MHR(C) 1173-56

[redacted]

UTILITY FLIGHT HANDBOOK

USAF SERIES
YF-105A
AIRCRAFT

NOTICE: This document contains information affecting the national defense of the United States within the meaning of the Espionage Laws, Title 18 U. S. C., Section 793 and 794. Its transmission or the revelation of its contents in any manner to an unauthorized person is prohibited by law.

PUBLISHED UNDER AUTHORITY OF THE SECRETARY OF THE AIR FORCE

Commanders are responsible for bringing this publication to the attention of all Air Force personnel cleared for operation of subject Aircraft.

2016 AF BAFB ALA 5/14/56 500

[redacted]

30 JANUARY 1956

56MOE-5607

LIST OF REVISED PAGES ISSUED

INSERT LATEST REVISED PAGES. DESTROY SUPERSEDED PAGES.

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

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

ADDITIONAL COPIES OF THIS PUBLICATION MAY BE OBTAINED AS FOLLOWS: USAF

USAF ACTIVITIES.—In accordance with Technical Order No. 00-5-2.

NAVY ACTIVITIES.—Submit request to nearest supply point listed below, using form NavAer-140: NASD, Philadelphia, Pa.; NAS, Alameda, Calif.; NAS, Jacksonville, Fla.; NAS, Norfolk, Va.; NAS, San Diego, Calif.; NAS, Seattle, Wash.; ASD, NSC, Guam.

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

TABLE OF CONTENTS

IMPORTANT . . . . .

In order that you will gain the maximum benefits from this handbook it is imperative that you read this page and the following page carefully.

SCOPE

This handbook contains all the information necessary for safe and efficient operation of the YF-105F. These instructions do not teach basic flight principles, but provide you with a general knowledge of the airplane, its flight characteristics, and specific normal and emergency operating procedures. Your flying experience is recognized, and elementary instructions have been avoided.

SOUND JUDGMENT

The instructions in this handbook are designed to provide for the needs of a crew 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 Handbook 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 Handbook. Clearance must be obtained from ARDC before any questionable operation is attempted which is not specifically covered in the Flight Handbook.

STANDARDIZATION

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

SECTION I

DESCRIPTION

TABLE OF CONTENTS

AIRPLANE

The YF-105A airplane was designed and built by Republic Aviation Corporation as a single place, swept wing and empennage fighter bomber. The airplane is capable of flight in the subsonic and sonic speed ranges at high altitude and is maneuverable in all speed ranges and at all altitudes. Power is supplied by a continuous flow axial type gas turbine engine equipped with an afterburner. The landing gear is a conventional tricycle type with steerable nose wheel. Irreversible primary flight controls are hydraulically actuated and provided with artificial feel systems to stimulate desired aerodynamic feel. Leading and trailing edge flaps are provided as high lift devices and a four sectional speed brake, installed on the aft end of the fuselage, and a drag chute are provided as drag increasing devices. Exceptionally large loads may be carried for the completion of various missions. Stores are carried in an enclosed bomb bay in the fuselage. This enables the airplane to accomplish a mission with the stores being undetected also keeping the airplane aerodynamically clean.

AIRPLANE DIMENSIONS

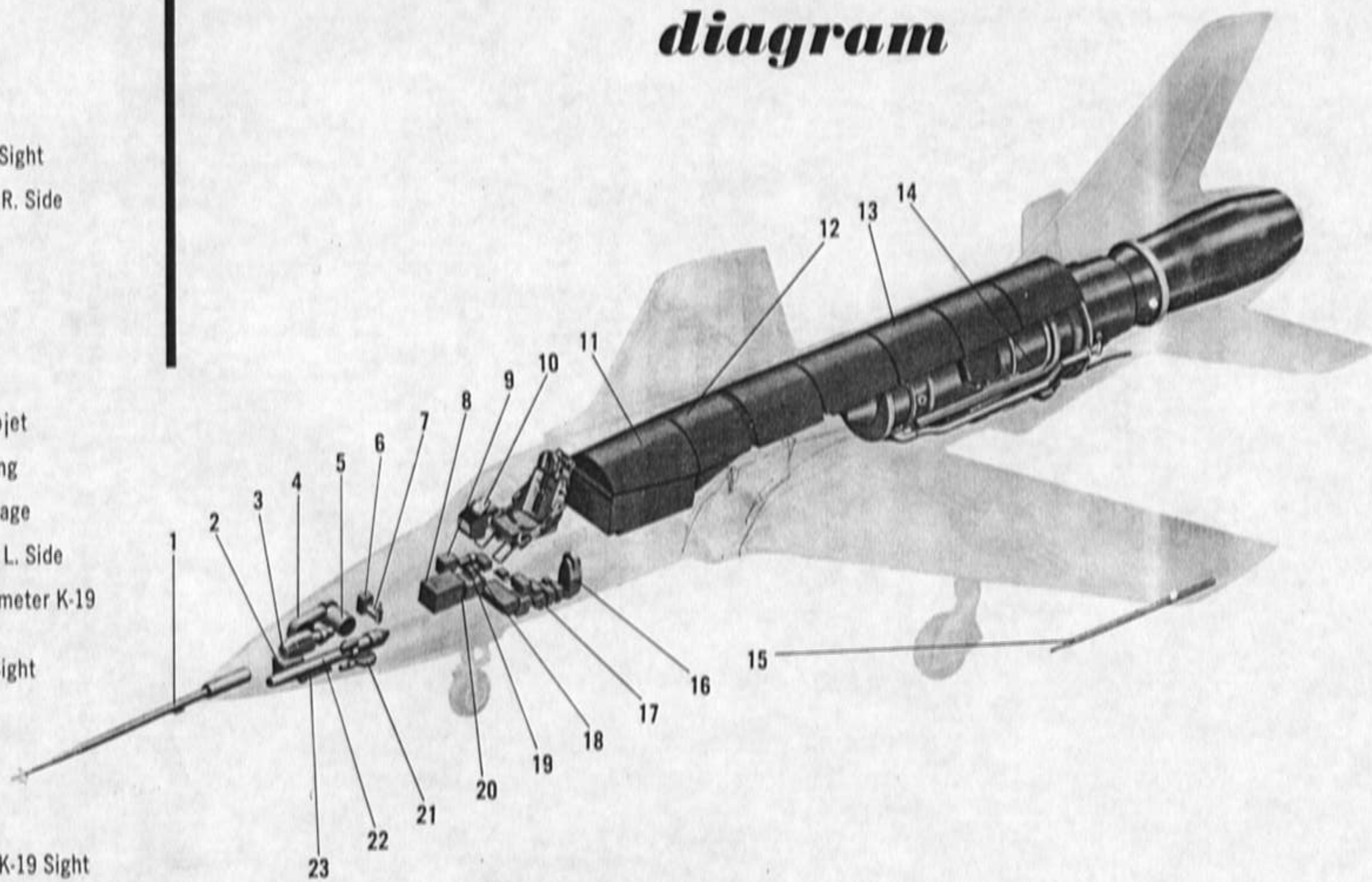
Length (including stabilizers)	61.03 ft.
Wing span	34 ft. 11.2 in.
Height (to top of fin-static)	17 ft. 6.38 in.
Tread	17 ft. 3.2 in.
Wheel Base	19 ft. 5.18 in.

AIRPLANE GROSS WEIGHT

Weight empty	20454 lbs.
Maximum take-off weight	41500 lbs.

GENERAL ARRANGEMENT diagram

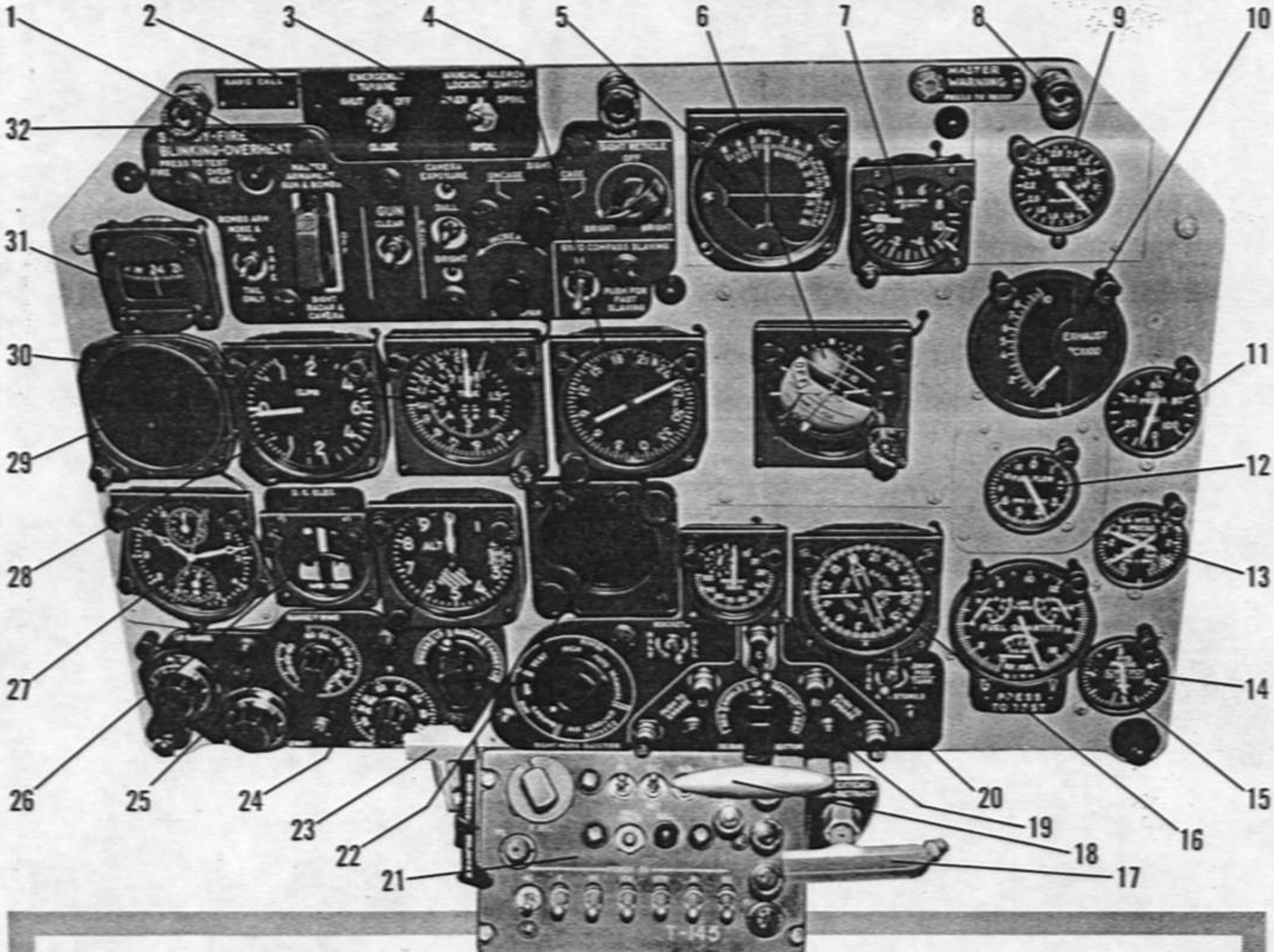
1. Test Boom
2. Battery
3. Alternator & Utility Hydraulic Pump
4. Airturbine
5. Hydraulic Reservoir
6. Inverter
7. Dynamotor
8. Bomb Computer K-19 Sight
9. Electronic Equipment R. Side
10. K-19 Sight Head
11. Fwd. Fuel Tank
12. Main Fuel Tank
13. Aft Fuel Tanks
14. P & W J-57-P-25 Turbojet
15. Pitot Tube — Left Wing
16. Air Conditioning Package
17. Electronic Equipment L. Side
18. Rate Gyro & Accelerometer K-19 Sight
19. Accelerometer K-19 Sight



