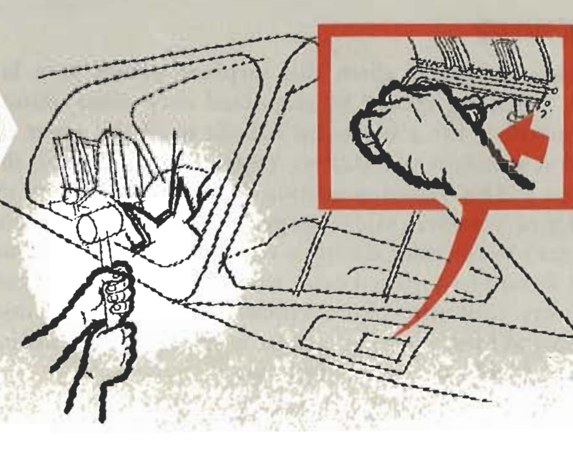
IF THE CANOPY HAS NOT BEEN RELEASED AND IF SPILLED FUEL IS IN THE VICINITY OF THE AIRPLANE, A FIRE MAY RESULT FROM A HOT POWDER SPARK WHEN THE CANOPY IS JETTISONED.

IF CANOPY FAILS TO JETTISON OR IF PRESENCE OF FUEL FUMES MAKES JETTISON INADVISABLE, GAIN ENTRANCE EITHER BY:

a. OPENING THE LATCH ACCESS DOOR LOCATED BELOW THE RIGHT-HAND WINDSHIELD AND PUSHING THE EXTERNAL CANOPY RELEASE HANDLE TO THE REAR TO UNLATCH THE CANOPY. AN EXTERNAL GRIP ON THE FORWARD LEFT-HAND SIDE OF THE CANOPY IS PROVIDED TO ASSIST IN RAISING THE CANOPY.

b. BREAKING CANOPY GLASS IF EXPEDIENCY OF ENTRY IS OF MAJOR IMPORTANCE. CANOPY GLASS SHOULD BE BROKEN IN THE LOWER FORWARD CORNER ON EITHER SIDE. IT IS ESSENTIAL TO STRIKE HARD BLOWS AS NEAR THE EDGE OF CANOPY FRAME AS POSSIBLE.

3



21871-1

Figure 3-6

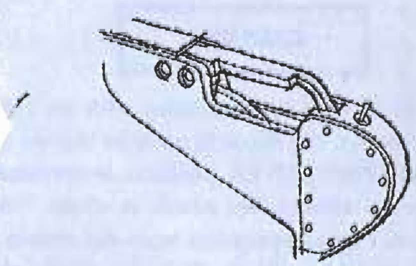
emergency entrance

WHEN ACCESS TO COCKPIT IS GAINED, CHECK POSITION OF EJECTION SEAT HANDGRIPS.

NOTE

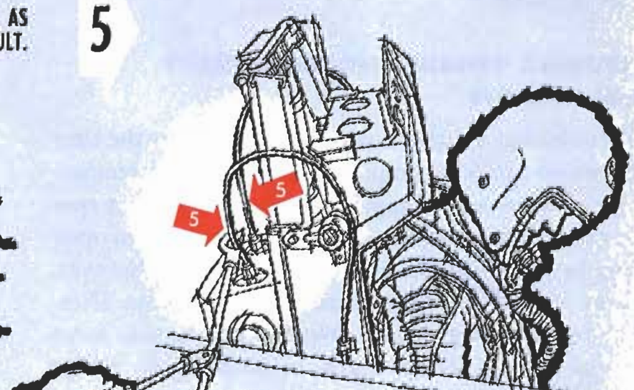
IF PILOT JETTISONED CANOPY IN PREPARATION FOR CRASH LANDING, THE CANOPY JETTISON HANDLE SHOULD HAVE BEEN USED: HOWEVER, RAISING EITHER HANDGRIP WILL ALSO JETTISON THE CANOPY. IF THE HANDGRIPS ARE RAISED, SEAT CATAPULT TRIGGERS ARE EXPOSED. SUBSEQUENT MOVEMENT OF EITHER TRIGGER WILL FIRE THE CATAPULT AND EJECT THE SEAT FROM THE AIRPLANE.

4



DISARM SEAT CATAPULT BY CUTTING WITH HEAVY DUTY CABLE CUTTERS, AS SHOWN, OR DISCONNECTING HOSE LEAD TO FITTING AT TOP OF SEAT CATAPULT.

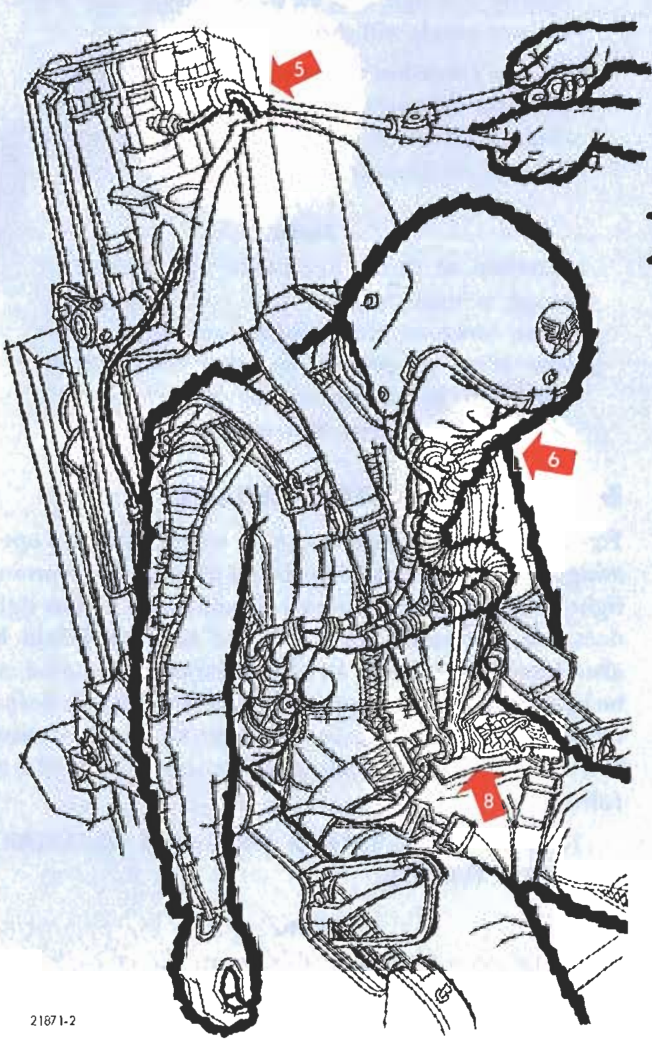
5



NOTE

ON AIRPLANES WITH ROCKET CATAPULT INSTALLED, DISARM THE SEAT BY PUSHING THE QUICK DISCONNECT FITTINGS, OR BY CUTTING WITH HEAVY DUTY CABLE CUTTERS AS SHOWN ABOVE.

5



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6 IF PILOT IS WEARING A PARTIAL PRESSURE SUIT AND HELMET, THE HELMET FACE PLATE MUST BE REMOVED PRIOR TO RELIEVING ANY PRESSURE TO THE SUIT. IF THE SUIT PRESSURE IS RELIEVED FIRST, IT IS POSSIBLE TO RUPTURE THE PILOT'S LUNGS BY FORCING AIR UNDER PRESSURE FROM THE STILL-PRESSURIZED HELMET. THE HELMET FACEPLATE MAY BE REMOVED BY PULLING DOWNWARD ON THE GREEN CORD BENEATH CHIN AND LIFTING FACEPLATE FREE.

7 PULL YELLOW SURVIVAL KIT "EMERGENCY RELEASE" HANDLE TO DISCONNECT THE HARNESS AND PERSONAL LEADS BUNDLE.

NOTE

- ON SOME AIRPLANES, THE RUBBER LIFE RAFT MAY INFLATE (IF INSTALLED). THE RAFT SHOULD BE PUNCTURED IF IT HINDERS REMOVAL OF THE PILOT.
- THE "HARNESS RELEASE" LEVER, AFT OF THE "EMERGENCY RELEASE" HANDLE, MAY ALSO BE USED TO DISCONNECT THE SURVIVAL KIT PARACHUTE ATTACHING STRAPS. HOWEVER, THE PERSONAL LEADS MUST THEN BE DISCONNECTED MANUALLY.

8 RELEASE SEAT BELT AND SHOULDER HARNESS.

2. If there is insufficient runway available to make a safe stop and takeoff can be safely continued at military thrust:
   a. **THROTTLE — INBOARD TO FULL MIL POWER.**

CAUTION

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 and a reduction of approximately four percent rpm will be noted. Partial failure of the exhaust nozzle to open may not be noted on the engine instruments. However, failure of several adjacent nozzle flaps when the afterburner ignites during takeoff may result in a side force which may cause a sudden lurch, or tendency to change direction of the airplane.

WARNING

If several adjacent nozzle flaps have failed and takeoff is continued, the side force may be so great that there will be insufficient rudder to control the airplane immediately after leaving the ground.

1. If exhaust gas temperature rises rapidly and rpm drops or if any directional change is noted when the afterburner is ignited during the takeoff roll:
   a. **THROTTLE — OFF.**
   b. Drag chute handle — Pull.
   c. Brakes — As required.

AFTERBURNER FAILURE DURING FLIGHT

1. Throttle – Inboard and hold five seconds.
   Move the throttle inboard to shut off fuel to the afterburner. Wait five seconds to clear afterburner.
2. **THROTTLE — AFTERBURNER TO RE-IGNITE AFTERBURNER (IF DESIRED).**
3. If afterburner fails to light – Throttle inboard.

WARNING

Do not make a second attempt to relight the afterburner if cause of afterburner failure is unknown, to prevent possibility of fire or explosion.

AFTERBURNER CUTOFF FAILURE

If the afterburner cannot be cut off through use of normal throttle motion, retard the throttle smartly to some point below approximately 90% rpm to mechanically terminate afterburner; then advance slowly to the minimum thrust which will sustain flight and land as soon as practicable.

CAUTION

- Do not pause at the afterburner actuation point when retarding the throttle; move the throttle smartly through this range to insure that the exhaust nozzle will close.
- During operation above 40,000 feet, avoid rapid throttle movement and do not reduce the throttle below 85% rpm to reduce the probability of compressor stalls.

Note

Selection of thrust just below the afterburner range is important for fuel saving considerations; however, afterburning is available in the case of a "go-around" or other critical flight condition by advancing the throttle smartly above approximately 90% rpm.

ENGINE OIL SYSTEM FAILURE

Failure of the engine oil system to supply sufficient operating pressure is indicated by the oil pressure-low warning light (master warning system). Illumination of this light does not necessarily mean that the airplane should be abandoned immediately. In most instances, the engine can be operated at reduced power for several minutes before ultimate engine failure. In the event of an oil pressure low indication, the following procedure should be followed:

1. **THROTTLE — RETARD (MINIMUM NECESSARY FOR FLIGHT).**

Note

Once the throttle is retarded to minimum necessary for flight, additional throttle movement could result in the engine freezing. Subsequent throttle movement should be used only in the interest of safety.

2. **AIRSPEED — 345 KIAS OR BELOW.**

   Airspeed should be reduced below the maximum RAT speed in the event the engine freezes and it is necessary to extend the RAT.

3. Avoid high g maneuvers.

4. **LAND AS SOON AS POSSIBLE. A FLAMEOUT PATTERN CAN BE USED IF THIS IS THE MOST EXPEDITIOUS MEANS OF GETTING ON THE RUNWAY.**

   Landing should be made on the nearest suitable runway, and the engine shut down immediately after taxiing off active runway.

5. If engine vibrations become excessive during flight, shut down engine.

   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 a forced landing.

Note

Maneuvers at less than one g or at negative g may cause the oil pressure-low warning light to illuminate during flight. This is permissible provided the duration of the maneuver does not exceed limitations specified in PROHIBITED MANEUVERS, Section V.

WARNING

- If engine oil system fails, expect ac and dc generator failure.
- Engine failure resulting from insufficient oil pressure may result in a frozen engine. As soon as the engine fails or is shut down, the RAT should be extended to supply flight control hydraulic pressure.

FUEL SYSTEM FAILURE

ENGINE STAGE FUEL PUMP FAILURE DURING TAKEOFF

In the event of failure of the engine stage fuel pump, as indicated by the engine fuel pump failure warning light (master warning system) the takeoff should be aborted if a safe stop can be made. If committed to takeoff, the afterburner stage fuel pump will provide all the fuel required by the engine and part of the fuel required by the afterburner; therefore, partial thrust will be obtained from continued afterburner operation.

ENGINE FUEL CONTROL FAILURE

Engine fuel control failure may be recognized by a drop in exhaust gas temperature, rpm, and fuel flow. As in any takeoff emergency, 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, use the following procedure:

1. Throttle — As required.

   Attempt to match throttle position to engine rpm.

WARNING

Compressor stall or engine flameout may occur due to introduction of excess fuel if the throttle position does not approximately correspond to engine rpm prior to selection of the emergency fuel system.

2. **FUEL CONTROL SWITCH — EMERGENCY.**

WARNING

- Avoid rapid throttle movements.
- Following an inflight normal fuel system failure, do not return the fuel control switch to NORMAL for the duration of the flight or flameout will result.

3. Control engine speed as necessary.
4. Land as soon as possible.

ENGINE STAGE FUEL PUMP FAILURE IN FLIGHT

In the event of failure of the engine stage fuel pump as indicated by the engine fuel pump failure warning light (master warning system), the afterburner stage fuel pump will supply fuel to the engine and afterburner operation should not be attempted.

AFTERBURNER FUEL PUMP FAILURE DURING TAKEOFF

In the event of failure of the afterburner stage fuel pump, afterburner failure will occur. The engine stage fuel pump will not supply fuel to the afterburner. Refer to AFTERBURNER FAILURE DURING TAKEOFF, this Section.

FUEL BOOST PUMP FAILURE

In the event complete boost pump failure is experienced during takeoff but 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 takeoff. With complete fuel boost pump failure, observe the following:

1. Avoid negative g maneuvers.
2. Under low fuel quantity conditions (low level warning light illuminated):
   a. Operate in a nose-high attitude (normal traffic pattern attitude is nose high).
   b. Avoid uncoordinated maneuvers.
   c. Avoid rapid decelerations (combinations of large thrust changes, speed brakes, and landing gear extensions).
3. Land as soon as practicable.

The above steps are to prevent uncovering the aft fuel intakes. A complete fuel boost pump failure will occur with loss of ac power.

FUEL BOOST PRESSURE-LOW WARNING

Fuel boost pressure L or R warning light will illuminate under the following conditions:

- An empty No. 3 tank.
- A dual failure of boost pumps on one side.
- Fuel shutoff valve not opened. (An electrical interlock makes boost pump operation dependent upon a fully open shutoff valve.)

If the fuel boost pressure-low warning light illuminates, in flight, proceed as follows:

1. Fuel selector switch — ENGINE (some airplanes).

   Check that fuel selector switch is selected to ENGINE position.

   Fuel shutoff valve switches — OPEN (other airplanes).
2. **PLAN THE FLIGHT FOR ONE-HALF THE INDICATED FUEL BEING AVAILABLE.**

   Initial action should be based on the assumption that only one-half of the total fuel on board the airplane at the time of fuel boost pump pressure-low warning will be available, if the warning is due to a fuel valve which is not fully open.
3. Verify quantity of remaining fuel in the No. 3 tank on the side which has indicated the warning.
   a. If the No. 3 tank is empty, shut off the empty tank by placing fuel selector switch to the other side, or, on some airplanes, by moving the appropriate fuel shutoff valve switch to CLOSE.

WARNING

If operation with a known empty tank is required, the fuel shutoff valve on the empty side should be closed. This is a precautionary measure to prevent air from entering the system in the event of a pump system malfunction on the side delivering fuel. On airplanes which do not incorporate the boost pump/shutoff valve interlock system, the boost pump switches should be turned off on the empty side.

   b. If the No. 3 tank indicates fuel remaining:
      (1) Fuel selector switch — Leave at ENGINE (some airplanes).
      Fuel shutoff valve switches — OPEN (other airplanes).
      (2) Failed boost pump switches — OFF.
      (3) **LAND AS SOON AS PRACTICABLE.**

Note

If continued flight is necessary, it may be possible to use the indicated fuel remaining. Turn off boost pumps on the good side and monitor remaining fuel carefully to determine if fuel is feeding from the failed side. If it is, conduct flight in accordance with complete FUEL BOOST PUMP FAILURE paragraph above. Prior to penetration, turn good boost pumps ON. (See system description, Section I.)

FUEL FLOW EQUALIZER FAILURE

Asymmetrical fuel flow can result from unequal fuel feeding through the fuel flow equalizer or from fuel flow equalizer failure. Minor unbalances from this source can be expected owing to allowable tolerances. Should extreme unbalanced fuel loading become apparent (500-lb. differential or greater), it is desirable to equalize the fuel load. If the unbalanced flow is allowed to continue without corrective action the airplane will exhaust the fuel from one side first. If an empty tank condition exists, refer to FUEL BOOST PRESSURE-LOW WARNING, this Section. To preclude the above, fuel loading may be balanced as follows:

1. Fuel selector switch — ENGINE (some airplanes).

   Fuel shutoff valve switches — OPEN (other airplanes).
2. Fuel quantity — Check (select each position on the fuel quantity selector switch).

   In the event the fuel low warning light illuminates on one side only, or abnormal wing heaviness is noted, the fuel quantity should be verified by the fuel quantity gage.
3. Boost pump switches – OFF on the low side. This will permit high rates of flow from the high side.
4. Boost pump switches – ON on the high side.
5. Fuel quantity gage selector switch – Monitor both low and high sides to insure that the low side has stopped feeding and the high side is feeding.

6. When fuel load is equalized, or when fuel low level warning light on the high side illuminates (less than approximately 570 lbs. of fuel remaining), proceed as follows:
   a. All boost pump switches – ON.
   b. Fuel quantity gage selector switch — TOTAL.

WARNING

If penetration or descent below 10,000 feet is initiated prior to correcting asymmetrical fuel flow, turn all boost pump switches ON.

FUEL TANK PRESSURIZATION FAILURE

If fuel tank pressurization fails, as indicated by illumination of either fuel tank pressure-low warning light, normal fuel transfer cannot be expected. Some fuel may transfer from the No. 1 and No. 2 tanks into No. 3 but the quantity will depend on flight attitude and altitude. Normally the only fuel which will feed to the engine will be that in No. 3. On airplanes which have No. 3 tank placarded on the fuel quantity gage selector switch, it is possible to read the amount of fuel remaining in the No. 3 tank on each side. On these airplanes, it is possible to monitor the No. 3 tanks and determine if fuel is feeding from the other two tanks without pressurization. The flight should be adjusted accordingly in event that only No. 3 tank is feeding the engine. On airplanes which have positions of L and R placarded on the fuel quantity gage selector switch, the fuel quantity gage indicates total fuel in either wing and it is not possible to determine if fuel is feeding from No. 1 and No. 2 tanks into No. 3. On these airplanes, if pressure fails, it is necessary to assume that only No. 3 tank fuel is available. Momentary illumination of the fuel tank pressure-low warning light will occur during rapid descent or high g acceleration. If the warning light illuminates steadily, proceed as follows:

1. Land as soon as practicable.
2. **DEPEND ON NO. 3 TANK FUEL ONLY.**
3. Monitor low level warning lights and fuel quantity in each wing.
4. Do not use afterburner.

ELECTRICAL POWER SYSTEM FAILURE

Note

- The airplane can be returned to base with complete electrical failure and a safe landing accomplished under visual flight conditions. When operating from the battery, all equipment not needed to maintain flight should be turned off. Usable battery power for continuous operation is approximately five to 20 minutes.
- When operating from battery power only, the UHF will not function after approximately five minutes due to lowering battery power. When the UHF stops functioning, it should be shut off to conserve battery power.

If ac and dc generators fail, the emergency ac generator will supply power to the ac essential bus and the battery will supply power to the dc essential bus. Since ac electrical power control relays which connect the ac generator to the ac bus are energized from the dc essential bus, complete electrical power failure will occur after battery power depletion (approximately five to 20 minutes). On airplanes with self-sustaining ac system*, a dc generator failure followed by an ac generator failure would also result in complete electrical power failure (except battery power). However, by momentarily placing the ac bus switch in the RESET position the emergency ac generator will start and supply power to the ac essential bus and the transformer-rectifier will again supply power to the emergency dc bus. When the ac bus switch is released, it returns to the ON position tying the ac emergency disconnect relay and emergency generator shutoff valve to the emergency dc bus.

WARNING

To conserve battery power, turn battery OFF if not required.

Should the transformer-rectifier fail after dc generator failure the battery may be used to power the emergency dc bus. Actuation of the battery switch to the TR FAIL position, ties the emergency dc bus to the dc essential bus (battery). Power will be supplied to the dc essential functions until battery power is depleted (five to 20 minutes).

AC GENERATOR FAILURE

If the main ac generator fails, the ac power failure warning light will illuminate. In the event of a failure, use the following procedure:

1. AC bus switch – EMER (some airplanes); RESET, then ON (other airplanes).
2. Radar master switch – OFF.
3. Boost pump switches – OFF.
4. Nesa switch – OFF.
5. Rain clear switch–RAINCLEAR (some airplanes); STDBY (other airplanes).
6. AC generator switch – RESET, then ON.

Note

If generator will not reset, check circuit breakers.

Airplanes modified by TCTO 1F-102-727.

If warning light extinguishes:

1. AC bus switch — NOR.
2. Boost pump switches — ON (one at a time).
3. Nesa switch — NORMAL (some airplanes); ON (other airplanes).
4. Radar master switch — As required.

If warning light remains on:

1. AC generator switch — OFF.
2. **LAND AS SOON AS PRACTICABLE.**

Note

In the event of main ac generator failure, the following primary flight aids, in addition to the pitot-static instruments, will be operable:

- All gyro flight instruments
- Gyro compass
- Interphone, UHF command, VOR, and marker beacon radio
- IFF radar

Refer to Section I for detailed list of electrically operated equipment.

DC GENERATOR FAILURE

If the dc generator fails, the dc power failure warning light will illuminate. In the event of failure, use the following procedure:

1. DC generator switch — RESET, then ON.

Note

If generator will not reset, check circuit breakers.

If warning light remains on:

1. DC generator switch — OFF.
2. Turn off all equipment not needed.
3. Battery switch — ON.

Note

- Airplane trim should be used as little as possible as it imposes a heavy drain on the battery.
- The battery will provide usable power for continuous operation for approximately five to 20 minutes.

4. Land as soon as practicable.

Note

After failure of the dc generator, the following primary flight aids, in addition to the pitot-static instruments, will be available until battery power depletion:

- All gyro flight instruments
- Gyro compass
- Interphone and UHF command
- VOR radio
- IFF radar

The UHF command radio and interphone will be operable after battery power depletion on airplanes with the self-sustaining ac system. Refer to Section I for a detailed list of electrically operated equipment.

AC AND DC GENERATOR FAILURE

If both normal ac and dc generator fail, follow same procedures prescribed for individual generator failure.

Note

The emergency ac generator will furnish power to the ac essential bus and 28-volt dc transformer-rectifier. The TR unit in turn furnishes dc power to the UHF command radio and with a failed dc generator, the battery furnishes all other dc power.

If the emergency ac generator or transformer-rectifier fails:

1. Battery switch — TR FAIL position.

Note

- On airplanes with self-sustaining ac system, placing the battery switch in TR FAIL position connects the dc emergency bus to dc essential bus (battery). UHF command radio and interphone will continue to operate until battery power is depleted.
- Complete loss of UHF and interphone occurs on airplanes prior to incorporation of self-sustaining ac system.

INSTRUMENT FAILURE

Failure of the main ac generator is indicated by the ac failure warning light and power can be restored to the primary flight and engine instruments by energizing the emergency ac generator. Failure of either of the transformers will also result in failure of these instruments.

CAUTION

Loss of ac power results in failure of the hydraulic pressure gage, fuel quantity gage, fuel flow indicator, directional indicator (slaved), engine pressure ratio gage and attitude indicator. These instruments will continue to register the condition that existed at the time of power failure.

There is no indication of transformer failure other than the loss of operation of equipment and no alternate source of power is provided.

HYDRAULIC POWER SYSTEM FAILURE

Note

It should be noted on Form 781 if the RAT has been extended in flight.

FAILURE OF ONE HYDRAULIC SYSTEM

Airplane flight characteristics encountered during flight with one hydraulic system inoperative reveal that high speed maneuvering capabilities are reduced. With one hydraulic system inoperative, elevator hinge moment is limited to the extent that the airplane is capable of pulling only a small percentage of maximum g limit. RAT extension is not required while flight control system operating pressure is available in either hydraulic system. In the event of primary pump failure and/or frozen engine, the RAT will provide immediate pressure to the primary hydraulic system and should be extended prior to entering the landing pattern. If the hydraulic pressure-low warning light flashes, indicating failure of one hydraulic system, use the following procedure:

1. Reduce airspeed.

   Reduce airspeed to below RAT maximum extension speed by immediate power reduction. Maintain airspeed below RAT maximum extension speed so that the emergency system will be readily available in the event of failure of the remaining system.

CAUTION

With one hydraulic system inoperative, hinge moment limitations prevent pulling the maximum g limit. For supersonic dive recovery, this becomes especially critical. Refer to Section VI for maximum g's available with one system inoperative and for dive recovery information.

2. Restrict flight to avoid need for speed brakes.

   Avoid need for speed brakes to conserve hydraulic pressure for flight control operation with primary hydraulic system inoperative, and because pressure will not be available for speed brakes operation with secondary hydraulic system inoperative.

3. Determine which system failed.

   Select PRI, then SEC with the hydraulic pressure gage selector switch and check hydraulic pressure gage to determine which system gives the low pressure indication.

CAUTION

If the secondary hydraulic system fails, the force required for initial stick movement may increase to as much as double the force normally required, and positive stick centering may be lost. When secondary hydraulic system failure occurs, the following will be inoperative:

- Pitch and yaw damper systems
- Emergency ac generator
- Speed brakes
- Nose wheel steering
- Normal landing gear extension system.

4. Land as soon as practicable.
5. If primary hydraulic system has failed, RAT handle — Pull (just after turn on final approach) & hold for four seconds.

   Hold the handle in fully extended position for four seconds to insure a satisfactory extension of the RAT.

6. If secondary hydraulic system has failed, emergency landing gear extension handle — Pull.

   If secondary hydraulic system has failed, use the emergency pneumatic system to extend gear. Use the normal gear extension if primary hydraulic system has failed.

CAUTION

- Normal landing gear extension with the primary hydraulic system inoperative should be made at a time when minimum use of flight controls is required.
- After emergency landing gear extension, do not attempt to retract gear.

FAILURE OF BOTH HYDRAULIC SYSTEMS

If the hydraulic pressure-low warning light illuminates steadily, use the following procedure:

1. Hydraulic pressure gage — Check.

   Verify that both systems have failed by checking the hydraulic pressure gage.

Note

In event of a frozen engine the hydraulic pressure gage will be inoperative. If the ac generator has failed and the engine is not frozen, the emergency ac generator should be energized. If flight instruments fuel quantity gage or fuel flow indicator operate, then secondary hydraulic pressure is available to operate the emergency ac generator hydraulic motor, and pressure indications should be noted.

2. **RAT — PULL AND HOLD FOR FOUR SECONDS.**

   Extend the RAT by pulling the hydraulic emergency power handle. Hold the handle in the fully extended position for four seconds to insure a satisfactory extension of the RAT.

WARNING

If airspeed is above RAT maximum extension speed and engine is frozen do not extend the speed brakes to slow the airplane. Speed brakes 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.

3. Check flight control operation by:
   a. Checking that hydraulic pressure-low warning light starts to flash indicating that the RAT is providing pressure through the primary system.
   b. Checking hydraulic pressure gage (if ac power is available).
   c. Moving the control stick and checking control response.

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 warnng light will be the only instrument for indication of hydraulic pressure.

4. **IF FLIGHT CONTROL OPERATION IS NOT POSSIBLE — EJECT (REFER TO EJECTION, THIS SECTION).**
5. If flight control is still possible but the pressure gage indicates both systems are below 800 psi and the hydraulic failure warning light is still illuminated steadily – Eject.
6. If pressure is available for flight control operation, and either ac or dc electrical power is available to give an indication that at least one system is operating, use the following procedure:
   a. Restrict flight to avoid violent maneuvers and need for speed brakes.
   b. Extend landing gear by emergency system.
   c. 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.

HIGH-PRESSURE PNEUMATIC SYSTEM FAILURE

If the "PNEU LOW" warning light illuminates in flight prior to usage of the high-pressure pneumatic system, land as soon as practical. Either wheel brakes or drag chute may not be available after landing. If a high-pressure pneumatic system failure is suspected, land as short as possible and use nose wheel steering as soon as the airplane has been slowed sufficiently. Apply brakes cautiously to determine if both brakes operate.

FLIGHT CONTROL EMERGENCY OPERATION

"HYD OIL HOT" WARNING LIGHT ILLUMINATION

On some airplanes* a "HYD OIL HOT" warning light may give advance warning of hydraulic fluid overheat conditions and possible subsequent flight control oscillations. If the "HYD OIL HOT" warning light illuminates, proceed as follows:

Before takeoff —

- Engine — Shut down.

  If the "HYD OIL HOT" warning light illuminates before takeoff, shut down the engine to prevent further damage to hydraulic system components.

During takeoff —

- Takeoff — ABORT.

  If the "HYD OIL HOT" warning light illuminates during takeoff, and runway and overrun conditions permit, abort the takeoff; refer to ABORT, this Section.

In flight —

1. Reduce airspeed to 220-240 KIAS and reduce engine rpm to minimum required to maintain this speed.
2. Flight controls and other hydraulically operated components. Use minimum necessary to maintain safe operating conditions.

**Airplanes modified by TCTO 1F-102-847.*

3. Do not extend the RAT if hydraulic pressure is available.

Note

Extension of the RAT will serve only to increase hydraulic temperatures in the primary system.

4. Land as soon as practicable.

FLIGHT CONTROL SYSTEM OSCILLATIONS

Flight control oscillations are primarily due to hydraulic fluid overtemperatures, but may also be due to failure of the damper system or failure of the artificial feel system. Gradual failure of either hydraulic system may be accompanied by 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 a movement of the control stick; however, if the oscillations were due to the damper system there will be no accompanying movement of the control stick. On some airplanes* a warning light, located on the master warning light panel, is provided to give an indication of hydraulic system overtemperatures. If the "HYD OIL HOT" warning light illuminates and uncontrollable oscillations are encountered, proceed as follows:

1. **REDUCE AIRSPEED TO 220-240 KIAS AND REDUCE ENGINE RPM TO MINIMUM NECESSARY TO MAINTAIN THIS SPEED.**
2. Flight mode — DIRECT MAN. Use minimum control stick movement; attempt to fly hands off for at least 30 seconds.

Note

Pilot induced oscillations, which may occur at transonic speeds when the flight mode is DIRECT MAN, are a phenomenon not related to flight control system oscillations.

3. Do not extend the RAT.

Note

Extension of the RAT will serve only to increase the hydraulic temperature in the primary system.

4. Select alternate flight modes (DIRECT MAN, DAMPER and PILOT ASSIST) and remain in the flight mode where oscillations are at a minimum.

Note

To aid in reducing hydraulic system fluid temperatures, use minimum control movement at all times. Oscillations may be more rapidly reduced in magnitude by releasing the control stick than by correcting for the oscillations with stick input.

5. If oscillations continue, utilize the following additional procedures:
   a. Use rudder to counteract roll.
   b. Maintain safe altitude while severe oscillations persist.
6. During low fuel conditions, keep nose attitude level.

Note

The heat exchanger utilizes fuel from the No. 3 fuel 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 descent to a landing.

7. Maintain safe altitude to evaluate nature of the system overheat.

Note

The system may cool if flight is continued under cautions dictated by the foregoing procedures.

8. When oscillations have been reduced to a safe minimum, land as soon as practicable.

Note

The oscillations may be caused by progressive failure of one hydraulic system, in which case complete failure of the malfunctioning system will usually terminate the oscillations.

TRIM FAILURE

If any of the three trim actuators should fail to operate the force required to perform normal maneuvers should be well within the normal physical capabilities of the pilot. Complete loss of dc power will cause the trim system to be inoperative.

PITCH AND YAW DAMPER SYSTEM FAILURE

The damper systems become inoperative if the main ac generator fails, the dc generator fails, or when secondary hydraulic system pressure fails. If any of these conditions occur the rudder and elevon servo actuators (extendible links) which are spring-loaded to center position will return to center and then serve as a fixed link. If a malfunction of the pitch and yaw damper system (i.e. failure

Airpalnes modifed by TCTO 1F-102-847.

to remain engaged or unusual oscillations) makes flight with the system off mandatory, the following precautions shall be observed:

- Roll maneuvers shall be restricted to positive quadrant rolls only.

Note

A positive quadrant roll is a roll to a 90-degree bank either side of level flight, or from a 90-degree bank in one direction to a 90-degree bank in the opposite direction passing through normal flight position.

- No uncoordinated maneuvers shall be performed, except during takeoff and landing.
- Rapid application of aileron shall be avoided, except during takeoff and landing.
- No intentional rolling push-downs shall be performed.

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, 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 low stick and/or rudder forces, excessive maneuvering, or pilot-induced oscillations. 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. Turn coordination, as normally applied through the yaw damper system, will not provide the desired amount of rudder to coordinate turns during flight with the intakes plugged. Flight under these conditions will probably require disengaging the yaw damper system. Elevator stick forces will be normal or low in all cases of "q" intake clogging. With the small intake clogged, elevator will be normal below .5 Mach, and low above that range, requiring caution at higher airspeeds to avoid overcontrolling. 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 normal during approach and landing. In the event of artificial feel system failure, proceed as follows:

1. **AIRSPEED — REDUCE TO 220-240 KIAS.**
2. Use minimum control stick movement; attempt to fly hands off for at least 30 seconds.
3. Yaw damper system—Disengage.

LANDING GEAR EMERGENCY OPERATION

EMERGENCY GEAR LOWERING PROCEDURE

The emergency landing gear lowering procedure is outlined on figure 3-7. Refer to Section VII for a discussion of landing gear operation.

WINDSHIELD FAILURE

If a panel cracks in flight and is still transparent (indicating outer layer failure only), cabin pressure need not be dumped. However, if the panel becomes white or opaque (indicating failure of the inner layer, which is the stress-bearing part of the windshield), then cabin pressure should be dumped immediately. If cabin pressure is released at an altitude of approximately 43,000 feet, the airplane oxygen system will automatically supply pressure to the partial pressure suit. The mission can be completed safely so long as there is sufficient oxygen in the airplane to furnish pressure to the partial pressure suit and oxygen for pilot breathing.

Note

Additional oxygen is available to the partial pressure suit through the oxygen bailout bottles, which can be actuated by manually pulling the green knob. Because of the limited duration of the bailout bottles, an immediate descent to a safe altitude is necessary.

In the event of windshield failure, proceed as follows:

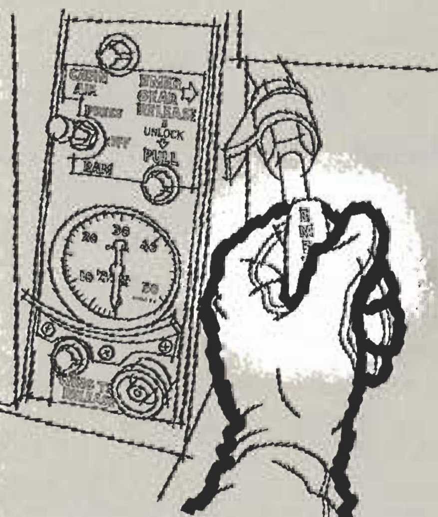
1. **CABIN AIR SWITCH — RAM (IF PANEL BECOMES OPAQUE).**
2. Nesa switches — OFF.

Note

If it is apparent during flight that the windshield has been or is being damaged, distorted, or discolored from uneven heat distribution, the cause may be due to failure of the power relay or control box. If such is the case, turning the Nesa switch to OFF will not remove electrical power from the windshield heating elements. It will be necessary to either pull the three-phase circuit breakers placarded "Windshield Anti-Ice," which are located on the left-hand aft circuit breaker panel, or place the ac bus switch to EMER, thus removing power from the Nesa heaters. The circuit breakers are difficult to reach in flight. Before moving the ac bus switch to EMER it must be determined whether the nature of the emergency is such that flight may be continued safely without additional items connected to the nonessential bus as they also will be rendered inoperative.

landing gear emergency lowering

1 REDUCE SPEED BELOW GEAR DOWN LIMIT SPEED.



2 PUSH DOWN (UNLOCK) THEN PULL THE LANDING GEAR EMERGENCY HANDLE APPROXIMATELY 2 INCHES.

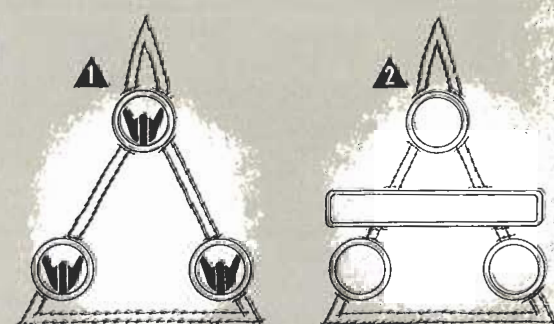
NOTE

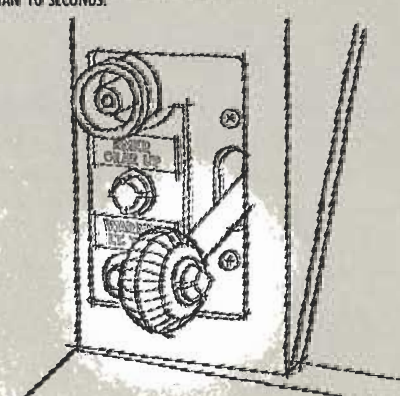
- ON AIRPLANES AF53-1791 THRU 56-972, THIS HANDLE IS PLACARDED "EMER GEAR RELEASE PULL" AND MAY BE PULLED STRAIGHT OUT. CHECK THAT SPRING CLIP OR DETENT ENGAGES IN BACK SIDE OF EMERGENCY LANDING GEAR EXTENSION HANDLE TO PREVENT INWARD MOTION OF THE HANDLE.
- ON AIRPLANES AF56-973 AND ON, THE HANDLE IS PLACARDED "EMER GEAR RELEASE—UNLOCK—PULL" AND AUTOMATICALLY LOCKS WHEN IN THE OUT POSITION.

3 CHECK LANDING GEAR POSITION INDICATORS FOR GEAR DOWN (WHEELS).

CAUTION

IN PLANNING THE LANDING WHEN THE GEAR IS TO BE LOWERED BY THE EMERGENCY SYSTEM, ONE MINUTE SHOULD BE ALLOWED FOR GEAR EXTENSION TIME ON AIRPLANES AF53-1791 THRU 56-1518 UNLESS MODIFIED BY T.O. 1F-102-655. ON AIRPLANES AF57-770 AND ON AIRPLANES MODIFIED BY T.O. 1F-102-655, EMERGENCY EXTENSION SHOULD NOT REQUIRE MORE THAN 10 SECONDS.





4 LANDING GEAR HANDLE DOWN.

NOTE

THE NORMAL LANDING GEAR HANDLE SHOULD BE IN THE DOWN POSITION PRIOR TO LANDING. IF THE REQUIREMENT FOR EMERGENCY GEAR EXTENSION IS DUE TO FAILURE OF THE SECONDARY HYDRAULIC SYSTEM, OR ENGINE FAILURE, THE EMERGENCY GEAR EXTENSION HANDLE SHOULD BE PULLED FULLY OUT PRIOR TO PLACING THE NORMAL GEAR HANDLE DOWN. THIS ACTION WILL CONSERVE ALL AVAILABLE HYDRAULIC PRESSURE FOR FLIGHT CONTROL OPERATION.

CAUTION

- NORMAL LANDING GEAR EXTENSION WITH THE PRIMARY HYDRAULIC SYSTEM INOPERATIVE SHOULD BE MADE AT A TIME WHEN MINIMUM USE OF FLIGHT CONTROLS IS REQUIRED.
- NOSE WHEEL STEERING IS INOPERATIVE WHEN GEAR IS EXTENDED BY THE EMERGENCY SYSTEM.

1 BEFORE MODIFICATION BY T.O. 1F-102-728.

2 AFTER MODIFICATION BY T.O. 1F-102-728.

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Figure 3-7

CANOPY WARNING LIGHT ILLUMINATED IN FLIGHT

Illumination of the canopy unlocked warning light while 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. Do not move the canopy latch handle.
2. Reduce airspeed to 230-240 KIAS.
3. Cabin air switch to OFF.
4. Return to base as soon as possible.

Note

The Emergency Abbreviated Check List is contained in T. O. 1F-102A-(CL)1-1.

TABLE OF CONTENTS

COCKPIT AIR-CONDITION AND PRESSURIZATION SYSTEM

The cockpit air-conditioning system (figure 4-1) uses air from the low-pressure pneumatic system to provide the pilot with a pressurized, temperature-controlled cockpit. For air-conditioning and cockpit pressurization, hot compressed air (engine bleed air) is ducted to the cockpit by a system which routes the air through a refrigeration unit and also provides a bypass around the refrigeration unit. In the refrigeration unit, the hot air is first routed through an air-to-air heat exchanger and then through a cooling turbine where it is cooled and expanded. A centrifugal flow fan, mounted on a common shaft with the cooling turbine wheel, absorbs the energy of the turbine wheel by drawing cooling ram air across the heat exchanger. Cold air from the refrigeration unit is mixed with hot bypassed air and then discharged into the cockpit through two discharge tubes on each side of the cockpit. Cockpit temperature control is achieved by automatically modulating a bypass valve which mixes the proportion of hot and cold air required to maintain the temperature selected by the pilot. Temperature sensing is accomplished by two thermostats, one located in the cockpit air inlet duct and the other in the outlet duct. When maximum cooling is required, all the air is routed through the refrigeration unit. Because full ram air is not flowing over the air-to-air heat exchanger, cockpit cooling is limited while the airplane is on the ground. In the event the normal air-conditioning system malfunctions during flight, a ram air system can be used to provide cockpit ventilation. This ram air ventilation system does not provide cockpit pressurization. A ram air pressure regulator and shutoff valve shuts off the ram air supply if ram air temperature or pressure becomes excessive. A check valve located in the air duct entering the cockpit on some airplanes* will prevent rapid depressurization of the cockpit in the event of engine flameout or duct failure. Pressurization of the cockpit is provided by regulating the discharge of air from the cockpit. The pressure regulator in the floor of the cockpit automatically controls the outflow of cockpit air to maintain a preset cockpit pressure schedule. See figure 4-2. This schedule provides for an unpressurized cockpit up to an altitude of 10,000 feet. As the airplane continues to climb, the cockpit altitude remains at

*AF 56-1275 thru -1316, -1332 & on, & airplanes modified by TCTO 1F-102-656.

low-pressure pneumatic and air conditioning systems

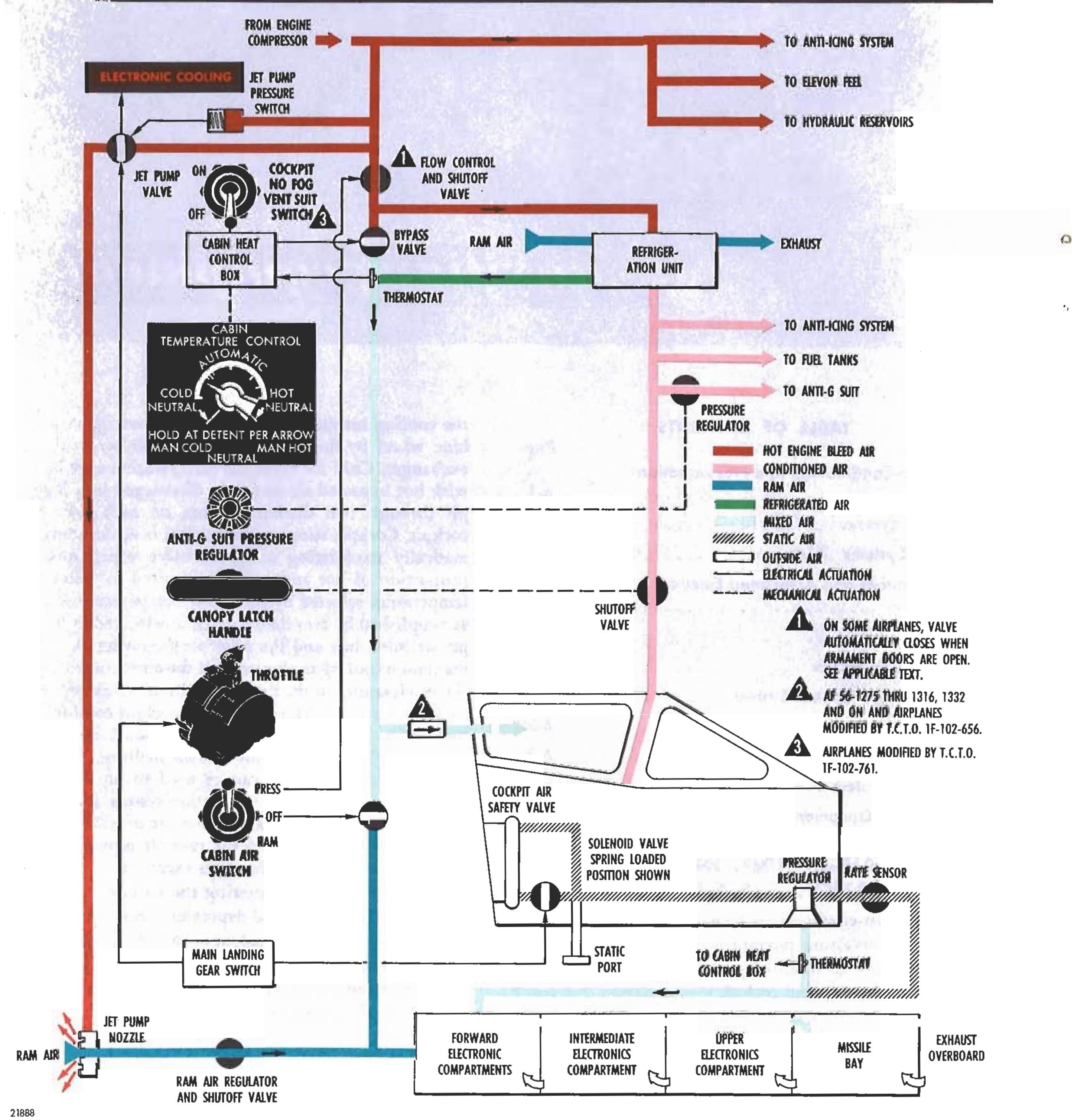


Figure 4-1

10,000 feet until an airplane altitude of approximately 26,500 feet is reached. Above this altitude a constant 5.0 psi differential pressure is maintained.

Note

Cockpit pressure fluctuations may be experienced at altitudes between 5000 feet and 15,000 feet in a speed range of 200 KIAS to 400 KIAS.

To prevent smoke and fumes from entering the cockpit, the bleed air shutoff valve shuts off the conditioned air supply to the cockpit on some airplanes* while the armament bay doors are opened for armament firing. Due to leakages, cockpit pressure may drop while the air supply is shut off. When pressurization is resumed, a rate sensor controls the speed at which the cockpit is repressurized. This prevents an initial pressure surge.

Note

Pressurization is resumed three to five seconds after the armament bay doors close or, if door closing is delayed by a misfire, by action of a time delay.

On other airplanes, in order to maintain cockpit pressurization during armament firing, cockpit pressurization is not shut off and some smoke or fumes in the cockpit may be experienced during firing.

CABIN AIR SWITCH

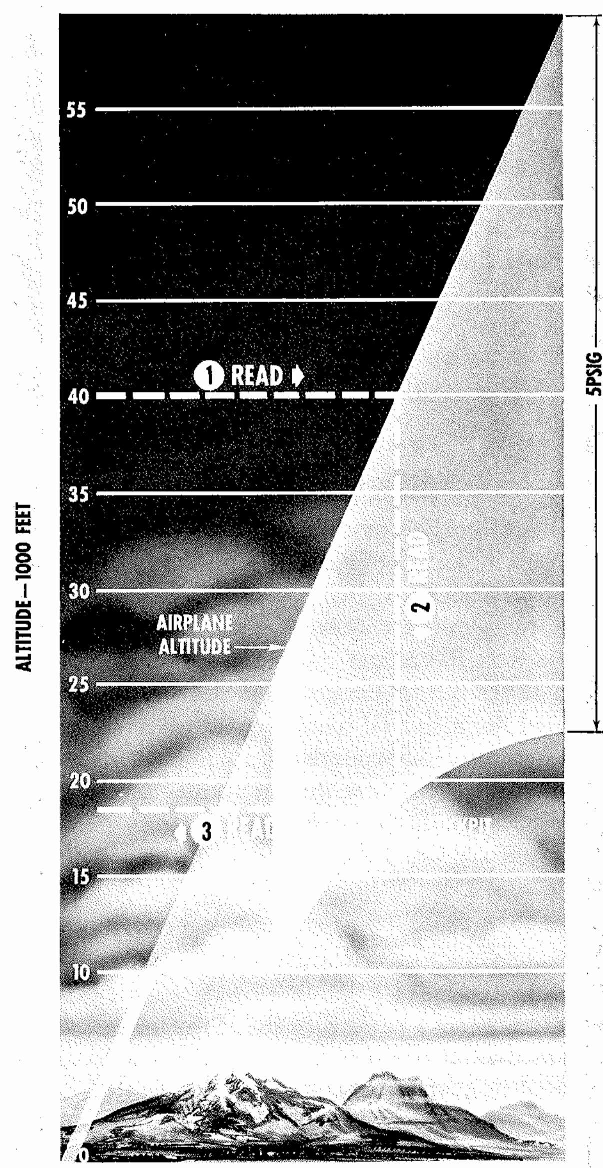
The cabin air switch (figure 1-24) is located on the left-hand auxiliary instrument panel. The switch is placarded "Cabin Air" and has three positions, PRESS, OFF, and RAM. When the switch is in the PRESS position, the ram air shutoff valve is closed to shut off the flow of ram air to the cockpit and the bleed air shutoff valve is open to route engine bleed air through the refrigeration and bypass unit to the cockpit. When the switch is in the RAM position, the ram air shutoff valve is open to route ram air to the cockpit, the bleed air shutoff valve is closed to shut off the engine bleed air supply, and the cabin safety valve is open to dump cockpit pressure. When the switch is in the OFF position both the ram air shutoff valve and the bleed air shutoff valve are closed to shut off the entire system. The cabin air switch receives power from the 28-volt dc essential bus.

CABIN TEMPERATURE CONTROL KNOB

The rotary cabin temperature control knob (figure 1-21) on the utility switch panel provides for automatic control of cockpit temperature within a range of 45°F to 100°F

*AF 53-1792, -1796, -1798 thru -1805, -1807 thru -1811, -1814 thru -1818, 54-1371 thru -1379, -1381 thru -1387, -1389, -1391 thru -1397, -1399, -1400, -1402 thru -1407, 55-3357 thru -3438, -3440 thru -3464, 56-957 thru -975, -977 thru -1088, -1090 thru -1098 unless modified by TCTO 1F-102-647.

cockpit pressurization schedule



EXAMPLE

(REFER TO DASH LINE) AIRPLANE ALTITUDE OF 40,000 FT EQUALS COCKPIT ALTITUDE OF APPROXIMATELY 17,000 FT.

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Figure 4-2

or for manual control of the system in the event of malfunction of the automatic control. Temperature control is accomplished by the positioning of the refrigeration unit bypass valve to mix hot bleed air with refrigerated air in the required proportions to meet either the automatic or manual demands of the system. The temperature control switch is ineffective when the cabin air switch is in RAM position. The temperature control switch has an AUTOMATIC range with a NEUTRAL position at both extremes of automatic selection. The knob can be held against light spring tension to the MAN HOT or MAN COLD position to override the automatic system. When the knob is released it will return to the NEUTRAL position.

Note

If too much pressure is applied to the switch during manual operation, the switch will pass through the manual position into a common neutral range. The switch will not function from this range and should be returned to its previous position.

The switch controls power from the 28-volt dc essential bus.

COCKPIT NO-FOG VENT SUIT SWITCH

The cockpit no-fog vent suit switch* is located on the utility switch panel (figure 1-21) and has ON and OFF positions. When the switch is placed in the ON position, cockpit temperature control is governed by the cockpit air inlet duct thermostat only. A steady temperature of 80° (±10°)F is then maintained in the cockpit and in the ventilated flight suit (if used), regardless of the position of the automatic temperature control knob. This temperature range will minimize the probability of cockpit fog from the air conditioning system. Power to the cockpit no-fog vent suit switch is supplied by the 28-volt dc essential bus.

CABIN PRESSURE ALTITUDE GAGE

The cabin (cockpit) pressure altitude gage (43, figure 1-4) is located on a panel forward of the left console and indicates cockpit pressure in feet above sea level. It is graduated from zero to 50,000 feet pressure altitude. The gage is vented only to pressure within the cockpit.

NORMAL OPERATION OF COCKPIT AIR-CONDITIONING AND PRESSURIZATION SYSTEM

1. Place cabin air switch in PRESS position.
2. Close canopy.
3. Cabin temperature knob as desired within the AUTOMATIC range.
4. Cockpit no-fog vent suit switch as required.

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, and outside visibility may also be completely obscured. Cockpit fog is most likely to occur on takeoff, but may also occur during landing or go-around. To prevent fog formation, the temperature control knob should be placed in the AUTOMATIC HOT range, or the cockpit no-fog vent suit switch should be placed to ON (if installed), prior to takeoff or landing.

EMERGENCY OPERATION OF COCKPIT AIR-CONDITIONING AND PRESSURIZATION SYSTEM

If automatic temperature control system becomes inoperative or if temperatures beyond the normal limits of the system are desired:

1. Hold cabin temperature control knob momentarily to MAN HOT or MAN COLD.
2. Repeat as necessary until desired change occurs.

If cabin pressurization fails and cabin altitude increases appreciably over actual altitude:

1. Determine cabin altitude.
2. If cabin altitude is above the actual altitude of the airplane, place cabin air switch to RAM, and/or reduce speed.

Note

When the cabin air switch is in RAM position, the following additional systems become inoperative: anti-g suit pressurization, radome anti-icing, and canopy sealing.

WARNING

If excessive cockpit fog occurs during takeoff, landing, or go-around, immediately place cabin air switch to RAM position to shut off the source of cockpit fog and to aid in more rapidly dissipating the fog. After the fog has cleared, or the possibility of fog formation no longer exists, resume normal operation of the cockpit air-conditioning and pressurization systems.

**Airplanes modified by TCTO 1F-102-761.*

DEFOGGING SYSTEM

Warm air from the low-pressure pneumatic system is used to defog the canopy panels and, on some airplanes*, the forward one-third of the windshield. Flow of this air is controlled by a switch on the utility switch panel. The windshields are defogged by electrically heating the Nesa glass panels. Refer to WINDSHIELD ANTI-ICING, this Section.

CANOPY DEFOG SWITCH

The canopy defog switch (figure 1-21) on the utility switch panel has CANOPY DEFOG and OFF positions. The switch controls 28-volt dc essential bus power to a motor driven valve in the hot air duct. When the switch is in CANOPY DEFOG position the valve is open and warm air is directed to the canopy and, on some airplanes*, to the forward one-third of the windshield. When the switch is in OFF position the valve is closed. Limit switches shut off the motor when the valve reaches the fully open or fully closed positions.

OPERATION OF CANOPY DEFOG SYSTEM

Note

During operation in high humidity conditions (ground dew point 20°C or higher) operate canopy defog continuously at altitude in order to insure a fog-free canopy during descent. For less severe humidity conditions, canopy defog switch need not be turned ON until just before descent.

The canopy defog system is either fully on or fully off, having no provisions for regulation by the pilot. To operate the system, proceed as follows:

1. To turn on the system, place the canopy defog switch in CANOPY DEFOG position.
2. To turn the system off, return the canopy defog switch to OFF.

Note

In the event of canopy defog system failure, hold the cabin temperature control knob to MAN HOT as necessary when canopy fogging occurs.

ANTI-ICING SYSTEMS

The airplane is equipped with an automatic anti-icing system that will provide automatic protection for the engine, intake duct leading edges, engine inlet guide vanes, radome, and ram air, "q" intakes on the fin if the system is armed. The automatic system may be overridden and operated manually in the event it malfunctions. Separate controls also provide for operation of the windshield anti-icing system, windshield rain clearing system, and the pitot-static head anti-icing system. Wing and fin anti-icing is not provided.

**AF 56-1275 thru -1316, -1332 & on.*

Note

The airplane configuration is such that wing and tail ice formations have no effect on stability and control. Engine thrust available is adequate to compensate for any drag increase.

ICE DETECTOR

An automatic ice detector probe is mounted in the engine intake duct to sense icing conditions and automatically turn on the ice protection system if the system is armed. When the ice protection system turns on, the detector immediately starts to de-ice. De-icing of the detector usually takes less than a second. If the probe becomes clogged or anti-ice heat to the probe remains on longer than approximately 20 seconds, the pilot is informed through the master warning system, and anti-ice detection is inoperative until the probe is unclogged. If, after the detector is de-iced, there is no further indication of ice for a period of one minute, the ice protection system will turn off automatically until the next time the detector senses ice.

ENGINE ANTI-ICING

The engine inlet guide vanes at the face of the compressor are anti-iced by engine bleed air from the low-pressure pneumatic system. The thermostatically controlled air flows through the hollow guide vanes. This system is a component part of the engine and is controlled through the automatic ice protection system.

INTAKE DUCT ANTI-ICING

The engine intake ducts are supplied with hot engine bleed air from the low-pressure pneumatic system. The thermostatically controlled bleed air flows through the double skin structure of the duct leading edges and is exhausted on the inside surface aft of the leading edge. The system is controlled through the automatic ice protection system.

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. Low-pressure pneumatic system air is used to force the glycol from the tank and through the porour 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 automatic ice protection system. See figure 1-33 for anti-icing fluid specifications.

RAM AIR "q" PRESSURE INTAKE ANTI-ICING

The ram air pressure intakes on the fin are electrically anti-iced by 28-volt dc power from the essential bus. The system is controlled through the automatic ice protection system, and is operative when the anti-ice switch is in MANUAL ON or AUTO. The intake heaters receive only partial power until the right main landing gear door is closed.

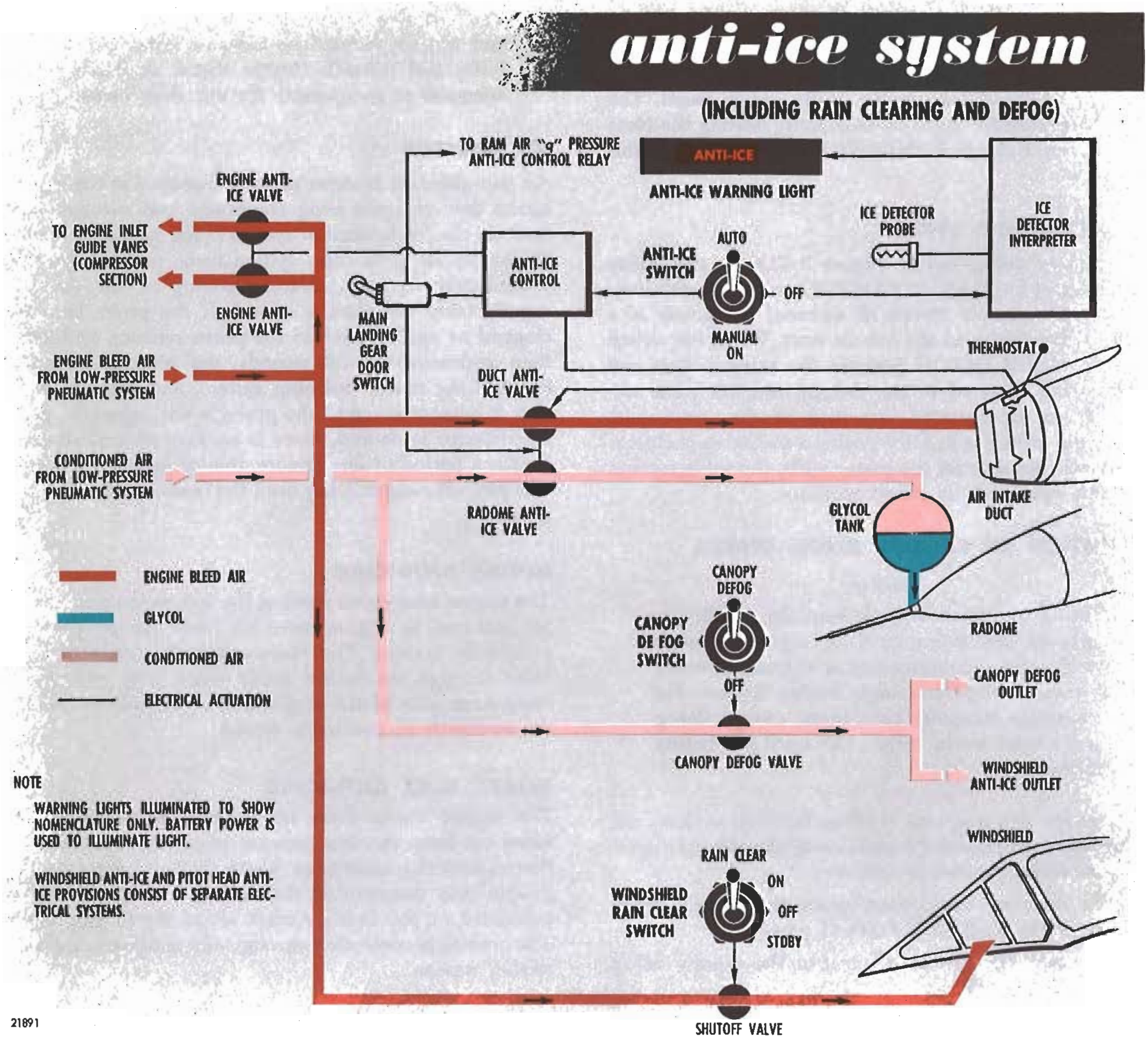


Figure 4-3

WINDSHIELD ANTI-ICING AND DEFOGGING

Anti-icing and defogging of the windshield panels is accomplished by electrical heating units imbedded in the Nesa panels. These units have temperature sensing elements to prevent overheating of the glass. On some airplanes anti-icing and defogging of the aft two-thirds is accomplished by electrical heating, the forward one-third uses hot air sent through ducts and distribution nozzle to the apex and is controlled by the canopy defog switch.

WINDSHIELD RAIN CLEARING SYSTEM

The windshield rain clearing system provides a high-velocity stream of hot air from the low-pressure pneumatic system to clear rain or snow from the left windshield panel. A switch in the cockpit enables the pilot to control the electrically operated shutoff valve in the rain clear hot air duct.

ANTI-ICE SWITCH

The anti-ice switch (figure 1-21) on the utility switch panel has three positions, AUTO, MANUAL ON, and OFF. When the switch is in AUTO position, 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 ice detection system through the ice detector. When the switch is in MANUAL ON position, the automatic features of the ice detection system are bypassed and all of the anti-icing systems which are controlled by the automatic ice detector system are activated regardless of icing conditions. On some airplanes*, it is necessary to pull the toggle switch out and then down to place the switch in MANUAL ON. On some airplanes** the system is deenergized if the anti-ice switch is in OFF position. On other airplanes, however, only the anti-ice warning light is deenergized and the anti-ice circuit remains energized as if in AUTO position. The anti-ice switch receives power from the 28-volt dc essential bus.

WINDSHIELD ANTI-ICE (NESA) SWITCH

The windshield anti-ice and defog system is controlled by switches located on the utility switch panel (figure 1-21). Some airplanes have a single switch, placarded "Nesa," with NORMAL and OFF positions. Placing the switch in NORMAL position connects 200/115-volt ac nonessential power to the Nesa transformers for both panels. Other airplanes† have two Nesa switches, one for the left and one for the right windshield panel. These switches are placarded "Nesa Control LH-RH" and provide the same functions for their respective panels as the single switch on the earlier configuration.

WINDSHIELD RAIN CLEAR SWITCH

The windshield rain clear switch is located on the utility switch panel (figure 1-21). On some airplanes, the switch has two positions, RAIN CLEAR, and OFF. In the RAIN CLEAR position, the switch supplies 28-volt dc essential bus power to the solenoid valve in the rain clear hot air duct. Other airplanes† have a third switch position, STDBY, which provides for automatic operation of the windshield rain clearing system if the Nesa system fails due to an ac power failure.

PITOT HEAT SWITCH

The pitot-static head anti-icing switch (figure 1-21) on the utility switch panel is placarded "Pitot Heat" and has ON and OFF positions. When in the ON position, the switch supplies 28-volt dc essential power to the pitot-static head on the nose boom.

ANTI-ICE SYSTEM WARNING LIGHT

A light on the warning light panel (figure 1-26) will illuminate to display "ANTI-ICE" when the anti-ice switch is OFF or when the airflow through the engine intake duct is under 40 knots with the switch in AUTO. The light will illuminate during normal operation (anti-ice switch AUTO) when the ice detector probe heater is on for 17 to 20 seconds and the probe openings remain clogged to indicate automatic anti-ice system is inoperative. The light will remain on until the probe is no longer clogged or the anti-ice switch is moved to MANUAL ON. The warning light is part of the master warning system and operates from the 28-volt dc essential bus.

**AF 56-1234 thru 56-1316, 56-1321 & on, & airplanes modified by TCTO 1F-102-664.*
***AF 56-1045 & on.*

NORMAL OPERATION OF ANTI-ICING SYSTEMS

The windshield anti-icing system, which also provides defogging, should be turned on as soon as ground electrical power is connected to the airplane and left on continuously until engine shutdown after completion of flight. This is necessary to insure proper defogging.

1. Place anti-ice switch in AUTO position. Under known or suspected icing conditions, place anti-ice switch in MANUAL ON.
2. Place windshield anti-ice (Nesa) switch in NORMAL (some airplanes); both Nesa switches ON (other airplanes)†.
3. Place rain clear switches in STDBY (some airplanes)†.
4. Move pitot heat switch to ON.

EMERGENCY OPERATION OF ANTI-ICING SYSTEMS

Automatic Anti-Icing System.

1. Place anti-ice switch to MANUAL ON position.

Windshield Anti-Icing System.

In the event of loss of ac electrical power to the laminated windshield, windshield defogging may be obtained by the following procedure:

1. If windshield is heated and power failure is noted immediately, place windshield rain clear switch to RAIN CLEAR.

   If the system has functioned normally, maintaining a heated windshield, and ac power failure is noted, the rain clearing system should be turned on immediately. Activation of the rain clearing system will provide an effective "heat block" for the heat energy stored in the left-hand windshield panel at the time of the power failure and will maintain the inside surface temperature of the left-hand panel.

2. If ac power failure is not noted until condensation has formed, proceed as follows:
   a. Windshield rain clear switch to RAIN CLEAR. Activation of the rain clearing system will initiate a heat addition process on the left-hand windshield outer surface.

†Airplanes modified by TCTO 1F-102A-562.

communications and associated electronic equipment

▲ SOME AIRPLANES—REFER TO APPLICABLE TEXT

TYPE	DESIGNATION	FUNCTION	PRIMARY OPERATOR	RANGE	LOCATION OF CONTROLS
Interphone Equipment	AN/AIC-10	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.	Pilot and ground crew.	Cockpit to ground crew.	Ground crew connection and amplifier in the nose wheel well.
UHF Command Radio	AN/ARC-34	Communications from airplane to airplane and from airplane to ground by UHF communications radio.	Pilot	Line of sight.	Left-hand console.
Visual Omni-range Receiver ▲	AN/ARN-14	Provides information for navigation and instrument low approach.	Pilot	Localizer approx. 45 miles and omni-directional 100 miles, depending on the flight altitude.	Right-hand console. Indicator on instrument panel.
Glide Slope Receiver ▲	AN/ARN-18	Indicates glide angle for automatic approach.	Pilot	Approximately 25 miles.	Right-hand console. Indicator on instrument panel.
Tacan ▲	AN/ARN-21	Used in conjunction with Tacan surface beacon to provide bearing and range information.	Pilot	Approximately 195 miles, line of sight.	Right-hand console. Indicators on instrument panel.
ILS Receiver ▲	AN/ARN-31	Receives ILS localizer and glide path transmission.	Pilot	Localizer approximately 45 miles, glide path approximately 25 miles.	Right-hand console. Indicator on instrument panel.
Marker Beacon Receiver	AN/ARN-12 or AN/ARN-32	Receives location marker signal on navigational beam.	Pilot		Indicator light on instrument panel.
IFF ▲	AN/APX-6A AN/APX-25	Automatic radar identification returned if challenged by surface or airborne sets. Can transmit emergency identification signal.	Pilot	Line of sight.	Right-hand console.

21909

Figure 4-4

b. Cabin temperature control knob to MAN HOT. Regulate cabin air 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 that was not preheated electrically, will require from two to four minutes when using the above procedure.
- On some airplanes*, the rain clearing system will come on automatically after an electrical power failure if the rain clear switch is in STDBY position.

COMMUNICATIONS AND ASSOCIATED ELECTRONIC EQUIPMENT

TABLE OF COMMUNICATION AND ASSOCIATED ELECTRONIC EQUIPMENT

See figure 4-4.

INTERPHONE — AN/AIC-10

An interphone system is installed to permit communications between the pilot and a ground crew through a nose wheel well connection, when the airplane is on the

**Airplanes modified by TCTO 1F-102A-562.*

ground, and also to provide an amplifier for the dynamic microphone for transmitting. The dynamic microphone is the push-to-talk type. The pushbuttons are located on the throttle grip (figure 1-7) and on the control stick grip (figure 1-20). There is no cockpit interphone control panel. The system is on whenever power is available from the 28-volt dc essential bus or, on some airplanes*, from the emergency dc bus.

UHF COMMAND RADIO — AN/ARC-34

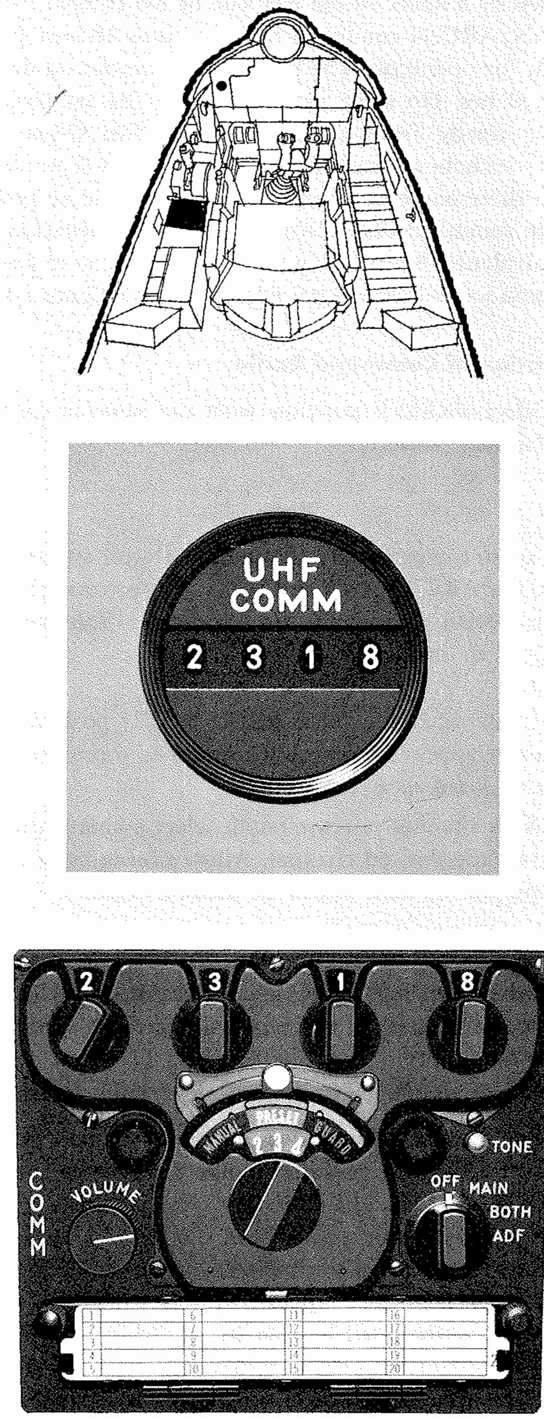
Note

No transmission will be made on emergency (distress) frequency channels except for emergency purposes in order to prevent transmission of messages that could be construed as actual emergency messages.

The AN/ARC-34 command radio set provides voice transmission and reception on 1750 frequencies in the range of 225.0 to 399.9 megacycles. The control panel for the set (figure 4-5) is on the left console and permits selection of any of 20 frequencies which can be preset in any order. A remote indicator** (44, figure 1-4) on the instrument panel, will indicate the frequency of the preset channel selected. In addition, an operating frequency can be set up manually without disturbing any of the preset frequencies. The set uses two receivers, a main and a guard receiver. The guard receiver is capable of covering the frequency range of 238.0 to 248.0 megacycles. Tuning of the guard receiver is set and fixed prior to installation. The functions of the set are selected by the four-position main control switch on the right side of the control panel. The switch has four positions: OFF, MAIN, BOTH, and ADF. When the switch is on MAIN position, the transmitter and receiver are operative on the same selected main frequency. The BOTH position allows transmission and reception on the main selected channels and simultaneous reception on the guard channel. The ADF position of the switch is inoperative in this installation. A button placarded "Tone" is adjacent to the main control switch. Holding the tone button down interrupts reception and provides a continuous tone transmission to aid ground stations in obtaining a direction finding bearing. A selector control above the channel selector knob is used to select the desired operating mode. The MANUAL position of the control permits the operating frequency to be changed to any desired frequency in the operating range of the set. The GUARD position selects the fixed guard frequency for the main receiver and transmitter, and PRESET is used to allow selection of any of the 20 preset frequencies. When the selector control is in PRESET, subsequent movement of the channel selector knob (at the center of the control panel) changes the frequency

*Airplanes modified by TCTO 1F-102-727.
**Airplanes modified by TCTO 1F-102-860.

uhf command radio control panel



21892

Figure 4-5

to the desired preset channel. A numerical indication of the selected channel appears in a window above the selector knob. A record of the frequencies that have been preset and assigned to the 20 channels can be noted on a plastic card provided for this purpose and located at the bottom of the control panel. The preset frequencies can be changed in flight if necessary. Audio volume is adjusted by a knob on the left side of the control panel. The AN/ARC-34 command radio requires 28-volt direct current for operation. The power source varies on different airplanes. On some early airplanes, the set receives power directly from the dc essential bus. Other airplanes* include a transformer-rectifier which is powered by the three-phase ac essential bus to produce dc power for the command radio. An additional modification on some airplanes** provides an emergency dc bus to power this radio set and some other equipment (see figure 1-12).

Operation of Command Radio

1. Select PRESET position with the selector control (sliding button).

Note

Due to the proximity of the fuel selector switch to the ARC-34 command radio selector control, be certain that the desired control is selected prior to movement.

2. Rotate main control switch to BOTH position and allow approximately two minutes to warm up main and guard receiver units.
3. With channel selector knob, select a channel other than the desired channel. Allow tuning cycle to be completed, then return to desired channel.

Note

Full power for reception and transmission is not available if the first channel selected is used.

4. Adjust volume control for desired audio level.
5. For manual selection of a frequency that is not included in the preset channels, set selector control to MANUAL. The four windows across the top of the panel should open. Turn the four knobs at the top of the panel until the numerals indicating the desired frequency appear in the windows. (The main control switch must be at MAIN or BOTH for manual frequency selection.) This procedure places the set in receive condition. Transmission on the same frequency is obtained by depressing the microphone button.

Note

- The microphone button should be released before changing transmitter frequency. Approximately four seconds should elapse before transmission begins on a new frequency.
- A thermal relay stops the drive motor after 50 to 125 seconds of continuous operation (switching from one channel to another without stopping). If this occurs, place the main control switch to the OFF position and allow for a 30-second cooling period. Then switch to the BOTH position and again select the desired channel.
- Do not attempt to tune the receiver to any frequency below 225 megacycles, as the radio will not operate in this range. If a frequency is set manually below 225 megacycles continuous operation of the channeling drive motor results.

6. To obtain transmission and reception of the guard frequency, move selector control to GUARD.
7. To turn off receiver-transmitter, move main control switch to OFF.

VOR RECEIVER — AN/ARN-14*

The VHF navigation receiving set consists of a control panel on the right-hand console, a course indicator on the instrument panel, and a radio magnetic indicator also on the instrument panel. The dynamotor for this set is powered from the 28-volt dc essential bus. Indicator power is from the 26-volt ac bus.

VOR Navigation Control Panel

The VHF Navigation control panel (figure 4-6) has a power switch with ON and OFF positions, two frequency selector controls, a frequency window, and a volume-control knob. Rotation of the notched outer frequency control wheel selects the frequency from 108 to 136 megacycles. The inner frequency selector is in the form of a rotary switch key and selects the frequency in tenths of megacycles. The frequency selected is shown in the frequency window.

Course Indicator

The course indicator (34, figure 1-4) has a bearing selector knob, a bearing window, a "TO-FROM" window, a course deviation indicator (vertical pointer), a glide-slope indicator (horizontal pointer), and a heading pointer. The bearing selector knob permits selection of a desired bearing, which is displayed in the bearing window. The "TO-FROM" window indicates whether the

**AF 57-770 & on, & airplanes modified by TCTO 1F-102A-540.*
***Airplanes modified by TCTO 1F-102-727.*

**AF 53-1791 thru 56-1241, -1317 thru -1358, -1369 thru -1392, -1394 & on, unless modified by TCTO 1F-102-719.*

bearing selected is a "to" or "from" radial. The course deviation indicator (CDI) shows the position of the airplane in relation to the desired bearing. The glide-slope indicator is used when the pilot is flying an instrument landing system approach. The heading pointer, actuated by the directional indicator (slaved) system, shows the heading of the aircraft relative to the selected bearing. The marker beacon light is illuminated when signals are received by the marker beacon radio. When the CDI is centered, the selected track will be maintained. If the pointer is deflected to the left or right, the airplane's heading is to the left or right the number of degrees indicated on the pointer scale. When radio signals are unreliable or too weak, an "OFF" flag appears at the end of the vertical bar or at the end of the horizontal bar, depending on which system is unreliable.

Radio Magnetic Indicator

The radio magnetic indicator (RMI) (11, figure 1-4) consists of a rotating compass card, actuated by the directional indicator (slaved) system, and a large and small pointer. Information from the flux valve compass transmitter (J-2 or J-4) turns the compass card under a heading arrow at the top of the indicator to indicate the magnetic heading of the airplane. The double-barred pointer is actuated by the VOR receiver, and points to the magnetic bearing of the omni station being received. The smaller pointer is wired in parallel with the larger one and travels with it.

Operation of VOR Receiver

1. Position power switch to ON.
2. Select desired frequency.
3. Adjust volume to desired level.
4. To turn equipment off, position power switch to OFF.

GLIDE-SLOPE RECEIVER — AN/ARN-18*

This set is used in conjunction with the localizer portion of the omni-directional range radio. Visual glide-slope indications are presented on the omni-range receiver course indicator (34, figure 1-4) by means of the horizontal pointer of that instrument. The glide-slope receiver is controlled from the navigation radio control panel (figure 4-6).

Operation of Glide-Slope Receiver

1. Place the power switch on the VHF navigation control panel, in the ON position.
2. Select the localizer frequency to be used.

vor-control panel

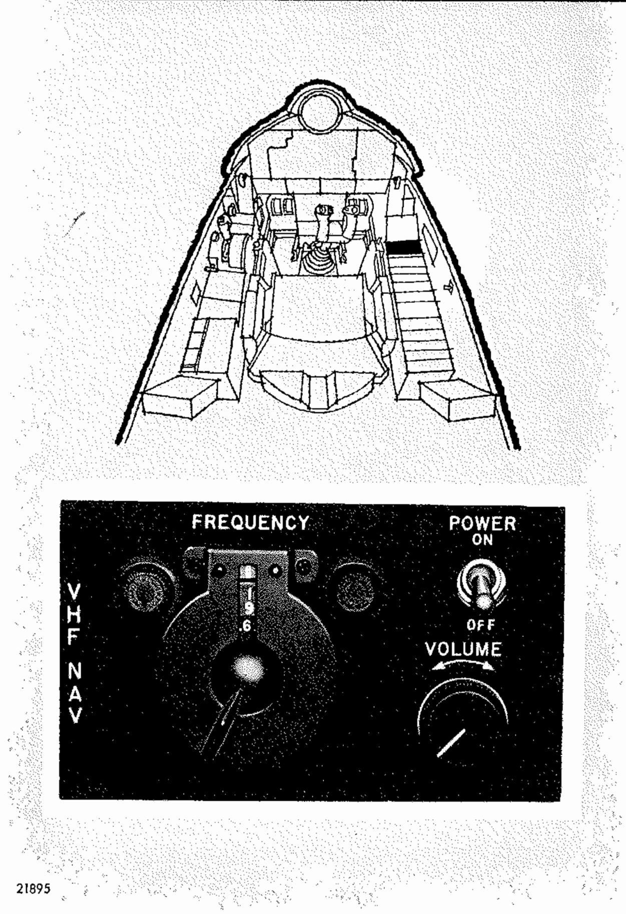


Figure 4-6

3. Observe glide-slope flight characteristics as indicated by the horizontal pointer of the course indicator.
4. To turn set off, place omni-range power switch in OFF position.

TACAN — AN/ARN-21**

This set is designed to operate in conjunction with a specifically constructed surface navigation beacon to form a radio navigation system called Tacan (TACtical Air Navigation). This system enables an equipped airplane to obtain continuous indications of its distance and bearing from any selected Tacan surface beacon located within a line-of-sight distance of approximately 195 nautical miles. The AN/ARN-21 utilizes the radio magnetic

AF 53-1791 thru 56-1241, -1317 thru -1358, -1369 thru -1392, -1394 & on, unless modified by TCTO 1F-102-719.

**AF 56-1242 thru -1316, -1359 thru -1368, -1393, & airplanes modified by TCTO 1F-102-719.*

tacan control panel

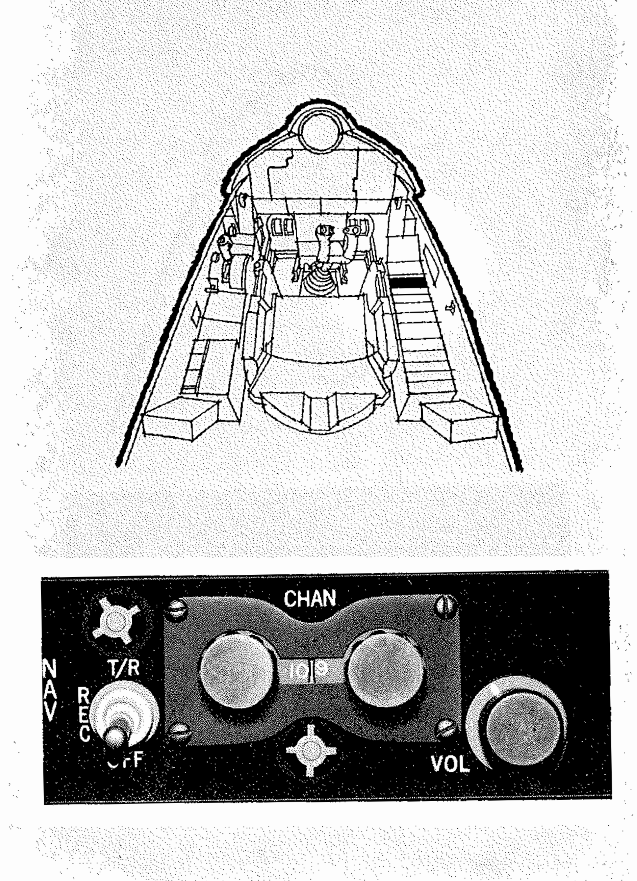


Figure 4-7

indicator, a range indicator, the course deviation indicator (CDI) (vertical pointer) of the course indicator, a control panel, and an instrument selector panel. The AN/ARN-21 transmits interrogation pulses, receives beacon pulses from the Tacan surface beacon and prepares the received information for display on the bearing and distance indicators. The system operates on the following UHF frequency ranges: transmitter—1025 to 1150 megacycles; receiver—962 to 1024 megacycles and 1151 to 1213 megacycles. There are 126 frequency channels, any one of which may be selected by setting the proper controls on the control panel. The equipment can operate at altitudes up to 50,000 feet. However, to prevent the possibility of arcing (flashover) at extreme altitudes, an altitude sensing switch is incorporated which shuts off the equipment automatically at a compartment pressure altitude of 45,000 feet ±2500 feet while climbing or restores operation at approximately 40,000 feet descending. The equipment is powered by the 28-volt dc nonessential bus and the 115-volt ac nonessential bus.

Note

Tacan will be inoperative with loss of either ac or dc generator.

Tacan Control Panel

The Tacan control panel (figure 4-7), located on the right-hand console, has a power switch with OFF, REC, and T/R positions, two channel selector knobs, a channel window and a volume-control knob. With the power switch in the REC position, the distance function of the set is disabled, and only bearing information is available. With the power switch in the T/R position, both bearing and distance information is displayed on the indicators. The left channel selector knob selects the first two figures of the Tacan beacon channel number, and the right channel selector knob selects the third number. The volume control knob is used to adjust the volume of aural identification signals received from the Tacan surface beacon.

Note

The Tacan surface beacon channels range from 00 to 129; however, the AN/ARN-21 is designed to operate only on channels 01 to 126.

Instrument Selector Panel

An instrument selector panel (figure 4-8) containing a two-position switch is located on the right-hand console. The switch controls the CDI of the course indicator. With the switch in TACAN position, the panel illuminates the area labeled TACAN and the CDI responds to AN/ARN-21 functions. With the switch in ILS position, the area surrounding ILS is illuminated, and the CDI responds to localizer signals used in conjunction with the AN/ARN-31 during an instrument landing system approach.

Radio Magnetic Indicator

The radio magnetic indicator (RMI) (11, figure 1-4) includes a rotating compass card and two pointers. The rotating card is actuated by the directional indicator (slaved) system. The pointers which are connected to function as a single unit, are actuated by the receiver portion of the AN/ARN-21. Azimuth signals from the Tacan surface beacon are received by the AN/ARN-21 and relayed to the RMI, causing the pointers to indicate the magnetic bearing of the Tacan surface beacon. With the control switch in the REC position, bearing information may be received even though the transmitter portion of the set is not energized. The set is so designed that when the correct bearing information cannot be determined or the equipment is not functioning satisfactorily, the indicator will "search" or rotate rapidly so that the pilot will be unable to derive unreliable information from the azimuth indicator.

Tacan Range Indicator and Altitude Switch Over-Ride

The Tacan range indicator panel (figure 4-9) is installed on the instrument panel and contains the range indicator and altitude switch over-ride. The range indicator displays the distance in nautical miles between the airplane and the Tacan surface beacon. The numerals in the window are controlled by the range circuits of the AN/ARN-21. These circuits measure the time elapsed between transmissions of the signal and the reception of the response signal from the Tacan surface beacon. The time difference is then converted into digital information, which is displayed on the range indicator in nautical miles. While the indicator is "searching" for the correct range or when the Tacan power switch is in REC position, the rotating numbers are partially covered by a red flag, which warns against reading incorrect distance indications. The altitude switch over-ride on the Tacan range indicator panel is provided to over-ride the automatic shutoff function of the altitude switch. The switch is placarded "Altitude SW Over-Ride" and has positions OFF and ON. Above 45,000 feet, when placed in the ON position, the altitude switch over-ride will restore operation of the Tacan equipment if the altitude switch has shut it off.

CAUTION

The altitude switch over-ride should not be used except in an emergency as arcing may occur in the Tacan equipment when operating above 45,000 feet.

Course Indicator

Signals received by the AN/ARN-21 from the surface beacon are relayed to the CDI (vertical pointer) of the course indicator (34, figure 1-4). Deviation of the airplane course either to the left or right of the transmitting beacon will be indicated by displacement of the CDI. For further information concerning course indicator, refer to "VOR Receiver—AN/ARN-14," this Section.

Operation of an AN/ARN-21

1. Power switch to either REC or T/R.
2. Instrument selector switch to TACAN.
3. Allow approximately 90 seconds warmup time after power is applied to nonessential buses. There is no delay when going from REC to T/R.
4. Select desired beacon channel.
5. Adjust volume to desired level.
6. To turn equipment off, position power switch to OFF.

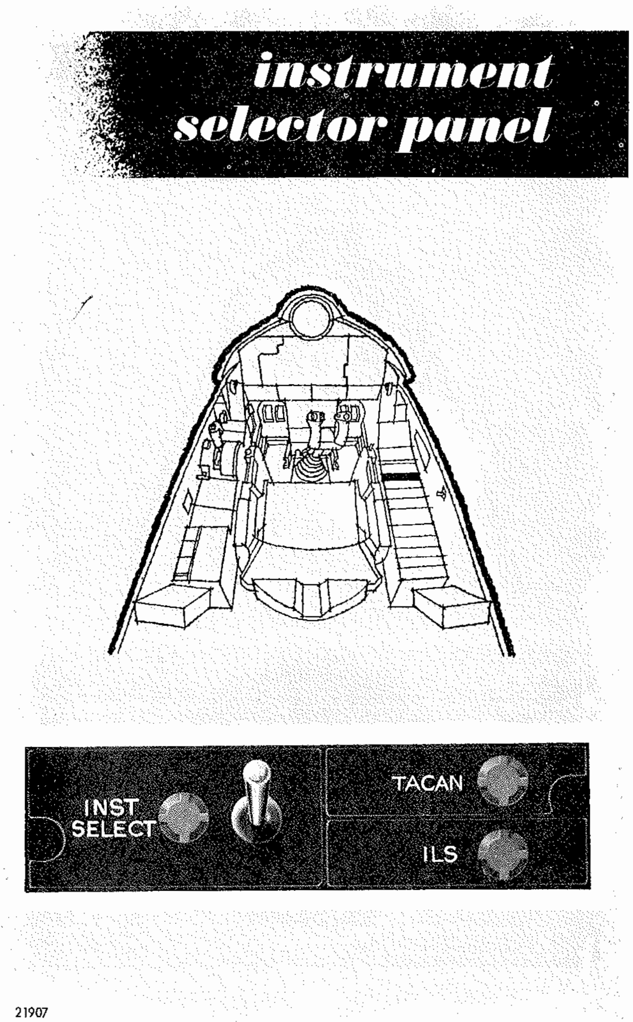
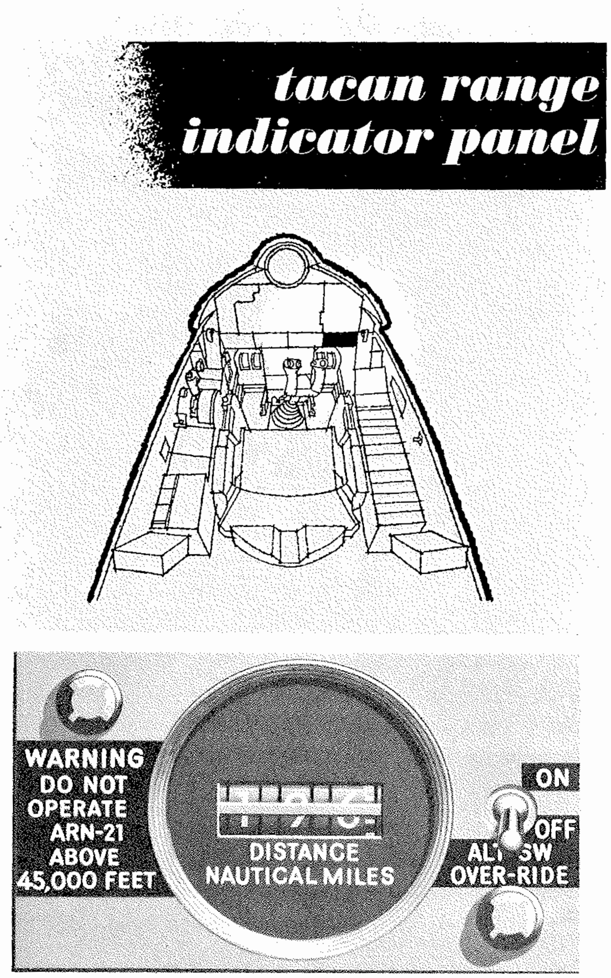


Figure 4-8

ILS RECEIVERS—AN/ARN-31*

This set consists of localizer and glide slope receivers, a control panel and an instrument selector panel. Indications from the AN/ARN-31 are displayed on the course indicator and give both vertical and horizontal guidance to a pilot making an instrument landing system approach. The localizer and glide slope operate 20 fixed frequency channels. The localizer frequencies range from 108.1 to 111.9 and the glide-slope from 329.3 to 335.0. The standard glide-slope frequency is automatically obtained when the desired localizer frequency is selected on the control panel. Two warning "OFF" flags are visible at the right end of the horizontal and at the lower end of the vertical bar on the course indicator whenever the signal level from the selected frequency is too weak to be reliable. The set is powered by the 28-volt dc nonessential bus and the 115-volt ac nonessential bus.

**AF 56-1242 thru -1316, -1359 thru -1368, -1393, 57-770 & on, & airplanes modified by TCTO 1F-102-719.*



21912

Figure 4-9

ILS Control Panel

The ILS control panel (figure 4-10) located on the right-hand console consists of a power switch, a frequency selector knob, a frequency window, and a volume control knob. Placing the power switch from OFF to POWER puts the set into operation. Turning the frequency selector knob selects any one of 20 localizer frequency channels and automatically obtains the desired glide-slope frequency. The localizer frequency selected is displayed in the window.

Instrument Selector Panel

A two-position switch (figure 4-8), on the right-hand console, controls relays that connect the course indicator to either the ILS receivers or the Tacan receiver. The switch must be in ILS position for operation of the AN/ARN-31 during an instrument landing approach. The instrument selector panel also contains a TACAN position which causes the vertical pointer (CDI) to respond to AN/ARN-21 functions. The instrument switch and relays receive power from the 115-volt ac nonessential bus.

WARNING

During an ILS approach, if disappearance of the glide-slope indicator warning "OFF" flag is delayed beyond normal expectation, it may be an indication that electrical power to the instrument selector relays is lost and the CDI information is being derived from a Tacan station. At the start of an ILS approach where the CDI warning "OFF" flag has disappeared but the glide-slope warning "OFF" flag is still visible, or at a locally prescribed check point, check to determine if there has been power loss to the relays by turning the bearing selector knob a few degrees away from the inbound heading. If the CDI responds to the bearing change, the signal was being received from a Tacan station and not the localizer. If the CDI does not respond to the inbound bearing change, the signal was being received from the localizer. After both warning "OFF" flags have disappeared, a subsequent power loss to the instrument selector relays will be detected by the horizontal needle warning "OFF" flag appearing and the horizontal needle will center itself and remain centered regardless of airplane movement.

Operation of AN/ARN-31

1. Place power switch in POWER position.
2. Place instrument selector switch in ILS position.
3. Select desired frequency.
4. Set is ready for operation when warning flags are no longer visible.
5. To turn set off, place power switch to OFF.

MARKER BEACON RECEIVER—AN/ARN-12 OR AN/ARN-32

Early airplanes* are equipped with AN/ARN-12 receiver and later airplanes** with AN/ARN-32 receiver. Both are fixed tuned receivers and perform the same navigational function. The marker beacon indicator light is on the course indicator. This equipment is in a standby condition at all times that dc power is supplied to the nonessential bus.

IDENTIFICATION RADAR (IFF-SIF)—AN/APX-6A, AN/APX-25

The AN/APX-6A, AN/APX-25 radar sets identify the airplane automatically according to prearranged modes

**AF 53-1791 thru 56-1274, -1317 thru -1331.*
***AF 56-1275 thru -1316, -1332 & on.*

of operation when challenged by suitably equipped ground or airborne units. Interrogation signals are decoded and coded signals sent in return. The identification radar sets also have means for transmitting a special distress signal. The AN/APX-25 identification radar can be set by ground technicians to decode challenges from either of two ground or airborne identification systems: the Mark X IFF, or the selective identification feature (SIF) systems. SIF provides a greater number of possible coded replies than does the Mark X system. When operating in SIF, in addition to the conventional IFF controls used with the Mark X IFF (AN/APX-6A radar installed or AN/APX-25 radar installed, but set for Mark X IFF), it will be necessary to use the coder group control panel placarded "SIF." The master switch on the IFF control panel receives power from 28-volt dc essential bus to operate a relay controlling power from the 115-volt ac essential bus which powers both radars.

IFF Control Panel

The IFF control panel (figure 4-11) is located on the right-hand console and has the IFF master control knob, a plunger button, and three toggle switches. The IFF master control knob is placarded "Master" and has positions EMERGENCY, NORM, LOW, STDBY, and OFF. Placing the IFF control knob to EMERGENCY selects the special distress signal. The OFF position deenergizes the identification radars. STDBY provides for set warmup with primary power on, but the receiver desensitized so that the set will not respond to interrogation signals. NORM position provides for normal set operation, while the LOW position provides a partial receiver sensitivity which allows the set to respond to only very strong interrogation signals. Selection is achieved by turning the IFF master control knob until the desired position appears below the selection arrow marker. To select EMERGENCY it is necessary to depress the plunger button in the lower left-hand corner of the control panel while turning the control knob. This button prevents accidental selection of the emergency position. Normal operating selections are called modes — mode 1, mode 2, and mode 3. Mode 1 is selected by placing the IFF master control knob in the NORM position. Selection of additional modes is accomplished by use of two mode selector toggle switches with positions MODE 2 — OUT, and MODE 3 — OUT.

Note

Combinations of mode selections available are: mode 1 and mode 2, mode 1 and mode 3 and modes 1, 2, and 3. Mode 1 is always used, alone or with any combination of modes.

The third toggle switch is provided to supply a special "identification of position feature." This toggle switch has three positions placarded I/P, OUT, and MIC. It is spring-loaded to OUT from the I/P position, and must be held in the I/P position. When placed in the

ils control panel

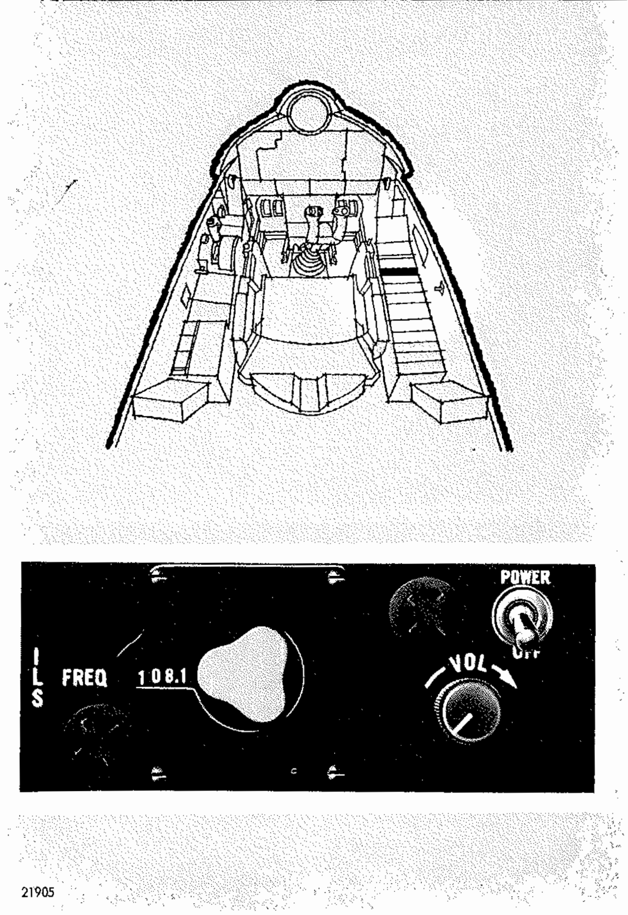


Figure 4-10

MIC position, the microphone button must also be held depressed. The identification radar features provided by the incorporation of this switch have been supplied to aid ground and airborne air-controllers with special position identification problems and should be used only as directed.

SIF Coder Control Panel

The SIF coder control panel (figure 4-11) is located immediately behind the IFF control panel, on the right-hand console, and is used in conjunction with the IFF control panel when the AN/APX-25 identification radar has been set to respond to SIF challenges. The selection of the mode of operation is done by positioning of the controls on the IFF control panel. The coder control panel is used to select the coded response to the selected mode 1, mode 3, or both. Coded SIF responses to a selected mode 2 are set on the ground and are provided automatically when mode 2 is selected on the IFF con-

iff control panel

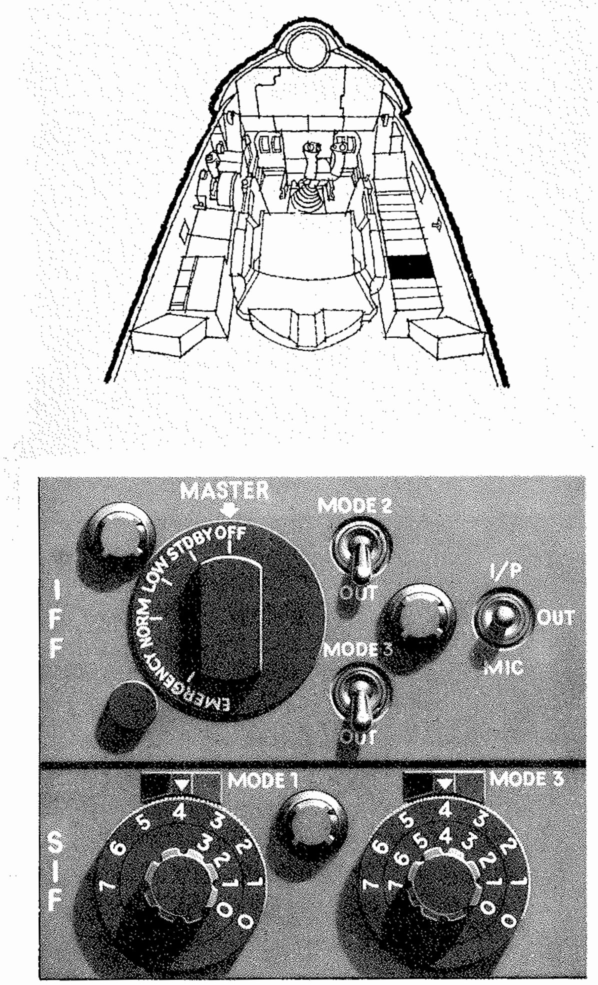


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Figure 4-11

trol panel if the AN/APX-25 has been set to respond to SIF interrogations. The coder control panel consists of two dual concentric selector knobs. The left-hand knob controls mode 1 SIF responses coding, the right-hand knob controls coded response to mode 3 SIF interrogations. Combinations of numbers on the inner and outer rings of each of the two selector knobs, aligned with the index markers, set the codes.

Operation of Identification Radar (IFF-SIF)

1. IFF master control knob — As desired.
2. Mode 2 and mode 3 selector switches – OUT, unless otherwise directed.

Note

Set I/P-OUT-MIC switch to OUT unless directed otherwise for identification of position.

3. SIF coder group controls — Set for proper codes as directed.
4. If in distress — Press dial stop plunger button and rotate master switch to EMERGENCY position. Set will then automatically transmit distress signals when interrogated.

LIGHTING EQUIPMENT

EXTERIOR LIGHTING

Exterior lighting* consists of 10 position lights, two landing lights and one taxi light. Three groups of position lights (fuselage, wing tip, and tail) are controlled by two switches in the cockpit, which provide for the selection of dim, bright, and flashing circuits. The landing and taxi lights are controlled by a single selector switch. A white light is located on each side of the upper fuselage above the wing root leading edge. A similar white light is located in the right and left armament bay flipper door, below the wing root leading edge. The white lights are not on a flasher circuit. One light is located in each wing tip. The right-hand light has green lenses flush with the upper and lower surfaces of the wing tip. The left-hand light has red lenses flush with upper and lower surfaces. The four tail lights, one white and one amber on each side, are located on the right and left sides of the tail cone fairings, above the inboard trailing edges of the elevons. The upper light on each side is amber and the lower one is white. The wing tip and tail lights are on an automatically cycled flasher circuit. One landing light is mounted on the inner side of each main landing gear fairing door, so that the lights are in duty position when the main landing gear is extended and the doors are fully opened. The taxi light is mounted on the inner side of the nose landing gear fairing door. Neither landing lights nor taxi light can be turned on if the nose landing gear is not down and locked.

Position Light Switches

Two position light switches* (figure 1-21) are located at the bottom right corner of the utility switch panel. All lights are turned on by selecting either STEADY or FLASH position with the left switch, and either BRIGHT or DIM position with the right switch (both must be used, as they are in series). The FLASH position energizes two wing and tail light circuits alternately at the rate of approximately 40 cycles per minute. During one cycle, each circuit is energized and deenergized once. One circuit consists of left and right white tail lights and left (red) and right (green) wing tip lights. The other cir-

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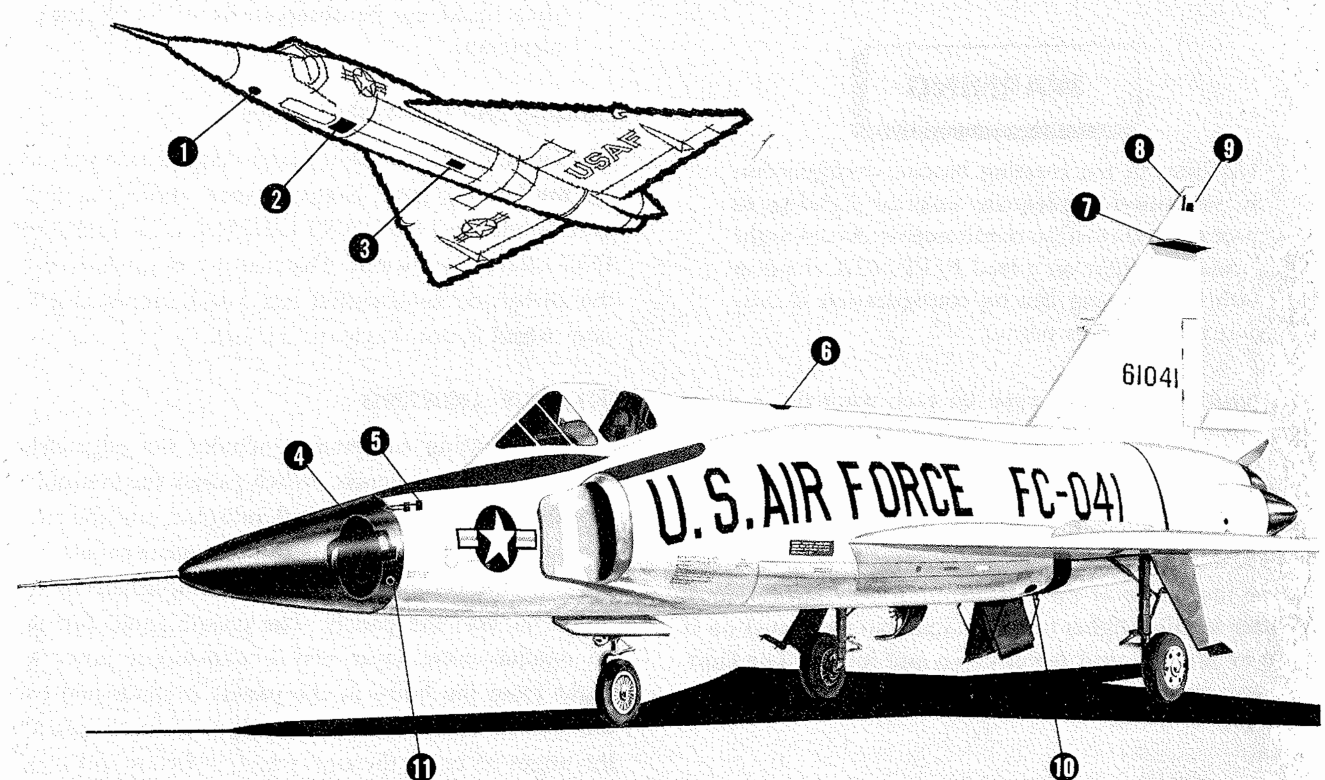
*AF 53-1791 thru 56-1429.

antenna locations



1. Lower IFF Antenna. (AN/APX-6A or AN/APX-25.)
2. Data Link Antenna.
3. Marker Beacon Antenna. (AN/ARN-12 or AN/ARN-32.)
4. Radar Antenna. (MG-10.)
5. Glide Slope Antenna. (AN/ARN-18 or AN/ARN-31.)
6. Upper IFF Antenna. (AN/APX-6A or AN/APX-25.)
7. VHF Antenna (AN/ARN-14) or Localizer of AN/ARN-31.
8. UHF Command Radio Antenna. (AN/ARC-34.)
9. Air-To-Air FIS (AN/APX-27) (PROVISIONS ONLY)
10. Tacan Antenna (AN/ARN-21).
11. Air-To-Air FIS (AN/APX-26) (PROVISIONS ONLY)

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Figure 4-12

cuit consists of left and right amber tail lights. If the flasher fails, the circuit consisting of the white tail lights and the wing tip lights will be energized and will remain on steady. The flasher switch controls the 115-volt single-phase ac essential bus power to all the lights, and also the 28-volt dc essential bus power to the flasher relays and controller. The dimmer switch controls only the 115-volt ac power to the lights.

EXTERIOR LIGHTING

Two retractable anticollision lights on the fuselage, two wing tip lights, two landing lights and one taxi light provide exterior lighting* for both inflight and ground operation. One of the red anticollision lights is located on the upper centerline of the fuselage aft of the cockpit and the other on the lower centerline of the fuselage aft of the main wheel well. The anticollision lights extend and illuminate whenever the position lights switch is in NAVIGATION position. They retract and turn off whenever the position lights switch is in FORMATION position or automatically when 335 KIAS is exceeded. On some airplanes**, to provide for more effective utilization of the rotating beacon lights as anticollision lights, the automatic extension and retraction feature has

*AF 53-1801, -1802, -1809, 56-987 and 56-1430 through 57-909.
**Airplanes modified by TCTO 1F-102-890.

been eliminated. The lights will extend in the NAVIGATION position or retract in the FORMATION positions respectively. With the anticollision lights retracted, a white light portion illuminates and provides lighting on both the upper and lower sides of the fuselage.

WARNING

Airplanes in the rotating beacon configuration do not provide adequate exterior lighting to meet night formation requirements. Avoid night formation flight on a lead F/TF-102A airplane with the rotating beacon configuration if mission requirements permit.

One position light is located on each wing tip, a green light on the right side and a red light on the left side. One landing light is mounted on the inner side of each main landing gear fairing door, so that the lights are in duty position when the main landing gear is extended and the doors are fully opened. The taxi light is mounted on the inner side of the nose landing gear fairing door. Neither landing lights nor taxi light can be turned on if the nose landing gear is not down and locked. The landing and taxi lights are controlled by a single selector switch.

Position Light Switches

Two position light switches* (figure 1-21) are located at the bottom right corner of the utility switch panel. The left switch has three positions: NAVIGATION, OFF, and FORMATION. In NAVIGATION position the red anticollision lights are turned on and extended, and the wing tip lights illuminate steadily. The right BRIGHT-DIM switch has no effect on the lights with the power switch in NAVIGATION position. In FORMATION position the anticollision lights are turned off and retracted, the wing tip lights illuminate steadily, and the white portion of the anticollision lights illuminate at the top and bottom of the fuselage. The lights may be dimmed in this position. There is no flasher position provided. The NAVIGATION position controls 28-volt dc nonessential bus power to the anticollision lights and the 115-volt ac essential bus power to the transformer for wing-tip lights. The FORMATION position controls 115-volt ac essential bus power to the transformer for wing tip lights and white fuselage lights.

Note

The rotating anticollision lights should be turned OFF during flight through conditions of reduced visibility where the pilot could experience vertigo as a result of the rotating reflections of the lights against the clouds. In addition, the lights would be ineffective as anticollision lights during these conditions since they could not be observed by pilots of other airplanes.

Landing and Taxi Light Switch

The landing and taxi light switch (figure 1-24) is located on the landing gear control panel, above the landing gear handle and has TAXI LIGHTS, OFF, and LANDING LIGHTS positions. The switch controls power from the 28-volt dc nonessential bus, but is inoperative if the nose wheel is not down and locked.

INTERIOR LIGHTING

Interior lighting equipment includes the edge-lighting for the instrument panels, switch panels and consoles, the thunderstorm lights, cockpit floodlights, and the standby compass light. Edge-lighting of instrument panels, switch panels, and console panels is accomplished by small lights set into the 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 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 lighting on the pilot's vision. Intensity of lighting in ac circuits is controlled by variable transformers, known as "powerstats." Intensity of dc lighting is controlled by a rheostat. Powerstat and rheostat knobs are identical in appearance.

Flight Instrument Panel Lights Rheostat

The flight instrument panel rheostat (33, figure 1-4) is located at the top center of the lighting control panel. Turning the knob clockwise from OFF to BRT controls the intensity of the fifteen panel lights that illuminate the flight instrument group in the center of the instrument panel, and the electrical control panel on some airplanes and also the standby compass light, when the compass light switch is in the ON position. Power to this rheostat comes from the 28-volt dc essential bus.

AF 56-1430 & on.

Forward Panel Lights Powerstat

A powerstat placarded "Fwd Panels" at the top right corner of the lighting control panel (33, figure 1-4) controls the intensity of the lighting for all of the engine instrument group on the instrument panel, and also for the armament control panel, the antenna scan control panel, the target altitude panel, the two small radar scope control panels at either side of the radar scope, the far left side of the instrument panel, the utility switch panel, and the lighting control panel. Turning the knob clockwise from OFF to BRT controls the intensity of the lighting on these panels. Power to this powerstat comes from the 115-volt ac essential bus.

Console Lights Powerstat

The console lights powerstat, located at the lower right corner of the lighting control panel (33, figure 1-4), is placarded "Console." Turning the knob clockwise from OFF to BRT controls the intensity of the lights on both consoles, and also the lights on the upright panels at the forward ends of the consoles. Power to this powerstat comes from the 115-volt ac essential bus.

Cockpit Floodlights Powerstats

Two floodlight powerstats are located at the left side of the lighting control panel (33, figure 1-4). One placarded "Inst Flood" controls intensity of the four red floodlights in two reflection shields aft of the main instrument panel. The other, placarded "Console Flood" controls the intensity of red floodlights above each console. Power to these powerstats comes from the 115-volt ac essential bus.

Thunderstorm Lights Switch

A switch placarded "Storm Lights" (figure 1-4) above the canopy unlocked warning light, at the extreme right end of the instrument panel, has ON and OFF positions and controls five white thunderstorm lights, located over the two consoles and aft of the instrument panel. When the switch is in ON position, the master warning dimming relay is cut out, causing the master warning light, master warning light panel, landing gear unsafe warning light, canopy unlocked warning light, hydraulic pressure low warning light, fire and overheat warning light, and takeoff trim light to be at full brilliance while the thunderstorm lights are on. On some airplanes* the canopy unlocked warning light is removed from the warning light dimming circuit and will appear at full brilliance any time the canopy is not locked. The thunderstorm light switch controls power from the 28-volt dc nonessential bus.

**Airplanes modified by TCTO 1F-102-773.*

LIQUID OXYGEN SYSTEM

The major components of the liquid oxygen system are a storage and converter unit, an oxygen pressure gage, a liquid oxygen content gage, an external filler valve, and an oxygen regulator. The storage and converter unit includes a five-liter (one liter equals about one quart) insulated storage container and a converter, which converts the liquid oxygen to gaseous oxygen and then supplies it to the oxygen regulator. The liquid oxygen content gage indicates, in liters, the supply of liquid oxygen in the storage container. The oxygen regulator is located on the left console on some airplanes and in the seat cushion-survival kit on other airplanes*. Gaseous oxygen is delivered from the converter to the regulator at a fairly constant pressure of about 70 psi, and the regulator, in turn, supplies the pilot with breathing oxygen. After a 24-hour boil-off period, the oxygen system supply allows for a maximum of 22.6 hours at a minimum demand and a minimum of 3.7 hours at a maximum demand. Oxygen duration at various altitudes is shown in figure 4-13. At sea level and with average temperature, a full supply of liquid oxygen dissipates through a relief valve in about five days, when the airplane remains on the ground and no demands are made on the system. 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.

LIQUID OXYGEN REGULATOR

A combination pressure-breathing, diluter-demand MD-1 oxygen regulator (figure 4-16) is mounted on the left console. Gaseous oxygen is supplied to the regulator at approximately 70 psi. The regulator reduces the oxygen pressure, mixes air with oxygen in varying amounts depending on altitude, temperature and pilot demand, and delivers it through a flexible tube to the pilot's mask or the K-1 helmet. At high altitudes, the regulator supplies positive-pressure breathing. Control of the oxygen system is accomplished by the use of three levers: the supply lever, the diluter lever, and the emergency toggle lever. The pilot receives an indication of system operation from the flow indicator and oxygen pressure gage located on the oxygen regulator panel.

Diluter Lever

The diluter lever, aft center on the regulator panel, has two positions, NORMAL and 100%. With the lever at NORMAL, the regulator mixes air with oxygen in varying amounts, according to altitude, and delivers it through a flexible tube to the pilot's mask or helmet. With the lever at 100%, the regulator delivers 100% oxygen regardless of altitude.

**AF 56-1275 thru -1316, -1332 & on, & airplanes modified by TCTO 1F-102-642.*

liquid oxygen duration chart-hours

USING PRESSURE BREATHING OXYGEN MASK TYPE MS22001 (A-13A)

CABIN ALTITUDE—FEET	5	4	3	2	1	BELOW 1
35,000 AND ABOVE	31.4	25.2	18.9	12.6	6.3	EMERGENCY DESCEND TO ALTITUDE NOT REQUIRING OXYGEN
	31.4	25.2	18.9	12.6	6.3	
30,000	22.7	18.1	13.6	9.0	4.5	
	23.3	18.7	14.0	9.3	4.7	
25,000	17.5	14.0	10.5	7.0	3.5	
	22.0	17.6	13.2	8.8	4.4	
20,000	13.3	10.7	8.0	5.3	2.7	
	25.0	20.0	15.0	10.0	5.0	
15,000	10.7	8.6	6.4	4.3	2.2	
	30.2	24.2	18.1	12.1	6.0	
10,000	8.6	6.9	5.2	3.4	1.7	
	30.2	24.2	18.1	12.1	6.0	
5,000	6.8	5.4	4.1	2.7	1.4	
	30.2	24.2	18.1	12.1	6.0	
SL	5.5	4.4	3.3	2.2	1.1	
	30.2	24.2	18.1	12.1	6.0	

LIQUID CONTENT—LITERS

- FIGURES ON GRAY INDICATE DILUTER LEVER—NORMAL OXYGEN
- FIGURES ON WHITE INDICATE DILUTER LEVER—100% OXYGEN
- ON AIRPLANES NOT EQUIPPED WITH SURVIVAL KIT, EMERGENCY TOGGLE SET AT EMERGENCY.

USING HIGH ALTITUDE PRESSURE COVERALLS MC-3 AND PRESSURE HELMET TYPE MA-2

CABIN ALTITUDE—FEET	5	4	3	2	1	BELOW 1
30,000 AND ABOVE	12.8	10.2	7.7	5.1	2.6	EMERGENCY DESCEND TO ALTITUDE NOT REQUIRING OXYGEN
25,000	9.8	7.8	5.9	3.9	2.0	
20,000	7.6	6.1	4.6	3.0	1.5	
15,000	6.0	4.8	3.6	2.4	1.2	
10,000	4.8	3.8	2.9	1.9	1.0	
5,000	3.9	3.1	2.3	1.6	0.8	
SL	3.2	2.6	1.9	1.3	0.7	

LIQUID CONTENT—LITERS

21898

Figure 4-13

Emergency Toggle Lever

The emergency toggle lever, aft left on the regulator panel, has two placarded positions, EMERGENCY and TEST, and an unmarked center (neutral) position. The toggle lever should remain in the neutral position at all times, unless an unscheduled pressure increase is required. Moving the toggle lever to EMERGENCY provides continuous positive pressure to the mask for emergency use. When the toggle lever is moved to TEST, it provides positive pressure to test the mask for leaks.

CAUTION

When positive pressures are required, it is mandatory that the oxygen mask be well fitted to the face. Unless special precautions are taken to insure against leakage, continued use of positive pressure under these conditions will result in the rapid depletion of the oxygen supply and extremely cold oxygen flowing to the mask.

Supply Lever

The supply lever, aft right on the regulator panel, is safety-wired to the ON position. This lever has ON-OFF positions and controls oxygen pressure to the regulator accordingly.

Note

If the safety wire is broken there should not be concern as long as the lever is at ON.

Pressure Gage and Flow Indicator

A pressure gage and a flow indicator are located on the oxygen regulator panel. The pressure gage shows gaseous oxygen supply pressure (pressure being furnished to the oxygen regulator inlet). The flow indicator consists of an oblong opening on the face of the regulator panel and shows black and white alternately during the breathing cycle.

Liquid Oxygen Content Gage

A liquid oxygen content gage is located just aft of the oxygen regulator panel. The gage provides an indication of the content of the storage container and is calibrated in liters from 0 to 5.

Note

The liquid oxygen content gage should read between 4 and 4½ liters when the system is fully charged. Do not be alarmed that the gage does not read five liters, since it is impossible to charge the liquid oxygen converter to five liters.

Liquid Oxygen System Preflight Check

Before takeoff the oxygen system should be checked as follows:

Note

This test procedure is applicable only for an initial preflight check of the system. Inflight tests or repeated tests made within short periods may produce false or misleading indications.

1. Attach oxygen hose as outlined in figure 4-15.
2. Check oxygen pressure gage at 70 to 110 psi.
3. Check liquid content gage at three liters minimum with partial pressure suit or two liters minimum without partial pressure suit.
4. Check supply lever ON.
5. Check oxygen regulator with the diluter valve first at NORMAL and then at 100% as follows: Remove mask and blow gently into the end of the regulator hose as during normal exhalation. There should be resistance to blowing. Little or no resistance to blowing indicates a leak or faulty operation.
6. With the oxygen mask connected to the regulator and the diluter lever at 100%, breathe normally into the mask and conduct the following checks:
   a. Observe flow indicator for proper operation.
   b. place emergency lever to EMERGENCY. A positive pressure should be supplied to the mask. Return emergency lever to center (neutral) position.
   c. Hold emergency lever in TEST position. A positive pressure should result within the mask. Hold breath to determine whether there is leakage around mask. Release emergency lever; positive pressure should cease.
7. Retain diluter lever in 100% position or return diluter lever to NORMAL as required.

Normal Operation of Liquid Oxygen System

1. Before each flight, be sure oxygen pressure gage reads at least 70 psi and liquid content gage shows a minimum of three liters with partial pressure suit or two liters without partial pressure suit. If content is below this minimum, have oxygen system serviced before takeoff.
2. See that oxygen supply lever is safety-wired in ON position.
3. See that diluter lever is at 100% or NORMAL position as required.

Note

- Above 30,000 feet, a vibration or wheezing sound may sometimes be noted in the mask. This noise is a normal characteristic of regulator operation and may be overlooked.
- Steady deep breathing will cause the quantity gage indicator to momentarily drop toward zero. This is normal and does not mean the oxygen supply is depleted.

Emergency Operation of Liquid Oxygen System

WARNING

A partial pressure suit should be worn for all flights above 45,000 feet ambient altitude.

If symptoms of hypoxia develop or if smoke or fumes enter the cockpit, proceed as follows:

1. If operating on NORMAL, move diluter lever to 100% position.
2. Push emergency lever forward to EMERGENCY position.
3. If oxygen regulator becomes inoperative, actuate emergency oxygen bailout bottle (which contains about a six-minute oxygen supply). Descend to a cockpit altitude below 10,000 feet as soon as possible.

LIQUID 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 to the regulator during normal operation from the oxygen converter at approximately 70 psi or during emergency operation from the bailout oxygen supply in the survival kit (refer to SURVIVAL KIT Section I). Bailout bottle pressure is approximately 1800 psi when fully charged and it is therefore necessary to reduce this pressure by means of a restrictor prior to delivery to the regulator. Two control units in the regulator, one for partial pressure suit capstan pressure and one for breathing and chest bladder pressure, regulate and deliver 100% oxygen to the pilot's mask or K-1 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 the oxygen with air) and therefore delivers 100% oxygen at all times. When using an oxygen mask instead of the partial pressure suit and helmet, it is necessary to use an adapter to reduce the amount of oxygen pressure between the regulator and the mask. The oxygen system is controlled by a single switch on the oxygen control panel.

Liquid Oxygen Control Panel

The liquid oxygen control panel* is located on the left-hand console and is placarded "Oxygen Supply." The panel contains a supply switch, a liquid oxygen content gage, and a pressure gage. The supply switch has ON-OFF positions and controls the flow of oxygen from the airplane supply to the oxygen regulator. The liquid oxygen content gage provides an indication of the content of the storage container and is calibrated in liters from 0 to 5.

Note

The liquid oxygen content gage should read between 4 and 4½ liters when the system is fully charged. Do not be alarmed that the gage does not read five liters as it is impossible to charge the liquid oxygen converter to five liters.

The pressure gage shows gaseous oxygen pressure from 0 to 500 psi. When oxygen is being used from the system, the pressure 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.

Press-to-Test Button

The oxygen system press-to-test button* is located on the forward edge of the survival kit. Depressing the button will provide positive pressure to the oxygen mask or pressure helmet to determine that the system is operating satisfactorily prior to takeoff. The press-to-test button is the only method of checking oxygen (other than decrease of liquid content and positive flow of oxygen through the system) as there is no blinker or flow indicator installed.

Oxygen Mask Adapter

An adapter* is provided with the survival kit and is to be used when the A-13 type oxygen mask is worn in lieu of the partial pressure helmet. The adapter consists of two separate units. One is an electrical plug which is to be plugged into the mask defog and communications leads in the personal equipment lead bundle on the survival kit. This unit will provide for a satisfactory receptacle for communications lead from the protective helmet and oxygen mask. The second unit is an adapter for the oxygen lead. This adapter provides a quick-disconnect connection for the oxygen mask hose and also serves as a pressure restrictor to reduce the oxygen pressure from the regulator to the mask.

Liquid Oxygen System Preflight Check

Before takeoff the oxygen system should be checked* as follows:

**AF 56-1275 thru -1316, -1332 & on, & airplanes modified by TCTO 1F-602-642.*

1. Before entering cockpit, check bailout bottle pressure gage (located in survival kit) at 1800 psi.
2. Connect oxygen mask adapter to end of oxygen hose in personal equipment lead bundle if oxygen mask is to be worn instead of partial pressure helmet.
3. Attach oxygen hose as outlined in figure 4-15.
4. Check oxygen pressure gage at 70 to 110 psi.
5. Check liquid content gage at three liters minimum with partial pressure suit or two liters minimum without partial pressure suit.
6. Check supply switch ON.
7. With the oxygen mask connected breathe normally into the mask. As a slight positive pressure is supplied at all times there should be no resistance to breathing.
8. If a partial pressure suit is worn, depress the press-to-test button on the front of the survival kit. A definite positive pressure should result within the face plate. Hold breath to determine whether there is leakage around face plate. Release the press-to-test button. Check for normal breathing.
9. If an A-13A oxygen mask is worn, depress the press-to-test button. Release button as soon as a buildup is felt.

Normal Operation of Liquid Oxygen System*

1. Perform oxygen system preflight check as outlined above prior to each flight. After completion of preflight check, oxygen system is ready for normal usage.
2. Oxygen supply switch ON after fastening mask or faceplate.
3. After landing when oxygen is no longer desired, turn oxygen supply switch OFF before opening mask or faceplate.

WARNING

The oxygen supply switch should neither be turned ON until immediately after the mask or faceplate is closed nor turned OFF until immediately before opening of the mask or faceplate. This will prevent a rapid depletion of the oxygen supply and also prevent low temperatures from damaging various components of the liquid oxygen system and causing personal injury.

liquid oxygen control panel

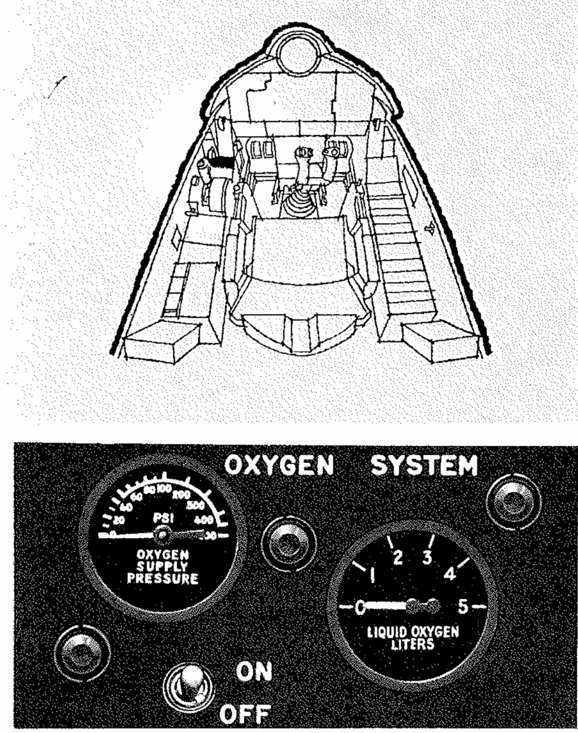


Figure 4-14

Emergency Operation of Liquid Oxygen System*

1. If symptoms of hypoxia develop, check oxygen hose connections and check supply switch ON.
2. If the ship's oxygen system is depleted or not supplying oxygen, actuate the bailout supply by pulling the green knob in the personal equipment lead bundle. Descend to a cockpit altitude below 10,000 feet within 12 minutes.

WARNING

Remove oxygen mask or faceplate when emergency oxygen supply is depleted.

**AF 56-1275 thru -1316, -1332 & on, & airplanes modified by TCTO 1F-102-642.*

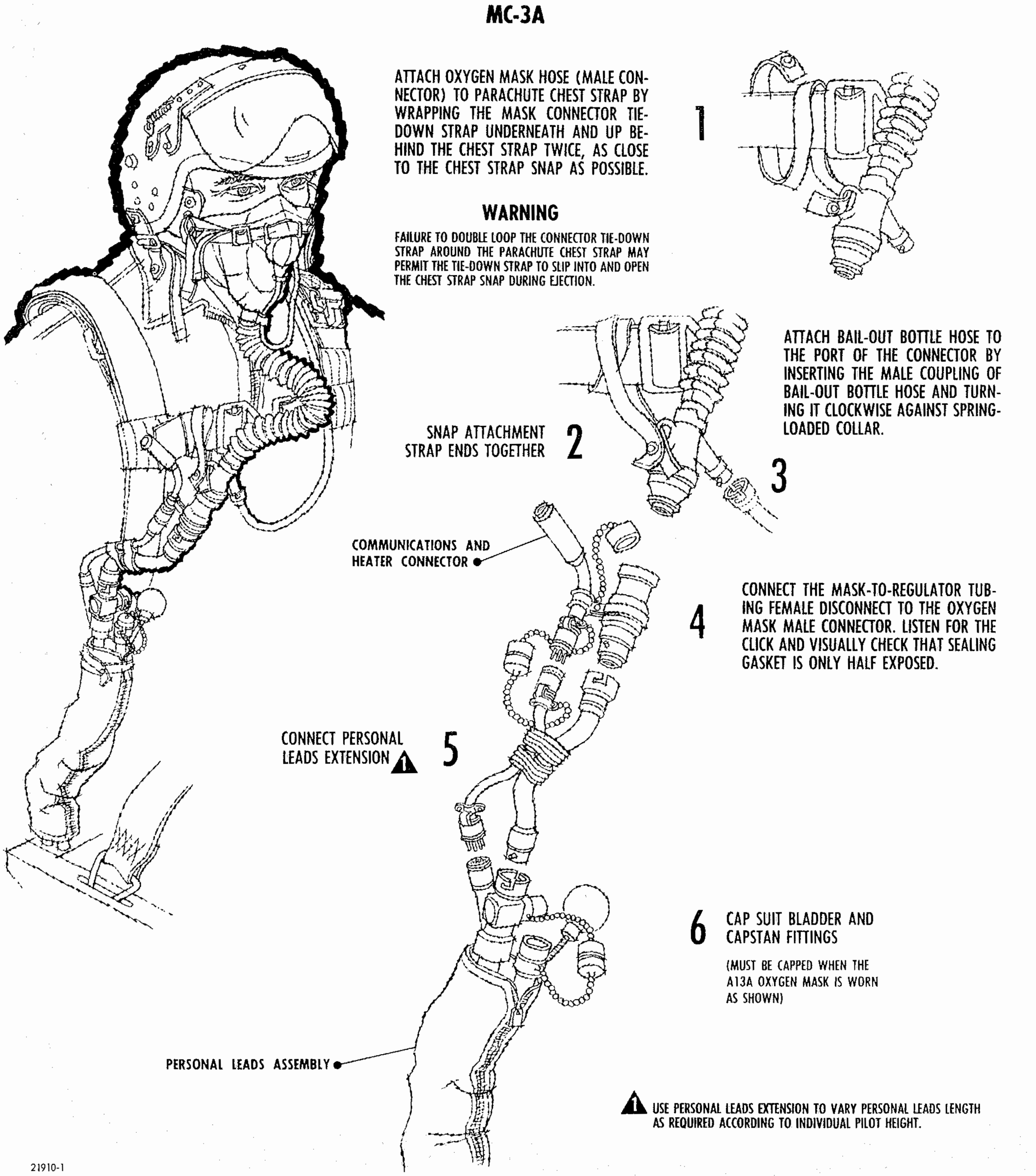
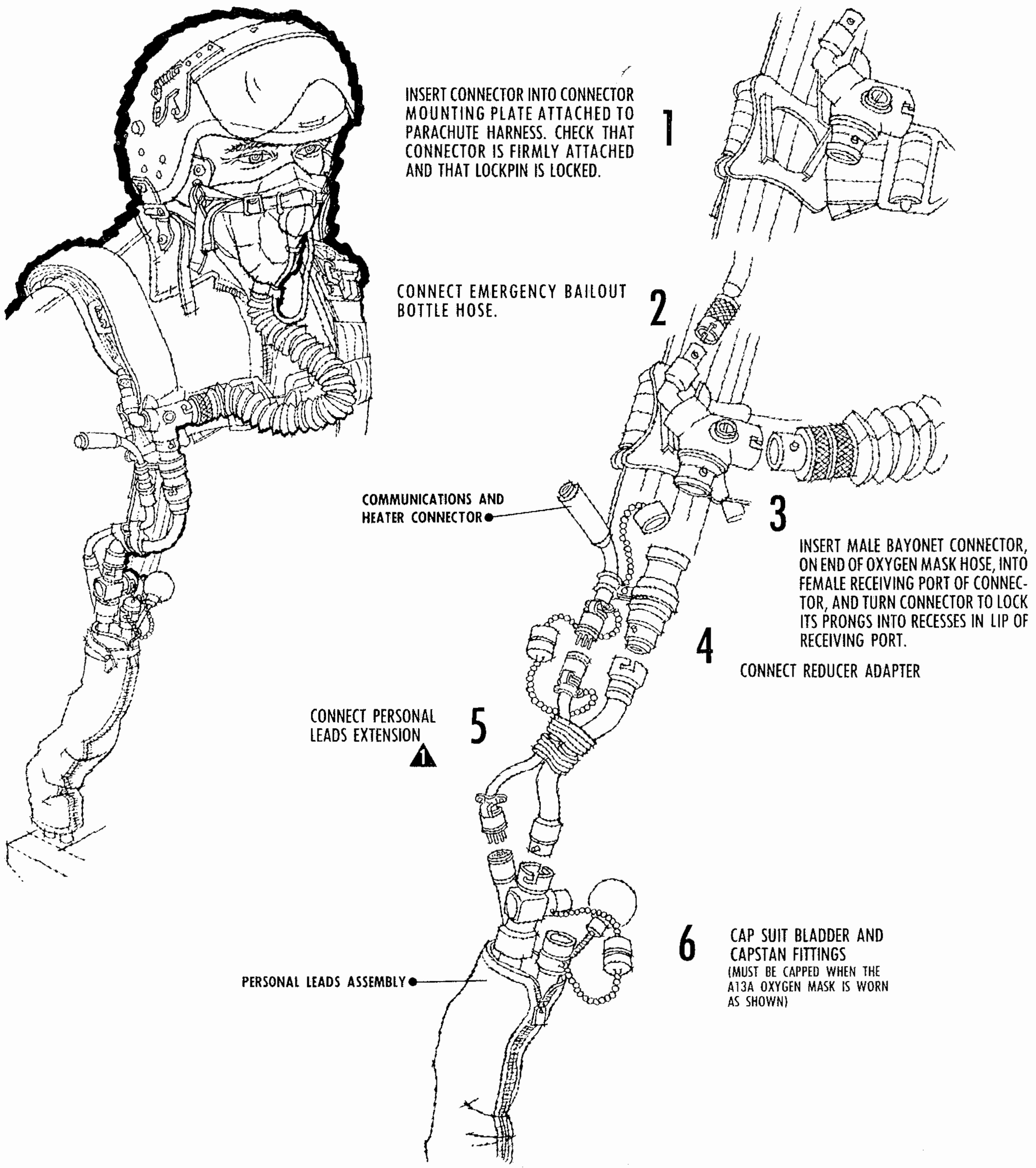

MC-3A
ATTACH OXYGEN MASK HOSE (MALE CONNECTOR) TO PARACHUTE CHEST STRAP BY WRAPPING THE MASK CONNECTOR TIE-DOWN STRAP UNDERNEATH AND UP BEHIND THE CHEST STRAP TWICE, AS CLOSE TO THE CHEST STRAP SNAP AS POSSIBLE.
1
WARNING
FAILURE TO DOUBLE LOOP THE CONNECTOR TIE-DOWN STRAP AROUND THE PARACHUTE CHEST STRAP MAY PERMIT THE TIE-DOWN STRAP TO SLIP INTO AND OPEN THE CHEST STRAP SNAP DURING EJECTION.
SNAP ATTACHMENT STRAP ENDS TOGETHER
2
ATTACH BAIL-OUT BOTTLE HOSE TO THE PORT OF THE CONNECTOR BY INSERTING THE MALE COUPLING OF BAIL-OUT BOTTLE HOSE AND TURNING IT CLOCKWISE AGAINST SPRING-LOADED COLLAR.
3
COMMUNICATIONS AND HEATER CONNECTOR
CONNECT THE MASK-TO-REGULATOR TUBING FEMALE DISCONNECT TO THE OXYGEN MASK MALE CONNECTOR. LISTEN FOR THE CLICK AND VISUALLY CHECK THAT SEALING GASKET IS ONLY HALF EXPOSED.
4
CONNECT PERSONAL LEADS EXTENSION 1
5
6
CAP SUIT BLADDER AND CAPSTAN FITTINGS
(MUST BE CAPPED WHEN THE A13A OXYGEN MASK IS WORN AS SHOWN)
PERSONAL LEADS ASSEMBLY
1 USE PERSONAL LEADS EXTENSION TO VARY PERSONAL LEADS LENGTH AS REQUIRED ACCORDING TO INDIVIDUAL PILOT HEIGHT.
21910-1

Figure 4-15

oxygen mask connection

CRU-8/P



21910-2

3. If wearing an A-13A oxygen mask and cabin pressure is lost, an immediate descent to 30,000 feet or below is mandatory.

WARNING

A partial pressure suit should be worn for all flights above 45,000 feet ambient altitude.

AUTOMATIC FLIGHT CONTROL SYSTEM

The automatic flight control system (AFCS) consisting of analog computing equipment which, when coupled with the airplane damping system and the MG-10A fire control system, provides automatic flight control of the airplane in three modes: pilot assist, attack, and landing approach. Command signals from the AFCS, combined with pitch-rate, roll-rate, yaw-rate, and effective elevon position signals deflect the control surfaces, through hydraulic actuators, to steer the desired course. The airplane damping system must be engaged and the MG-10A master switch must be in STBY or ON before automatic flight is possible. During all automatic modes of operation, the control stick follows the motion of the control surfaces. To prevent application of excessive aileron, limit switches are incorporated to disengage the AFCS switch and the pitch damper switch when aileron travel exceeds 2.5°. Pitch g limiting is provided to prevent airplane acceleration limits from being exceeded and to prevent subjecting the pilot to an uncomfortable number of g's in all modes of operation. Refer to Section V for AFCS limitations.

PITCH G LIMITER

Airplanes having AFCS are equipped with a pitch g limiter which prevents the automatic flight control system from subjecting the airplane to excessive pitch g forces. The preset limits are 4½ g's positive and 1½ g's negative or a positive pitch rate of 45° per second and negative pitch rate of 15° per second. If the preset limits are reached, the pitch g limiter removes electrical power from the solenoid-held switches of the automatic flight control system (all modes) and the pitch damper circuit. The yaw damper remains engaged to aid the pilot in maintaining coordinated flight after the automatic flight control system is deactivated. A preflight check of the limiter should be made prior to all flights by means of a test switch on the utility switch panel. This check may also be performed in flight. Power to the pitch g limiter originates from the 115-volt ac essential bus.

PILOT ASSIST MODE

The pilot assist mode relieves the pilot of routine steering tasks by performing conventional autopilot functions such as: (1) altitude or pitch attitude hold, and (2) heading or bank attitude hold. Primary inputs to AFCS in this mode are from the MG-10A vertical gyro, the airplane directional indicator, and ambient air pressure signals. The pilot may select whether he wishes to hold pitch attitude or altitude by positioning a switch on the AFCS control panel. Pitch attitude will be maintained with this switch in the OFF position. Pitch attitude is held by mixing signals from the vertical gyro and a pitch memory servo which registers the attitude at the time of engaging. If altitude hold is selected, a barometric altitude control unit is engaged. This unit produces an error signal proportional to ambient air pressure changes which is used to hold the altitude as it was when the unit was engaged. Altitude hold function utilizes the airplane static pressure sensing system, and, accordingly, the static pressure sensing errors will be reflected in the form of pitch changes while operating in the transonic region. After passing through the transonic zone, pitch changes will continue as altitude hold seeks the altitude held at time of engagement. Heading hold function may be selected by means of a switch on the AFCS panel. If the airplane's wings are within 5° of level flight when AFCS and heading hold are engaged, signals from the airplane compass and heading memory servo maintain the heading. If the bank angle exceeds 5° and heading hold is engaged, or any time the heading hold is off, the vertical gyro and the bank memory servo combine to produce an error signal to hold the bank attitude. The heading hold switch must be engaged at any time heading hold is desired. Heading hold errors will be evident as a result of airplane compass (J-4) precession. Large changes of altitude, pitch attitude, bank attitude, or heading may be made manually by momentary release of AFCS by using a momentary interrupt trigger on the control stick. However, if altitude hold has been selected, depression of the trigger will return this switch to OFF. Small changes of altitude, and bank attitude (if prevailing bank attitude exceeds 5°) may be made by using the elevon trim button to "beep" in signals to the AFCS. The heading hold switch must be OFF to maintain bank angles of less than 5°. 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 engaged, the airplane will return to level flight attitude. If altitude hold has been selected, trimming in pitch will return this switch to OFF. When the trim button is used, the pitch and roll references will change at a constant rate as long as the button is displaced. With AFCS engaged, the pilot assist mode will function until the radar is locked on or the AILAS is engaged. When engaging AFCS, if the system does not function properly, monitor circuits prevent engaging and return the switch to OFF.

ATTACK MODE

The AFCS attack mode automatically steers the airplane on the computed lead collision course for rocket or missile firing. With AFCS in operation, the radar attack is performed automatically after the desired armament is selected and the radar is locked on. The automatic run continues until armament firing occurs unless AFCS is manually disengaged by the pilot or a collision warning signal automatically disengages the attack mode. Loss of lock-on or appearance of the pullout signal on the radar scope returns AFCS to pilot assist (attitude hold) mode. The preselected armament will fire automatically during all automatic attack runs. During automatic operation, electronic limiting provides g limits of +3.3 g to +5. g and limits the command voltage to a value within these g limits. In the attack mode, the primary inputs to AFCS are the same as those to the steering dot on the pilot's scope. AFCS translates the azimuth and elevation steering signals into requests for control surface movement. Control surfaces are deflected to steer the airplane toward the correct lead collision course at a rate proportional to the off-course error signal. Thus, in the case of large steering error, the airplane turns rapidly initially, then as the error decreases, the rate of turn decreases proportionally. During the attack phase, the elevon trim button is inoperative and cannot be used to beep or feed trim signals to vary the airplane attitude.

AUTOMATIC INSTRUMENT LANDING APPROACH SYSTEM (AILAS)

In this mode, the airplane is automatically flown throughout an ILS approach. Throttle control, landing gear actuation, and flareout and landing are manually accomplished by the pilot. Throughout the approach, the course indicator should be monitored for proper operation of the system. A light in the AILAS pushbutton switch illuminates to indicate engagement. The automatic approach consists of two phases: constant altitude and glide-slope. The constant altitude phase extends from AILAS engagement to glide-slope entry. The glide-slope phase begins with glide-slope entry and extends to termination of the automatic approach. Preparatory to initial AILAS engagement, the pilot must set the proper runway heading into the ILS course indicator. This 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 on any heading in the localizer engage area which is a circle of four miles diameter centered 12 miles from the runway. However, an entry of 45° or less to the localizer heading is recommended to insure early localizer capture and minimum overshoot. Altitude at engagement should be 1500 feet above runway altitude as AILAS steering sensitivity during glide-slope descent is reduced as a function of sensed barometric pressure increase equivalent to 1500 feet. Prior to glide-slope entry, bank angle is limited to 33° (±3°). Glide-slope entry is indicated by airplane pitch change in desponse 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. During the glide-slope phase,

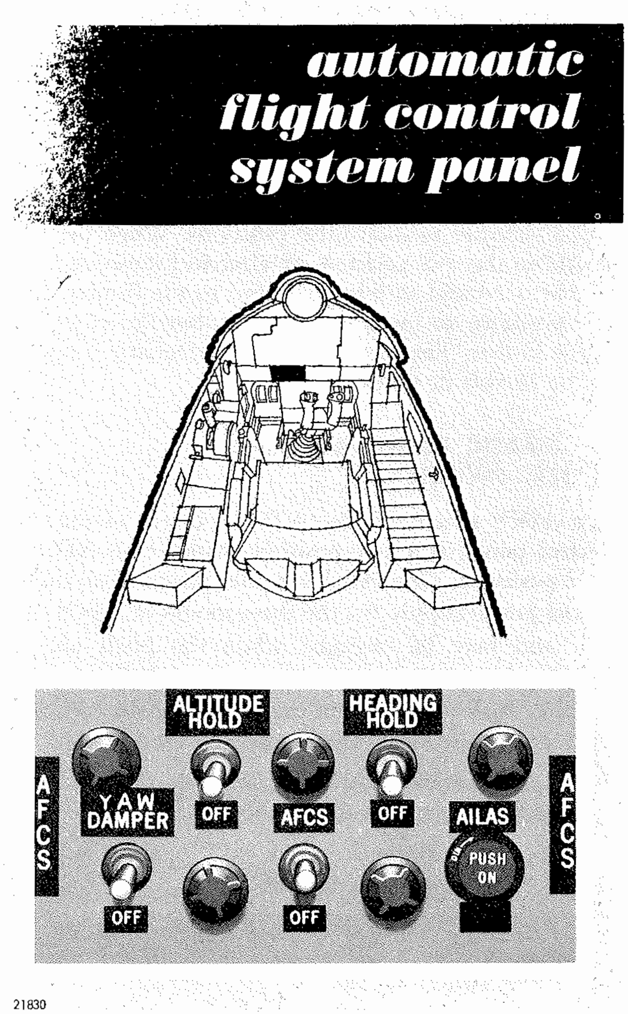


Figure 4-16

bank angle limiting is reduced to 15°. When visual contact with the runway is established, the momentary interrupt trigger should be held depressed, and the landing should be completed manually.

CAUTION

Flare and landing must be accomplished manually as the landing gear will not withstand the impact at 170 KIAS along the 3° glide-slope.

An ILS localizer signal interruption or failure prior to glide-slope entry or any ILS signal interruption will cause AILAS disengagement either immediately or at glide-slope coincidence, and flight control will automatically revert to pilot-assist mode. If normal ILS signals are restored before glide-slope entry, AILAS mode may be reengaged in the normal manner. Refer to AUTOMATIC APPROACH (AILAS), Section IX, for automatic approach procedures.

PITCH G LIMITER TEST SWITCH

On later airplanes, a test switch (figure 1-21) on the utility switch panel placarded "Pitch G Limiter Test" has positions +G, −G, and a spring-loaded (center) position. When the automatic flight control system is engaged, placing the g limit test switch to either the +G or −G position will actuate the g limiter, thus disengaging holding solenoids to return the pitch damper switch, and all AFCS switches to their OFF positions. When the switch is held in the +G position, an electrical input equivalent to that produced at five g's is fed to the limiter. In the −G position, an input signal equivalent to −2g's is fed to the limiter. The pitch g limiter test switch is operated by the 28-volt dc nonessential bus.

AUTOMATIC FLIGHT CONTROL SYSTEM SWITCH

The AFCS switch (figure 4-16), located on the AFCS control panel, is a two-position switch with AFCS and OFF positions. This switch supplies power from the integrated power supply for the three modes of AFCS operation and may be engaged when the pitch and yaw dampers are on and the radar master switch is in STBY or ON. With the switch in the solenoid-held AFCS position, the following conditions may be established:

1. Attack mode is automatically selected upon radar lock-on except in the SNAKE mode.
2. Landing approach mode will operate when the AILAS switch is engaged.
3. The pilot assist mode will function at all other times.

The AFCS switch receives power from the 28-volt dc nonessential bus.

ALTITUDE HOLD SWITCH

The altitude hold switch (figure 4-16), located on the AFCS control panel, is a two-position switch with ALTITUDE HOLD and OFF positions. During the pilot assist mode of operation, pressure altitude will be maintained when the switch is in the solenoid-held ALTITUDE HOLD position. With the switch in the OFF position, pitch attitude of the airplane will be maintained. The holding solenoid which engages the switch will release, and the switch will return to OFF under the following conditions:

1. Switch manually returned to OFF.
2. Momentary interrupt trigger is depressed.
3. The airplane is beep-trimmed in pitch.
4. Fire control system lock-on is obtained.
5. AILAS is engaged. During the constant altitude phase of AILAS, the switch will return to OFF but the system will continue to hold the engage altitude until glide-slope entry.
6. Emergency damper disconnect button is depressed.

The altitude hold switch receives power from the 28-volt dc nonessential bus.

AILAS BUTTON

The AILAS button (figure 4-16) on the AFCS control panel is a pushbutton switch placarded "AILAS" with instructions to PUSH ON on the face of the switch. When the switch is pushed in for engaging a green light illuminates, indicating successful engagement. The indicator light may be dimmed by clockwise rotation. When the switch is engaged, AILAS follows instrument landing system localizer and glide-slope signals to direct the airplane throughout an ILS approach. The AILAS switch can be engaged only if AFCS is engaged, and ILS signal is received, and the airplane is below the glide-slope. This switch receives power from the 28-volt dc nonessential bus.

HEADING HOLD SWITCH

The AILAS button (figure 4-16) on the AFCS control panel is a pushbutton switch placarded "AILAS" with instructions to PUSH ON on the face of the switch. When the switch is pushed in for engaging, a green light illuminates, indicating successful engagement. The indicator light may be dimmed by clockwise rotation. When the switch is engaged, AILAS follows instrument landing system localizer and glide-slope signals to direct the airplane throughout an ILS approach. The AILAS switch can be engaged only if AFCS is engaged, and ILS signal is received, and the airplane is below the glide-slope. This switch receives power from the 28-volt dc nonessential bus.

1. Switch manually returned to OFF.
2. AFCS switch returned to OFF.
3. Emergency damper disconnect button depressed.

The heading hold switch is powered by the 28-volt dc nonessential bus.

AUTOMATIC FLIGHT CONTROL SYSTEM PREFLIGHT

After starting engine, perform the following checks prior to all AFCS flights:

1. Radar master switch — STBY or ON.
2. Artificial horizon—Appears within about 30 seconds.
   Check for appearance of artificial horizon on scope and that the vertical gyro erects in about 60 seconds.
3. Pitch and yaw damper systems — Engage.
4. AFCS switch — AFCS; should engage in about 90 seconds.
   Check that no objectionable stick movement occurs when the switch is placed to AFCS.

5. Longitudinal beep trim — Check.

   Check longitudinal beep trim by trimming NOSE UP and NOSE DN, checking that control stick follows trim button displacement freely.

6. Pitch trim follow up — Check.

   Check longitudinal elevon trim followup by placing AFCS switch to OFF while stick position is trimmed to approximately three inches up or down elevon. Stick movement should be slight.

7. Lateral beep trim — Check.

   Check lateral beep trim by trimming RWD and LWD, noting that control stick follows trim button displacement freely. When aileron application exceeds 2.5° (control surface deflection), note that aileron limit switches cause pitch damper and AFCS switches to return to OFF and stick returns to neutral. Re-engage pitch damper and AFCS switches.

8. Momentary interrupt trigger — Depress; check manual flight.

   Depress momentary interrupt trigger and check that manual flight control is available. Release trigger.

Note

If aileron application exceeds 2.5° while checking for manual control, it will be necessary to re-engage the AFCS switch.

9. Override — Check.

   Check for override of AFCS by rapidly moving the control stick.

Note

If aileron application exceeds 2.5° while checking for override of the system, it will be necessary to re-engage the pitch damper switch and the AFCS switch.

10. Pitch g limiter test switch — +G; note that pitch damper and AFCS switches return to OFF. Re-engage switches.
11. Pitch g limiter test switch — −G; note that pitch damper and AFCS switches return to OFF. Re-engage switches.
12. Emergency damper disconnect button — Depress.

    Depress the emergency damper disconnect button and check that the AFCS and pitch and yaw damper switches disengage.

13. Takeoff trim — Set.

AUTOMATIC FLIGHT CONTROL SYSTEM NORMAL OPERATION

Engaging Procedure

Note

AFCS automatically disengages during vertical gyro erection. If the artificial horizon display on the pilot's scope does not function properly or will not cage, AFCS should not be engaged. The vertical gyro that furnishes the artificial horizon display also furnishes signals to the automatic flight control system for pitch and bank attitudes and, when not operating properly, may cause incorrect and erratic control surface movement.

1. Radar master switch—STBY or ON.
2. Pitch and yaw damper systems—Engage.
3. Trim for desired flight attitude.
4. Artificial horizon—Appears within about 30 seconds.

   Check for appearance of artificial horizon on scope and that vertical gyro erects in about 60 seconds.

5. With momentary interrup trigger depressed, AFCS switch — AFCS.
6. Momentary interrupt trigger — Release to engage AFCS.
7. Heading hold switch — HEADING HOLD.
8. Altitude hold switch — ALTITUDE HOLD.

Note

The altitude hold switch should be engaged in approximately level flight. Engaging while climbing or descending will result in an overshoot condition before damping to a constant altitude.

When operating in altitude hold, the altitude hold switch will return to OFF if the airplane is beep-trimmed in pitch, momentary interrupt trigger is depressed, or if auto attack or AILAS is engaged. The attack mode is automatically engaged when the pilot assist mode of AFCS is in operation, proper armament is selected, and radar lock-on is obtained. Refer to NORMAL OPERATION OF FIRE CONTROL SYSTEM, this Section, for operation of the attack mode.

Disengage Procedure

Momentary disengaging is available by depressing the momentary interrupt trigger. If it is not desired to re-engage AFCS immediately, any of the following actions will disengage the system:

1. AFCS switch — OFF.

2. Radar master switch to WARM or OFF.
3. Pitch and yaw damper systems — Disengage.
4. Emergency damper disconnect button — Depress.

Note

Depressing the emergency damper disconnect button also disengages the damper systems.

Abbreviated Check List

Refer to T. O. 1F-102A-(CL)1-1 for the Abbreviated Check List of the above procedures.

NAVIGATION EQUIPMENT

STANDBY COMPASS

Refer to INSTRUMENTS in Section I.

DIRECTIONAL INDICATOR (SLAVED)—J-2 COMPASS SYSTEM

This system* consists of a flux valve transmitter installed near the tip of the left wing, a control gyro and amplifier installed in the nose wheel well, and the rotating dial element of the radio magnetic indicator (refer to COMMUNICATIONS AND ASSOCIATED ELECTRONIC EQUIPMENT, this Section) on the instrument panel. The system is basically a gyro-stabilized compass that is automatically kept on a true magnetic north heading by signals from the flux valve transmitter, which detects the north-south flow of the earth's magnetic field. The directional gyro control contains an electrically driven gyro, having a spin axis tangent to the earth's surface. The gyro is also slaved to the flux valve transmitter, which puts in corrections referenced to the earth's magnetic meridian. The system provides an indication of magnetic headings without northerly turning error, oscillation or swinging. The compass system is operable when ac and dc essential bus power are available.

Note

Should ac power fail, dc power to the system will also be cut off by the interlock relay. However, dc power failure will not disconnect ac power to the system.

For the first three or four minutes of operation, the gyro is on a fast slaving cycle and will precess rapidly. During this time it should align with the magnetic heading. The gyro then begins a slow slaving cycle. The directional indicator (slaved) is free from drift and requires no resetting in normal operation.

Note

After the gyro reaches operating speed, the indicator should be checked against the standby compass indication to make sure the indicator does not show a 180-degree ambiguity. The system is not operating properly if such ambiguity exists.

AF 53-1791 thru 55-3379 unless modified by TCTO 1F-102A-546.

Indicator readings are incorrect if the airplane exceeds 85 degrees of climb, dive, or bank.

Directional Indicator (Slaved) Fast Slave Button

A fast slave button at the top left side of the instrument panel provides a means of stabilizing the gyro after it has been upset by overbanking or acrobatics. Depressing the button interrupts 28-volt dc essential bus power to the compass. When the button is released, power is restored and the fast slaving cycle is initiated to permit faster gyro recovery to the true heading.

CAUTION

To avoid damage to the slaving torque motor, the fast slave button should not be used too frequently. Allow ten minutes between actuations, and hold button depressed no longer than two seconds.

Directional Indicator (Slaved) Slaving Switch

A switch on the top left side of the instrument panel has two positions, NORMAL and DESLAVE. When in DESLAVE position the switch applies dc power to a relay in the amplifier unit to open the circuit that slaves the control gyro to the flux valve transmitter. This switch is used in polar regions where the excessive dip in the earth's magnetic lines of force causes compass indications to become inaccurate. With the switch in DESLAVE position the compass system may still be used temporarily as a turn indicator if conventional procedures for making gyro drift corrections are employed. Except for the special circumstances noted, the switch should always be on NORMAL.

Directional Indicator (Slaved) Correction Card

A correction card and holder are located on the left side of the cockpit above the console, and forward of the throttle quadrant.

DIRECTIONAL INDICATOR (SLAVED) — J-4 COMPASS SYSTEM

The J-4 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 J-4 compass system consists of a directional control gyro, an amplifier-servo assembly, a control panel (figure 4-17)

AF 55-3380 & on, & airplanes modified by TCTO 1F-102A-546.

J-4 compass control panel

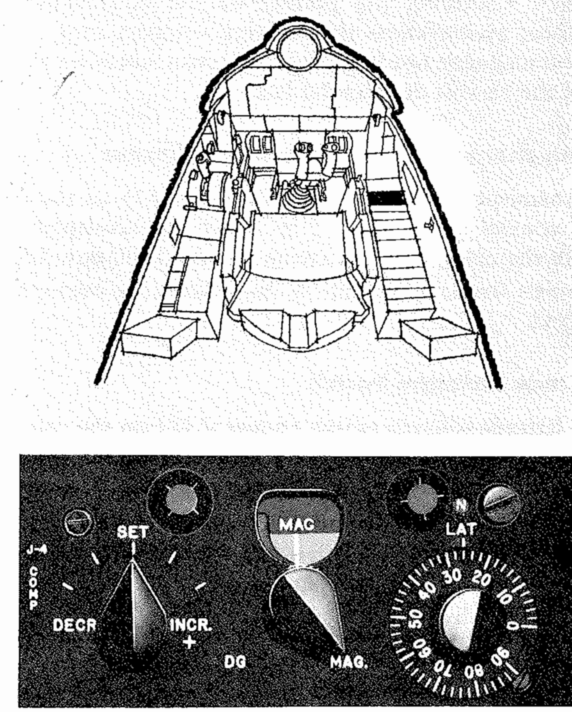


Figure 4-17

on the right-hand console, and the rotating compass card of the radio magnetic indicator on the instrument panel. The two modes of operation, "magnetic slaved" and "directional gyro," provide accurate directional reference for all latitudes. Magnetic slaved mode may be used at all latitudes; however, a severe magnetic distortion occurs when operating near the magnetic poles. When in magnetic slaved mode the system is basically a gyro-stabilized compass slaved to the flux valve (remote compass) transmitter. This mode provides magnetic headings 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 polar regions. When in directional gyro mode, the system is free of magnetic influence and operates 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 due to earth's rotation 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 J-4 compass system also serves as a directional reference for the automatic flight control system and supplies information to the VHF navigation indicators. The system is powered by the 200/115-volt and 26-volt, 400-cycle ac essential bus and the 28-volt dc essential bus. On some airplanes the J-4 compass control panel lights will illuminate whenever the 28-volt dc essential bus is energized. On later airplanes* the control panel lights are controlled by the console light switch located on the lighting control panel (33, figure 1-4). The system is energized whenever power is supplied to the essential buses.

Directional Indicator (Slaved) Correction Card

A correction card and holder are located on the left side of the cockpit above the console, and forward of the throttle quadrant.

Function Selector Switch

The two-position function selector switch (figure 4-17), located on the J-4 compass control panel, has positions DG and MAG. DG position selects directional gyro mode; MAG selects magnetic slaved mode. The switch receives power from the 28-volt dc essential bus.

Synchronizer "SET" Switch and Synchronization Indicator

The synchronizer "set" switch (figure 4-17), located on the J-4 compass control panel, provides a manual means to fast slave, or synchronize, the rotating compass card of the radio magnetic indicator to the correct magnetic heading when the system is in magnetic slaved operation. When in directional gyro mode, the "set" switch is used to position the compass card to the desired gyro heading. The switch has a spring-loaded SET position at the top center and may be moved, against spring tension, to the left "DECR −," position or to the right, "INCR +," position. Two intermediate dash marks on either side of SET position indicate slow or fast synchronization. The first dash mark on either side provides a slewing rate of ½ rpm and the second mark provides a slewing rate of seven to nine rpm. When in DG operation, the direction of movement of the "set" switch is determined by whether an increased or decreased heading is desired. When in MAG operation the direction of displacement of the "set" switch is determined by a synchronization indicator (figure 4-17) above the function selector switch on the control panel. When the function selector switch is in MAG the synchronization indicator is exposed and is placarded MAG. A center dash mark at the bottom of the indicator is opposed by a pointer which is secured at the top

*AF 57-770 & on.

(pendulum-like) and is caused to swing to either a "+" indication on the left or a "−" indication on the right by signals from the flux valve transmitter. When the pointer is off center to the "+" side, the "set" switch is moved to the "INCR +" direction to the desired rate of slewing until the synchronization pointer swings down to the center position. If the pointer is indicating "−," the knob is moved to the "DECR −" position to center of the pointer. Centering the pointer of the synchronization indicator synchronizes the rotating compass card to the correct magnetic heading. The "set" switch receives power from the 28-volt dc essential bus.

Hemisphere Selector Screw and Indicator

The hemisphere selector screw (figure 4-17) 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 Switch

The latitude selector switch (figure 4-17) 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 switch and dial are operative in DG mode 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 switch receives power from the 28-volt dc essential bus.

NORMAL OPERATION OF THE J-4 COMPASS

Magnetic Slaved Mode

1. Select MAG with the function selector switch.
2. Allow approximately two minutes warmup time after power is applied to the essential buses. When power is initially applied, or when switching from DG, a fast slaving 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 2° per minute.
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 "set" switch to center the synchronization pointer. The switch may be used at any time to obtain synchronization and will not produce a hard-over signal if used during automatic flight control system operation.

CAUTION

Do not operate the "set" switch continuously for more than 30 seconds to avoid overheating the slew motor.

Directional Gyro Mode

1. Allow approximately 12 minutes warmup time after power is applied to the essential buses.
2. Select the desired hemisphere with the hemisphere selector screw. Check the "N" or "S" in the hemisphere selector indicator window.
3. Select the latitude in which the airplane will be operating with the latitude selector switch.
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.

Note

As the airplane passes through different latitudes in flight, the latitude selector switch should be rotated to the new latitude every two degrees of latitude change.

ARMAMENT EQUIPMENT

Note

Rerer to the Confidential Supplement, T.O. 1F-102A-1A, for armament information and for the following figures pertaining to armament:

- *Armament Control Panel*— Figure 4-21.
- *Armament Selection Table*— Figure 4-22.

FIRE CONTROL SYSTEM

All information concerning the fire control system is contained in the Confidential Supplement, T. O. 1F-102A-1, except the following:

ELECTRONIC COOLING

During normal flight conditions, the eletronics compartments are cooled by ram air, with the cockpit exhaust air aiding in cooling the forward, upper and intermediate electronics compartments. A pressure-and-temperature-controlled shutoff valve in the ram air duct maintains air pressure at approximately two inches Hg above ambient, and the shutoff valve will close if the temperature of the incoming ram air reaches approximately 160-165°F. When the ram air valve closes due to high inlet temperature, cockpit exhaust air is automatically distributed through the ram air ducts for cooling of the forward, upper, and intermediate compartments. Ground cooling of the electronic compartments with the engine operating is provided by a reverse flow of air in the ram air distribution ducts. In the aft electronics compartment and IFF units, this reverse flow of air is started by the engine compressor creating a partial vacuum in the engine intake ducts, which causes a reverse flow of air through the aft electronic compartment and IFF units. Air in the aft compartment comes from the main wheel well, and air in the tail section is drawn through the IFF units. The vacuum in the intake ducts also causes a purge valve between the left engine inlet duct and the upper electronics compartment to open and draw air from the open nose wheel well through the upper and intermediate electronics compartments. A reverse air flow created by a jet pump is the only ground cooling means for the forward electronics compartments, and the jet pump also assists in cooling the intermediate and upper compartments. When the engine rpm is approximately 72% or below, the jet pump valve opens and allows engine bleed air to flow forward and overboard through the left boundary-layer ram air duct. A partial vacuum is thus created in the ram air duct, which draws cooling outside air from the nose wheel well through the forward, intermediate, and upper electronics compartments. Operation of the jet pump is normally audible to the pilot. A pressure switch senses bleed air duct pressure and prevents the jet pump valve from opening when the duct pressure exceeds 30 to 35 psi. Structure overheat detection probes, near the jet pump nozzles, sense excessive fuselage skin temperatures and automatically close the jet pump valve when excessive temperatures exist. Malfunction of the jet pump system is indicated by an electronics cooling warning light on the warning light panel. During standby conditions with the engine not running, a fitting in the nose wheel well provides for connection of a ground cooling unit.

Note

Refer to ELECTRONICS COOLING, Section V, for cooling limitations.

ELECTRONIC COOLING WARNING LIGHT

An electronic cooling warning light is provided which will illuminate and display "ELECTRONIC COOLING" (16 figure 1-26) to indicate a malfunction in the electronic cooling system. Any time the light illuminates it indicates a malfunction, either in the electronic compartment cooling as intended, or a malfunction in the electrical circuit of the warning light itself. Illumination of the light in flight could indicate the following cooling system malfunctions:

- An electrical malfunction.
- Ram air regulator is closed and ram air pressure downstream of the regulator above two inches Hg.
- Ram air regulator open and ram air temperature above 160°F.
- Jet pump valve open.
- Structural overheat.

Note

Frequently a flashing or flickering light may be experienced; however, only a steady light is significant indication that proper electronic cooling is not provided, and the fire control system should be turned off.

Some airplanes* have no provision for warning of electronic compartment cooling system trouble while airborne, and in these airplanes illumination of the electronic cooling warning light during flight could indicate only a malfunction in the warning light electrical circuit. This illumination would impose no restriction in the operation of the electronic components in these airplanes. For most airplanes** with the electronic cooling warning light modification, armament firing will have no effect upon the electronic cooling warning light operation. On some airplanes†, however, it is normal for the electronic cooling warning light to illuminate briefly during armament firing, and if there is no malfunction in the system it will go out shortly after firing. During ground operations illumination of the electronic cooling warning light will indicate a malfunction on all airplanes** incorporating the electronic cooling warning light modification. There is one exception. With electrical power and the engine furnishing no rpm, it is normal for the electronic cooling warning light to illuminate. If there is no malfunction, the light should go out when external cooling or engine rpm is applied. With other airplanes‡, a check must be made to determine the true existence of a malfunction when the light appears. If the light illuminates when the throttle is above 72%, retard the throttle to below 72%; if the light remains on, there is malfunction in the electronic cooling system and the fire control system should be turned to WARM or OFF. If the light goes out, there is no malfunction and the throttle may be advanced to desired rpm. (Refer to ELECTRONIC COOLING, Section V, for limitations.) If the light illuminates after advancing the throttle during takeoff, disregard as it is normal and the light will go out when the landing gear is up.

**AF 53-1791, -1792, -1794 thru -1796, -1798, -1800 thru -1805 53-1807 thru 54-1387, -1389, -1391 thru -1400, 54-1402 thru 55-3372.*

***All airplanes after compliance with TCTO 1F-102-713.*

†AF 53-1793, -1797, -1799, -1806, 54-1388, -1390, -1401, 55-973 & on, prior to compliance with TCTO 1F-102-713.

‡All airplanes not modified by TCTO 1F-102-713.

NORMAL OPERATION OF THE FIRE CONTROL SYSTEM

Only basic operating procedures are given in this section. Data link procedures are not included. For additional information on fire control system operation and techniques, refer to applicable technical order.

Operating Procedure

Note

Referto ELECTRONICS COOLING, Section V, for cooling limitations which must be observed prior to operation of the fire control system.

1. Armament safety switch — SAFE.
2. Armament selector switch — SNAKE.
3. Igniter control switch — TRAINING (safety-wired).
4. Radar master switch — STBY.

Note

Check that the range trace and artificial horizon appear on the scope in approximately 30 seconds after turning master switch to STBY from OFF or when turning from WARM to STBY.

5. Radar mode selector switch — As desired.
6. Elevation scan control — As desired.
7. Elevation vernier wheel — DETENT.
8. Azimuth scan control — As desired.
9. Antenna elevation scan button — As desired.
10. Anti-clutter switch — OUT.
11. ATOT switch — NORMAL (center) (some airplanes).
12. Anti-jam switch — NORMAL (some airplanes).

    Auto-tune/missile anti-chaff switch — NORMAL (center) (other airplanes).
13. Estimated range rate knob — 0 (some airplanes).
14. Nose-tail switch — As required.
15. Radar master switch — ON.

    When master switch is placed to ON (after delay period has elapsed), check that range trace widens to approximately 1/8 inch and that "grass" or noise appears on the scope to indicate that the set is transmitting.
16. Search intensity — Adjust.

    After radar is determined to be transmitting, the search intensity should be adjusted as follows:

    a. IF Gain control knob — Counterclockwise to stop.

    b. Search intensity wheel — Adjust until range trace is barely visible.

    c. IF Gain control knob — Clockwise to optimum position.
17. Attack intensity wheel — As desired.

    Adjust attack intensity by adjusting brightness of artificial horizon display.

Automatic Search Operation

1. Azimuth scan knob—As desired.

   The azimuth scan knob should be set at B if azimuth of target is unknown. If azimuth is known, use scan knob to obtain best target return.
2. Elevation scan knob—As desired.

   Elevation scan knob should be set to scan desired vertical area.
3. Radar mode selector switch — As desired.
4. Anti-clutter switch — As desired.
5. Anti-jam switch — As desired (some airplanes).

   Auto-tune/missile anti-chaff switch — As desired other airplanes).

   If jamming is detected, select AUTO-TUNE position to obtain radar frequency clear of jamming.
6. Data link GCI station channel selector switch — Assigned channel.
7. Data link airplane channel selector switch — Assigned channel.

Manual Search and Lock-On

Lock-on is obtained after the target is within range by using the following procedure:

1. Action trigger—Depress.

   Squeeze the action trigger to gain manual control of the antenna after azimuth and elevation position of the target is noted.
2. Target—Spotlighted.

   By means of lateral movement of the left control grip and adjustment of the antenna elevation wheel, obtain best possible target return.
3. Range gate marker—Coincide with target.

   When the range gate marker and the target coincide, the attack display should appear showing lock-on has been obtained.
4. Action trigger — Released.

   After lock-on is obtained, release the action trigger and hand control.

Automatic Track; Snake Mode Operation

1. Armament safety switch — SAFE.
2. Armament selector switch — SNAKE.
3. Igniter control switch — TRAINING (safety-wired).

4. Steering dot — Centered.

   After lock-on, a pursuit course is flown by centering the steering dot in the reference circle. To perform an identification pass, steer with the dot centered and maintain the desired rate of closure.

WARNING

The first indication of a possible collision course is evidenced by range circles failing to shrink at the proper time. The pullout maneuver should be executed if a collision course is suspected. The pullout signal will appear when the airplane is 200 feet below and to the right and 200 yards astern of the target.

When it is desired to fly a formation course on a leading airplane, maintain the range circle at a constant diameter, keep the rate of closure gap in the range circle at zero, and maintain a constant position steering dot.

Automatic Track; Missile Firing

1. Armament selector switch — MISSILE selection as desired.
2. Armament safety switch — ARMED.

Note

Normally the desired armament is selected prior to the attack phase during automatic search operation but should be rechecked after lock-on.

3. Igniter control switch —TACTICAL.
4. Lock-on.
5. Steering dot — Centered.

   Fly lead collision course by centering the steering dot within the reference circle.
6. Armament trigger — Depress to SECOND DETENT after reference circle collapses to ¼ inch.

   It will be necessary to depress the armament trigger for firing unless an automatic attack is accomplished.

WARNING

When using AFCS to accomplish automatic attack, armament firing is automatic and use of the armament trigger is not required. If the wrong target was selected, or if for any reason it is desired to abort the attack run, depress manual mode trigger or auto-search button and place the armament safety switch to SAFE to prevent automatic firing.

7. Estimate target range after range circle starts to shrink.

   The range circle will start to shring at 25,000-yard range. Depending on speed and altitude of the airplane, missile launch range will be approximately 3½ to 1½ miles.

WARNING

- If the range circle fails to shrink at proper rate or time with normal rate of closure or if range trace fails to move to the center of scope, a collision course may be indicated and the pullout maneuver should be executed.
- If armament firing is initiated during transient negative pitch maneuvers (not including steady-state negative g maneuvers), it is possible to strike the airplane nose section with the missiles. Therefore, missiles should not be fired during or immediately following rapid application of nose down elevator or sudden release of nose up elevator.

8. Continue to fly course until fire signal (X) appears on scope.

   The fire signal (X) appears at time of missile launch and disappears from the scope in three seconds. When the signal disappears, the pullout maneuver should be executed.