20. Sensitivity Amplifier K-19 Sight
21. Antenna, AS-578/ARA-25
22. Gun T-171 Model "D"
23. Emergency Hydraulic Accumulator

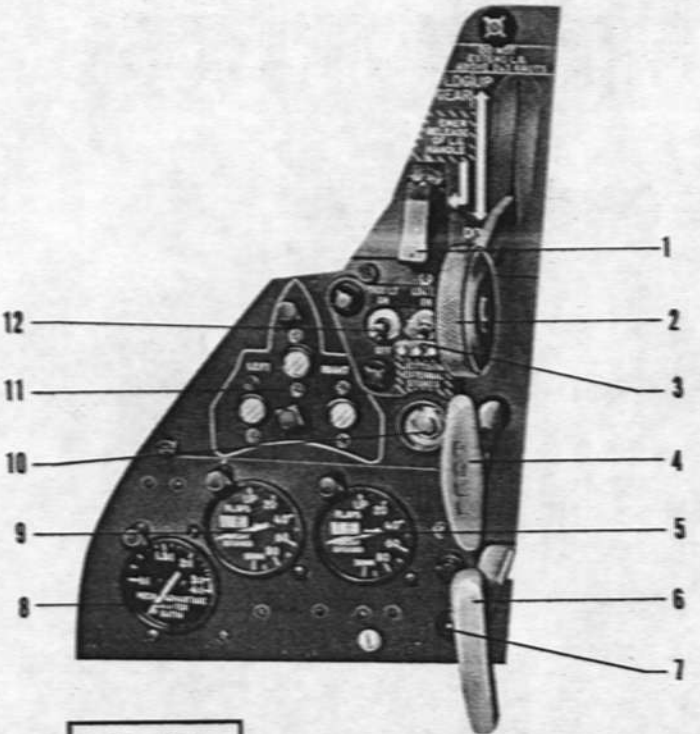
Figure 1—2

INSTRUMENT PANEL



1. Master Armament Panel
2. Emergency Turbine Control Switch
3. Manual Aileron Lock-out Switch
4. Slaved Gyro Magnetic Compass Indicator
5. Fire Control Roll Indicator
6. Attitude Indicator
7. Accelerometer
8. Master Warning Light
9. Pressure Ratio Gage
10. Exhaust Gas Temperature Indicator
11. Engine Oil Pressure Gage
12. Fuel Flowmeter
13. Hydraulic Pressure Gage—Controls
14. Hydraulic Pressure Gage—Utility
15. Fuel Quantity Gage
16. Radio Compass Indicator
17. Refueling Probe Control
18. Belly Tank Emergency Release
19. Fire Control Selector Panel
20. Tachometer
21. Special Store Control Panel
22. Course Indicator
23. Internal Store Release Control
24. Altimeter
25. Fire Control Setting Panel
26. Turn and Bank Indicator
27. Clock
28. Rate of Climb Indicator
29. Flight Command Indicator
30. True Airspeed and Machmeter
31. Standby Magnetic Compass
32. Engine Overheat and Fire Warning Light

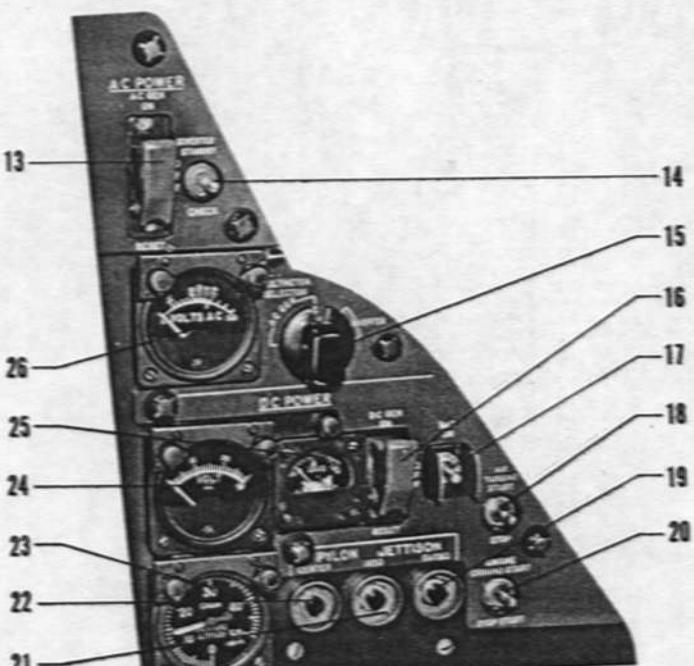
Figure 1–3



left

SIDE PANELS

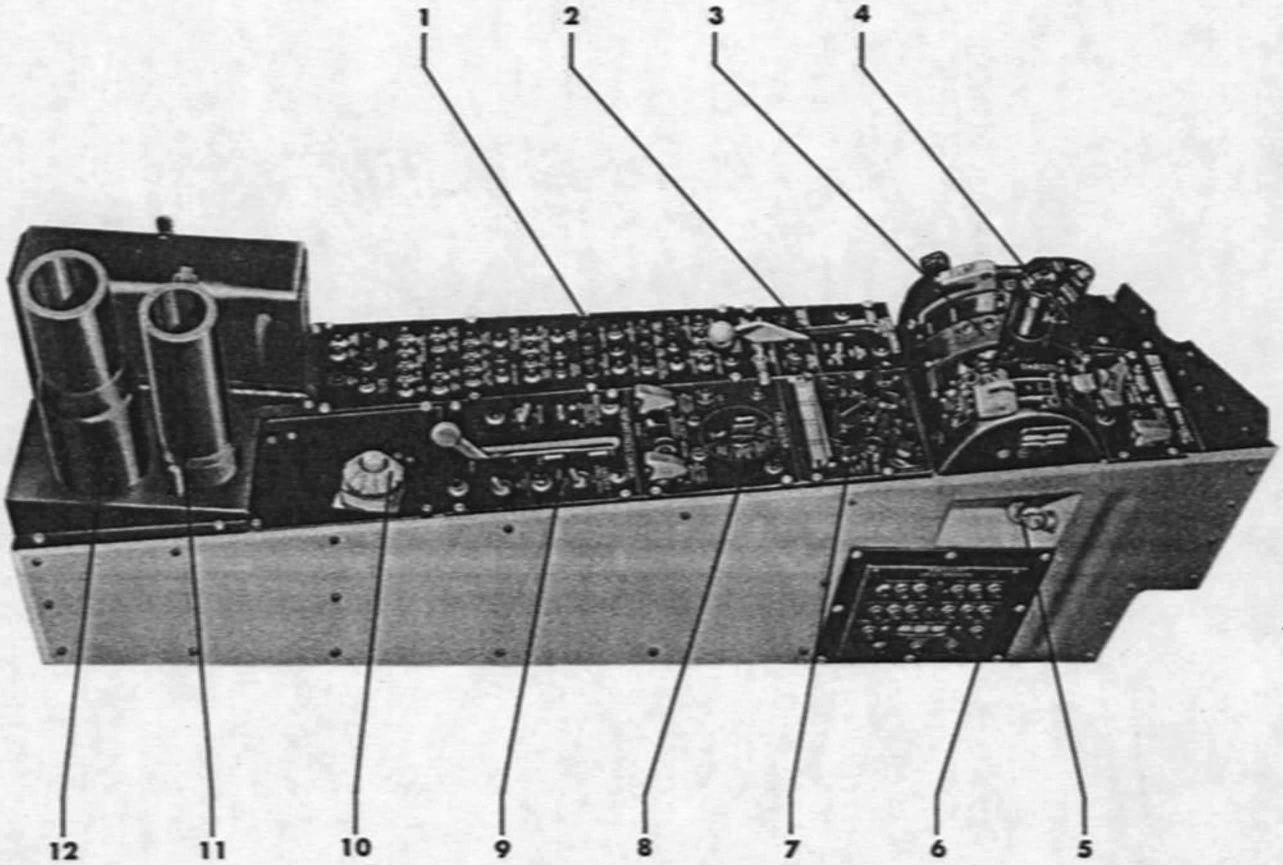
right



1 Landing Gear Emergency Release Switch
2 Landing Gear Selector Handle
3 Landing Light Switch
4 Landing Gear Emergency Extension Control
5 Trailing Edge Flap Indicator
6 Emergency Brake Control
7 Landing Gear Light and Horn Test Switch
8 Stabilator Mechanical Advantage Indicator
9 Leading Edge Flap Indicator
10 Jettison External Stores Switch
11 Landing Gear Position Indicators
12 Taxi Light Switch
13 A-C Generator Switch
14 A-C Inverter Switch
15 Voltmeter Selector Switch
16 D-C Generator Switch
17 Battery Switch
18 Air Turbine Switch
19 Outboard Pylon Jettison Switch
20 Engine Ground Start Switch
21 Inboard Pylon Jettison Switch
22 Center Pylon Jettison Switch
23 Cabin Pressure Indicator
24 D-C Voltmeter
25 Loadmeter
26 A-C Voltmeter

Figure 1–4

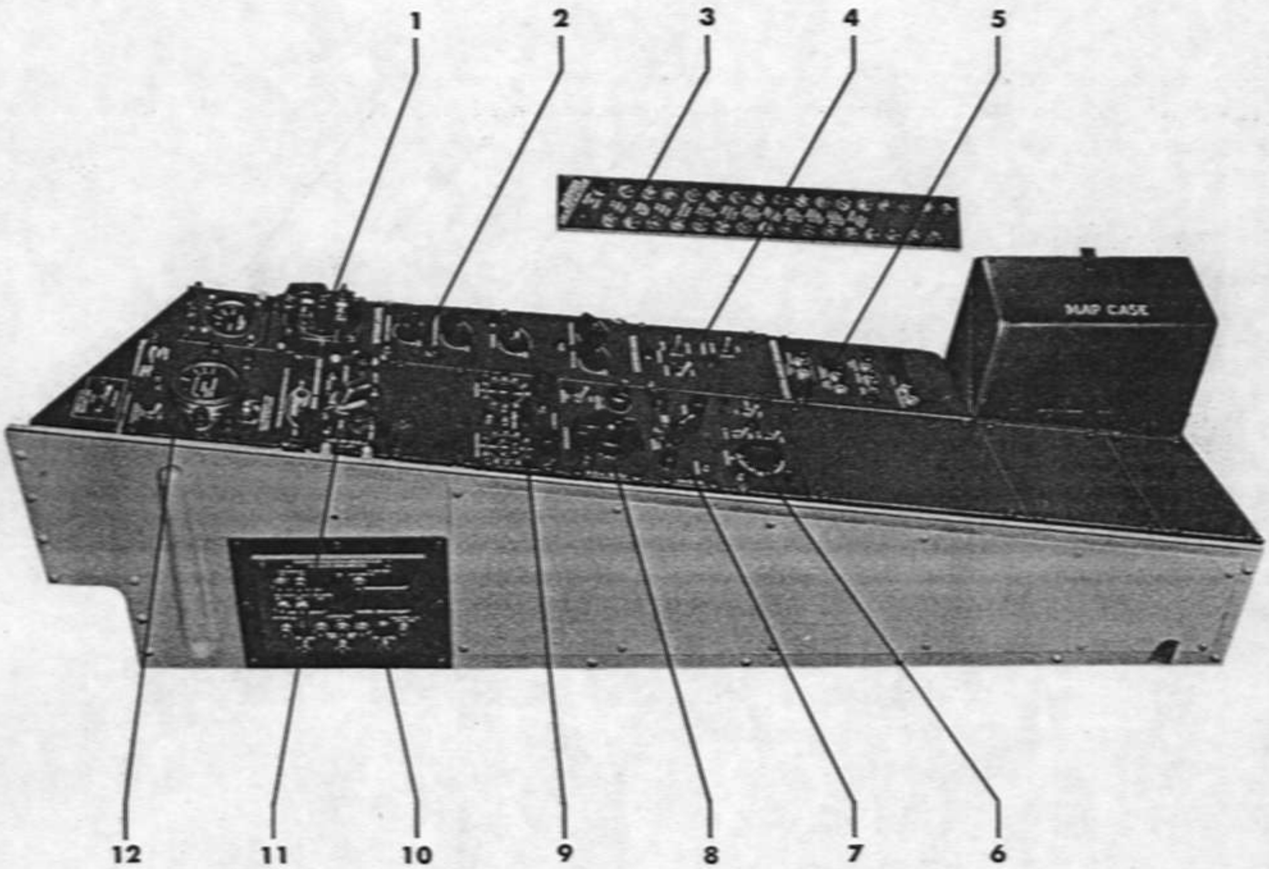
LEFT HAND CONSOLE



1. Circuit Breaker Panel (See figure 1–14)
2. Flight Control Panel (See figure 1–18)
3. Throttle Quadrant (See figure 1–8)
4. Drag Chute Control Panel (See figure 1–19)
5. Canopy Jettison Control
6. Circuit Breaker Panel (See figure 1–14)
7. Command Radio Control Panel (See figure 4–4)
8. Fuel System Control Panel (See figure 1–11)
9. Temperature and Pressure Control Panel (See figure 4–3)
10. Anti-g Valve
11. Pilot's Relief Container
12. Liquid Container