Note

During pullout, do not exceed the limits of radar lock-on or do not intentionally break lock before missile impact. The radar maintains lock-on after the pullout signal appears, to provide return radiated energy for the radar missile guidance system. If lock-on is broken, no energy will be provided to guide the missile, resulting in a probable miss and a free missile.

9. Auto-search button — Depress after missile impact.

   After missile impact, depress the auto-search button to restore automatic searching.

Automatic Track; Rocket Firing

1. Armament selector switch — ROCKET or RAD.
2. Armament safety switch — ARMED.
3. Igniter control switch —TACTICAL.

Note

Normally the desired armament is selected prior to the attack phase during automatic search operation but should be rechecked after lock-on.

4. Lock-on.
5. Steering dot — Centered.

   Fly lead collision course by centering the steering dot within the reference circle.
6. Armament trigger — Depressed to SECOND DETENT and reference circle collapses to 1/4 inch.

   On some airplanes it will be necessary to depress the armament trigger for firing unless an automatic attack is accomplished.

WARNING

When using AFCS, armament firing is automatic and use of the armament trigger is not required. If the wrong target was selected, or if for any reason it is desired to abort the attack run, depress the momentary interrupt trigger or auto-search button and place the armament safety switch to SAFE to prevent automatic firing.

7. Estimate target range after range circle starts to shrink.

   The range circle will start to shrink at 5000-yard range.

WARNING

- If the range circle fails to shrink at proper rate or time with normal rate of closure, or if range trace fails to move to the center of scope, a collision course may be indicated and the pullout maneuver should be executed.
- If armament firing is initiated during transient negative pitch maneuvers (not including steady-state negative g maneuvers), it is possible to strike the airplane nose section with the rockets. Therefore, rockets should not be fired during or immediately following rapid application of nose down elevator or sudden release of nose up elevator.

8. Continue to fly course until fire signal (X) appears on scope.

   When firing rockets, the pullout maneuver should be executed as soon as the signal appears.

WARNING

During rocket firing attacks, the rocket travel is considerably less than for a missile on missile runs and less horizontal clearance is provided. Pullout maneuvers should be planned in advance and correctly executed to avoid possible collision.

9. Auto-search button — Depress after pullout.

   After pullout, depress the auto-search button to restore automatic radar search.

AFCS Automatic Attack

An attack run using either missiles or rockets may be performed automatically by using AFCS. To make an AFCS attack, the AFCS should be operating in pilot assist mode and armament selected prior to lock-on. After lock-on, AFCS "fades" into complete control of the airplane in 1/2 to 4 1/2 seconds.

Note

During AFCS attacks, steering signals are provided from the fire control system. System noise, if present, will be reflected in AFCS steering performance.

Snap-Up Attack

A snap-up attack can be performed by using manual steering (without AFCS) and normal missile procedures. This attack is a lead collision attack except that the target can be as much as 15,000 feet above the airplane. Steering is accomplished by ignoring the elevation steering error until the reference circle collapses. At this time, as rapidly and smoothly as possible, the nose of the airplane should be pulled up to center the steering dot in elevation. At the time of snap-up, the dot should be centered in azimuth. The armament trigger should be squeezed when the reference circle collapses. After the climb attitude is established and the steering dot centered, the missiles are launched by a signal from the computer or clock timer and continue to climb for impact with the target. For a typical snap-up attack, see figure 4-18.

Ground Map Operation

In MAP mode, single bar scan of an area 200 nautical miles forward of the airplane is provided. However, close-in ground mapping can be obtained in other search modes if the airplane is low enough. With the fire control system operating in automatic search, proceed as follows for ground map operation:

1. Radar mode selector switch — MAP.
2. Azimuth scan knob — As desired.
3. Elevation scan switch — DN to provide desired range coverage.
4. IF gain control knob — As required.

   Adjust the IF gain control knob for optimum setting.

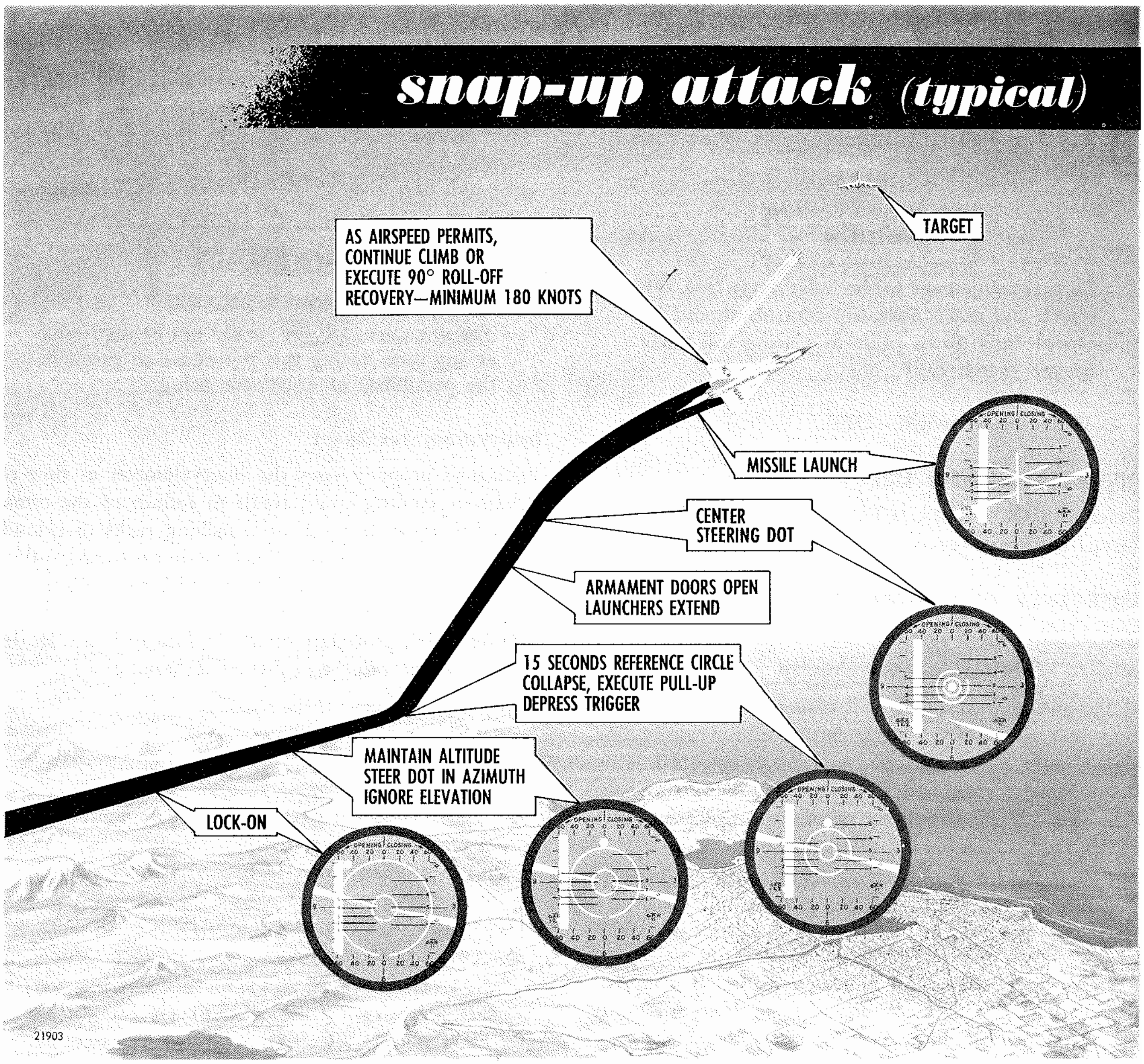


Figure 4-18

5. To expand display, position strobe then move AAI beacon expand switch to BCN EXP.

   To expand display, position strobe at the lower limit of the desired 20 mile sector by use of the hand control; then hold the AAI beacon expand switch to BCN EXP.

Beacon Operation

With the fire control system operating in automatic search, use the following procedure for beacon operation.

1. Radar mode selector switch – BCN.
2. Azimuth control knob – As desired.
3. Elevation control knob – DN to provide desired range coverage.
4. IF Gain control knob – As required.

   Adjust the IF gain control knob for maximum legibility.
5. To expand display, position strobe then move AAI beacon expand switch to BCN EXP.

   To expand display, position strobe at the lower limit of the station return by use of the hand control; then hold the AAI beacon expand switch to BCN EXP to read the station identification.

Prior to Acrobatics or Landing

1. Radar master switch—STBY.

   For acrobatics, landing pattern, and taxiing after landing, the master switch should be in STBY.

2. Armament safety switch—SAFE.
3. Armament selector switch—SNAKE.

Normal Shutdown (After Parking)

1. Search and attack intensity controls—Fully counterclockwise.

CAUTION

To prevent damage to the radar scope face, the search and attack intensity controls should be turned fully down prior to placing the radar master switch OFF.

2. Radar master switch – OFF.

ABBREVIATED CHECK LIST

Refer to T.O. 1F-102A-(CL)1-1 for Abbreviated Check List of the above procedure.

EMERGENCY OPERATION OF FIRE CONTROL SYSTEM

Misfire Warning Light Illuminated

If the misfire warning light illuminates, it is an indication that the intervalometer has produced an ignitor signal and a missile has not been launched (the doors remain open and the selected launchers remain extended). If this condition occurs, the following procedure should be followed:

1. Auto-search button – Depress to break "lock-on."
2. Armament safety switch – SAFE.
3. Armament selector switch – SNAKE.
4. Normally, launchers remain extended if a misfire occurs. If a flight condition exists wherein a clean airplane configuration is necessary, launchers may be retracted after five minutes. In an emergency, the launchers may be retracted after 30 seconds.

Incomplete Armament Cycle

If the launchers remain extended or the armament bay doors do not close for any reason other than a misfire as described above, the misfire warning light will not illuminate. Use the following procedure in attempting to close doors:

1. Depress auto-search button to break lock-on and wait 15 seconds.
2. Armament selector switch to opposite intervalometer and hold for five seconds (if missiles previously selected, select rockets or vice versa).
3. Armament safety switch – ARMED (hold for five seconds).
4. Igniter control switch – TACTICAL.
5. Armament safety switch – SAFE.
6. Armament selector switch – SNAKE.

   If this procedure does not result in closing the armament doors, land with doors open. Refer to LANDING WITH ARMAMENT DOORS OPEN, Section III.

WARNING

The armament trigger should not be depressed at any time during this procedure to preclude the possibility of armament firing.

Intervalometer Reset

Failure to properly reset the intervalometer at time of armament loading could result in failure of the armament bay doors to open or launching racks to extend, causing an aborted attack. This condition would follow a completed radar attack during which the pullout signal was received but the missiles or rockets did not fire. The following procedure should be followed to reset the intervalometer prior to subsequent radar attacks.

WARNING

The armament trigger should not be depressed at any time during this procedure to preclude the possibility of accidental armament firing.

1. Armament safety switch – SAFE.
2. Armament selector switch – ROCKET TRIGGER SALVO (if on last attack rockets were selected) or MISSILE TRIGGER SALVO (if on last attack missiles were selected). Hold for five seconds.
3. Armament safety switch – ARMED (hold two seconds).
4. Armament safety switch – SAFE.
5. Armament selector switch – SNAKE.
6. Proceed with next attack.

ABBREVIATED CHECK LIST

Refer to T.O. 1F-102A-(CL)1-1 for Abbreviated Check List of the above procedures.

COMPUTING OPTICAL SIGHT

The windshield mounted optical sight* (figure 4-19) is used for firing missiles from a pure pursuit course or rockets from a lead pursuit course. The optical sight

**AF 56-1137 & on.*

is generally utilized when radar operation is unsuccessful because of heavy ground return or enemy jamming. The sight head consists of controls for operation and a retractable reflector glass which may be lowered into position when desired. When the intended target has been identified as to type (heavy or medium bomber) the desired target reticle may be selected. Depressing armament trigger to the first detent position will cause a reticle to appear on the reflector glass with a missile selection. The reticle consists of a center dot and two concentric circles and corresponds in size to the type of target selected. The inner circle is used for missile-firing attacks and the outer circle for rocket-firing attacks. In making a rocket-firing attack, an optical selection on the armament control panel will cause the radar antenna to be taken out of automatic search and cause the reticle to appear on the reflector glass. After tracking the target for a few seconds with the antenna caged by means of a switch on the control stick yoke, angular rate voltages build up to establish the amount of lead angle required. The antenna is then uncaged and the computor output then causes the antenna and the reticle position (which is slaved to the antenna) to be precessed to the correct lead angle. Tracking the target with the reticle will then fly the interceptor on a lead pursuit course. During missile firing with the optical sight, depressing the armament trigger to the first position causes the reticle to appear, and will cause the antenna to be fixed along the radar boresight line and remain there. Throughout the missile attack no lead or deflection angle is supplied and the sight operates as a fixed sight.

Note

In addition to the optical sight control discussed below refer also to ARMAMENT TRIGGER, RADAR MASTER SWITCH, and ARMAMENT SELECTOR SWITCH, this Section.

Brightness Selector Knob

The brightness selector knob (figure 4-19) on the face of the optical sight, is used to select the desired brightness of the reticle image. Turning the knob clockwise increases the brightness of the reticle. Turning the knob fully counterclockwise will remove the reticle.

Target Selector Knob

The target selector knob (figure 4-19) is located on the face of the optical sight and is placarded "Target Selector." This knob has four positions of which two are operable. The selection of either of the placarded positions, HVY or MED, provides for display of the correct size reticle on the reflector glass. Selection of HVY

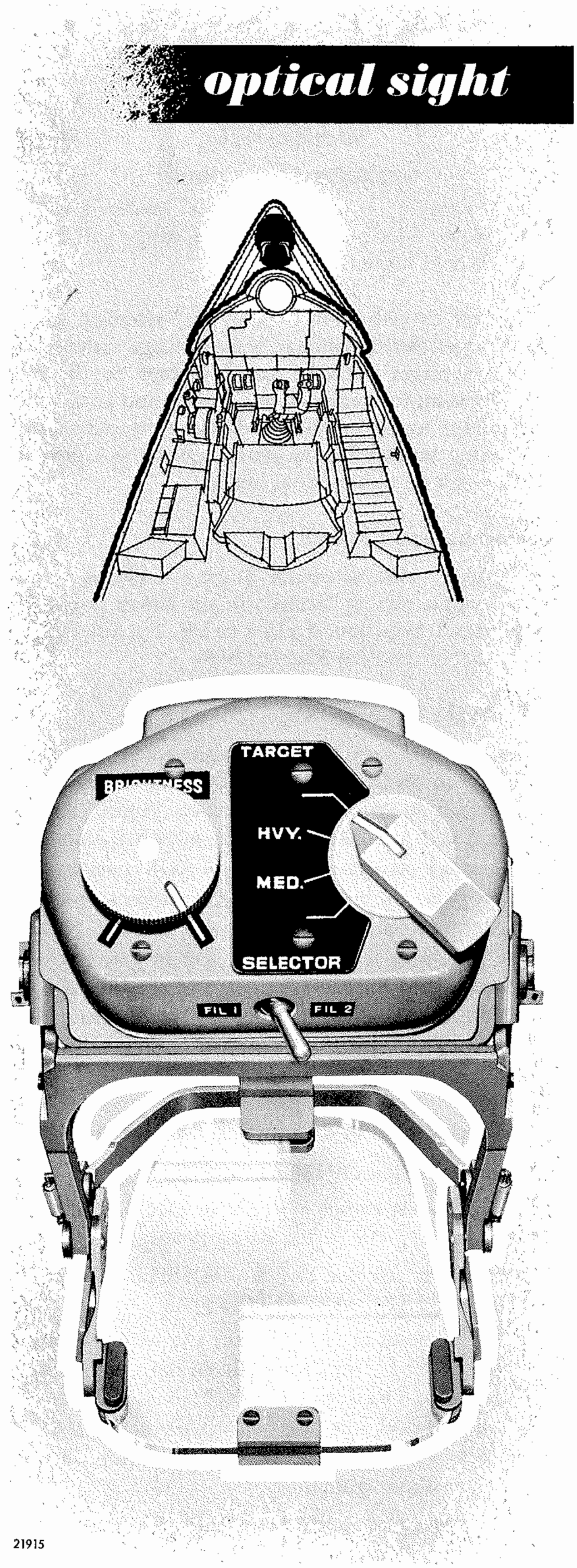


Figure 4-19

displays a reticle which is compatible to operation against a heavy bomber. Selection of MED displays the correct sized reticle for attacking a medium bomber.

WARNING

When making passes on airplanes smaller than the target selector setting, closing range will be closer than normal.

In addition to providing reticle size, selection of the desired target also introduces a fixed range voltage and an airplane-target overtake speed voltage to the computer subsystem for rocket-firing attacks, and an overtake speed voltage to the missile auxiliaries subsystem during missile-firing attacks. The target selector knob is powered by the 28-volt dc nonessential bus.

Filament Selector Switch

The filament selector switch (figure 4-19) is a two-position toggle switch located on the lower portion of the sight head. Selection of FIL 1 or FIL 2 position selects one element of the dual filament bulb.

Optical Sight Cage Switch

The optical sight cage switch (figure 1-20) is a toggle switch located on the platform extending forward from the control stick yoke. It is placarded "Sight" and has positions CAGE and UNCAGE. This switch operates only during optical rocket attacks. In CAGE position, the switch provides for caging the radar antenna along the antenna reference line. When the switch is in UNCAGE and the reticle is maintained on the target, gyros on the antenna sense the rate of change of the line-of-sight, and provides a signal to the computer to determine lead angle. The optical sight cage switch is powered by the 28-volt dc nonessential bus.

Normal Operation of Computing Optical Sight

MISSILE FIRING – OPTICAL

1. Radar master switch – STBY or ON, Delay period elapsed.
2. Armament selector switch – MISSILE selection as desired (RAD, ALL, or OPT).

   This system is primarily for use with the optical missile or for obtaining launch range information for radar missiles in ATOT pursuit attacks. If only optical tracking is available and a RAD or ALL selection is made, the radar missiles will fire without guidance.
3. Armament safety switch – ARMED.
4. Igniter control switch – TACTICAL.
5. Target selector knob – HVY or MED.
6. Armament trigger – Held depressed to FIRST DETENT – reticle appears.

   A minimum of 16 seconds is required for missile preparation prior to firing.

Note

To avoid potentially disabling strain on missile components, do not exceed a two-minute total warmup time. If it appears that the closing portion of the attack will require more than two minutes, let the 13-second hold period expire by releasing the armament trigger until closing time has been reduced. The 16-second preparation time must then be re-accomplished.

7. Brightness knob – As desired.
8. Track target with reticle.
9. When wings of target fill small reticle, armament trigger—Second detent.

WARNING

When making an attack on airplanes smaller than the target selector setting, firing range will be closer than normal.

ROCKET FIRING – OPTICAL

1. Radar master switch – STBY or ON, delay period elapsed.
2. Armament selector switch – ROCKETS – OPT. Reticle appears.
3. Armament safety switch – ARMED.
4. Igniter control switch – TACTICAL.
5. Target selector knob – HVY or MED.
6. Brightness knob – As desired.
7. Optical sight cage switch – CAGE.
8. Track target with reticle.

   Track the target with reticle, establishing and holding a smooth turning rate for four to six seconds minimum.
9. Optical sight cage switch – UNCAGE.
10. Track target with reticle.
11. When wings of target fill large reticle, armament trigger – Depress to second detent.

WARNING

When making an attack on airplanes smaller than the target selector setting, firing range will be closer than normal.

ABBREVIATED CHECK LIST

Refer to T.O. 1F-102A-(CL)1-1 for abbreviated check list of the above procedures.

MISCELLANEOUS EQUIPMENT

PILOT'S MASK DEFOG RHEOSTAT

The pilot's mask defog rheostat (figure 4-20) is located near the aft end of the left console. The control knob is placarded "Mask Defog." The word INCREASE and an arrow indicate that more heat is available to the mask when the knob is turned in a clockwise direction. On later airplanes,* to accommodate use of the MA-2 pressure helmet, the rheostat has been removed and a step-switch installed. The control knob and its function remains the same on these airplanes. The system is powered by the 28-volt dc essential bus.

CAUTION

Use the minimum amount of heat necessary to prevent or remove any accumulation of moisture on the faceplate. It is not necessary that heat be felt on the face. The faceplate heat rheostat should be at maximum heat just long enough to remove moisture and then returned to the minimum heat required to prevent moisture accumulation.

ANTI-G SUIT PROVISIONS

Low-pressure pneumatic system air, cooled by the cockpit air conditioning system, is used to pressurize the pilot's anti-g suit. Duct pressure is reduced to anti-g suit pressure by a type M-8 regulator in the left-hand console. The 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. On some airplanes a manual control enables the pilot to control suit pressures within the pressure range of the regulator output. On other airplanes, varying pressures within the pressure range of the regulator are furnished automatically. The regulator also incorporates a manual control button. When the button is depressed, unregulated low-pressure pneumatic system air pressurizes the anti-g suit. When the button is released, the suit is depressurized. This feature of the regulator may be used by the pilot during extended flights to prevent fatigue by creating a massaging effect on the body. Operation of the manual button does not test the automatic feature of the regulator.

Anti-G Suit Control Knob

A large round knob (figure 4-20) on the left console, aft of the pilot's mask defog rheostat, is placarded "Anti-g Suit" and has arrows pointing to LO and HI positions. On some airplanes turning the knob clockwise increases g suit pressure.

AF 57-770 & on, & airplanes modified by TCTO 1F-102A-570.

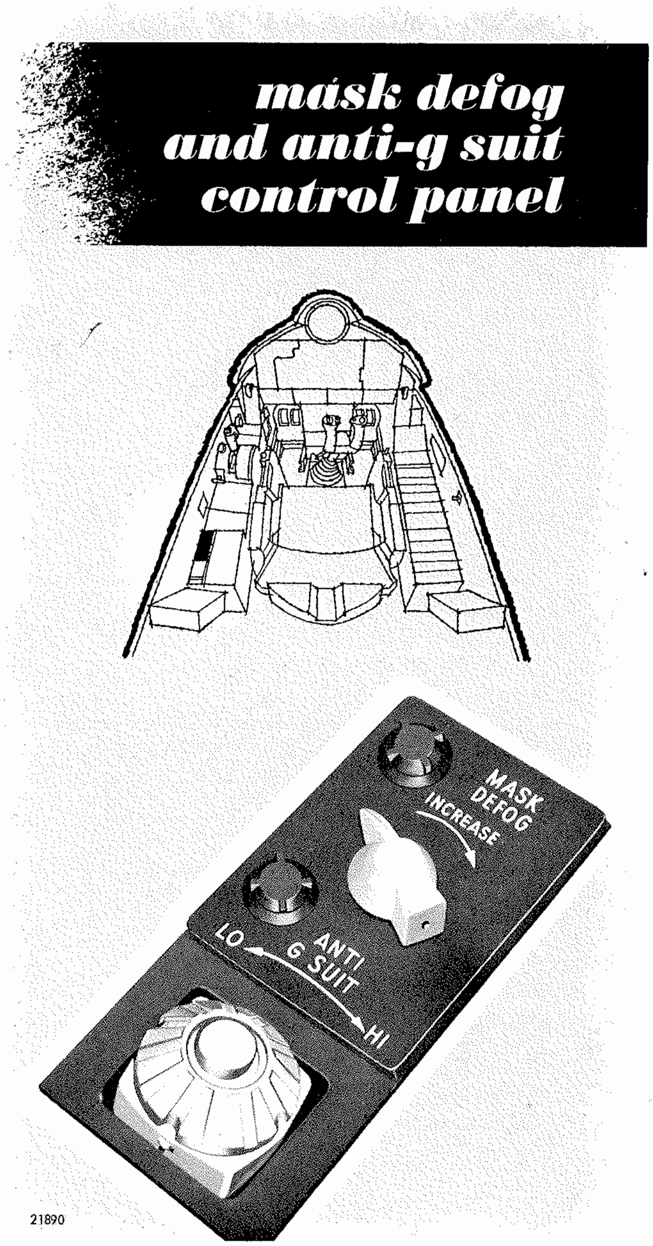


Figure 4-20

SPARE LAMPS

Spare lamps are stowed in holders at the aft end of the left console. The holder is placarded "Spare Lamps."

CHECK LIST

A check list (figure 1-3) is printed on a hinged shelf which folds out of the way when not in use. This shelf is located on the right side of the cockpit, just under the canopy sill. The check list can be illuminated at night by three lights on the shelf, controlled by the console light switch. The lights receive power from the 115-volt ac essential bus.

FLIGHT REPORT, MAP AND DATA CASE

A combination flight report holder and map and data case is located inboard of the spare lamp holder, at the aft end of the left console.

SHOULDER HARNESS STOWAGE STRAP

Some airplanes are equipped with strap assemblies on the left and right-hand sides of the cockpit sill, providing a means of stowing pilot's shoulder harness.

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 that these limitations be closely adhered to.

ENGINE LIMITATIONS

Normal engine limitations are shown in figure 5-1. Additional information concerning operating limitations is given in figure 5-2 for starting, acceleration, idle, maximum continuous, military, and maximum thrust. On some airplanes,* these limitations are listed on a placard located on the left-hand side panel above the compass and altimeter correction cards. 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 fully forward and inboard (nonafterburning).

**Airplanes modified by TCTO 1F-102-707.*

CAUTION

- During operation above 40,000 feet the engine may develop a rough operating condition, or compressor stall if the throttle is moved rapidly or rpm is reduced below 85%. Subsequent movement of the throttle may not eliminate this condition. The probability of a flameout also exists.
- In order to maintain engine EGT within the desired limits, all training flights should be limited to 45,000 feet as far as practical. When required to maneuver above 45,000 feet, climb must be accomplished at a minimum indicated Mach of .93.
- Under standard day conditions, the limit EGT will coincide with the ultimate ceiling of the airplane.
- When climbing above 40,000 feet, monitor EGT continuously. Any time the EGT is observed to be at or approaching limits, the throttle must be retarded or the indicated Mach number increased.

ENGINE OVERSPEED

The maximum permissible engine speed is 102% rpm. Any engine speed in excess of 102% should be noted on Form 781. If the engine speed exceeds 102% rpm, it must be inspected for damage. Speeds in excess of 104% rpm necessitate engine overhaul.

instrument markings

FUEL GRADE JP-4

EXHAUST TEMPERATURE
J57-23 ENGINE
AF 56-957 AND ON

- (green) 470°C TO 610°C CONTINUOUS OPERATION.
- (red) 670°C MAXIMUM DURING FLIGHT (EXCEPT ENGINE ACCELERATION.)
- (red) 680°C MAXIMUM DURING ENGINE ACCELERATION.
  (2 MINUTE GROUND OPERATION)
  (2 MINUTES FLIGHT OPERATION)

REFER TO FIGURE 5-2 FOR ADDITIONAL INFORMATION.

EXHAUST TEMPERATURE
J57-23 ENGINE
AF 53-1791 THRU 55-3464

- (green) 470°C TO 610°C CONTINUOUS OPERATION.
- (red) 670°C MAXIMUM DURING FLIGHT (EXCEPT ENGINE ACCELERATION.)
- (red) 680°C MAXIMUM DURING ENGINE ACCELERATION.
  (2 MINUTE GROUND OPERATION)
  (2 MINUTES FLIGHT OPERATION)

REFER TO FIGURE 5-2 FOR ADDITIONAL INFORMATION.

HYDRAULIC PRESSURE

- (red) 1000 PSI MINIMUM OPERATING PRESSURE
- (green) 2500 PSI TO 3050 PSI CONTINUOUS OPERATION
- (red) 3050 PSI MAXIMUM OPERATING PRESSURE

TACHOMETER

- (green) 90% TO 98% CONTINUOUS OPERATION.
- (red) 102% MAXIMUM PERMISSIBLE RPM.

MACHMETER-AIRSPEED

NOTE
MAXIMUM SPEED POINTER DOES NOT INDICATE ANY LIMIT.

- (yellow) 240 KNOTS IAS MAXIMUM GEAR DOWN

REFER TO TEXT FOR MAXIMUM SPEED LIMITS.

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Figure 5-1

engine operating limits

OPERATING CONDITIONS — J57-23	MAXIMUM OBSERVED EXHAUST GAS TEMP. (C) S.L. TO 30,000 FT.	MAXIMUM OBSERVED EXHAUST GAS TEMP. (C) ABOVE 30,000 FT.	TIME LIMITS GROUND OPERATION	TIME LIMITS FLIGHT OPERATION
MAXIMUM (EXCEPT ENGINE ACCELERATION)	640	670	1 MINUTE	15 MINUTES
MILITARY	630	660	5 MINUTES ⚠1	30 MINUTES
MAXIMUM CONTINUOUS	580	610	5 MINUTES ⚠1	CONTINUOUS
IDLE	340	—	—	—
STARTING	630	630	—	—
ACCELERATION	680	680	2 MINUTES	2 MINUTES

NOTE
IF TEMPERATURE LIMITS ARE EXCEEDED, MAKE ENTRY IN FORM 781 STATING DURATION AND PEAK TEMPERATURES.

⚠1 REFER TO "COOLING LIMITATIONS" TEXT THIS SECTION

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Figure 5-2

COOLING LIMITATIONS (GROUND OPERATIONS)

If the afterburner is not operated, the engine can be run for a total of five minutes at power above approximately 77% rpm. In addition, the engine can be run for one minute at maximum power (military plus afterburning). If these ground operating limits are closely approached, five minutes operation at idle thrust is required to allow for engine cooling before a schedule which will approach the limits can be repeated. If the engine has been operated for longer than one minute above 85% rpm in the five-minute period prior to shutdown, it will be necessary to operate the engine at idle rpm for a period of five minutes to stabilize engine temperatures. Immediately prior to shutdown, operate the engine at 70% rpm for 30 seconds to scavenge the oil system. Make entry on Form 781 if temperature limits are exceeded and state the time and temperature peaks.

AFTERBURNER LIMITATION

Do not attempt to start afterburner below 83% of military thrust.

EMERGENCY FUEL

Aviation gasoline (Mil-G-5572) or JP-5 (Mil-J-5624) may be used as an emergency fuel. Aviation gasoline, when used, should be the lowest available grade and the temperature of the fuel in the tanks at take-off must not be greater than 70°F.

CAUTION

When aviation gasoline is used it must be diluted 2.5% by volume with oil, Specification Mil-O-6082, grade 1100, and flight restricted to altitudes below 35,000 feet JP-5 fuel freezes at minus 55°F; therefore when using this fuel, flight must be restricted to altitudes where temperatures below this point will not be encountered.

ENGINE OIL

The maximum allowable engine oil consumption is four pints per hour.

AIRSPEED LIMITATIONS

MAXIMUM ALLOWABLE AIRSPEEDS

Some airplanes* are limited to 655 KIAS or Mach 1.25, whichever occurs first, due to inlet duct vibration at high indicated airspeeds. Other airplanes**, except AF 53-1813, are cleared of this structural deficiency and may be flown at design limit speed of 655 KIAS or Mach 1.5, whichever occurs first. AF 53-1813 is limited to 600 knots or Mach 1.25, whichever occurs first, due to a non-production vertical fin.

LANDING GEAR EXTENSION SPEED

The maximum allowable airspeed with landing gear extended is marked by a yellow radial on the airspeed indicator. The airplanes are limited to 240 KIAS with the landing gear extended.

DRAG CHUTE DEPLOYMENT SPEED

The maximum allowable airspeed at which the drag chute may be deployed is 160 KIAS. If the drag chute is deployed at speeds above this limit it will be automatically released from the airplane.

CAUTION

To prevent loss of the drag chute or to prevent slowing the airplane to excessively slow speeds, the drag chute must not be deployed in flight.

RAT LIMIT SPEED

The maximum allowable airspeed at which the RAT can be extended or operated is 345 KIAS. Extension of the RAT at speeds above 345 KIAS is likely to cause overspeed or structural damage. Satisfactory emergency hydraulic pressure will not be maintained at airspeeds below 125 KIAS.

OTHER OPERATIONAL LIMITATIONS

AUTOMATIC FLIGHT CONTROL SYSTEM (AFCS)

Operation of all modes of AFCS is temporarily prohibited below 12,000 feet at speeds above 480 KIAS on some airplanes.† Operation of the attack mode of AFCS is prohibited when external tanks containing fuel are installed.

AF 53-1791, -1792, -1795 thru -1803, -1805 thru -1812, -1814 thru -1818, 54-1371 thru -1387, -1389, -1391 thru -1397, -1399, -1400, 55-3357 thru -3373, unless modified by TCTO 1F-102A-524.

***AF 53-1793, -1794, -1804, 54-1388, -1390, -1398, -1401, 55-3374 & on & airplanes modified by TCTO 1F-102A-524.*

†80% limit (refer to PROHIBITED MANEUVERS, this Section).

ELECTRONICS COOLING

Ground Operation

During engine runup, the jet pump will cut out at approximately 72% rpm. Without jet pump cooling, radar equipment will begin to suffer damage within one minute. On the ground, operation of the radar with radar master switch in STBY or ON is prohibited, with engine rpm above 72%, except during takeoff. With engine rpm below 72%, operation of the radar in STBY or ON, is limited to seven minutes. When the radar is required as soon as the airplane is airborne, the following procedures should be followed:

1. If airplane is taxied with radar master switch in WARM, place master switch to ON and wait 30 seconds before advancing throttle for takeoff.
2. If airplane is taxied with radar master switch in STBY or ON (engine rpm below 72%) takeoff should be made within seven minutes of the radar master switch turned to WARM.

Inflight Operation

During hot day conditions (100°F or warmer at sea level), flight at either military power or loiter (maximum endurance) throttle settings below 5000 feet are restricted to seven minutes with the MG-10A radar master switch in ON or STANDBY positions. If flight at these throttle settings exceeds seven minutes, the radar master switch must be in OFF or WARM positions to prevent excessive heat from damaging electronics equipment.

Note

- The radar master switch must be in the WARM or OFF position during flight below 10,000 feet at maximum speed during hot weather operation to prevent damage to electronic equipment from overheating.
- If the radar master switch is placed in WARM position, the pilot assist and automatic functions of AFCS will be rendered inoperative.

EXTERNAL TANKS

External Tank Rubbing Strips

Installation of external tanks is prohibited until satisfactory rubbing strips are installed on the main landing gear fairing doors and the external tanks to prevent damage to the door and tank from contact with each other.

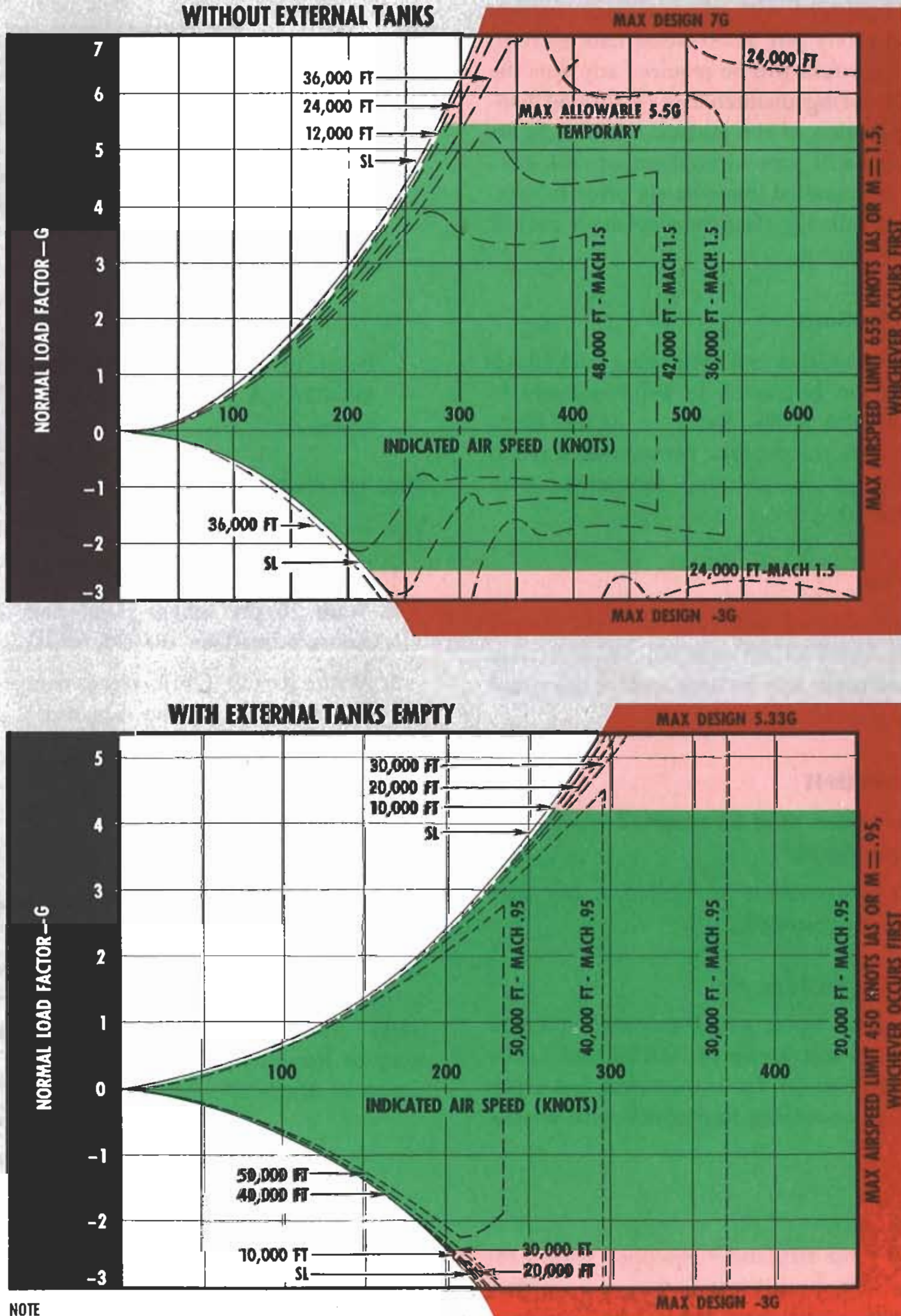
External Tank Fuel Transfer Switch

The external tank fuel transfer switch must remain in the OFF position until internal fuel quantity indicates 5500 pounds or less to maintain cg within limits. On airplanes not equipped with an external tank fuel transfer switch it will be necessary, prior to flight with full external tanks, to check center of gravity in accordance with

operating flight limits

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
FUEL GRADE: JP-4



NOTE
- FLIGHT CAPABILITY FOR VARIOUS ALTITUDES INDICATED BY DOTTED LINES.
- REFER TO TEXT FOR MAXIMUM ALLOWABLE AIRSPEEDS AND LOAD FACTORS.

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Figure 5-3

manual of weight and balance data, T.O. 1-1B-40, and either ballast the airplane or off-load fuel if it is found that the aft center of gravity is beyond the permissible limit.

External Tank Jettison Warning Lights and Safety Pin

On airplanes not equipped with external tank jettison warning lights and safety pin, inadvertent tank ejection is possible. Extreme caution will be required any time the cockpit is occupied during maintenance or ground handling. Until incorporation of the jettison warning lights and safety pin, which will prevent inadvertent tank ejection, tanks should be installed immediately prior to takeoff and removed immediately after the airplane is parked and the engine is shut down.

Maneuvering Limitations

With external tanks installed and containing fuel, maximum aileron deflection is limited to approximately 1/3 aileron (3°) above 270 KIAS. Refer to ACCELERATION LIMITATIONS, this Section, for symmetrical and asymmetrical (rolling) maneuvering limitations with external tanks installed.

Airspeed Limitation

Maximum allowable airspeed with external tanks (empty or containing fuel) is 450 KCAS or Mach .95, whichever occurs first. External tanks may be jettisoned at any speed within the operating speed range.

ARMAMENT EQUIPMENT

The following limitations must be observed when operating armament equipment:

1. Firing of the armament is prohibited at any time external tanks are installed.

Armament Gear Limitations

With missile bay doors open, avoid excessive roll and rudder-induced maneuvers. Refer to ACCELERATION LIMITATIONS, this Section, for symmetrical and asymmetrical (rolling) maneuvering limitations with missile bay doors open.

Rocket Firing

Rocket firing from some airplanes* equipped with 2.75-inch rockets is prohibited until installation of a satisfactory debris deflector on each blast pan in the armament door. Possible damage which could be incurred when firing without a debris deflector is:

- Damage to the missile or missile radome.
- Dents or holes in the missile bay beams.
- Damaged electrical wiring, pneumatic lines, or hydraulic lines.

CAUTION

It is also possible for debris from a forward 2.75-inch rocket to prevent the aft rocket from leaving the tube, resulting in destruction of the armament bay door when aft rocket is ignited. For practice rocket firing missions, ensure that only the forward 2.75-inch rockets are loaded.

Note

Refer to the Confidential Supplement, T.O. 1F-102A-1A, for figure 5-3, Missile Firing Limitations.

TIRE LIMITATIONS

The maximum allowable tire ground speeds are:

1. With 18 ply rubber tread tires—174 knots.
2. With 20 ply rubber tread tires—195 knots (195 knots printed on the side wall).
3. With 20 ply fabric tread tires—217 knots (217 knots printed on the side wall).

Note

Refer to the Appendix for takeoff tire ground limit speeds.

ELEVATOR FEEL-FORCE REGULATION (SHELLY INTELLIGENCE UNIT)

If the pin on elevator feel-force intelligence unit is found to be extended or retracted (not flush with case of the unit) ±0.06 inch (approximately 1/16 inch) during exterior inspection, the airplane should not be flown in excess of Mach .90.

CAUTION

Shaft retraction directs unregulated ram air into the elevator feel-force cylinder. The increased pressure in the cylinder produces higher stick forces, but the airplane remains controllable at restricted speeds. If inspection shaft extends from hole more than 0.06 inch, the regulator must be replaced.

*AF 53-1791, -1792, -1794 *thru* -1796, -1798 *thru* -1805, -1807 *thru* -1818, 54-1371 *thru* -1389, -1391 *thru* -1400, -1402 *thru* -1407, 55-3357 *thru* -3464, 56-957 *thru* -1136 *unless modified by* TCTO 1F-102-609.

SPEED BRAKES

Speed brakes operation is temporarily restricted, on some airplanes* as follows:

1. Flight with speed brakes open is prohibited at speeds above 300 KIAS.
2. Speed brakes should not be used during abrupt maneuvers below 300 KIAS.
3. Use of speed brakes should be avoided while flying in close proximity of other aircraft.

CANOPY

When taxiing airplanes equipped with a canopy hold button, do not exceed 60 knots ground speed or 90 KIAS, whichever is less. On airplanes without a canopy hold button, the canopy hold-open rod must be used and taxi speed limited to 30 knots ground speed. Taxi speeds in excess of the above may result in damage to the canopy or airplane structure.

PROHIBITED MANEUVERS

SPINS

Intentional spins are prohibited due to excessive loss of altitude and the probability of exceeding engine temperature limits. (Refer to SPINS, Section VI.)

NEGATIVE G MANEUVERS

Maneuvers at less than one g or at a negative g for more than three seconds duration are prohibited on airplanes without an oil recirculating system. The recirculating system will supply satisfactory oil pressure to the constant-speed electrical power system during negative g conditions. On modified airplanes** which incorporate a recirculating system, negative g maneuvers are limited to 15 seconds maximum, of which only 10 seconds may be at zero g. This limitation is necessary to ensure adequate engine lubrication.

OTHER MANEUVERING LIMITATIONS

AILERON DEFLECTION LIMITATIONS

Note

- There are no aileron deflection restrictions on any airplanes during takeoff and landing when airspeed is below 270 KIAS.
- At approximately ¾ aileron deflection, an additional 10 pounds resistance (feel force) is imposed by added spring tension and must be overcome to obtain full aileron.

**AF 53-1791 thru 57-823, 57-825 thru 57-830, -832, -833, -835, -837, -838, -840, & -841, unless modified by TCTO 1F-102-725.*

***AF 53-1791, -1793, 56-1083 & on, & airplanes modified by TCTO 1F-102-602.*

Airplanes With Enlarged Vertical Fin

1. Yaw damper and turn coordinator on:
   a. Above 450 KIAS below 20,000 feet—¾ aileron (½ aileron on some airplanes†).
   b. All other conditions—No restriction.
2. Yaw damper and turn coordinator off:
   a. Above 450 KIAS below 20,000 feet—½ aileron.
   b. All other conditions—¾ aileron except for takeoff and landing.

Airplanes With Small Vertical Fin

1. Yaw damper and turn coordinator on or off:
   a. Above 450 KIAS below 35,000 feet—¾ aileron.
   b. Above 480 KIAS below 12,000 feet—½ aileron.
   c. All other conditions—No restriction.

Yaw Damper and Turn Coordinator Off

When operating the airplane with the yaw damper and turn coordinator off, the following restrictions shall be observed to prevent exceeding the current structural limitations of the airplane:

1. Roll maneuvers shall be restricted to positive quadrant rolls only.

Note

A positive quadrant roll is a roll to a 90° bank either side of level flight or from a 90° bank in one direction to a 90° bank in the opposite direction passing through level flight position.

2. No uncoordinated maneuvers shall be performed except during takeoff and landing.
3. Rapid application of aileron shall be avoided except during takeoff and landing.
4. No intentional rolling pushdowns shall be performed.

Roll Maneuvers

When executing rolls through greater than 180°, elevator position should not be varied from the position selected at entry, except to prevent the airplane from exceeding limit asymmetrical load factors.

Note

If elevator is inadvertently applied during rolls and excessive pitch or yaw maneuvers are encountered, do not attempt to fight the response. Neutralize all controls until control is regained.

†80% load factor limit.

Airplanes With Small Vertical Fin

In addition to the other maneuvering limitations, the following restrictions apply to airplanes with the small tail:

1. Snap rolls or snap maneuvers are prohibited.
2. Uncoordinated maneuvers are prohibited.
3. High rates of roll must be avoided to prevent inertia coupling (see Section VI).
   a. Roll rates for positive quadrant rolls are not restricted (roll to a 90° bank either side of level flight or from a 90° bank in one direction to a 90° bank in the opposite direction passing through level flight position).
   b. For full roll maneuvers (360°) with the sideslip angle transducer inoperative (flight mode selector switch to DIRECT MAN), the airplane is limited to a ½ aileron deflection or a maximum roll rate of 80° per second. With sideslip angle transducer inoperative (flight mode selector switch to MAN), the airplane is limited to ¾ aileron deflection or a maximum roll rate of 120° per second.

ACCELERATION LIMITATIONS

Some airplanes are restricted to 80% of design load factors until certain structural modifications are accomplished, at which time they are cleared for operation at 100% design load factors. Those airplanes that are limited to 80% of design load factors are as follows:

- AF 54-1375, 54-1390, 55-3430, 56-992, and 56-995 unless modified by TCTO 1F-102-633A.
- AF 53-1799 unless modified by TCTO 1F-102-600.
- AF 53-1793, 53-1799, 53-1817, and 54-1403 unless modified by TCTO 1F-102-639.
- AF 53-1794, 53-1797, 53-1799, and 53-1806 unless modified by TCTO 1F-102A-538.

Figure 5-3 represents operating flight limits for symmetrical loads. Asymmetrical (rolling) maneuvers impose higher loads on the airplane and therefore require more restrictive g limitations. The following tables show symmetrical and asymmetrical maneuvering limitations:

Missile Bay Doors Open

Symmetrical		Rolling Pullouts
Negative	Positive	Positive
−0.8g*	+2.4g*	
−1.0g	+3.0g	

**80% Limits.*

External Tanks Containing Fuel

Symmetrical		Rolling Pullouts
Negative	Positive	Positive
−0.8g*	+2.4g*	0 g to 1.8g*
−1.0g	+3.0g	0 g to 2.3g

**80% Limits.*

External Tanks Empty or Over 4200 Pounds Internal Fuel

Symmetrical		Rolling Pullouts
Negative	Positive	Positive
−2.2g*	+4.2g*	0 g to 3.1g*
−2.7g	+5.3g	0 g to 3.9g

**80% Limits.*

Clean Airplane, Less Than 4200 Pounds Fuel

Symmetrical		Rolling Pullouts
Negative	Positive	Positive
−2.4g*	+5.6g*	0 g to 4.0g*
−3.0g	+7.0g	0 g to 5.0g

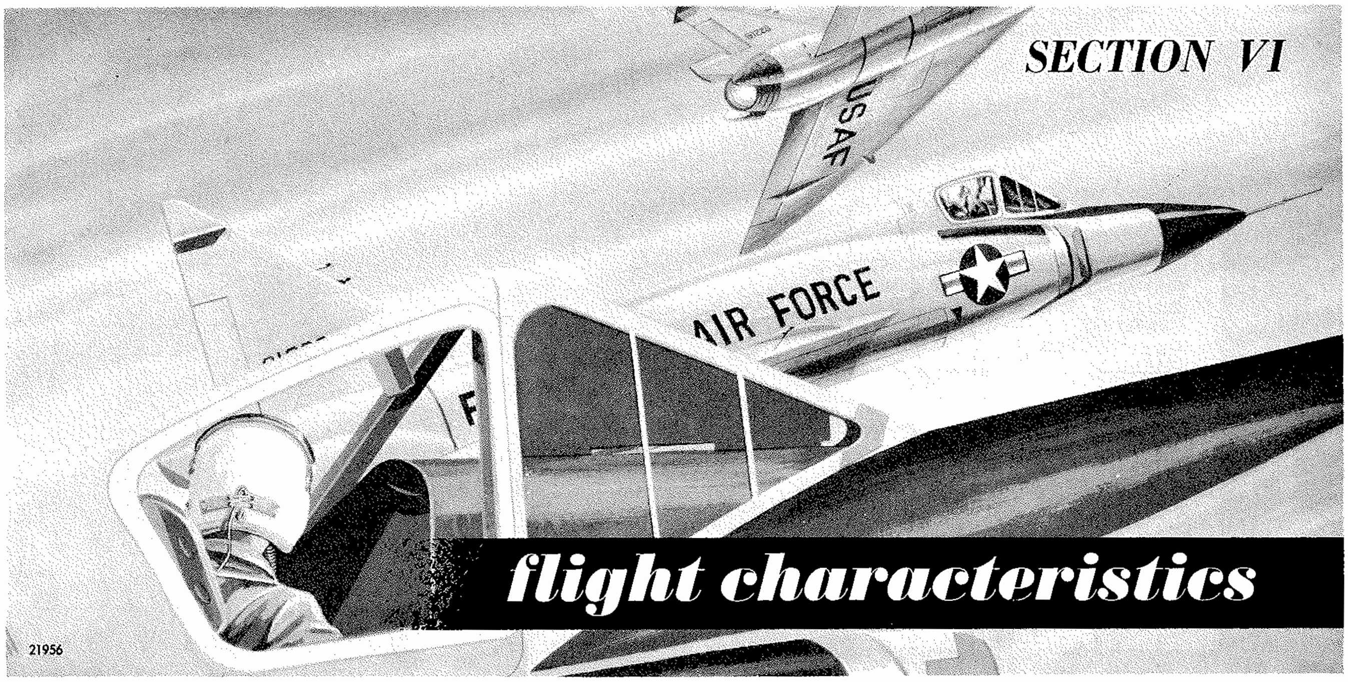
**80% Limits.*

CENTER OF GRAVITY LIMITATIONS

The only in-flight control of cg position is the firing of armament and the transfer of external fuel. (Refer to EXTERNAL TANKS, this Section.) The cg position at takeoff is therefore important and should be closely checked and maintained within limits either by armament loading or ballast. CG position at takeoff approaches the aft limit and moves forward during flight and is normally near the forward limit for landing. Firing of armament soon after takeoff may result in the cg position shifting beyond the aft limits. (Refer to Handbook of Weight and Balance Data, T.O. 1-1B-40.)

WEIGHT LIMITATIONS

The design of the airplane precludes the possibility of overloading; therefore, the maximum gross weight will not be exceeded for takeoff as long as standard armament and full external tanks are carried. At landing weights (less than 22,000 pounds) the rate-of-sink at touchdown is limited to a maximum of 540 feet per minute, to prevent structural damage to the landing gear. If the gross weight at touchdown exceeds 22,000 pounds or if empty external wing tanks are still on, the designed rate of sink at touchdown is 300 feet per minute. Refer to FLIGHT WITH EXTERNAL TANKS in Section VI for additional information.



SECTION VI

flight characteristics

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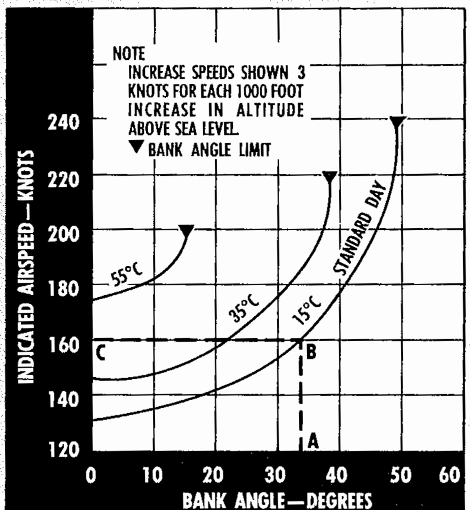
The airplane has a conventional turbojet airplane response to change of thrust; however, the change of thrust in the operating range can be relatively large for a small throttle movement due to engine bleed valve operation. Optimum performance is obtained when the cg is maintained close to the aft limit due to reduced trim drag. 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. 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. The maneuvering capabilities of the airplane are generally good; however the roll restrictions as established in Section V are imposed to provide satisfactory operation of the airplane and, at the same time, reduce the possibility of encountering inertial coupling. Later airplanes are provided with an increased area vertical fin to eliminate roll restrictions imposed because of inertial coupling dangers. Flight tests indicate that airplanes with the increase of 40% vertical fin area have less inertial coupling effects and better handling and general flying qualities than the earlier airplanes without this design improvement. With incorporation of the increased area fin, performance of the airplane is not perceptibly changed. The general handling characteristics of the airplane during flight with a dead engine are good in the low speed ranges. However, when a flameout occurs (or is induced) at airspeeds above 220 KIAS, the interruption of airflow through the engine inlet ducts will create a duct rumble which causes an annoying buffet or heavy vibration within the airplane. The severity of this disturbance will increase proportionally to the indicated airspeed, and is only slightly perceptible at best glide speed of 220 KIAS.

STALLS

The handling characteristics of this airplane at low speeds are excellent, with good control response below the recommended minimum speeds of the airplane, and no "stall," in the usual sense of the word, is encountered. A stall in this airplane may be defined as loss of control about any one axis. See figure 6-1 for the recommended minimum speeds based on normal gross weight and sea level altitude. The data are presented for maximum and military thrust and show the variation of minimum speed with bank angle, load factor and ambient air temperature. A correction factor is included to account



LEVEL FLIGHT MINIMUM SPEEDS WITH MILITARY THRUST

BANK ANGLE	LOAD FACTOR
0°	1.0 G
30°	1.2 G
45°	1.4 G
60°	2.0 G

EXAMPLE

A INITIAL BANK ANGLE =33°.
B OUTSIDE AIR TEMPERATURE =15°C.
C MINIMUM AIRSPEED =160 KNOTS IAS.

ALTITUDE REQUIRED TO RECOVER FROM DESCENT IN ADJUSTING THRUST FROM THAT SPECIFIED TO MILITARY THRUST

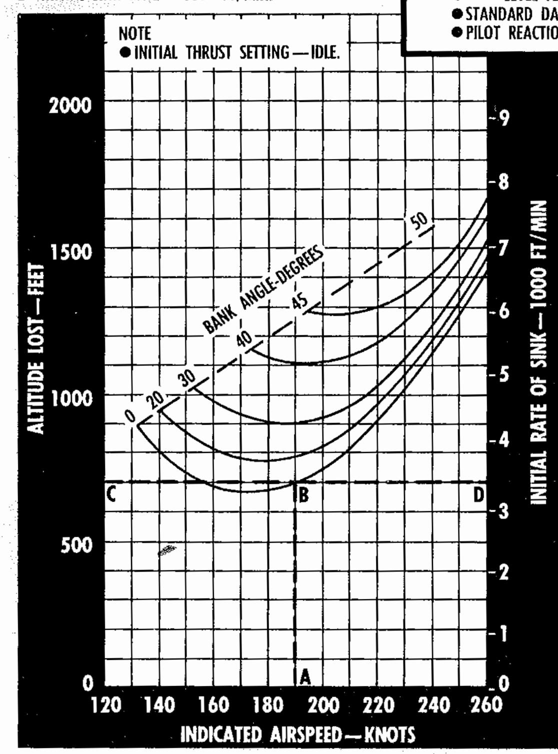
EXAMPLE

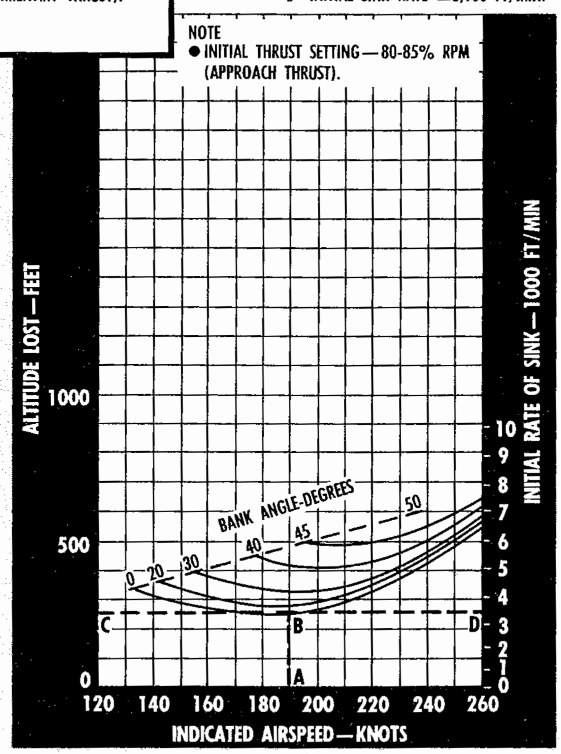
A INITIAL AIRSPEED =190 KNOTS IAS.
B INITIAL BANK ANGLE =0°.
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT =700 FEET.
D INITIAL SINK RATE =3300 FT/MIN.

NOTE
- ALTITUDE LOST IS HIGHLY DEPENDENT ON PILOT TECHNIQUE AND CURVES ARE THEREFORE APPROXIMATIONS.
- ---LEVEL FLIGHT MINIMUM SPEED (MILITARY THRUST).
- STANDARD DAY.
- PILOT REACTION TIME =2 SECONDS.

EXAMPLE

A INITIAL AIRSPEED =190 KNOTS IAS.
B INITIAL BANK ANGLE =0°.
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT =250 FEET.
D INITIAL SINK RATE =3,100 FT/MIN.





21957-1

Figure 6-1

minimum speeds

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
FUEL GRADE: JP-4

CONDITIONS

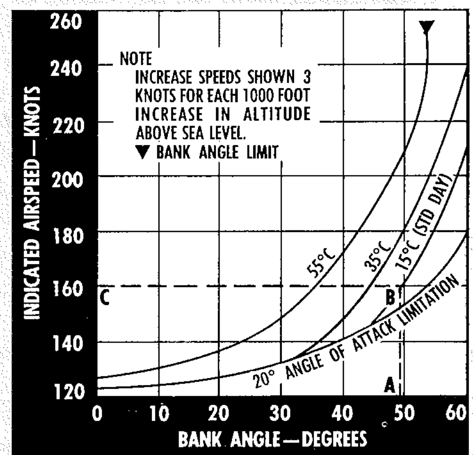
- GROSS WEIGHT EQUALS 27,500 LBS
- C G 27.5%
- AMBIENT TEMPERATURE AS INDICATED
- ALTITUDE—SEA LEVEL

CONFIGURATION

- LANDING GEAR DOWN
- SPEED BRAKES OUT

LEVEL FLIGHT MINIMUM SPEEDS WITH MAXIMUM THRUST

EXAMPLE
A INITIAL BANK ANGLE =49°.
B OUTSIDE AIR TEMPERATURE =15°C.
C MINIMUM AIRSPEED =160 KNOTS IAS.

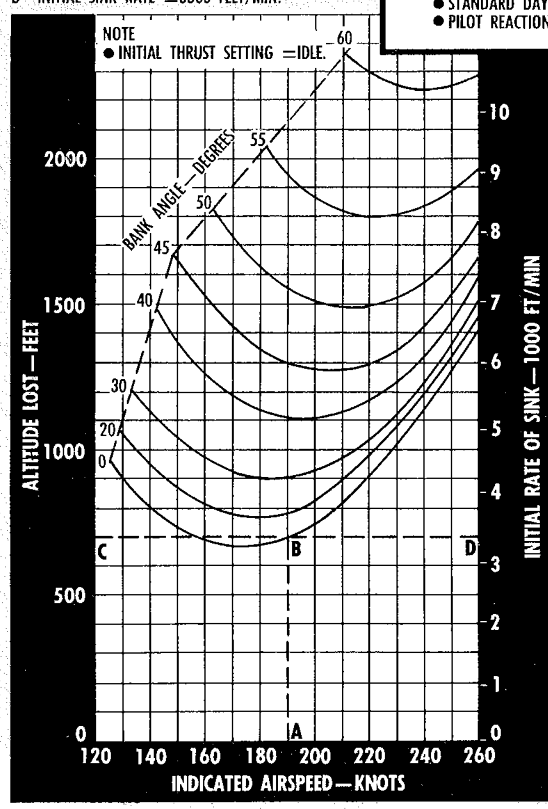


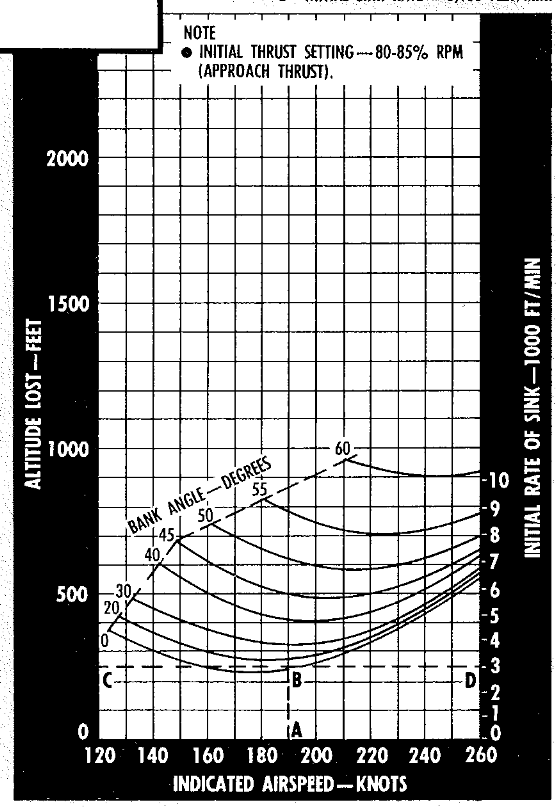
ALTITUDE REQUIRED TO RECOVER FROM DESCENT IN ADJUSTING THRUST FROM THAT SPECIFIED TO MAXIMUM THRUST

EXAMPLE
A INITIAL AIRSPEED =190 KNOTS IAS.
B INITIAL BANK ANGLE =0°.
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT =700 FEET.
D INITIAL SINK RATE =3300 FEET/MIN.

NOTE
- ALTITUDE LOST IS HIGHLY DEPENDENT ON PILOT TECHNIQUE AND CURVES ARE THEREFORE APPROXIMATIONS.
- ---LEVEL FLIGHT MINIMUM SPEED (MAXIMUM THRUST).
- STANDARD DAY.
- PILOT REACTION TIME =2 SECONDS.

EXAMPLE
A INITIAL AIRSPEED =190 KNOTS IAS.
B INITIAL BANK ANGLE =0°.
C ALTITUDE REQUIRED TO REGAIN LEVEL FLIGHT =250 FEET.
D INITIAL SINK RATE =3,100 FEET/MIN.





21957-2

for altitudes greater than sea level. Additional data are presented in the form of rate-of-sink and altitude required to recover from descent with idle thrust and approach thrust (80-85% rpm) using maximum thrust are military thrust for recovery. During level flight, at a gross weight of 25,000 pounds, a warning buffet starts at approximately 170 KIAS, and increases slightly to a moderate buffet at 125 KIAS. With military thrust at 30,000 feet, a rate-of-sink develops at approximately 160 KIAS, and very high rates-of-sink are encountered at lower speeds. As the airspeed is reduced below 110 knots, the ailerons should not be used for lateral control of the airplane as the adverse yaw induced may inadvertently cause a spin entry. Rudder control alone should be used to maintain lateral directional control to stall speed of 95 KIAS. At this speed, the airplane may fall off in a nose-down spiral or, if back pressure is held, enter a spin. Recovery from a stall, or the approach to a stall, is easily effected by relaxing back pressure on the stick; however, considerable altitude (up to 6000 feet) will be lost. The recommended minimum speed for this airplane is 125 knots. Below this speed, lateral control effectiveness falls off rapidly and very high sink rates are encountered (up to 10,000 feet per minute). At altitudes above 35,000 feet, engine compressor stalls are usually encountered below 115 KIAS, and if allowed to persist, the exhaust gas temperature limits may be exceeded (Refer to COMPRESSOR STALL, Section III.)

SPINS

Note

Intentional spins in this airplane are prohibited due to excessive loss of altitude and the probability of exceeding engine temperature limitations.

The possibility of entering inadvertent spins cannot be overlooked in any operational fighter-type airplane, although the pre-spin warnings of the F-102A should greatly reduce the probability of entering a spin. Spin warning characteristics, as in stalls, are airframe buffet, loss of lateral directional control, high rate of sink, and probable engine compressor stalls. As the airspeed is allowed to decrease well below the recommended minimum airspeed or a g load is held through severe buffet, the airplane will fall off laterally in either direction and spin if continued back stick pressure is held. The airplane will spin in an oscillatory manner and may change direction of rotation several times when up elevator and neutral aileron controls are held. If aileron is applied and held in one direction, the airplane will spin in the opposite direction, i.e., left aileron will induce a spin to the right. Rudder deflection at spin entry will determine the initial direction of the spin if applied or held in before the angle of attack has reached 32° nose up.

SPIN RECOVERY

The recommended procedure for recovery from an inadvertent spin is as follows:

1. Throttle IDLE.
2. Release all controls until the nose has dropped to well below the horizon and the rotation has stopped.
3. Apply power and up elevator to complete the recovery only after 130 KIAS has been reached. At this airspeed the elevons and rudder are effective and may be used to maintain desired altitude.

If the rotation continues for more than one turn after release of the controls, the recovery can be effected immediately by applying one-half aileron in the direction of the spin (left aileron to recover from a spin to the left).

Note

- Oscillation in roll may mask the direction of rotation; if the aileron does not cause recovery in one turn, reverse the direction of application.
- As the spin breaks, the nose will pitch down and rotation will stop simultaneously. As soon as this occurs, neutralize aileron to stop roll rate.

After recovery from a spin, during which engine compressor stalls were encountered, return to base at minimum power and land. If the airplane has been spun or fully stalled this fact shall be entered on Form 781 so that the engine can be inspected for damage.

FLIGHT CONTROLS

SPEED BRAKES

The hydraulically operated speed brakes may be used to slow the airplane at all speeds. When extending the speed brakes, a slight nose-up airplane trim change occurs which becomes more pronounced at higher airspeeds. When retracting the speed brakes, a nose-down trim change occurs. Refer to the Appendix for descents with speed brakes and to Section V for speed brakes limitations.

LONGITUDINAL CONTROL FORCES

Stick force is proportional to stick deflection and varies with altitude and Mach number. Caution should be used during accelerated flight conditions as some elevator stick force lightening will occur as speed decreases in the transonic region (Mach .9 to 1.05).

LATERAL CONTROL FORCES

Lateral controls are designed so that stick force is a constant function of stick displacement. However, ailerons are most effective at approximately Mach .9 and decrease in effectiveness with an increase or decrease of speed. Control response near the neutral position is inherently sensitive but not excessive. The new pilot checking out may have a tendency to overcontrol slightly before becoming accustomed to stick feel in the proximity of the neutral position.

RUDDER CONTROL FORCES

The rudder pedal force varies with indicated airspeed. Full rudder deflection is available at approach and taxi speeds with approximately 50 pounds pedal force. At high indicated airspeeds approximately five times this force is required for one-quarter rudder deflection.

LEVEL FLIGHT CHARACTERISTICS

LOW SPEED

The recommended airspeeds for takeoff, approach and landing, and their relationship to gross weight, ambient air temperature and altitude contained in this handbook are above any unsafe flight attitude. Handling characteristics within these speed ranges are excellent and control response is positive. However, high angle of attack, associated with low speeds in the approach and landing configuration, together with the effective control response may mask the development of high sink rates. Since this airplane does not exhibit conventional stalling characteristics, it is possible, in high-altitude flight, particularly in maneuvering, to generate a condition of high drag which is beyond the capability of the engine to overcome. The possibility of entering this condition must be anticipated, since recovery may require more altitude than is available, particularly in the landing approach. Figure 6-1 shows the relationship of rate-of-sink and altitude lost in recovery from descent in the low-speed region. Data are included for initial thrust settings of idle and approach thrust (80-85% rpm) using maximum and military thrust for recovery. Engine thrust is not always a substitute for airspeed, particularly in the low speed range, so do not rely on engine thrust alone to get out of trouble. As is shown in figure 6-1, there is a minimum speed associated with the various airplane and atmospheric conditions of operation. It is very important to obtain at least this speed prior to any high-altitude maneuvering flight. A speed higher than the minimum is advisable at least until more complete familiarity with the airplane characteristics is attained. 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 speeds makes it possible to obstruct forward visibility along the flight path with the nose of the airplane. Refer to Appendix for takeoff and landing speeds.

MEDIUM SPEED

Medium speed flight characteristics are conventional.

HIGH SPEED

Whenever flight into the supersonic region is anticipated, the pitch and yaw damper systems should be engaged to aid in maintaining coordinated flight through the high-speed stages.

CAUTION

- Flight into the transonic and supersonic speed ranges should not be attempted with the yaw damper inoperative. With the yaw damper disengaged, a snaking or oscillating motion will appear in the transonic speed range.
- If flight with the damper systems inoperative is mandatory, refer to OTHER MANEUVERING LIMITATIONS, Section V.

As speed increases through the transonic range and into the supersonic speed range, a nose-down trim change (nose tucks down slightly) occurs. Some stick force lightening may occur when decelerating through the Mach 1.0 and Mach .95 region. Because of the excellent response to aileron action throughout the entire speed range of the airplane, large aileron deflections in the high-speed range should not be attempted until the pilot is thoroughly familiar with the response. When using maximum thrust at low altitude, it is possible to exceed temporary or design speed limitations (refer to Section V).

Note

See figure 6-4 which shows maximum Mach number capabilities.

MANEUVERING FLIGHT

The present-day design requirements for an interceptor are principally for speed with reduced capabilities for maneuvering. With this design the mass of the fuselage as compared to short wingspread creates a new phenomenon. This phenomenon, inertia coupling, is discussed in the following paragraph.

INERTIA COUPLING

The large lateral and normal load factor response during an abrupt aileron roll maneuver has been referred to as the inertial coupling effect. When a maneuver is performed so that a combined rolling and yawing rotational velocity exists, a pitching motion results due to the inertia effects. Similarly a combined pitching and rolling rotational velocity results in a yawing motion. This phenomenon can be illustrated by considering a rod to represent the distributed mass along the length of the fuselage. If the rod is rotating about an axis which is inclined to the axis of the rod at some pitch angle, the centrifugal force or inertia effects will tend to cause the rod to increase this pitch angle (i.e. becomes a fly weight). The aerodynamic characteristic which opposes the increase of this pitch angle is the static stability of the airplane. This effect is present on all airplanes but has become more pronounced on present day design. The automatic stability augmentation system of this airplane provides adequate protection from structural damage provided that the abrupt aileron maneuvers are performed

characteristics at low speeds

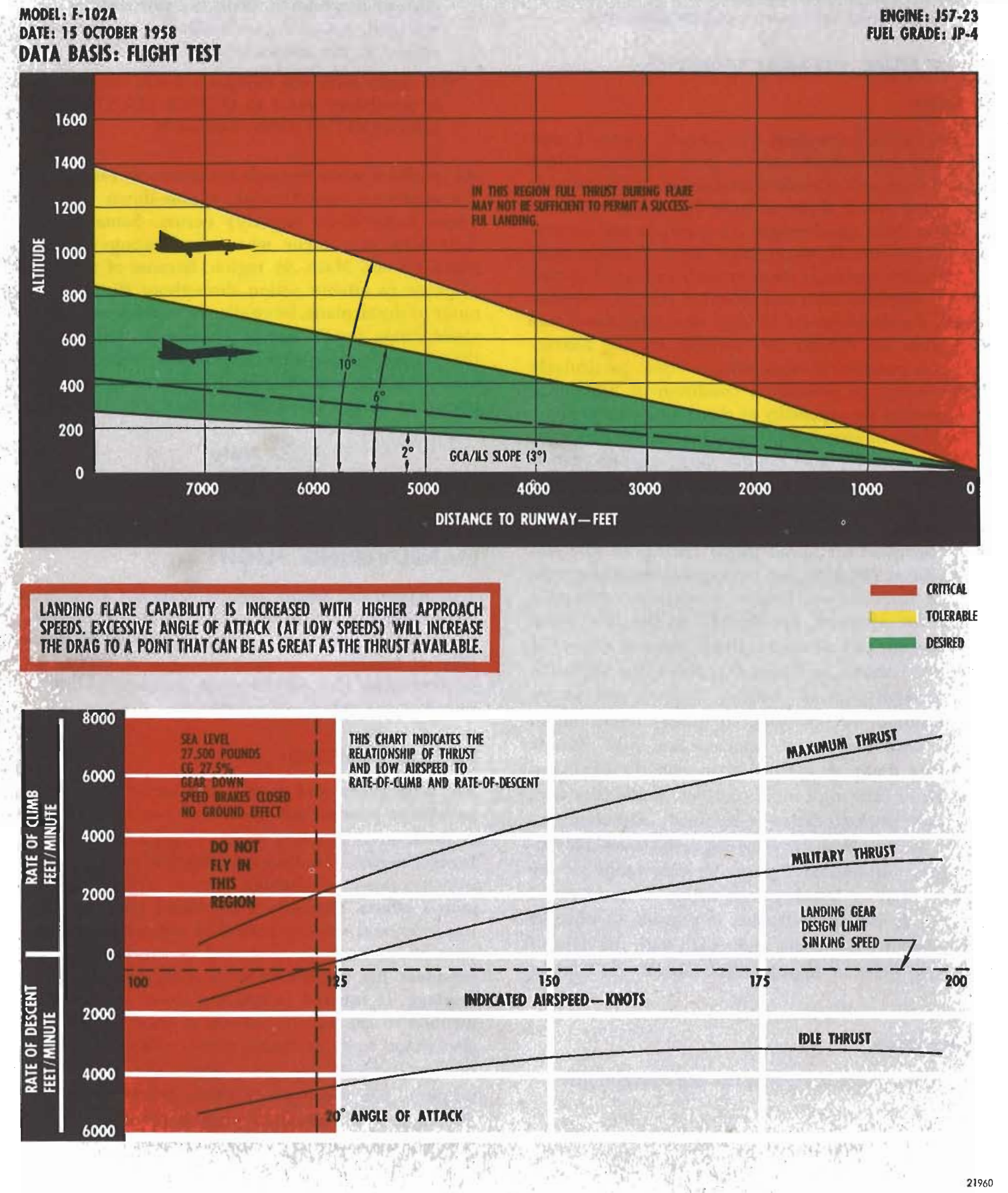


Figure 6-2

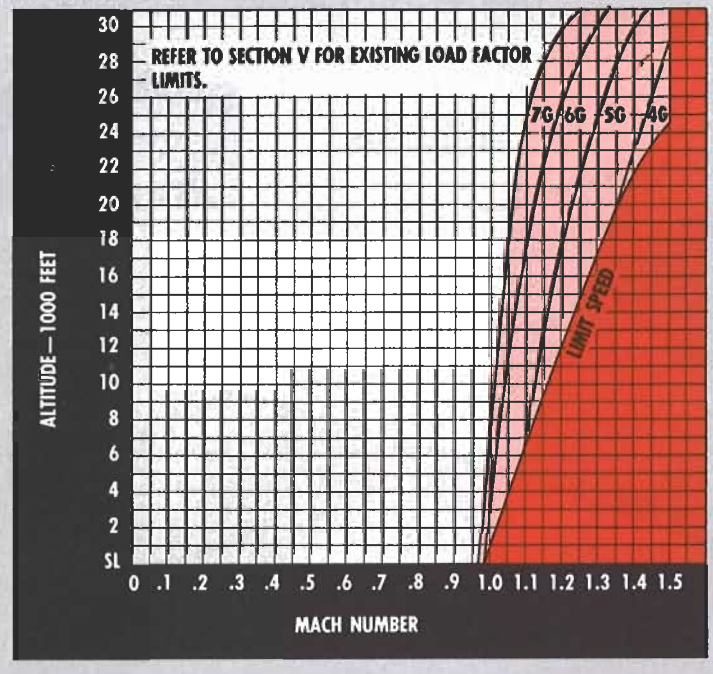
hinge moment limitation

MODEL: F-102A
DATE: 26 MARCH 1957
DATA BASIS: FLIGHT TEST

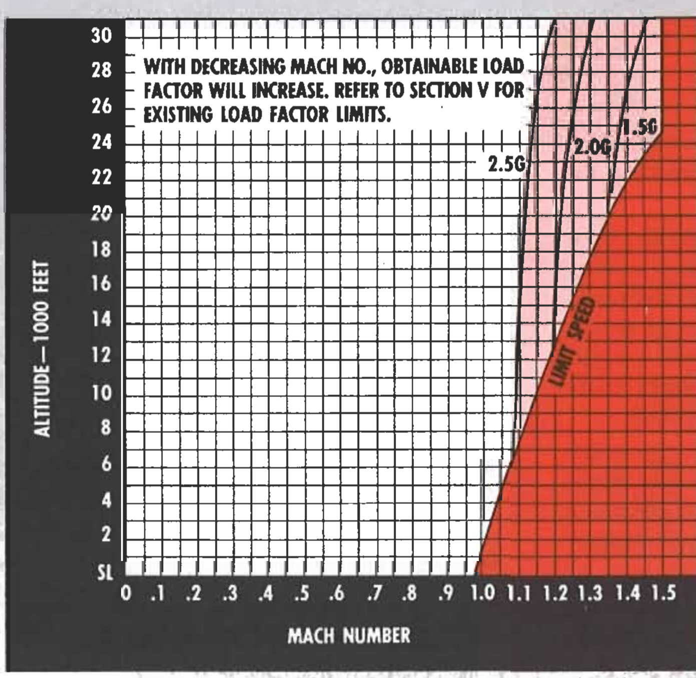
CONFIGURATION: CLEAN
GROSS WEIGHT: 25,000 POUNDS
C.G.: 28.5% M.A.C.

ENGINE: J57-23
FUEL GRADE: JP-4

IDLE THRUST



OBTAINABLE LOAD FACTORS
(BOTH HYDRAULIC SYSTEMS OPERATING)



OBTAINABLE LOAD FACTORS
(SINGLE HYDRAULIC SYSTEM OPERATING)

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Figure 6-3

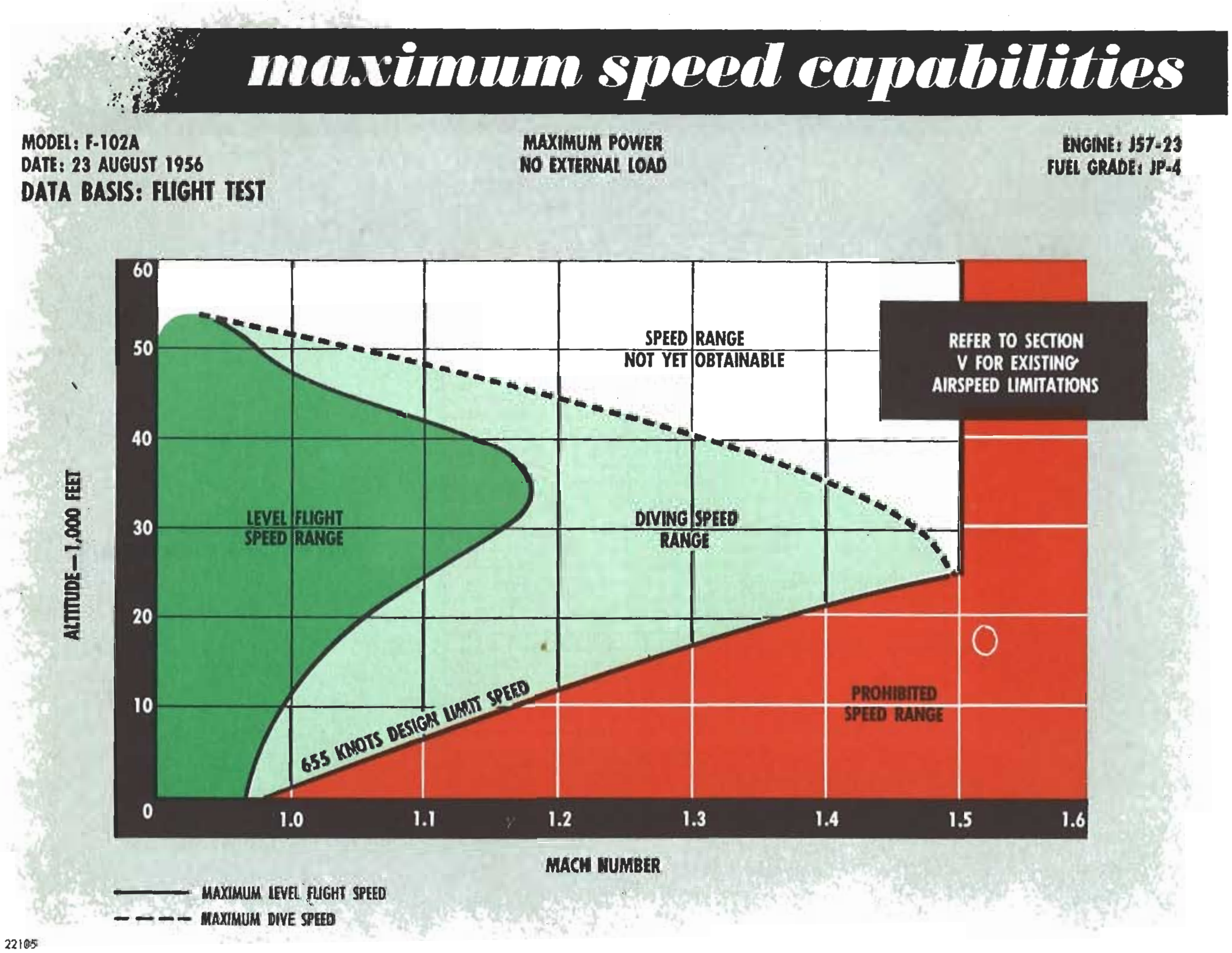


Figure 6-4

in accordance to the operating limitations. However, the following provides additional information regarding the roll behavior of the airplane:

a. Adverse yaw (against the turn) characteristics exist with nose-high angles of attack during rolling maneuvers, whereas complementary (with the turn) yaw characteristics exist with low angles of attack. The crossover between adverse yaw and complementary yaw characteristics for level flight entry conditions occur at approximately 268 KIAS.
b. Application of back stick (up elevator) aggravates the divergence tendency of the adverse yaw flight conditions, whereas forward stick (down elevator) aggravates flight conditions where complimentary yaw characteristics exist.
c. During the critical response roll maneuver, the normal load factor is opposite to the pitch rate, i.e. negative load factor with positive pitch rate or vice versa. Under these conditions the application of elevator governs the pitch rate and can greatly increase the normal load factor.

If limit roll rates are exceeded and inertia coupling is encountered, the recommended control action is to use aileron only to stop the roll rate. Do not attempt to fight the pitch and yaw, but neutralize the elevator and rudder.

DIVES

Diving flight characteristics are similar to high-speed level flight characteristics. A tuck-under tendency noted in the transonic and lower supersonic speed ranges is also present when diving. To minimize the tuck-under when dive speed approaches Mach 1.3, correction should be made with manual trim or back stick application.

DIVE RECOVERY

When diving in the speed range between maximum level flight speed and limit dive speed, a roll-off, generally to the right (approximately 20° per second), will be encountered during recovery below 35,000 feet, when attempting a maximum load factor pullout. The roll-off is caused by some airplane asymmetry which requires aileron trim.

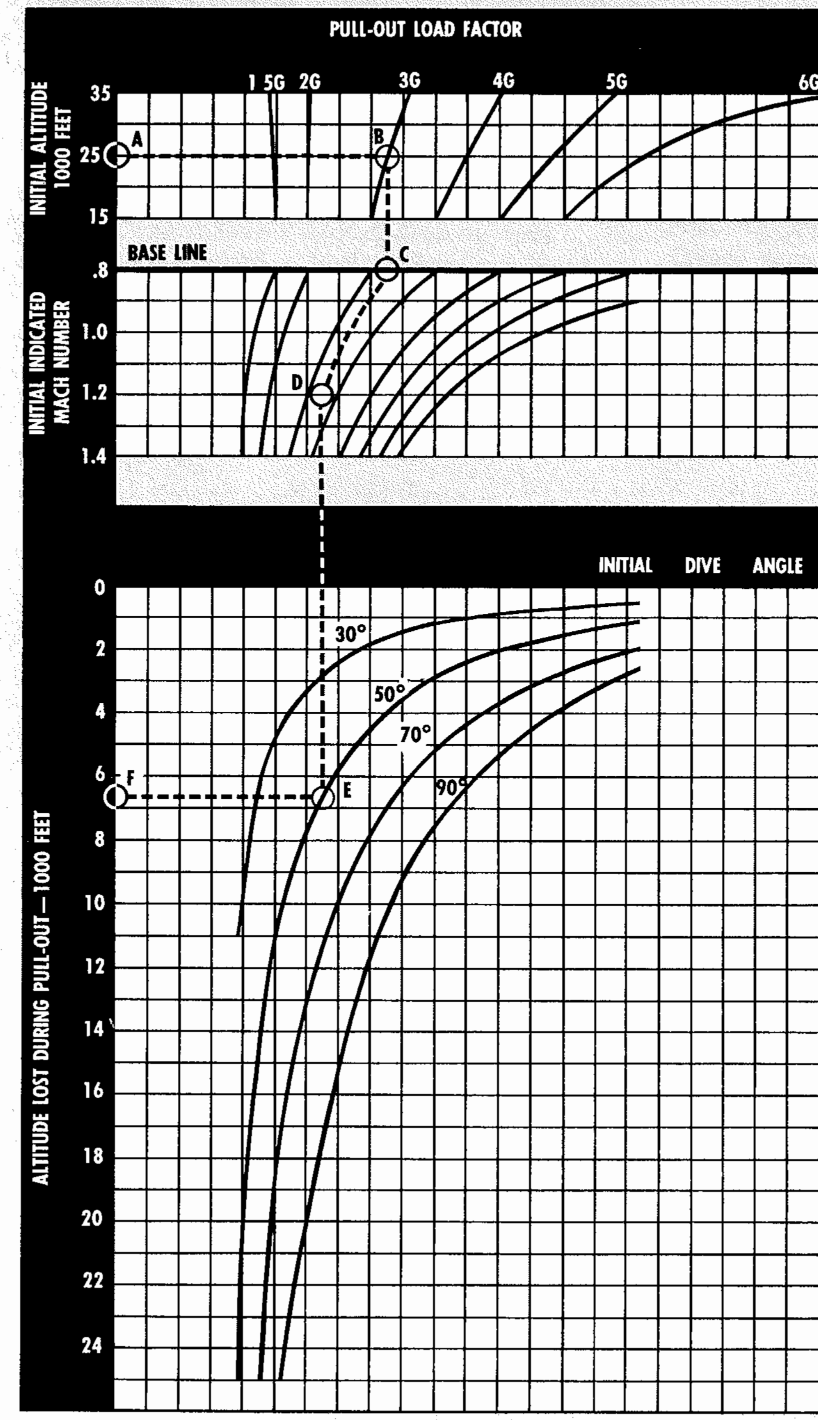
dive recovery chart

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: SPEED BRAKES OPEN
(GROSS WT.: 24,500 LB.)
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4

IDLE THRUST



21959

EXAMPLE

A. IS ALTITUDE AT START OF PULL-OUT (25,000 FEET).
B. IS PULL-OUT LOAD FACTOR (3G).
C. IS MACH NUMBER BASE LINE.
D. IS MACH NUMBER AT START OF PULL-OUT (1.2).
E. IS DIVE ANGLE AT START OF PULL-OUT (50°).
F. IS ALTITUDE LOST DURING PULL-OUT (6,700 FEET).