Figure 1–5

RIGHT HAND CONSOLE



1. Ground Position Indicator
2. Interior Lights Control Panel (See figure 4—9)
3. Warning Lights Panel (See figure 1—12)
4. Exterior Light Control Panel (See figure 4—10)
5. Circuit Breaker Panel (See figure 1—14)
6. Radar Control Panel—IFF (See figure 4—5)
7. Radar Control Panel—APW-11 (See figure 4—6)
8. Radio Compass Control Panel (See figure 4—7)
9. Ground Position Indicator Control Panel (See figure 4—8)
10. Circuit Breaker Panel (See figure 1—15)
11. Auto Pilot Control Panel (See figure 4—14)
12. Oxygen Regulator (See figure 4—11)

Figure 1—6

ENGINE

The airplane is powered by a continuous-flow axial type gas turbine engine USAF Model J57-P-25 equipped with an afterburner. Rated sea level static thrust of the engine is 10,200 pounds without afterburner and 16,000 pounds with afterburner. The engine has a multistage, two unit axial-type compressor, an eight-unit semi-annular combustion chamber and a three stage turbine. The compressor section consists of a nine stage low pressure unit and a seven stage high pressure unit. The rotor assembly of each unit is mechanically independent of the other. The high pressure compressor rotor is connected to, and driven by, the first stage turbine wheel by means of a hollow shaft. A through shaft, which passes through the hollow shaft, independently joins the low pressure compressor rotor to the combined second and third stage turbine wheels. A compressor air bleed system is utilized to direct part of the low pressure compressor air overboard at low engine rpm to provide surge free, fast engine accelerations. The compressor air bleed system is actuated automatically and is controlled by a governor driven by the low pressure compressor rotor. A thrust decay system, consisting of doors at the aft end of the afterburner, is provided to reduce thrust during engine idle operation. During idle operation, the engine produces too much thrust for landing, therefore, the thrust decay system doors are automatically opened to reduce the thrust to a satisfactory level.

ENGINE FUEL SYSTEM

Fuel flow to the engine is supplied from the airplane fuel system, is controlled by the throttle and regulated by the engine fuel control system (figure 1–7). This system includes the engine driven fuel pump unit, the hydromechanical fuel controller and the afterburner system.

FUEL PUMP UNIT

The engine driven fuel pump unit (figure 1–7) consists of three individual pumps, i.e., a common centrifugal pump, a spur gear type engine fuel pump and a spur gear type afterburner fuel pump. The unit also includes an afterburner shut-off valve and a fuel transfer valve. The pumps are mechanically driven by the engine high pressure rotor and individual shear sections are provided in the drive shafts of each pump so that a failure of any pump need not cause failure of the other two. All fuel from the tanks passes through the centrifugal pump which provides the required fuel pressure boost if the fuel tank booster pumps fail. The engine pump maintains fuel pressure for the engine fuel control and the afterburner system and also acts as a standby emergency pump. When the afterburner is not in use the afterburner shut-off valve is closed routing the total output of the afterburner pump back to the centrifugal pump discharge. A warning light illuminates if the gear type engine pump fails in which case the transfer valve automatically ports afterburner pump output to the engine fuel control to sustain engine operation. If the afterburner pump fails, the engine pump cannot supply fuel for afterburner operation.

ENGINE FUEL CONTROL

An engine driven, hydromechanical fuel control, incorporating both the main and emergency fuel control systems, regulates fuel flow to the engine combustion chambers. For any given throttle setting the fuel flow is automatically compensated for variations in flight conditions. The main system adjusts fuel flow for altitude changes, schedules fuel flow to protect the engine from overspeed and overtemperature conditions during rapid engine accelerations and also prevents compressor surge or stall. During rapid decelerations of the engine, the main system maintains a minimum fuel flow to prevent engine flame-out. The emergency fuel control system, which provides regulation of engine fuel flow, if the main system fails, must be selected by the pilot. There are no provisions for automatic transfer to the emergency fuel control system in the event of failure of the main fuel control system. When operating on the emergency system, indicated by illumination of an indicator light, the main system is inoperative and fuel flow is metered by a throttle valve in the fuel control. This valve is connected directly to the throttle control. The emergency system is compensated only for changes in altitude. Fail safe stops, for take-off protection are incorporated in the fuel control. These stops replace an automatic emergency transfer system which, from past experience, is considered too unreliable because of its complexity to furnish the required protection without incurring additional hazards. The fail safe stops are armed during the last 7½ degrees of throttle travel and are intended to supply a normally adequate measure of safety during take-off when the time for manually switching to emergency is at a minimum. If a fuel control failure, which decreases fuel flow, is experienced while these stops are armed, there will be a variable decrease in thrust but not to an extent which gives less resultant thrust than that which would be attained at Military thrust for 6000 feet on a standard hot day. In order to prevent over-fueling, the engine, with resultant overspeed and overtemperature, these stops must be disengaged at or below 7000 feet altitude by retarding the throttle below the take-off position.

FUEL PRESSURIZING AND DUMP VALVE

The fuel pressurizing and dump valve is located in the engine fuel system between the fuel control and the engine burners. The pressurizing valve is designed to

ENGINE FUEL CONTROL *schematic*

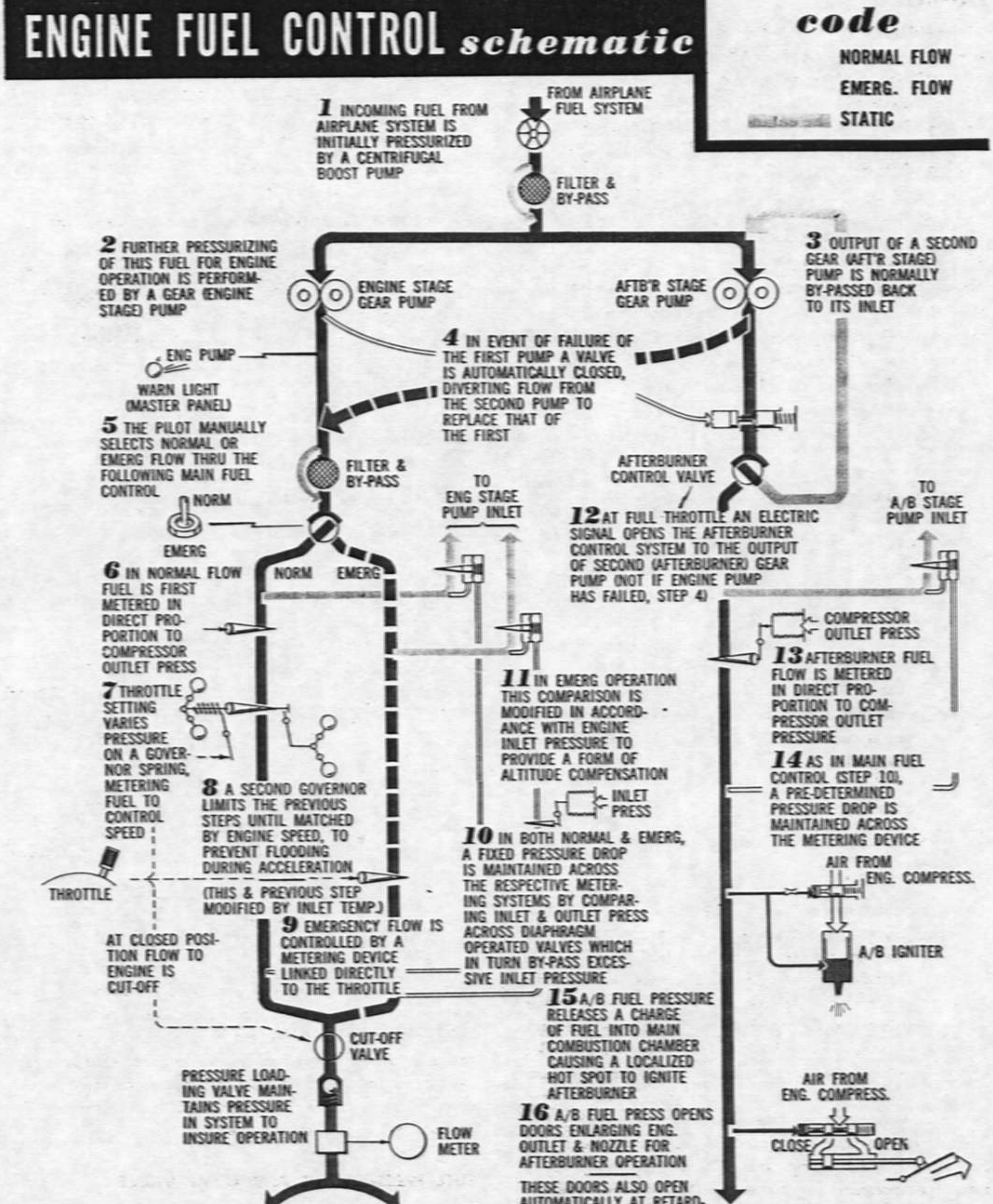


Figure 1—7

Figure 1–8

maintain fuel pressure in the system to assure satisfactory operation. The dump valve is closed during engine operation. When the engine is stopped the dump valve automatically opens to drain the engine fuel manifold.

ENGINE FUEL SYSTEM CONTROLS

THROTTLE CONTROL

Engine power is controlled by the throttle control (3, figure 1–5) which is located in a quadrant on the left console. The throttle is mechanically linked to the engine fuel control. The throttle control rotates on two axes so that movement is forward and aft and at certain points can be moved outboard or inboard. The throttle positions and their functions are as follows: The OFF, or outboard aft position closes the fuel shut-off valve in the fuel control and de-energizes the engine ignition circuit. The control is then moved forward then inboard to the IDLE position. This arms the engine ignition circuit and opens the fuel shut-off valve in the engine fuel control. From IDLE to the MILITARY position, with the throttle inboard, the fuel flow is adjusted for required engine output. At approximately 83 per cent rpm the throttle can be moved outboard. This opens the afterburner control valve, powered by the d-c primary bus, to direct fuel to the afterburner fuel system and ignite the afterburner. To move the throttle forward of the MILITARY position the MILITARY BY-PASS button is depressed while the throttle is advanced to the TAKE-OFF position. The TAKE-OFF position arms the fail safe stops in the fuel control for take-off protection. These stops must be disengaged at or below 7000 feet by retarding the throttle to the MILITARY position. It is not necessary to depress the MILITARY BY-PASS button to retard the throttle from the TAKE-OFF position.