Figure 6-5

The trim is lost when maximum elevon hinge movement is obtained, resulting in insufficient elevator deflection to develop full g capabilities. See figure 6-3 for maximum load factors obtainable before encountering roll-off. Dive recovery should be executed with caution since an accelerometer is not installed on all airplanes, and no dive should be attempted when pullout would exceed a load factor of 4.0 g's.

Note

In recovery from a supersonic dive, minimum altitude loss is experienced if afterburner is turned off and speed brakes are opened.

FLIGHT WITH EXTERNAL TANKS

General flight characteristics with external tanks installed are basically the same as for the clean airplane. The normal differences associated with increased gross weight will be noted. Takeoff roll will be slightly increased and nose wheel lift-off speed should be a few knots higher. With one tank containing fuel and the other empty, no particular problem should be encountered in flight. Landing with either or both external tanks full or partially full is not recommended, and if attempted, close attention to airplane gross weight and sink rate must be observed. External tanks may be jettisoned at any speed within the tanks-on operating range. Though some lift may be experienced as the tanks release, there is no tendency to pitch-up. If either or both tanks are full when jettisoned, a sudden shock or vibration is experienced within the airplane as a result of the tank ejector mechanism reacting against the inertia of the heavy tank. This is characteristic of the ejector mechanism and should cause no concern.

Note

Refer to EXTERNAL TANKS, Section V, for airspeed and acceleration limitations with external tanks installed.

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COMBUSTION START—SECOND ATTEMPT

If a second attempt to start is to be made on airplanes equipped with a combustion starter and no external power is available, it may be necessary to manually replenish the starter fuel flask. This would be necessary if a false start had occurred. Without external power, ac power is not available and as a result the starter fuel flask would not be automatically replenished. The following procedure should be followed to manually replenish the starter fuel flask:

1. Gain entrance to the fuel flask through the right-hand engine access door.
2. Remove bleed line cap cover from starter flask bleed line.
3. Apply approximately ten psi of air to fuel flask bleed line.
4. Listen for bottoming of the plunger in starter flask.
5. Remove air source and allow fuel to gravity feed from fuel tanks to fill starter fuel flask. Use a suitable container to collect fuel drainage from fuel flask.
6. Install bleed line cap cover.
7. Check for leaks.

ENGINE COMPRESSOR BLEED SYSTEM

An automatically controlled compressor air bleed system on the engine reduces the possibility of compressor instability and resulting compressor stalls due to surges of low-speed compressor airflow acceleration. At times when the forward (low-speed) compressor supplies a greater volume of air than can be used by the aft compressor, air is vented to atmosphere from excess 9th stage air, overboard from the 9th stage manifold through a bleed valve and ducting to openings in the upper fuselage skin. A compressor bleed governor, driven by the low-speed compressor, senses changes both in rotor speed and in air pressure and temperature in the engine compressor inlet. This intelligence is transmitted to a bleed valve actuator which opens or closes the bleed valve, according to information received. The governor is mechanical, and the temperature sensing unit operates through a capillary. The valve actuating mechanism is pneumatic and uses air bled from the 16th stage of the high-speed compressor. The engine compressor bleed system is independent of any of the airplane systems.

FUEL BOOST PUMP LOW FUEL LEVEL OPERATION

On lowering the landing gear or on application of equivalent decelerating force, the fuel in the tanks will momentarily displace forward and unport the aft boost pumps and "T" check valve bell mouth. If neither forward pump is operating in fuel, air will be introduced into the fuel system and result in a flameout. Basically, the fuel system will sustain a constant flow of fuel as long as either forward boost pump is operating in fuel or both "T" check valves are submerged in fuel. If one side has been depleted of fuel as indicated by the low level warning light and the boost pump low-pressure light, the fuel valve to the empty side will be closed. In this condition, the concern is with the forward pump and "T" check valve on the side supplying fuel. In flight, boost pump failures are few but unawareness of probable consequences is a hazard. A boost pump that is a potential failure will normally continue operating satisfactorily until stopped. Once stopped, it may not restart. During descent with low fuel level in the No. 3 tanks, the following is applicable:

1. Whenever a penetration is made with a comparatively low fuel level in the No. 3 tanks, 600 pounds or less per tank, then the positive operation of the forward pumps should be established.
2. If the aft boost pumps are momentarily deactivated and the boost pump low-pressure warning light does not illuminate, this is an indication of proper functioning of respective forward boost pump.
3. If either forward pump is not available, then a relatively nose-high attitude must be maintained.

For additional information on fuel management, refer to Section III.

HIGH-PRESSURE PNEUMATIC CAPABILITY

Basic design concept in providing air supply for starting the engine is to provide a start capability at a remote base where a high-pressure compressor is not available. This concept presumes that a mission has been accomplished (including three armament passes consisting of three door cycles and two launchers cycles), and that the airplane requires only refueling before a return to the prime base, but does not allow for any emergencies or air motoring requirements of the combustion starter. When landing at a remote base where a high-pressure compressor is not available and it is essential that a start be made, air motor is to be minimized under the following conditions:

1. If the airplane has made two or three separate armament passes.
2. If a warning light indicating low pressure in the high-pressure pneumatic system is illuminated.

Note

- To minimize air motoring of the combustion starter during the starting sequence, place the throttle outboard to START, then depress the ignition button and immediately move inboard to the OFF, then IDLE position.
- There is a possibility of air motoring with the airplane's air supply system after two armament passes, but this would be marginal and may result in a "hung" start.
- Do not turn off battery power when landing at a remote base if an air compressor is not available and a combustion start is to be attempted. Do not reset the armament switch until after engine start to prevent the cylinders from refilling with air needed for the combustion starter.

Under all other conditions, air motoring of the combustion starter is mandatory. Given the most extreme conditions of air usage, leakage, and thermal loss, a starter capability should exist from 15 to 45 minutes after touchdown. Under more favorable conditions, this capability should exist over a much longer period due to a usual thermal gain after touchdown. (See figure 7-1.)

Note

- If a combustion start is made using the airplane supply prior to a firing mission, there will not be sufficient air left for armament firing.
- Low pressure in the system will affect rudder feel. Feel will not increase above that normally attained at approximately 380 knots indicated. If faster airspeeds are attained it is recommended that the rudder not be used because rudder feel will give the pilot a misleading indication of the stresses on the tail assembly.

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 ENGINES. 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.

LANDING GEAR

Emergency landing gear extension is accomplished by pulling the landing gear emergency extension handle. When the landing gear emergency extension handle is pulled, electrical power to the normal landing gear handle is shut off; therefore the landing gear handle can

high-pressure pneumatic capability

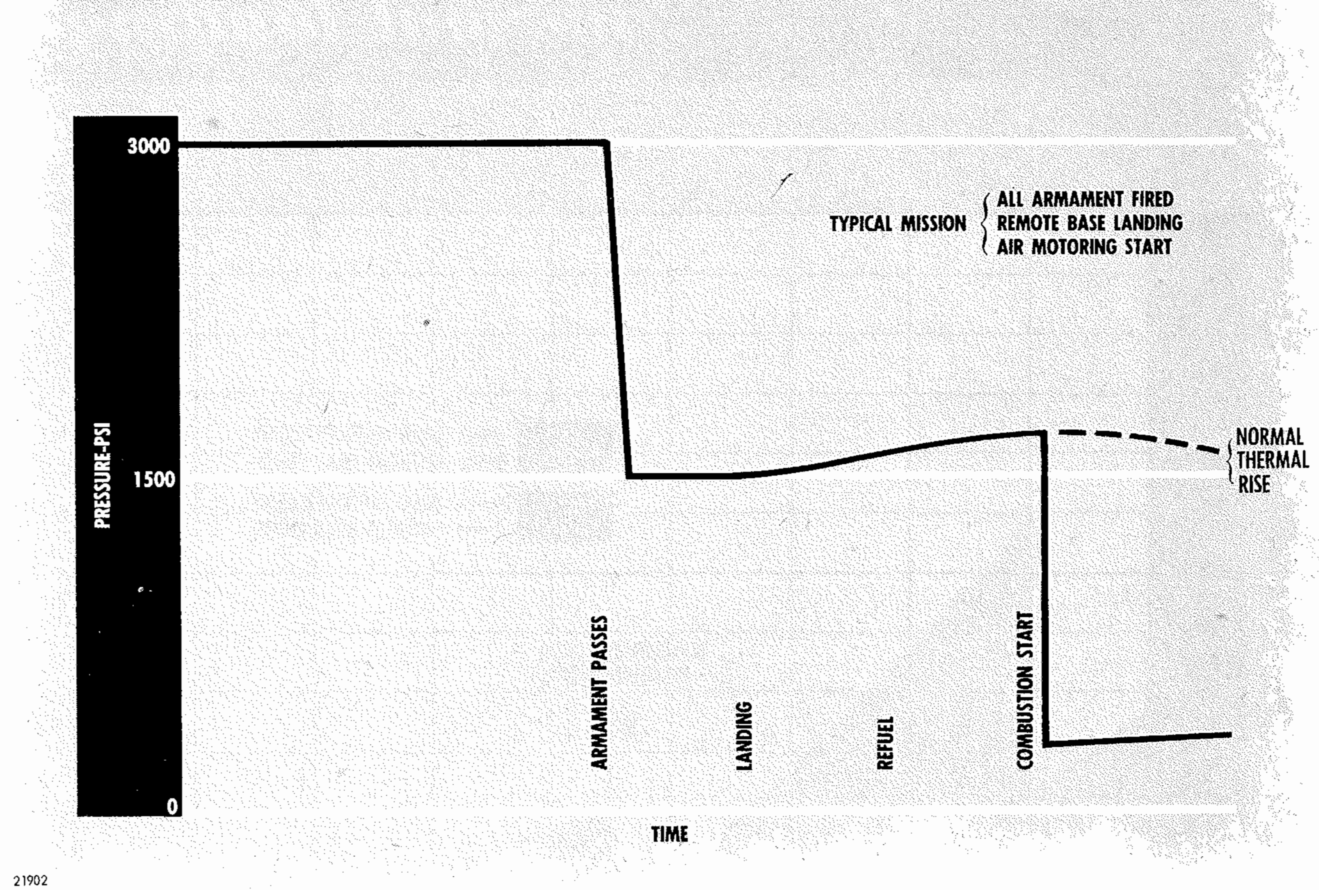


Figure 7-1

be in either the UP or DOWN position whenever the emergency extension handle is pulled, and the gear will remain extended by the emergency system. This action mechanically opens a pneumatic shutoff valve to supply high-pressure pneumatic system pressure to the wheel well door and gear actuating cylinders which will open the doors and extend the gear. The air pressure is routed through cylinder mounted shuttle valves which prevent intermixing of air and hydraulic fluid during extension. A spring clip is installed to lock the landing gear emergency extension handle in the fully pulled position. The emergency extension system should be utilized whenever the normal extension system fails to extend the gear, in event of complete electrical failure, failure of the secondary hydraulic system, or in the event of engine failure. Since the landing gear system operates from the secondary hydraulic system, failure of the primary hydraulic system would not influence normal gear extension; however, using the emergency system with only the primary hydraulic system inoperative may cause damage to the secondary hydraulic system.

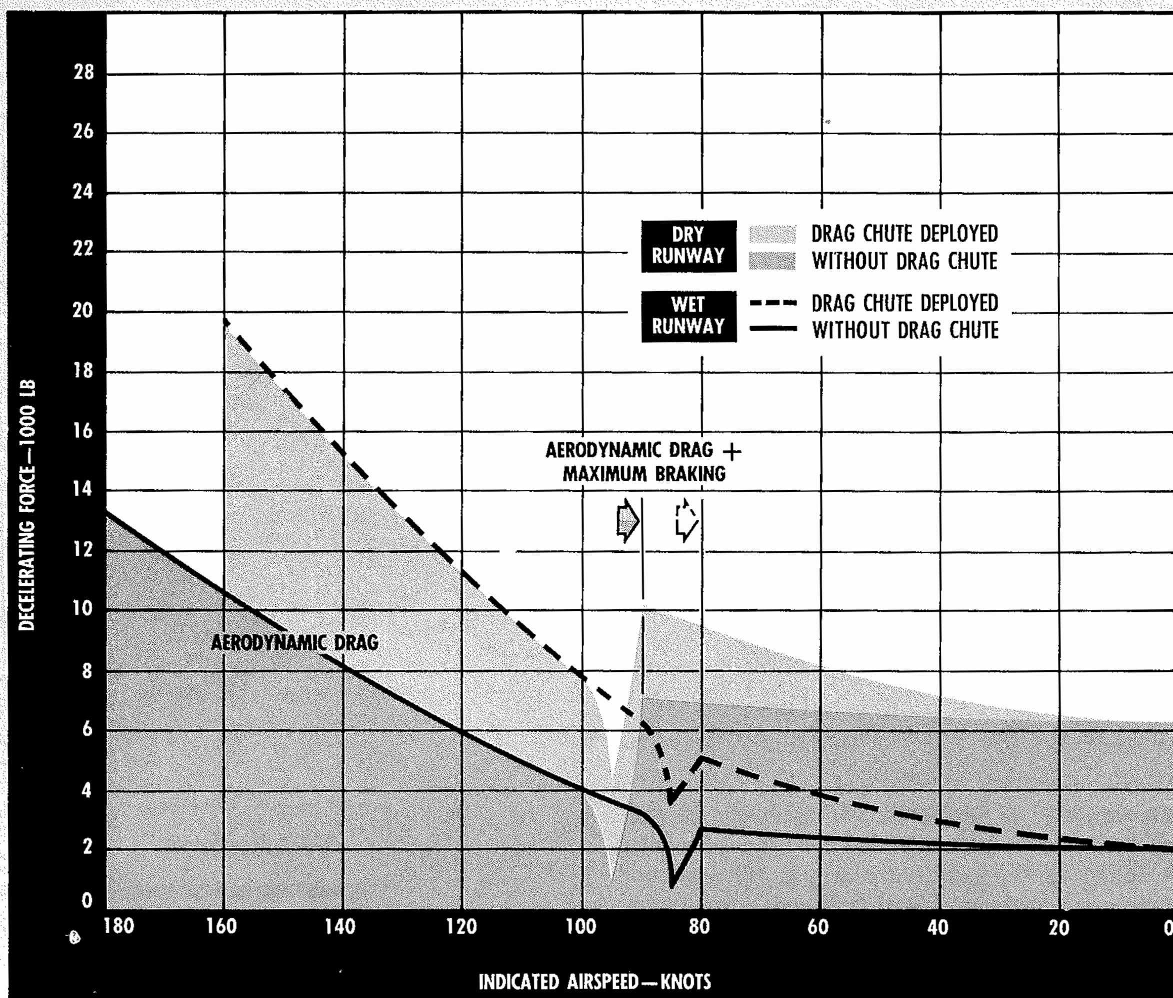
WARNING

- Normal landing gear extension with the primary hydraulic system inoperative should be made at a time when minimum use of flight controls is required.
- In planning the landing when the gear is to be lowered by the emergency system, one minute should be allowed for gear extension time on some airplanes.* On other airplanes** emergency extension should not require more than 10 seconds.

The above considerations would also apply in the event of engine failure even though the engine is windmilling and partial system pressure is available from the primary

**AF 53-1791 thru 56-1518 unless modified by TCTO 1F-102-655.*
***AF 57-770 & on, & airplanes modified by TCTO 1F-102-655.*

landing deceleration chart normal landing



CONDITIONS

- SEA LEVEL
- STANDARD DAY
- NO WIND
- 21,500 LB GROSS WEIGHT
- BRAKING COEFFICIENT OF FRICTION $(\mu) = 0.3$ DRY RUNWAY, 0.1 WET RUNWAY
- NOSE HIGH ATTITUDE TO 100 KNOTS FOR DRY RUNWAY AND TO 90 KNOTS FOR WET RUNWAY
- MAXIMUM BRAKING BEGUN AT 95 KNOTS FOR DRY RUNWAY AND AT 85 KNOTS FOR WET RUNWAY
- SPEED BRAKES OPEN

21920

Figure 7-2

and secondary hydraulic systems. There are no provisions for retracting the landing gear pneumatically; therefore, landing gear retraction should not be attempted following an emergency extension.

USE OF WHEEL BRAKES

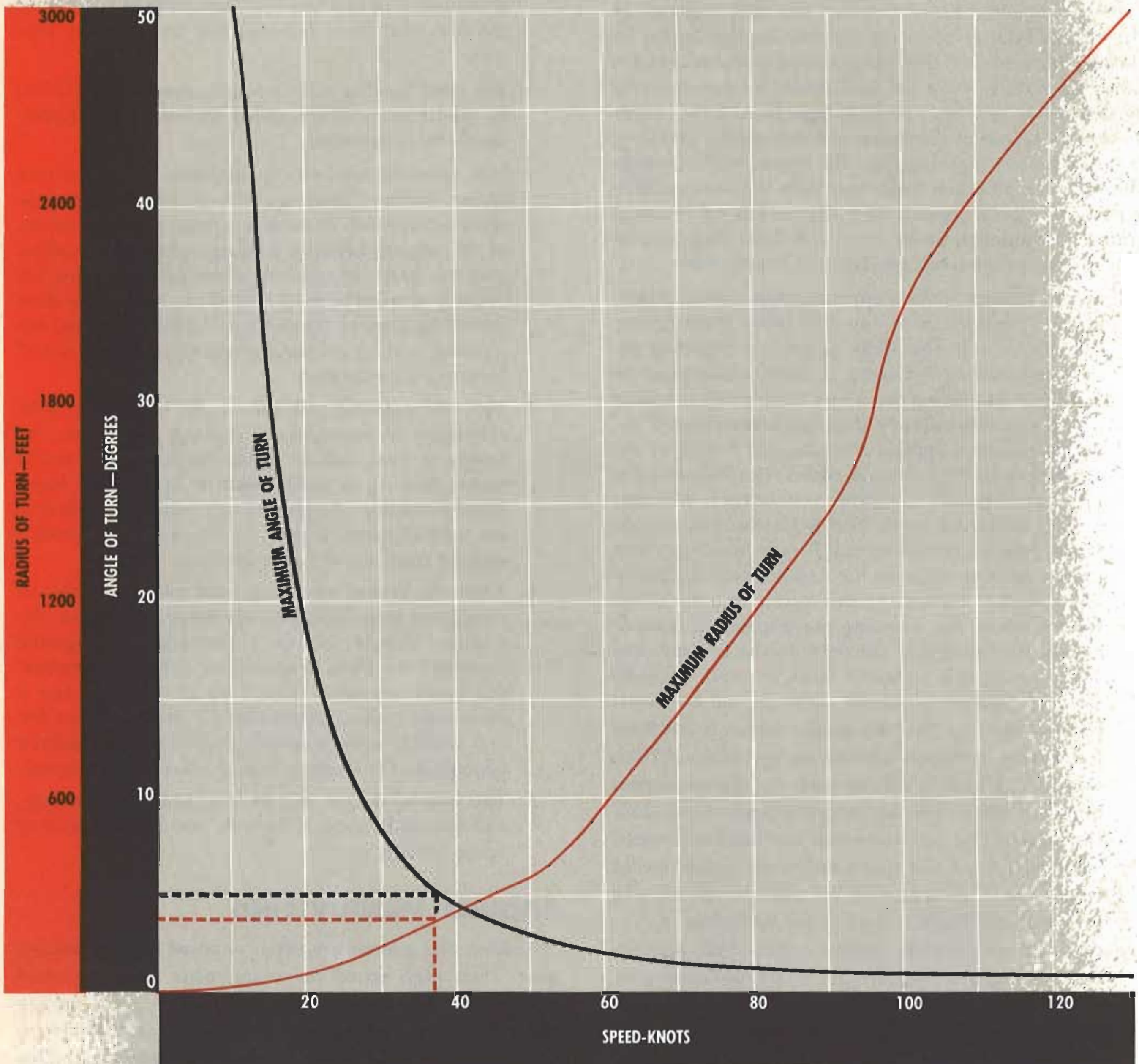
To reduce maintenance difficulties and accidents due to wheel brake failure, it is necessary that airplane brakes be treated with respect. The most common mistakes that are made that reduce brake life and reliability are stopping the airplane as quickly as possible regardless of the length of the runway; use of the brakes consistently for speeding up taxiing turns; and dragging the brakes while taxiing. When applying brakes, there may be a slight time delay between the pedal release and the release of braking action. To minimize brake wear, the following precautions should be observed insofar as is practicable.

1. Use extreme care when applying brakes immediately after touchdown or at any time when there is considerable lift on the wings to prevent skidding the tires and causing flat spots. A heavy brake pressure can result in locking the wheel 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 wheels. A wheel once locked in this manner immediately after touchdown will not become unlocked as the load is increased as long as brake pressure is maintained. Proper braking action cannot be expected until the tires are carrying heavy loads. Brakes, themselves, can merely stop the wheel from turning, but stopping the airplane is dependent on the friction of the tires on the runway. For this purpose, it is easiest to think in terms of coefficient of friction which is equal to the frictional force divided by the load on the wheel. It has been found that optimum braking occurs with approximately a 15 to 20% rolling skid; i.e. the wheel continues to rotate but has approximately 15 to 20% slippage on the surface so that the rotational speed is 80 to 85% of the speed which the wheel would have were it in free roll. As the amount of skid increases beyond this amount, the coefficient of friction decreases rapidly so that with a 75% skid the friction is approximately 60% of the optimum and, with a full skid, becomes even lower. There are two reasons for this loss in braking effectiveness with skidding. First, the immediate action is to scuff the rubber, tearing off little pieces which act almost like rollers under the tire. Second, the heat generated starts to melt the rubber and the molten rubber acts as a lubricant. If one wheel is locked during application of brakes, there is a very definite tendency for the airplane to turn away from that wheel, and further application of brake pressure will offer no corrective action. Since the coefficient of friction goes down when the wheel begins to skid, it is apparent that a wheel, once locked, will never free itself until brake pressure is reduced so that the braking effect on the wheel is less than the turning moment remaining with the reduced frictional force.
2. If maximum braking is required after touchdown, lift should first be decreased as much as possible by dropping the nose before applying brakes. This procedure will improve braking action by increasing the frictional force between the tires and the runway.
3. For short landing rolls, a single, smooth application of the brakes with constantly increasing pedal pressure is most desirable.
4. It is recommended that a minimum of 15 minutes elapse between landings where the landing gear remains extended in the slip stream, and a minimum of 30 minutes between landings where the landing gear has been retracted, to allow sufficient time for cooling between brake applications. Additional time should be allowed for cooling, if brakes are used for steering, cross-wind taxiing operation, or a series of landings is performed.
5. The full landing roll should be utilized to take advantage of aerodynamic braking and to use the brakes as little and as lightly as possible. Aerodynamic braking is most effective during the high-speed portion of the ground roll, when wheel brakes are least effective. Figure 7-2 illustrates the benefits derived from aerodynamic braking.
6. After the brakes have been used excessively for an emergency stop and are in the heated condition, the airplane should not be taxied into a congested parking area. Peak temperatures occur in the wheel and brake assembly from 5 to 15 minutes after a maximum braking operation. To prevent brake fire and possible wheel assembly explosion, the specified procedures for cooling brakes should be followed.
7. The brakes should not be dragged when taxiing, and should be used as little as possible for turning while taxiing.

GROUND MANEUVERING

Excessive side g-loads are often imposed on the landing gear when high speed turns are made during ground operation. This occurs most frequently while taxiing prior to takeoff and during the landing roll. Excessive side g-loads cause undue strain and eventual damage to the landing gear. As indicated in figure 7-3, the maximum permissible angle of turn decreases and the radius of turn increases sharply at higher speeds. As an example, at 15 knots the maximum angle of turn is 30 degrees and the radius of turn 35 feet. With an increase of speed to 37 knots for the same condition, the angle of turn is reduced to five degrees and the radius of turn increases to 230 feet. Because of the unreliability of the airspeed indicator at slow speeds, the airplane should be slowed down as much as possible prior to turning off the active

ground maneuvering chart



EXAMPLE:
(SEE DOTTED LINE)
KNOTS: 37
ANGLE OF TURN: 5 DEGREES
RADIUS OF TURN: 230 FEET

21917

Figure 7-3

runway after landing. As an added precaution, it is recommended the last taxi way be used whenever possible. This will decrease tendencies of making high-speed turns that would subject the landing gear to excessive side g-loads. Considerable runway is required to reduce speed sufficiently for making turns. Under certain conditions with maximum braking first applied at touchdown and with no wind effect, approximately 2100 feet is required to slow the airplane from 134 knots to 30 knots (134 knots is used as touchdown speed in this example) with the drag chute deployed, and approximately 3255 feet without drag chute assistance. More than half the landing roll with drag chute, and two-thirds without drag chute, is at speeds of 90 knots or more. It can be seen that much less braking force can be applied at high speeds, and that considerable runway is used before braking is most effective. For information concerning proper use of wheel brakes, refer to USE OF WHEEL BRAKES, this Section.

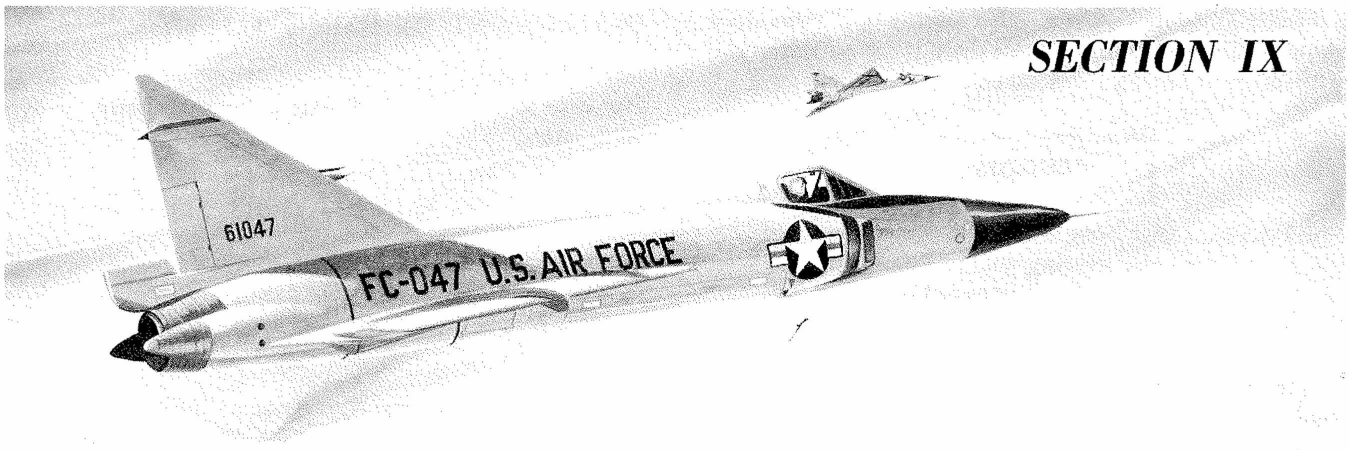
PARTIAL PRESSURE SUIT DEPRESSURIZATION

When the partial pressure suit is pressurized during ground check or inadvertently due to a crash landing or malfunction at low altitude, the pressure should always be released from the helmet before it is released from the suit. This can best be accomplished by opening (unlocking) the faceplate. The pressure is released simultaneously from helmet and suit following use of the test button on the kit and on some types of preflight test consoles. With the MC-3 and MC-4 suits, there is little immediate difficulty if pressure is first released from the suit, but with previous partial pressure suits, lung damage can result. It is the responsibility of each pilot who wears an altitude suit to be thoroughly familiar with donning and connecting his suit, conducting a preflight inflation with available testers, and being thoroughly familiar with the operation and care of his particular type suit.

SECTION VIII—CREW DUTIES

Not applicable to this airplane.

SECTION IX



all weather operation

21984

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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. Any discussion relative to systems operations is covered in Section VII.

INSTRUMENT FLIGHT PROCEDURES

These procedures and techniques pertain primarily to instrument flight conditions and are in addition to normal procedures. Because navigation facilities and terrain vary at each base, this information is intended to serve as a guide to commanders in establishing instrument flight procedures. This airplane is capable of supersonic speeds during instrument flight conditions, thus demanding a relatively high degree of instrument proficiency and conscientious preflight planning. Fuel requirements for completion of instrument letdown, approach procedures, and possible diversion to alternate bases are much greater than for VFR flights and must be included in preflight planning. Only the essential navigation equipment is installed since ground-controlled guidance replaces some of the more conventional navigation aids found in most other types of airplanes. With the existing equipment, three types of instrument approaches may be made; ILS, GCA, VOR, and Tacan (some airplanes).

INSTRUMENT TAKEOFF

Complete the normal TAXI and BEFORE TAKEOFF checks as prescribed in Section II and check pitot heat ON. After aligning airplane visually with runway center line, check directional indicator (RMI) and standby magnetic compass against runway heading. Set attitude indicator at 5° nose-down. 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 heading with nose wheel steering until rudder becomes effective at approximately 80 knots. The directional indicator is primarily for directional control, but reference should be made to runway centerline and runway lights if possible. Beginning at 125 KIAS apply back pressure and rotate the airplane from 5° nose-down pitch attitude to 15° nose-up pitch on the attitude indicator. Airplane will become airborne with pitch change. Raise landing gear when a positive climb indication is noted. Maintain approximately 12° pitch attitude on the attitude indicator until a positive climb is indicated on the vertical velocity indicator. Maintain 12° pitch attitude on the attitude indicator and a positive climb on the vertical velocity indicator until the climb schedule is reached.

Note

In maximum thrust takeoffs, the altimeter and vertical velocity indicator may lag or even indicate a descent in altitude just before breaking ground and during the initial climbout. This altimeter error is the result of disturbed pressure ahead of the airplane due to acceleration and high angle of attack. The altimeter will indicate correctly after the airplane reaches approximately 300 feet of altitude.

INSTRUMENT CLIMB

If the tactical situation permits, use military climb schedule when turbulence or rain are encountered. Upon reaching climb schedule, increase pitch attitude to approximately 18° on the attitude indicator.

CAUTION

Afterburner climb through rain can result in damage to the radome and wing fences. This schedule also results in a high pitch attitude for instrument flight.

INSTRUMENT CRUISING FLIGHT

Keep the airplane trimmed for straight and level flight. For ease and precision of flight, limit all turns to 30° bank angle.

RADIO AND NAVIGATION EQUIPMENT

Refer to Section IV for radio and navigation equipment installed.

DESCENT

The optimum thrust for fuel economy during descents is idle. Instrument descents can be flown without difficulty at any speed; though for maximum ease of handling, a constant-speed letdown (275 KIAS) is recommended.

WARNING

Steep descents, high indicated Mach numbers, and high angles of bank should be avoided to maintain positive control of the airplane at all times. Limiting bank angle to 30° at all altitudes and rates of descent is recommended.

RADAR RECOVERY

Radar recovery can be accomplished with a minimum amount of fuel and time. Radar recovery procedure to be used following interceptions in IFR conditions should be practiced in VFR and IFR weather to develop and improve the teamwork of the pilot and radar controller. When a GCA or ILS approach is required, the descent from the inbound cruising altitude should be started at a sufficient distance to permit a straight-in at the recommended airspeed plus three to four miles for deceleration and changing to approach configuration before turn-on point (gate) to final approach. See figure 9-3 for recommended procedures.

JET PENETRATION

Omni penetrations can be made by using various techniques. For ease of operation, letdowns using 85% rpm, 275 KIAS, gear up, and speed brakes out are recommended. The exact procedure for jet penetrations in the "Terminal Flight Information (High Altitude)" will vary at different bases because of terrain differences, airway locations, and conflicting traffic control zones. Consequently, fuel requirements may vary as conditions vary. Letdown procedure at destinations should be carefully checked and fuel allowances made a part of the preflight planning. See figures 9-1 and 9-2 for the recommended techniques for typical penetration procedures.

Note

If weather is below minimum for an omni low approach, request ILS or GCA prior to beginning letdown. Make decision to proceed to alternate while still at altitude, if possible.

HOLDING

Based upon minimum fuel consumption consistent with ease of handling, the holding data are as follows:

ALTITUDE	KIAS	% RPM	FUEL FLOW (lbs/hr)	MAXIMUM BANK ANGLE
40,000	232	88 ± 2	2200-2500	20°
20,000	225	85 ± 2	2400-2750	30°

INSTRUMENT APPROACHES

The airplane has good handling characteristics during instrument approaches. At low rpm and low indicated airspeeds, response to throttle movement and acceleration is slow. Therefore, use relatively high thrust settings in the approach configuration and maintain at least 170 KIAS. In rain or snow, use of the windshield rain clearing system will greatly increase forward visibility. If it is necessary to execute a missed approach, military thrust should be added and speed brakes closed.

Note

Afterburner should be used only if necessary, because of high fuel consumption rate.

Raise the gear when a definite climb has been established and maintain at least 200 KIAS. Execute the established missed approach procedure as published in the "Terminal Flight Information (High Altitude)" or as directed by the GCA controller.

TACAN

Establish recommended final approach airspeed with gear extended after reaching the gate. Thrust required for level flight is approximately 85% rpm. Upon reaching the gate, extend speed brakes and descend to minimum altitude.

RADIO APPROACHES

Normally an omni approach would be required only if the airplane is not VFR after reaching minimum penetration altitude and no GCA or ILS is available. Refer to the "Terminal Flight Information (High Altitude)" for the local procedures for the standard jet instrument approach.

GCA APPROACH

GCA procedures vary at each base due to location of radio fixes and local terrain. Fuel may be conserved by waiting until final approach before using speed brakes and lowering landing gear. In heavy precipitation it is difficult for the GCA operators to keep the airplane visible in the precipitation clutter on the radar scope. This may result in initiating a missed approach.

ILS APPROACH

Check the "Terminal Flight Information (High Altitude)" for local procedures. During an ILS approach, maintain airplane attitude with the basic flight instruments and monitor the ILS indicator for reference to the localizer and glide-slope. If localizer interception is from a radio fix or GCI control and is at an angle of 90° or less, allow sufficient distance out to perform final cockpit check, slow to approach speed, and establish correct beam heading. See figure 9-4.

WARNING

During an ILS approach, if disappearance of the glide-slope indicator warning "OFF" flag is delayed beyond normal expectation, it may be an indication that electrical power to the instrument selector relays is lost and the CDI information is being derived from a Tacan station. At the start of an ILS approach where the CDI warning "OFF" flag has disappeared but the glide-slope warning "OFF" flag is still visible, or at a locally prescribed check point, check to determine if there has been power loss to the relays by turning the bearing selector knob a few degrees away from the inbound heading. If the CDI responds to the bearing change, the signal was being received from a Tacan station and not the localizer. If the CDI does not respond to the inbound bearing change, the signal was being received from the localizer. After both warning "OFF" flags have disappeared, a subsequent power loss to the instrument selector relays will be detected by the horizontal needle warning "OFF" flag appearing and the horizontal needle will center itself and remain centered regardless of airplane movement.

AUTOMATIC APPROACH (AILAS)

Automatic approaches (see figure 9-4) are flown by using the automatic flight control system (AFCS). For detailed operating instructions of the automatic flight control system, refer to Section IV. Refer to Section V for AILAS mode limitations. Use the following procedures for an automatic approach:

1. Radar master switch—STBY. This positions radar antenna downward clear of the ILS glide-slope antenna.

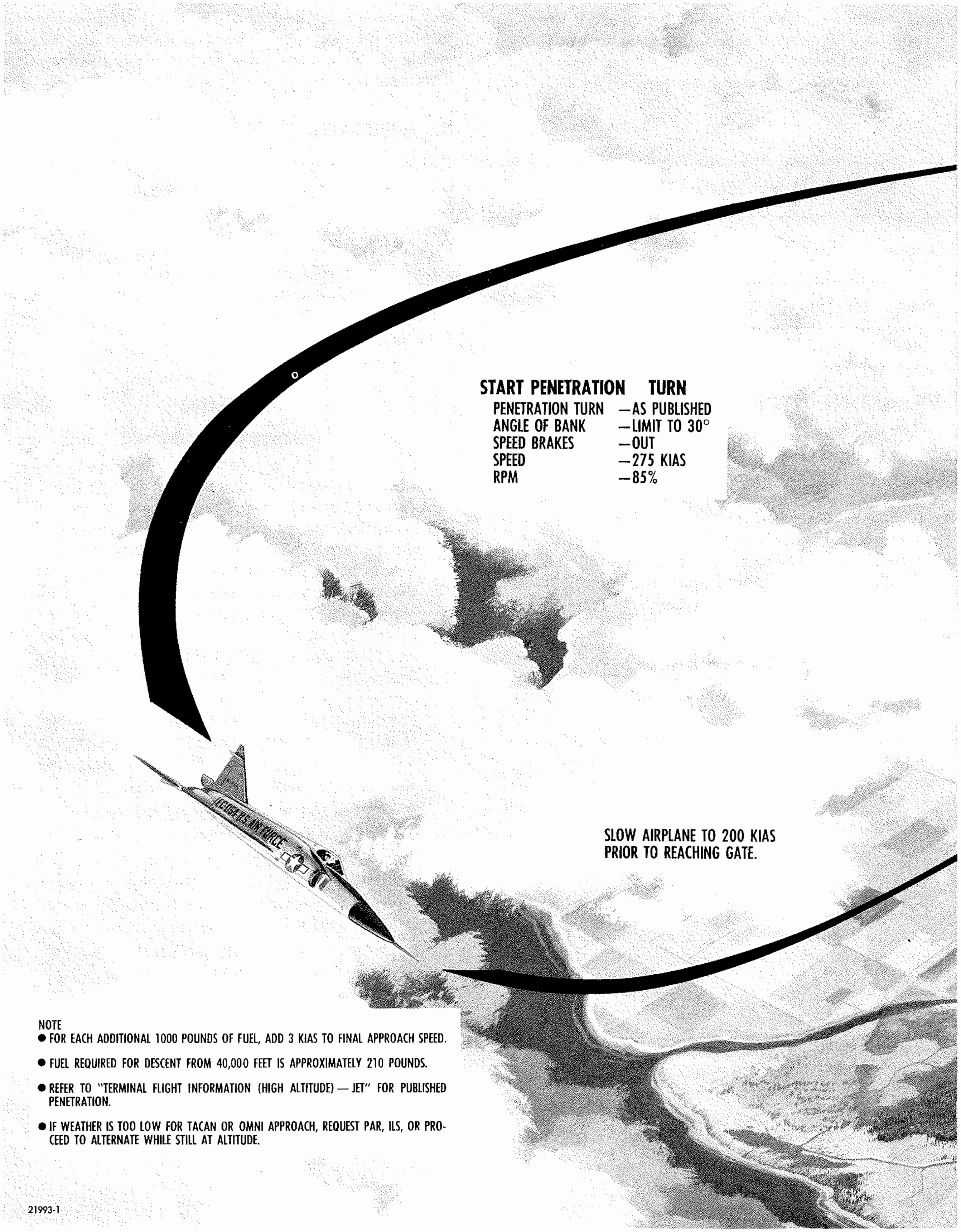
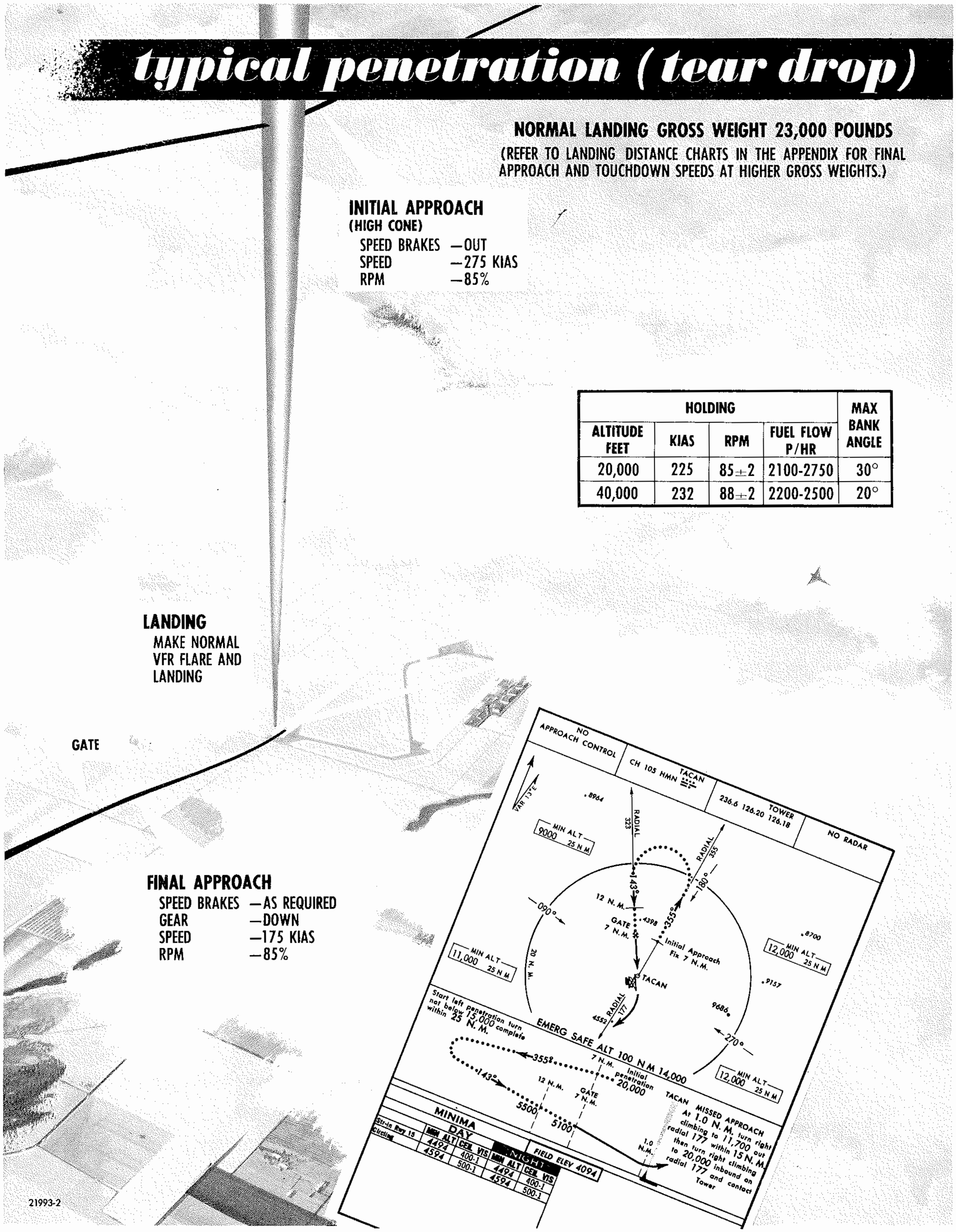
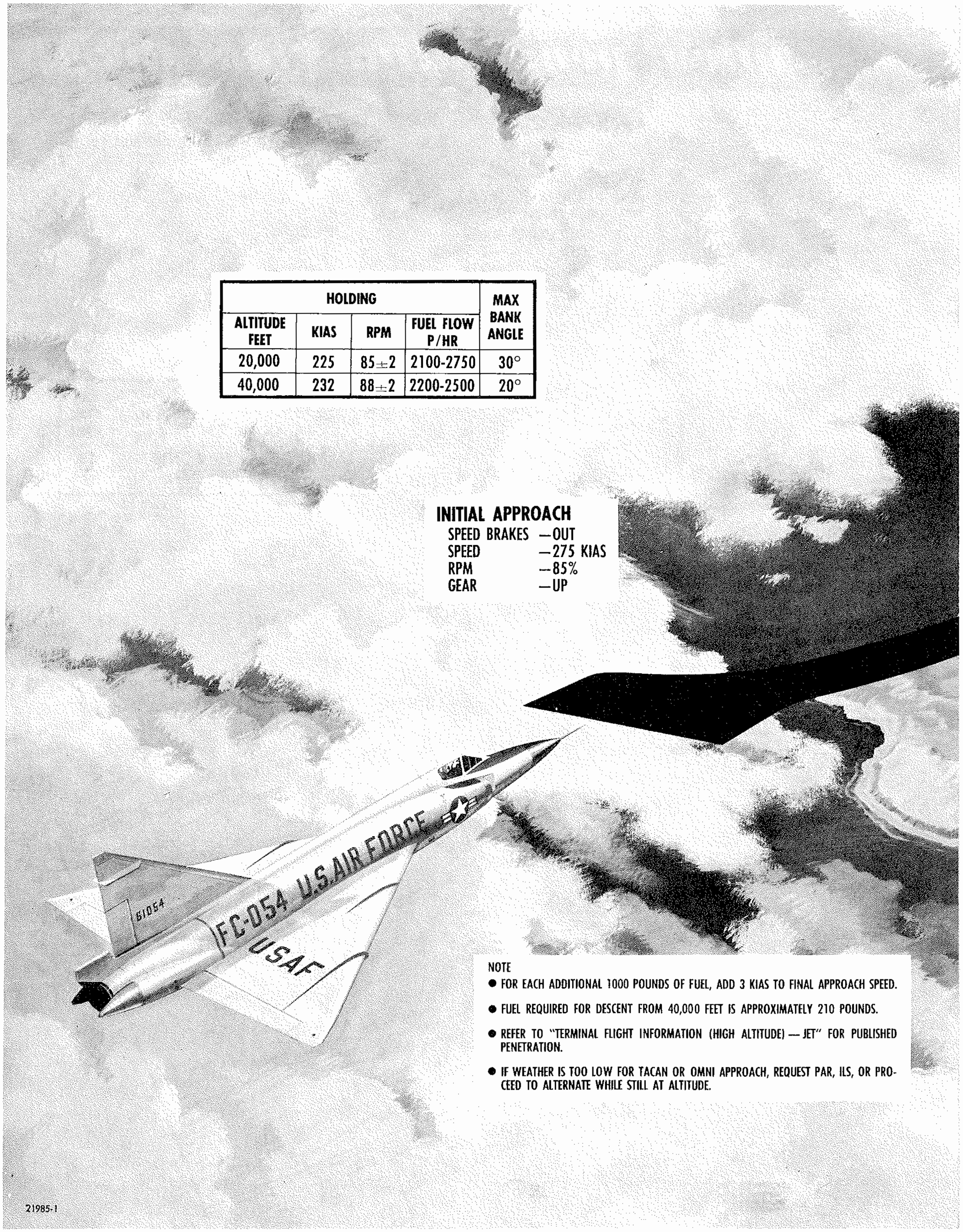

START PENETRATION TURN
PENETRATION TURN —AS PUBLISHED
ANGLE OF BANK —LIMIT TO 30°
SPEED BRAKES —OUT
SPEED —275 KIAS
RPM —85%
SLOW AIRPLANE TO 200 KIAS PRIOR TO REACHING GATE.
NOTE
● FOR EACH ADDITIONAL 1000 POUNDS OF FUEL, ADD 3 KIAS TO FINAL APPROACH SPEED.
● FUEL REQUIRED FOR DESCENT FROM 40,000 FEET IS APPROXIMATELY 210 POUNDS.
● REFER TO "TERMINAL FLIGHT INFORMATION (HIGH ALTITUDE) — JET" FOR PUBLISHED PENETRATION.
● IF WEATHER IS TOO LOW FOR TACAN OR OMNI APPROACH, REQUEST PAR, ILS, OR PROCEED TO ALTERNATE WHILE STILL AT ALTITUDE.
21993-1

Figure 9-1

typical penetration (tear drop)



HOLDING				MAX BANK ANGLE
ALTITUDE FEET	KIAS	RPM	FUEL FLOW P/HR	
20,000	225	85±2	2100-2750	30°
40,000	232	88±2	2200-2500	20°



HOLDING				MAX BANK ANGLE
ALTITUDE FEET	KIAS	RPM	FUEL FLOW P/HR	
20,000	225	85±2	2100-2750	30°
40,000	232	88±2	2200-2500	20°

INITIAL APPROACH

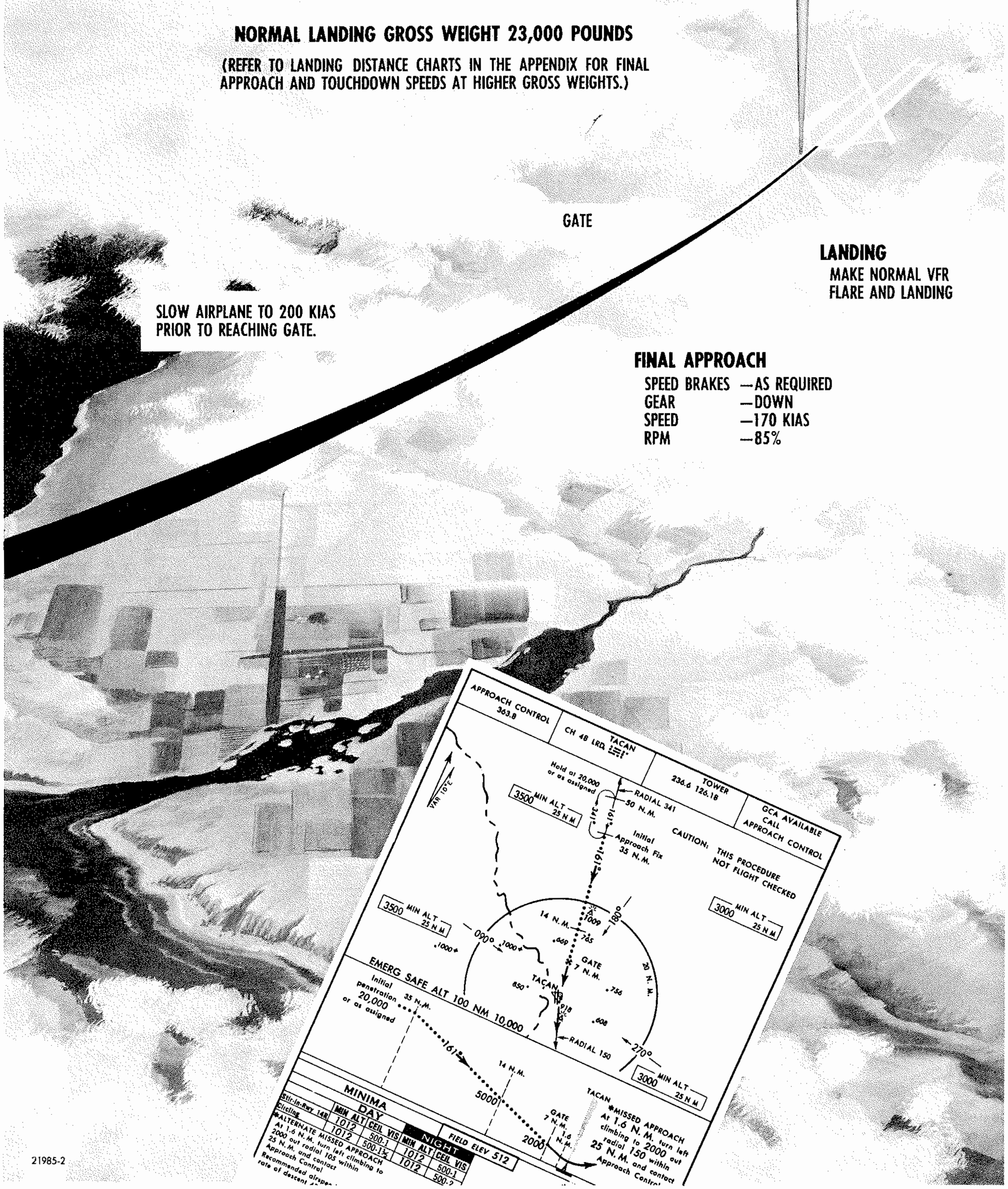
SPEED BRAKES —OUT
SPEED —275 KIAS
RPM —85%
GEAR —UP

NOTE

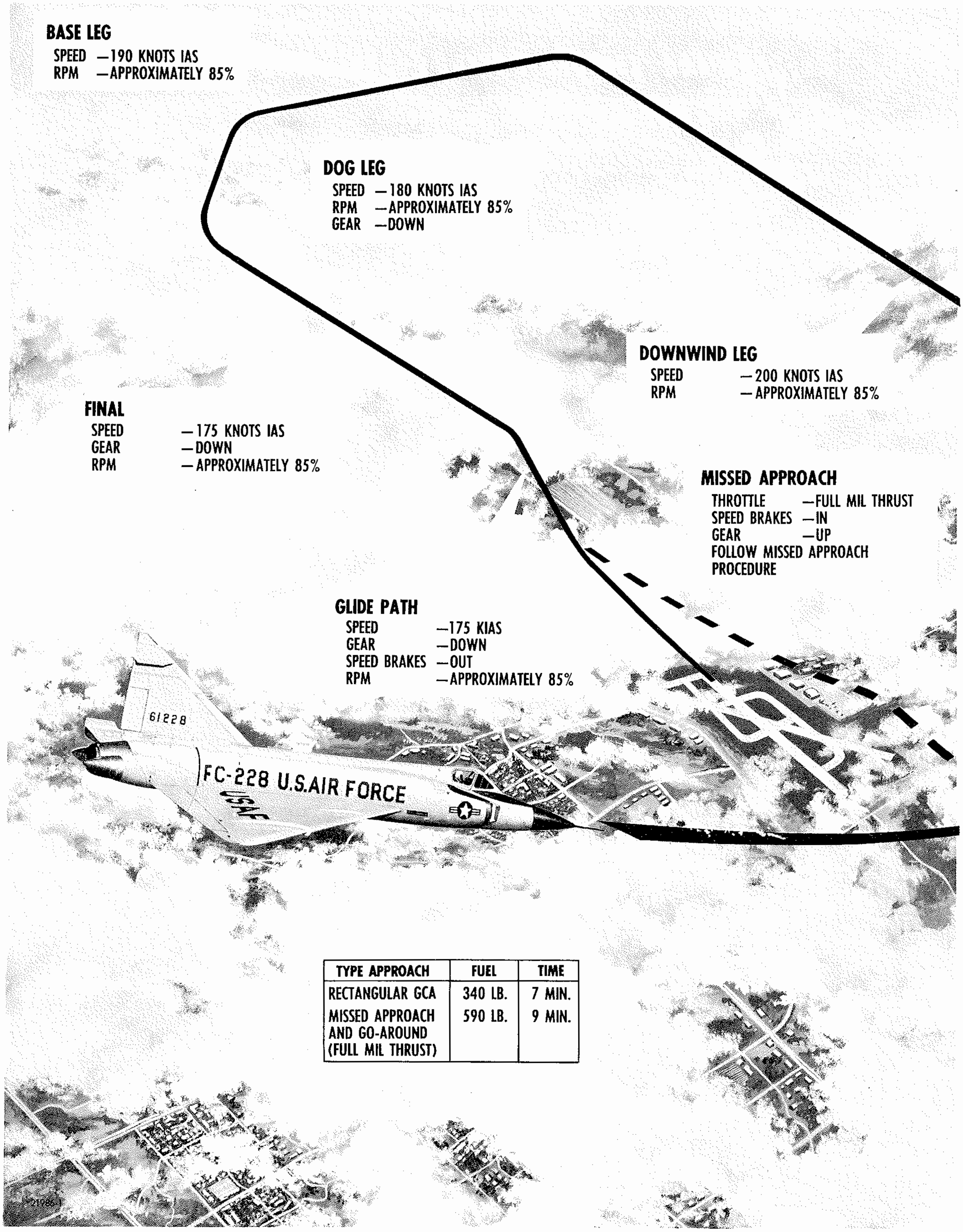
- FOR EACH ADDITIONAL 1000 POUNDS OF FUEL, ADD 3 KIAS TO FINAL APPROACH SPEED.
- FUEL REQUIRED FOR DESCENT FROM 40,000 FEET IS APPROXIMATELY 210 POUNDS.
- REFER TO "TERMINAL FLIGHT INFORMATION (HIGH ALTITUDE) — JET" FOR PUBLISHED PENETRATION.
- IF WEATHER IS TOO LOW FOR TACAN OR OMNI APPROACH, REQUEST PAR, ILS, OR PROCEED TO ALTERNATE WHILE STILL AT ALTITUDE.

Figure 9-2

typical penetration (straight in)

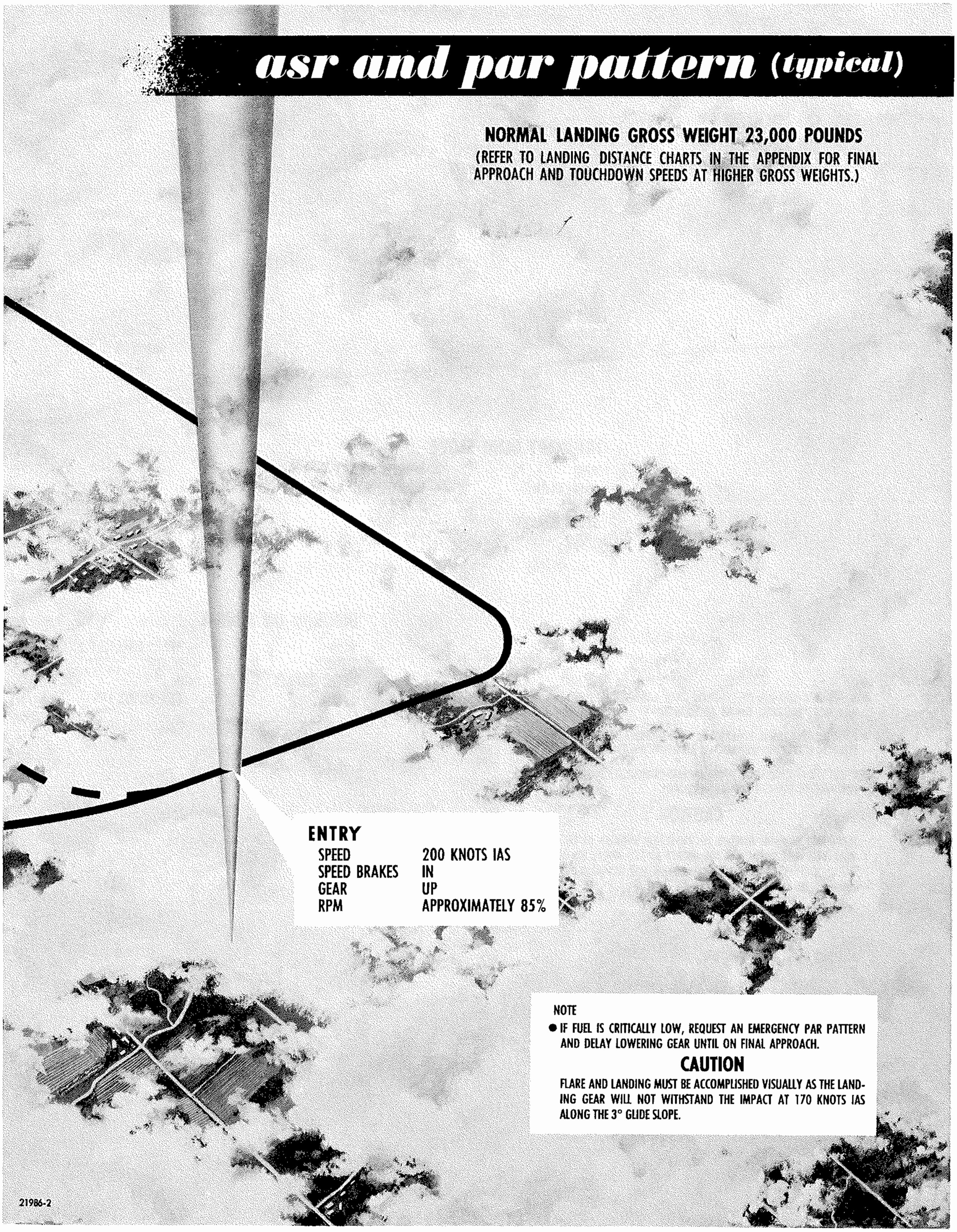


21985-2



TYPE APPROACH	FUEL	TIME
RECTANGULAR GCA	340 LB.	7 MIN.
MISSED APPROACH AND GO-AROUND (FULL MIL THRUST)	590 LB.	9 MIN.

Figure 9-3


asr and par pattern (typical)
NORMAL LANDING GROSS WEIGHT 23,000 POUNDS
(REFER TO LANDING DISTANCE CHARTS IN THE APPENDIX FOR FINAL APPROACH AND TOUCHDOWN SPEEDS AT HIGHER GROSS WEIGHTS.)
ENTRY
SPEED 200 KNOTS IAS
SPEED BRAKES IN
GEAR UP
RPM APPROXIMATELY 85%
NOTE
● IF FUEL IS CRITICALLY LOW, REQUEST AN EMERGENCY PAR PATTERN AND DELAY LOWERING GEAR UNTIL ON FINAL APPROACH.
CAUTION
FLARE AND LANDING MUST BE ACCOMPLISHED VISUALLY AS THE LANDING GEAR WILL NOT WITHSTAND THE IMPACT AT 170 KNOTS IAS ALONG THE 3° GLIDE SLOPE.
21986-2

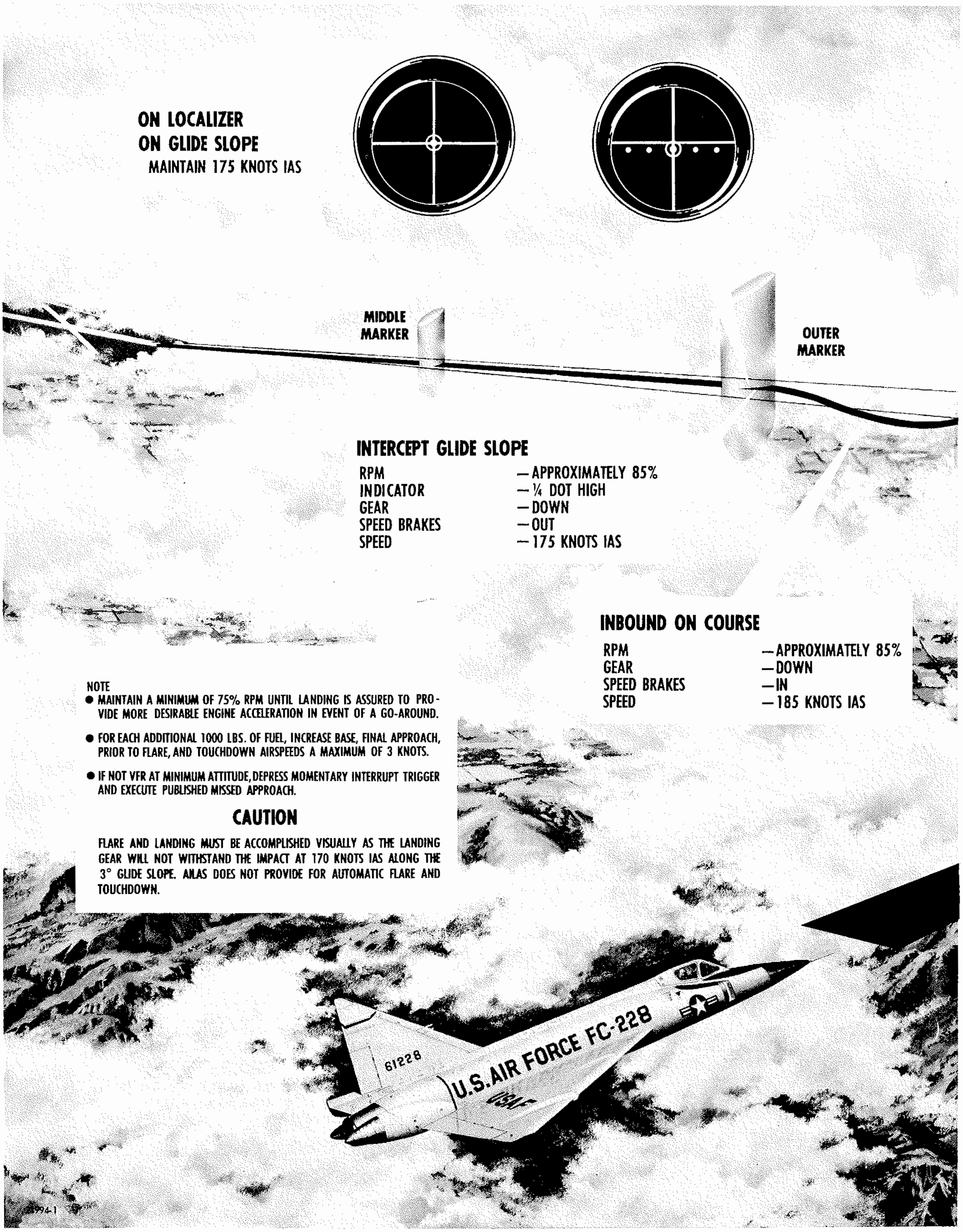
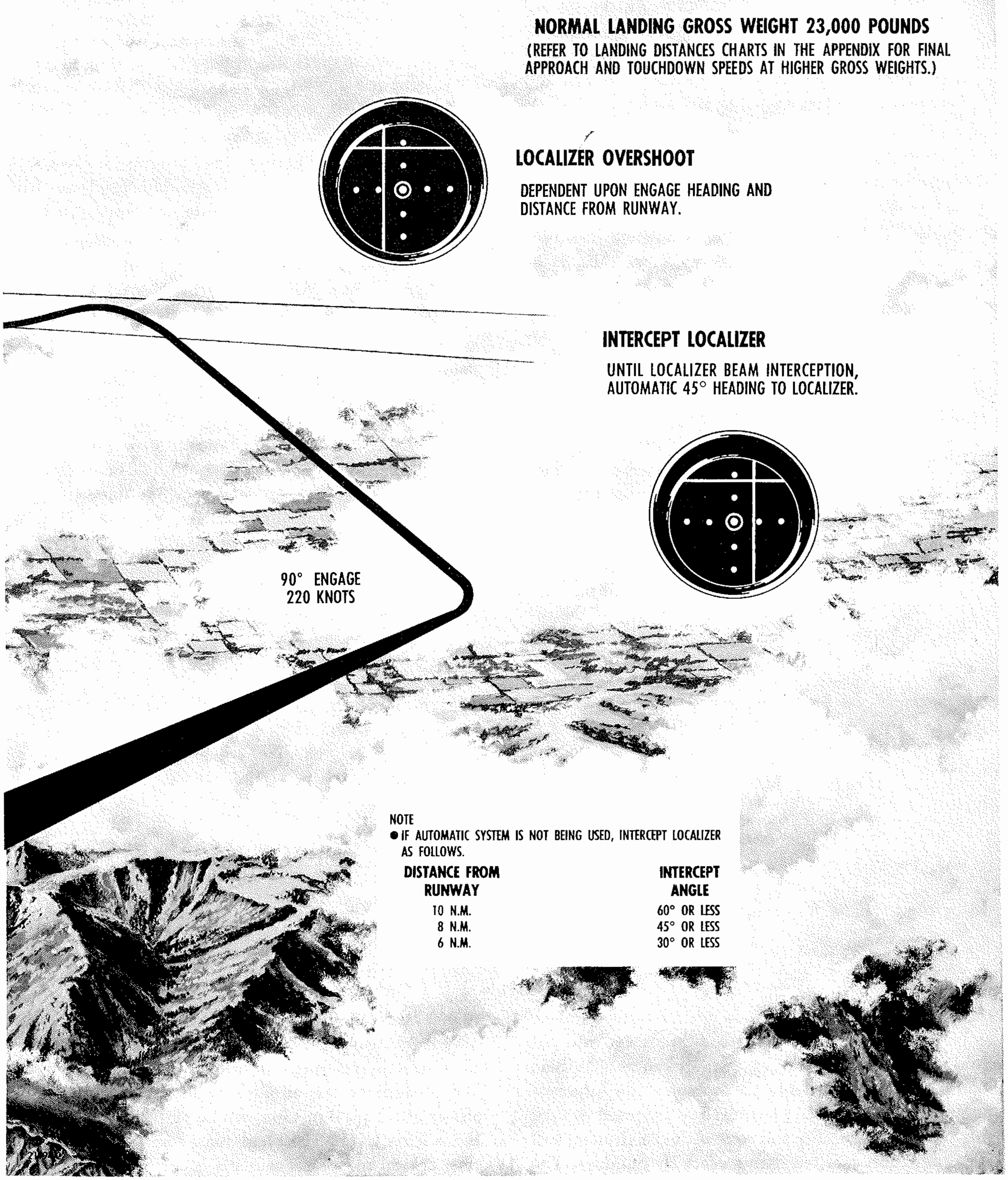

ON LOCALIZER
ON GLIDE SLOPE
MAINTAIN 175 KNOTS IAS
MIDDLE MARKER
OUTER MARKER
INTERCEPT GLIDE SLOPE
RPM —APPROXIMATELY 85%
INDICATOR —¼ DOT HIGH
GEAR —DOWN
SPEED BRAKES —OUT
SPEED —175 KNOTS IAS
INBOUND ON COURSE
RPM —APPROXIMATELY 85%
GEAR —DOWN
SPEED BRAKES —IN
SPEED —185 KNOTS IAS
NOTE
● MAINTAIN A MINIMUM OF 75% RPM UNTIL LANDING IS ASSURED TO PROVIDE MORE DESIRABLE ENGINE ACCELERATION IN EVENT OF A GO-AROUND.
● FOR EACH ADDITIONAL 1000 LBS. OF FUEL, INCREASE BASE, FINAL APPROACH, PRIOR TO FLARE, AND TOUCHDOWN AIRSPEEDS A MAXIMUM OF 3 KNOTS.
● IF NOT VFR AT MINIMUM ATTITUDE, DEPRESS MOMENTARY INTERRUPT TRIGGER AND EXECUTE PUBLISHED MISSED APPROACH.
CAUTION
FLARE AND LANDING MUST BE ACCOMPLISHED VISUALLY AS THE LANDING GEAR WILL NOT WITHSTAND THE IMPACT AT 170 KNOTS IAS ALONG THE 3° GLIDE SLOPE. AILAS DOES NOT PROVIDE FOR AUTOMATIC FLARE AND TOUCHDOWN.
U.S. AIR FORCE FC-228
61228
USAF

Figure 9-4

automatic approach (ailas)



2. AFCS engaged and operational in pilot assist mode.
3. Set ILS runway heading in course indicator.
4. Set function selector knob on J-4 compass control panel to MAG.

   This is necessary or the compass will be phased to another heading reference.
5. VOR receiver on, frequency set (some airplanes); ILS receiver (AN/ARN-31) on, frequency set, and instrument selector switch to ILS (other airplanes).
6. Check that the localizer and glide-slope warning "OFF" flags are not showing and that the airplane is below the glide-slope (glide-slope indicator above center).
7. Enter AILAS four-mile engage circle, which is centered 12 miles from the runway, at an altitude of 1500 feet above the runway. AILAS may be engaged on any heading; however, for early localizer capture and minimum localizer overshoot, the entry heading should be 45° or less to the localizer course.
8. Airspeed 220 KIAS, landing gear up.
9. AILAS button depressed. Check green light illuminated, dim as desired.

   The AILAS light will remain illuminated as long as the AILAS remains engaged; the light will go out upon disengagement. While under AILAS control, the elevon trim switch is inoperative. AILAS limits bank angle to 33 (±3°) until glide-slope entry.
10. Inbound on localizer, reduce speed to 185 KIAS. The pilot may assume manual control at any time during the approach by depressing the momentary interrupt trigger. Prior to glide-slope entry, manual control is available without disengaging AILAS by depressing the momentary interrupt trigger (engage light remains on). Upon release of the momentary interrupt trigger, the system reverts to the AILAS mode.
11. As the glide-slope indicator progresses down to 1/4 width from center, extend speed brakes, lower landing gear, and reduce speed to final approach speed.
12. Check for glide-slope capture by noting airplane pitch response to the glide-slope signal following coincidence. Following glide-slope interception, AILAS limits bank angle to 15 degrees.

When visual contact with the runway is established, depress and hold momentary interrupt trigger, flare out and touch down. With AILAS engaged prior to glide-slope capture, if the localizer signal is lost (warning flag appears), AILAS will automatically disengage and the AFCS will revert to pilot assist attitude hold. If either localizer or glide-slope is lost after glide-slope entry, the AILAS will automatically disengage and the AFCS will revert to pilot assist attitude hold.

ice and rain

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 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 drag from surface ice buildups. Surface icing will reduce 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 GCA or ILS) should be used to minimize the surface ice 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, a rapid rise in exhaust gas temperature and possible compressor stall (refer to COMPRESSOR STALL, Section III). The possibility of this occuring is reduced by the absence of inlet screens and the relatively clean, unrestricted intake. However, should sufficient ice buildup occur to restrict airflow, the first indication will be a compressor stall. Immediate action should be taken to relieve the stall and change to an altitude where icing is less severe. 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 operating at all times when flight under icing conditions is anticipated.

Note

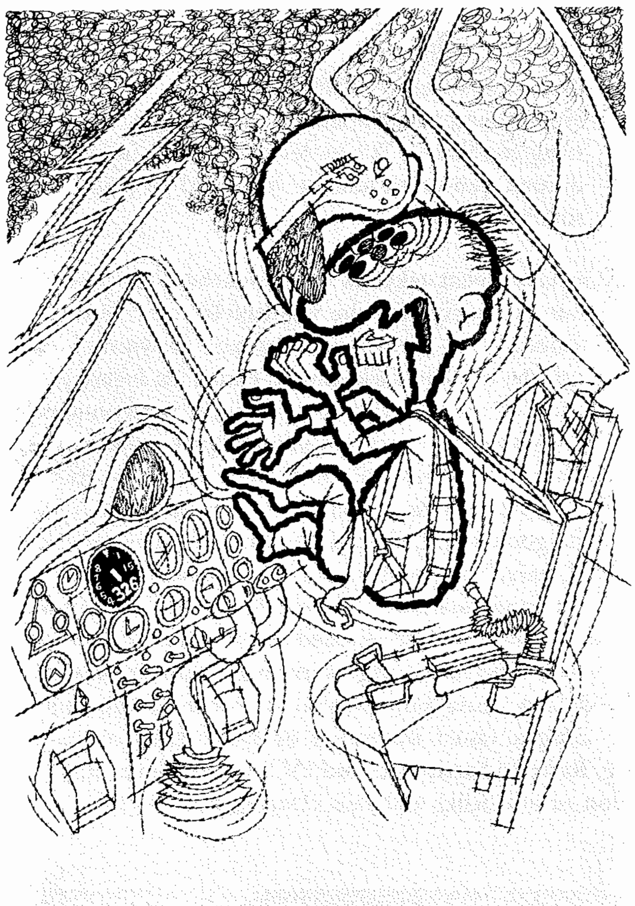
Engine operation below approximately 90% rpm may not supply sufficient heat to keep the engine compressor inlet clear of ice under severe icing conditions. When descending in icing conditions, power should be increased to provide sufficient heat.

When instrument takeoffs or approaches are to be made and rain is anticipated, the windshield rain clearing system should be in operation to increase forward visibility through the left-hand windshield panel. To insure proper operation of the rain clearing system, it is recommended that the system be energized momentarily prior to 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 clearing equipment.

CAUTION

- Flight through areas of turbulent air, hailstones, or thunderstorms should be avoided whenever possible. Flight under these conditions increases the possibility of compressor stall or engine flameout and can result in damage from hail or turbulence.
- Above 40,000 feet, a rough operating condition or compressor stall may result if the throttle is moved rapidly, or rpm is reduced below 85%. Subsequent movement of the throttle may not eliminate this condition.

The most serious consequences of flight through severe turbulence and thunderstorms is the increased possibility of engine flameout. Of particular importance, is the "crystal ice" compressor stall. At high altitudes, ice crystals associated with the area around and in thunderstorms can cause compressor stall and probable flameout. The ice crystals do not settle on the intake duct but go into



AN AIRSPEED OF 275 TO 325 KNOTS IAS, NOT TO EXCEED .88 MACH, IS RECOMMENDED FOR SAFE COMFORTABLE TURBULENT AIR PENETRATION.

22082

the engine with the air. The crystals are then heated during the compression process, and become ingested water. The ingested ice crystals reduce the compressor stall margin and compressor stall or flameout follows. There are several other factors associated with turbulence and thunderstorms which are also conducive to flameout:

- High liquid content of cumulus buildups.
- Icing of engine intake ducts or inlet guide vanes.
- Increased angles of attack caused by turbulence, resulting in marginal engine performance.
- Above 40,000 feet, the surge margin of the engine is reduced and there is poor air distribution across the face of the compressor.

CAUTION

Flying in turbulence or hail may increase inlet distortion. At higher altitudes this distortion can result in engine surge and possible flameout. However, normal engine air start may be accomplished.

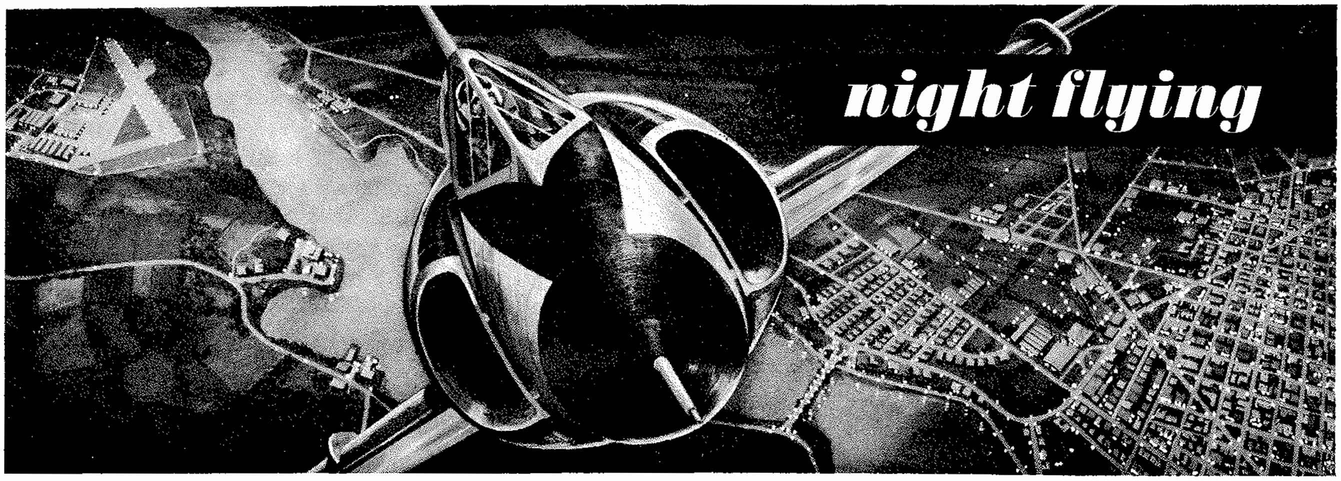
If thunderstorm areas cannot be avoided, use ground radar (GCI) or the fire control system radar (search mode) to determine and avoid the most intense storm areas. Establish a safe, comfortable penetration speed of 275 to 325 KIAS (not to exceed Mach .88) if such speed does not interfere with accomplishment of the mission. However, design limit load factors will not be exceeded at any speed below the design limit speed. Prior to weather penetration, make certain that pitot heat is on and that the anti-ice switch is in MANUAL ON position. The engine anti-ice system prevents ice formation but is not fully effective unless used before ice buildup occurs. Exhaust gas temperature and engine pressure ratio gauges should be monitored continuously during penetration to detect possible engine malfunction. Exhaust gas temperature indication alone may come too late to take timely action in preventing flameout. On airplanes having all points ignition capability, which permits engine ignition to be initiated regardless of throttle position (non afterburning), depress and hold the ignition button during the period of thunderstorm penetration. Under these circumstances, continuous use of the ignition system is permissible for a period not to exceed 10 minutes. After 10 minutes the ignition system must be turned off for a cooling period of at least 10 minutes.

CAUTION

- Continuous use of the ignition system in excess of 10 minutes, or consecutive continuous usages without a 10-minute interim cooling period may result in damage to the ignition system which will render it inoperable for restarts. Energizing the ignition system will not prevent compressor surge or stall, but it will aid in preventing flameout.
- During the period of ignition on, it is essential that EGT and EPR be monitored closely for possible increase above allowable limits. In event of high EGT and/or EPR, reduce thrust and altitude and maintain adequate airspeed for optimum engine inlet airflow.

Note

In event the ignition system is energized continuously for a period in excess of three minutes, an entry should be made on Form 781 noting the period of continuous ignition operation.



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.

WARNING

Airplanes in the rotating beacon configuration do not provide adequate exterior lighting to meet night formation requirements.

To insure satisfactory cold weather operation, utilize the normal operating procedures outlined in Section II in conjunction with the following additions:

BEFORE ENTERING THE AIRPLANE

Failure to remove snow and ice accumulated on airplanes while on the ground can result in serious aerodynamic and structural effects when flight is attempted. Depending on the weight and distribution of the snow and ice, takeoff distances and climb-out performances can be adversely affected. This roughness, pattern, and location of the snow and ice can affect stall speeds and handling characteristics to a dangerous degree. To eliminate these hazards and insure satisfactory performance, check that all snow and ice is removed from the airplane surfaces. Assure that all protective covers are removed.

BEFORE STARTING ENGINE

When attempting a normal start, if start is not obtained, the start should be aborted and a second start initiated immediately.

Note

In order to make an immediate second start attempt, the airplane must be supplied with external ac power to resupply fuel to the starter fuel flask. If no external power is available, refer to COMBUSTION START, SECOND ATTEMPT, Section VII.

If the second start fails, the procedure may be repeated after the ground compressor has been recharged.

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.

WARNING

In cold weather, make sure all instruments have warmed up sufficiently to insure normal operation. Check for sluggish instruments during taxiing.

TAXIING INSTRUCTIONS

Increase space between airplanes while taxiing to provide safe stopping distance and to prevent icing of airplane surfaces by melted snow and ice in the jet blast of preceding airplanes. Taxi speed should be reduced when taxiing on slippery surfaces to avoid skidding. Emergency fuel system may be used to reduce thrust if normal idle causes too high a taxi speed.

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, and ice fog from other jet engines can considerably reduce visibility prior to takeoff or during takeoff roll. Brakes alone cannot be relied upon to prevent skidding of the airplane when operating on a slippery surface above approximately 85% rpm. It will be necessary to restrain the airplane by methods other than brakes if a normal full military thrust check is to be made before the airplane begins the takeoff roll.

CAUTION

If no means of restraining the airplane is available and the instruments are to be checked on the takeoff roll, do not hold the brakes while the engine is accelerating. It is possible to lose control of the airplane if one wheel slides ahead of the other during engine acceleration.

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, the cabin pressure switch should be placed to RAM and not returned to PRESS until above approximately 3000 feet.

TAKEOFF

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 much steeper than normal. Afterburner takeoffs may produce an uncomfortably steep initial climb angle.

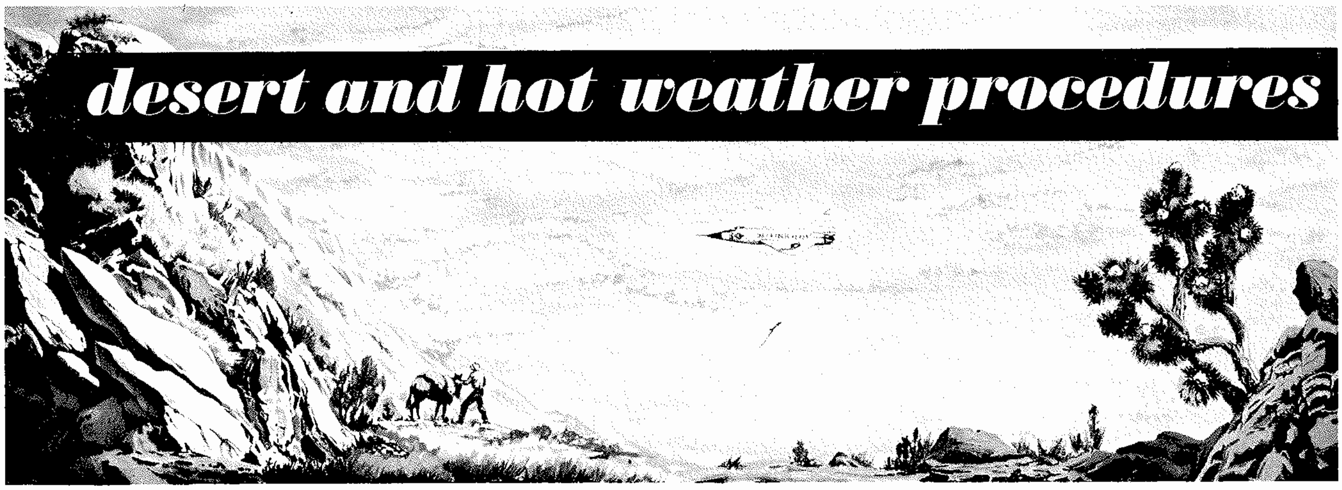
LANDING

Apply brakes carefully on icy runways, especially when some dry spots exist, to prevent skidding. Normal drag chute operation should be used.

CAUTION

When landing in a cross wind on icy runways, be prepared to jettison the drag chute should a weather-cocking tendency develop. Refer to Section II.

desert and hot weather procedures



In general, desert and hot weather 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. Takeoff and landing ground roll will be considerably increased in hot weather.

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 over inflated from the heat. Make sure that all protective covers are removed.

ON ENTERING AIRPLANE

Check the cockpit for excessive accumulation of dust or sand. Check instruments and control for moisture from high humidity, and ground heat them if necessary to dry them. Complete as much of preflight cockpit check as possible before operating, to avoid prolonged ground running.

BEFORE TAKEOFF

Note

Do not attempt takeoff in a sandstorm or dust storm. Park airplane cross-wind, shut down the engine, and have crew chief install protective covers.

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 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 switch to the RAM position, returning it to PRESS position after becoming airborne and above approximately 3000 feet.

CAUTION

It is imperative that takeoff be made at recommended speeds. Refer to Appendix for takeoff distances required at varying gross weights, temperatures, and field elevations. When outside air temperature is high, do not lift from runway too soon, as more than the usual takeoff distance will be required to obtain takeoff speed.

DURING FLIGHT

Note

During operation in high humidity conditions (ground dew point 20°C or higher), operate canopy defog continuously at altitude in order to insure fog-free canopy during descent. For less severe humidity conditions, canopy defog need not be turned on until just before descent.

DESCENT

Check that the windshield and canopy defog system is on at least four minutes before any rapid descent from altitude to prevent fogging and frosting of the windshield and canopy.

Note

Under high humidity conditions, the windshield defog system may not be capable of keeping the windshield clear of moisture.

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.

BEFORE LEAVING AIRPLANE

Leave the canopy open to permit air circulation in the cockpit unless blowing sand or dust is expected. Have the crew chief install the protective covers from the pitot boom, ram air intakes, engine intake ducts, and tailpipe.