FUEL SYSTEM SELECTOR SWITCH

The fuel system selector switch (figure 1–9) directs the normal or the emergency fuel flow through the engine fuel control. The switch is powered from the d-c primary bus and has two positions NORMAL and EMERGENCY. The NORMAL position directs fuel flow through the main fuel control and the engine is protected against overspeed and overtemperature conditions during rapid engine accelerations, compressor surge or compressor stalls. During rapid decelerations the engine is protected against flame-out. The EMERGENCY position is selected in event of main fuel control failure and directs fuel flow through the emergency fuel control. During emergency operation the fuel flow is compensated only for changes in altitude.

CAUTION

Since the emergency system does not offer the automatic overspeed, overtemperature, flame-out and compressor stall prevention features of the main fuel control system, rapid throttle movements should be avoided during operation on the emergency system.

FUEL PUMP PRESSURE WARNING LIGHT

A placard type warning light (figure 1—12) marked ENG FUEL PUMP OUT will illuminate if the pressure output of the main engine fuel pump drops below psi. The light is powered from the d-c primary bus.

ENGINE STARTER AND IGNITION SYSTEMS

The engine is equipped with a pneumatic starter that requires the application of an external compressed air source for operation. Actuation of the engine starter controls opens a valve that directs compressed air from the external source to actuate the starter. A centrifugal switch closes the air valve to automatically disengage the starter at approximately 50 per cent engine rpm. A control is also provided to disengage the starter if the engine fails to start. Engine ignition is supplied by high tension units powered by the primary bus and connected to two igniter plugs. The ignition system is used only during engine starting as combustion is continuous after the engine has been started. The ignition circuit is deenergized when the starter is automatically disengaged at approximately 50 per cent engine rpm. If the engine fails to start the ignition circuit can be de-energized. A switch is provided to energize the ignition system during air starts. As the afterburner is ignited by a "hot streak ignition", no electrical ignition system is provided for this unit.

ENGINE STARTER AND IGNITION CONTROLS

ENGINE GROUND START SWITCH

The engine starter is controlled by a three position switch (20, figure 1—4) marked ENGINE GROUND START, off and STOP START. If the d-c primary bus is energized, holding the engine start switch in the GROUND START position momentarily, opens an air valve so that the starter can be actuated by air from an external source when the throttle control from OFF to IDLE energizes the engine ignition circuit. An electrical holding relay keeps the starter and ignition circuits energized until engine rpm reaches approximately 50 per cent. Both circuits are then automatically opened. If an engine start is unsuccessful the starter and ignition circuits can be de-energized by momentarily placing the engine start switch in the STOP START position. These two circuits are for ground operation only.

AIR START SWITCH

A momentary contact push button type switch (figure 1—11) marked AIR START is provided to supply engine ignition for engine air starts when the engine is windmilling. The switch is powered from the d-c primary bus and connected to a time delay relay which supplies engine ignition for 20 seconds after actuation.

ENGINE INDICATORS

EXHAUST TEMPERATURE GAGE

The exhaust temperature gage (10, figure 1—3), located on the instrument panel, indicates engine exhaust temperature in degrees centigrade. Gage indications are received from thermocouples mounted directly downstream of the third stage turbine. The temperature indicator system is of the self generating type and therefore does not require power from the airplane electrical system.

TACHOMETER

The tachometer (20, figure 1—3), located on the instrument panel, registers engine speed in percentage of the approximate maximum rpm of the high pressure compressor rotor. The tachometer receives its power from a tachometer generator that is geared to the engine accessory section driven by the high pressure compressor rotor and is therefore independent of the airplane electrical system.

PRESSURE RATIO GAGE

The pressure ratio gage (9, figure 1—3) indicates the ratio of total turbine discharge pressure to compressor inlet total pressure. Thrust indication is read on this gage in lieu of the tachometer. One per cent variation in rpm results in approximately five per cent variation in thrust at the higher power settings while one per cent variation in turbine discharge pressure ratio results in only approximately one and one-half per cent variation in thrust. Thus, increased accuracy of power setting is obtained when turbine discharge pressure is utilized. When using turbine discharge pressure to check power prior to take-off, a thrust overshoot may be noted when the throttle control is advanced to MILITARY 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; therefore it may be advantageous to utilize this excess thrust for take-off.

OIL PRESSURE GAGE

The oil pressure gage (11, figure 1—3) indicates oil pump discharge pressure above oil breather pressure.

ENGINE AFTERBURNER SYSTEM

The engine is equipped with an afterburner to produce increased thrust for maximum power. Because of its high fuel consumption the afterburner is intended to be used for short operational periods only, such as during takeoff or during critical climb or flight conditions. Operation of the afterburner is controlled by the throttle.

When the throttle is moved outboard into AFTERBURNER an electrically operated shut-off valve in the engine-driven fuel pump unit is opened to direct fuel to the afterburner fuel metering valve, which meters fuel in proportion to engine airflow. The metered fuel flow from this valve passes to the afterburner igniter and the afterburner fuel nozzles. Concurrently, fuel is directed to the exhaust nozzle control to open the nozzle for afterburner operation. Since afterburner fuel flow is determined by the altitude and airspeed compensated afterburner fuel metering valve, which is governed by altitude and engine speed, there is no direct control of this fuel flow. However, during afterburner operation, a thrust variation ranging between the maximum available thrust and the equivalent of approximately 50 per cent afterburning can be obtained by advancing or retarding the throttle 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, throttle movement must be made carefully to prevent engine overspeed and overtemperature.

VARIABLE EXHAUST NOZZLE

The two-position, finger type exhaust nozzle mounted at the end of the tail pipe, is positioned to provide the proper exhaust nozzle area for either normal or afterburner engine operation. The nozzle fingers are moved to the full open position during afterburner operation and are positioned to the minimum nozzle opening area when the afterburner is not in use. During normal operation at idle engine rpm the nozzle fingers also open automatically to provide a thrust decay system for landing and taxiing. Positioning of the nozzle fingers is accomplished automatically by means of the exhaust nozzle control unit, and no emergency override control is included.

EXHAUST NOZZLE CONTROL UNIT

Opening and closing of the exhaust nozzle is controlled by afterburner fuel pressure and actuated by air from the engine compressor. When the throttle is moved outboard to AFTERBURNER, the electrically operated shut-off valve in the fuel pump unit is opened, permitting fuel from the afterburner fuel pump to enter the exhaust nozzle control unit. This fuel pressure moves a valve within the control unit, which directs compressor discharge air pressure to the nozzle actuators to open the fingers. Moving the throttle inboard from AFTERBURNER closes the shut-off valve, so that fuel pressure is no longer supplied to the nozzle control unit. The valve in the control unit is then positioned to direct engine compressor air to close the fingers for normal engine operation.

AFTERBURNER IGNITER

When the afterburner system is engaged, fuel from the afterburner fuel metering valve is directed to the igniter unit. This fuel flow actuates the igniter unit, which then injects a small amount of fuel into one burner of the engine combustion section, 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 nozzles. The igniter is actuated only when full pressure is built up within the afterburner manifold, so that fuel is available at the nozzles when the igniter introduces fuel to the burner for ignition. No repeater mechanism is incorporated in the igniter, and the unit does not recycle until the afterburner fuel pressure is shut-off and then restored. If afterburner light-up is not obtained within 3 seconds at sea level (5 seconds at altitude) after the throttle is moved outboard to the AFTERBURNER range, the throttle should be moved momentarily inboard from this position and then returned outboard to AFTERBURNER to recycle the igniter.

ENGINE ANTI-ICING SYSTEM

The engine anti-icing system is fully automatic and cannot be controlled by the pilot. When ice starts to form on the engine inlet vanes, the vacuum formed by the compressor operates a bellows to energize electrical circuits to operate motors to open the valves and allow the air from the compressor 16th stage to be ducted to the inlet vanes. Air flow regulators, consisting of a temperature-sensitive bimetallic coil attached to a butterfly valve, control the flow of air. When the ice is removed and normal inlet pressure conditions return, the bellows operated switch returns to its normal position and energizes the valve motors to close the valves and air flow to the inlet vanes is cut-off. A warning light is provided to indicate to the pilot when an icing condition is present. The anti-icing system is powered from the d-c primary bus.

ENGINE ICING WARNING LIGHT

The engine icing warning light (figure 1–12) is a placard type light marked ENGINE ICING and powered from the d-c primary bus. The light will illuminate together with the master warning light when an engine icing condition exists and will go out when the icing condition is eliminated. The master warning light can be put out by depressing the master warning light reset switch.

AIRPLANE FUEL SYSTEM

code schematic

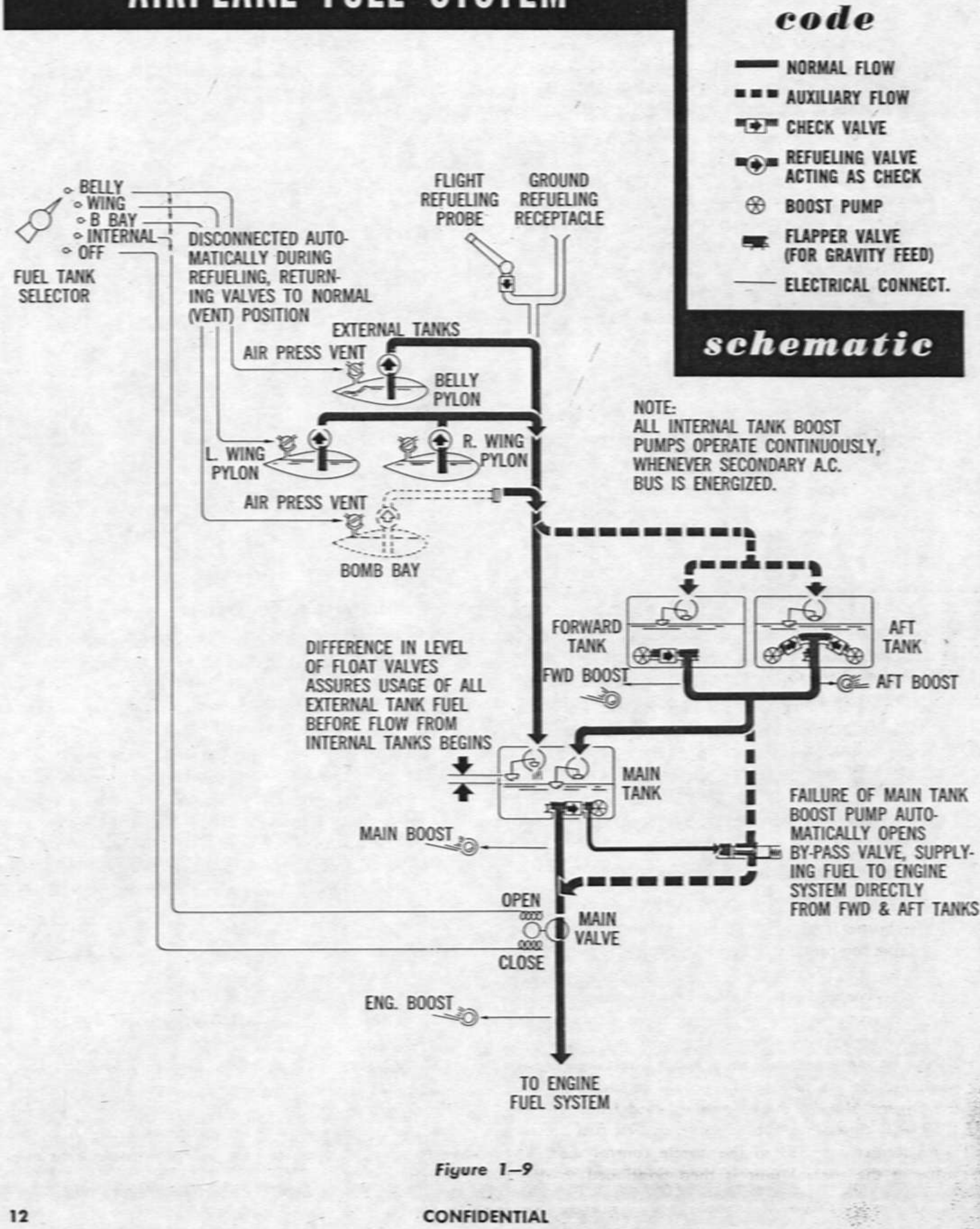


Figure 1–9

OIL SYSTEM

The dry-sump, recirculating, pressure-type engine oil system is supplied from a 5.5 US gallon tank mounted on the left side of the engine compressor section. Usable oil supply is 3 US gallons. Oil flows from the tank to the gear-type pump, which supplies oil under pressure to lubricate and cool bearings and gears within the engine. Scavenged oil is picked up by 6 gear-type pumps, and is sent through a ram air cooled oil cooler and then to a fuel-cooled oil cooler. The oil is then returned to the tank to repeat the oil flow cycle. The air-cooled cooler is the secondary cooler in the oil system and is provided to augment the fuel cooled unit, particularly at high altitudes where fuel flow is inadequate for proper oil cooling. Normal operation of the air-cooled cooler thermostatic valve, by-passes the oil around the air oil cooler and routes it through the fuel oil cooler on the engine. When oil temperatures exceed the rated value of the air oil cooler thermostatic valve, the valve closes the by-pass and routes the oil through the air oil cooler.

AIRPLANE FUEL SYSTEM

The airplane fuel system (figure 1–9) consists of three bladder type tanks (main, forward and aft) premanently installed in the fuselage, a non-jettisonable metal tank installed in the bomb bay, a metal pylon tank mounted under each wing and a metal tank mounted on a pylon on the fuselage center line. Fuel from the main, forward and aft tanks is transferred by means of booster pump in each tank while transfer of fuel from the bomb bay tank and the external tanks is accomplished by means of air pressure bled from the engine compressor. Fuel transfer from the internal tanks is automatic when the booster pumps are energized but the bomb bay tank and external tank transfer must be manually selected. Normally fuel is supplied to the engine from the main tank, provided the main tank booster pump is in operation. Pressure from the booster pump also keeps a valve in a line from the forward and aft tank in the closed position. Should the main tank booster pump fail, spring pressure will open this valve and allow fuel flow from the forward and aft tanks to be directed to the engine automatically. A selector switch is provided to open the fuel supply line to the engine and also select pressurization of the external bomb bay tanks. During normal operation fuel flow is from the main tank to the engine. The main tank fuel is replenished by selecting the external and bomb bay tanks in succession until the fuel level in the main tank drops below the external transfer valve level. Fuel is then transferred automatically to the main tank simultaneously from the forward and aft internal tanks until the supply is depleted. The remaining fuel in the main tank is then transferred to the engine. A fuel quantity indicator is provided to record the fuel remaining in the various tanks. Warning lights are also provided to warn the pilot of malfunction or condition of the components of the fuel system.

BOOSTER PUMPS

A booster pump is installed in both the main and forward fuel tanks while the aft fuel tank is equipped with two booster pumps. The pumps will start to operate as soon as the secondary a-c bus in energized and will continue to operate as long as power is available. There are no switches provided to turn the pumps off. The main tank booster pump supplies fuel to the engine fuel system. The differential pressure between the main tank and the main tank booster pump is used to hold a valve closed in a supply line from the forward and aft tanks to the engine fuel supply. In the event of main tank booster pump failure, this valve automatically opens by spring pressure and allows the output of the forward and aft booster pumps to be directed to the engine fuel system. A flapper type check valve in the main booster pump outlet will be closed by the fuel pressure to prevent fuel from being pumped into the main tank during this type operation. Each booster pump is equipped with a flapper check valve which will be held closed by pressure from other boosters in case of failure. If all booster pumps fail, the flapper valves will open and gravity feed of fuel (assisted by suction from the engine fuel pump) will take place.

FUEL TANK SELECTOR

The fuel tank selector (figure 1–11) is a rotary type switch which electrically controls various fuel tank feeds when manually positioned. There are five positions: namely, OFF, MAIN TANK, BOMB BAY TANK, EXT WING TANKS and BELLY TANK. When positioned in the BELLY TANK, EXT WING TANKS or BOMB BAY TANK position, the respective pressure vent valve is opened to pressure to allow the tank to transfer fuel to the internal tanks. The fuel shut-off valve between the airplane and engine fuel systems is also opened allowing fuel flow from the main tank to the engine. When positioned to MAIN TANK only the fuel shut-off valve is opened allowing fuel flow to the engine fuel system. In this position fuel will not transfer from any of the external tanks. The OFF position closes the fuel shut-off valve cutting off all fuel flow to the engine. All circuits connected to this switch are powered by the d-c primary bus.

JETTISONING EXTERNAL TANKS

JETTISON EXTERNAL STORES SWITCH

The jettison external stores switch (10, figure 1–4) is a push button switch marked JETTISON EXTERNAL STORES. When depressed, d-c primary bus power is

FUEL TANK CAPACITIES U.S. GALS.

TANK	USABLE FUEL	FULLY SERVICED	EXPANSION SPACE	TOTAL VOLUME
MAIN	300.0	303.0	9.5	312.5
FORWARD	403.0	408.0	12.9	420.9
AFT	447.0	452.0	14.1	466.1
BOMB BAY	350.0	351.7	14.2	365.9
INBOARD PYLON				
RIGHT		450.0		
LEFT		450.0		
FUSELAGE PYLON		450.0		

USABLE FUEL TOTALS

MAXIMUM INTERNAL	1500.0
MAXIMUM INTERNAL PLUS EXTERNAL	2850.0

Figure 1—10

FUEL SYSTEM

Control Panel

Figure 1–11

supplied to explosive cartridges in each pylon (outboard, inboard and center) consequently jettisoning all pylons and stores. If the d-c primary bus de-energized, battery bus power is automatically supplied to the explosive cartridges when the switch is depressed.

PYLON JETTISON

Three push button type pylon jettison switches (22, 21, 19, figure 1–4) are marked CENTER, INBD and OUTBD. As any one of the switches is depressed, d-c primary bus power is supplied to explosive cartridges in the respective pylon, thereby jettisoning the pylon with any attached stores. If d-c primary bus power is not available battery bus power is automatically supplied to the cartridges.

JETTISON WING TANKS SWITCH

The jettison wing tanks switch (figure 1–11) is a two position toggle switch guarded in the off position. When the cover guard is lifted and the switch positioned to the JETTISON WING TANKS position, d-c primary bus power is supplied to explosive cartridges in the inboard pylons releasing the inboard pylon tanks. If d-c primary bus power is not available battery bus power is automatically supplied to the circuit. A safety circuit is incorporated in the system which prevents dropping bombs from the inboard pylons by accidentally actuating the jettison wing tanks switch. The circuit also prevents dropping fuel tanks from the inboard pylons if the bomb release switch is depressed.

JETTISON BELLY TANK SWITCH

The jettison belly tank switch (figure 1–11) is a two position toggle switch guarded in the off position. When the cover guard is lifted and the switch positioned to the JETTISON BELLY TANK position, d-c primary bus power is supplied to explosive cartridges in the fuselage pylon. If d-c primary bus power is not available battery bus power is automatically supplied to the circuit.

FUEL SYSTEM INDICATORS

FUEL QUANTITY INDICATOR

The fuel quantity indicator (15, figure 1–3) provides the means of measuring electrically the fuel in the internal fuel cells, the bomb bay tank and the wing pylon tanks. No means of indicating fuel quantity is supplied for the belly tank. Total fuel weight for all internal and external tanks (not including belly tank) is indicated on the outside scale of the fuel quantity indicator which is marked TOT. FUEL. Smaller individual scales and pointers marked L. H. PYL, R.H. PYL and B.B. AUX indicate the contents in the respective tanks. The fuel quantity indicator system is powered from the a-c primary circuit and fuel quantity readings are corrected for temperature and fuel density variations.

FUEL FLOW INDICATOR

The fuel flow indicator (12, figure 1–3) is powered from the a-c primary circuit and shows the rate of fuel flow from the engine fuel control to the engine burners in pounds per hour. The instrument does not register fuel flow to the afterburner fuel system.

MASTER WARNING LIGHT AND RESET SWITCH

The master warning light (8, figure 1–3) is powered from the primary bus and is marked MASTER WARNING. This light will illuminate together with any individual warning light located above the right console. The master warning light is located so as to be easily visible to the pilot. When the light illuminates, the pilot checks the warning light panel above the right console to establish which system is malfunctioning. After it is ascertained which system has caused the light to illuminate, the reset switch is depressed. This extinguishes the master warning light which will then be ready for indication of another warning. The individual warning light will remain illuminated.

FUEL LEVEL LOW WARNING LIGHT

The fuel level low warning light (figure 1–12) is a placard type light powered from the d-c primary bus and marked FUEL LEVEL LOW. The light will illuminate together with the master warning light when the total fuel remaining in the airplane is approximately 3000 pounds.

FUEL INLET PRESSURE LOW WARNING LIGHT

The fuel inlet pressure low warning light (figure 1–12) is a placard type light powered from the d-c primary bus and marked FUEL INLET PRESS LOW. The light will

WARNING LIGHTS Panel

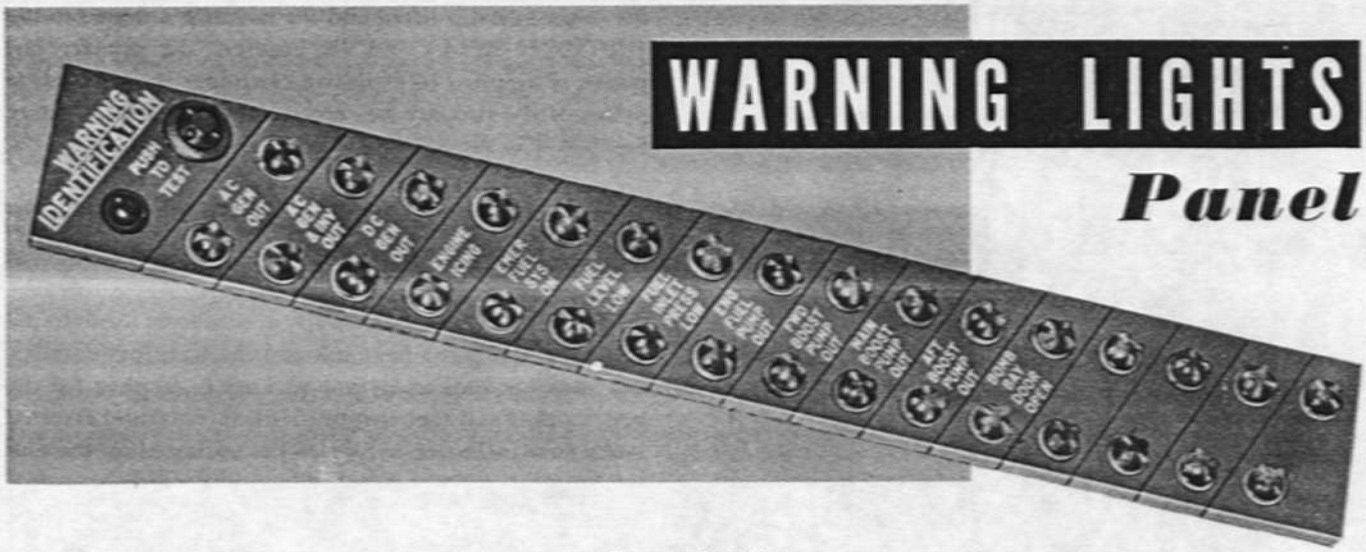


Figure 1–12

illuminate together with the master warning light when the differential pressure in the fuel line from the fuel shut-off valve to the engine fuel system and the main tank vent pressure is between 1 and 4 psi.