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PART 1 INTRODUCTION

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INTRODUCTION

Note

Appendix I contains performance data for operation with or without external fuel tanks. The appendix is divided into nine parts, each pertaining to a specific phase of flight, and contains all charts (clean or with tanks) pertinent to that phase. The information is applicable to airplanes with either the Case X or the Case XX wing. In general, performance of the two configurations is similar. However, at high altitudes, airplanes of the Case XX configuration will provide significantly better performance than those having the Case X wing. Flight tests have demonstrated that the Case XX wing provides a 4200-foot increase in maximum combat ceiling and a Mach .06 speed increase at 50,000 feet, over the Case X wing configuration.

The charts shown in this Section present the flight-tested performance for this airplane. These data are based on operation with the J57-P-23 engine and are representative of airplanes in the production configuration. Performance data will vary among airplanes and pilots. Performance deteriorates slightly with age of the airplane. The engine, for example, even though it is trimmed to thrust output specified, will show an increase in fuel flow with an accumulation of hours. Performance of the airplane also deteriorates with improperly fitting door panels, damaged or dented wing and fuselage skin areas. The term "clean configuration" as applied to these charts implies no optional external stores (wing tanks) and no drag producing components extended, such as speed brakes or landing gear. In all cases, the ICAO standard atmosphere is used (except on takeoff and landing charts where conditions are generalized).

MACH NUMBER CORRECTION

To determine either true or indicated Mach number, see figure A1-1 and enter at left with true Mach number. Follow across to diagonal, then read down to obtain indicated Mach number. Reverse procedure to obtain true Mach number. Example: True Mach number equals 0.700. Enter from left at 0.700, follow across to diagonal, and read down for indicated Mach number, 0.680. The reverse procedure for an indicated Mach number of 0.700 gives a true Mach number of 0.725. These relationships are true regardless of altitude or free air temperature. Above Mach 1.025, true Mach number equals indicated Mach number.

AIRSPEED CORRECTION

Figure A1-2 shows the airspeed correction in terms of indicated airspeed and indicated altitude.

Note

Indicated airspeed (IAS) is the instrument reading. Calibrated airspeed (CAS) is the IAS corrected for installation error. Equivalent airspeed (EAS) is the CAS corrected for compressibility effects, and true airspeed (TAS) is the EAS corrected for atmospheric density effects.

Indicated Mach number is the instrument reading and true Mach number is the indicated Mach number corrected for installation error. In all cases, IAS and indicated Mach number are instrument readings without instrument corrections applied.

ALTITUDE CORRECTION

The correction for installation error on altitude is presented in two forms. Figure A1-3 shows the altitude correction in terms of IAS and figure A1-4 shows the correction in terms of indicated Mach number. Indicated altitude is the instrument reading without instrument error considered. True pressure altitude is obtained from figure A1-3 by entering the column for IAS and reading down to the line for indicated altitude. The corrective increment is added to the indicated altitude. Example: 200 KIAS at an altitude of 40,000 feet gives an increment of 420 feet. The true pressure altitude is then 40,420 feet. If the chart was entered with true pressure altitude the reverse would be true. Example: 200 KIAS at an altitude of 40,000 feet gives an increment of 420 feet. The indicated altitude would then be 39,580 feet. These corrections are also available on an indicated Mach number basis from figure A1-4. Read down the column for indicated Mach number to the line for indicated altitude and add the correction. Example: Indicated Mach number equals 0.78 and indicated altitude equals 40,000 feet. A linear interpolation between 0.60 and 0.80 indicated Mach number gives a correction of 420 feet. True pressure altitude is then 40,420 feet. If the chart was entered with true pressure altitude, the correction would be subtracted and the indicated altitude would be 39,580 feet.

COMPRESSIBILITY CORRECTION TO CALIBRATED AIRSPEED

Figure A1-5 converts calibrated airspeed to equivalent airspeed. CAS $-\Delta V_c =$ EAS. Mach number may be determined by interpolation. True airspeed may be obtained by multiplying EAS times $1/\sqrt{\sigma}$. The term, $1/\sqrt{\sigma}$, may be obtained from standard atmosphere tables. For this case an ambient temperature greater than standard would result in a conservative true airspeed. The opposite is true for a temperature lower than standard. True airspeed may be determined more accurately by use of figure A1-6, density altitude chart. As noted by guide lines the value of $1/\sqrt{\sigma}$ is determined for a known pressure altitude and ambient temperature.

TRUE AIRSPEED—TRUE MACH NUMBER AND TEMPERATURE CHART

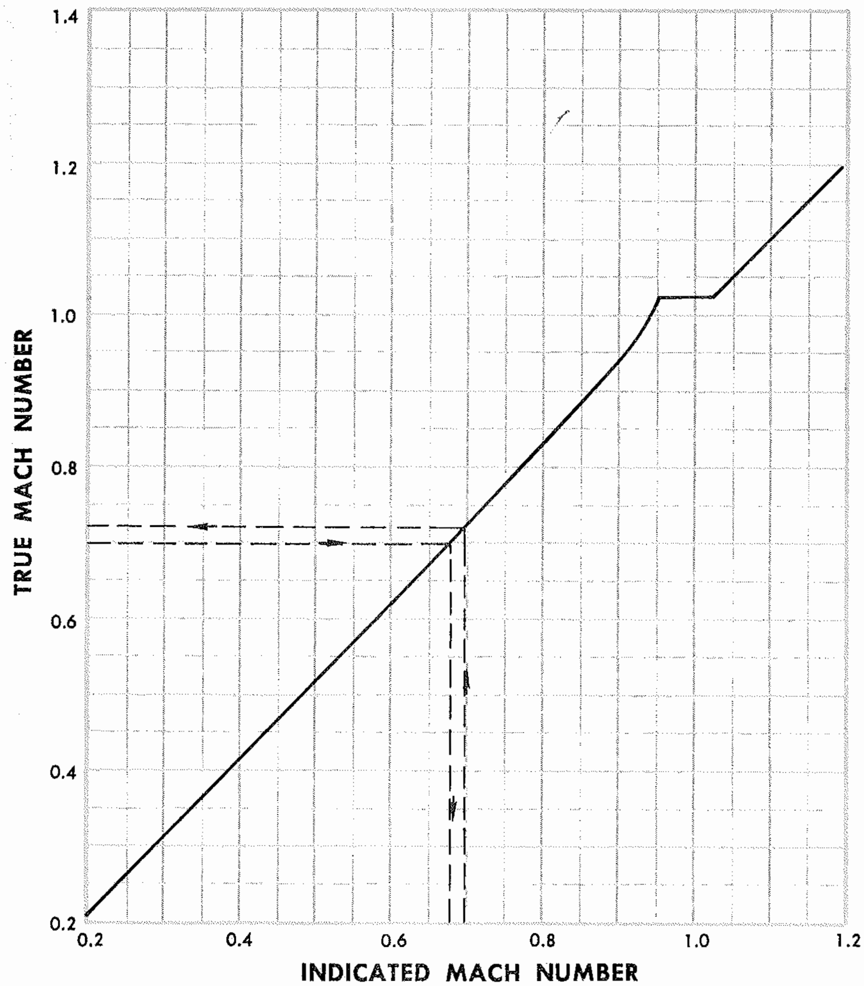
Figure A1-7 shows true airspeed as a function of temperature and true Mach number. The total temperature scale represents the direct dial reading minus instrument correction, if any. The lines of constant temperature are true ambient conditions (adiabatic temperature rise removed). This chart ties together four functions: true airspeed, true Mach number, total air temperature, and ambient air temperature. Any two of these functions will determine the remaining two. Example: At an ICAO standard pressure altitude of 30,000 feet, the ambient temperature is −44.4°C (figure A1-7 or figures A1-8 and A1-9). At a true Mach number of 0.95 the true airspeed is 559 knots and the total temperature is −5.6°C. Conversely, an indicator reading of −5.6°C total temperature at a true Mach number of 0.95 shows the true airspeed to be 559 knots.

mach number correction

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

ENGINE: J57-23

(INSTALLATION ERROR)



NOTE

ENTER AT LEFT WITH TRUE MACH NUMBER THEN READ DOWN TO OBTAIN INDICATED MACH NUMBER

ENTER AT BOTTOM WITH INDICATED MACH NUMBER THEN READ LEFT FOR TRUE MACH NUMBER

AT MACH 1.025 AND ON, TRUE AND INDICATED MACH ARE EQUAL

22001B

Figure A1-1

airspeed correction

(INSTALLATION ERROR)

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23

IAS KNOTS	INDICATED ALTITUDE—1000 FEET												
	SL	5	10	15	20	25	30	35	40	45	50	55	
100	5	5	4	4	4	4	4	4	4	3	3	3	
150	6	6	5	5	5	5	5	5	5	4	4	4	SUBSONIC
200	7	7	6	6	6	6	6	6	6	6	6	6	
250	8	8	7	7	7	7	7	7	7	7	3	0	
300	9	8	8	8	8	8	8	8	13	0	0		SUPERSONIC
350	9	9	9	9	9	9	9	4	0				
400	10	10	10	10	10	12	0	0					
450	11	11	11	11	19	0							
500	12	12	12	21	0								
550	13	14	19	0									
600	14	17	0										
650	16	0											
700	0												
750													

ADD CORRECTION TO OBTAIN CAS

22002B

Figure A1-2

altitude correction

(INSTALLATION ERROR)

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23

TRUE OR INDICATED ALTITUDE — FT	IAS—KNOTS 100	150	200	250	300	350	400	450	500	550	600	650	700
SL	35	80	130	190	255	340	450	590	750	940	1185	1510	0
5,000	45	95	150	215	295	395	535	695	875	1100	1770	0	
10,000	50	110	170	240	335	460	615	800	1030	2000	0		
15,000	60	125	195	280	390	540	725	940	2000	0			
20,000	70	140	220	320	450	630	855	1885	0				
25,000	85	155	245	370	545	760	1170	0					
30,000	95	175	285	445	655	945	160						
35,000	105	200	335	535	795	850	0						
40,000	130	240	420	670	1690	0							
45,000	150	295	500	900	0								
50,000	185	395	675	1000									
55,000	220	490	955	0									

SUBSONIC
SUPERSONIC

ADD CORRECTION TO OBTAIN TRUE PRESSURE ALTITUDE
SUBTRACT CORRECTION TO OBTAIN INDICATED ALTITUDE

22003B

Figure A1-3

altitude correction

INSTALLATION ERROR

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23

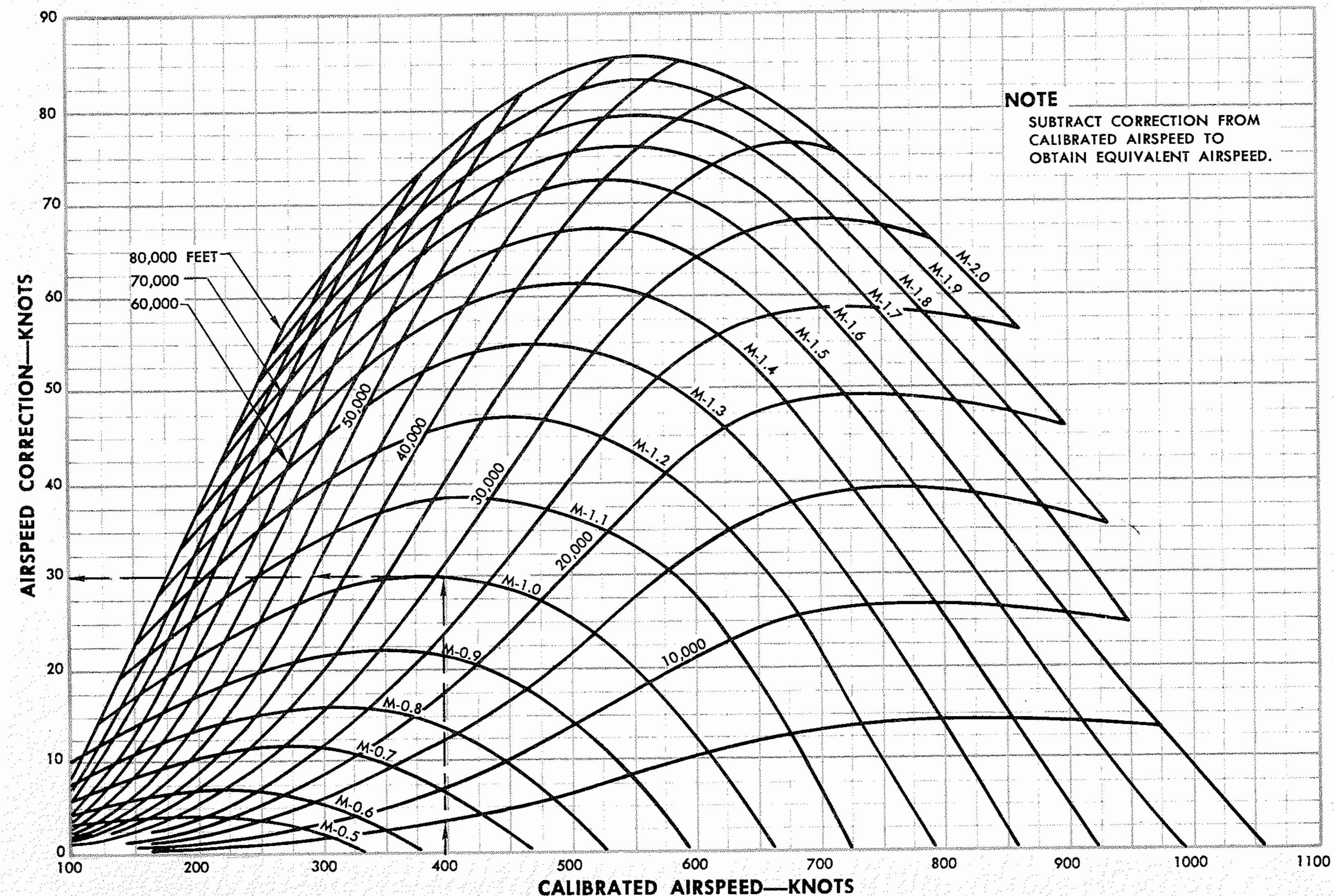
TRUE OR INDICATED ALTITUDE — FT	INDICATED MACH NUMBER .2	.4	.6	.8	.85	.9	.925	.95	.96	.98	1.0	1.016
SL	15	70	290	570	655	900	1155	1655	1780	1300	575	0
5,000	25	90	270	540	630	865	1110	1585	1680	1290	540	0
10,000	30	105	260	525	605	840	1070	1530	1570	1285	480	0
15,000	25	100	255	510	585	810	1030	1470	1525	1255	460	0
20,000	5	85	245	495	565	775	1000	1415	1490	1225	425	0
25,000	5	80	230	480	545	750	965	1365	1455	1175	415	0
30,000	10	75	220	460	520	720	910	1315	1415	1125	410	0
35,000	10	80	210	445	505	695	895	1275	1385	1100	410	0
40,000	10	85	210	440	495	680	875	1250	1355	1070	405	0
45,000	10	85	215	440	490	680	885	1240	1340	1070	400	0
50,000	15	90	225	440	490	690	890	1240	1330	1070	400	0
55,000	25	105	240	450	500	700	895	1260	1370	1100	415	0

ADD CORRECTION TO OBTAIN TRUE PRESSURE ALTITUDE
SUBTRACT CORRECTION TO OBTAIN INDICATED ALTITUDE

22004B

Figure A1-4

compressibility correction to calibrated air speed

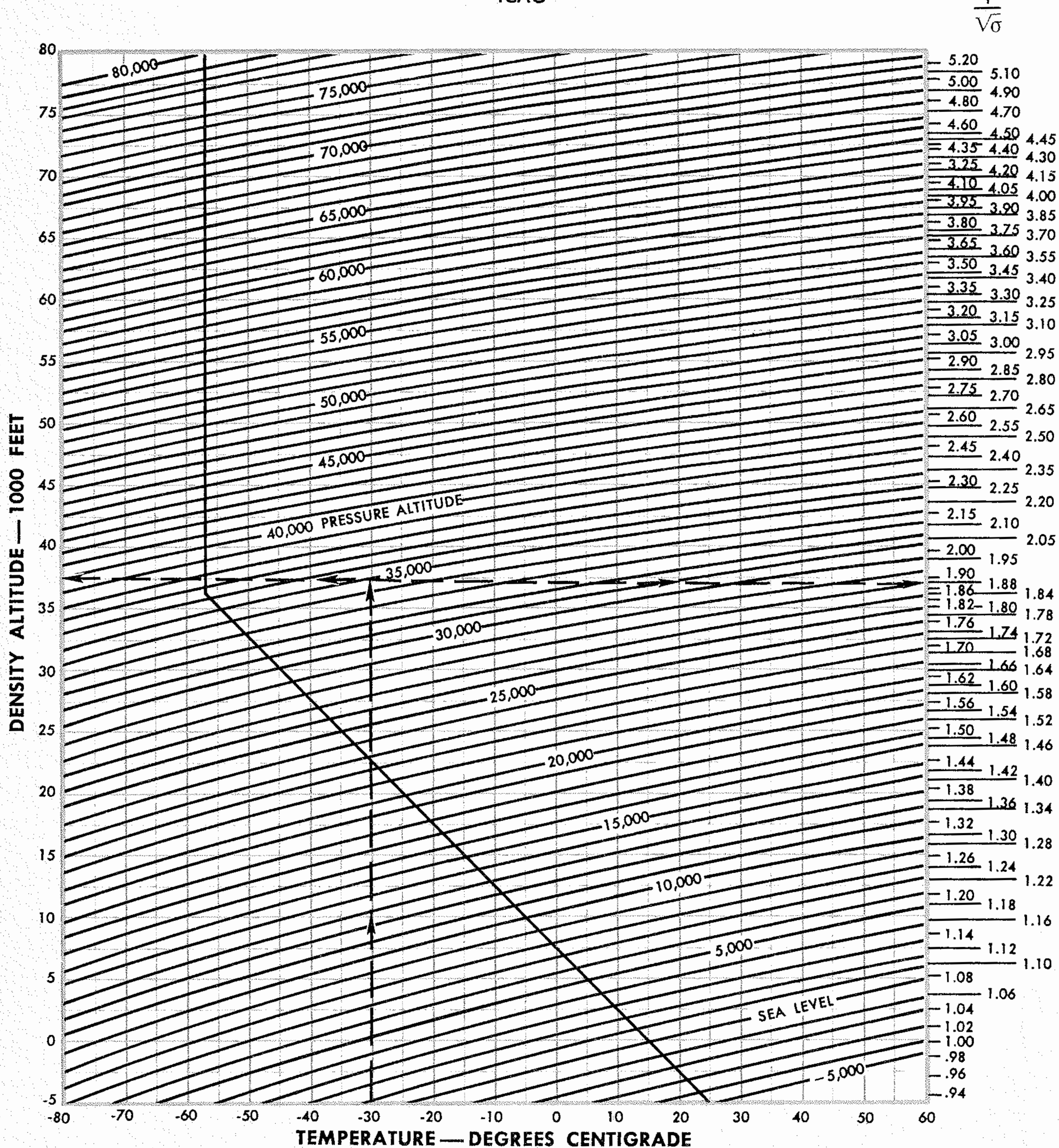


22005C

Figure A1-5

density altitude chart

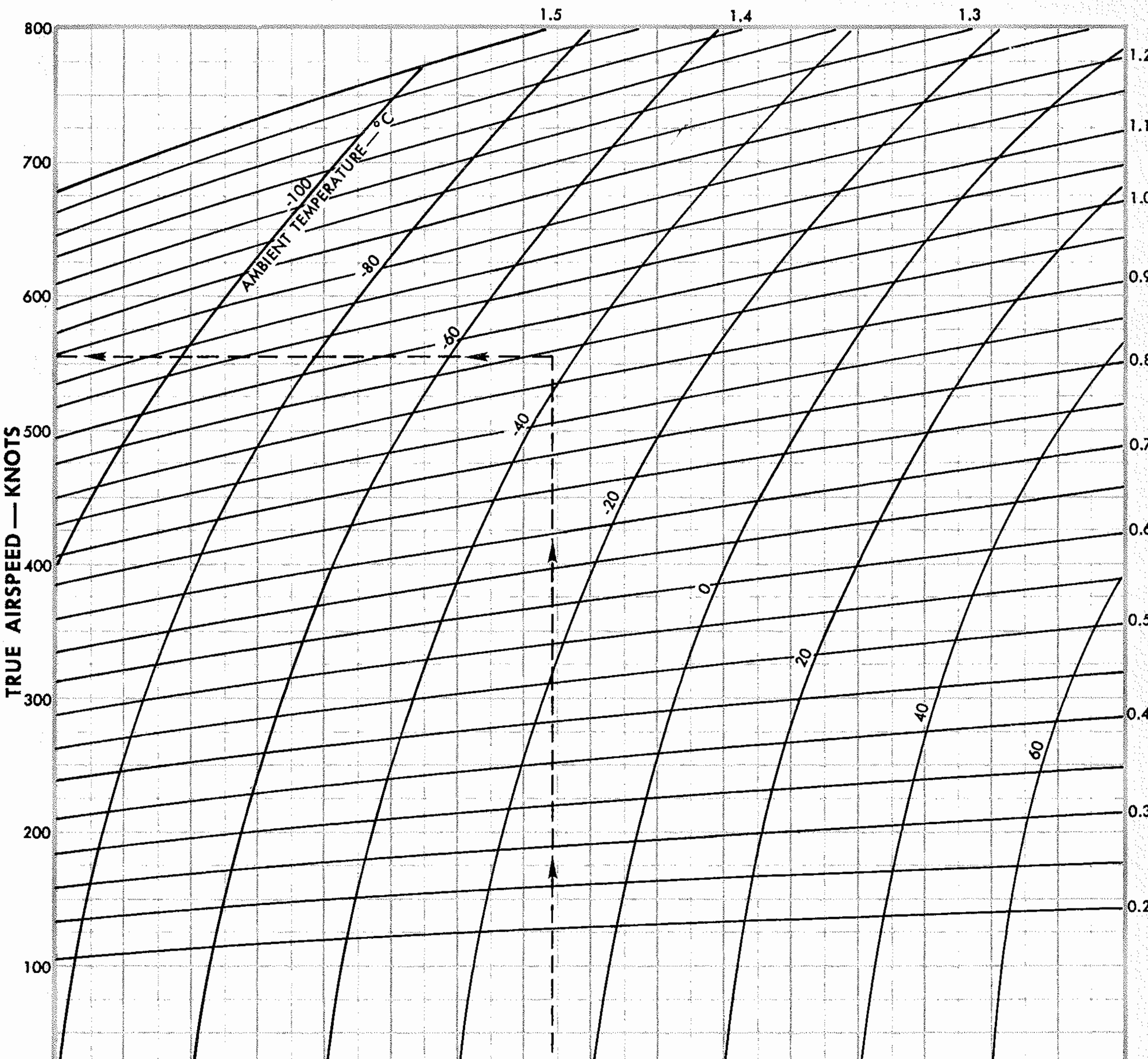
ICAO



22006C

Figure A1-6

true airspeed • mach number temperature chart



REMARKS

(1) TEMPERATURE RECOVERY FACTOR = 0.95

$T_{AMBIENT} = T_{TOTAL} / 1 + .19\ M^2$

REFER TO TEXT FOR EXAMPLE

22007C

Figure A1-7

standard atmosphere table

STANDARD SL CONDITIONS:
TEMPERATURE=15°C (59°F)
PRESSURE=29.921 IN. Hg (2116.216 LB/SQ FT)
DENSITY=.0023769 SLUGS/CU FT
SPEED OF SOUND=1116.89 FT/SEC (661.7 KNOTS)

CONVERSION FACTORS:
1 IN. Hg=70.727 LB/SQ FT
1 IN. Hg=0.49116 LB/SQ IN.
1 KNOT=1.151 MPH
1 KNOT=1.688 FT/SEC

ALTITUDE FEET	DENSITY RATIO σ	$\frac{1}{\sqrt{\sigma}}$	TEMPERATURE °C	TEMPERATURE °F	SPEED OF SOUND KNOTS	PRESSURE IN Hg	PRESSURE RATIO δ
SL	1.000	1.0000	15.000	59.000	661.7	29.921	1.0000
1,000	.9711	1.0148	13.019	55.434	659.5	28.856	.9644
2,000	.9428	1.0299	11.038	51.868	657.2	27.821	.9298
3,000	.9151	1.0454	9.056	48.302	654.9	26.817	.8962
4,000	.8881	1.0611	7.076	44.735	652.6	25.842	.8637
5,000	.8617	1.0773	5.094	41.169	650.3	24.896	.8320
6,000	.8359	1.0938	3.113	37.603	648.7	23.978	.8014
7,000	.8106	1.1107	1.132	34.037	645.6	23.088	.7716
8,000	.7860	1.1279	—0.850	30.471	643.3	22.225	.7428
9,000	.7620	1.1456	—2.831	26.905	640.9	21.388	.7148
10,000	.7385	1.1637	—4.812	23.338	638.6	20.577	.6877
11,000	.7155	1.1822	—6.793	19.772	636.2	19.791	.6614
12,000	.6932	1.2011	—8.774	16.206	633.9	19.029	.6360
13,000	.6713	1.2205	—10.756	12.640	631.5	18.292	.6113
14,000	.6500	1.2403	—12.737	9.074	629.0	17.577	.5875
15,000	.6292	1.2606	—14.718	5.508	626.6	16.886	.5643
16,000	.6090	1.2815	—16.699	1.941	624.2	16.216	.5420
17,000	.5892	1.3028	—18.680	—1.625	621.8	15.569	.5203
18,000	.5699	1.3246	—20.662	—5.191	619.4	14.942	.4994
19,000	.5511	1.3470	—22.643	—8.757	617.0	14.336	.4791
20,000	.5328	1.3700	—24.624	—12.323	614.6	13.750	.4595
21,000	.5150	1.3935	—26.605	—15.889	612.1	13.184	.4406
22,000	.4976	1.4176	—28.587	—19.456	609.6	12.636	.4223
23,000	.4806	1.4424	—30.568	—23.022	607.1	12.107	.4046
24,000	.4642	1.4678	—22.549	—26.588	604.6	11.597	.3876
25,000	.4481	1.4938	—34.530	—30.154	602.1	11.103	.3711
26,000	.4325	1.5206	—36.511	—33.720	599.6	10.627	.3552
27,000	.4173	1.5480	—38.492	—37.286	597.1	10.168	.3398
28,000	.4025	1.5762	—40.474	—40.852	594.6	9.725	.3250
29,000	.3881	1.6052	—42.455	—44.419	592.1	9.297	.3107
30,000	.3741	1.6349	—44.436	—47.985	589.5	8.885	.2970
31,000	.3605	1.6654	—46.417	—51.551	586.9	8.488	.2837
32,000	.3473	1.6968	—48.398	—55.117	584.4	8.106	.2709
33,000	.3345	1.7291	—50.379	—58.683	581.8	7.737	.2586
34,000	.3220	1.7623	—52.361	—62.249	579.2	7.382	.2467
35,000	.3099	1.7964	—54.342	—65.816	576.6	7.041	.2353
36,000	.2981	1.8315	—56.323	—69.382	574.0	6.712	.2243
36,089	.2791	1.8347	—56.500	—69.700	573.7	6.683	.2234
37,000	.2843	1.8753				6.397	.2138
38,000	.2710	1.9209				6.097	.2038
39,000	.2583	1.9677				5.811	.1942
40,000	.2462	2.0155	—56.500	—69.700	573.7	5.538	.1851

22008-1B

Figure A1-8

standard atmosphere table

STANDARD SL CONDITIONS:
TEMPERATURE=15°C (59°F)
PRESSURE=29.921 IN. Hg 2116.216 LB/SQ FT
DENSITY=.0023769 SLUGS/CU FT
SPEED OF SOUND=1116.89 FT/SEC 661.7 KNOTS

CONVERSION FACTORS:
1 IN. Hg=70.727 LB/SQ FT
1 IN. Hg=0.49116 LB/SQ IN.
1 KNOT=1.151 MPH
1 KNOT=1.688 FT/SEC

ALTITUDE FEET	DENSITY RATIO σ	$\frac{1}{\sqrt{\sigma}}$	TEMPERATURE °C	TEMPERATURE °F	SPEED OF SOUND KNOTS	PRESSURE IN Hg	PRESSURE RATIO δ
41,000	.2346	2.0654	—56.500	—69.700	573.7	5.278	.1764
42,000	.2236	2.1148				5.030	.1681
43,000	.2131	2.1662				4.794	.1602
44,000	.2031	2.2189				4.569	.1527
45,000	.1936	2.2728				4.355	.1455
46,000	.1845	2.3281				4.151	.1387
47,000	.1758	2.3848				3.956	.1322
48,000	.1676	2.4428				3.770	.1260
49,000	.1597	2.5022				3.593	.1201
50,000	.1522	2.5630				3.425	.1145
51,000	.1451	2.6254				3.264	.1091
52,000	.1383	2.6892				3.111	.1040
53,000	.1318	2.7546				2.965	.09909
54,000	.1256	2.8216				2.826	.09444
55,000	.1197	2.8903				2.693	.09001
56,000	.1141	2.9606				2.567	.08578
57,000	.1087	3.0326				2.446	.08176
58,000	.1036	3.1063				2.331	.07792
59,000	.09877	3.1819				2.222	.07426
60,000	.09414	3.2593				2.118	.07078
61,000	.08972	3.3386				2.018	.06746
62,000	.08551	3.4198				1.924	.06429
63,000	.08150	3.5029				1.833	.06127
64,000	.07767	3.5881				1.747	.05840
65,000	.07403	3.6754				1.665	.05566
66,000	.07055	3.7649				1.587	.05305
67,000	.06724	3.8564				1.513	.05056
68,000	.06409	3.9502				1.442	.04819
69,000	.06108	4.0463				1.374	.04592
70,000	.05821	4.1447				1.310	.04377
71,000	.05548	4.2456				1.248	.04171
72,000	.05288	4.3488				1.190	.03976
73,000	.05040	4.4545				1.134	.03789
74,000	.04803	4.5633				1.081	.03611
75,000	.04578	4.6738				1.030	.03442
76,000	.04363	4.7874				0.982	.03280
77,000	.04158	4.9039				0.935	.03126
78,000	.03963	5.0231				0.892	.02980
79,000	.03777	5.1454				0.850	.02840
80,000	.03600	5.2706	—56.500	—69.700	573.7	0.810	.02707

22008-2B

Figure A1-9

PART 2 TAKEOFF

TABLE OF CONTENTS

TAKEOFF DATA CARD

Prior to each takeoff compute all data contained on the Takeoff Data Card as shown below.

F-102A
TAKEOFF DATA CARD

CONDITIONS

Runway Length ______________
Wind ______________
Outside Air Temp ______________
Pressure Altitude ______________
Gross Weight ______________

TAKEOFF

Engine Pressure Ratio ______________
Acceleration Check ______________
Takeoff Distance ______________
Takeoff Speed ______________
Initial Climb Speed ______________

LANDING IMMEDIATELY AFTER TAKEOFF

Final Approach Speed ______________
Prior to Flare Speed ______________
Touchdown Speed ______________
Landing Ground Roll:
Wheel Brakes Only ______________
Drag Chute Deployed ______________

Completion of both takeoff and landing data is necessary before takeoff, in the event of emergency landing after takeoff. The data for "Landing Immediately After Takeoff" should be based on the takeoff gross weight. The takeoff and landing card nomenclature definitions are as follows:

1. **Conditions.**
   a. Runway Length. Usable length of runway in feet.
   b. Wind. The wind component parallel to the runway.
   c. Outside Air Temperature. Runway air temperature in degrees centigrade.
   d. Pressure Altitude. Field pressure altitude obtained by setting altimeter to 29.92 inches Hg and reading altimeter dial.
   e. Gross Weight. Gross weight of airplane in pounds at start of takeoff roll or at start of final approach, whichever is applicable.
2. **Takeoff Data.**
   a. Engine Pressure Ratio. The pressure ratio of pitot pressure and engine turbine discharge pressure. The pressure ratio setting or the maximum and minimum limits are obtained from the Takeoff Check Chart, figure A2-5.
   b. Takeoff Distance. Ground roll to takeoff speed in feet (figure A2-7 or A2-8).
   c. Takeoff Speed. Indicated airspeed in knots at which the airplane leaves the ground (figure A2-9 or A2-10).
   d. Initial Climb Speed. Indicated airspeed in knots at start of climb for maximum rate-of-climb. See figure A3-2 without external tanks.
3. **Landing Immediately After Takeoff.**
   a. Approach Speed. Recommended indicated airspeed in knots for final approach (figure A7-3 or A7-4).
   b. Prior to Flare Speed. Recommended indicated airspeed in knots for the latter portion of final approach just prior to the flare.
   c. Touchdown Speed. Indicated airspeed in knots at which the airplane contacts the runway.
   d. Landing Ground Roll:
      (1) Wheel Brakes Only. Distance in feet from airplane touchdown to full stop with no drag chute and speed brakes open, using wheel brakes only (figures A7-3 and A7-5).
      (2) Drag Chute Deployed. Distance in feet from airplane touchdown to full stop with drag chute (inflated at touchdown), speed brakes open, and using wheel brakes (figures A7-4 and A7-6).

TAKEOFF CHECK CHART

Figure A2-5 shows the pressure ratio of pitot pressure and turbine discharge pressure. The minimum and maximum limits represent the range of pressure ratio that must be obtained for takeoff. The target limit represents that pressure to be set in the takeoff window of the pressure ratio gage. When this limit is set in the takeoff window, the extreme ends of the reference mark will indicate the minimum and maximum limits. When the throttle is advanced to FULL MIL POWER, the pressure ratio gage should indicate within the extreme ends of the reference mark, with the exhaust gas temperature and rpm remaining within limits.

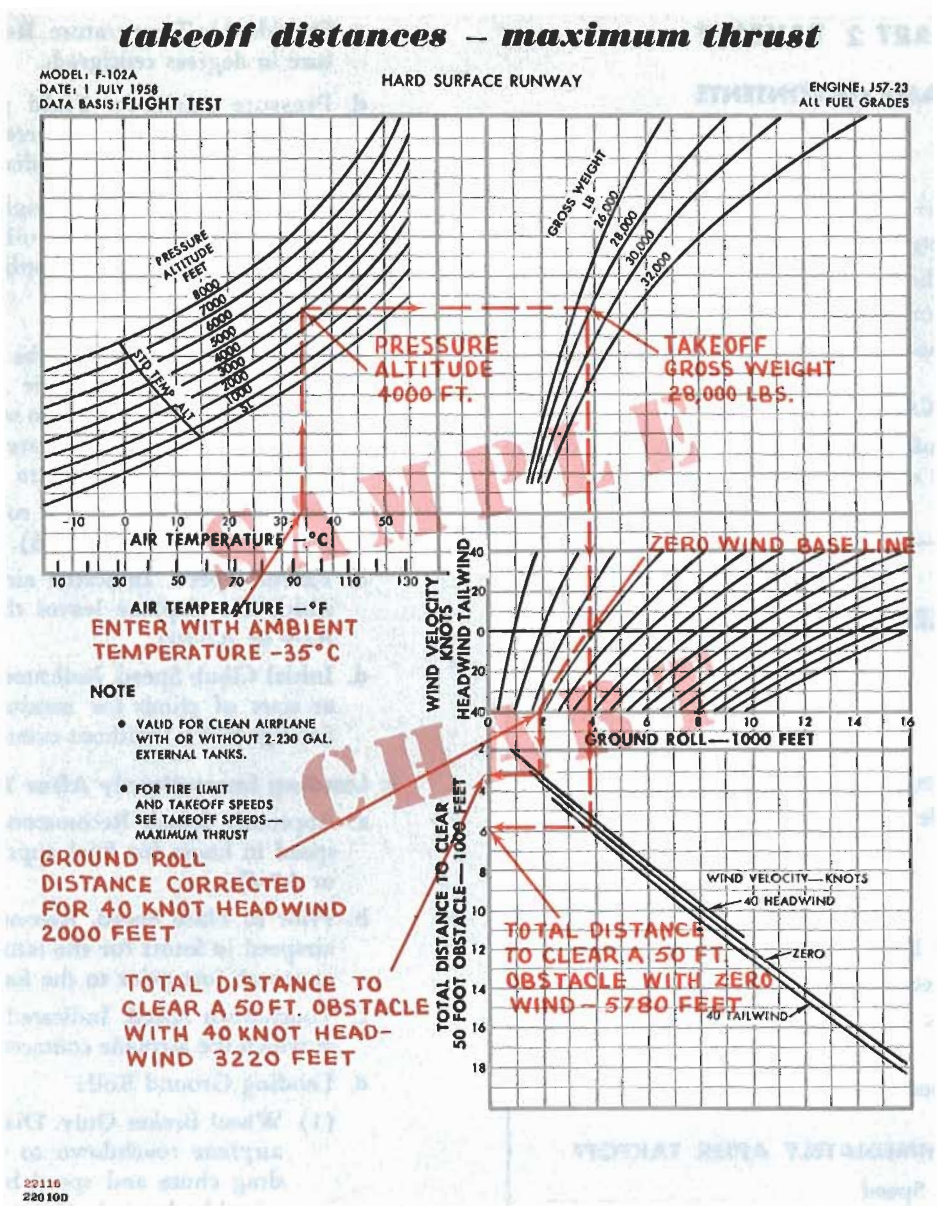


Figure A2-1

Use

Engine pressure ratio is determined by entering the ambient temperature scale for existing temperature condition and proceeding vertically to the minimum target, and maximum takeoff limits. From this point, read to the left for the pressure ratio setting.

TAKEOFF DISTANCE

Maximum thrust takeoff distances are shown in figure A2-7 and military thrust takeoff distances are shown in figure A2-8. The takeoff distance charts show both ground run distance and total distance over a 50-foot obstacle for takeoff from a hard-surface runway. Various conditions of gross weight, ambient air temperature, pressure altitude, and wind are shown. These data are valid for the airplane with or without external fuel tanks within the weight range shown.

Use

Enter the chart with the ambient air temperature and read up to the pressure altitude. Read horizontally to the right for the takeoff gross weight and then down to the ground run distance with zero wind. Correct ground run for wind and proceed down to appropriate base lines and read to the left for the total distance to clear 50-foot obstacle.

Sample Problem

Find the maximum thrust takeoff distance for an airplane weighing 28,000 pounds, a temperature of 35°C and a 40-knot headwind:

a. Enter figure A2-1 with the ambient temperature −35°C.

b. Read up to the pressure altitude — 4000 feet.

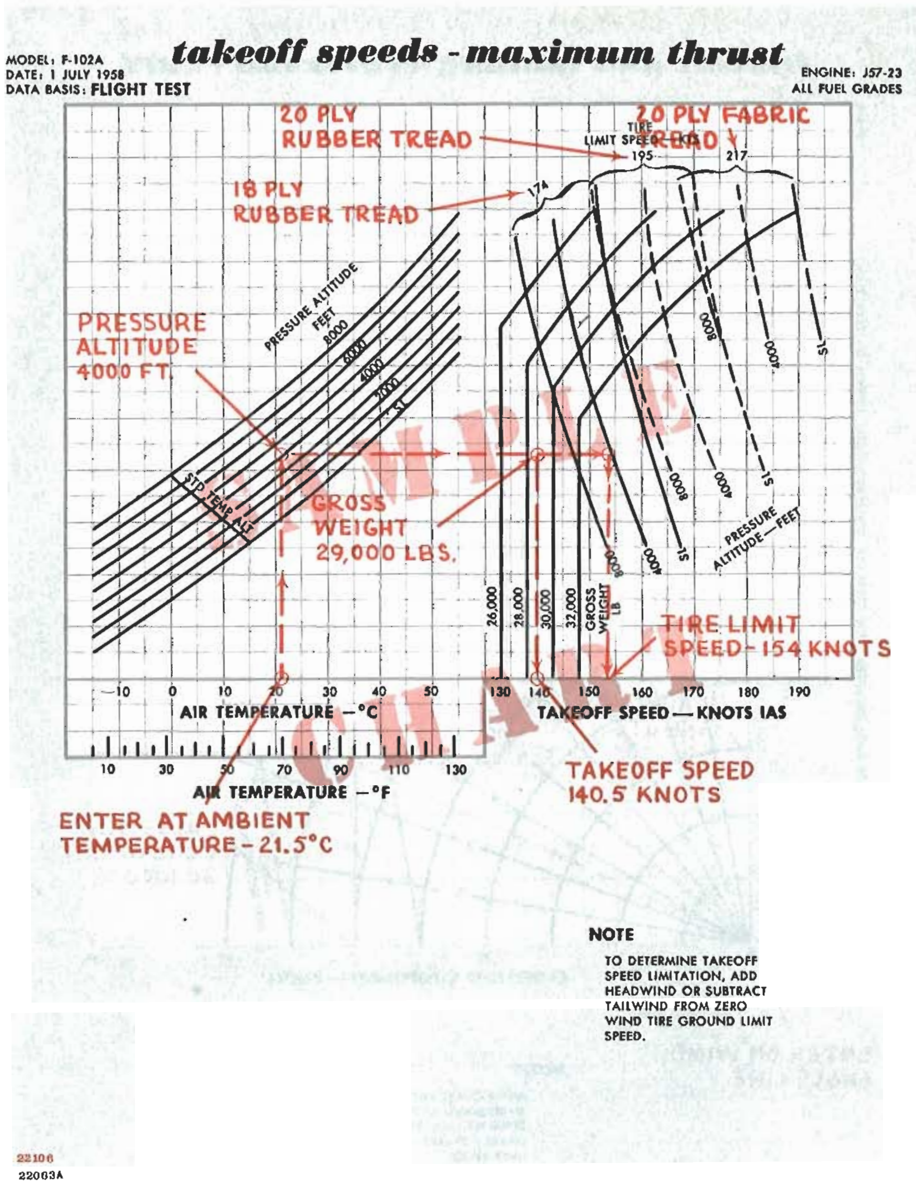


Figure A2-2

c. Read horizontally to the right for the takeoff gross weight -- 28,000 pounds.

d. Proceed down to the ground run wind base line — zero wind.

e. Correct ground run for headwind by following parallel to a wind guide line to 40-knot head-wind.

f. Read down to ground run distance for 40-knot headwind — 2000 feet.

g. Read down to baseline for total distance to clear a 50-foot obstacle and 40-knot headwind.

h. Read horizontally to the left for total distance to clear a 50-foot obstacle with 40-knot head-wind — 3220 feet.

TAKEOFF SPEEDS

Takeoff speeds are shown in figures A2-9 and A2-10 for maximum thrust and military thrust, respectively. These charts provide the normal takeoff speeds for various gross weights, ambient air temperatures, and pressure altitudes. Tire limit speeds are also provided on the charts.

Use

Enter the chart with the ambient air temperature and read up to the pressure altitude. Read horizontally to the right to the takeoff gross weight, then down to the take-off speed.

Sample Problem

Find the takeoff speed for a maximum thrust takeoff at 4000 feet field elevation. Ambient temperature is 21.5°C. Airplane gross weight is 29,000 pounds:

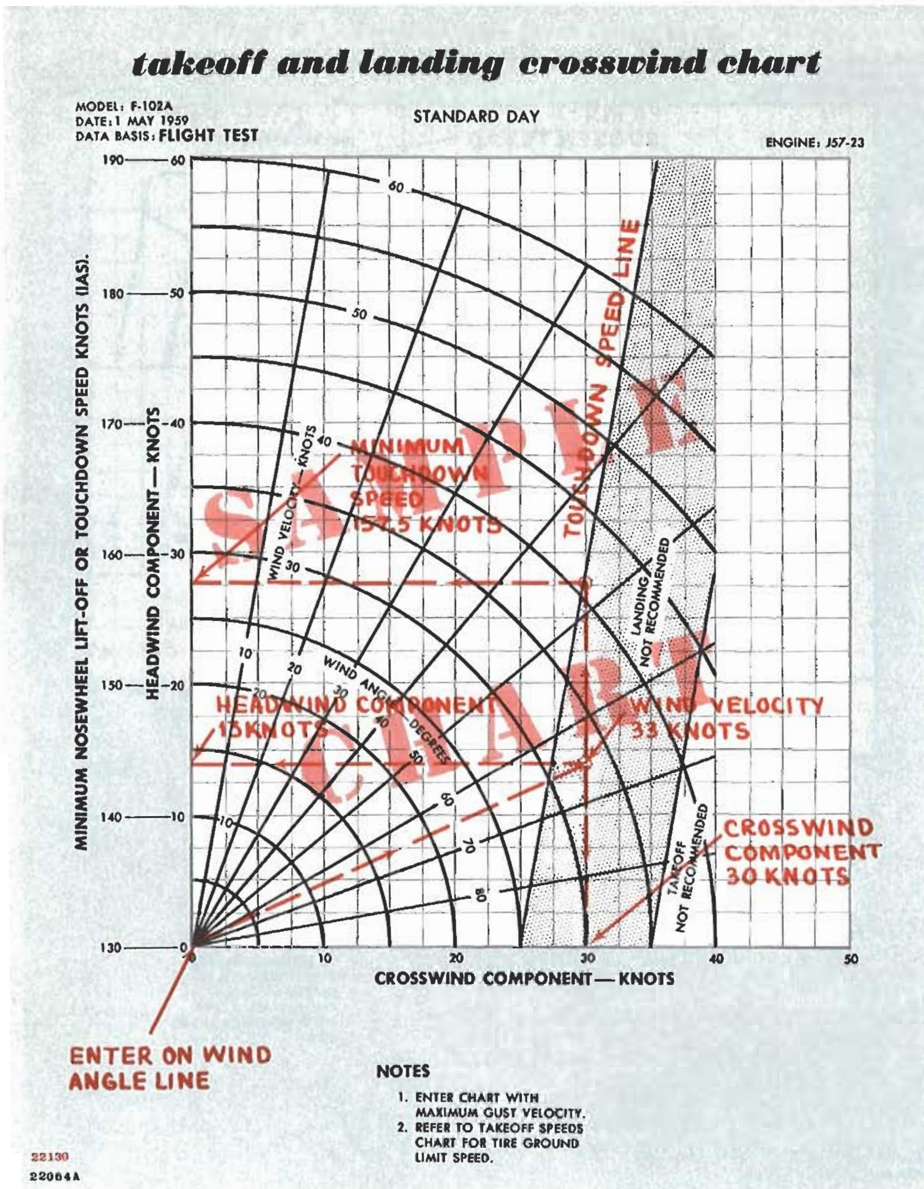


Figure A2-3

a. Enter figure A2-2 at ambient temperature — 21.5°C.

b. Read up to pressure altitude — 4000 feet.

c. Read across to a gross weight of 29,000 pounds.

d. Read down to a takeoff speed of 140.5 KIAS.

Note

The takeoff speed is less than tire limit speed. Do not take off when takeoff speed is higher than tire limit speed.

TAKEOFF AND LANDING CROSSWIND CHART

The Takeoff and Landing Cross-Wind Chart (figure A2-6) is used to resolve a surface wind of known velocity and direction into its headwind and crosswind component and to determine minimum recommended takeoff and landing speeds under various crosswind conditions.

Use

To resolve a surface wind of known velocity and direction into its headwind and cross-wind components, enter the chart on the wind angle line and follow it to the wind velocity circle. Then proceed horizontally to the left scale and read the velocity of the headwind component, and vertically downward to the bottom scale and read the velocity of the crosswind component.

Sample Problem

Given: A wind of 33 knots, 65° relative to runway heading.

Determine: The headwind and cross-wind components.

a. Enter figure A2-3 on the wind angle line — 65°.

b. Proceed to the wind velocity circle — 33 knots. Determine that takeoff is recommended.

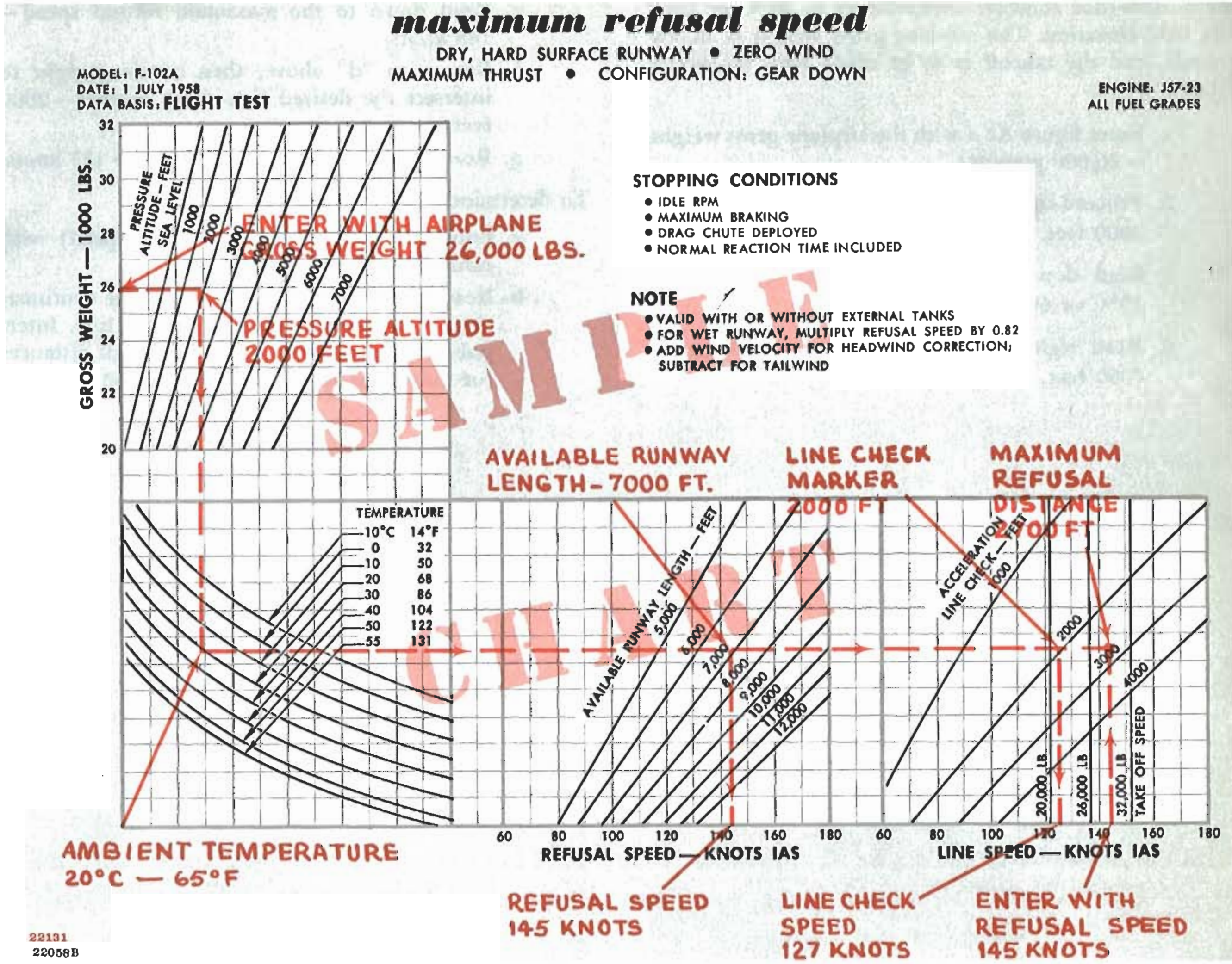


Figure A2-4

c. Move horizontally to the left and read the head-wind component — 13 knots.

d. Move vertically downward and read the cross-wind component — 30 knots.

MAXIMUM REFUSAL SPEED

Maximum Refusal Speed Charts, figure A2-11 and A2-12, show the highest speed to which the airplane can be accelerated (assuming normal acceleration) and still be stopped within the remaining runway length. The refusal speed charts take into account gross weight, pressure altitude, temperature, available runway length and pilot reaction time. They are presented for a dry, hard surface runway with a correction factor noted for a wet runway. Line speed is used to allow the takeoff to be monitored at marked runway distances for acceleration line check markers. In the event of subnormal acceleration, a decision to continue or abort the takeoff may be made prior to attaining the refusal speed.

Use

To determine the refusal speed, enter the chart at the upper left block with the airplane gross weight and proceed to the right to intersect the pressure altitude line. Read down to the ambient temperature line in the block directly below. Then read to the right along the atmospheric conditions line and intersect the applicable available runway length. At this point read down to the maximum refusal speed. Return to the atmospheric conditions line and continue right to the runway marker line that will be used for an acceleration line check. Read down to the speed that should be attained at this point in the takeoff roll. To determine refusal distance, enter the right block (line check speed) with the refusal speed and read up to a point that intersects the continuation of the atmospheric conditions line. At this point interpolate between the acceleration line check distances for maximum refusal distance.

Sample Problem

Find the maximum refusal speed, the line check speed at the 1000-foot runway marker, and the refusal distance for a 7000-foot runway. Temperature is 20°C at 2000 feet field elevation. The airplane gross weight is 26,000 pounds, and the takeoff is to be made with maximum thrust:

a. Enter figure A2-4 with the airplane gross weight — 26,000 pounds.

b. Proceed right to intersect the pressure altitude — 2000 feet.

c. Read down to the ambient air temperature, 20°C or 68°F.

d. Read right to the available runway length — 7000 feet.

e. Read down to the maximum refusal speed — 145 knots.

f. Return to "d" above, then continue right to intersect the desired line check marker — 2000 feet.

g. Read down to the line check speed — 127 knots.

To determine refusal distance:

a. Enter the right block (line check speed) with refusal speed — 145 knots.

b. Read up to a point that intersects the continuation of the atmospheric conditions line. Interpolate between acceleration line check distances for maximum refusal distance — 2700 feet.

takeoff check chart

MILITARY THRUST ● (NON-AFTERBURNING)

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23
FUEL GRADE: JP-4

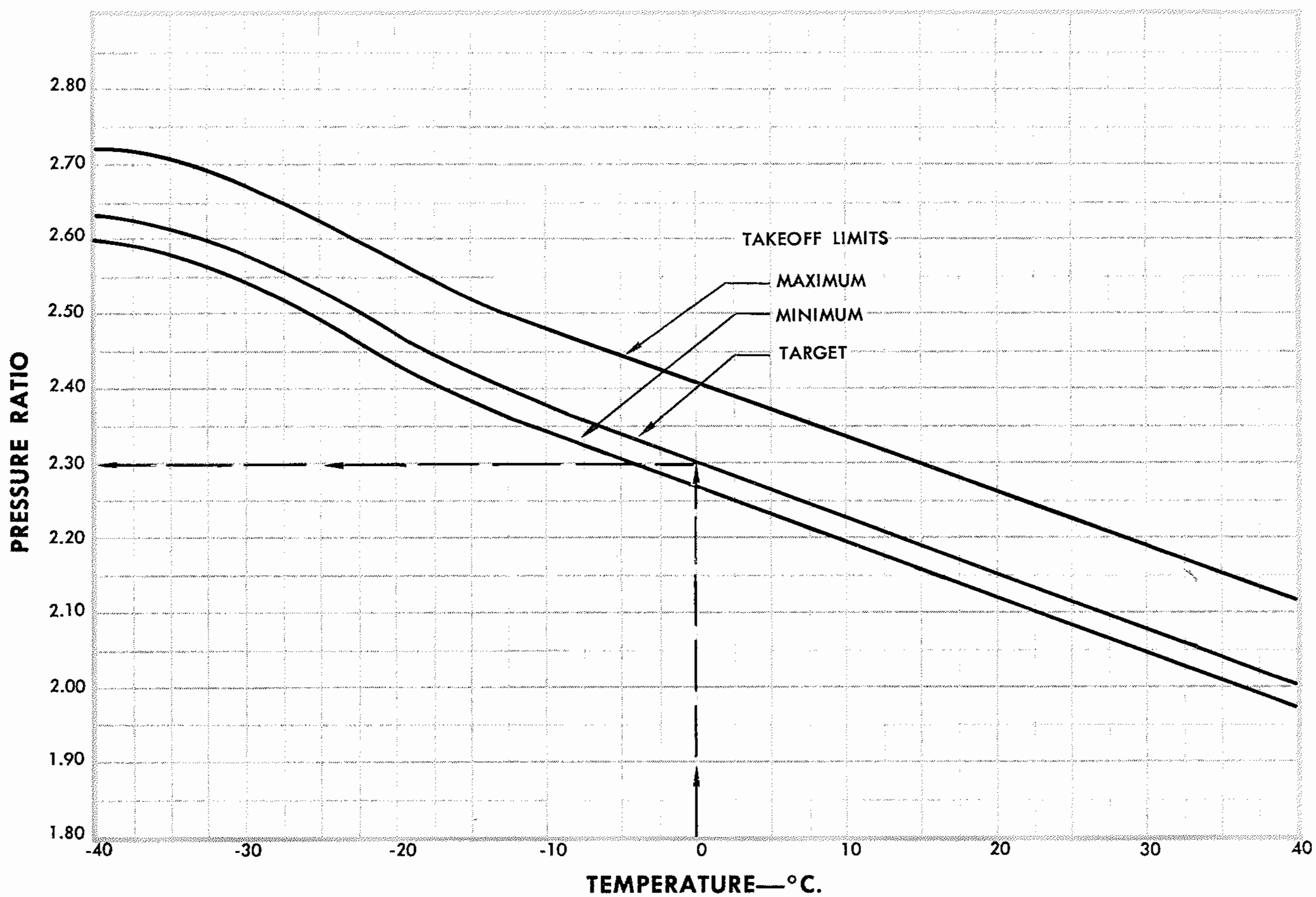


Figure A2-5

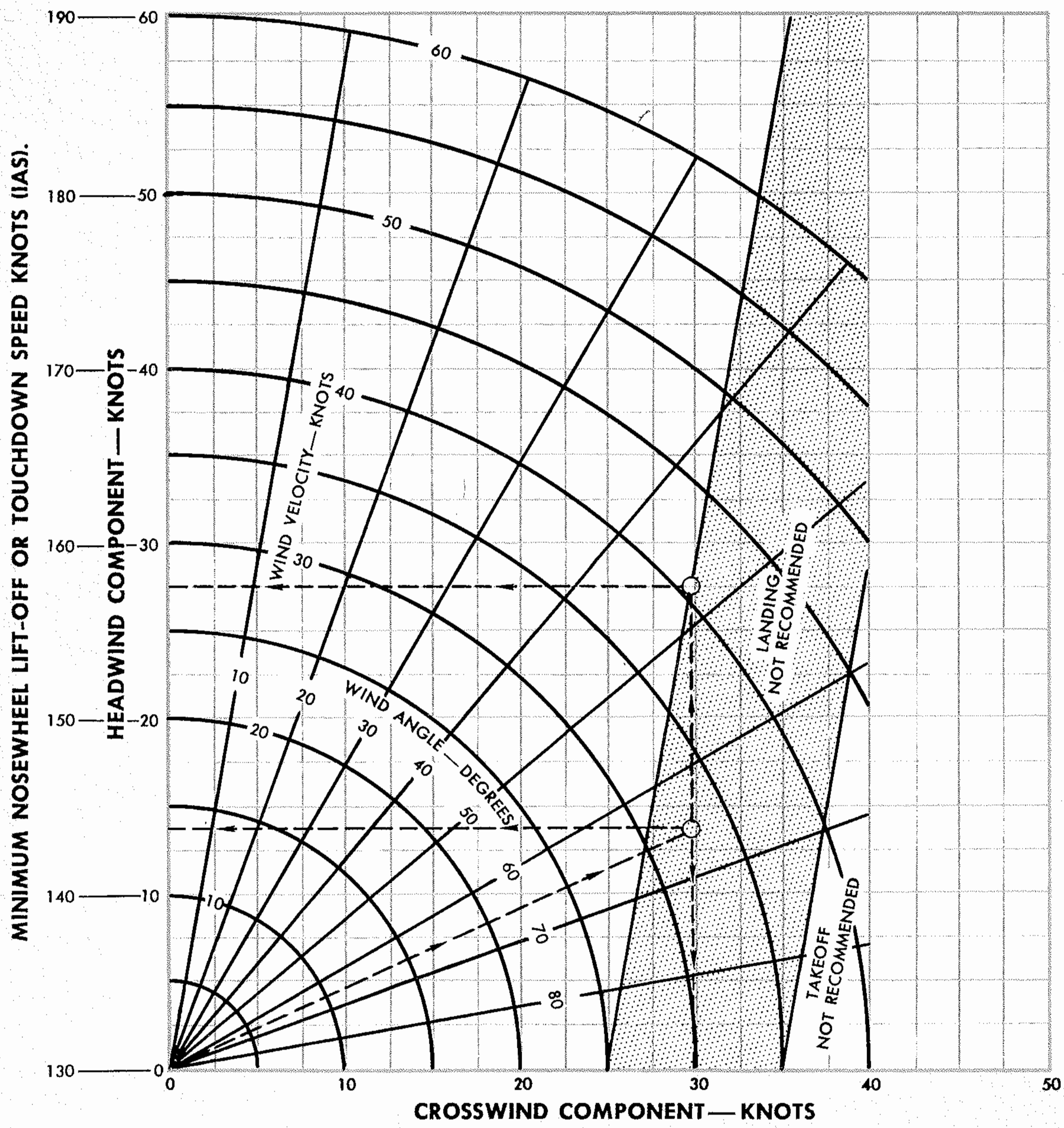
22009D

takeoff and landing crosswind chart

MODEL: F-102A
DATE: 1 MAY 1959
DATA BASIS: FLIGHT TEST

STANDARD DAY

ENGINE: J57-23



NOTES

1. ENTER CHART WITH MAXIMUM GUST VELOCITY.
2. REFER TO TAKEOFF SPEEDS CHART FOR TIRE GROUND LIMIT SPEED.

22064A

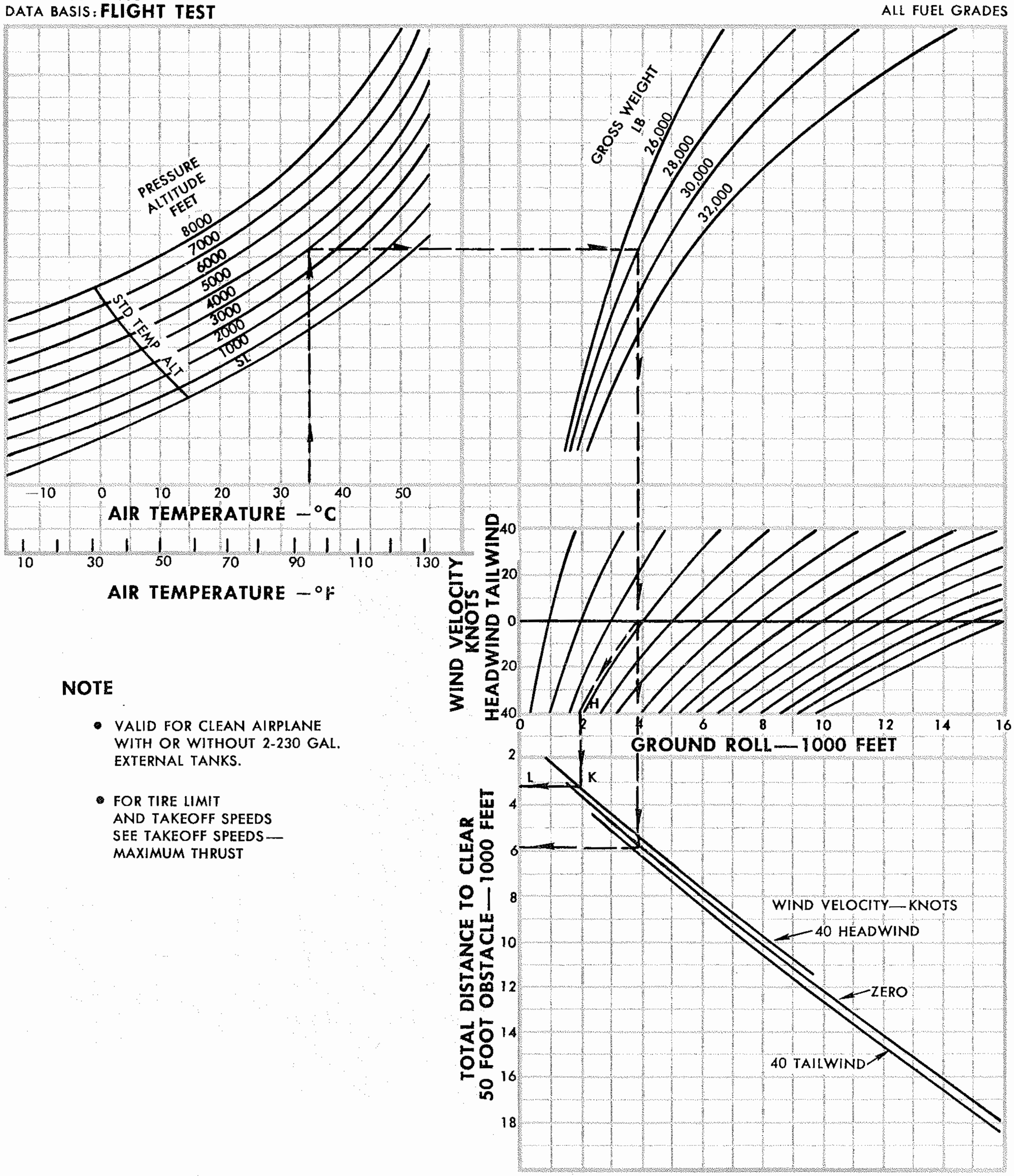
Figure A2-6

takeoff distances – maximum thrust

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

HARD SURFACE RUNWAY

ENGINE: J57-23
ALL FUEL GRADES



NOTE

- VALID FOR CLEAN AIRPLANE WITH OR WITHOUT 2-230 GAL. EXTERNAL TANKS.
- FOR TIRE LIMIT AND TAKEOFF SPEEDS SEE TAKEOFF SPEEDS—MAXIMUM THRUST

22010D

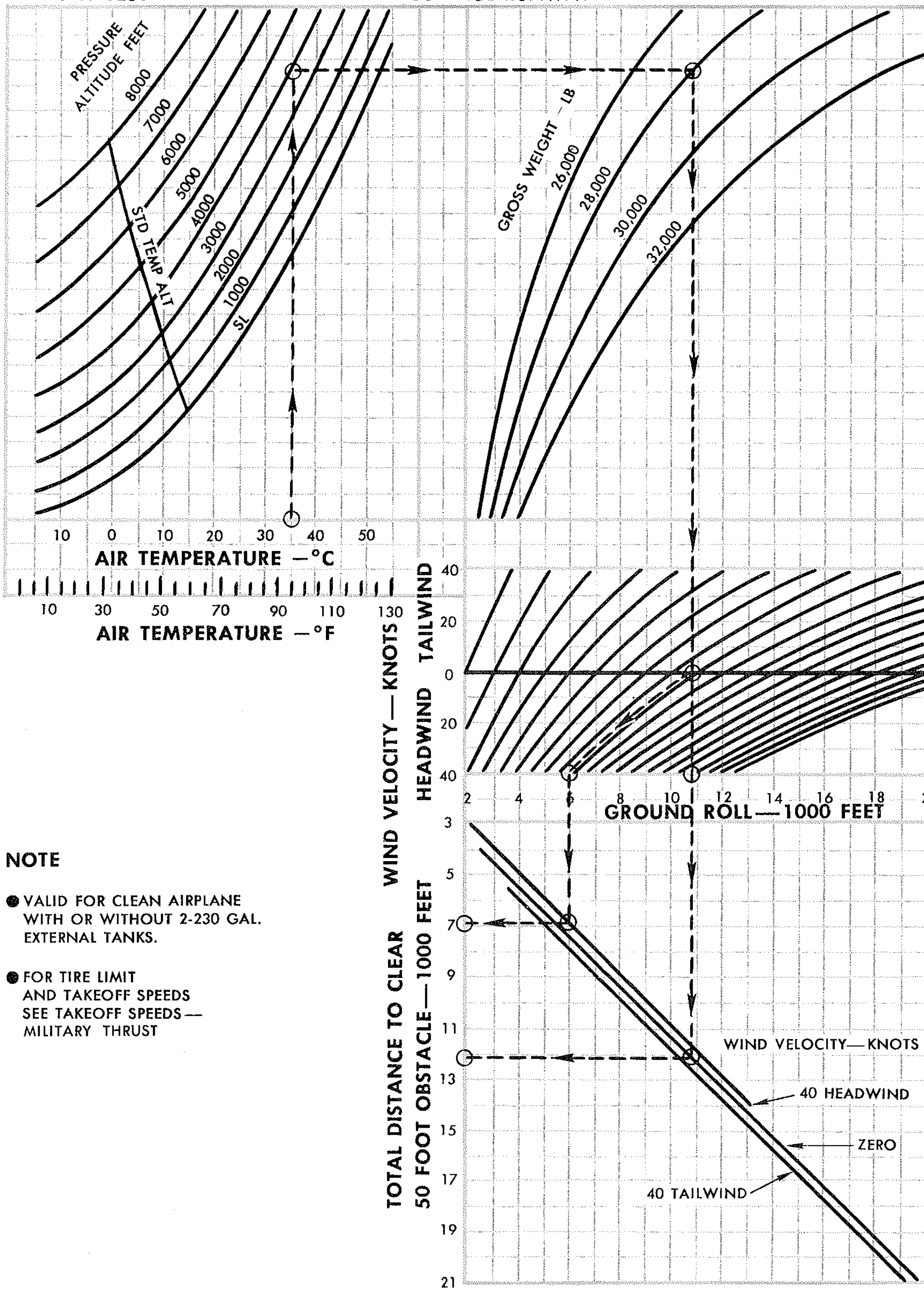
Figure A2-7

takeoff distances military thrust

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

HARD SURFACE RUNWAY

ENGINE: J57-23
ALL FUEL GRADES



NOTE

- VALID FOR CLEAN AIRPLANE WITH OR WITHOUT 2-230 GAL. EXTERNAL TANKS.
- FOR TIRE LIMIT AND TAKEOFF SPEEDS SEE TAKEOFF SPEEDS— MILITARY THRUST

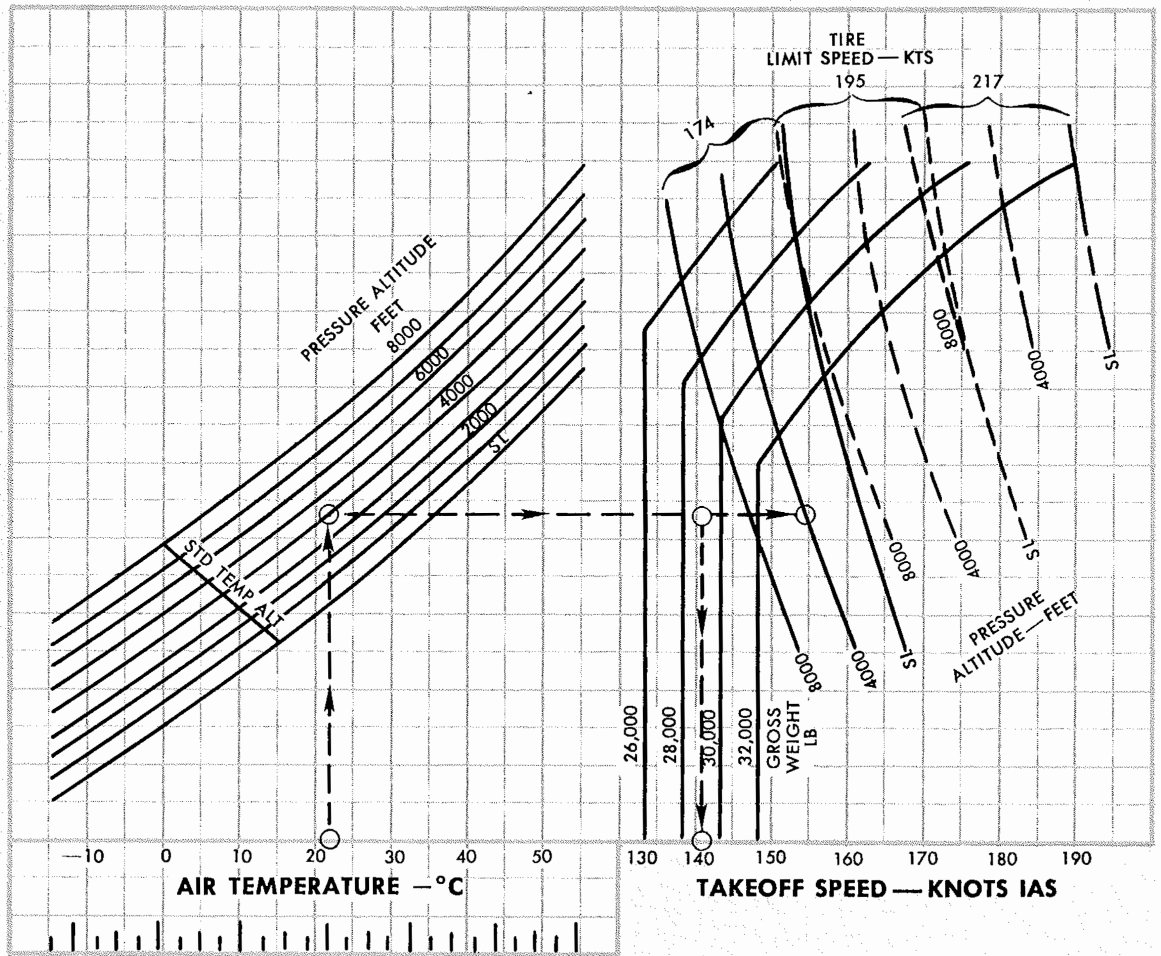
22057D

Figure A2-8

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

takeoff speeds - maximum thrust

ENGINE: J57-23
ALL FUEL GRADES



NOTE

TO DETERMINE TAKEOFF SPEED LIMITATION, ADD HEADWIND OR SUBTRACT TAILWIND FROM ZERO WIND TIRE GROUND LIMIT SPEED.

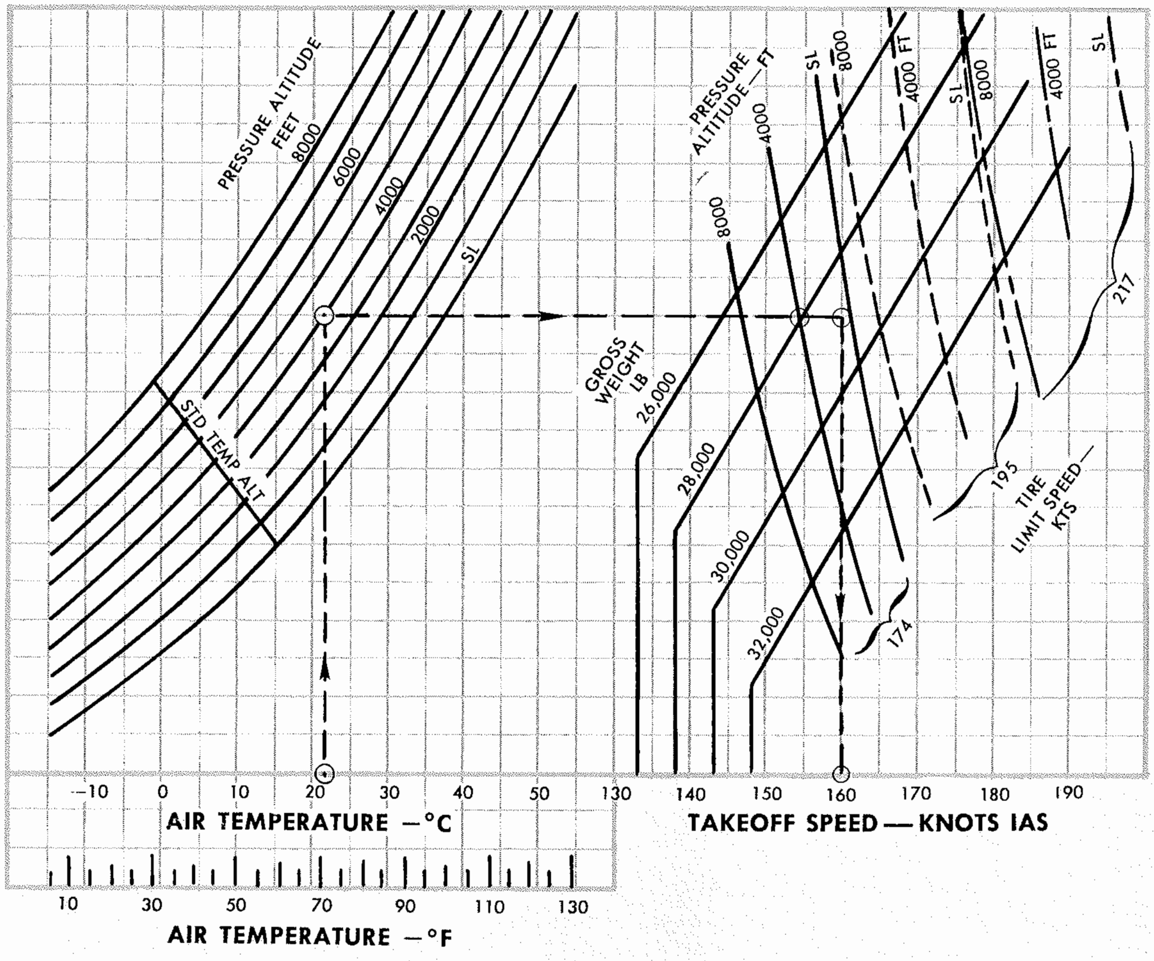
22063A

Figure A2-9

takeoff speeds – military thrust

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23
ALL FUEL GRADES



NOTE
TO DETERMINE TAKEOFF SPEED LIMITATION, ADD HEADWIND OR SUBTRACT TAILWIND FROM ZERO WIND TIRE GROUND LIMIT SPEED.

22062A

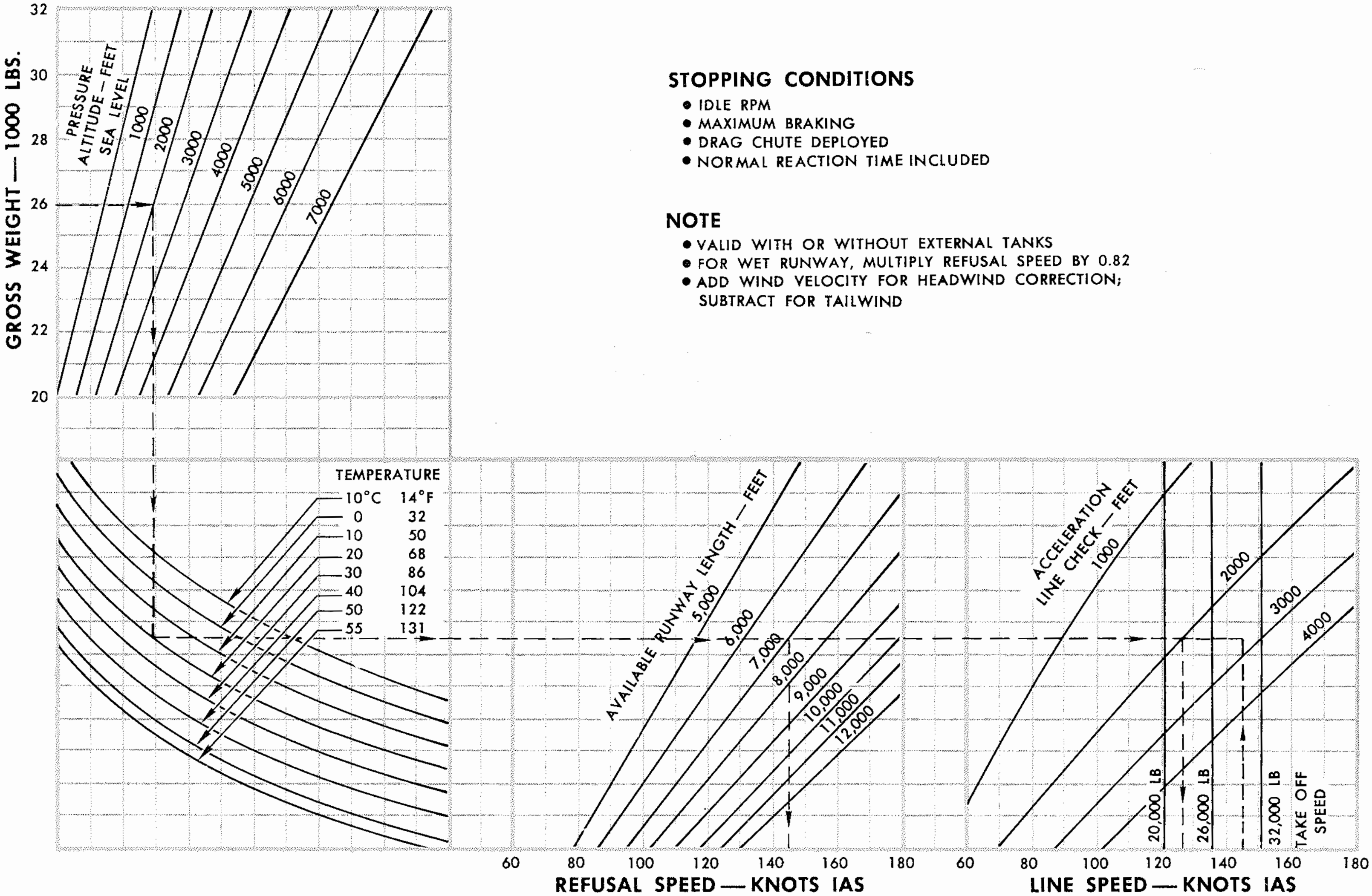
Figure A2-10

maximum refusal speed

DRY, HARD SURFACE RUNWAY • ZERO WIND
MAXIMUM THRUST • CONFIGURATION: GEAR DOWN

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23
ALL FUEL GRADES



STOPPING CONDITIONS

- IDLE RPM
- MAXIMUM BRAKING
- DRAG CHUTE DEPLOYED
- NORMAL REACTION TIME INCLUDED

NOTE

- VALID WITH OR WITHOUT EXTERNAL TANKS
- FOR WET RUNWAY, MULTIPLY REFUSAL SPEED BY 0.82
- ADD WIND VELOCITY FOR HEADWIND CORRECTION; SUBTRACT FOR TAILWIND

Figure A2-11

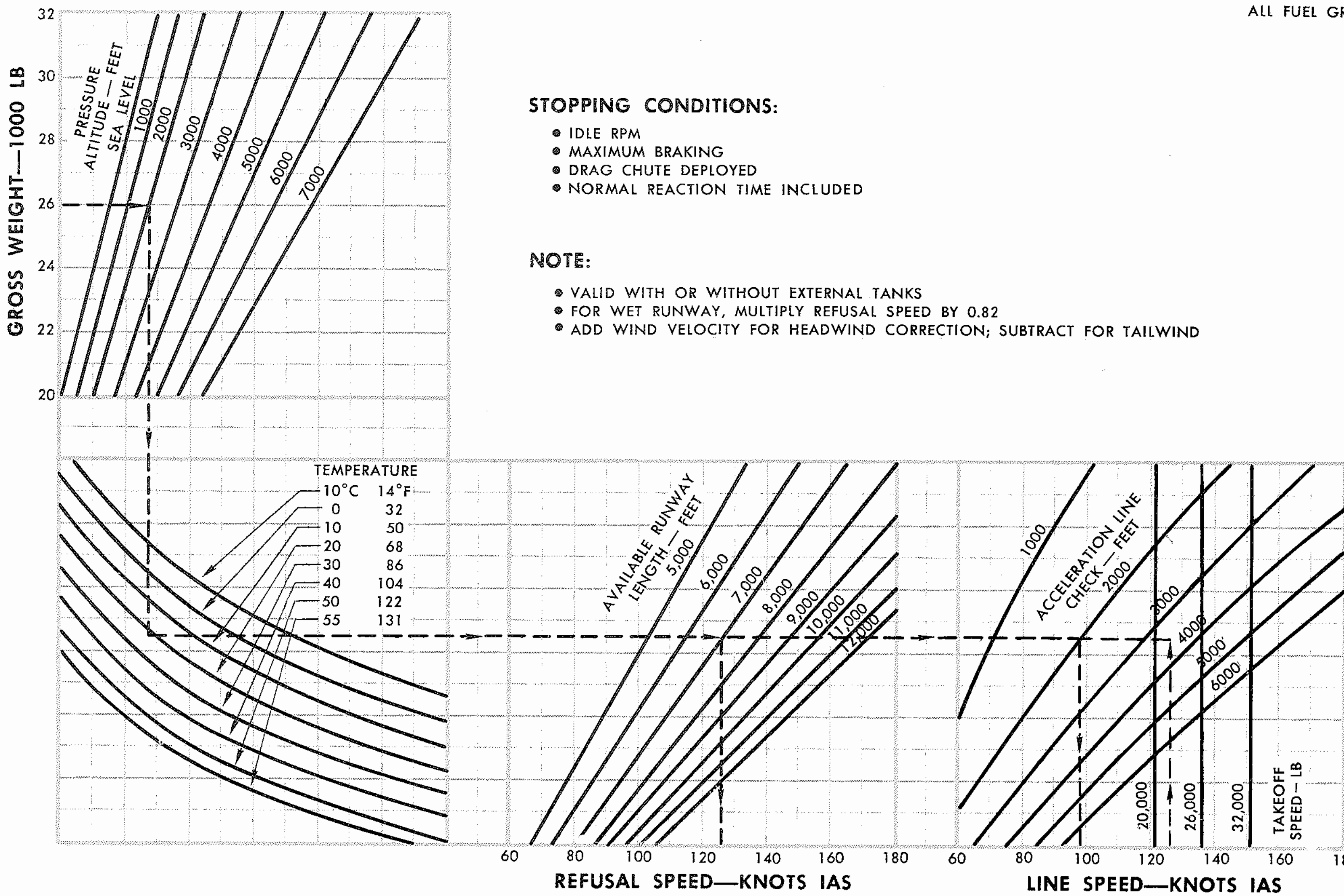
22058B

maximum refusal speed

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

DRY, HARD SURFACE RUNWAY • MILITARY THRUST • ZERO WIND
CONFIGURATION: GEAR DOWN

ENGINE: J57-23
ALL FUEL GRADES



STOPPING CONDITIONS:

- IDLE RPM
- MAXIMUM BRAKING
- DRAG CHUTE DEPLOYED
- NORMAL REACTION TIME INCLUDED

NOTE:

- VALID WITH OR WITHOUT EXTERNAL TANKS
- FOR WET RUNWAY, MULTIPLY REFUSAL SPEED BY 0.82
- ADD WIND VELOCITY FOR HEADWIND CORRECTION; SUBTRACT FOR TAILWIND

Figure A2-12

22059B

PART 3 CLIMB

TABLE OF CONTENTS

CLIMB

Figures A3-2 thru A3-5 are graphical presentations of the climb performance for military thrust and maximum thrust, with or without external tanks. These charts plot gross weight versus distance in climb with lines of constant altitude and guide lines showing how fuel is used during climb. An auxiliary scale of time-to-climb is also shown. Any inflight climb problem may be solved by entering the particular chart at a known gross weight and altitude, then following up, parallel to the nearest guide line, to the desired altitude condition. The distance, time, and fuel used for this climb segment are determined by the differences between initial and final altitude conditions. The chart also shows altitude conditions for cruise, combat, and service ceiling where applicable. The climb schedule for true and calibrated speed and Mach number is shown in table form.

Use

To obtain time, distance, and fuel to climb: enter the chart with the gross weight and altitude at the start of the climb. Follow parallel to the nearest gross weight guide line up to the altitude at end of the climb. Then read directly down to the gross weight scale for the gross weight at end of climb, to the left for distance in climb, and to the right for time-to-climb. The difference between the initial gross weight and the gross weight at end of climb is the fuel used. The figures obtained must then be corrected for temperature variation if a difference exists between actual and standard day conditions. Multiply each percent constant (found in table on the chart) by the number of degrees variation, and apply this percentage correction to the respective values for time, distance, and fuel.

Note

The correction factor is added when temperature is above standard, and subtracted when temperature is below standard.