BOOSTER PUMP WARNING LIGHTS

Three booster pump warning lights (figure 1–12) are placard type lights powered from the d-c primary bus and are marked FWD BOOST PUMP OUT, MAIN BOOST PUMP OUT and AFT BOOST PUMP OUT. The lights will illuminate together with the master warning light when the differential pressure between the respective booster pump inlet and outlet is between 1 and 4 psi. The aft booster pump will indicate failure of either or both of the booster pumps in the aft fuel tank.

BELLY TANK EMPTY INDICATOR LIGHT

The belly tank empty indicator light (figure 1–12) is powered from the primary bus and will illuminate when the belly tank is empty. This light is provided as the fuel quantity indicator has no provision for fuel quantity remaining in the belly tank.

EXPLOSION DETECTION AND SUPPRESSION SYSTEM

The explosion detection and suppression system is incorporated in each of the internal fuel tanks. This system prevents explosions within the internal tanks which may be caused by enemy action. The system consists of an explosion detector and squib-operated suppression-material capsules in each tank. When light strikes the detector, a d-c primary bus source fires the squibs exploding and distributing the suppression-material within the fuel cell.

ELECTRICAL POWER SUPPLY SYSTEM

The airplane is equipped with two basic electrical power supply systems, namely a direct-current and an alternating-current system. All circuits are protected by a circuit breaker which is accessible from the cockpit.

CIRCUIT BREAKER PANELS

Circuit breaker panels (figure 1–14 and 1–15) are provided to protect the various electrical circuits in the airplane. The circuit breakers are arranged in panels and each panel series a major electrical circuit. This arrangement facilitates locating and identifying each circuit breaker.

DIRECT-CURRENT SYSTEM

The 28 volt d-c system (figure 1–13) is powered from a 400 ampere, engine-driven generator with a 24 volt battery as a standby source of d-c power. The system also incorporates an external power receptacle for the accommodation of an external power cart. Electrical power is distributed through a three bus system consisting of a battery bus, a primary bus and a secondary bus. The battery bus services emergency equipment and remains energized regardless of the battery switch position or generator operation. The primary bus services equipment essential to flight and is energized by the battery, the generator, or the external power receptacle. The secondary bus services equipment not essential to flight and is energized by the generator or the external power receptacle. Therefore, in the event of generator failure in flight, all d-c energized equipment not essential to flight will be automatically cut out since the d-c secondary bus will cease to be energized, and battery power will be conserved for d-c primary bus equipment.

BATTERY SWITCH

The battery switch (17, figure 1–4) is a two position switch marked BAT ON and OFF. The BAT ON position connects the airplane's battery to the d-c primary bus.

ELECTRICAL SYSTEM *schematic*

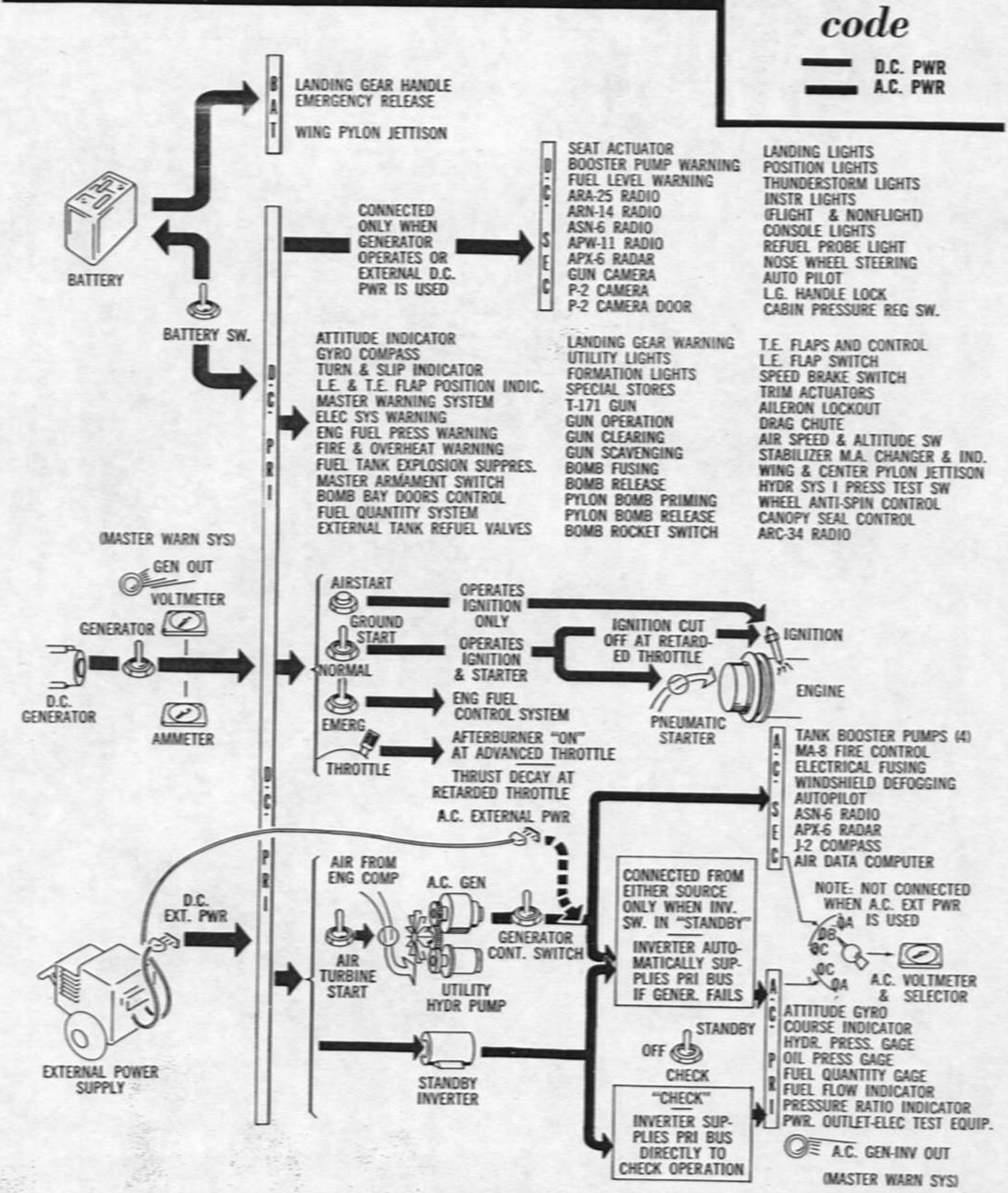


Figure 1–13

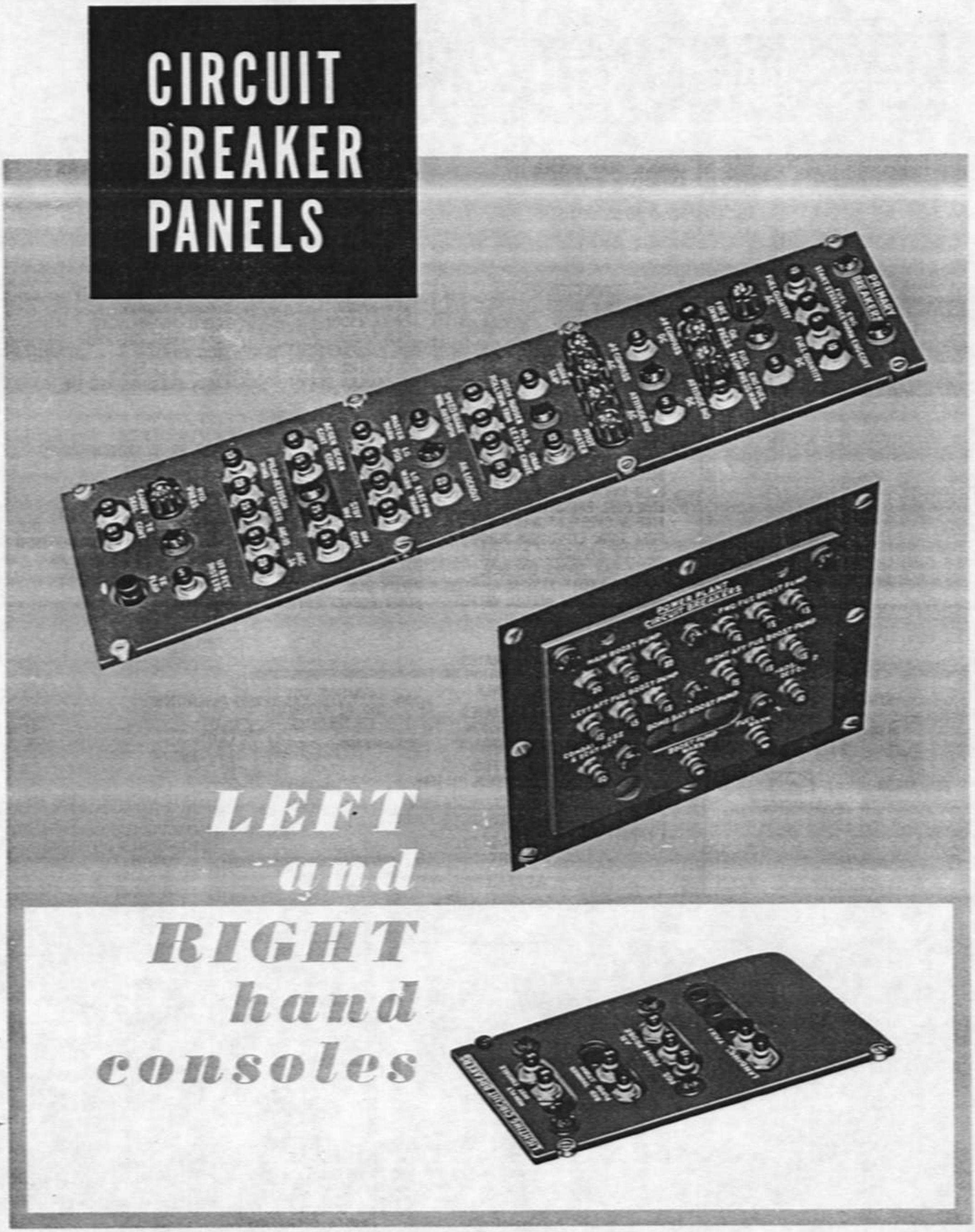
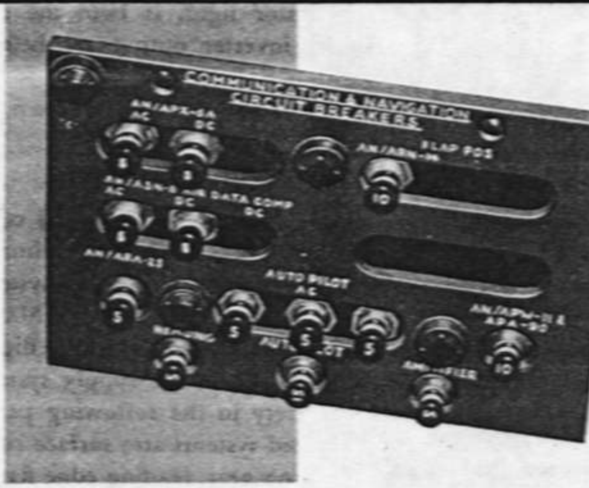


Figure 1–14

CIRCUIT BREAKER PANEL



Right Hand Console

Figure 1–15

When the battery switch is in the OFF position the d-c primary bus can only be energized when a d-c external power source is connected to the airplane.

GENERATOR SWITCH

The generator switch (16, figure 1–4) is a three position switch marked DC GEN ON, OFF and RESET and guarded in the ON position. If the generator voltage is too high or too low, the generator is automatically disconnected from the electrical circuit. If the generator has cut-out due to variations in voltage, the generator switch is positioned to RESET for a few seconds, to reset the generator field control relay and then returned to ON. If the generator does not operate after positioning the generator switch to RESET and returning to the ON position, the OFF position is selected to disconnect the generator from the electrical system. The ON position connects the generator to the electrical system whenever the generator speed is high enough to close the reverse current relay.

LOADMETER

The loadmeter (25, figure 1–4) is marked LOAD and indicates the load being drawn from the generator in per cent from 0 to 100 per cent, with provisions for an additional 25 per cent reading to indicate over-load, and a minus 10 per cent reading to indicate discharge.

VOLTMETER

The d-c voltmeter (24, figure 1–4) indicates voltages output of the generator.

GENERATOR OUT INDICATOR LIGHT

The generator out indicator light (figure 1–12) is a placard type light marked DC GEN OUT. Whenever the reverse current relay opens, the generator out indicator will illuminate together with the master warning light which shows that the generator is disconnected from the electrical system. The master warning light can be extinguished by depressing the master warning reset switch. However, the generator out indicator will remain illuminated.

EXTERNAL POWER RECEPTACLE

The d-c external power receptacle energizes the d-c primary and secondary busses for ground operation. The a-c primary bus is also energized by using the standby inverter for checking a-c powered equipment necessary for flight. The a-c secondary bus cannot be energized by the d-c external power receptacle.

ALTERNATING CURRENT SYSTEM

The a-c electrical system (figure 1–13) is powered by a 3 phase, 120/208 volt, 400 cycle air driven generator with a 3 phase, 115 volt standby inverter as an emergency source of a-c power. The generator is driven by an air turbine in conjunction with the utility hydraulic pump. The air turbine is driven by air bled from the engine compressor and can be controlled from the cockpit. Electrical power is distributed through a two bus system, namely, the a-c primary bus and the a-c secondary bus. The a-c primary bus is energized by the a-c generator or the standby inverter and supplies power to a-c equipment requiring single phase current. The a-c secondary bus is energized by the a-c generator and supplies power to equipment requiring three phase current.

AIR TURBINE SWITCH

The air turbine switch (18, figure 1–4) is a two position switch marked AIR TURBINE START and STOP. The START position supplies d-c primary bus power to open a valve in the compressed air line from the engine which supplies compressed air to operate the air turbine. The air turbine powers the a-c generator and the utility hydraulic pump. The OFF position stops the supply of air to the turbine making it inoperative. In the event of d-c primary bus failure the valve in the compressed air line will close and the turbine will be inoperative.

A-C GENERATOR SWITCH

The a-c generator switch (13, figure 1–4) is a three position switch marked AC GEN ON, OFF and RESET and is guarded in the ON position. If the a-c generator voltage is too high or too low the generator is automatically disconnected from the electrical system. If the generator has cut-out, due to over voltages, the generator switch is positioned to RESET for a few seconds, to reset the generator field control relay, and then returned to the ON position. If the generator is not operating after positioning the generator switch to RESET and returning to the ON position the OFF position is selected to disconnect the generator from the electrical system.

INVERTER SWITCH

The inverter switch (14, figure 1–4) is a three position switch marked STANDBY, OFF and CHECK. The STANDBY position automatically supplies d-c primary bus power to operate the a-c inverter in the event of failure of the a-c generator. The STANDBY position must be selected for the a-c generator to supply power to the a-c primary bus. The inverter powers the a-c primary bus. During ground servicing or maintenance when the engine and a-c generator are inoperative, the inverter would automatically operate from the d-c primary bus, therefore, the OFF position is provided so that the inverter can be turned off. The CHECK position is a momentary contact position which is used to check the operation of the inverter during preflight inspections. This position energizes the inverter directly from the d-c primary bus.

VOLTMETER SELECTOR SWITCH

The voltmeter selector switch (15, figure 1–4) is a rotary type switch which is provided so that the various a-c busses may be selected and their voltage checked on a common voltmeter. Three switch positions, marked AC GEN (A, B and C) are provided to select the three phases of the a-c secondary bus. When positioned to any of these positions the voltage at the respective bus will be read on the a-c voltmeter. Two switch positions marked INVERTER (A and C) are provided to select the two single phase circuits of the a-c primary bus. When positioned to either of the two positions the voltage at the respective bus will be read on the a-c voltmeter.

VOLTMETER – AC

The a-c voltmeter (26, figure 1–4) is provided to indicate the a-c voltage on each of the a-c system busses. The instrument limits range from 0 to 150 volts. The reading will indicate voltage on the bus selected by the position of the voltmeter selector switch.

A-C GENERATOR OUT INDICATOR LIGHT

The a-c generator out indicator light (figure 1–12) is a placard type light marked AC GEN OUT and powered from the primary bus. The light will illuminate together with the master warning light if the a-c generator output is below normal and will remain illuminated as long as the a-c output is low. The master warning light can be extinguished by depressing the master warning light reset switch.

A-C GENERATOR AND INVERTER OUT INDICATOR LIGHT

The a-c generator and inverter out indicator light (figure 1–12) is a placard type light marked AC GEN & INV OUT and powered from the primary bus. The light will illuminate, together with the master warning light and the a-c generator out indicator light, if both the a-c generator and the standby inverter output are below normal. The master warning light can be extinguished by depressing the master warning light reset switch.

HYDRAULIC POWER SUPPLY SYSTEM

The hydraulic power supply system (figure 1–16) consists of three individual systems, namely, the primary system, the emergency system and the utility system operating at pressures of approximately 3000 psi. Maximum safety margins are incorporated for primary flight control actuation. Each hydraulic power supply system will be described in its entirety in the following paragraphs. Hydraulically operated systems are; surface control actuators, spoilers, landing gear, leading edge flaps, speed brakes, refueling probe, bomb bay doors, nosewheel steering and wheel brakes. These systems are described under applicable headings.

PRIMARY HYDRAULIC POWER SUPPLY SYSTEM

The primary hydraulic power supply system (figure 1–16) is divided into two individual systems, designated as the primary number one and primary number two systems. The two systems power the stabilator, rudder, spoilers and aileron actuators and are located on opposite sides of the airplane. Each system pressure is approximately 3000 psi and powers one side of the tandem surface control actuators delivering one half of the required load. With failure of one of the primary hydraulic systems, control of the airplane is maintained by pressures from the other primary hydraulic system but maneuverability is restricted in that the actuators, with only one side powered, apply a force which can support half of the maximum possible hinge movement. In the event of the failure of one primary system due to pump failure, the side of the tandem actuators normally powered by the failed pump, will by-pass hydraulic fluid while the other side of the actuator continues to operate. Each primary system is powered by an identical engine driven hydraulic pump. The pumps are variable displacement type, employing nine axial pistons, reciprocating in a rotary cylinder block. The block is enclosed in a swivelling yoke which is actuated by a pressure control assembly for varying the piston stroke. Full flow is available up to maximum regulated pressure. When maximum regulated pressure is reached, the piston stroke is reduced and the pumps automatically assume an unloaded condition (zero flow to system). An airless type reservoir, located in the main landing gear wheel well, is provided for each system. The reservoir employs system pressure acting on a pressurizing piston to maintain the necessary head of fluid at the pump inlet port. At the reservoir full level position, a spring applies a load of 5 psi on the reservoir fluid. When the engine starts, system pressure will pressurize the reservoir fluid with an additional 25 psi giving a total reservoir pressure of 30 psi which satisfies the pump inlet pressure requirements. Reservoir volume at filling is small due to the absence of any reservoir volumetric changes during operation of any of the actuators. A dual hydraulic pressure indicator powered from the a-c primary bus, indicates system one and system two pressures. A manually operated transfer valve is installed in the number one primary system to enable the pilot to direct hydraulic pressure from the emergency hydraulic system to the primary one system for surface control actuator operation in the event of engine failure.

EMERGENCY HYDRAULIC SYSTEM CONTROL

The emergency hydraulic system control (2, figure 1–5) has two positions TURBINE EXTENDED and TURBINE RETRACTED. This control manually positions a valve in the number one primary hydraulic system to direct hydraulic pressure from the emergency hydraulic turbine to the surface control actuators through number one system plumbing. Full description of the control will be made under the emergency hydraulic system.

HYDRAULIC PRESSURE GAGE

The hydraulic pressure gage (13, figure 1–3) is a dual indicator with two scales and two pointers. The pointer and scale on the left side of the instrument indicates pressure in the number one primary system. The other scale and pointer indicates pressure in the number two primary system. The scale limits are from 0 to 4000 psi. The instrument is powered from the a-c primary bus.

EMERGENCY HYDRAULIC POWER SUPPLY SYSTEM

The emergency hydraulic system (figure 1–17) is provided to power the primary flight controls in the event of engine failure or failure of the primary hydraulic power supply system. Emergency hydraulic system pressure is supported by a ram air driven variable displacement pump that is lowered into the airstream when the emergency system is selected. Emergency hydraulic pressure is supplemented by utility system pressure. An emergency system accumulator is provided to supply hydraulic pressure to the primary control actuators during the time it takes for the air driven pump to be extended and build up pressure. The accumulator pressure is maintained by the utility system pump.

EMERGENCY HYDRAULIC SYSTEM CONTROL

The emergency hydraulic system control (figure 1–18) is a two position control marked TURBINE EXTENDED and TURBINE RETRACTED. When placed in the TURBINE EXTENDED position, a transfer valve in the number one primary hydraulic system is manually positioned to shut off the line from the number one primary pump and direct emergency pressure from the air driven turbine into the number one primary system plumbing. A valve is also manually operated to supply utility system pressure to lower the air driven pump into the air steam. In this position the utility system pressure augments emergency pressure for primary flight control actuation. The TURBINE RETRACTED position manually positions the transfer valve to shut-off emergency hydraulic pressure and open the number one primary hydraulic pump to the number one system plumbing. The air driven pump is also retracted into the fuselage by utilizing utility system pressure. In this position the utility system pressure keeps the emergency hydraulic reservoir charged at all times. Emergency hydraulic system pressure will be indicated on the number one primary system hydraulic pressure gage. A switch is provided to cut-off utility system pressure from the emergency system so that the emergency system pressure alone can be checked.

UTILITY HYDRAULIC POWER SUPPLY SYSTEM

The utility hydraulic power supply system (figure 1–16) is provided to power all hydraulic systems except the primary flight controls. However, during emergency hydraulic system operation the utility power supply system supplements the emergency system for actuation of the primary flight controls. Utility system pressure is maintained by a variable displacement pump driven by an air turbine in conjunction with the a-c generator. The air turbine is driven by compressed air which is bled from the engine compressor. An airless type reservoir is provided for the utility system. The reservoir employs system pressure acting on a pressurizing piston to maintain the necessary head of fluid at the pump inlet port. At the reservoir full level position, a spring applies a load of 5 psi on the reservoir fluid. When the air turbine starts, system pressure will pressurize the reservoir fluid with an additional 30 psi giving a total reservoir pressure of 35 psi which satisfies the pump inlet pressure requirements. Reservoir volume at filling is small due to the absence of any reservoir volumetric changes during operation of any of the systems. The utility system pressure output is also directed to pressurize the emergency system accumulator. A valve is provided in the line between the utility pressure system and the emergency system so that utility pressure can be shut-off and the emergency system pump output pressure can be checked.