Sample Problem

Find the time, distance, and fuel used for a military thrust climb to an altitude of 35,500 feet immediately after takeoff from sea level. Airplane gross weight is 27,750 pounds.

a. Enter the chart in figure A3-1 with the gross weight at sea level — 27,750 pounds.

b. Follow parallel to the nearest gross weight guide line up to an altitude of 35,500 feet.

c. Read straight down to the gross weight at end of climb 26,680 pounds. The difference between 27,750 pounds and 26,680 is fuel used — 1070 pounds.

d. Read left to the distance in climb — 78.3 nautical miles.

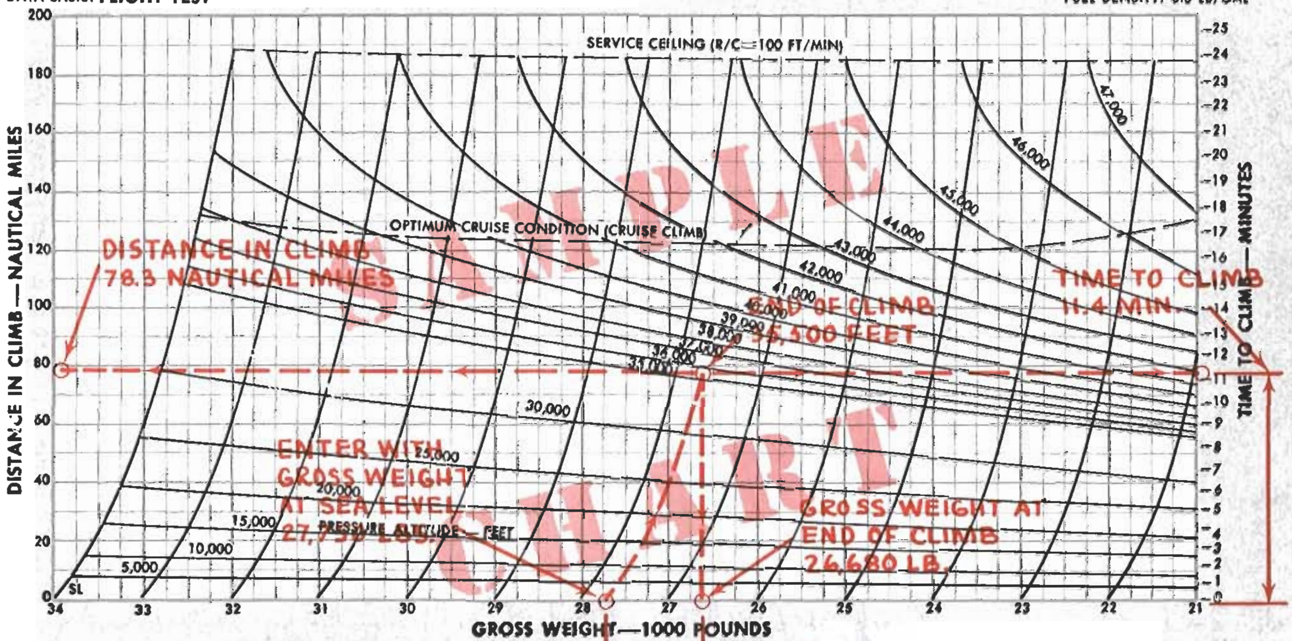
e. Read right to the time-to-climb — 11.4 minutes.

climb-military thrust

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

CONFIGURATION: CLEAN ● STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



CLIMB SPEEDS					
PRESSURE ALTITUDE FEET	IND MACH NO.	IAS KNOTS	TRUE MACH NO.	TAS KNOTS	CAS KNOTS
50,000	.87	213	.90	516	219
45,000	.87	238	.90	516	245
40,000	.87	269	.90	516	276
35,000	.87	302	.90	519	310
30,000	.79	303	.82	483	311
25,000	.72	306	.75	451	314
20,000	.66	307	.68	419	315
15,000	.61	309	.63	393	317
10,000	.56	313	.58	370	321
5,000	.52	318	.54	351	326
SL	.48	322	.50	331	331

CLIMB SCHEDULE

	PERCENT CORRECTION PER DEGREE TEMPERATURE VARIATION	
	F	C
TIME	0.8	1.4
DISTANCE	0.9	1.6
FUEL	0.8	1.4

NOTE

THE CORRECTION FACTOR IS ADDED WHEN TEMPERATURE IS ABOVE STANDARD, AND SUBTRACTED WHEN TEMPERATURE IS BELOW STANDARD.

22122
22013D

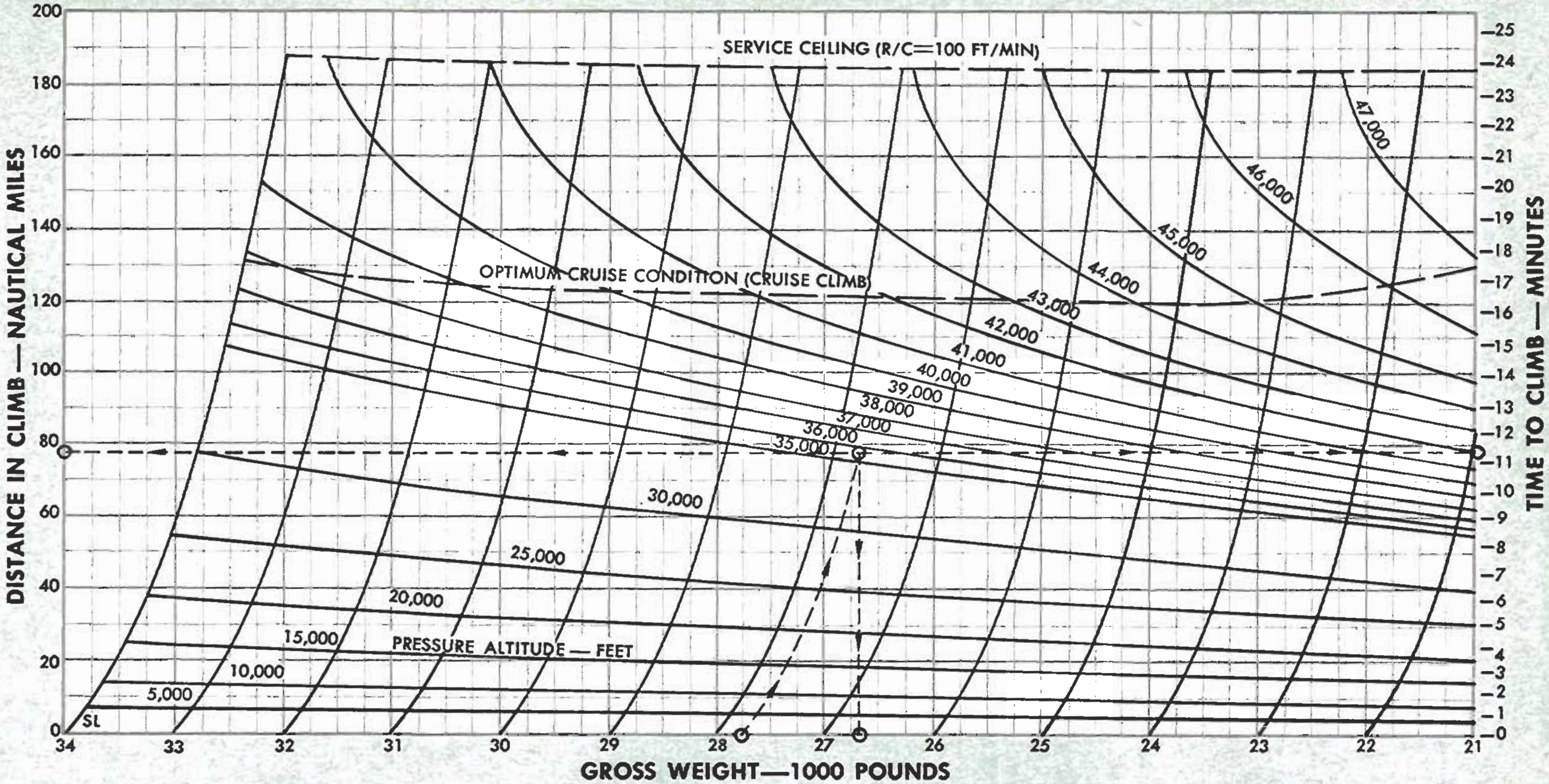
Figure A3-1

climb-military thrust

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

CONFIGURATION: CLEAN • STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



CLIMB SPEEDS

PRESSURE ALTITUDE FEET	IND MACH NO.	IAS KNOTS	TRUE MACH NO.	TAS KNOTS	CAS KNOTS
50,000	.87	213	.90	516	219
45,000	.87	238	.90	516	245
40,000	.87	269	.90	516	276
35,000	.87	302	.90	519	310
30,000	.79	303	.82	483	311
25,000	.72	306	.75	451	314
20,000	.66	307	.68	419	315
15,000	.61	309	.63	393	317
10,000	.56	313	.58	370	321
5,000	.52	318	.54	351	326
SL	.48	322	.50	331	331

	PERCENT CORRECTION PER DEGREE TEMPERATURE VARIATION	
	F	C
TIME	0.8	1.4
DISTANCE	0.9	1.6
FUEL	0.8	1.4

NOTE

THE CORRECTION FACTOR IS ADDED WHEN TEMPERATURE IS ABOVE STANDARD, AND SUBTRACTED WHEN TEMPERATURE IS BELOW STANDARD.

22013D

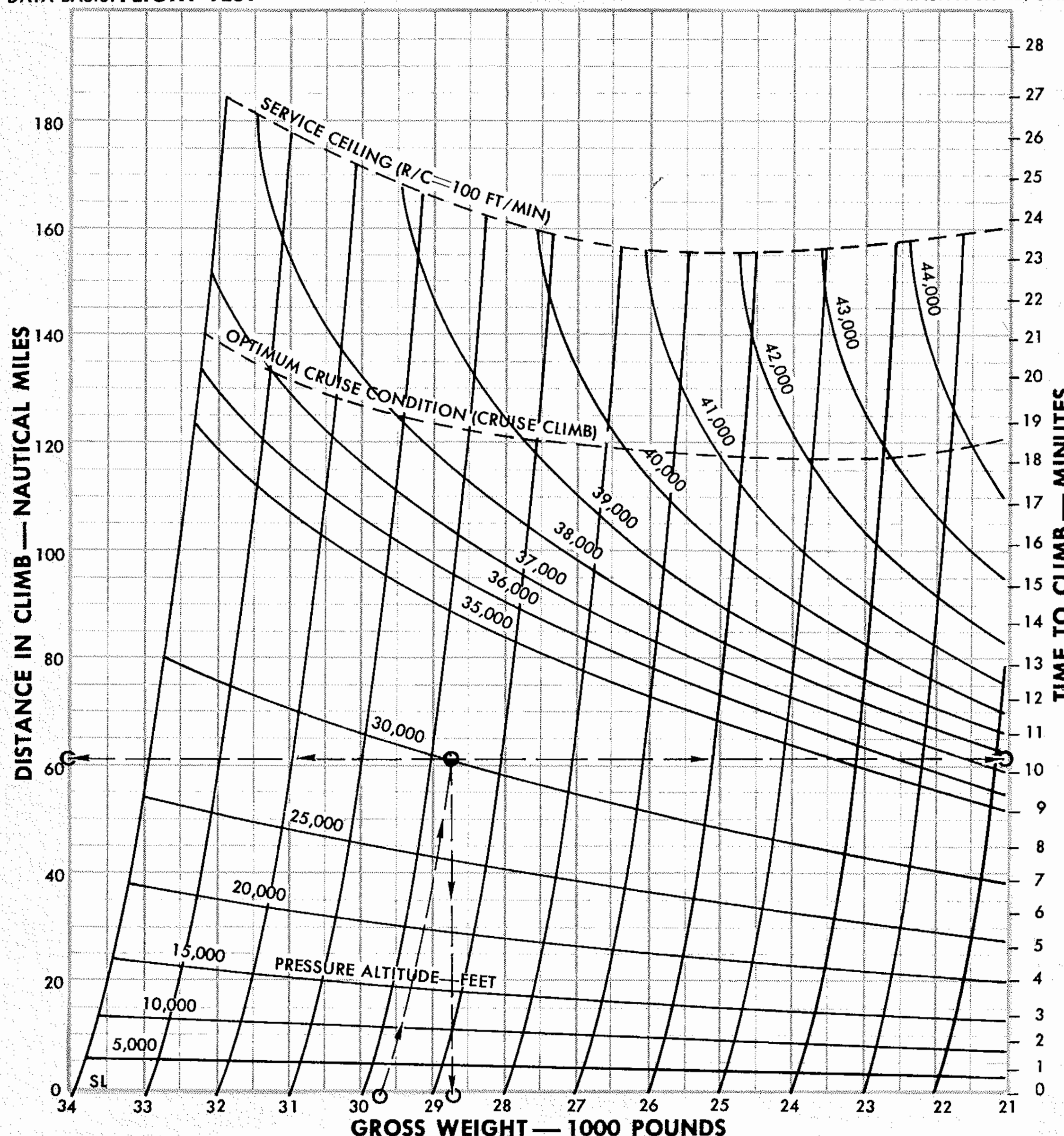
Figure A3-2

climb-military thrust

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD DAY

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



	PERCENT CORRECTION PER DEGREE TEMPERATURE VARIATION	
	F	C
TIME	0.8	1.5
DISTANCE	0.9	1.6
FUEL	0.9	1.6

NOTE:
THE CORRECTION FACTOR IS ADDED WHEN TEMPERATURE IS ABOVE STANDARD, AND SUBTRACTED WHEN TEMPERATURE IS BELOW STANDARD.

CLIMB SPEEDS

PRESSURE ALTITUDE FEET	IND MACH NO.	IAS KNOTS	TRUE MACH NO.	TAS KNOTS	CAS KNOTS
45,000	.78	210	.81	462	216
40,000	.78	237	.81	462	244
35,000	.78	266	.81	464	273
30,000	.70	266	.73	427	273
25,000	.64	270	.67	400	275
20,000	.59	274	.61	376	281
15,000	.55	277	.57	354	285
10,000	.51	284	.53	337	292
5,000	.48	288	.49	320	292
SL	.45	295	.46	304	304

22019E

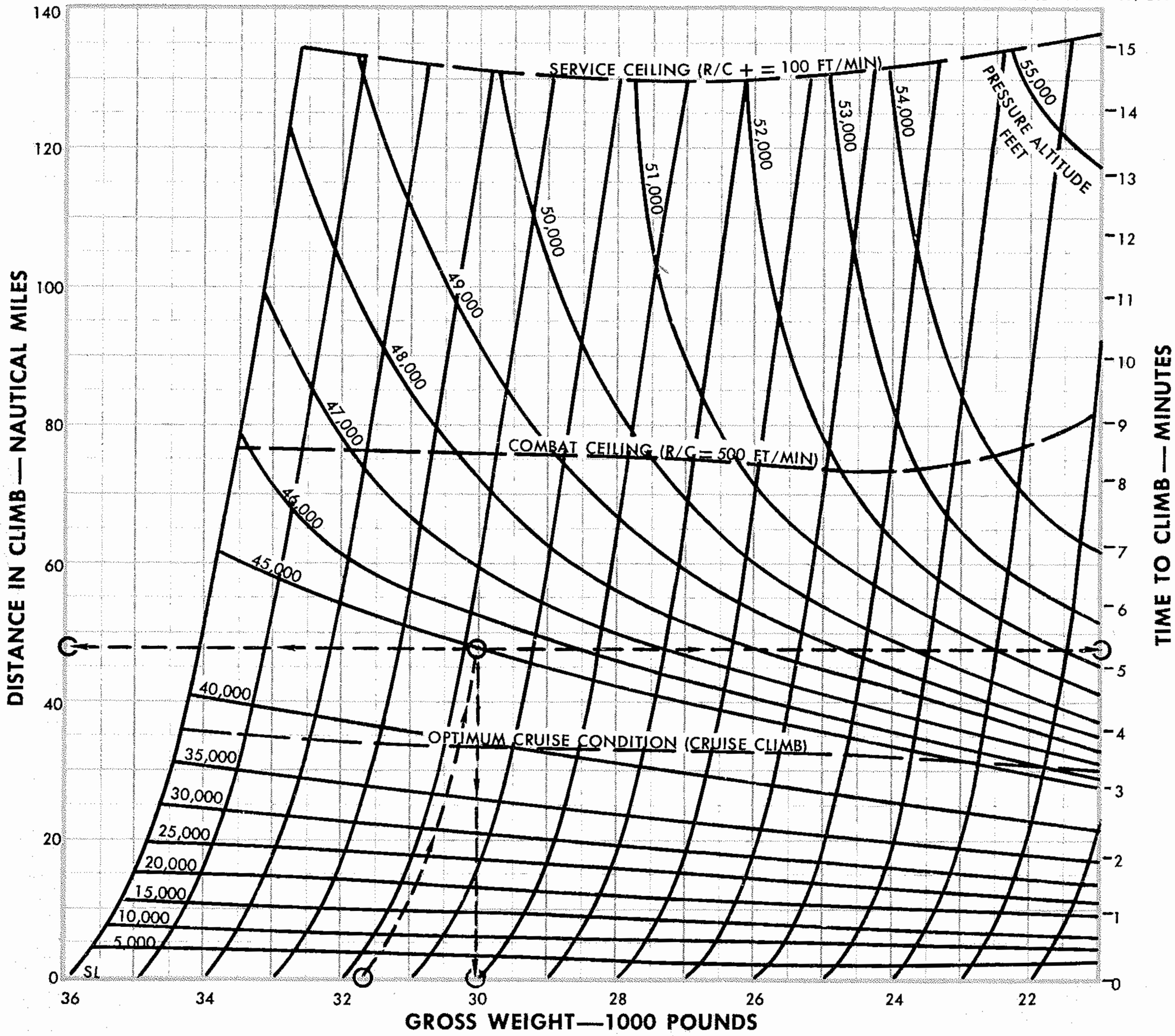
Figure A3-3

climb-maximum thrust

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

CONFIGURATION: CLEAN • STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



	PERCENT CORRECTION PER DEGREE TEMPERATURE VARIATION	
	F	C
TIME	0.8	1.5
DISTANCE	0.8	1.5
FUEL	0.9	1.6

NOTE:

THE CORRECTION FACTOR IS ADDED WHEN TEMPERATURE IS ABOVE STANDARD, AND SUBTRACTED WHEN TEMPERATURE IS BELOW STANDARD.

CLIMB SPEEDS

PRESSURE ALTITUDE FEET	IND MACH NO.	IAS KNOTS	TRUE MACH NO.	TAS KNOTS	CAS KNOTS
55,000	.89	197	.93	533	203
50,000	.89	222	.93	533	228
45,000	.89	249	.93	533	256
40,000	.89	279	.93	533	287
35,000	.89	313	.93	536	322
30 000	.89	347	.93	547	356
25,000	.89	386	.93	558	396
20,000	.89	424	.92	566	435
15,000	.88	464	.92	576	476
10,000	.88	503	.92	585	516
5,000	.88	545	.91	593	558
SL	.88	588	.91	602	602

22011E

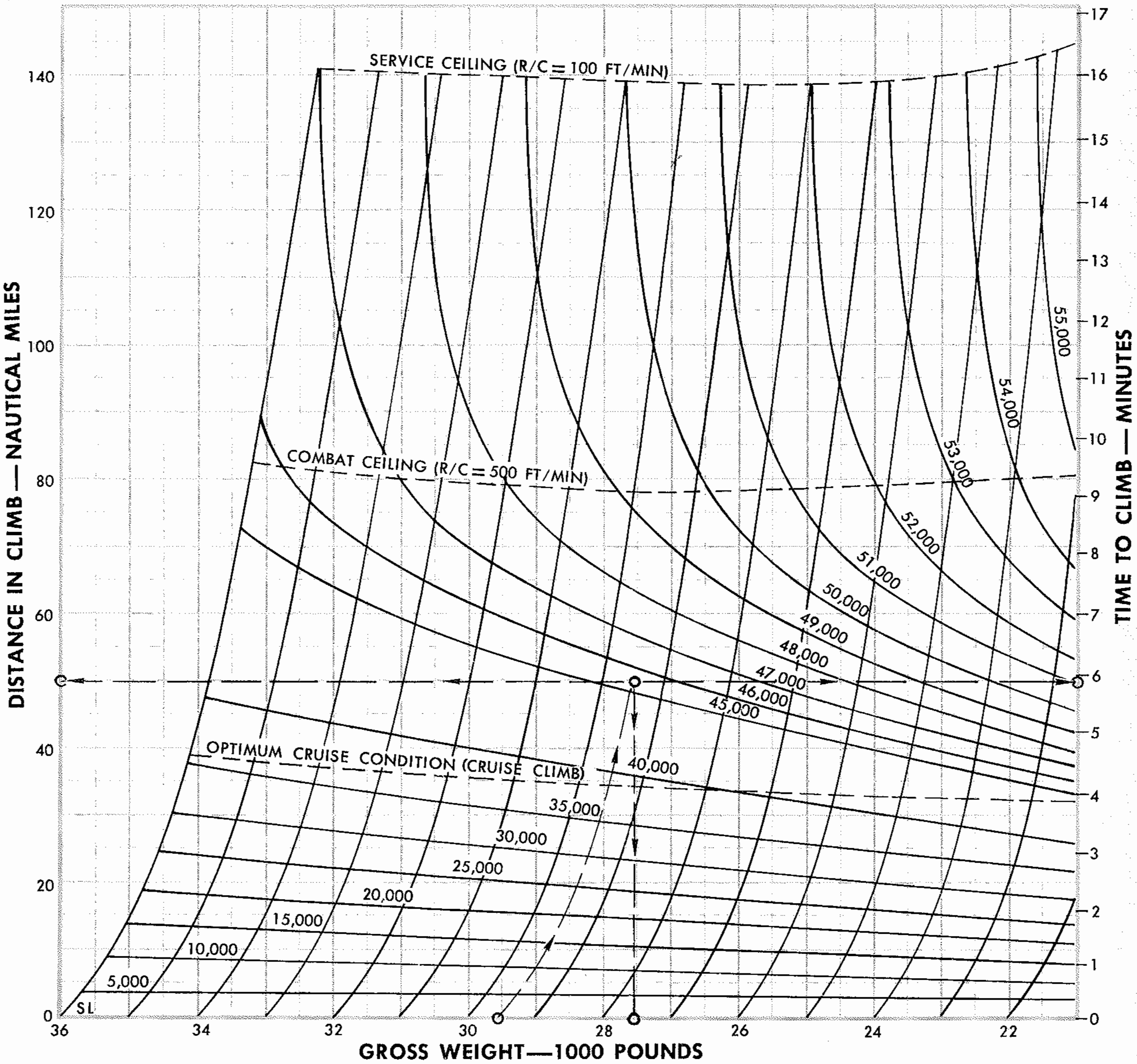
Figure A3-4

climb-maximum thrust

CONFIGURATION: TWO 230-GAL EXTERNAL TANKS ● STANDARD DAY

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL.



	PERCENT CORRECTION PER DEGREE TEMPERATURE VARIATION	
	F	C
TIME	0.6	1.2
DISTANCE	0.6	1.2
FUEL	0.6	1.1

NOTE:
THE CORRECTION FACTOR IS ADDED WHEN TEMPERATURE IS ABOVE STANDARD, AND SUBTRACTED WHEN TEMPERATURE IS BELOW STANDARD.

CLIMB SPEEDS

PRESSURE ALTITUDE FEET	IND MACH NO.	IAS KNOTS	TRUE MACH NO.	TAS KNOTS	CAS KNOTS
55,000	.88	193	.92	525	199
50,000	.88	217	.92	525	223
45,000	.88	244	.92	525	251
40,000	.88	274	.92	525	282
35,000	.88	308	.92	527	316
30,000	.87	334	.90	530	343
25,000	.84	362	.88	528	371
20,000	.82	389	.85	522	399
15,000	.79	409	.82	511	419
10,000	.75	424	.78	496	435
5,000	.71	434	.73	475	445
SL	.66	438	.68	449	449

22017D

Figure A3-5

PART 4 RANGE

TABLE OF CONTENTS

MISSION PROFILE

Mission profile charts are shown for the clean airplane in figure A4-4, for the airplane carrying external fuel tanks in figure A4-5, and for the airplane with external fuel tanks dropped when empty in figure A4-6. These charts are basically a plot of altitude versus distance. They show the initial military thrust climb path to the optimum cruise climb condition with reference lines of time and fuel used for constant altitude cruise and for the optimum cruise climb. The time and fuel required to take off, climb, and cruise a given distance at any specified altitude is rapidly determined by a linear interpolation between reference lines. The fuel reference lines include takeoff allowances as noted but the time reference lines do not include any takeoff allowance. No allowance is made for landing reserves. The altitude for maximum range does not apply to short range flights. The cruise altitude for a short range mission is determined by the maximum range point along a constant fuel line in the vicinity of the desired range. The effect of wind may be closely approximated by computing the overall air distance required in cruise (neglecting the wind effect on climb since very little time is spent in this phase of the mission). The chart includes tabular data concerning speeds, fuel flows and engine power conditions for cruise at constant altitude and the cruise-climb. The climb schedule is tabulated along the altitude scale. Takeoff weights are noted; the effect of weight change due to fuel used are included.

Use

The fuel and time required to fly a given air distance at a specified altitude are determined by entering the chart with the desired air distance, reading up to the specified altitude, and interpolating (if necessary) between reference lines for fuel and time. Given a specific fuel quantity, the reverse procedure may be used to determine the air distance. For a given fuel quantity the maximum available range is obtained by locating the fuel quantity available for the flight (total fuel quantity less desired fuel reserve for loiter, descent, and landing) on the fuel reference lines (interpolating if necessary), then following the fuel reference line to the altitude at which the air distance is a maximum. The reverse procedure is used to find the minimum fuel required to fly a given air distance. To obtain the fuel and time required to fly a given ground distance with head or tail winds, correct the ground distance to an equivalent air distance and use this distance in the chart.

The recommended procedure for wind correction is as follows:

1. Compute average true airspeed (TAS) (Distance, time with zero wind conditions).
2. Apply wind to obtain ground speed (Subtract headwind; add tailwind).
3. Compute time with wind effect (Distance + ground speed).
4. To obtain actual air distance (Multiply time with wind effect × TAS).
5. Enter profile chart at cruise altitude and corrected air distance — interpolate for fuel required.

Sample Problem

Find the minimum fuel and the time required to fly a distance of 250 nautical miles with 30-knot headwind. Takeoff gross weight is 28,150 pounds.

a. Enter figure A4-1 with the air distance — 250 nautical miles.

b. Read up to the altitude at which the fuel reference lines indicate maximum range (cruise-climb). Fuel required (no wind) — 2995 pounds. Time — 31.0 minutes.

c. Subtract headwind — fuel required with 30-knot headwind is 3100 pounds and time is 33.1 minutes.

To obtain this range, the mission profile could be flown with a military thrust climb to the optimum cruise altitude (approximately 38,000 feet) following takeoff.

INTERCEPT PROFILE

The intercept profile for the clean airplane, figure A4-7, is presented in a manner similar to the mission profiles. The major differences are: (1) climb with maximum thrust and (2) constant altitude cruise at military thrust. The intercept profile should be restricted to missions that require minimum time.

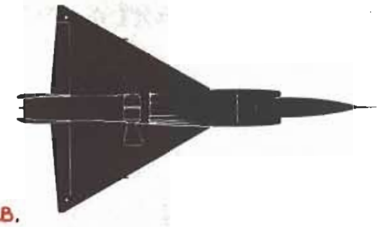
Use

The intercept profile chart is used in the same manner as the mission profile charts.

OPTIMUM RETURN PROFILE

The optimum return profiles are shown in figure A4-8 for the clean airplane, and figure A4-9 for airplanes carrying external fuel tanks. These profiles indicate the minimum fuel required to fly a given air distance; or conversely, the maximum distance that can be flown with a given amount of fuel remaining at any given altitude.

mission profile

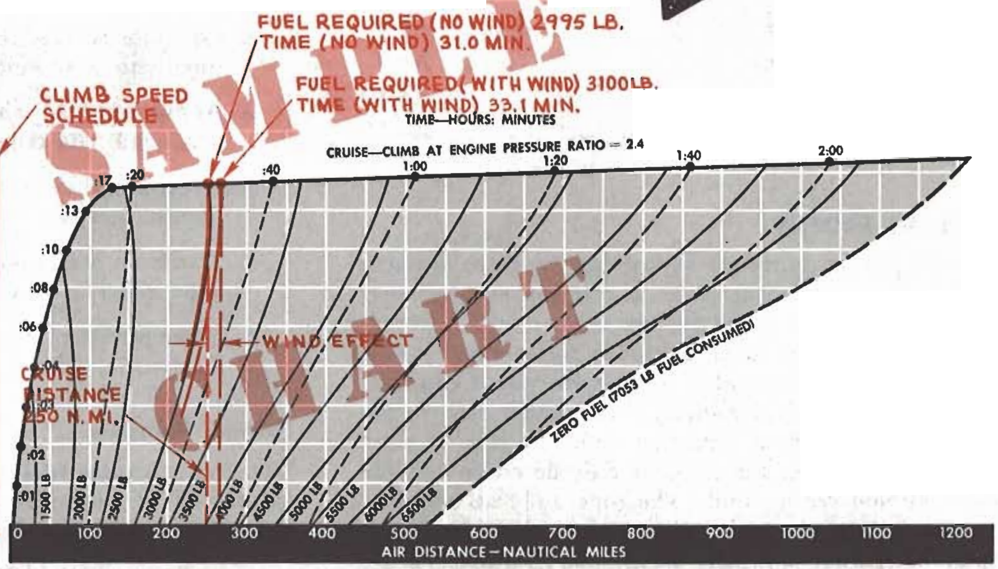


MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
TAKEOFF GROSS WEIGHT: 28,150 LBS.
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

MILITARY THRUST CLIMB					
TAS KNOTS	CAS KNOTS	IAS KNOTS	IND MACH NO.	TRUE MACH NO.	ALT 1000 FEET
516	219	213	.87	.90	50
516	245	238	.87	.90	45
516	276	269	.87	.90	40
519	310	302	.87	.90	35
483	311	303	.79	.82	30
451	314	306	.72	.75	25
419	315	307	.66	.68	20
393	317	309	.61	.63	15
370	321	313	.56	.58	10
351	326	318	.52	.54	5
331	331	322	.48	.50	SL



CRUISE SETTINGS

CRUISE CLEAN CONFIGURATION							
ALTITUDE FEET	TRUE MACH NO.*	IND MACH NO.*	IAS KNOTS*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*	ENGINE PRESSURE RATIO*
CRUISE-CLIMB	.92	.89	251	259	530	2415	2.44
45,000	.91	.87	242	250	520	2440	2.36
40,000	.90	.87	270	278	518	2410	2.18
35,000	.83	.80	276	284	480	2430	1.98
30,000	.74	.72	271	279	437	2480	1.84
25,000	.67	.65	272	280	405	2600	1.72
20,000	.62	.59	274	282	378	2740	1.61
15,000	.56	.54	276	284	352	2845	1.53
10,000	.51	.49	275	283	326	2940	1.45
5,000	.47	.46	277	285	307	3135	1.39
SL	.43	.42	279	287	287	3300	1.34

*APPROXIMATE

TAKEOFF FUEL ALLOWANCE

● ● ● ● TIME TO CLIMB
— — — ZERO FUEL REMAINING
——— FUEL CONSUMED
——— CRUISE-CLIMB FLIGHT PATH
—·—·— TIME (START, TAXI AND TAKEOFF NOT INCLUDED)

22117
22047

NOTE
1. Fuel allowance for start, taxi, and takeoff (1130 lbs) included
2. No allowance or reserves made for loiter descent or landing
3. Use Military Thrust for climb (See Military Thrust Climb Chart for detail information)
4. Cruise at recommended Mach No.

Figure A4-1

The return flight path requires a military thrust climb from lower altitudes, to the optimum return altitude and either a cruise-climb or a constant altitude cruise as required. For short range returns or low fuel quantities, the return is made at the initial altitude. The recommended return flight paths are indicated on the charts by different shaded areas and notes. Operating conditions for climb and cruise are shown in the tables along with the profile.

Note

Fuel required for loiter, descent, or landing is not included in these profiles.

Use

To use the charts, enter the respective chart at the desired return distance, read up to the initial airplane altitude and interpolate for the fuel required to return. The desired fuel reserves must be added to the fuel required. If the chart is entered with the fuel available for return at the initial airplane altitude, read down for the return distance. In this case, the fuel reserve must be subtracted from the total fuel available before entering the chart. Wind effects are computed in the same manner as discussed under the mission profiles.

Sample Problem

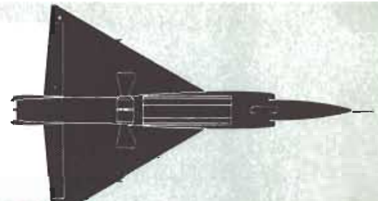
Find the fuel required to return a distance of 300 nautical miles under 30-knot wind conditions.

a. Enter figure A4-2 at the return distance—300 nautical miles.
b. Read up to the initial altitude—35,000 feet.
c. Read the fuel required for return (no wind)—1330 pounds. Fuel required with wind—1270 pounds. Add required reserve fuel to the return fuel to get total fuel required.

The flight path would be as follows:

a. Climb with military thrust to cruise altitude 45,800 feet using schedule as indicated in the climb table.
b. Cruise-climb to destination—use cruise table.
c. Descend to base and land.

optimum return profile

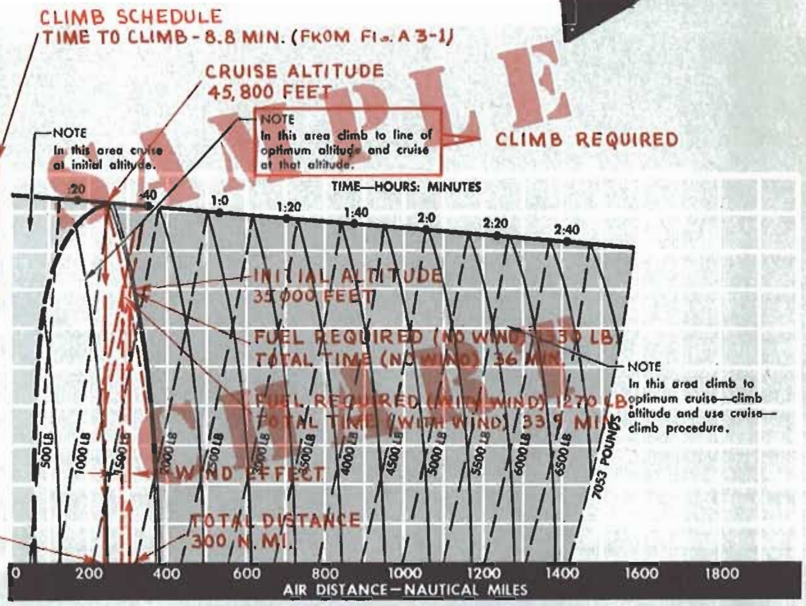


MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
TAKEOFF GROSS WEIGHT: 28,150 LBS.
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

MILITARY THRUST CLIMB					
TAS KNOTS	CAS KNOTS	IAS KNOTS	IND MACH NO.	TRUE MACH NO.	ALT 1000 FEET
516	219	213	.87	.90	50
516	245	238	.87	.90	45
516	276	269	.87	.90	40
519	310	302	.87	.90	35
483	311	303	.79	.82	30
451	314	306	.72	.75	25
419	315	307	.66	.68	20
393	317	309	.61	.63	15
370	321	313	.56	.58	10
351	326	318	.52	.54	5
331	331	322	.48	.50	SL



CRUISE TABLE
TRUE AIRSPEED FOR
CRUISE AT 45,800 FT.
529 KNOTS

CRUISE-CLEAN CONFIGURATION							
ALTITUDE FEET	TRUE MACH NO.*	IND MACH NO.*	IAS KNOTS*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*	ENGINE PRESSURE RATIO*
CRUISE-CLIMB	.92	.89	251	259	530	2415	2.44
45,000	.91	.87	242	250	520	2440	2.36
40,000	.90	.87	270	278	518	2410	2.18
35,000	.83	.80	276	284	480	2430	1.98
30,000	.74	.72	271	279	437	2480	1.84
25,000	.67	.65	272	280	405	2600	1.72
20,000	.62	.59	274	282	378	2740	1.61
15,000	.56	.54	276	284	352	2845	1.53
10,000	.51	.49	275	283	326	2940	1.45
5,000	.47	.46	277	285	307	3135	1.39
SL	.43	.42	279	287	287	3300	1.34

*APPROXIMATE

22121
22051

- — — — — FUEL REQUIRED
- ● ● ● TIME AT CRUISING ALTITUDE
- ━ ━ ━ LINE OF BEST RANGE FOR CONSTANT ALTITUDE
- ——— CLIMB PATH GUIDE LINES
- ━━━ LINE OF BEST RANGE FOR CRUISE-CLIMB ALTITUDE

NOTE:
(1) Fuel required at any point includes Military Thrust climb to cruise—climb altitude (if below that).
(2) No allowance made for loiter, descent or landing.
(3) Best cruise condition is determined by intersection of climb path guide line and lines of best range.
(4) Cruise at recommended Mach No.

Figure A4-2

NAUTICAL MILES PER 1000 POUNDS OF FUEL

Charts of nautical miles per 1000 pounds of fuel are provided in figures A4-10 through A4-29 for the clean airplane and for the airplane with external tanks. These charts present the constant altitude cruise specifics as a function of Mach number at 5000-foot altitude intervals from sea level to 45,000 feet. Various gross weights covering the weight range of the airplane have been considered. Recommended cruise speeds for maximum range cruise and for maximum endurance are shown for the instantaneous weights. Engine fuel flow covering the full range of operation are included on the charts. Corrections to the nautical miles per pound of fuel under wind conditions may be made as indicated on the charts.

Note

Correction factors are included on figures A4-18 through A4-25 to provide emergency cruise information. These correction factors provide range and optimum cruise speeds when the armament bay doors are open and aft armament launchers extended. Reducing the suggested normal optimum cruise speed by 13% will provide the emergency cruise speed. Reduce the suggested normal miles per pound figure for a given altitude by 40% for airplanes with a clean configuration, and by 37% for airplanes with external tanks, to arrive at the more accurate mile-per-pound range information.

Use

To find the nautical miles per 1000 pounds of fuel for a given cruise speed and gross weight, enter the chart with the desired cruise speed, read up to the gross weight, and then to the left for the nautical miles per 1000 pounds of fuel. Fuel flow may be determined by interpolation. Engine thrust settings are determined by interpolation between lines of constant engine pressure ratio.

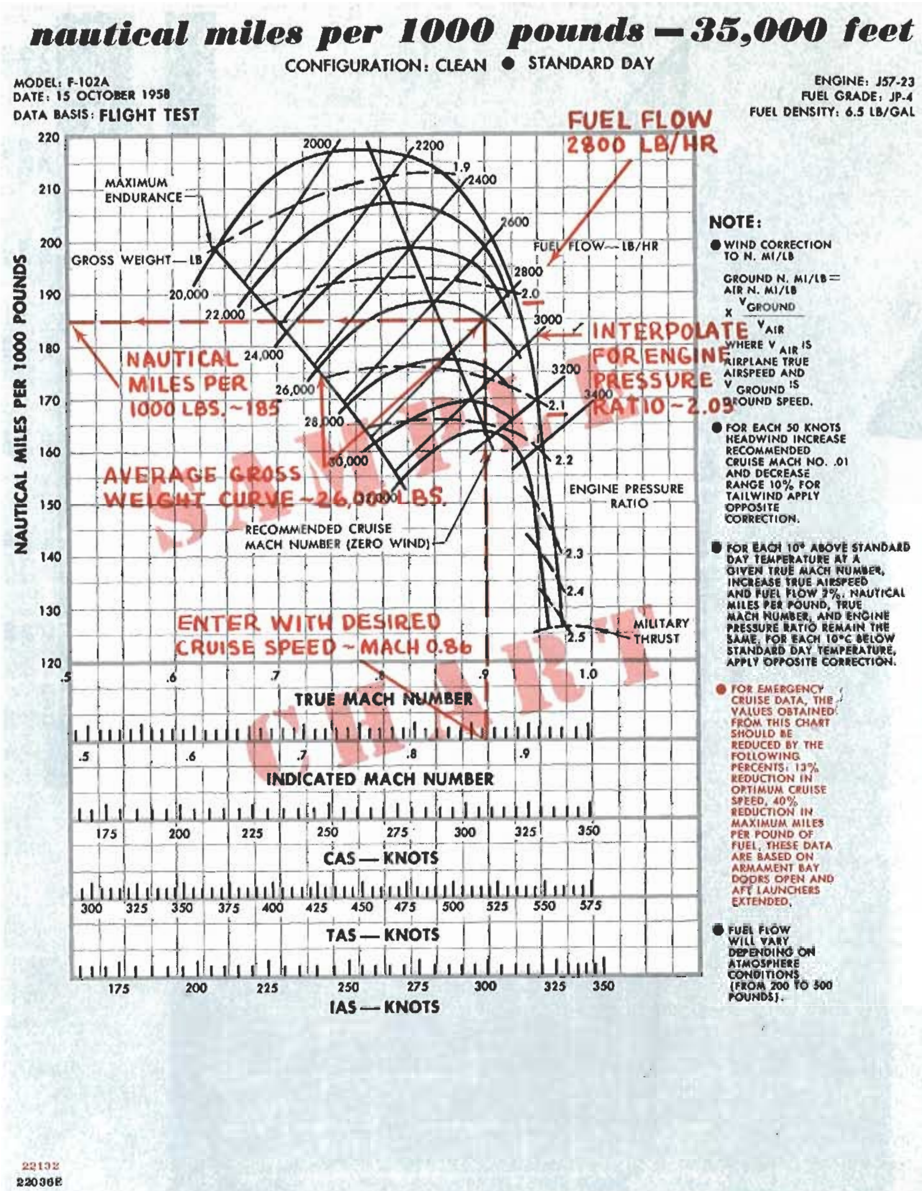


Figure A4-3

Sample Problem

Find the distance that can be flown with 1000 pounds of fuel at an altitude of 35,000 feet and at 0.86 Mach number. The airplane is in the clean configuration and the average gross weight is 26,000 pounds.

a. Enter figure A4-3 (Nautical Miles Per 1000 pounds 35,000 Feet — Clean) with the cruise speed Mach 0.86.

b. Read up to the average gross weight — 26,000 pounds. At this point fuel flow and engine pressure ratio can be interpolated — 2800 pounds per hour fuel flow and an engine pressure ratio setting of approximately 2.03.

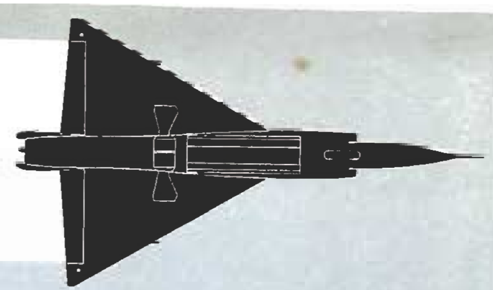
c. Read to the left for the nautical miles per 1000 pounds — 185 nautical miles.

mission profile

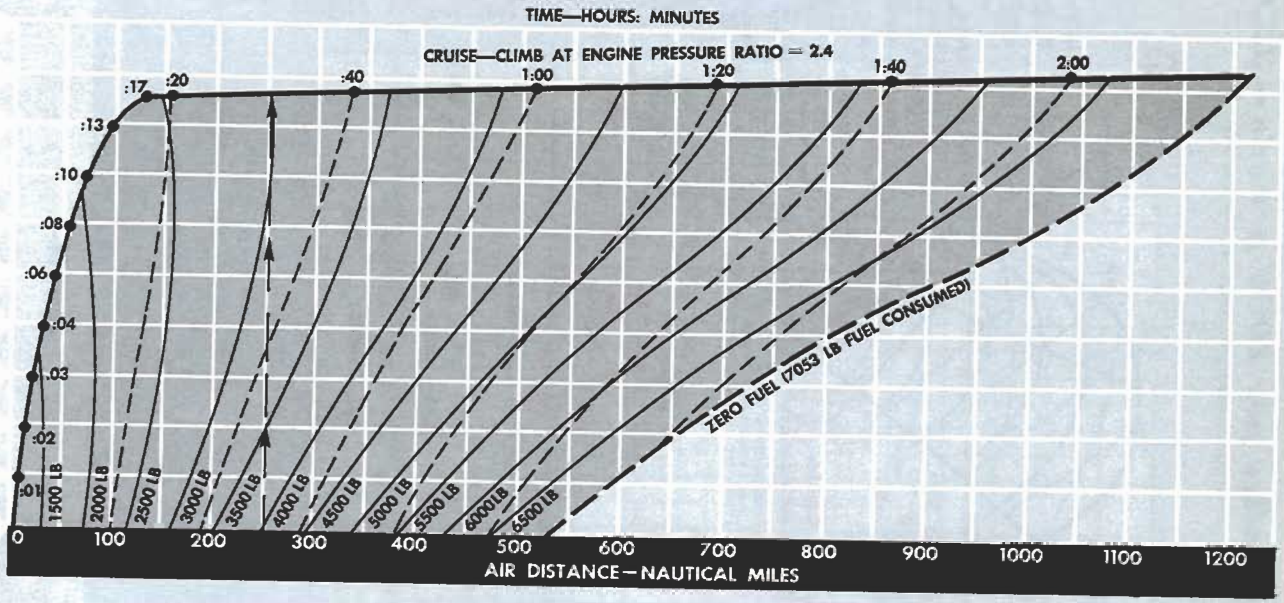
MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
TAKEOFF GROSS WEIGHT: 28,150 LBS.
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



MILITARY THRUST CLIMB					
TAS KNOTS	CAS KNOTS	IAS KNOTS	IND MACH NO.	TRUE MACH NO.	ALT 1000 FEET
516	219	213	.87	.90	50
516	245	238	.87	.90	45
516	276	269	.87	.90	40
519	310	302	.87	.90	35
483	311	303	.79	.82	30
451	314	306	.72	.75	25
419	315	307	.66	.68	20
393	317	309	.61	.63	15
370	321	313	.56	.58	10
351	326	318	.52	.54	5
331	331	322	.48	.50	SL



CRUISE CLEAN CONFIGURATION							
ALTITUDE FEET	TRUE MACH NO.*	IND MACH NO.*	IAS KNOTS*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*	ENGINE PRESSURE RATIO*
CRUISE-CLIMB	.92	.89	251	259	530	2415	2.44
45,000	.91	.87	242	250	520	2440	2.36
40,000	.90	.87	270	278	518	2410	2.18
35,000	.83	.80	276	284	480	2430	1.98
30,000	.74	.72	271	279	437	2480	1.84
25,000	.67	.65	272	280	405	2600	1.72
20,000	.62	.59	274	282	378	2740	1.61
15,000	.56	.54	276	284	352	2845	1.53
10,000	.51	.49	275	283	326	2940	1.45
5,000	.47	.46	277	285	307	3135	1.39
SL	.43	.42	279	287	287	3300	1.34

*APPROXIMATE

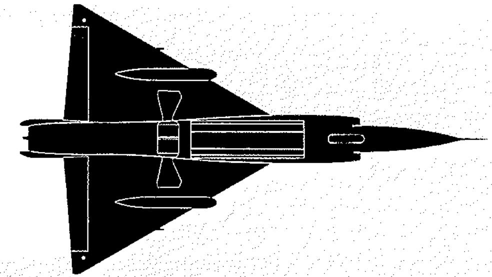
● ● ● ● TIME TO CLIMB
— — — ZERO FUEL REMAINING
——— FUEL CONSUMED
——— CRUISE-CLIMB FLIGHT PATH
– – – – TIME (START, TAXI AND TAKEOFF NOT INCLUDED)

22047

NOTE
1. Fuel allowance for start, taxi, and takeoff (1130 lbs) included
2. No allowance or reserves made for loiter descent or landing
3. Use Military Thrust for climb (See Military Thrust Climb Chart for detail information)
4. Cruise at recommended Mach No.

Figure A4-4

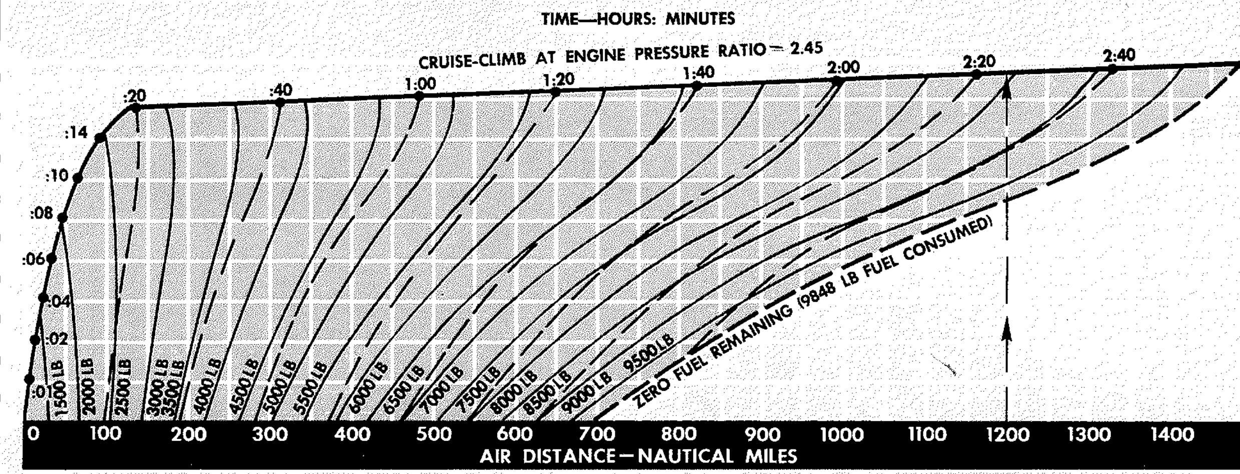
mission profile



MODEL: F-102A | CONFIGURATION: TWO – 230 GALLON EXTERNAL TANKS | ENGINE: J57-23
DATE: 15 OCTOBER 1958 | TAKEOFF GROSS WEIGHT: 31,276 LBS. | FUEL GRADE: JP-4
DATA BASIS: FLIGHT TEST | STANDARD DAY | FUEL DENSITY: 6.5 LB/GAL

MILITARY THRUST CLIMB

TAS KNOTS	CAS KNOTS	IAS KNOTS	IND MACH NO.	TRUE MACH NO.	ALT 1000 FEET
462	216	210	.78	.81	45
462	244	237	.78	.81	40
464	273	266	.78	.81	35
427	273	266	.70	.73	30
400	277	270	.64	.67	25
376	281	274	.59	.61	20
354	285	277	.55	.57	15
337	292	284	.51	.53	10
320	296	288	.48	.49	5
304	304	295	.45	.46	SL



CRUISE—TWO 230 GAL TANKS

ALTITUDE FEET	TRUE MACH NO.*	IND. MACH NO.*	IAS KNOTS*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*	ENGINE PRESSURE RATIO*
CRUISE-CLIMB	.89	.86	263	270	510	2850	2.45
40,000	.89	.86	266	273	510	2840	2.39
35,000	.83	.80	275	283	477	2840	2.20
30,000	.72	.70	265	272	427	2850	1.98
25,000	.65	.63	265	272	392	2860	1.85
20,000	.59	.57	265	272	365	2975	1.71
15,000	.54	.52	263	270	335	3100	1.61
10,000	.49	.47	263	270	312	3220	1.52
5,000	.45	.43	264	271	291	3320	1.44
SL	.41	.40	265	272	272	3440	1.38

*APPROXIMATE

● ● ● TIME TO CLIMB
— — — ZERO FUEL REMAINING
——— FUEL CONSUMED
——— CRUISE-CLIMB FLIGHT PATH
– – – – TIME (START, TAXI AND TAKEOFF NOT INCLUDED)

NOTE
1. Fuel allowance for start, taxi, and takeoff (1130 lbs) included.
2. No allowance or reserves made for loiter, descent or landing.
3. Use Military Thrust for climb (See Military Thrust Climb Chart for detail information).
4. Cruise at recommended Mach No.

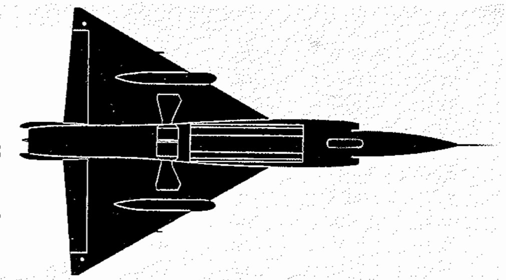
22048

mission profile

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO-230 GALLON EXTERNAL TANKS
(DROPPED WHEN EMPTY)
TAKEOFF GROSS WEIGHT: 31,276 LBS.
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



MILITARY THRUST CLIMB					
TAS KNOTS	CAS KNOTS	IAS KNOTS	IND MACH NO.	TRUE MACH NO.	ALT 1000 FEET
516	219	213	.87	.90	50
516	245	238	.87	.90	45
516	276	269	.87	.90	40
464	273	266	.78	.81	35
427	273	266	.70	.73	30
400	277	270	.64	.67	25
376	281	274	.59	.61	20
354	285	277	.55	.57	15
337	292	284	.51	.53	10
320	296	288	.48	.49	5
304	304	295	.45	.46	SL

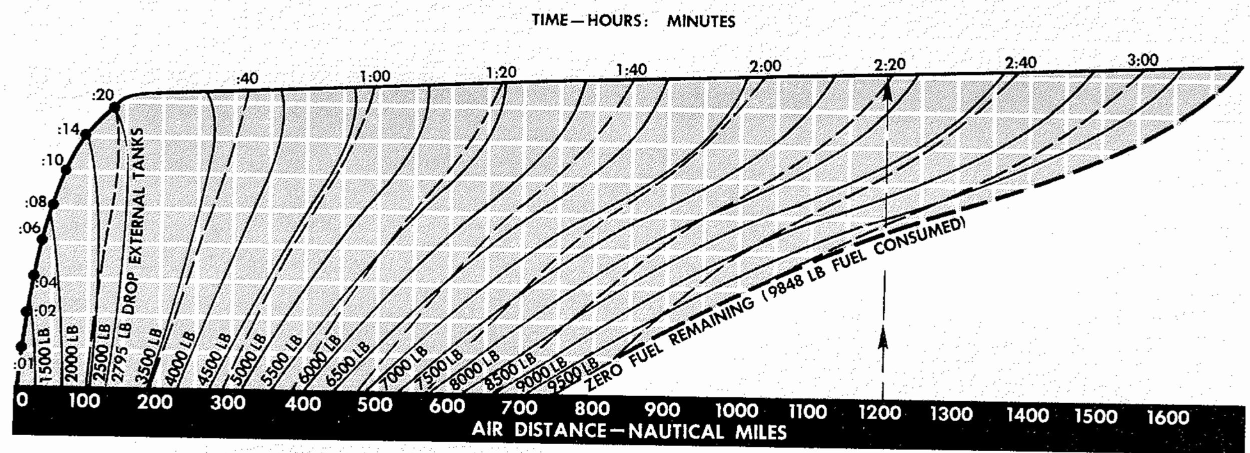


Figure A4-6

CRUISE-CLEAN CONFIGURATION							
ALTITUDE FEET	TRUE MACH NO.*	IND MACH NO.*	IAS KNOTS*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*	ENGINE PRESSURE RATIO*
CRUISE-CLIMB	.92	.89	251	259	530	2415	2.44
45,000	.91	.87	242	250	520	2440	2.36
40,000	.90	.87	270	278	518	2410	2.18
35,000	.83	.80	276	284	480	2430	1.98
30,000	.74	.72	271	279	437	2480	1.84
25,000	.67	.65	272	280	405	2600	1.72
20,000	.62	.59	274	282	378	2740	1.61
15,000	.56	.54	276	284	352	2845	1.53
10,000	.51	.49	275	283	326	2940	1.45
5,000	.47	.46	277	285	307	3135	1.39
SL	.43	.42	279	287	287	3300	1.34

CRUISE—TWO 230 GAL TANKS							
ALTITUDE FEET	TRUE MACH NO.*	IND MACH NO.*	IAS KNOTS*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*	ENGINE PRESSURE RATIO*
CRUISE-CLIMB	.89	.86	263	270	510	2850	2.45
40,000	.89	.86	266	273	510	2840	2.39
35,000	.83	.80	275	283	477	2840	2.20
30,000	.72	.70	265	272	427	2850	1.98
25,000	.65	.63	265	272	392	2860	1.85
20,000	.59	.57	265	272	363	2975	1.71
15,000	.54	.52	263	270	335	3100	1.61
10,000	.49	.47	263	270	312	3220	1.52
5,000	.45	.43	264	271	291	3320	1.44
SL	.41	.40	265	272	272	3440	1.38

*APPROXIMATE

22049

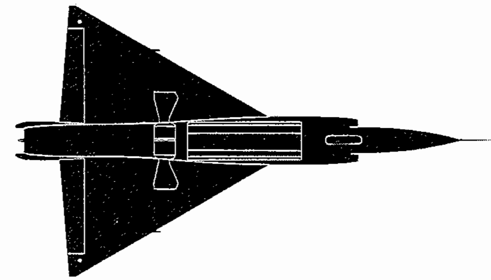
● ● ● TIME TO CLIMB
— — — ZERO FUEL REMAINING
——— FUEL CONSUMED
——— CRUISE-CLIMB FLIGHT PATH
- - - - TIME (START, TAXI AND TAKEOFF NOT INCLUDED)

NOTE

1. Fuel allowance for start, taxi and takeoff 1130 lbs included.
2. No allowance or reserves made for loiter, descent or landing.
3. Use Military Thrust for climb (See Military Thrust Climb Chart for detail information).
4. Cruise at recommended Mach No.

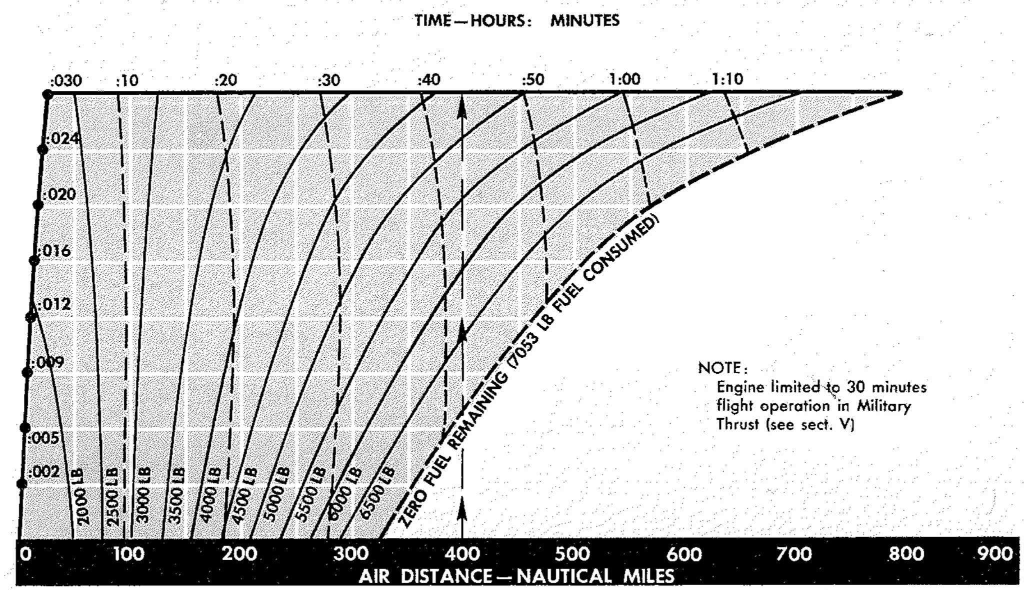
Figure A4-7

intercept profile



MODEL: F-102A | CONFIGURATION: CLEAN | ENGINE: J57-23
DATE: 15 OCTOBER 1958 | TAKEOFF GROSS WEIGHT: 28,150 LBS. | FUEL GRADE: JP-4
DATA BASIS: FLIGHT TEST | STANDARD DAY | FUEL DENSITY: 6.5 LB/GAL

MAXIMUM THRUST CLIMB					
TAS KNOTS	CAS KNOTS	TRUE MACH NO.	IAS KNOTS	IND. MACH NO.	ALT 1000 FEET
533	256	.93	249	.89	45
533	287	.93	279	.89	40
536	322	.93	313	.89	35
547	356	.93	347	.89	30
558	396	.93	386	.89	25
566	435	.92	424	.89	20
576	476	.92	464	.88	15
585	516	.92	503	.88	10
593	558	.91	545	.88	5
602	602	.91	588	.88	SL



NOTE:
Engine limited to 30 minutes flight operation in Military Thrust (see sect. V)

NOTE:
1. Fuel allowance for start, taxi and take-off 1130 lbs included.
2. No allowance or reserves made for loiter, descent or landing.
3. Use Military plus Afterburner Thrust for climb.
4. Cruise at Military Thrust.

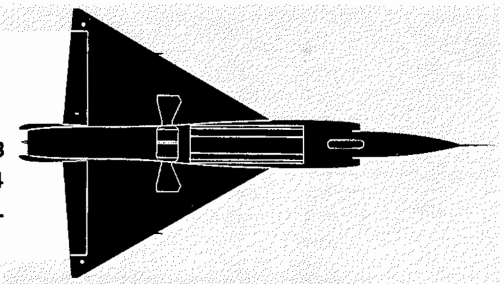
CRUISE CLEAN							
ALTITUDE FEET	TRUE MACH NO.*	IND MACH NO.*	IAS KNOTS*	CAS KNOTS*	TAS KNOTS*	ENGINE PRESSURE RATIO*	FUEL FLOW LB/HR*
40,000	.96	.92	288	297	547	2.52	3340
35,000	.97	.93	324	334	559	2.52	4430
30,000	.96	.92	360	371	567	2.35	5120
25,000	.96	.92	399	411	576	2.24	5880
20,000	.95	.91	438	450	584	2.09	6670
15,000	.94	.90	474	486	588	1.98	7440
10,000	.91	.88	502	514	582	1.87	8200
5,000	.88	.86	525	538	571	1.80	9000
SL	.84	.81	542	555	555	1.76	10,270

*APPROXIMATE

● ● ● TIME TO CLIMB
▬ ▬ ▬ ZERO FUEL REMAINING
——— FUEL CONSUMED
— — — TIME (START, TAXI, AND TAKEOFF NOT INCLUDED)

22050

optimum return profile



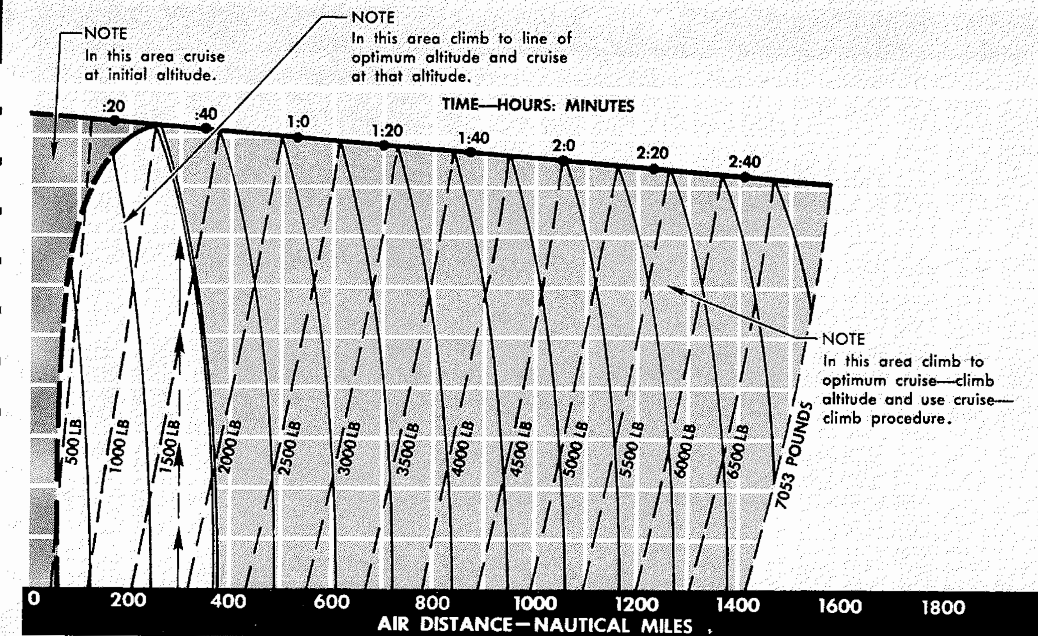
MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
TAKEOFF GROSS WEIGHT: 28,150 LBS.
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

MILITARY THRUST CLIMB

TAS KNOTS	CAS KNOTS	IAS KNOTS	IND MACH NO.	TRUE MACH NO.	ALT 1000 FEET
516	219	213	.87	.90	50
516	245	238	.87	.90	45
516	276	269	.87	.90	40
519	310	302	.87	.90	35
483	311	303	.79	.82	30
451	314	306	.72	.75	25
419	315	307	.66	.68	20
393	317	309	.61	.63	15
370	321	313	.56	.58	10
351	326	318	.52	.54	5
331	331	322	.48	.50	SL



CRUISE-CLEAN CONFIGURATION

ALTITUDE FEET	TRUE MACH NO.*	IND MACH NO.*	IAS KNOTS*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*	ENGINE PRESSURE RATIO*
CRUISE-CLIMB	.92	.89	251	259	530	2415	2.44
45,000	.91	.87	242	250	520	2440	2.36
40,000	.90	.87	270	278	518	2410	2.18
35,000	.83	.80	276	284	480	2430	1.98
30,000	.74	.72	271	279	437	2480	1.84
25,000	.67	.65	272	280	405	2600	1.72
20,000	.62	.59	274	282	378	2740	1.61
15,000	.56	.54	276	284	352	2845	1.53
10,000	.51	.49	275	283	326	2940	1.45
5,000	.47	.46	277	285	307	3135	1.39
SL	.43	.42	279	287	287	3300	1.34

*APPROXIMATE

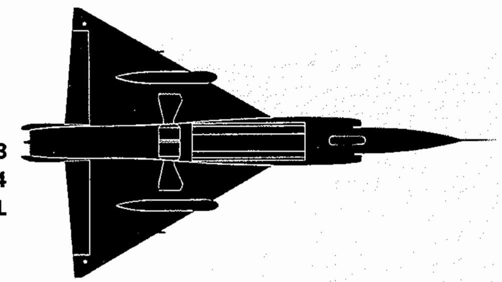
22051

- – – – FUEL REQUIRED
- ● ● ● TIME AT CRUISING ALTITUDE
- — — — LINE OF BEST RANGE FOR CONSTANT ALTITUDE
- ——— CLIMB PATH GUIDE LINES
- ━━━ LINE OF BEST RANGE FOR CRUISE-CLIMB ALTITUDE

NOTE:
(1) Fuel required at any point includes Military Thrust climb to cruise—climb altitude (if below that).
(2) No allowance made for loiter, descent or landing.
(3) Best cruise condition is determined by intersection of climb path guide line and lines of best range.
(4) Cruise at recommended Mach No.

Figure A4-8

optimum return profile

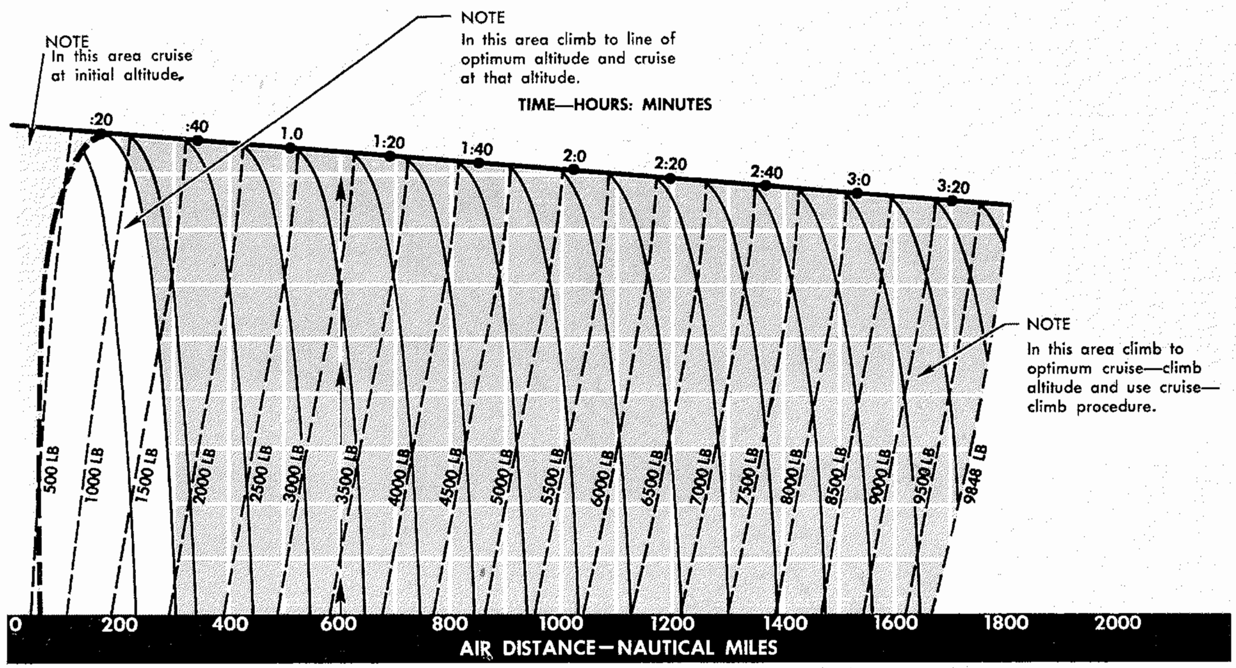


MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO-230 GALLON EXTERNAL TANKS
STANDARD DAY
GROSS WEIGHT: 31,276 LBS.

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

MILITARY THRUST CLIMB					
TAS KNOTS	CAS KNOTS	IAS KNOTS	IND MACH NO.	TRUE MACH NO.	ALT 1000 FEET
462	216	210	.78	.81	45
462	244	237	.78	.81	40
464	273	266	.78	.81	35
427	273	266	.70	.73	30
400	277	270	.64	.67	25
376	281	274	.59	.61	20
354	285	277	.55	.57	15
337	292	284	.51	.53	10
320	296	288	.48	.49	5
304	304	295	.45	.46	SL



CRUISE—TWO 230 GAL TANKS							
ALTITUDE FEET	TRUE MACH NO.*	IND MACH NO.*	IAS KNOTS*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*	ENGINE PRESSURE RATIO*
CRUISE-CLIMB	.89	.86	263	270	510	2850	2.45
40,000	.89	.86	266	273	510	2840	2.39
35,000	.83	.80	275	283	477	2840	2.20
30,000	.72	.70	265	272	427	2850	1.98
25,000	.65	.63	265	272	392	2860	1.85
20,000	.59	.57	265	272	363	2975	1.71
15,000	.54	.52	263	270	335	3100	1.61
10,000	.49	.47	263	270	312	3220	1.52
5,000	.45	.43	264	271	291	3320	1.44
SL	.41	.40	265	272	272	3440	1.38

*APPROXIMATE

22052

● ● ● TIME AT CRUISING ALTITUDE
— — — LINE OF BEST RANGE FOR CONSTANT ALTITUDE
———— CLIMB PATH GUIDE LINES
———— LINE OF BEST RANGE FOR CRUISE-CLIMB ALTITUDE
– – – – FUEL REQUIRED

NOTE
(1) Fuel required at any point includes Military Thrust climb to cruise—climb altitude (if below that).
(2) No allowance made for loiter, descent or landing.
(3) Best cruise condition is determined by intersection of climb path guide line and lines of best range.
(4) Cruise at recommended Mach No.

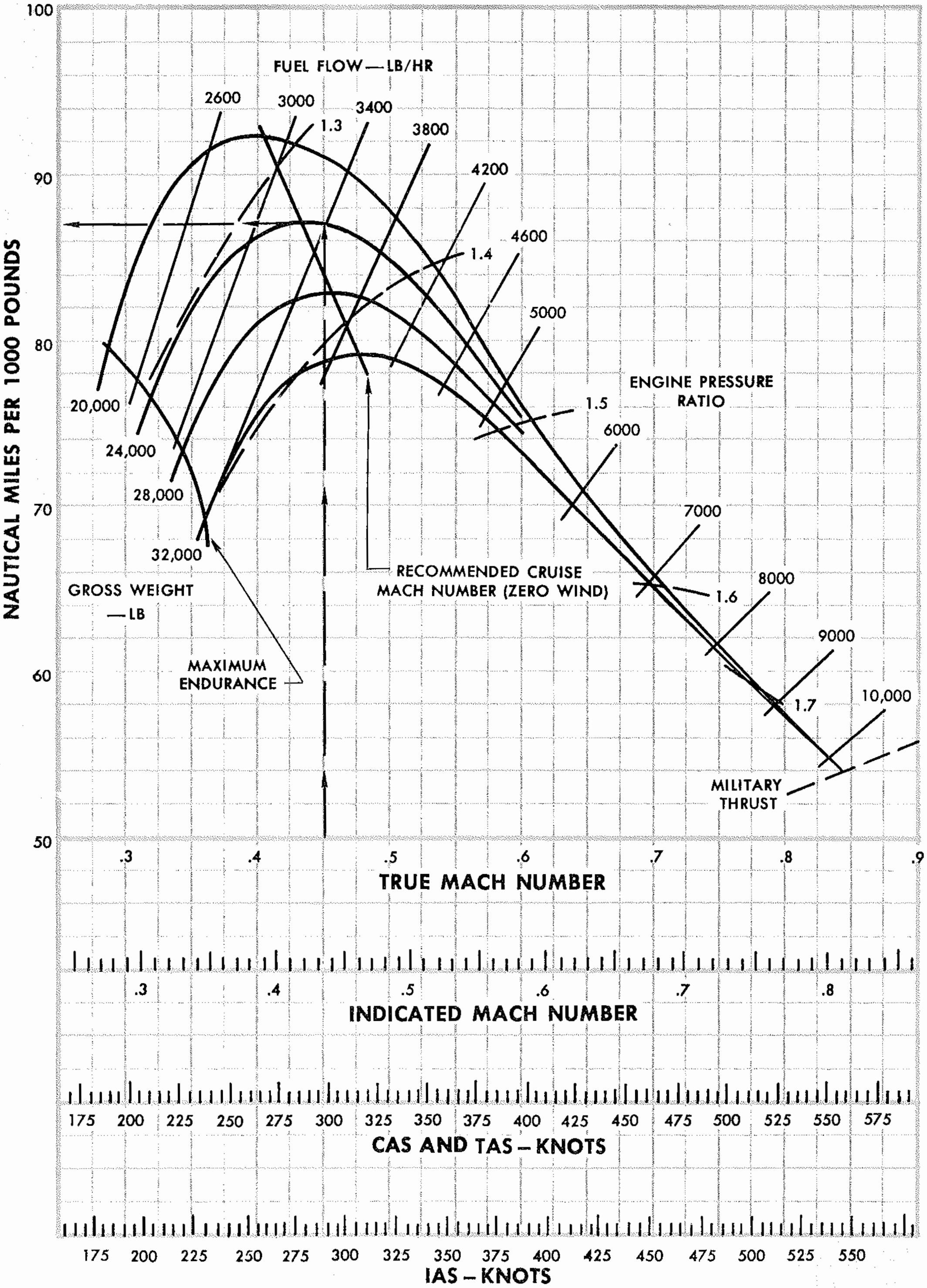
Figure A4-9

nautical miles per 1000 pounds—sea level

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .02 AND DECREASE RANGE 16%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22029C

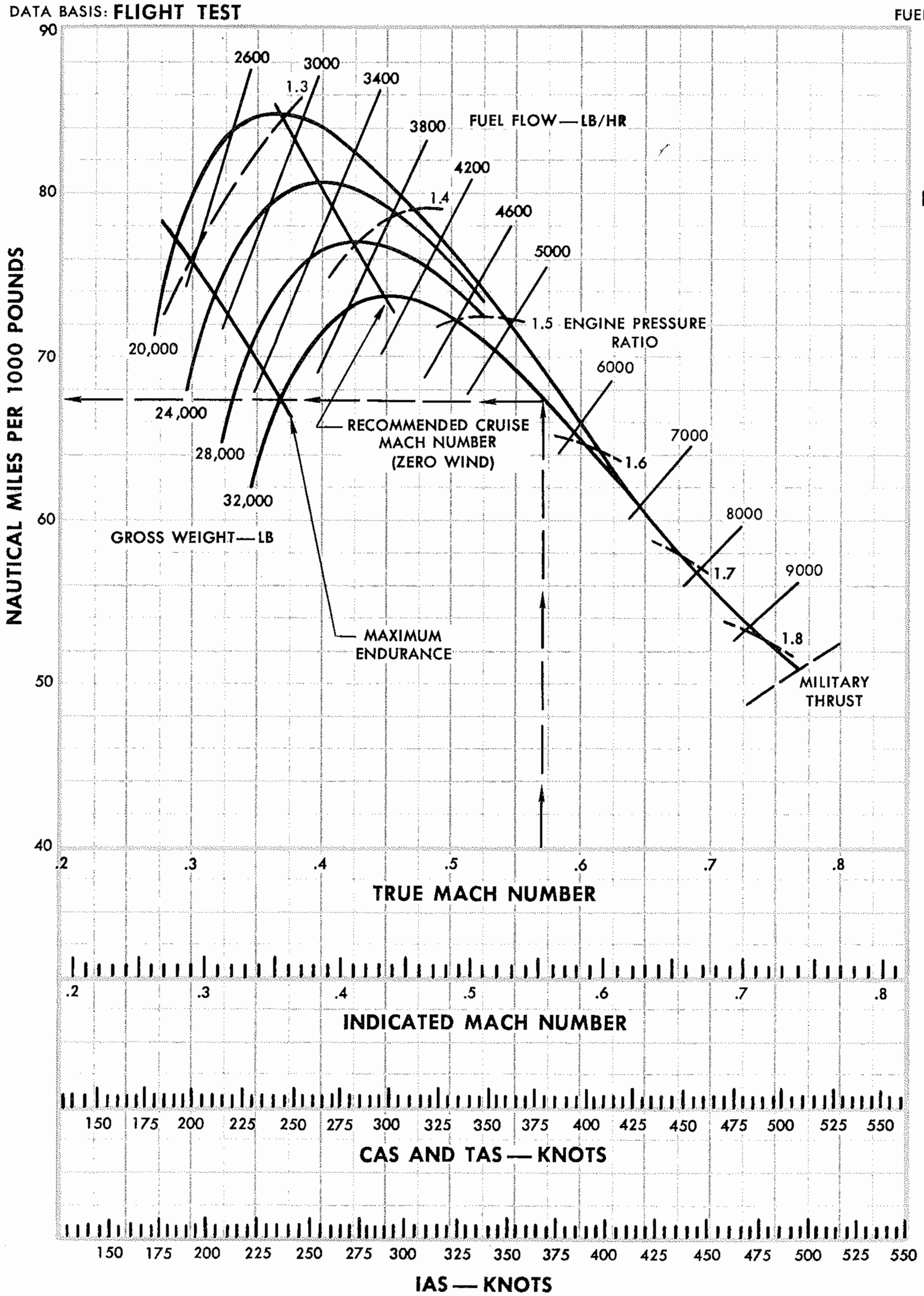
Figure A4-10

nautical miles per 1000 pounds - sea level

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD DAY

MODEL F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

ENGINE J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 18%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22038C

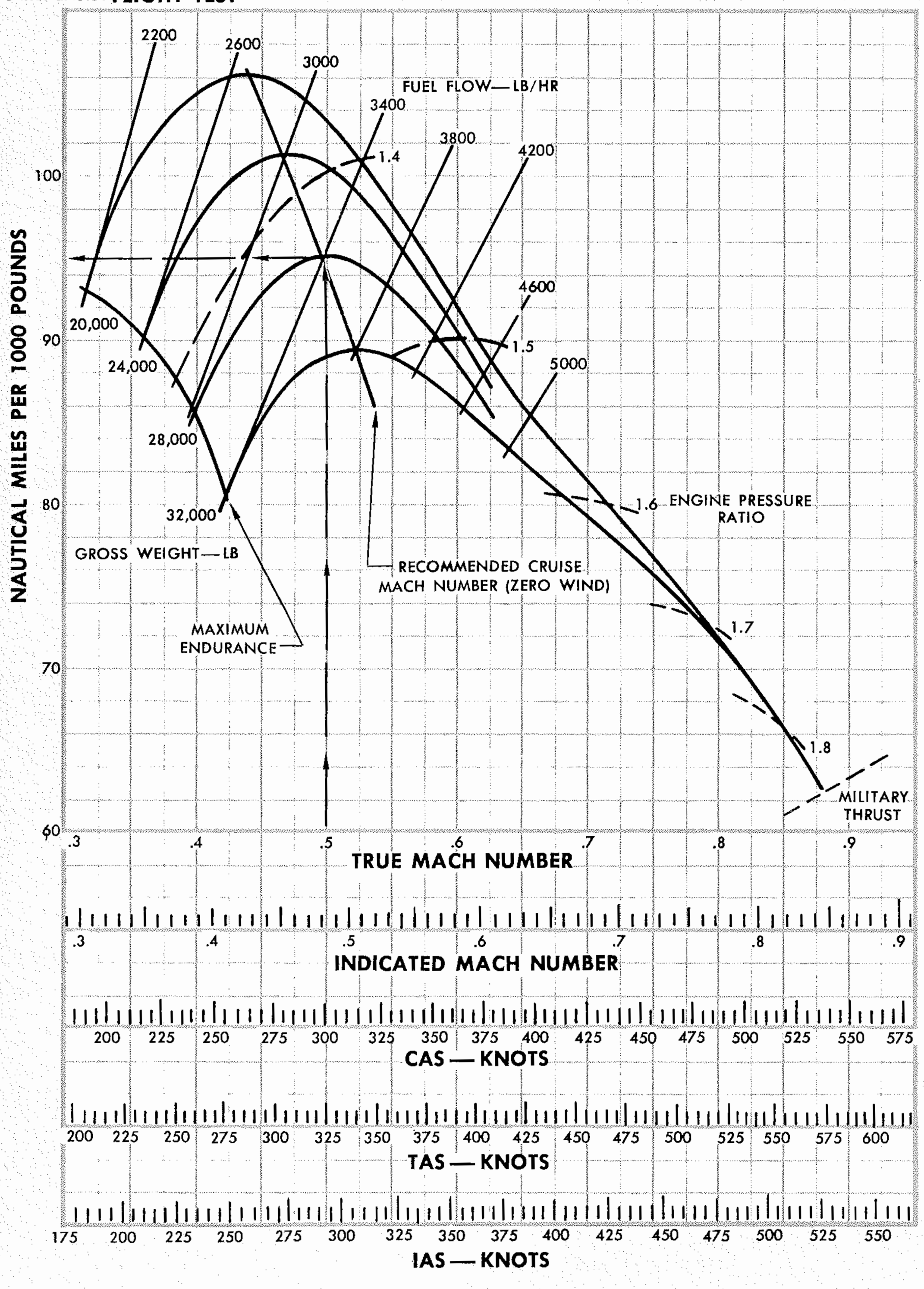
Figure A4-11

nautical miles per 1000 pounds — 5,000 feet

CONFIGURATION: CLEAN
STANDARD DAY

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL.



NOTE:

- WIND CORRECTION TO N. MI/LB
  GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$
  WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. AND DECREASE RANGE 15% FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10° ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22030C

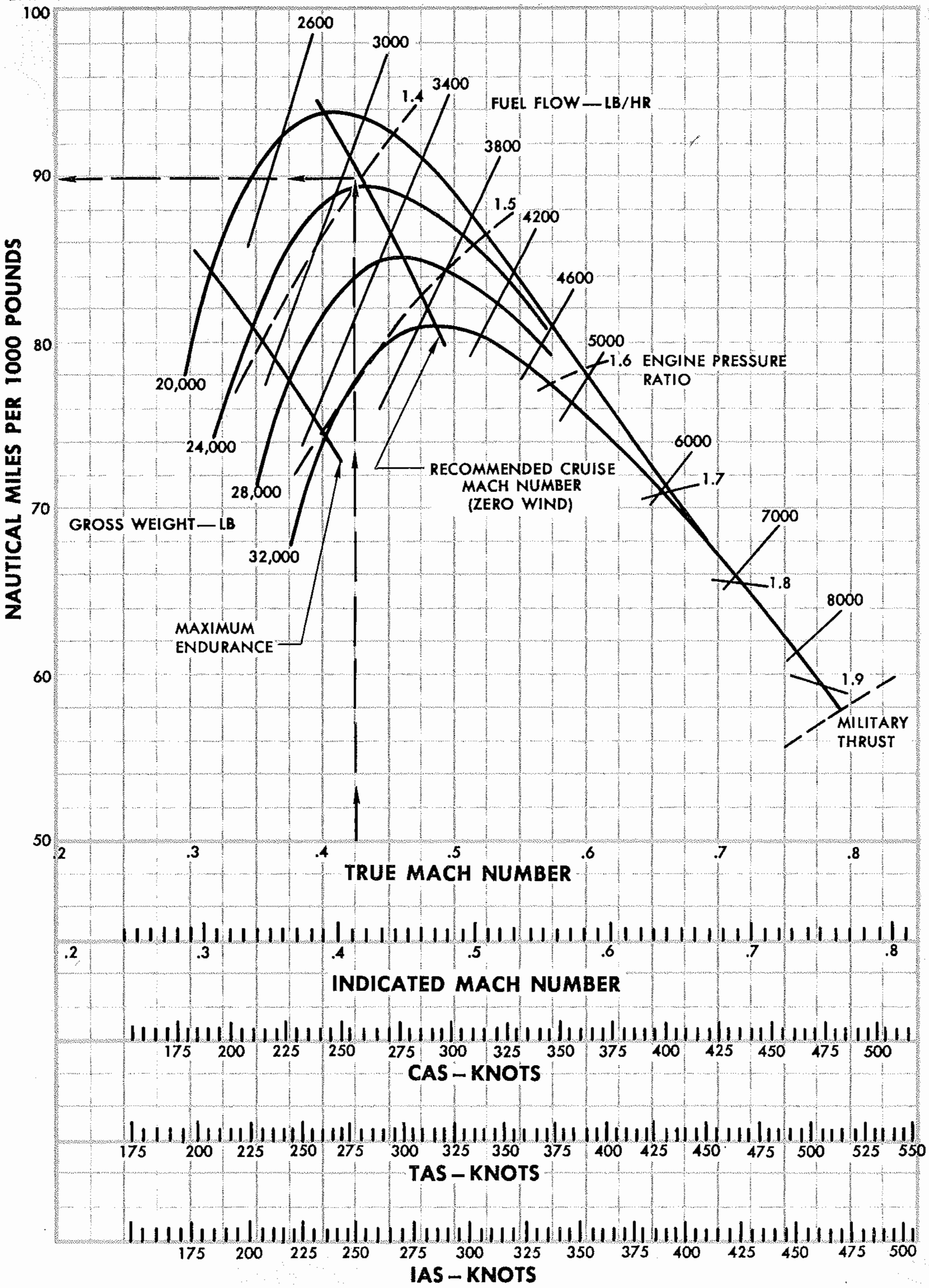
Figure A4-12

nautical miles per 1000 pounds—5000 feet

CONFIGURATION: TWO 230 GALLON EXTERNAL TANKS
STANDARD DAY

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 17%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22039C

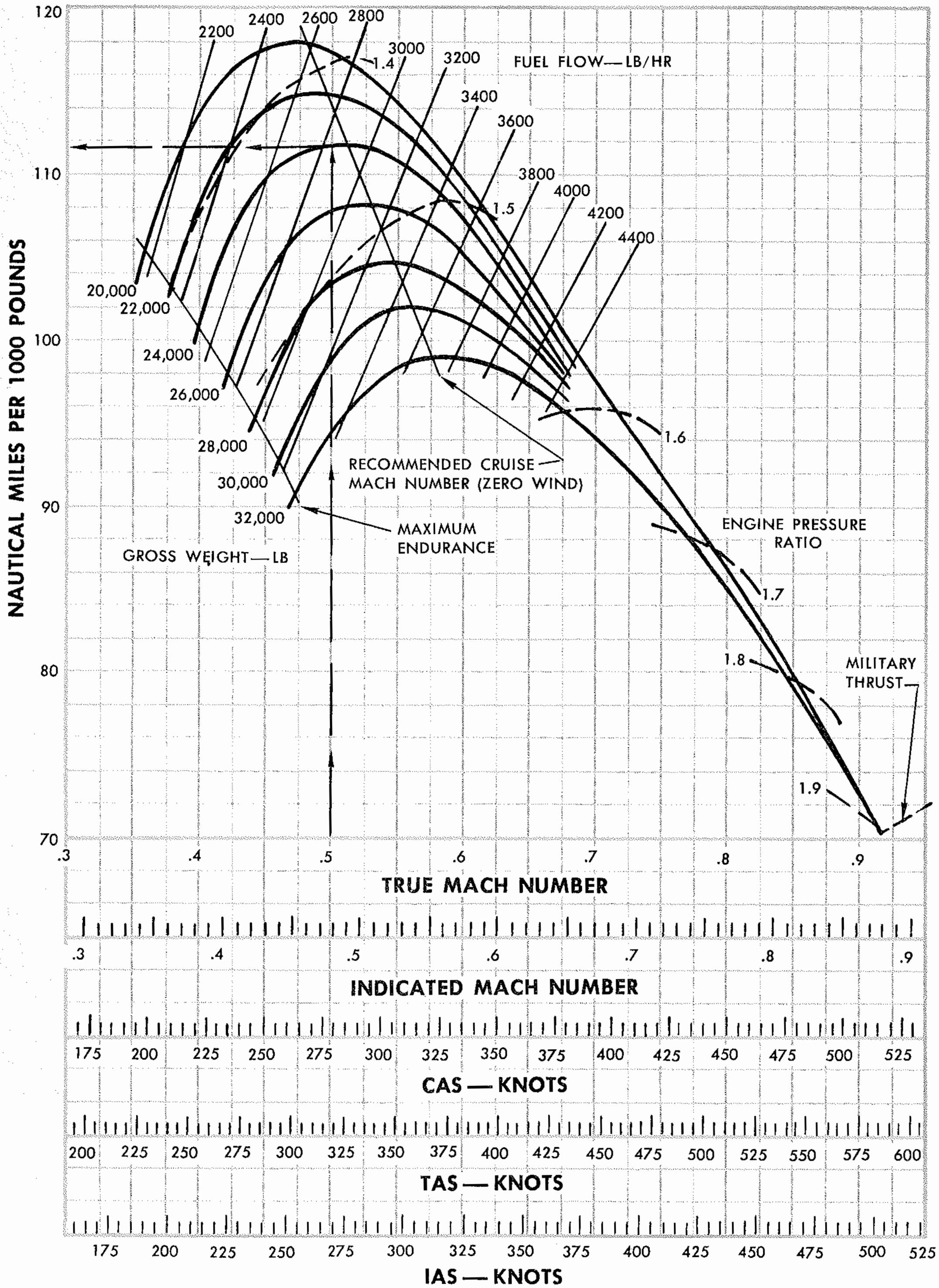
Figure A4-13

nautical miles per 1000 pounds — 10,000 feet

CONFIGURATION: CLEAN
STANDARD DAY

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL.



NOTE:

- WIND CORRECTION TO N. MI/LB

  GROUND N. MI/LB = AIR N. MI/LB $\times \dfrac{V_{GROUND}}{V_{AIR}}$

  WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.

- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .02 AND DECREASE RANGE 15% FOR TAILWIND APPLY OPPOSITE CORRECTION.

- FOR EACH 10° ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.

- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22031D

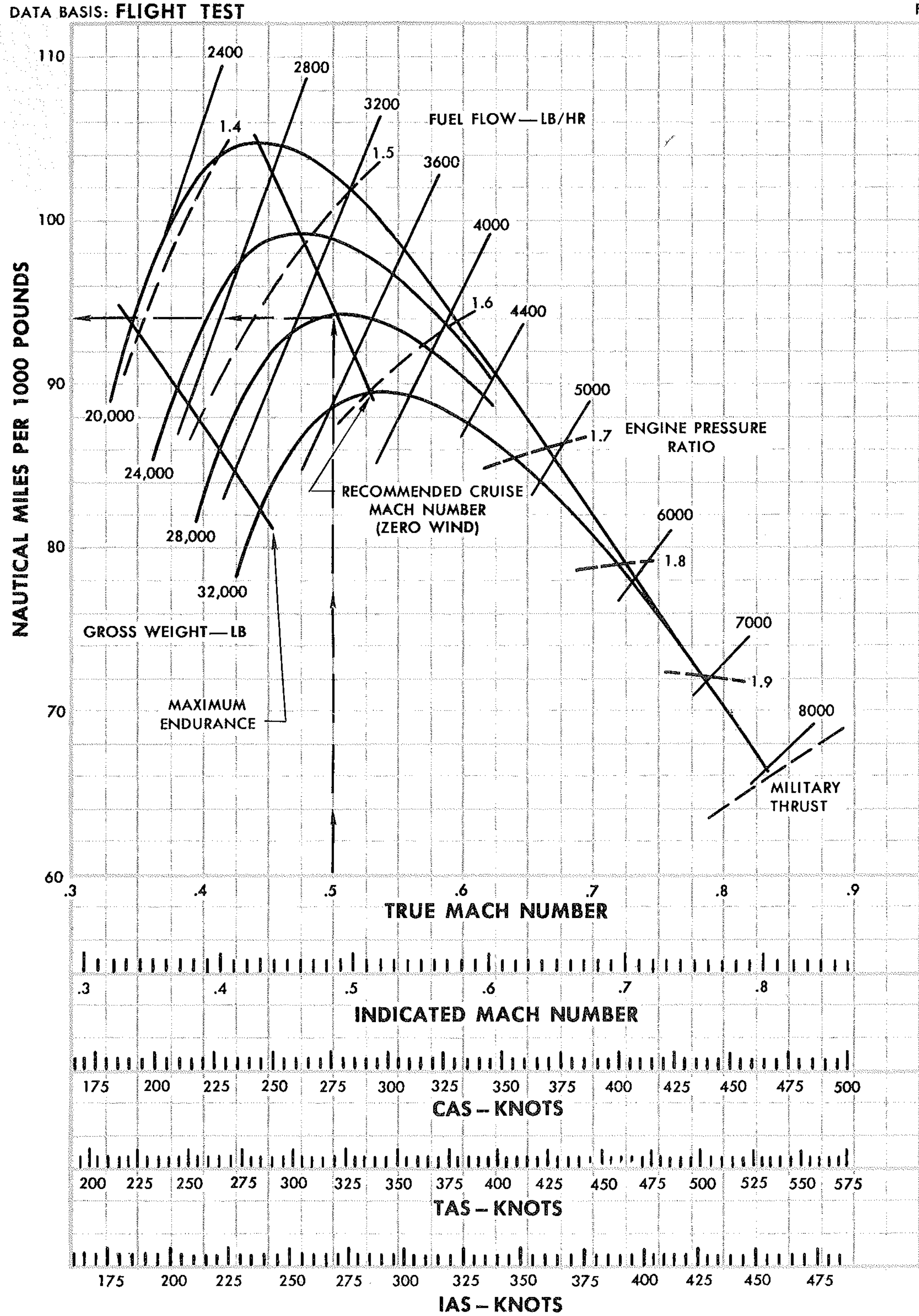
Figure A4-14

nautical miles per 1000 pounds—10,000 feet

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD DAY

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 16%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22040C

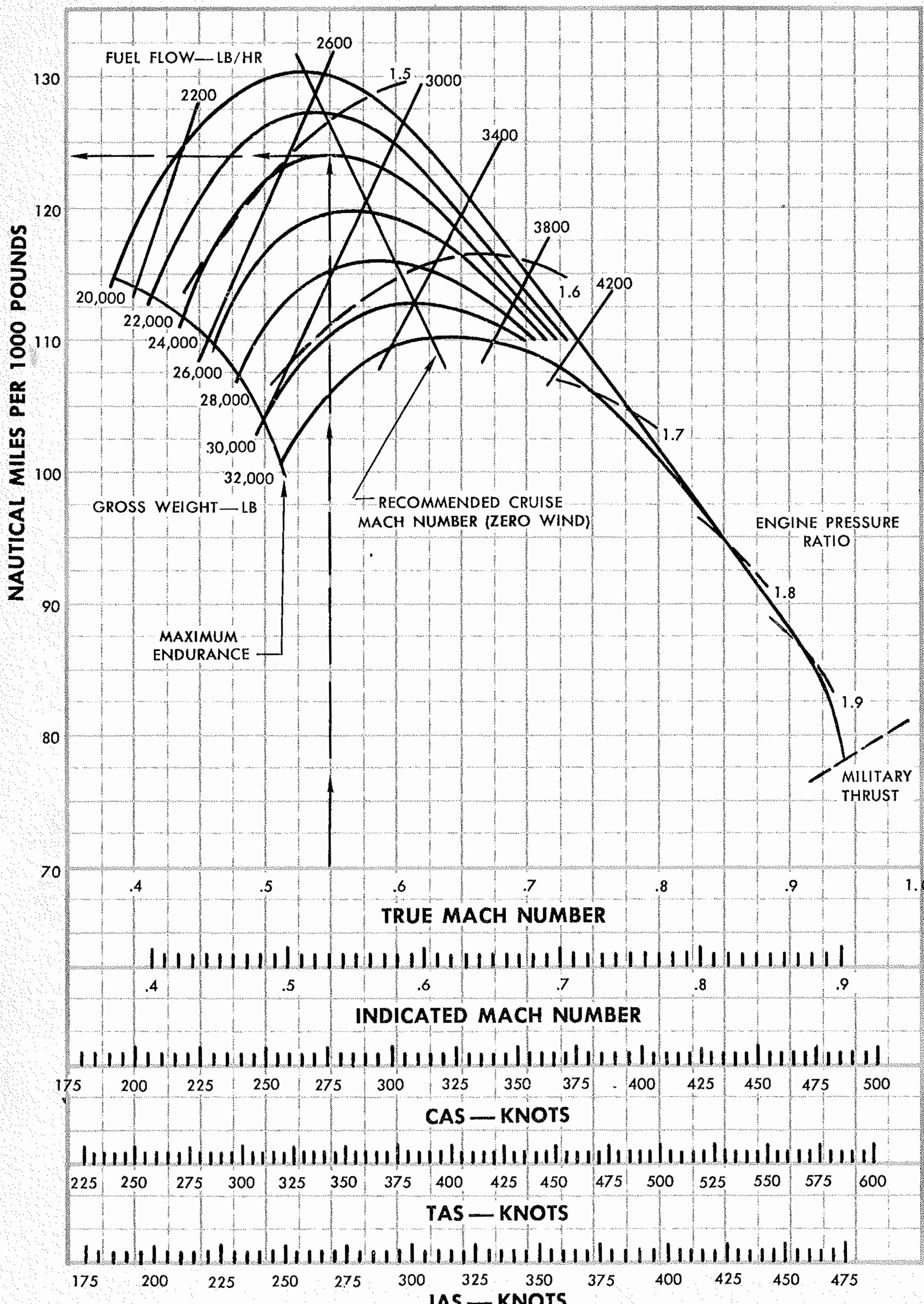
Figure A4-15

nautical miles per 1000 pounds—15,000 feet

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

CONFIGURATION: CLEAN ● STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE:

- WIND CORRECTION TO N. MI/LB

  GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$

  WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.

- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .015 AND DECREASE RANGE 14% FOR TAILWIND APPLY OPPOSITE CORRECTION.

- FOR EACH 10° ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.

- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

DATA BASIS: **FLIGHT TEST**
DATE: 15 OCTOBER 1958

22032D

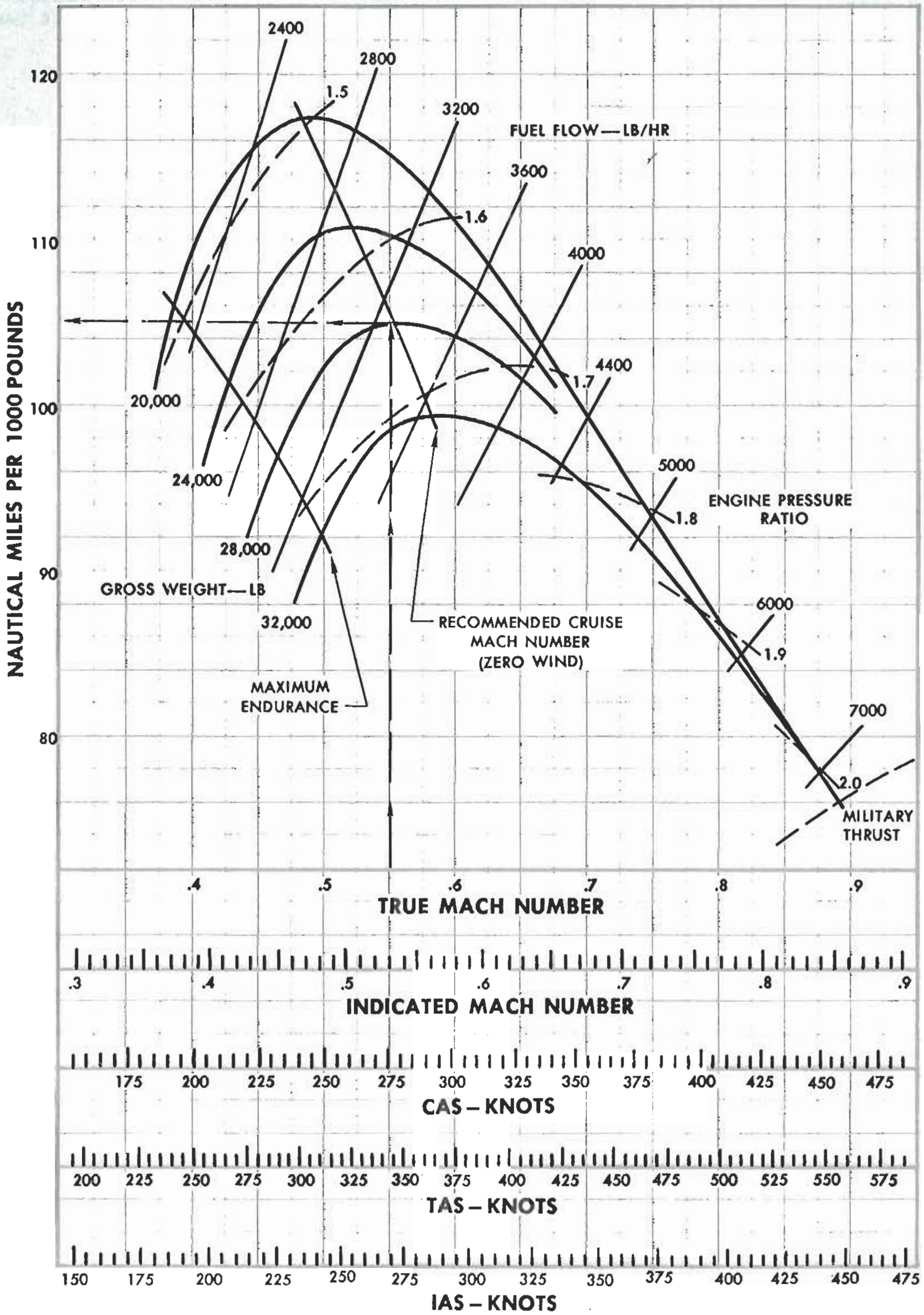
Figure A4-16

nautical miles per 1000 pounds—15,000 feet

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 15%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22041C

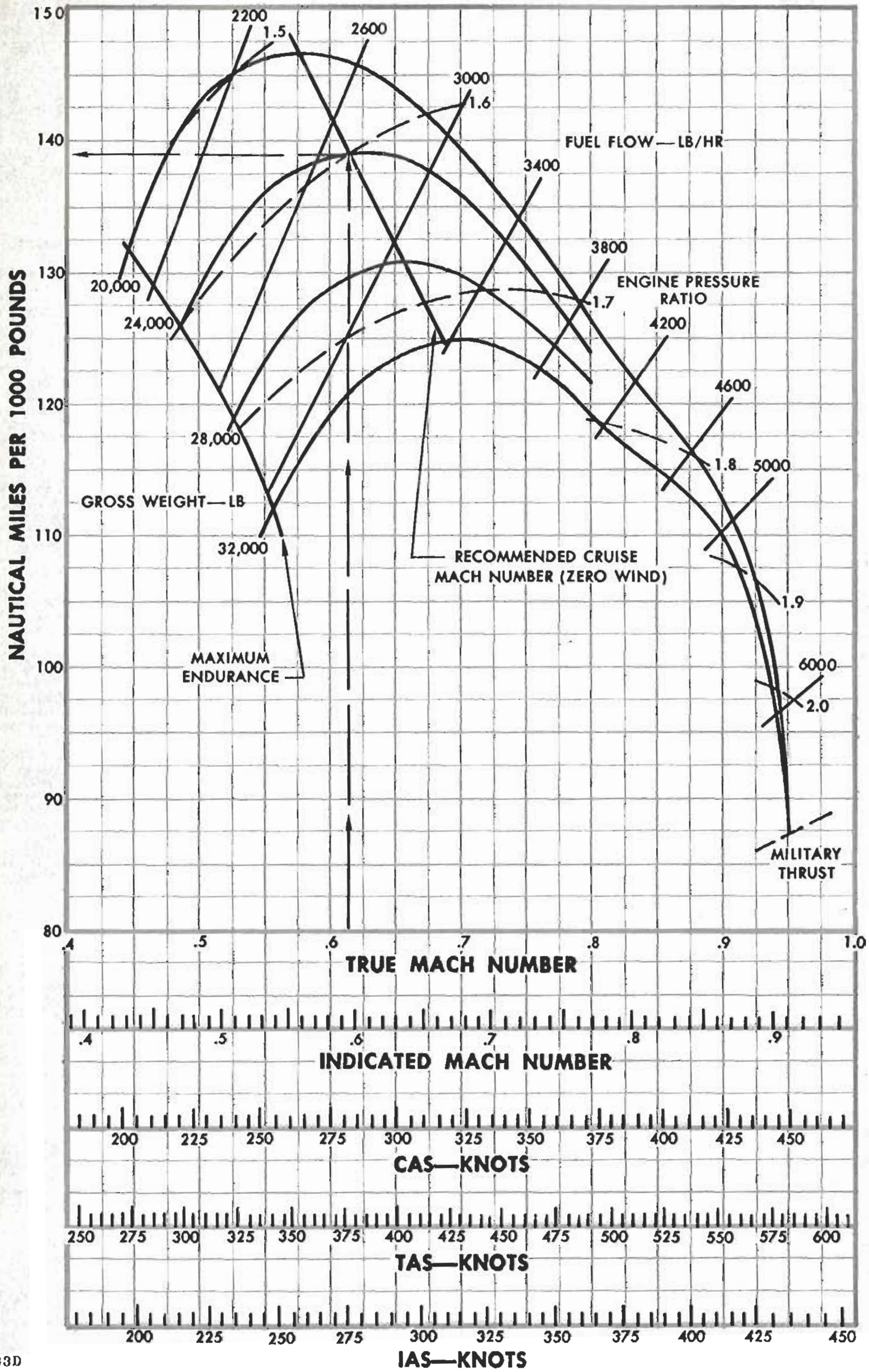
Figure A4-17

nautical miles per 1000 pounds—20,000 feet

CONFIGURATION: CLEAN • STANDARD DAY

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB /GAL



22033D

NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 11%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 40% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

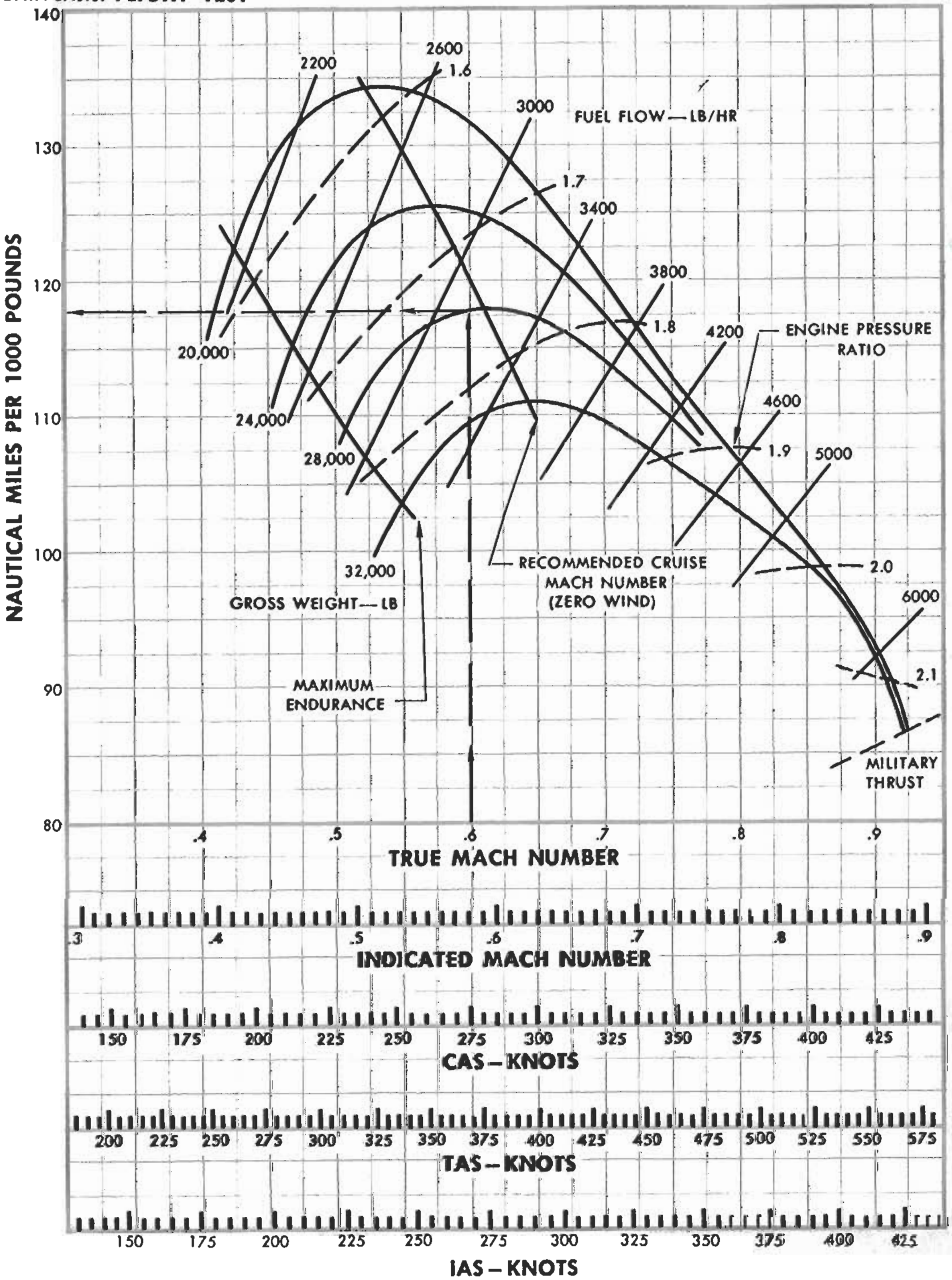
Figure A4-18

nautical miles per 1000 pounds—20,000 feet

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS • STANDARD DAY

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \dfrac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 14%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 37% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22042D

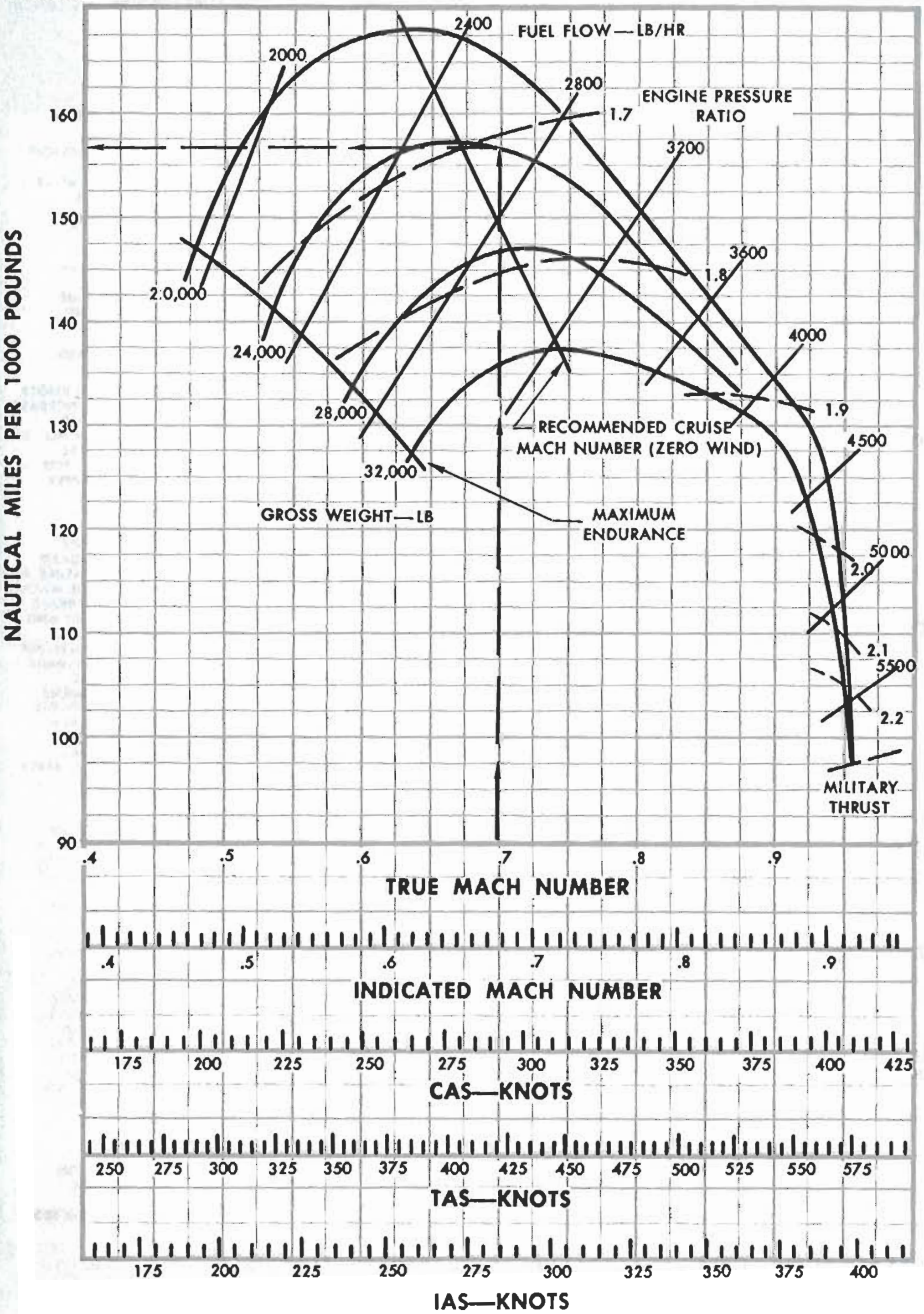
Figure A4-19

nautical miles per 1000 pounds—25,000 feet

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN • STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 12%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 40% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22034D

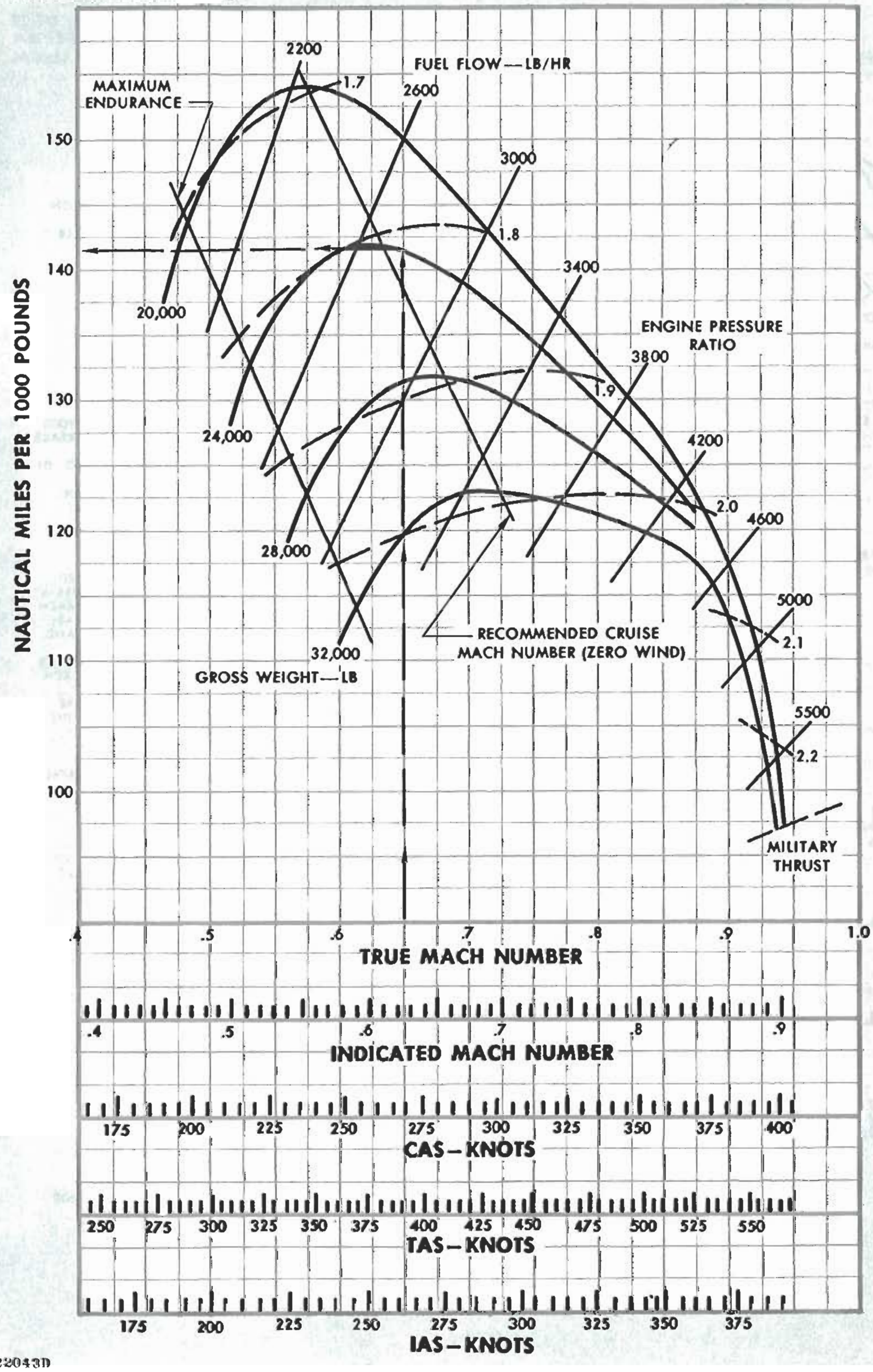
Figure A4-20

nautical miles per 1000 pounds—25,000 feet

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230 GALLON EXTERNAL TANKS
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 13%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 37% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22043D

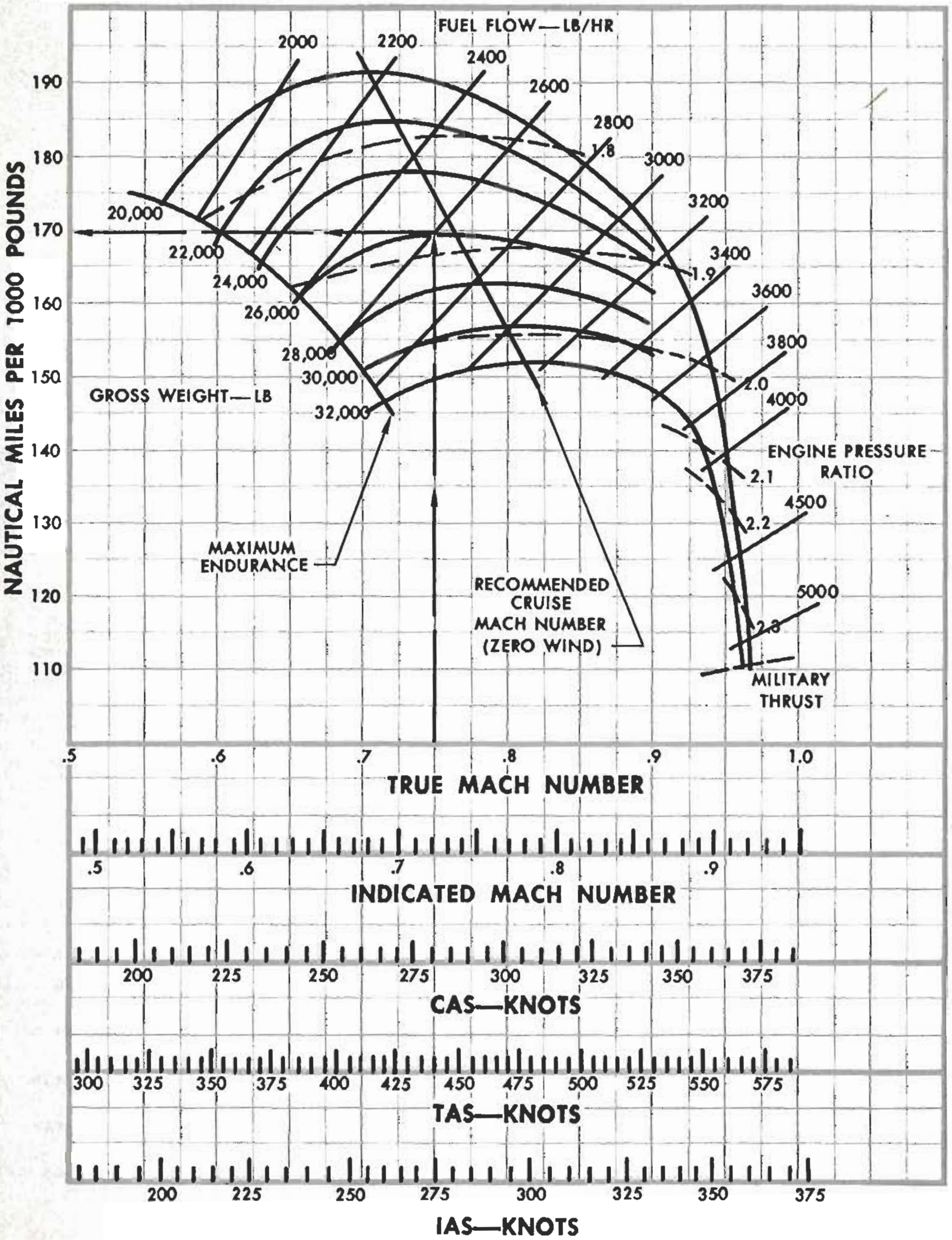
Figure A4-21

nautical miles per 1000 pounds—30,000 feet

CONFIGURATION: CLEAN • STANDARD DAY

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB X $\frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 12%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 40% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

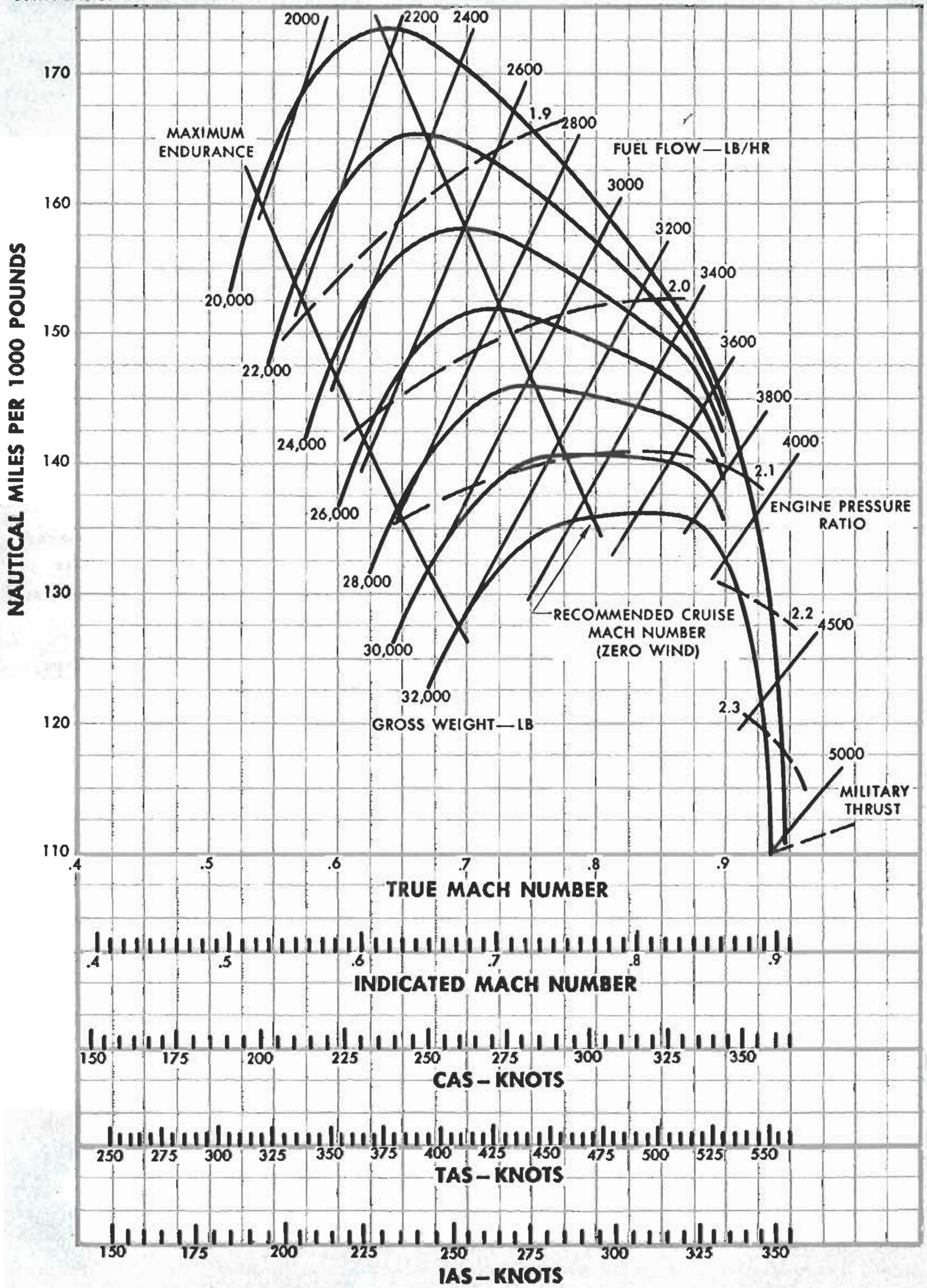
Figure A4-22

nautical miles per 1000 pounds—30,000 feet

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD DAY

MODEL: F-102A
DATA: 1 JULY 1958
DATA BASIS: FLIGHT TEST

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



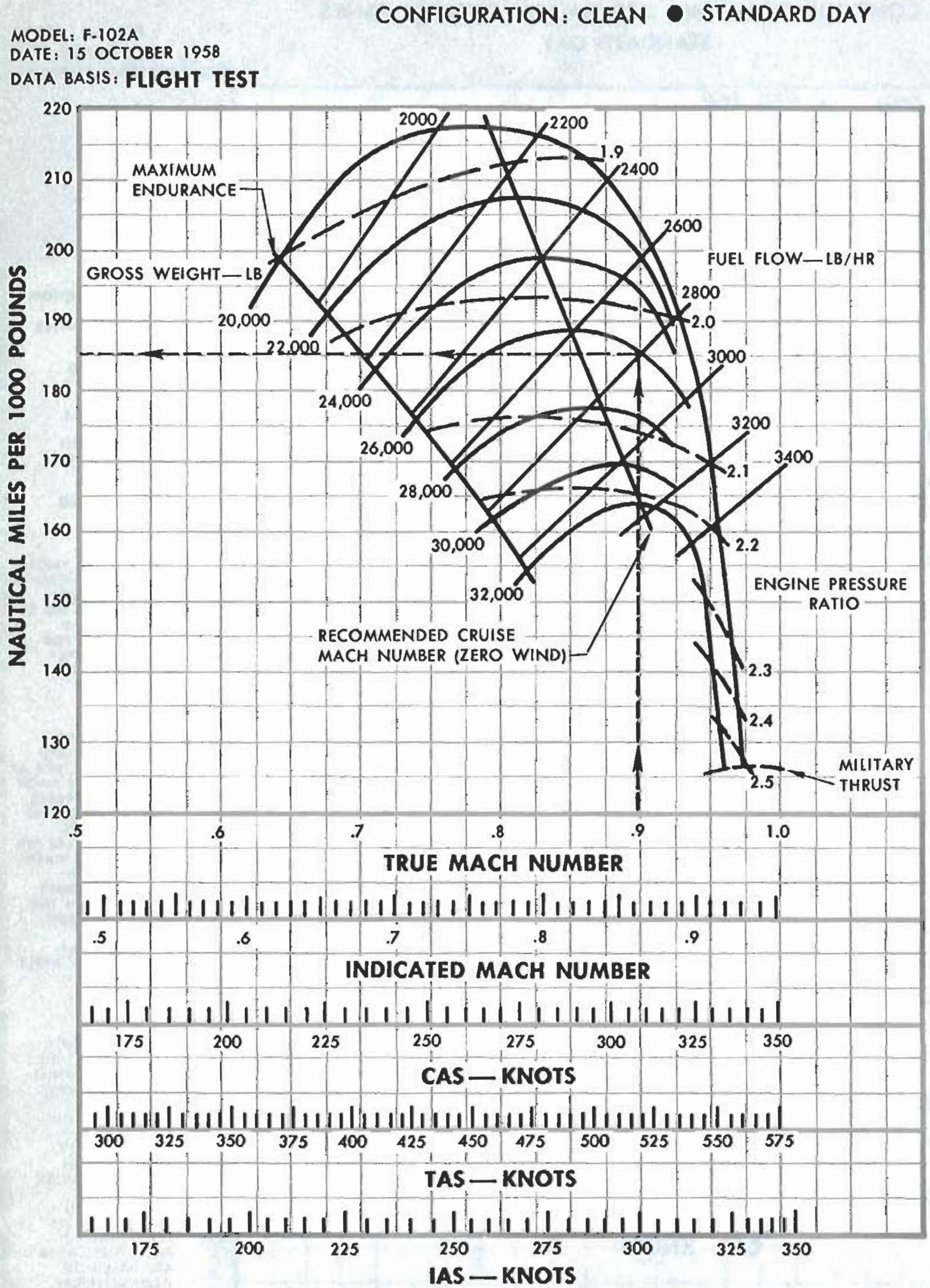
NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. 01 AND DECREASE RANGE 12%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 37% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22044D

Figure A4-23

nautical miles per 1000 pounds — 35,000 feet



NOTE:

- WIND CORRECTION TO N. MI/LB

  GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$

  WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 10% FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10° ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 40% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

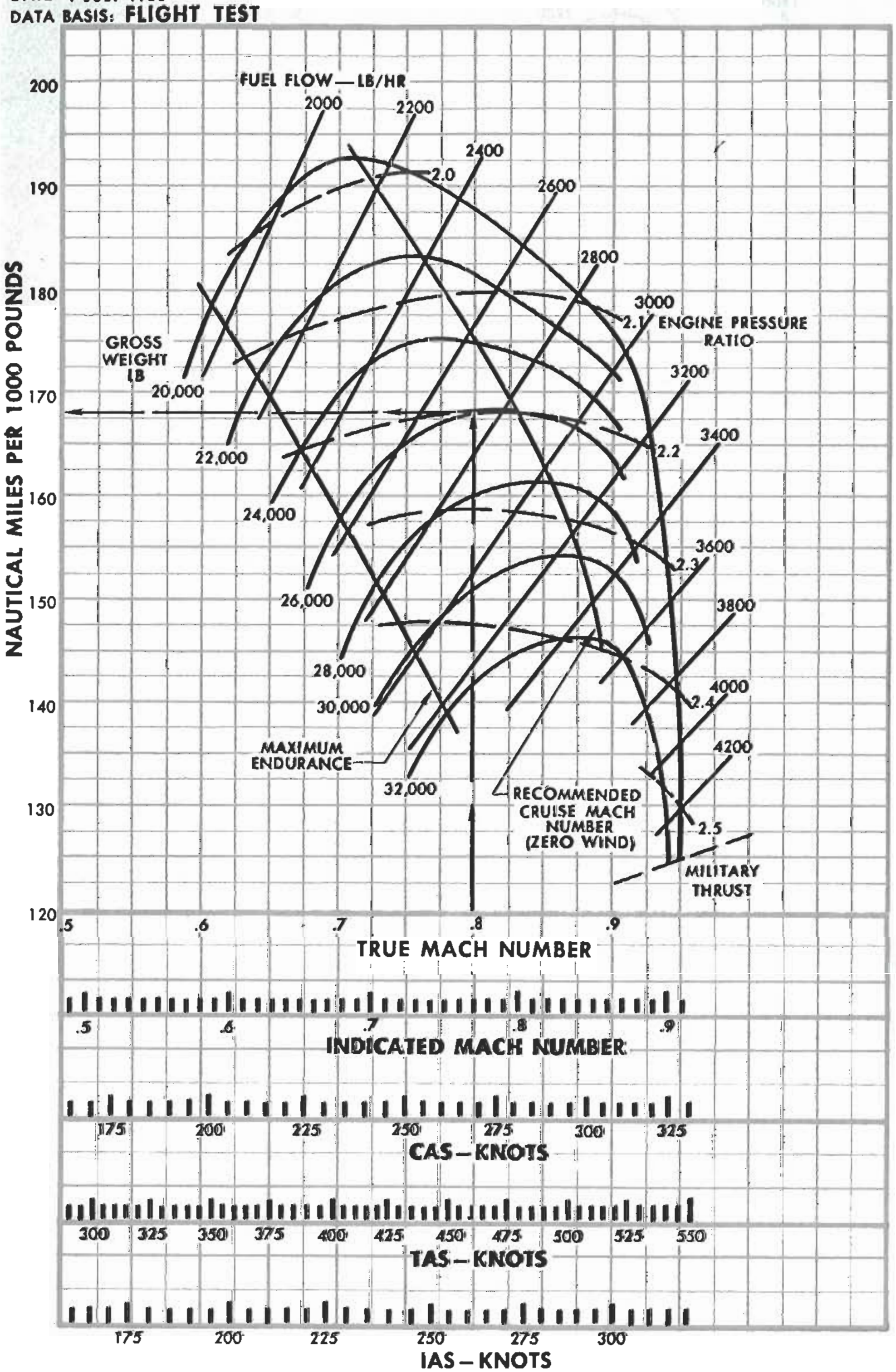
Figure A4-24

nautical miles per 1000 pounds—35,000 feet

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE:

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 11%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FOR EMERGENCY CRUISE DATA, THE VALUES OBTAINED FROM THIS CHART SHOULD BE REDUCED BY THE FOLLOWING PERCENTS: 13% REDUCTION IN OPTIMUM CRUISE SPEED, 37% REDUCTION IN MAXIMUM MILES PER POUND OF FUEL. THESE DATA ARE BASED ON ARMAMENT BAY DOORS OPEN AND AFT LAUNCHERS EXTENDED.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22045 D

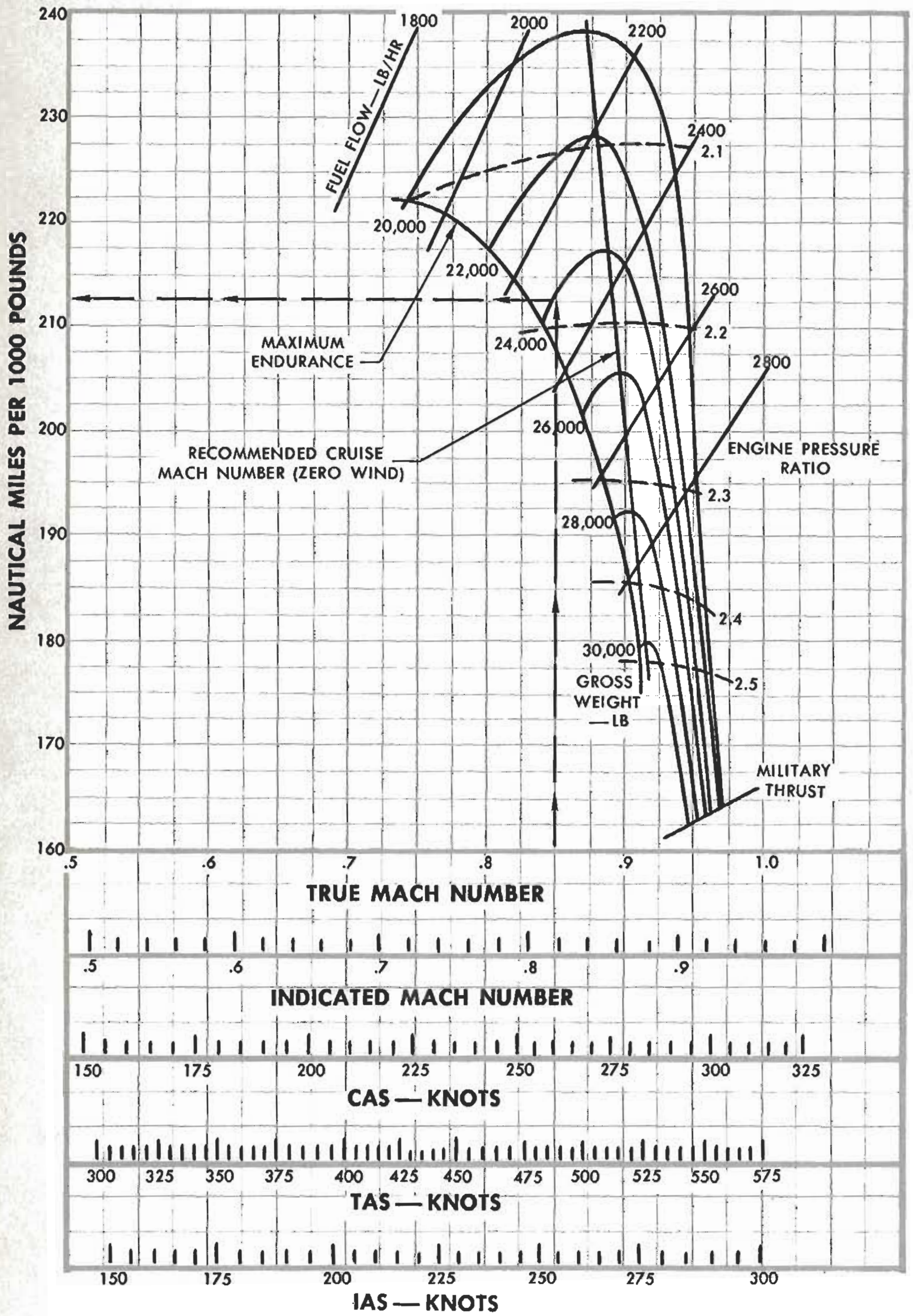
Figure A4-25

nautical miles per 1000 pounds — 40,000 feet

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS:

CONFIGURATION: CLEAN ● STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE:

- WIND CORRECTION TO N. MI/LB

  GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$

  WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.

- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 10% FOR TAILWIND APPLY OPPOSITE CORRECTION.

- FOR EACH 10° ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.

- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

22037D

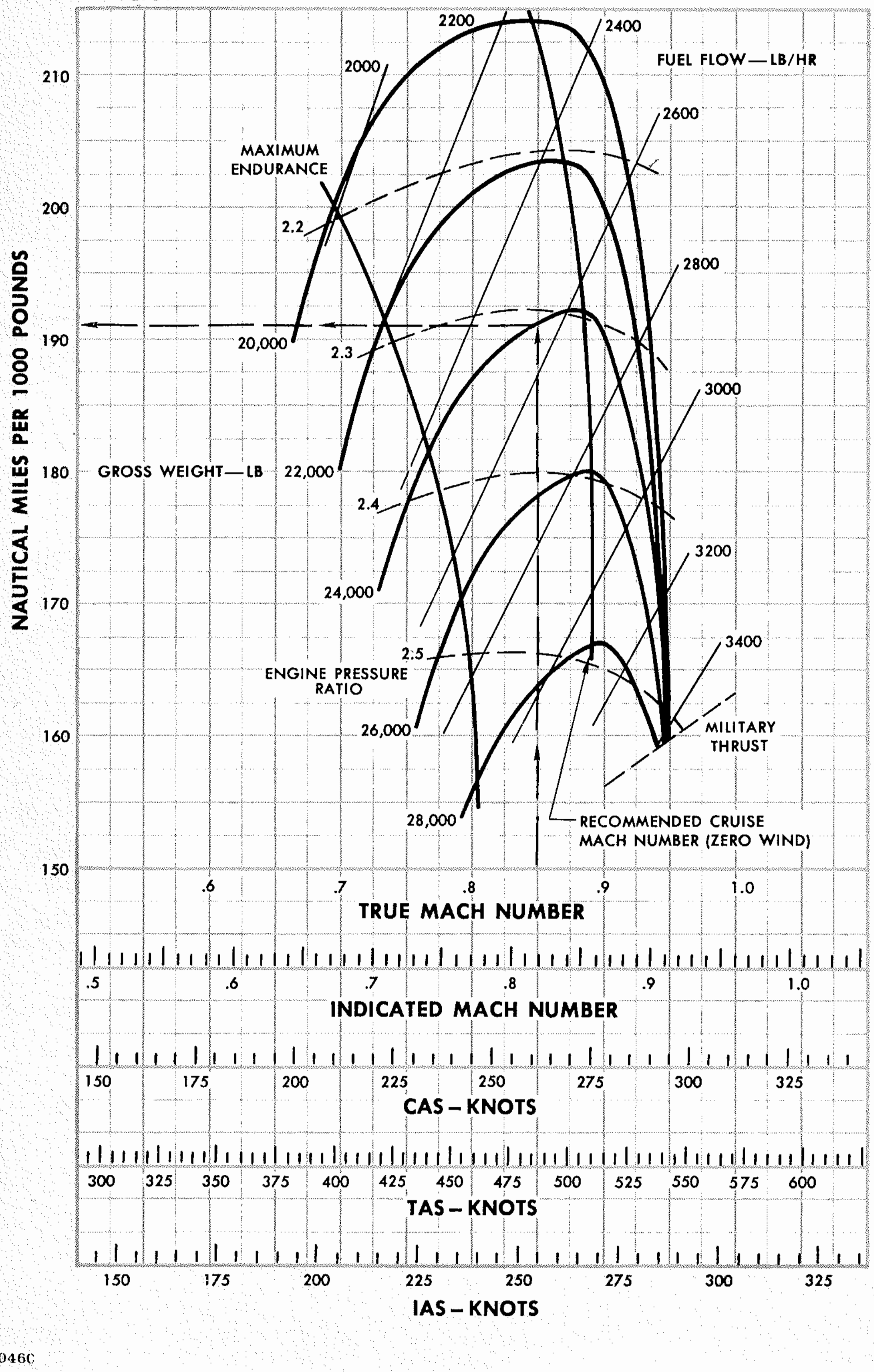
Figure A4-26

nautical miles per 1000 pounds—40,000 feet

MODEL: F-102A
DATA: 1 JULY 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 10%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

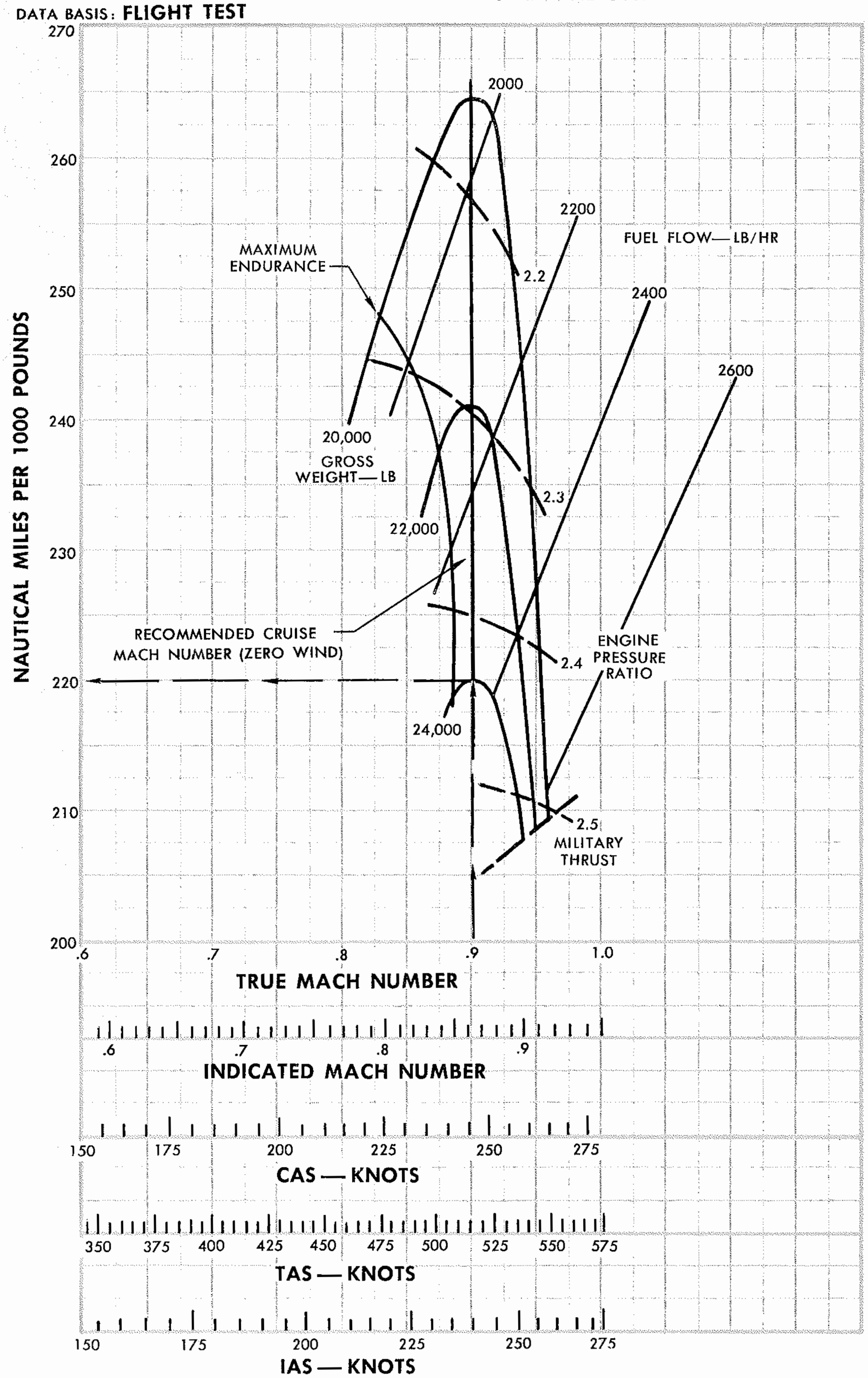
Figure A4-27

nautical miles per 1000 pounds — 45,000 feet

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

CONFIGURATION: CLEAN
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL.



NOTE:

- WIND CORRECTION TO N. MI/LB

  GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$

  WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.

- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 10% FOR TAILWIND APPLY OPPOSITE CORRECTION.

- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.

- FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).

21983A

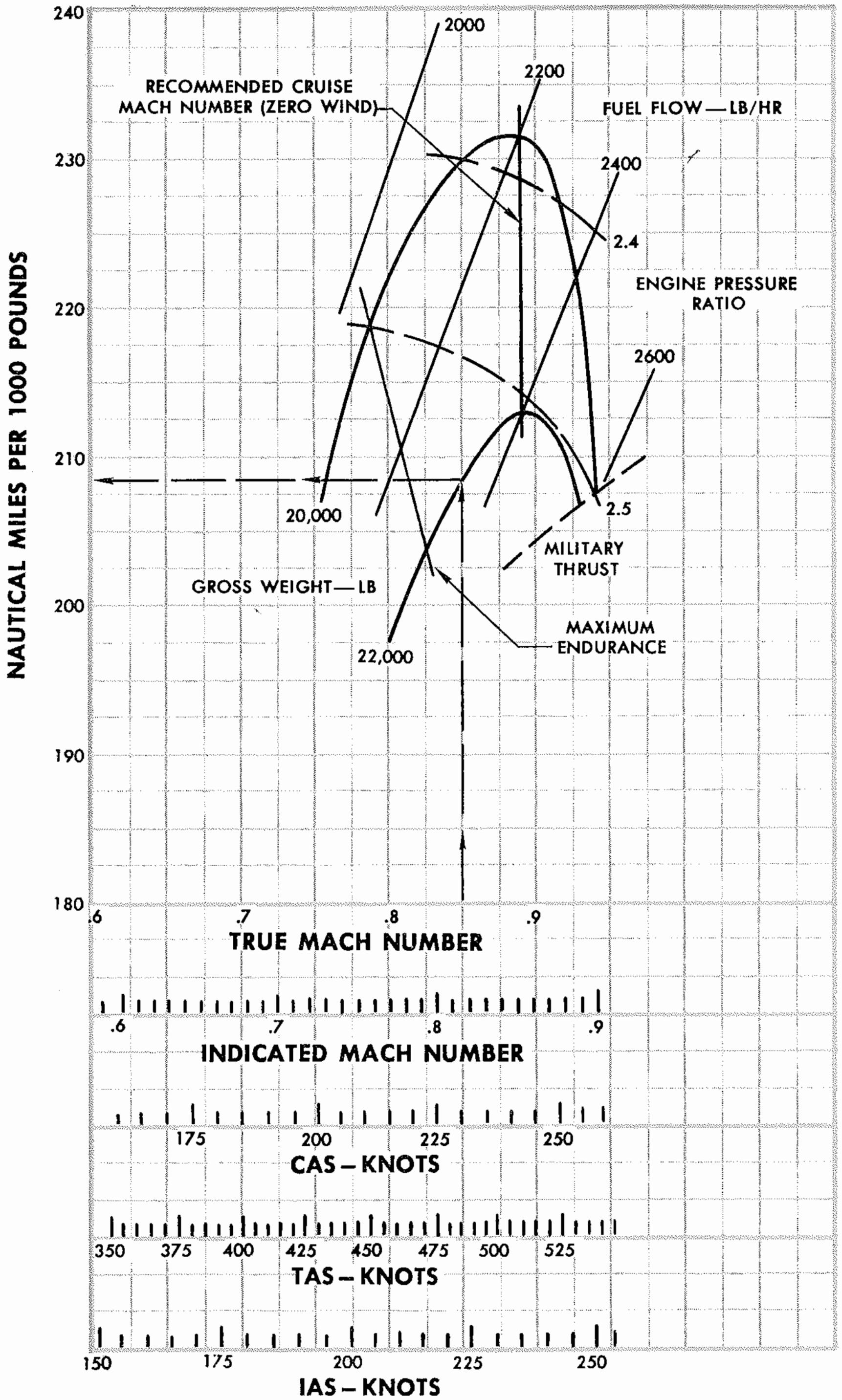
Figure A4-28

nautical miles per 1000 pounds—45,000 feet

CONFIGURATION: TWO 230-GALLON EXTERNAL TANKS
STANDARD DAY

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



NOTE

- WIND CORRECTION TO N. MI/LB GROUND N. MI/LB = AIR N. MI/LB $\times \frac{V_{GROUND}}{V_{AIR}}$ WHERE V_{AIR} IS AIRPLANE TRUE AIRSPEED AND V_{GROUND} IS GROUND SPEED.
- FOR EACH 50 KNOTS HEADWIND INCREASE RECOMMENDED CRUISE MACH NO. .01 AND DECREASE RANGE 10%. FOR TAILWIND APPLY OPPOSITE CORRECTION.
- FOR EACH 10°C ABOVE STANDARD DAY TEMPERATURE AT A GIVEN TRUE MACH NUMBER, INCREASE TRUE AIRSPEED AND FUEL FLOW 2%. NAUTICAL MILES PER POUND, TRUE MACH NUMBER, AND ENGINE PRESSURE RATIO REMAIN THE SAME. FOR EACH 10°C BELOW STANDARD DAY TEMPERATURE, APPLY OPPOSITE CORRECTION.
- **FUEL FLOW WILL VARY DEPENDING ON ATMOSPHERIC CONDITIONS (FROM 200 TO 500 POUNDS).**

22060A

Figure A4-29

PART 5 ENDURANCE

TABLE OF CONTENTS

MAXIMUM ENDURANCE PROFILE

Figures A5-2 and A5-3 show the constant altitude endurance characteristics for the clean airplane and the airplane with external fuel tanks, respectively. These charts are plots of endurance time as a function of altitude for various amounts of fuel remaining. The recommended operating speeds are tabulated, at 5000-foot intervals, on the chart.

Use

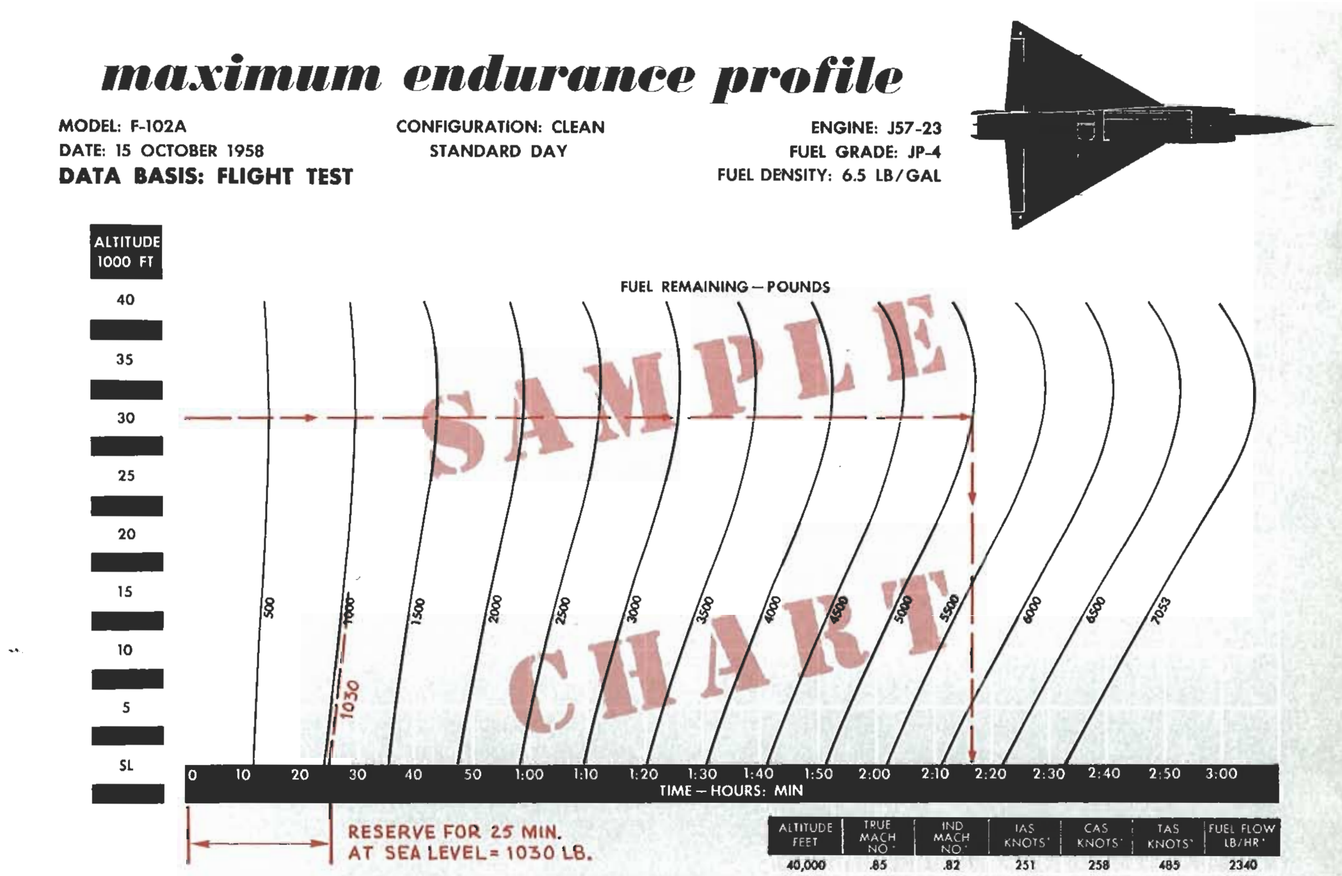
Maximum endurance time available for a specified amount of fuel may be found by entering the chart at the endurance altitude, reading right to the desired fuel remaining, and then down to the maximum endurance time available. Conversely, fuel required for endurance for a specific time may be determined by entering the chart with the endurance time, reading up to the endurance altitude and interpolating if necessary for the fuel required.

Note

Fuel allowance for descent and landing has not been included.

Sample Problem

Find the maximum endurance time available with 5000 pounds of fuel remaining for endurance at 30,000 feet. The airplane is in its clean configuration:



ALTITUDE FEET	TRUE MACH NO.*	IND MACH NO.*	IAS KNOTS*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*
40,000	.85	.82	251	258	485	2340
35,000	.72	.69	233	240	412	2280
30,000	.64	.62	232	238	375	2280
25,000	.54	.53	219	225	327	2330
20,000	.49	.47	218	224	301	2420
15,000	.44	.43	216	223	278	2500
10,000	.40	.39	215	222	257	2580
5,000	.37	.35	213	220	237	2650
S.L.	.32	.31	205	212	212	2780

*APPROXIMATE

22119
22053

Figure A5-1

a. Enter figure A5-1 at the endurance altitude—30,000 feet.
b. Proceed right to the fuel remaining—5000 pounds.
c. Read down for the maximum endurance time—2 hours and 17 minutes.

OPTIMUM MAXIMUM ENDURANCE PROFILE

The maximum possible endurance time available for a given fuel quantity at any initial altitude is shown in figure A5-4 for the clean airplane and in figure A5-5 for the airplane with external tanks. These charts are based on an optimum flight path from any starting altitude. The basic flight path consists of a military thrust climb or an idle thrust descent to the optimum endurance altitude (where required), a loiter at constant altitude, and a maximum range descent to sea level with idle thrust. Notes and guide lines on the charts indicate the proper flight path to be used. Tabular data include speeds for climb, descent, and loiter, and also for loiter fuel flows at 5000-foot altitude intervals.

Use

Mission planning with the optimum maximum endurance profile is the same as with the optimum profile except that cruise distance is replaced by loiter time, and the flight path utilizes an idle thrust descent.

Note

Fuel allowance for landing is not included.

COMBAT ALLOWANCE CHART

The combat allowance chart, figure A5-6 shows the relationship between time and fuel consumed for military thrust high-speed and maximum thrust high-speed operation. Basically these charts show time as a function of altitude for various amounts of combat fuel available. Supplementary charts of fuel flow in pounds per minute as a function of altitude for the same speed and thrust conditions are also shown.

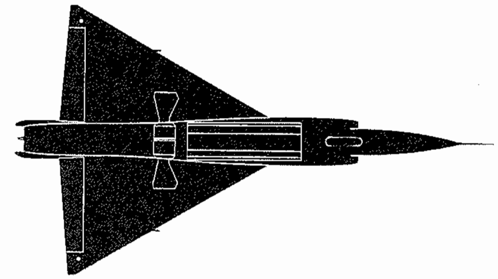
Use

To find the combat time available for a given amount of fuel, enter the chart at the combat altitude, read to the right to the allotted fuel, and then read down for the time available. For maximum or military thrust fuel flows, enter the chart at the desired altitude and proceed horizontally to the respective fuel flow curve and then down to get the fuel flow in pounds per minute.

Note

Does not include fuel to accelerate to combat Mach number.

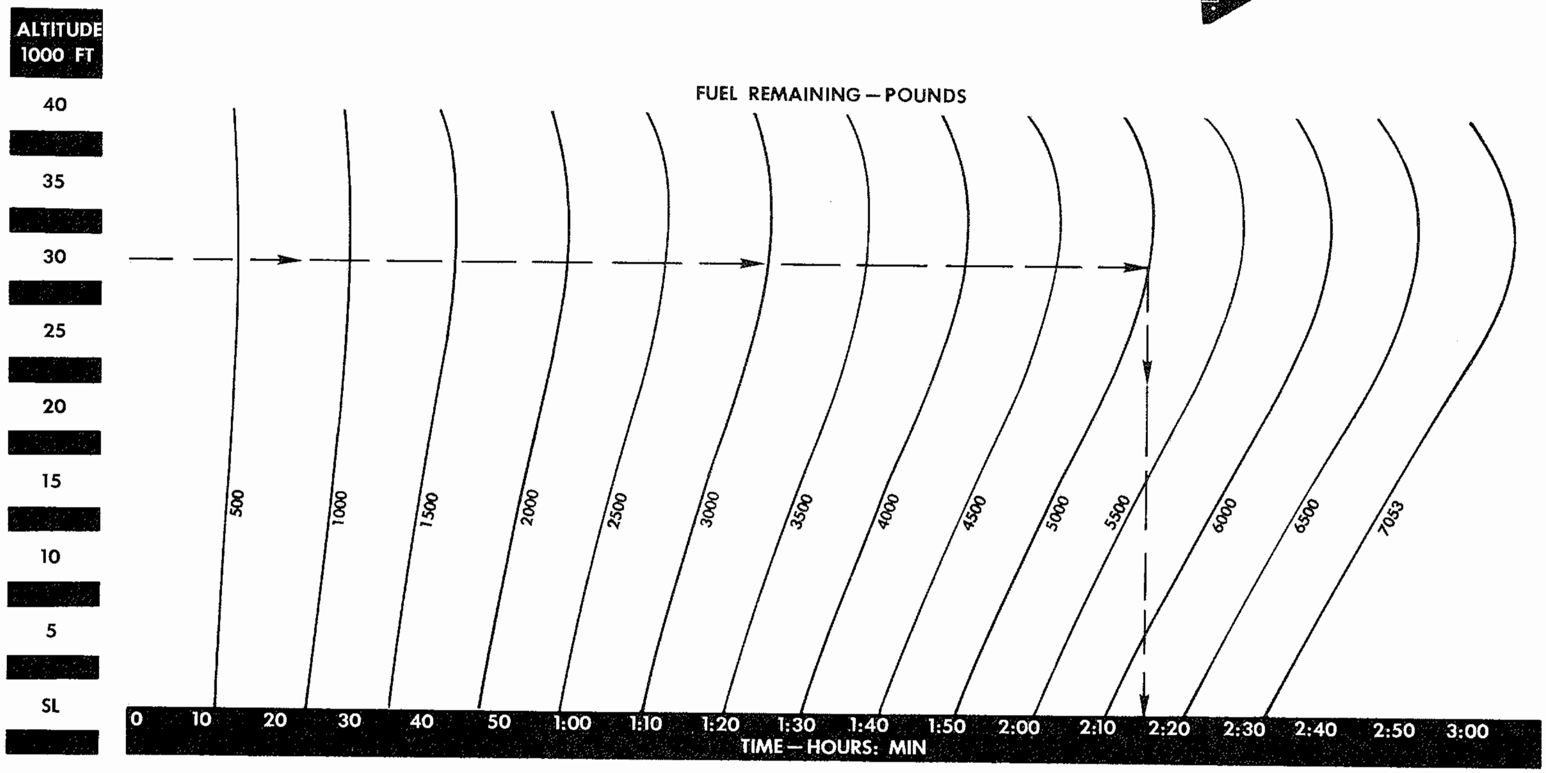
maximum endurance profile



MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL



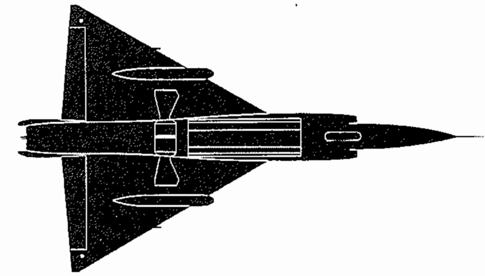
ALTITUDE FEET	TRUE MACH NO.*	IND MACH NO.*	IAS KNOTS*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*
40,000	.85	.82	251	258	485	2340
35,000	.72	.69	233	240	412	2280
30,000	.64	.62	232	238	375	2280
25,000	.54	.53	219	225	327	2330
20,000	.49	.47	218	224	301	2420
15,000	.44	.43	216	223	278	2500
10,000	.40	.39	215	222	257	2580
5,000	.37	.35	213	220	237	2650
S.L.	.32	.31	205	212	212	2780

*APPROXIMATE

22053

Figure A5-2

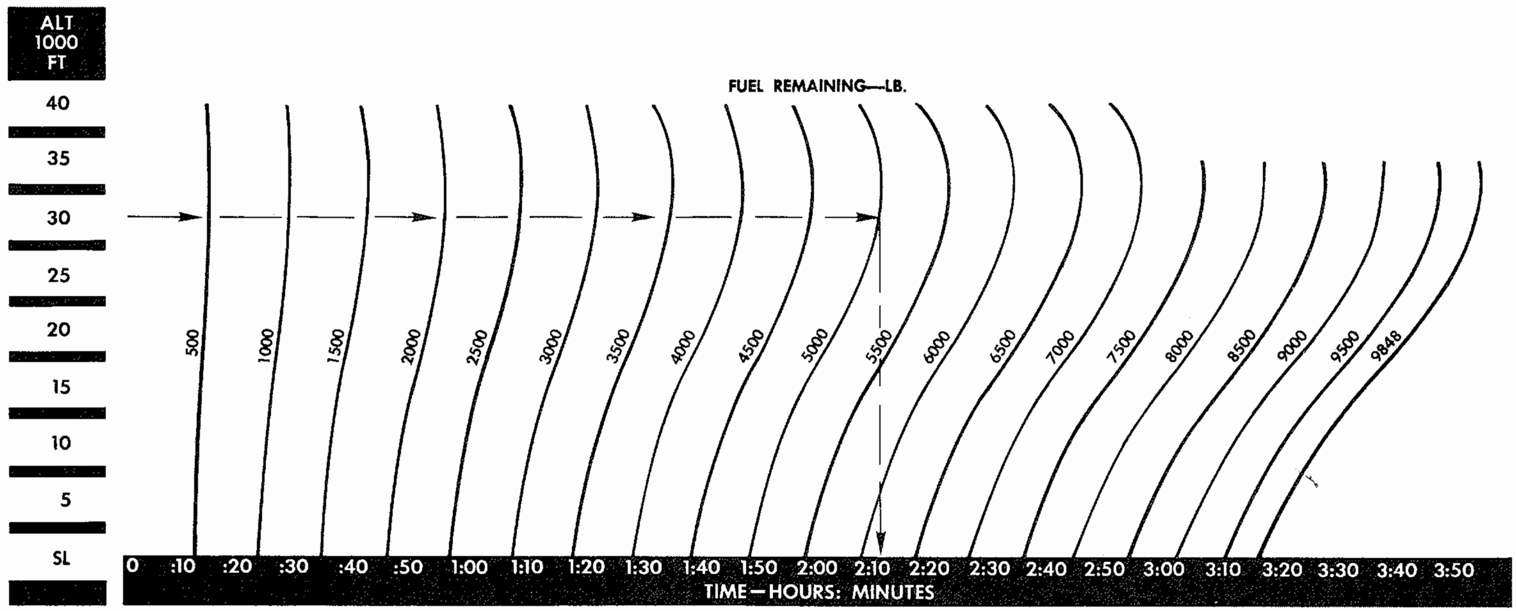
maximum endurance profile



MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO – 230 GALLON EXTERNAL TANKS
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

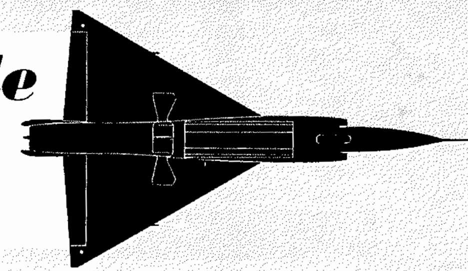


ALTITUDE FEET	TRUE MACH NO.	IND. MACH NO.	IAS KNOTS	CAS KNOTS	TAS KNOTS	FUEL FLOW LB/HR
40,000	.79	.76	234	240	455	2670
35,000	.70	.68	229	235	404	2560
30,000	.62	.60	227	233	368	2580
25,000	.55	.54	223	229	333	2600
20,000	.49	.48	218	224	302	2700
15,000	.44	.43	216	223	278	2800
10,000	.40	.38	211	218	253	2890
5,000	.36	.33	210	217	234	2960
S.L.	.33	.32	221	228	228	3040

Figure A5-3

22054

optimum maximum endurance profile

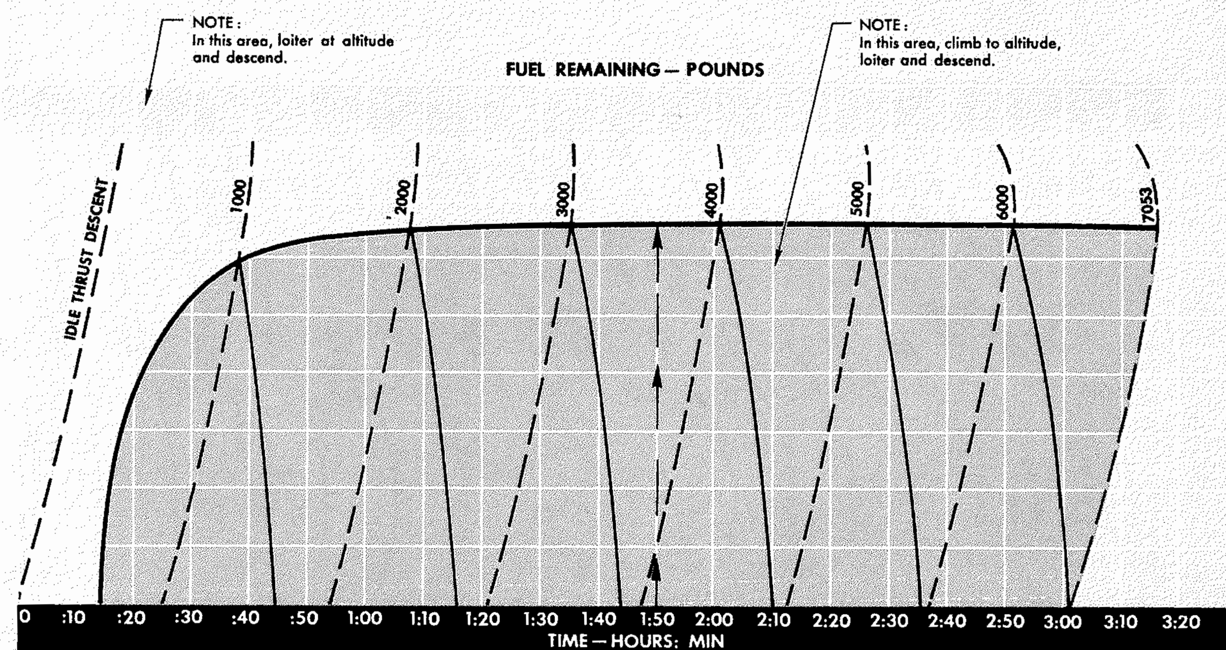


MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD DAY
CLIMB-LOITER-DESCEND

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

MILITARY THRUST CLIMB					
TAS KNOTS	CAS KNOTS	IAS KNOTS	IND MACH NO.	TRUE MACH NO.	ALT 1000 FEET
516	276	269	.87	.90	40
519	310	302	.87	.90	35
483	311	303	.79	.82	30
451	314	306	.72	.75	25
419	315	307	.66	.68	20
393	317	309	.61	.63	15
370	321	313	.56	.58	10
351	326	318	.52	.54	5
331	331	322	.48	.50	SL



ALTITUDE FEET	TRUE MACH NO.*	IND MACH NO.*	IAS KNOTS*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*
40,000	.85	.82	251	258	485	2340
35,000	.72	.69	233	240	412	2280
30,000	.64	.62	232	238	375	2280
25,000	.54	.53	219	225	327	2330
20,000	.49	.47	218	224	301	2420
15,000	.44	.43	216	223	278	2500
10,000	.40	.39	215	222	257	2580
5,000	.37	.35	213	220	237	2650
S. L.	.32	.31	205	212	212	2780

*APPROXIMATE

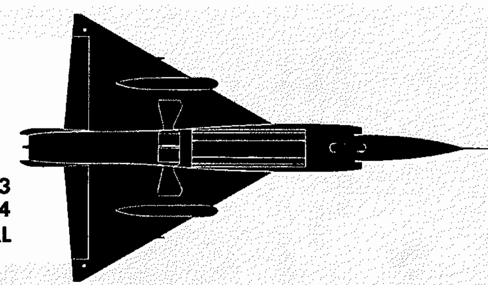
—— OPTIMUM LEVEL FLIGHT ENDURANCE ALTITUDE
—— CLIMB PATH GUIDE LINES
— — — FUEL CONSUMED OR AVAILABLE

NOTE:
1. Fuel required at any point includes military thrust climb if required, to altitude, loiter and descend to sea level.
2. No allowance or reserve made for landing.

22055

Figure A5-4

optimum maximum endurance profile



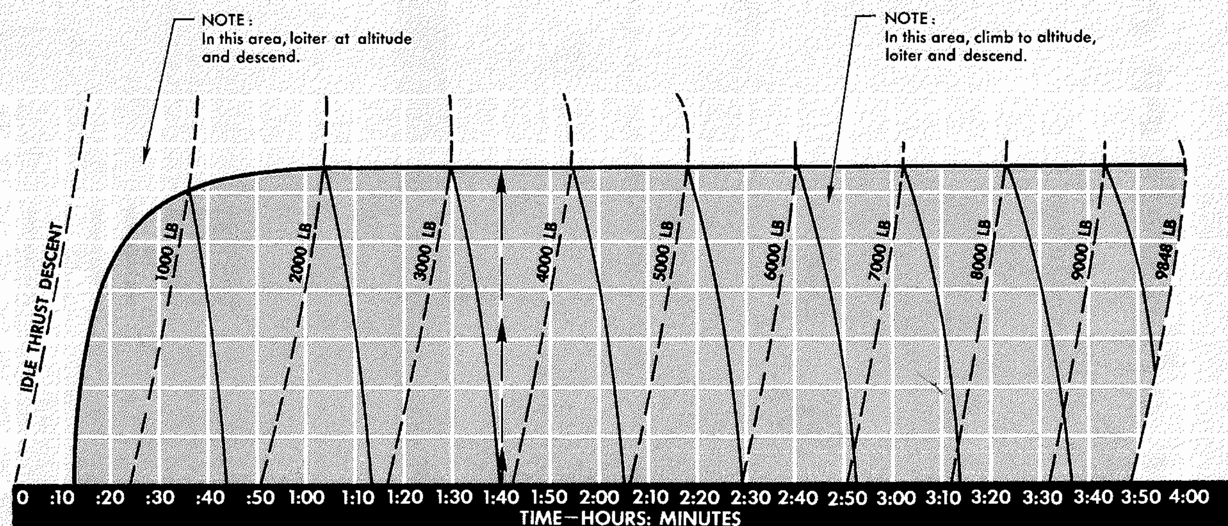
MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: TWO-230 GALLON EXTERNAL TANKS
STANDARD DAY
CLIMB-LOITER-DESCEND

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL

MILITARY THRUST CLIMB

TAS KNOTS	CAS KNOTS	IAS KNOTS	IND MACH NO.	TRUE MACH NO.	ALT 1000 FEET
462	244	237	.78	.81	40
464	273	266	.78	.81	35
427	273	266	.70	.73	30
400	277	270	.64	.67	25
376	281	274	.59	.61	20
354	285	277	.55	.57	15
337	292	284	.51	.53	10
320	296	288	.48	.49	5
304	304	295	.45	.46	SL



ALTITUDE FEET	TRUE MACH NO.*	IND MACH NO.*	IAS KNOTS*	CAS KNOTS*	TAS KNOTS*	FUEL FLOW LB/HR*
40,000	.79	.76	234	240	455	2670
35,000	.70	.68	229	235	404	2560
30,000	.62	.60	227	233	368	2580
25,000	.55	.54	223	229	333	2600
20,000	.49	.48	218	224	302	2700
15,000	.44	.43	216	223	278	2800
10,000	.40	.38	211	218	253	2890
5,000	.36	.35	210	217	234	2960
SL	.33	.32	221	228	228	3040

*APPROXIMATE

——— OPTIMUM LEVEL FLIGHT ENDURANCE ALTITUDE
——— CLIMB PATH GUIDE LINES
– – – – FUEL CONSUMED OR AVAILABLE

NOTE:
1. Fuel required at any point includes Military Thrust Climb (if required) to altitude, loiter and descend to sea level.
2. No allowance or reserve made for landing.

22133

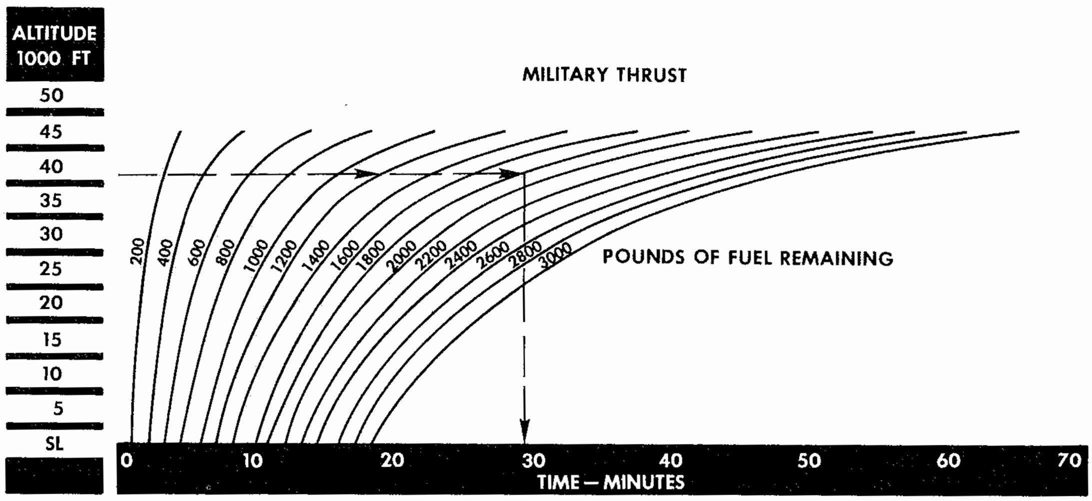
Figure A5-5

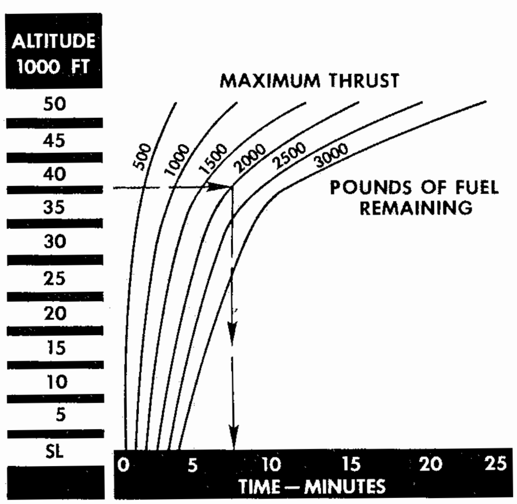
combat allowance chart

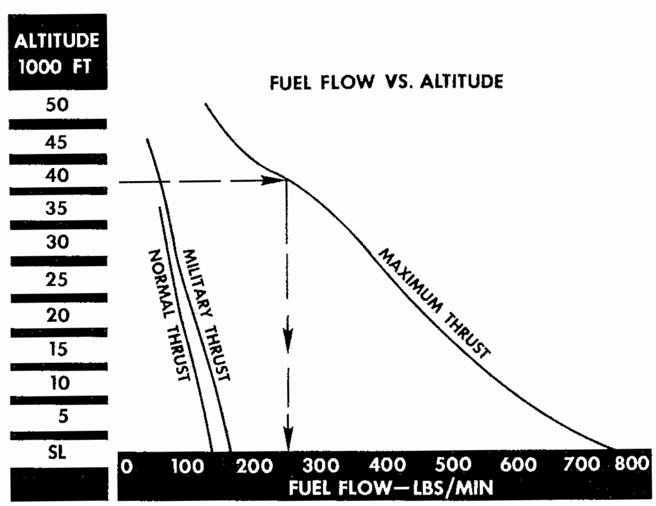
MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: FLIGHT TEST

CONFIGURATION: CLEAN
STANDARD DAY

ENGINE: J57-23
FUEL GRADE: JP-4
FUEL DENSITY: 6.5 LB/GAL







22056

Figure A5-6

PART 6 DESCENT

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DESCENTS

Figures A6-2 through A6-4 graphically illustrate the descent characteristics. The data shown are fuel used, rate of descent, time to descend and distance traveled versus pressure altitude for various airplane configurations. Speed schedules are noted on the charts and in the remarks.

Use

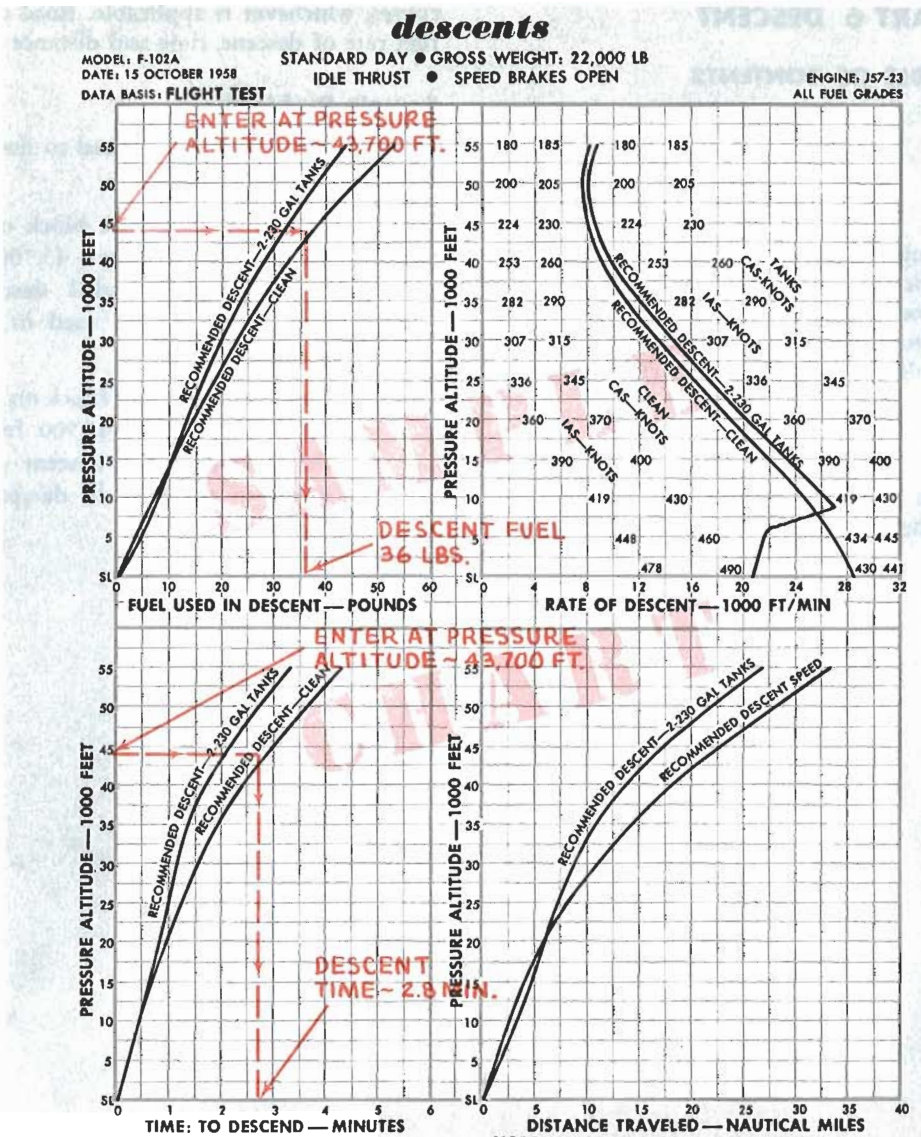
Enter charts on the altitude scale and proceed right to the "recommended descent" or "maximum range descent" curves, whichever is applicable. Read down for required fuel rate of descent, time and distance traveled.

Sample Problem

Find the time and fuel required to descend from 43,700 feet clean configuration.

a. Enter the upper left block on figure A6-1 on the pressure altitude — 43,700 feet. Read right to the "recommended descent — clean" line, then down to fuel used in descent. Read 37 pounds of fuel.

b. Enter the lower left block on figure A6-1 on the pressure altitude — 43,700 feet. Read right to the "recommended descent — clean" line, then down to the time to descend line. Read 2.8 minutes.



REMARKS
RECOMMENDED DESCENT—
DESCEND AT SPEEDS NOTED ABOVE.

NOTE DURING OPERATION ABOVE 40 000 FEET.
DO NOT REDUCE ENGINE RPM BELOW 85%.
(REFER TO SECTIONS V, VII & IX.)

22118
22025D

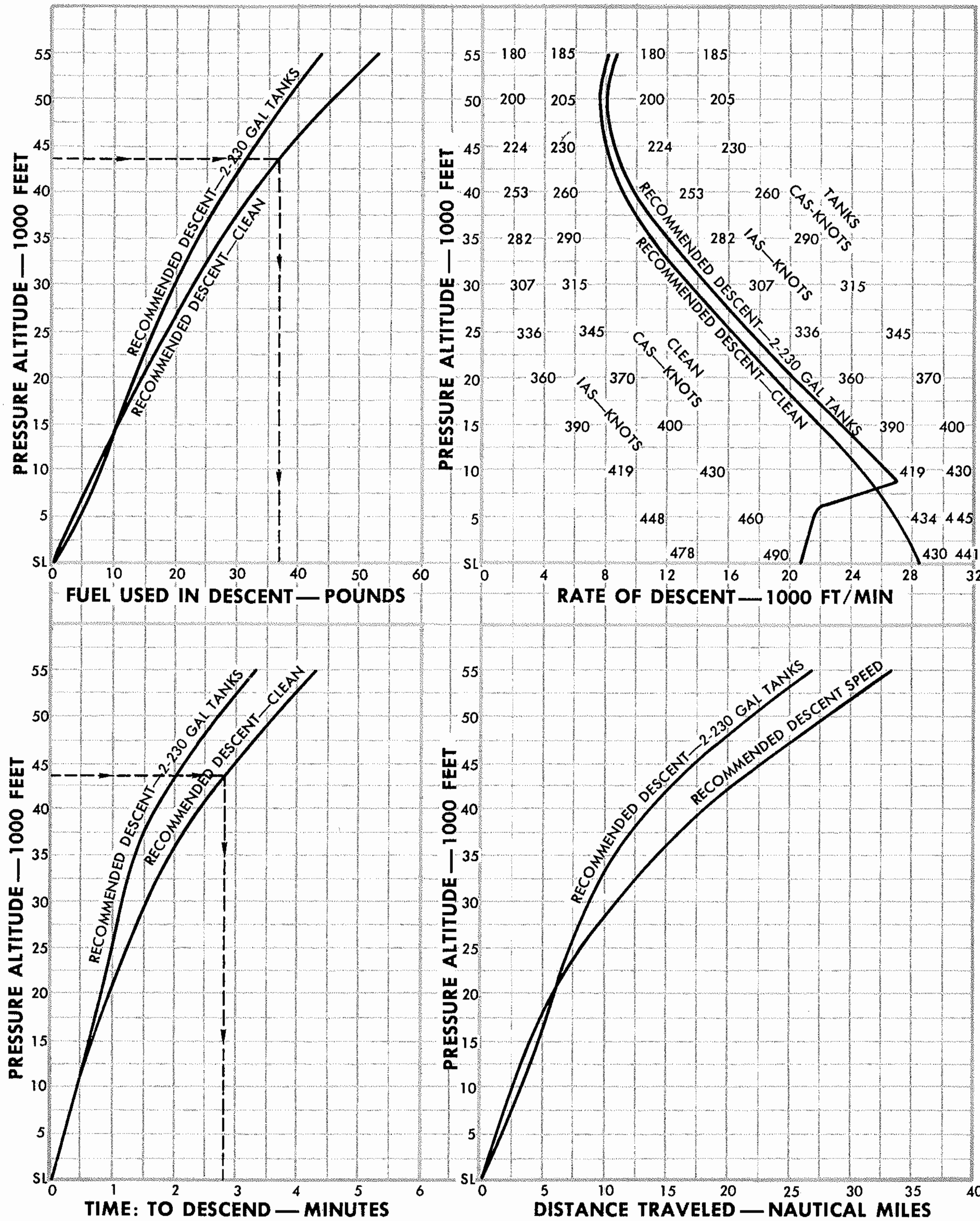
Figure A6-1

descents

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: **FLIGHT TEST**

STANDARD DAY ● GROSS WEIGHT: 22,000 LB
IDLE THRUST ● SPEED BRAKES OPEN

ENGINE: J57-23
ALL FUEL GRADES



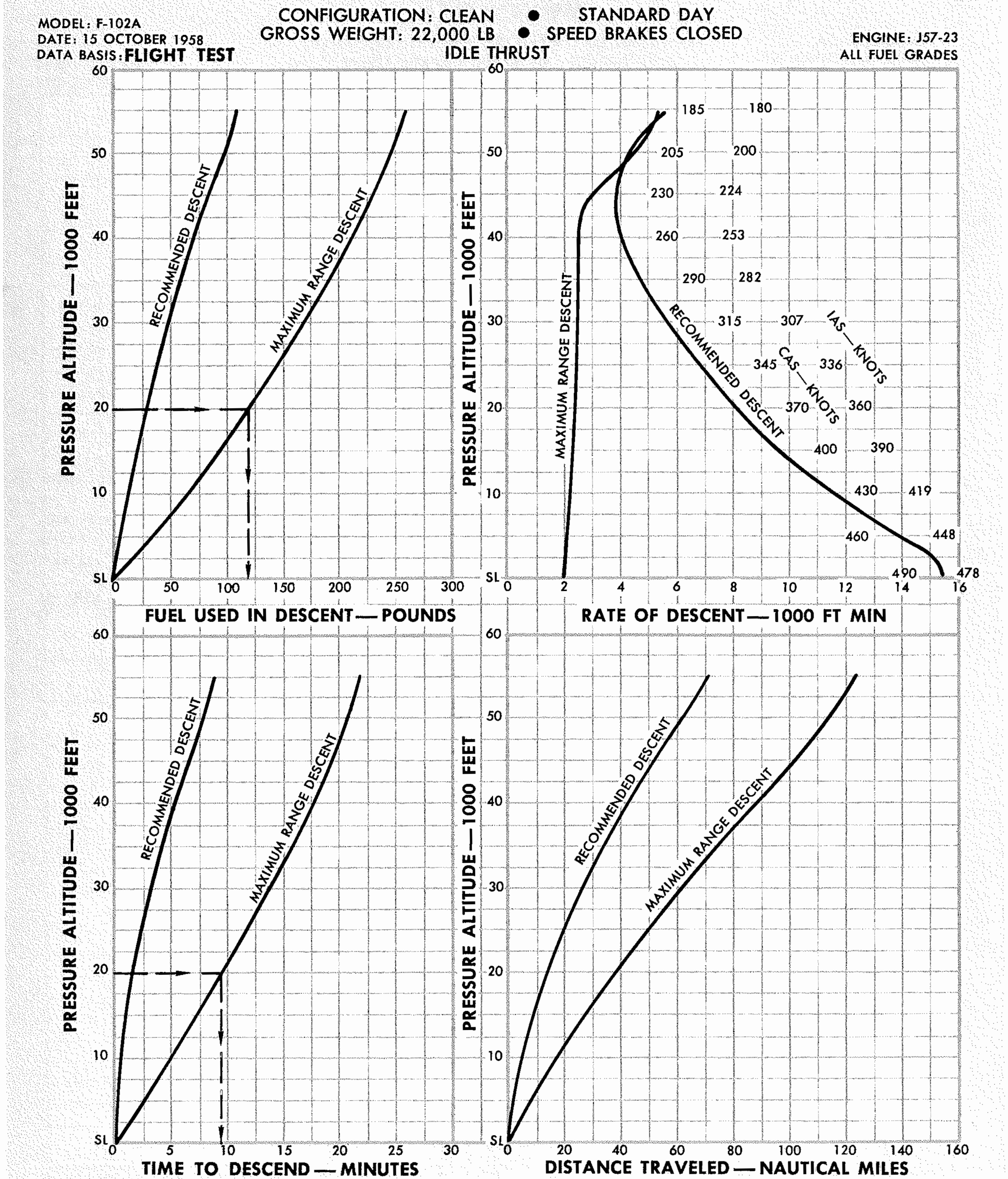
REMARKS
RECOMMENDED DESCENT—
DESCEND AT SPEEDS NOTED ABOVE.

NOTE DURING OPERATION ABOVE 40 000 FEET.
DO NOT REDUCE ENGINE RPM BELOW 85%.
(REFER TO SECTION V.)

22025D

Figure A6-2

descents



REMARKS

RECOMMENDED DESCENT—DESCEND AT SPEEDS NOTED ABOVE.
MAX RANGE DESCENT—DESCEND AT 210-230 KNOTS IAS, NOT TO EXCEED .85 215-235 KNOTS CAS IND. MACH NUMBER.

NOTE DURING OPERATION ABOVE 40,000 FT. DO NOT REDUCE ENGINE RPM BELOW 85%. (REFER TO SECTION V.)

22023C

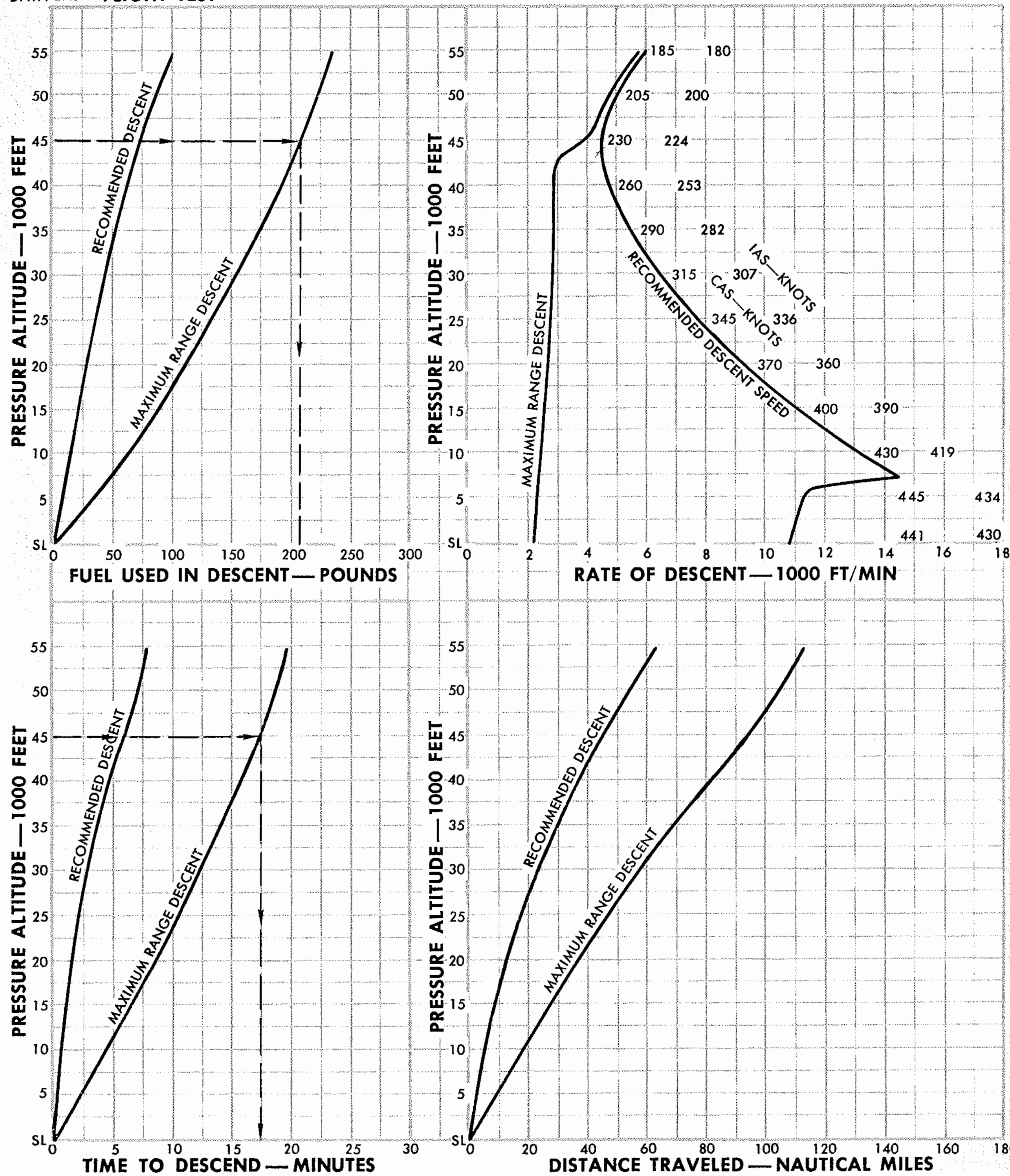
Figure A6-3

descents

CONFIGURATION: 2-230 GAL. TANKS ● GROSS WEIGHT: 22,000 LB
IDLE THRUST ● SPEED BRAKES CLOSED

MODEL: F-102A
DATE: 15 OCTOBER 1958
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23
ALL FUEL GRADES



REMARKS

RECOMMENDED DESCENT—DESCEND AT SPEEDS NOTED ABOVE
MAXIMUM RANGE DESCENT—DESCEND AT 210-230 KNOTS IAS NOT TO EXCEED 215-235 KNOTS CAS; .85 IND. MACH NUMBER.

NOTE DURING OPERATION ABOVE 40,000 FT. DO NOT REDUCE ENGINE RPM BELOW 85%. (REFER TO SECTION V.)

22024D

Figure A6-4

PART 7 APPROACH AND LANDING

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LANDING DISTANCES

Landing distances are shown for two basic configurations: figure A7-3, speed brakes open, and figure A7-4, speed brakes open with drag chute deployed at touchdown. The data are valid for the airplane with or without external tanks within the weight range shown when landing at recommended speeds. These charts account for the effects of ambient temperature, pressure altitude, headwind and tailwind, braking coefficient of friction, and landing over a 50-foot obstacle. The variation in ground roll due to increased touchdown speeds and wet runway is shown in figures A7-5 and A7-6. These data may be used with the normal landing charts to determine the corrected ground roll for a wet runway condition or increased landing speed.

Use

Enter the chart at the given ambient temperature, then proceed vertically to intersect the field pressure altitude. Proceed horizontally to the estimated gross weight line, then descend to the wind base line. Follow the wind guide line to the applicable headwind or tailwind, then descend to the desired braking effect line (interpolate if necessary). Proceed horizontally to ground roll scale and read ground roll. Continue horizontally to the corresponding braking line (interpolate if necessary), then vertically down to determine total distance to clear a 50-foot obstacle.

Sample Problem

Find the ground roll and total landing distance to clear a 50-foot obstacle for an airplane weighing 23,000 pounds and using the drag chute. Temperature is 35°C; headwind is 20-knots.

a. Enter figure A7-1 with the ambient air temperature — 35°C.

b. Read up to the pressure altitude — 3000 feet.

c. Read horizontally to the right for the landing gross weight — interpolate for 23,000 pounds.

d. Read down to the zero wind base line, then follow wind guide line to correct for headwind — 20 knots.

e. Continue down to braking line — between medium and light. Interpolate for braking corrected for 20-knot headwind.

f. Read left on ground roll scale — 3450 feet.

g. Continue left to the corresponding braking line, then vertically to determine total distance to clear 50-foot obstacle — 6800 feet.

h. Determine from recommended speeds table that touchdown speed is 140 KIAS (interpolate between touchdown speeds for 22,000 pounds and 24,000 pounds).

To find landing distance corrected for wet runway:

a. Enter the 20-knot increase in touchdown speed chart, figure A7-2, with ground roll distance of 2600 feet (obtained from figure A7-1) with zero wind and heavy braking correction.

b. Proceed vertically to wet runway lines and interpolate for 20-knot headwind.

c. Read left and read corrected ground roll distance — 5000 feet.

LANDING DATA CARD

F-102A
LANDING DATA CARD

CONDITIONS

RUNWAY LENGTH	______________
WIND	______________
OUTSIDE AIR TEMP	______________
PRESSURE ALTITUDE	______________
GROSS WEIGHT	______________

LANDING

FINAL APPROACH SPEED	______________
PRIOR TO FLARE SPEED	______________
TOUCHDOWN SPEED	______________
LANDING GROUND ROLL:	
Wheel Brakes Only	______________
Drag Chute Deployed	______________

The Landing Data Card must be filled out in conjunction with the Takeoff Data Card. Landing Data Card nomenclature definitions are as follows:

Conditions

1. Runway Length. Usable length of runway in feet.
2. Wind. The wind component parallel to the runway.
3. Outside Air Temperature. Runway air temperature in degrees Centigrade.
4. Pressure Altitude. Field pressure altitude obtained by setting altimeter to 29.92 inches Hg and reading altimeter dial.
5. Gross Weight. Gross weight of airplane in pounds at start of final approach.

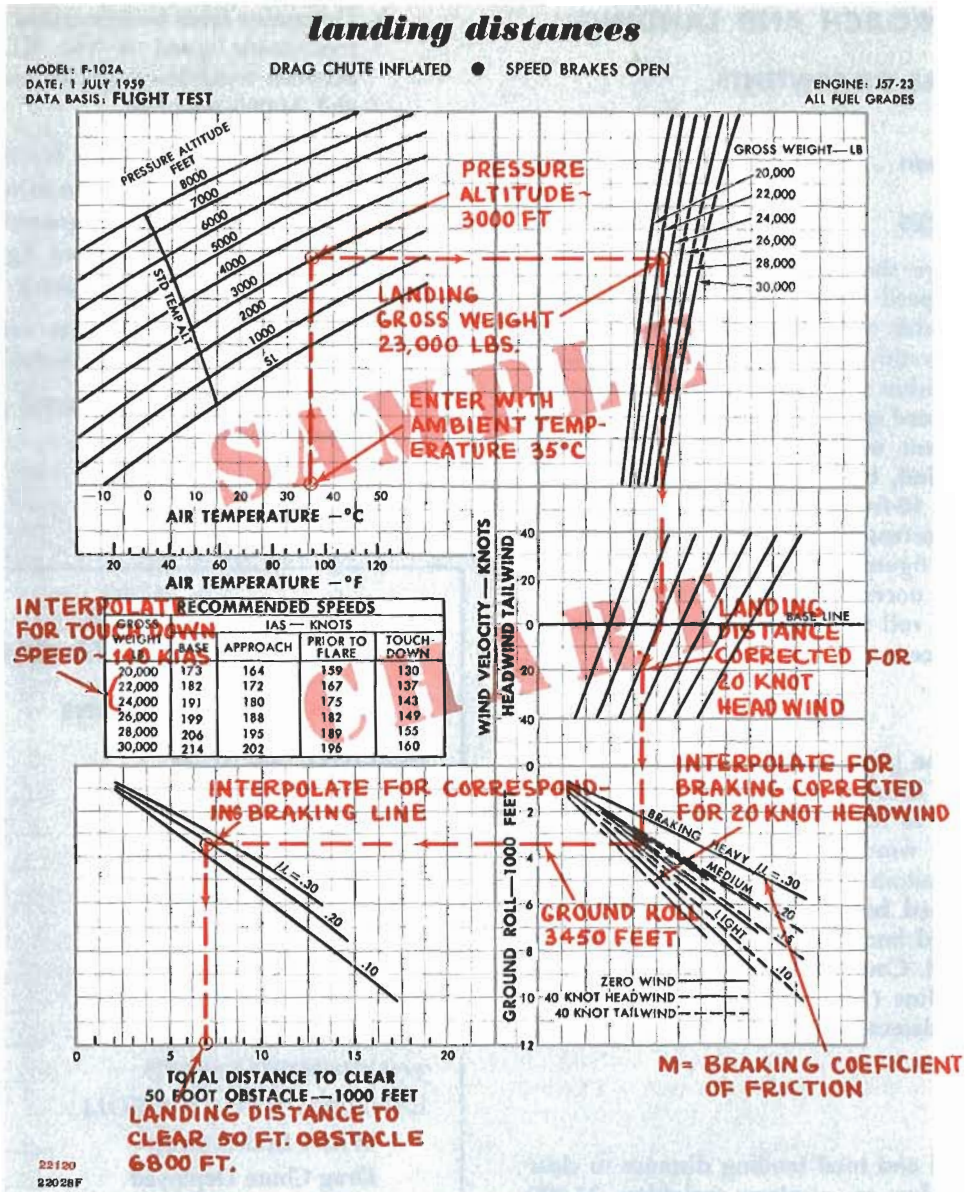


Figure A7-1

Landing

1. Final Approach Speed. Recommended indicated airspeed in knots for the latter portion of final approach is included in the landing distance charts (figure A7-3 or A7-4).
2. Prior To Flare Speed. Recommended prior to fiare speeds for various gross weights are included in figures A7-3 and A7-4.
3. Touchdown Speed. Recommended touchdown speeds (speed at time wheels contact runway) are included in figures A7-3 and A7-4.
4. Landing Ground Roll:
   a. Wheel Brakes Only. Distance in feet from airplane touchdown to full stop with no drag chute and speed brakes open, using wheel brakes only (figure A7-3).
   b. Drag Chute Deployed. Distance in feet from airplane touchdown to full stop with drag chute inflated at touchdown, speed brakes open, and using wheel brakes (figure A7-4).

CROSS-WIND LANDINGS

The minimum touchdown speeds for a given cross-wind condition are determined from the Takeoff and Landing Cross-Wind Chart, figure A2-6.

Use

Enter the chart on the wind direction line and follow it to the wind velocity circle, then proceed vertically to intercept the landing touchdown speed line. From the point of interception, move horizontally to the left scale and read the minimum touchdown speed. Compare this

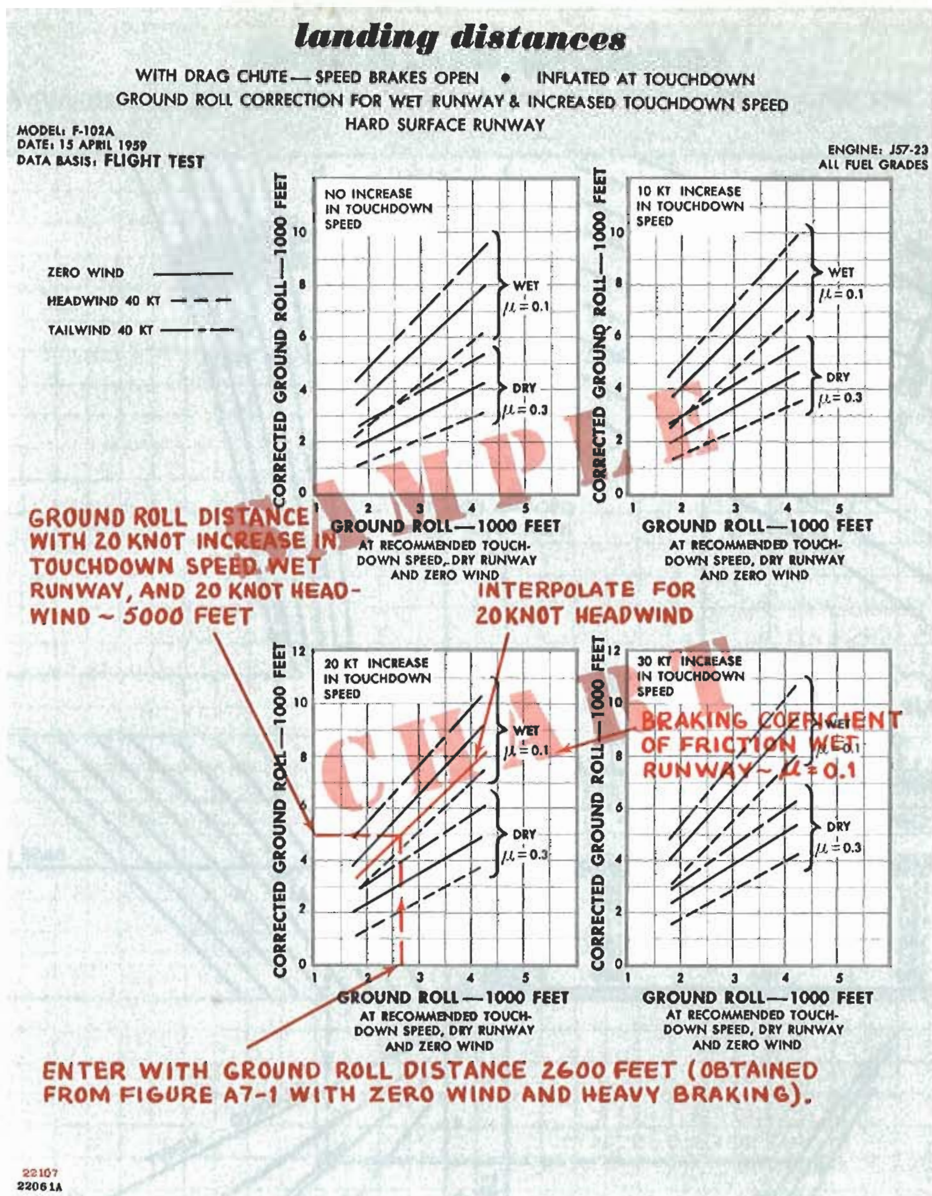


Figure A7-2

speed with the recommended touchdown speed as determined from the Landing Distance Chart, figure A7-3 or A7-4, and use whichever speed is higher.

Note

If landing touchdown speed is increased over that speed shown on the Landing Distance Chart, ground roll will be increased.

Sample Problem

Given: Wind of 33 knots, 65° relative to runway heading.
Determine: The minimum landing touchdown speed with this wind condition.

a. Enter figure A2-3 on the wind direction line — 65°.

b. Proceed to the wind velocity circle — 33 knots.

c. Move vertically to intercept the touchdown speed line.

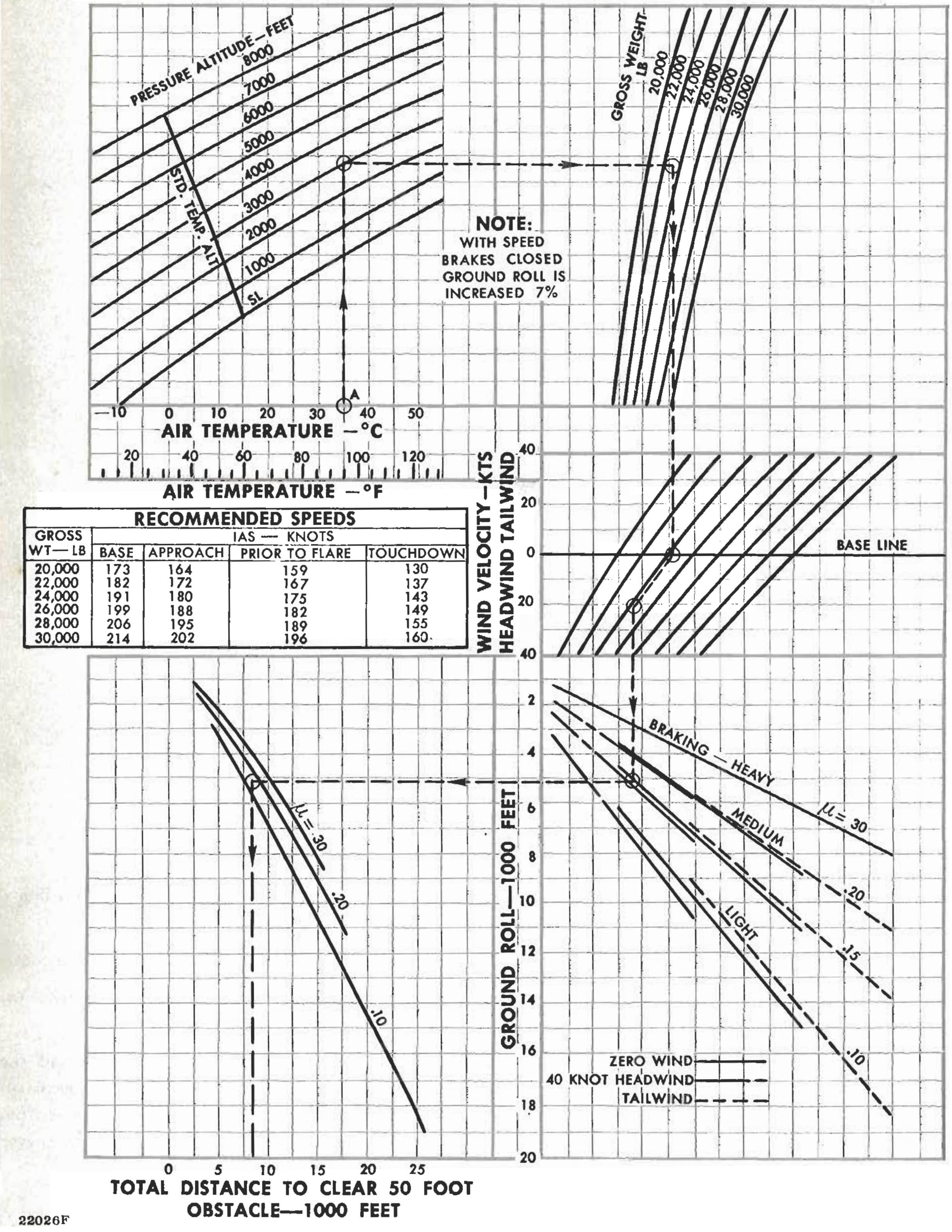
d. Move horizontally to the left and read the minimum touchdown speed — 157.5 knots. Compare this speed with the speed determined from the Landing Distance Chart and use whichever speed is higher.

landing distances

MODEL: F-102A
DATE: 1 JULY 1958
DATA BASIS: **FLIGHT TEST**

NO DRAG CHUTE — SPEED BRAKES OPEN ● HARD SURFACE DRY RUNWAY

ENGINE: J57-23
ALL FUEL GRADES



RECOMMENDED SPEEDS				
GROSS WT—LB	IAS — KNOTS			
	BASE	APPROACH	PRIOR TO FLARE	TOUCHDOWN
20,000	173	164	159	130
22,000	182	172	167	137
24,000	191	180	175	143
26,000	199	188	182	149
28,000	206	195	189	155
30,000	214	202	196	160

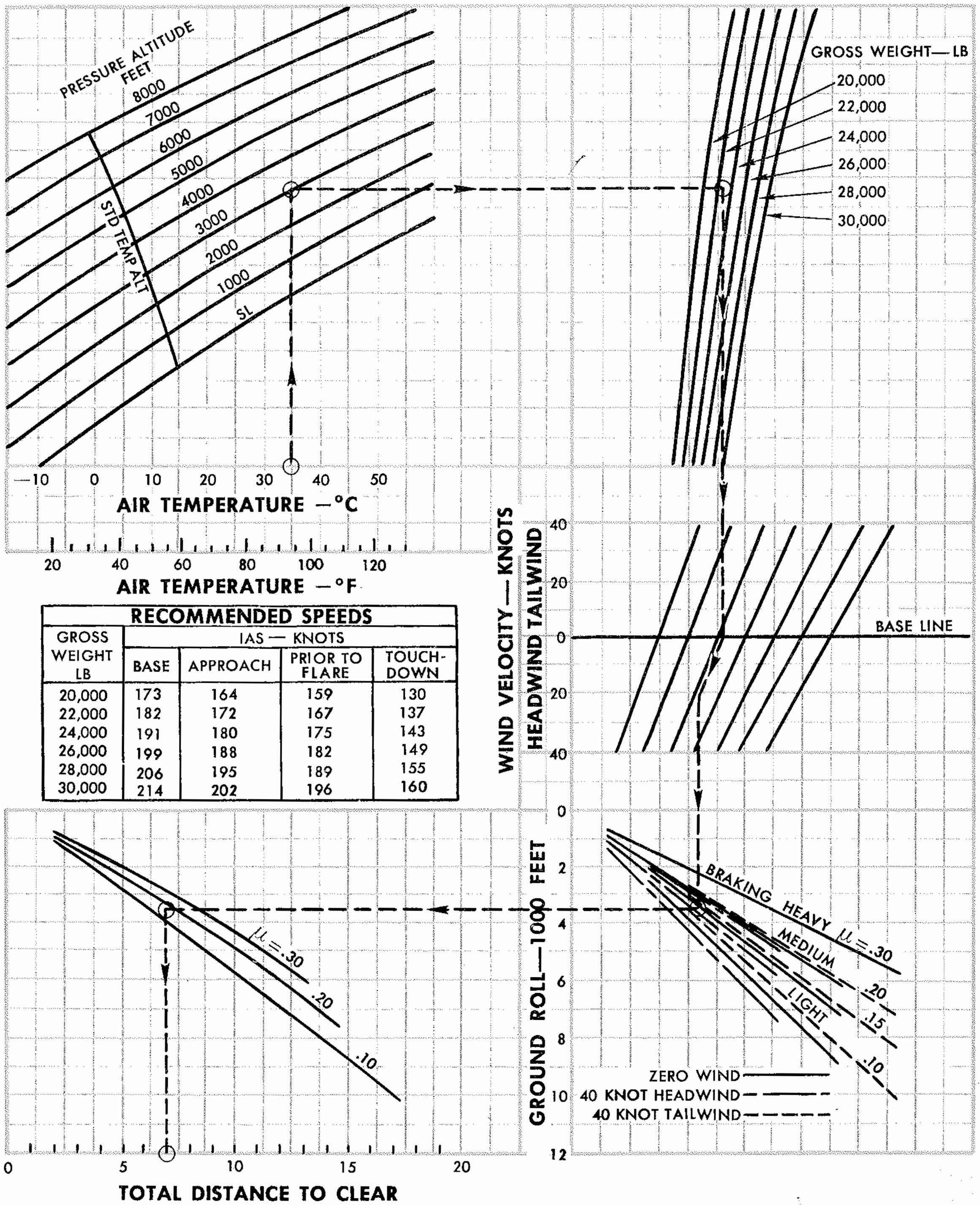
Figure A7-3

landing distances

MODEL: F-102A
DATE: 1 JULY 1959
DATA BASIS: **FLIGHT TEST**

DRAG CHUTE INFLATED ● SPEED BRAKES OPEN

ENGINE: J57-23
ALL FUEL GRADES



RECOMMENDED SPEEDS				
GROSS WEIGHT LB	IAS — KNOTS			
	BASE	APPROACH	PRIOR TO FLARE	TOUCH-DOWN
20,000	173	164	159	130
22,000	182	172	167	137
24,000	191	180	175	143
26,000	199	188	182	149
28,000	206	195	189	155
30,000	214	202	196	160

22028F

Figure A7-4

landing distances

MODEL: F-102A
DATE: 15 APRIL 1959
DATA BASIS: **FLIGHT TEST**

NO DRAG CHUTE — SPEED BRAKES OPEN ● HARD SURFACE RUNWAY
CORRECTION FOR WET RUNWAY & INCREASED TOUCHDOWN SPEED

ENGINE: J57-23
ALL FUEL GRADES

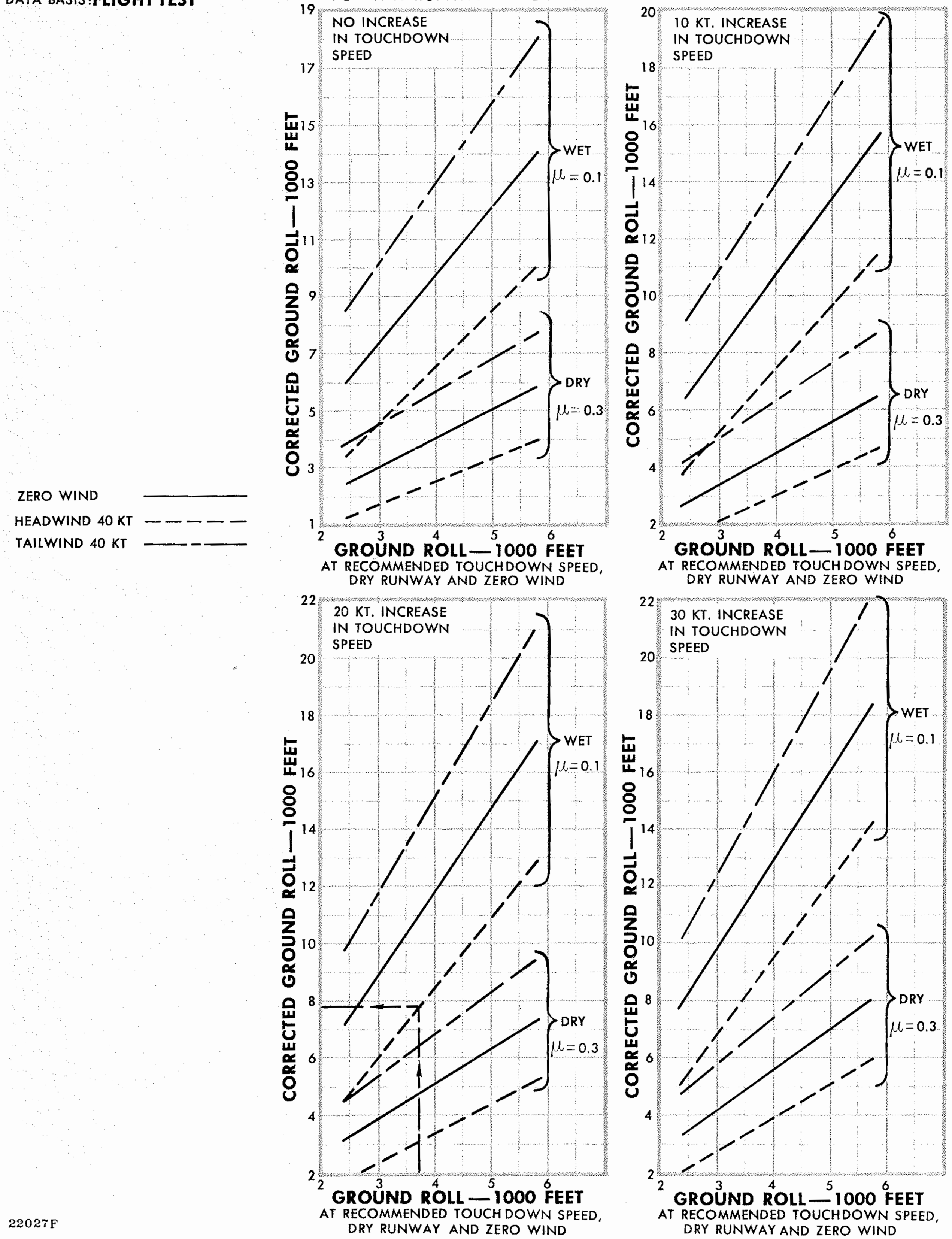


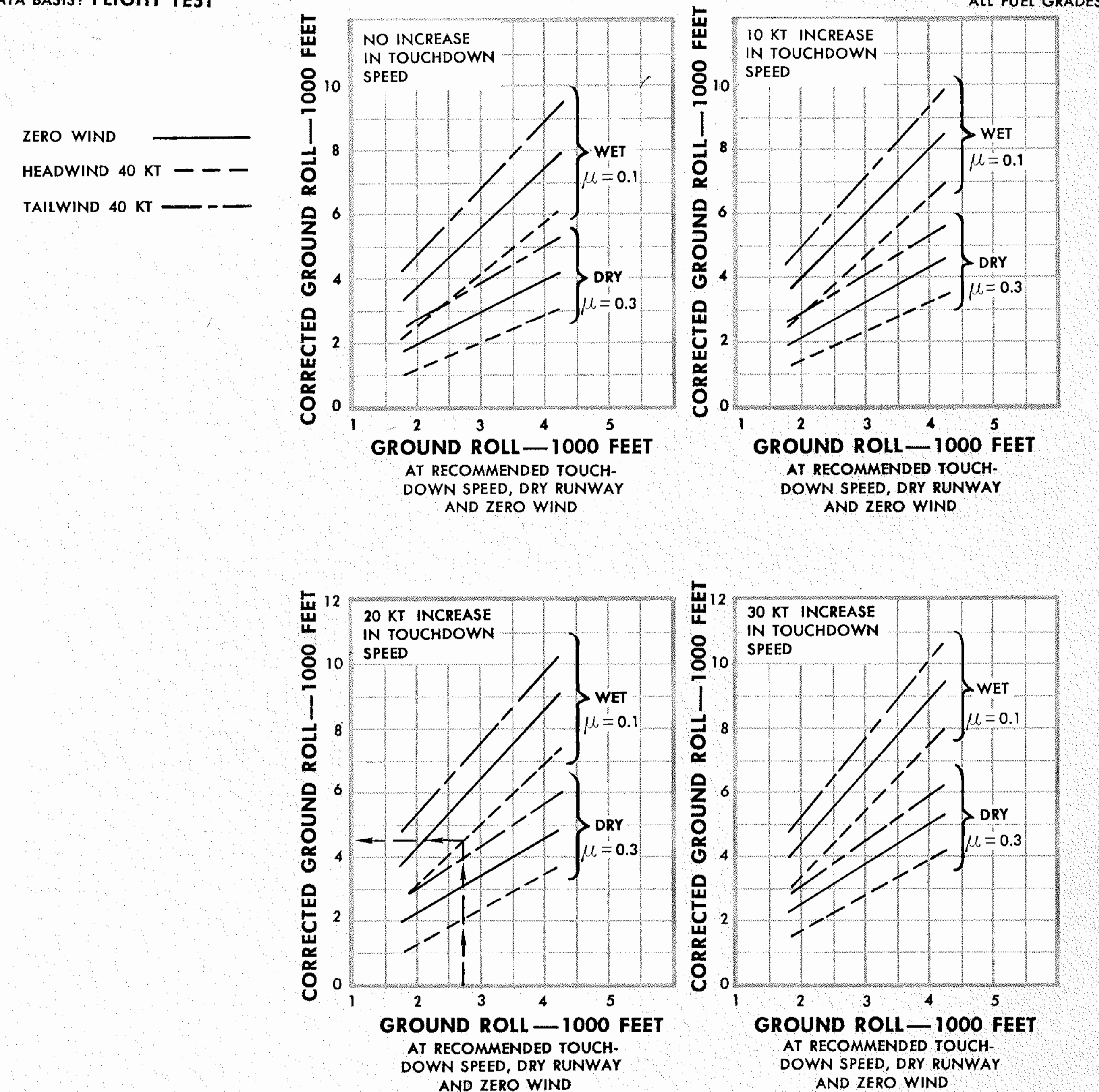
Figure A7-5

landing distances

SPEED BRAKES OPEN ● WITH DRAG CHUTE INFLATED AT TOUCHDOWN
GROUND ROLL CORRECTION FOR WET RUNWAY & INCREASED TOUCHDOWN SPEED
HARD SURFACE RUNWAY

MODEL: F-102A
DATE: 15 APRIL 1959
DATA BASIS: **FLIGHT TEST**

ENGINE: J57-23
ALL FUEL GRADES



22061A

Figure A7-6

PART 8 COMPUTER

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MB-8 COMPUTER

The MB-8 computer consists of three metal and two plastic discs, of which the three metal discs, supplied through regular Air Force channels, are a standard item, good for any airplane; however, they are useless without the plastic "data" discs. The plastic "data" discs contain the airplane performance. The MB-8 computer is designed to solve simple level-flight cruise control problems for jet airplanes. However, exclusive use for preflight planning is not recommended. The use of the Flight Manual results in far more comprehensive results. The greatest advantage of the computer lies in its simplicity of operation and convenient size; therefore, certain compromises are involved. The computer is designed for an average gross weight. This will result in a lower-than-indicated miles per pound of fuel with a subsequent higher rate of fuel flow at the beginning of flight and higher-than-indicated miles per pound with a subsequent lower rate of fuel flow during the final portion of the flight, giving an average miles per pound as indicated on the computer. The back or "tabulator" side (figure A8-1) of the MB-8 computer shows the cruise data in the "MAX RANGE" window, listing combinations of fuel remaining at selected pressure altitude. This data can be used as a quick range check. Range data for both optimum cruise and cruise at constant altitude is given, thereby providing a quick and yet fairly comprehensive picture of the range potential. This data is very similar to the information given in the optimum return profile of the Flight Manual. A second window displays the time, fuel, and distance required for climb or descent, while a third window frames the recommended altitude-speed schedule for these maneuvers. Notice that a black background is used for one configuration, while the other has a white background. Turn to the front or "working side" of the computer and begin at the center, working toward the outer edge. Keep in mind the six factors of range:

1. Speed
2. Altitude
3. Fuel
4. Distance
5. Time
6. Wind

Notice the opposing "pie-shaped" windows which allow the center plastic discs to show through. The black-bordered window is for the black background data. Notice that the window is divided into altitudes, with an index line through the center of the windows. The outer edge of this first (metal) disc is divided into a logarithmic scale labeled "Fuel Quantity — Pounds" (referred to as the fuel disc). With this type of scale, the "1000" mark can mean 1, 10, 100, 1000, or 10,000 pounds, etc., depending on the magnitude of the other factors in the range problem. Using the tab provided, rotate the fuel disc counterclockwise so that the index line for any selected altitude passes across the speed lines which show through the respective window. The first speed line encountered is the recommended speed for maximum range. Further counterclockwise rotation results in passing over increasing speeds until the maximum speed line in the series is reached. In progressing from speed for maximum range to maximum speed, the index passes over a speed line coded as a solid dot with a vertical line passing through it. This is the computer setting for maximum endurance. The speed for maximum endurance is quoted at the extreme right of the maximum range speed line. This coded point is used together with the quoted speed to obtain maximum endurance information. Another coded speed line (diamond with a vertical line) is the maximum speed for normal thrust (maximum continuous thrust). To help understand the position of these speed points on the computer, examine a typical nautical-miles-per-pound-of-fuel (specific range) curve which presents these same points graphically.

Note

The speeds shown on the MB-8 computer are CAS or true Mach number; therefore, any indicated speeds should be corrected for installation error before speeds are entered on the computer. Because of numerous variables and possible modification of the airspeed indicating system, the indicated airspeed and Mach number are not incorporated in the MB-8 computer.

The second disc (plastic) is a performance data disc around which is placed a logarithmic scale labeled "Air Nautical Miles." Refer to this disc as the "distance" disc. The placement of the speed lines previously described maintains the proper relationship between the distance and fuel discs. Note that any specific relationship between the fuel and distance discs for a selected speed and altitude will give the specific range or nautical air miles per pound of fuel; i.e., the air miles at the 1000-pound mark are actually nautical air miles per 1000 pounds of fuel. The tab on the distance disc is a special shape with the straight edge of the tab acting as a "wiper" or cursor on the third (metal) disc. Mention will be made of this cursor in the discussion of the "time" disc. The third disc of the front face of the computer is referred to as the "time" disc. On this disc is printed a series of concentric scales, of which the inner scale is labeled "Hours — Minutes." The succeeding scales are speed scales (Mach number or calibrated airspeed) for selected altitudes; i. e., altitudes corresponding to the altitudes listed on the fuel disc.

MB-8 computer

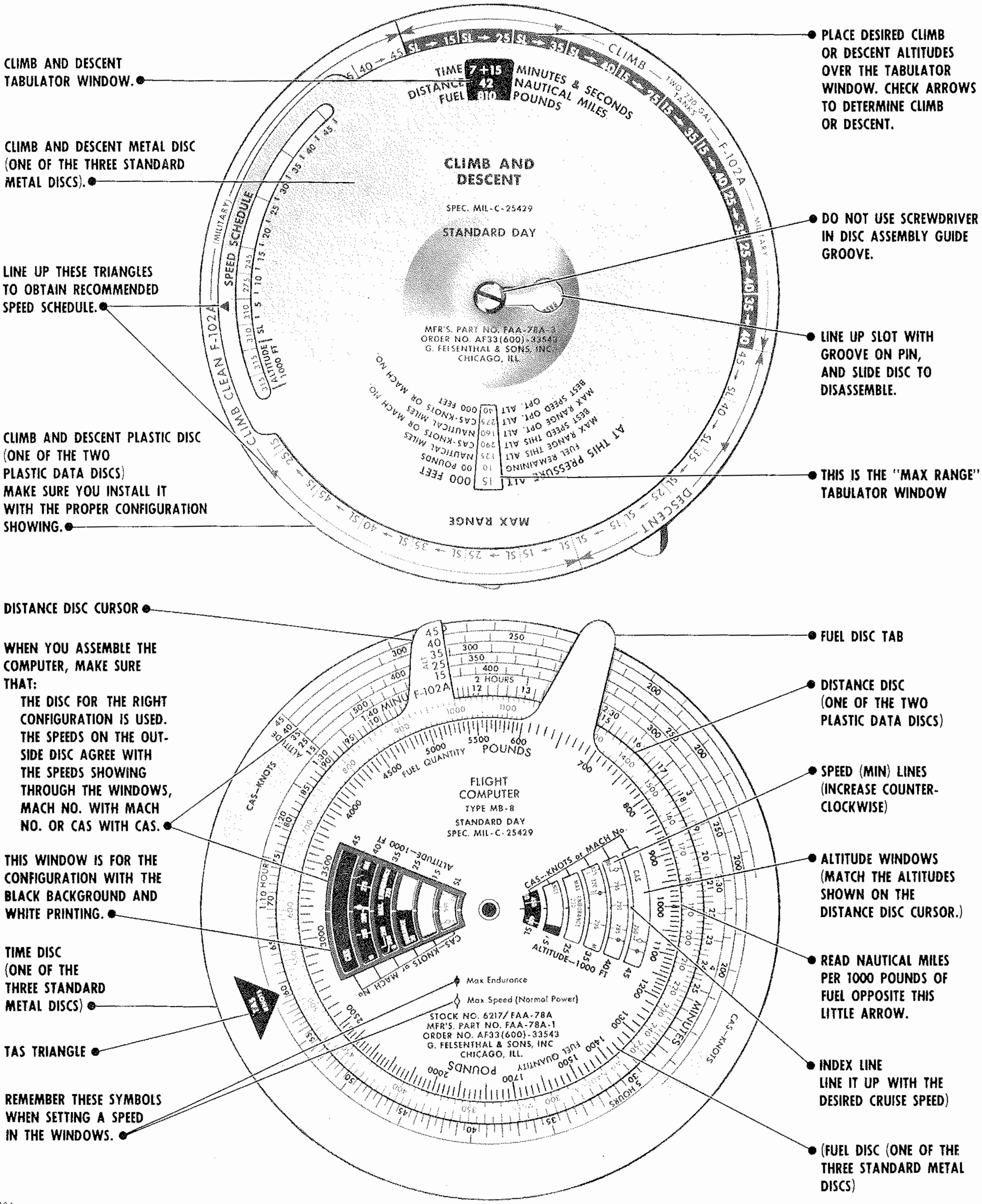


Figure A8-1

Note

Make sure when the computer is assembled that this disc displays speed in the same terms as the speed appearing in the window of the fuel disc; either Mach number or CAS.

Notice the large black triangle on this disc, labeled "TAS – Knots," with the apex at the 60-minute mark of the time scale. For standard atmosphere conditions, the true airspeed in knots may be obtained from the distance disc opposite this TAS triangle. That is, when the distance disc cursor is aligned with a speed on the time disc at some altitude, the corresponding true airspeed is indicated by the TAS triangle (using the distance scale as a speed scale). This conversion feature is useful in making corrections for wind.

Wind Corrections

The front face of the computer can be adjusted for wind in the following manner:

1. Do not change the relationship of the fuel disc and the time disc. Pinch the fuel disc tab against the outer edge of the time disc. This will still permit rotation of the distance disc.
2. Rotate the distance disc until the ground speed (TAS + wind) on the "Air Nautical Miles" scale of the distance disc is aligned with the TAS triangle. The "Air Nautical Miles" scale then becomes ground nautical miles, and the fuel required to travel any ground distance is obtained from the fuel disc, while the time is read from the time disc.

WARNING

The speeds now appearing in the window of the fuel disc and on the time disc under the index cursor no longer apply. The original speed must be maintained in flight.

3. To determine the winds while in flight, a system of check points can be utilized. Rotate the distance disc until the distance between check points is aligned with the elapsed time on the "Minutes" scale of the time disc. The ground speed is then read on the distance disc opposite the TAS triangle. The fuel required to travel any selected ground distance is obtained from the time disc. The speeds appearing in the window of the fuel disc no longer apply and should be ignored. The original speed must be maintained.

Fuel Flow Corrections

Variations in the fuel consumption characteristics due to battle damage, small changes in configurations, differences in engines, formation flight, etc., may be accounted for in the following manner:

1. Determine the fuel flow from the flowmeter.
2. Do not change the relationship of the distance disc and the time disc (set from ground speed).
3. Rotate the fuel disc until the rate of fuel flow, read from the flowmeter, is aligned with the TAS triangle.
4. Determine distance and time for selected fuel quantities from the respective discs.

Summary

Variations in rate of fuel flow of an average magnitude of +5% of that indicated on the computer can be expected on the initial portion of the flight and −5% on the final portion when flying at the speed recommended for maximum range. These variations show up plainly when the maximum range shown on the tabulator side of the computer is compared with the range obtained on the fuel-distance side. The tabulator side will indicate a greater distance because this data considers the change in airplane gross weight as fuel is consumed, whereas the indicated specific range on the fuel-distance side of the computer is an average value and results in a slightly conservative distance. The true airspeeds presented on the computer are based on standard atmospheric conditions. An allowance for the difference in this true airspeed and the true airspeed for the actual atmospheric condition can be made by the wind correction method described previously. The rate of fuel flow is also based on standard atmospheric conditions. However, the difference in fuel flow need not be corrected, since the air range calculated on the flight computer is normally independent of air temperature when the speed is properly indexed.

PART 9 MISSION PLANNING

TABLE OF CONTENTS

TAKEOFF PLANNING

The takeoff planning procedures presented in the takeoff speed, takeoff distance, and maximum refusal speed charts are based on runways marked at 1000-foot intervals. The Takeoff Planning Chart (figure A9-1) presents an explanation of the terms used in conjunction with these charts. Acceleration line check speed points are positions along the runway where normal acceleration may be checked. Refusal speed and refusal distance, assuming normal acceleration, are the speeds and distances which can be attained and still stop the airplane in the remaining runway length. All refusal speed and distance data contain reaction time of both the pilot and the applicable airplane systems. Stopping distance is the runway length required to bring the airplane to a safe stop using speed brakes, drag chute, and heavy braking.

MISSION PLANNING

The charts presented in Parts 1 through 8 provide the performance information necessary to plan many different types of missions. The following sample problems illustrate how the performance charts may be integrated to form a complete tactical mission flight plan:

Note

The following sample problems are an exercise in use of the performance charts. They are not intended ot reflect actual or propsed tactical missions employing this airplane.

SAMPLE PROBLEMS

RADIUS MISSION

Determine combat time available for a 250 nautical mile radius mission against a target at 35,000 feet altitude. Conditions are as follows:

a. Takeoff with full internal fuel load (30-knot headwind for takeoff).

b. Climb with military thrust to optimum cruise-climb altitude.

c. Follow cruise-climb procedure to target area.

d. Descend to target altitude for combat.

e. Combat with maximum thrust.

f. Climb with military thrust to optimum cruise-climb altitude.

g. Return to base following cruise-climb procedure.

h. Descend to sea level.

i. Land with reserves equivalent to 25 minutes endurance at sea level speeds for maximum endurance.

1. *Takeoff with full internal fuel load* (7053 pounds). Fully loaded gross weight is 28,150 pounds with full internal fuel load. The takeoff distance and speed as shown in figures A2-7 and A2-9 for sea level standard conditions with a 30-knot headwind are as follows:

   (a) Takeoff gross weight is 28,150 pounds.

   (b) Ground run distance is 1400 feet.

   (c) Total distance over 50-foot obstacle is 2800 feet.

   (d) Takeoff speed is 138 KIAS.

2. *Climb with military thrust to optimum cruise-climb altitude and follow cruise-climb procedure to target area.* Figure A4-4 is the mission profile chart for the clean configuration and combines military thrust climb with cruise characteristics. This chart includes the takeoff allowance, 1130 pounds. For zero wind conditions the total fuel required to attain 250 nautical miles is 2995 pounds. The time required is 31.0 minutes. With a 30-knot headwind the following corrections are applied.

   (a) Average true airspeed (TAS) (250 nautical miles $\div \frac{31.0}{60} = 484$ knots).

   (b) With a 30-knot headwind the true ground speed is 454 knots (484 knots −30 knots).

   (c) Time with wind effect is 0.551 hours or 33.1 minutes (250 nautical miles ÷ 454 knots).

   (d) Equivalent air distance traveled is 266 nautical miles (484 knots × 0.551 hours).

   (e) Actual fuel required is obtained by re-entering the mission profile chart with the equivalent air distance traveled. Fuel required (wind corrected) is 3100 pounds. Time required (wind corrected) is 33.1 minutes. The final gross weight is 25,050 pounds at an altitude of 43,700 feet. The climb schedule is noted on the chart for military thrust. The engine pressure ratio is noted as 2.40 along the optimum cruise-climb flight path at an indicated Mach number of 0.89.

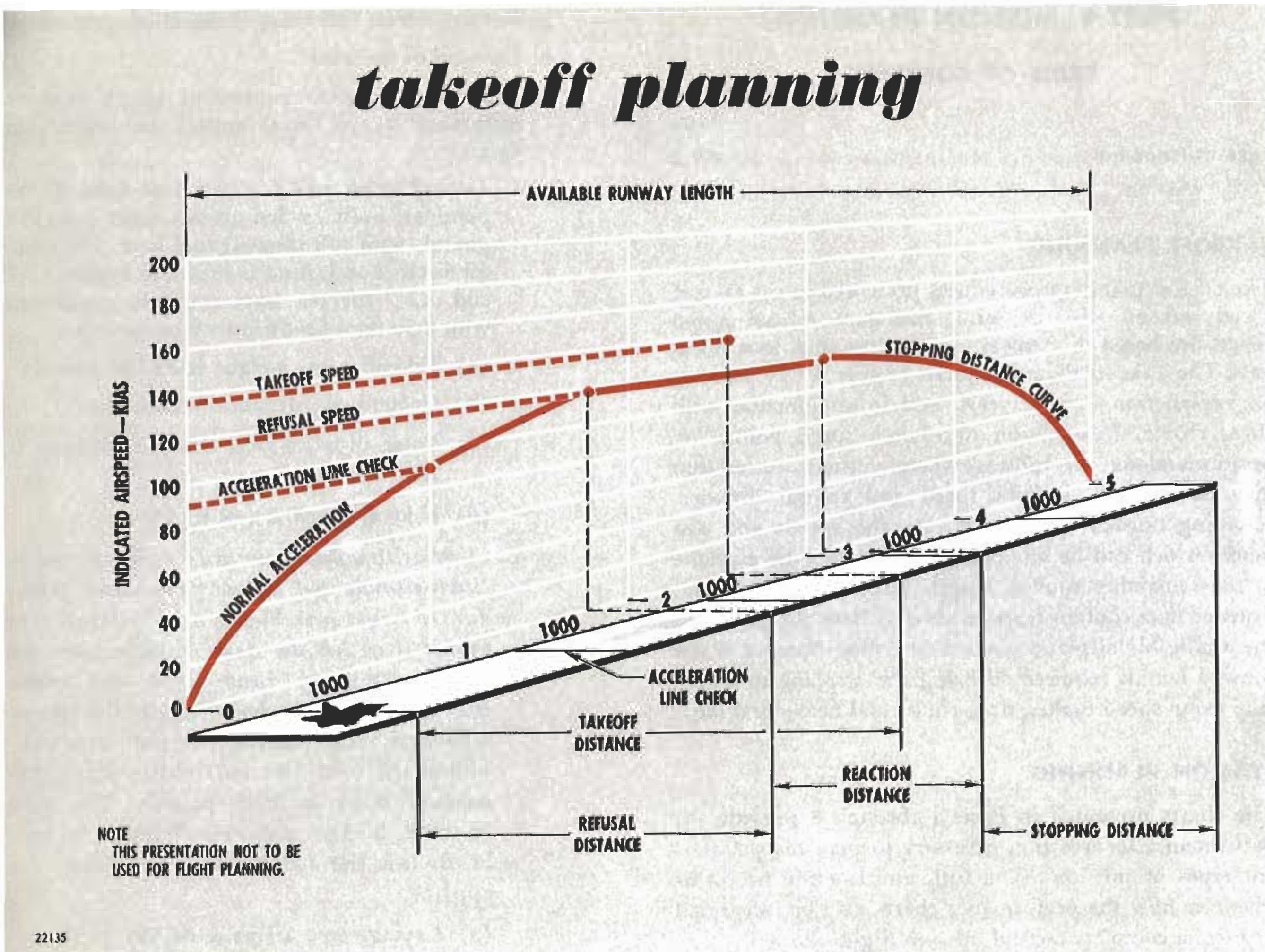


Figure A9-1

Note

At optimum cruise-climb conditions the thrust setting (engine pressure ratio) and indicated Mach number for optimum cruise is unaffected by wind conditions. At lower altitudes refer to figure A4-10 through A4-29 to obtain the effect of wind on true cruise Mach number and engine pressure ratio at optimum cruise conditions.

3. *Descend to target altitude for combat.* A spiral descent is assumed, hence no distance is considered in descending to the combat altitude, 35,000 feet. For the recommended descent, figure A6-2 shows approximately 11 pounds of fuel used in 1.0 minute with speed brakes open. Total time to reach the combat point is then 34-1 minutes (33.1 minutes + 1.0 minute). The gross weight at start of combat is then 25,039 pounds.
4. *Land with reserves equivalent to 25 minutes endurance at sea level speeds for maximum endurance.* Figure A5-2 shows 1030 pounds of fuel for the reserve requirements at sea level. When added to the empty weight of the airplane (21,097 pounds), this results in a landing weight of 22,127 pounds. The landing distances as shown in figure A7-4 for sea level standard conditions with a 30-knot headwind and with wheel brakes, speed brakes and drag chute are as follows:
   (a) Landing weight is 22,127 pounds.
   (b) Ground roll distance is 1500 feet.
   (c) Total distance over 50-foot obstacle is 3900 feet.
   (d) Approach speed is 168 KIAS.
   (e) Touchdown speed is 137 KIAS.

To illustrate the use of the landing distance chart for correcting ground roll for wet runway and increased touchdown speed (figure A7-6), assume that a wet runway condition exists upon return from the mission and touchdown is made at a speed 10 knots greater than recommended. It is first necessary to obtain the zero wind ground roll distance from the dry runway landing distance chart; in this case 2080 feet. Entering the ground roll correction chart (figure A7-6) with 2080 feet, the corrected ground roll is found to be 3300 feet.

5. *Climb with military thrust to optimum cruise-climb altitude and return to base following cruise-climb procedure.* The optimum return profile is shown on figure A4-8. This chart combines climb and cruise characteristics in the optimum form to minimize return fuel requirements. It is assumed that the combat is terminated at a distance of 300 nautical miles at the initial combat altitude, 35,000 feet. Figure A4-8 shows 1330 pounds of fuel required to return a distance of 300 nautical miles at zero wind conditions. This includes the climb to optimum cruise altitude. By following the climb guide lines to the line of best range for constant altitude it is noted the return cruise is accomplished at an altitude of 45,800 feet. The time required, no wind, is the sum of the climb and cruise times. The climb time, approximated from figure A3-2, is 8.8 minutes. This value is determined by using the landing weight plus reserve previously determined and adding the fuel required for the return climb and cruise segments. This results in a gross weight of 23,457 pounds (22,127 pounds + 1330 pounds) at 35,000 feet altitude at start of climb. The cruise time is determined by the remaining distance at 45,800 feet and the true airspeed for cruise at that altitude. The true airspeed is found by interpolation in the cruise table. Cruise time (no wind) is 27.2 minutes (240 nautical miles × 60 ÷ 530 knots). The total time required, no wind, is then 36.0 minutes (8.8 minutes + 27.2 minutes). The effect of a 30-knot tailwind results in the following corrections:
   (a) Average true airspeed (TAS) is 500 knots (300 nautical miles $\div \frac{36.0}{60}$ hours).
   (b) With a 30-knot tailwind the true ground speed is 530 knots (500 knots + 30 knots).
   (c) Time with wind effect is 0.565 hours or 33.9 minutes (300 nautical miles ÷ 530 knots).
   (d) Equivalent air distance traveled is 283 nautical miles (540 knots × 0.565 hours).
   (e) Actual fuel required is found by re-entering the profile chart with the equivalent air distance traveled. Fuel required (wind corrected) is 1270 pounds. Time required (wind corrected) is 33.9 minutes.
6. *Descend to sea level.* For the recommended descent, figure A6-2 indicates approximately 40 pounds of fuel used in 3.0 minutes with speed brakes open. The total time to return and descend to sea level is then 36.9 minutes (33.9 minutes + 3.0 minutes).
7. *Combat with maximum thrust.* To determine the combat time available it is necessary to compute the total fuel requirements thus far and determine the fuel available for the combat phase of the mission. Fuel required for return from the point where combat is terminated is the sum of the return climb and cruise segments, the descent from cruise altitude to sea level and the reserve allowance. This amount is 2340 pounds of fuel (1270 pounds + 40 pounds + 1030 pounds). Added to the empty weight of the airplane (21,097 pounds) results in a gross weight at the end of the combat phase of 23,437 pounds. The gross weight at the start of combat was found previously to be 25,039 pounds. The difference of these two gross weights – 1602 pounds is the fuel available for combat. Figure A5-6 shows a combat time available of 5.7 minutes for 1602 pounds of fuel at 35,000 feet.

Figure A9-2 is a graphic presentation of the gross weight vs distance for the radius mission.

LOCAL INTERCEPT TYPE MISSION

Determine loiter time available for the following combat mission:

a. Takeoff with full load including external tanks.
b. Climb to combat ceiling with maximum thrust (tanks dropped in climb when empty).
c. Combat for 5 minutes at combat ceiling.
d. Descend to altitude for maximum endurance.
e. Loiter at optimum altitude.
f. Land with reserve fuel equal to 20 minutes at sea level speeds for maximum endurance.
   1. *Takeoff with full load including external tanks.* Fully loaded gross weight is 31,276 pounds including external tanks. The total fuel load is 1515 gallons or 9848 pounds at 6.5 pounds/gallon. Of this total, 430 gallons or 2795 pounds, is usable external tank fuel. The takeoff distance and speed, (figures A2-7 and A2-9) for sea level standard zero wind conditions are as follows:
      (a) Takeoff gross weight is 31,276 pounds.
      (b) Ground run distance is 2750 feet.
      (c) Total distance over 50-foot obstacle is 4500 feet.
      (d) Takeoff speed is 146 KIAS.

As noted on the mission profile charts the takeoff fuel allowance is 1130 pounds. The fuel required for warmup, taxi takeoff and accelerate-to-climb speed is considered to be within this allowance.

   2. *Climb to combat ceiling (rate-of-climb = 500 feet/minute) with maximum thrust.*
      (a) Climb — Maximum thrust with external tanks – figure A3-5.

radius mission

GROSS WEIGHT VS. DISTANCE

MODEL: F-102A
CLEAN CONFIGURATION

ENGINE: J57-23
STANDARD DAY

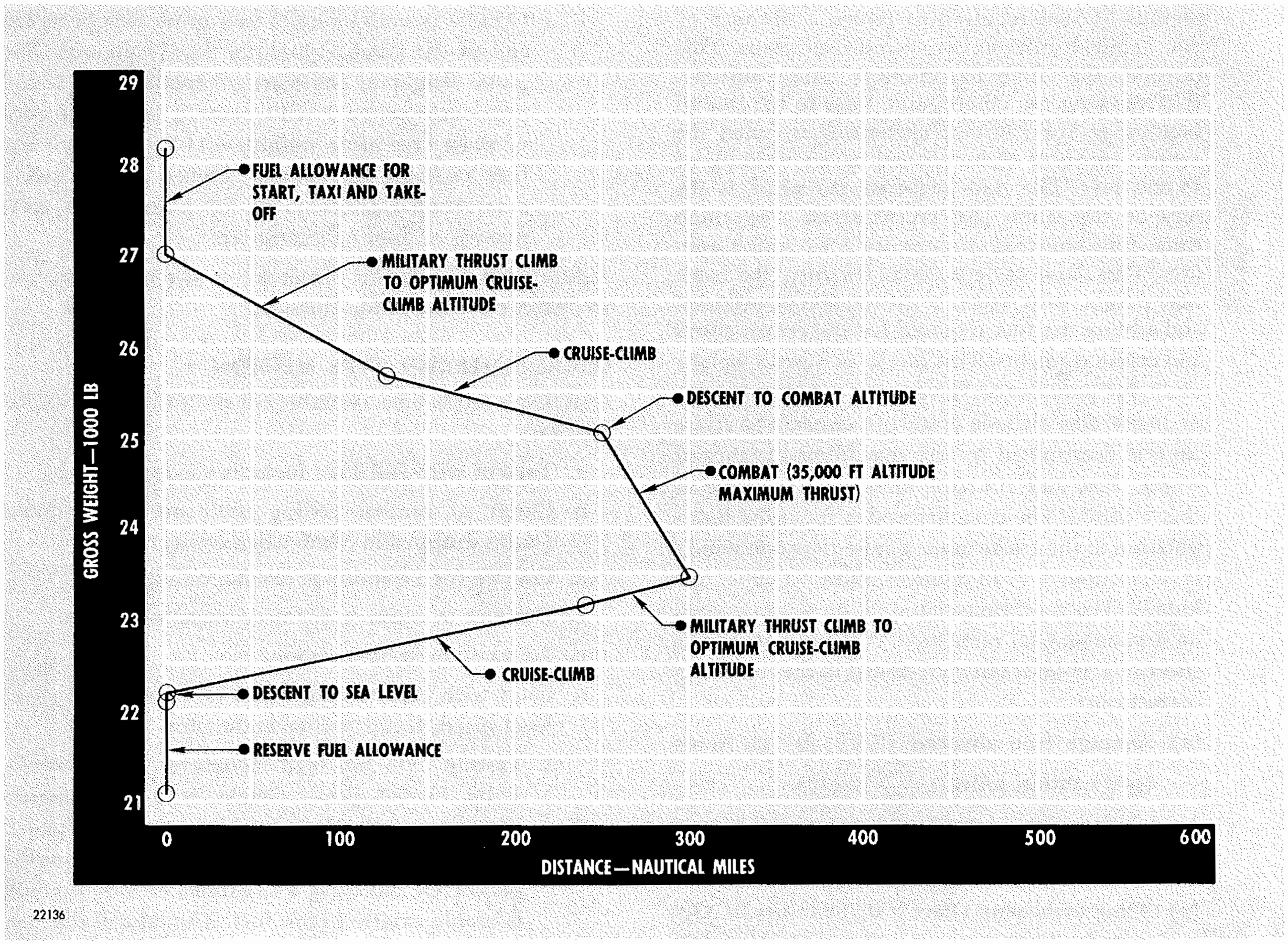


Figure A9-2

(1) Gross weight at start of climb is 30,146 pounds (31,276 pounds—1130 pounds).

(2) Enter figure A3-5 with 30,146 pounds at sea level. Follow parallel to guide lines to combat ceiling.

Note

Total fuel required to attain combat ceiling is greater than external tank capacity. Therefore, drop external tanks when empty and continue climbing with clean airplane configuration. Fuel remaining in external tanks at the start of climb is 1665 pounds (2795 pounds − 1130 pounds). Therefore, 28,481 pounds is the gross weight at the end of climb with external tanks (30,146 pounds − 1665 pounds = 28,481 pounds).

(3) With an end weight of 28,481 pounds an altitude of 38,000 feet is attained. The distance traveled is 36 nautical miles and the time for climb is 4.3 minutes.

(b) Climb-maximum thrust clean configuration – figure A3-4.

(1) Gross weight after dropping external tanks is 28,150 pounds (28,481 pounds −331 pounds).

(2) Enter figure A3-4 at initial conditions of 28,150 pounds gross weight at an altitude of 38,000 feet. The distance index for this condition is 28.0 nautical miles and the time index is 3.0 minutes.

(3) Proceed parallel to guide lines to the final condition, combat ceiling. At this point the distance index is 75.0 nautical miles and the time index is 8.4 minutes.

(4) Therefore, distance and time for this climb segment is established by the differences between respective index numbers:

(a) Climb distance segment is 47 nautical miles (75.0 nautica l miles − 28.0).

(b) Time in climb for this segment is 5.4 minutes (8.4 minutes − 3.0 minutes).

(5) The fuel used in climb is the difference between the initial weight and the final weight as determined by following parallel to the guide lines up to combat ceiling. 28,150 pounds initial weight − 27,350 pounds final weight gives 800 pounds of fuel used.

(6) Interpolation between lines of constant altitude at the combat ceiling point gives an altitude of 50,100 feet.

The total time to climb is the summation of the initial climb with tanks plus the remaining climb segment with the clean airplane (4.3 minutes + 5.4 minutes) and is 9.7 minutes — similarly, a summation of the distance covered in the two climb segments (36 nautical miles + 47 nautical miles) results in a total distance in climb of 83 nautical miles. The total fuel used (including takeoff allowance) is 2795 pounds from tanks plus 800 pounds of internal fuel for a total of 3595 pounds. The fuel remaining is 6253 pounds at a gross weight of 27,350 pounds.

Note

Adhere to climb schedules applicable to airplane configurations. Schedules are tabulated on all pertinent charts.

3. *Combat for 5 minutes at combat ceiling.* Combat fuel is determined from figure A5-6. At an altitude of 50,100 feet 600 pounds of fuel is required at maximum power. Gross weight at the end of combat is 26,750 pounds (27,350 pounds − 600 pounds).

4. *Descend to altitude for maximum endurance.* Reduce engine rpm to 85% and descend at the speed schedule for the recommended descent, figure A6-3. As noted on figure A5-4, the approximate altitude for maximum endurance is 35,000 feet. The time and fuel used for this descent is determined by the differences between initial and final altitude. The initial fuel index is 100 pounds and the final index at 35,000 feet altitude is 60 pounds. Fuel used is then 40 pounds and the final gross weight is 27,310 pounds (27,350 − 40 pounds). The initial time index is 8.0 minutes and the final index is approximately 4.0 minutes which results in a total descent time of 4.0 minutes.

Note

The use of figures A6-2 and A6-3 for weights greater than 22,000 pounds results in conservative fuel requirements.

5. *Land with reserve fuel equal to 20 minutes at sea level speeds for maximum endurance — reference figure* A4-10. The empty gross weight is 21,097 pounds for the clean airplane. As a first approximation enter figure A4-10 with the empty weight along the line for maximum endurance. Interpolate for fuel flow and determine the fuel quantity required for 20 minutes endurance (2430/60 × 20 = 810 pounds). Add one-half of this quantity to the empty weight and re-enter the chart at the new gross weight, 21,502 pounds. This then establishes a second approximation for the average endurance fuel flow, 2480 pounds/hour. The fuel required is then 2480/60 × 20 = 825 pounds. The landing weight, including reserve fuel is then 21,097 + 825 = 21,922 pounds. The landing distances, figure A7-4, for sea level standard zero wind conditions are as follows:

(a) Landing weight is 21,922 pounds.

(b) Ground roll distance is 2100 feet.

(c) Total distance over 50-foot obstacle is 5100 feet.

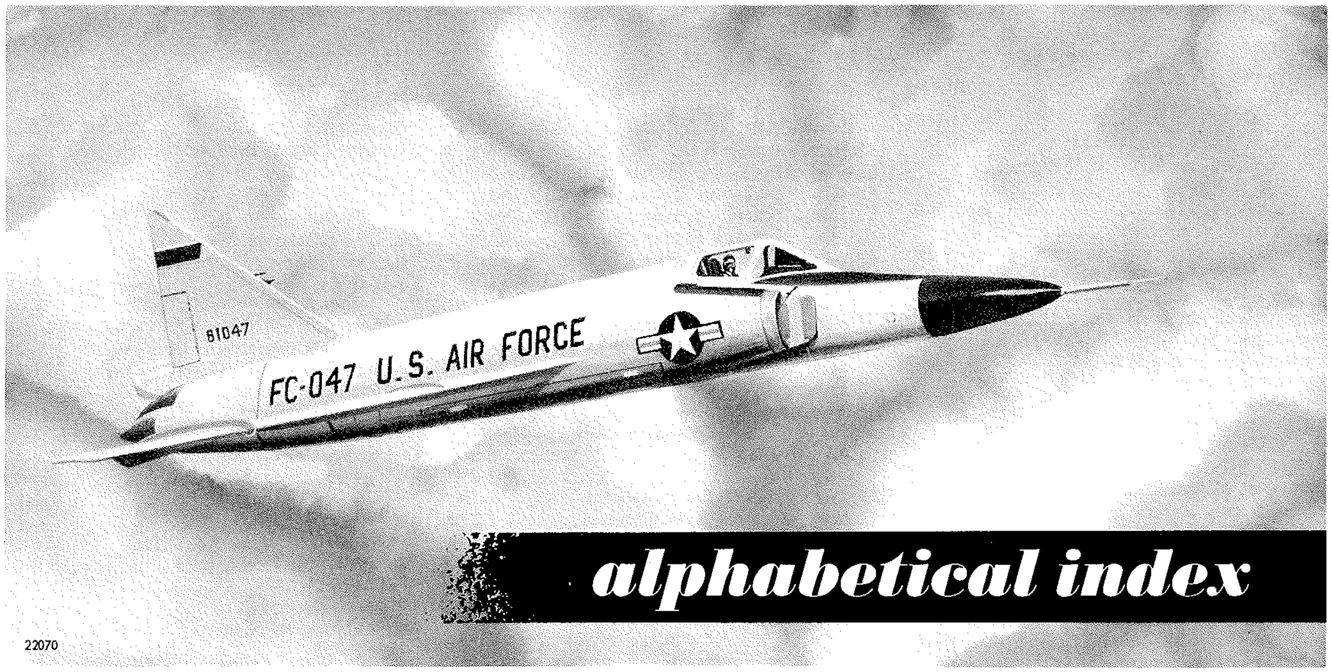
(d) Approach speed is 168 KIAS.

(e) Touchdown speed is 137 KIAS.

6. *Loiter at optimum altitude.* The fuel available for loiter is determined by the difference between landing weight, 21,922 pounds, and the weight after descent to loiter altitude 27,310 pounds. The fuel quantity is 5388 pounds, at an altitude of 35,000 feet. Figure A5-4 shows a loiter time of 2 hours 37 minutes including a descent time of 16 minutes from loiter altitude to sea level.

Note

Although this is not considered a radius mission a potential radius of 83 nautical miles is possible, (83 nautical miles total climb distance compared to a total of 104 nautical miles possible during descent, excluding distance made good during loiter phase.)



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