HYDRAULIC SYSTEM *schematic*

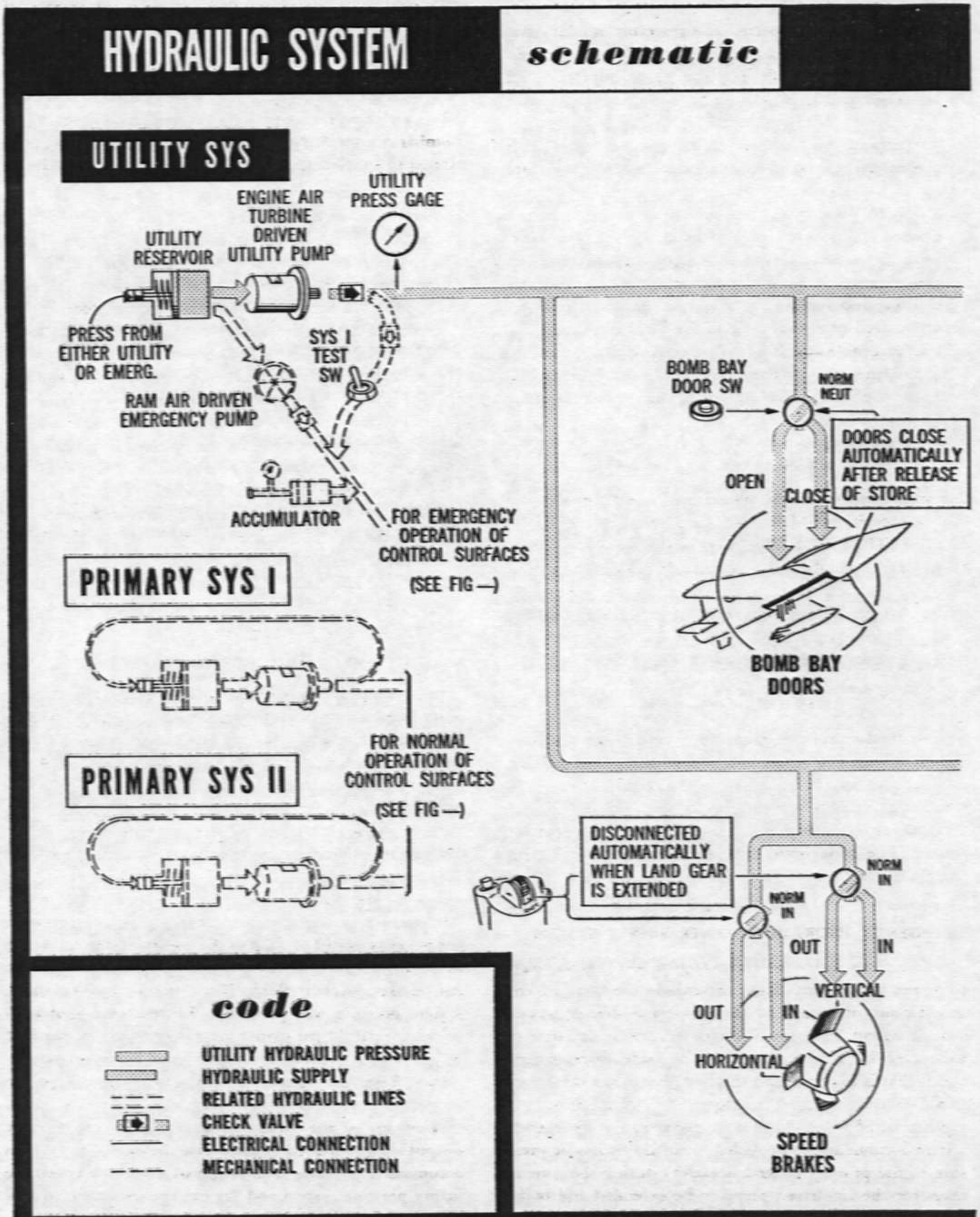


Figure 1-16 (Sheet 1 of 2)

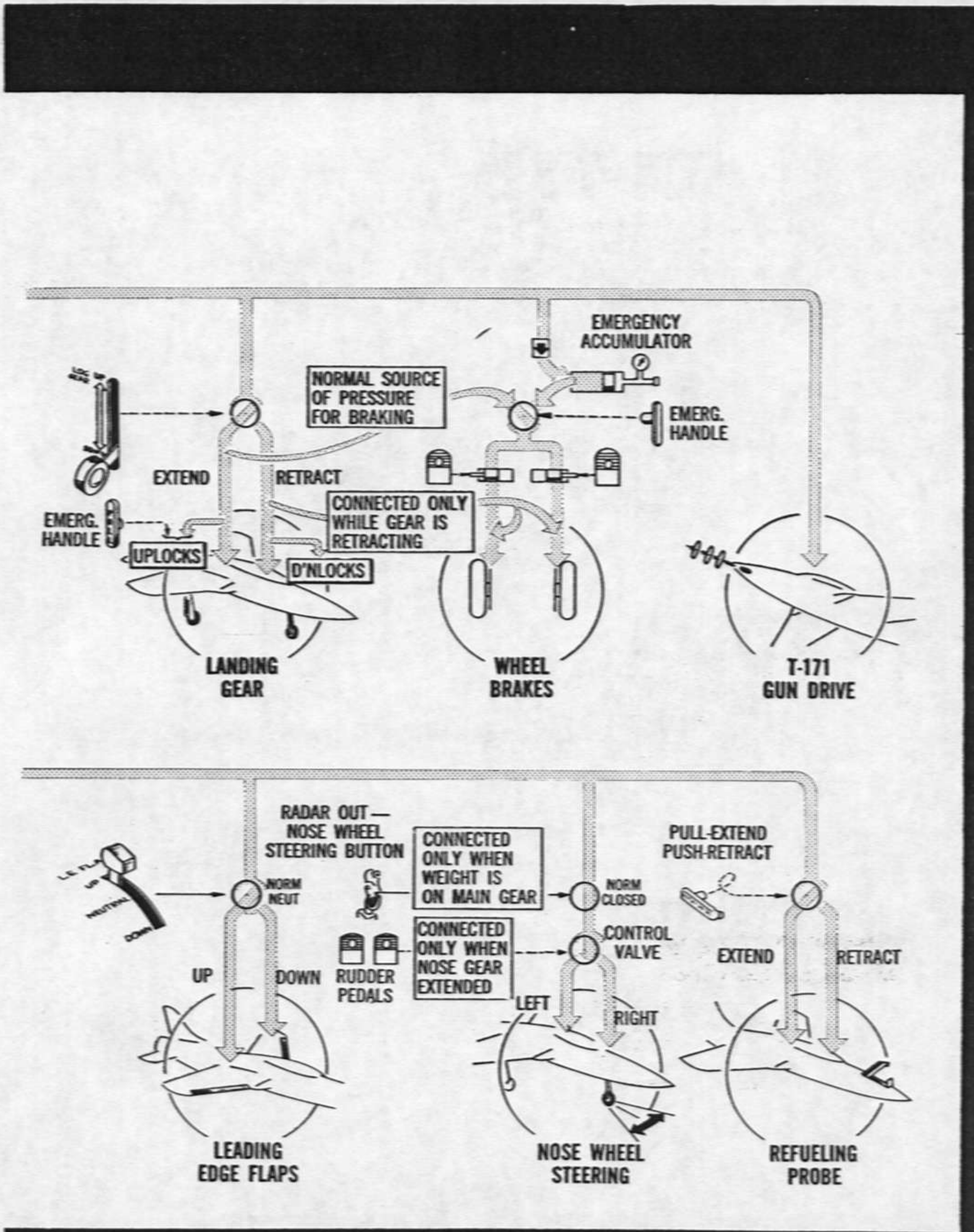


Figure 1—16 (Sheet 2 of 2)

FLIGHT CONTROL

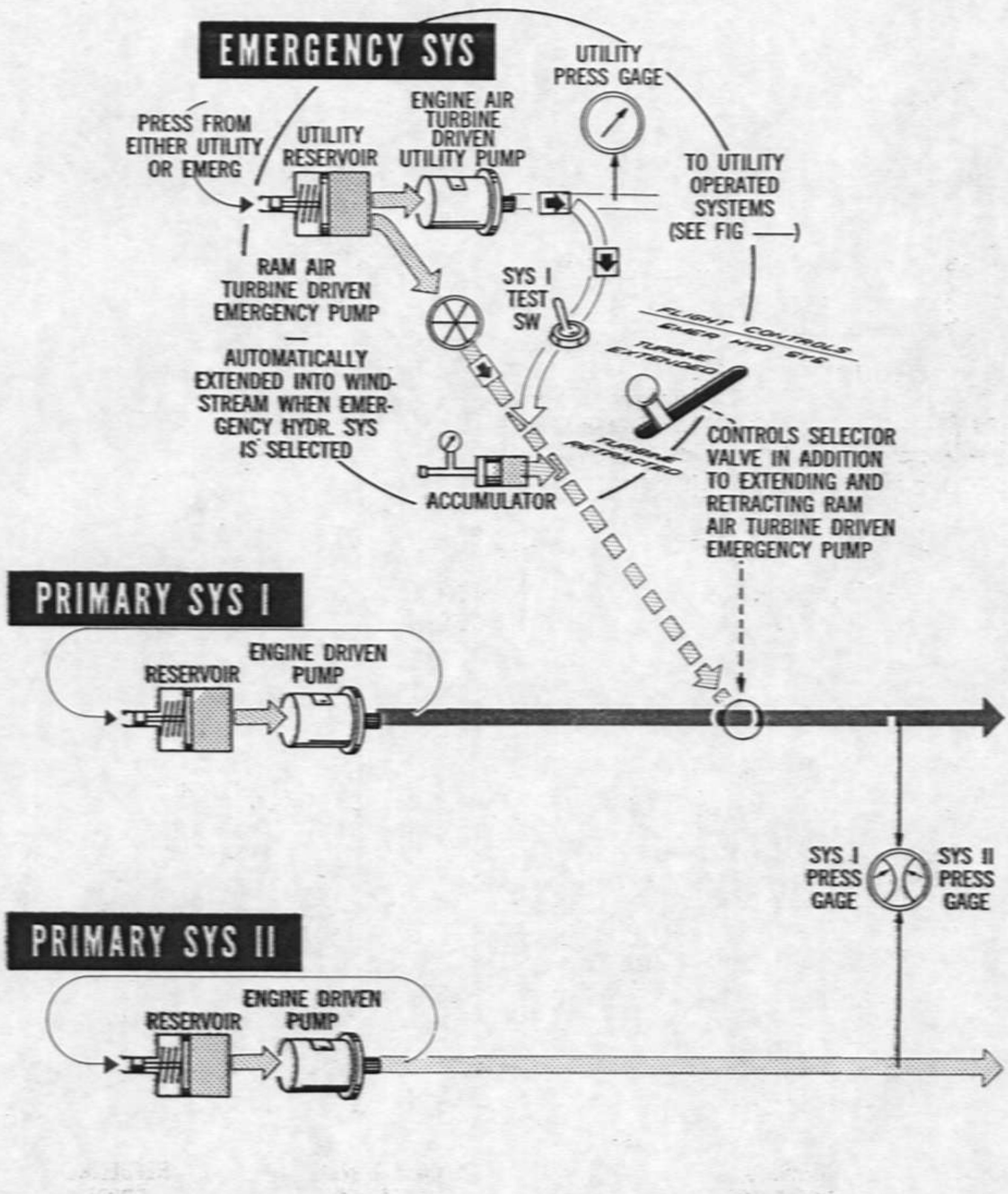


Figure 1-17 (Sheet 1 of 2)

hydraulic system schematic

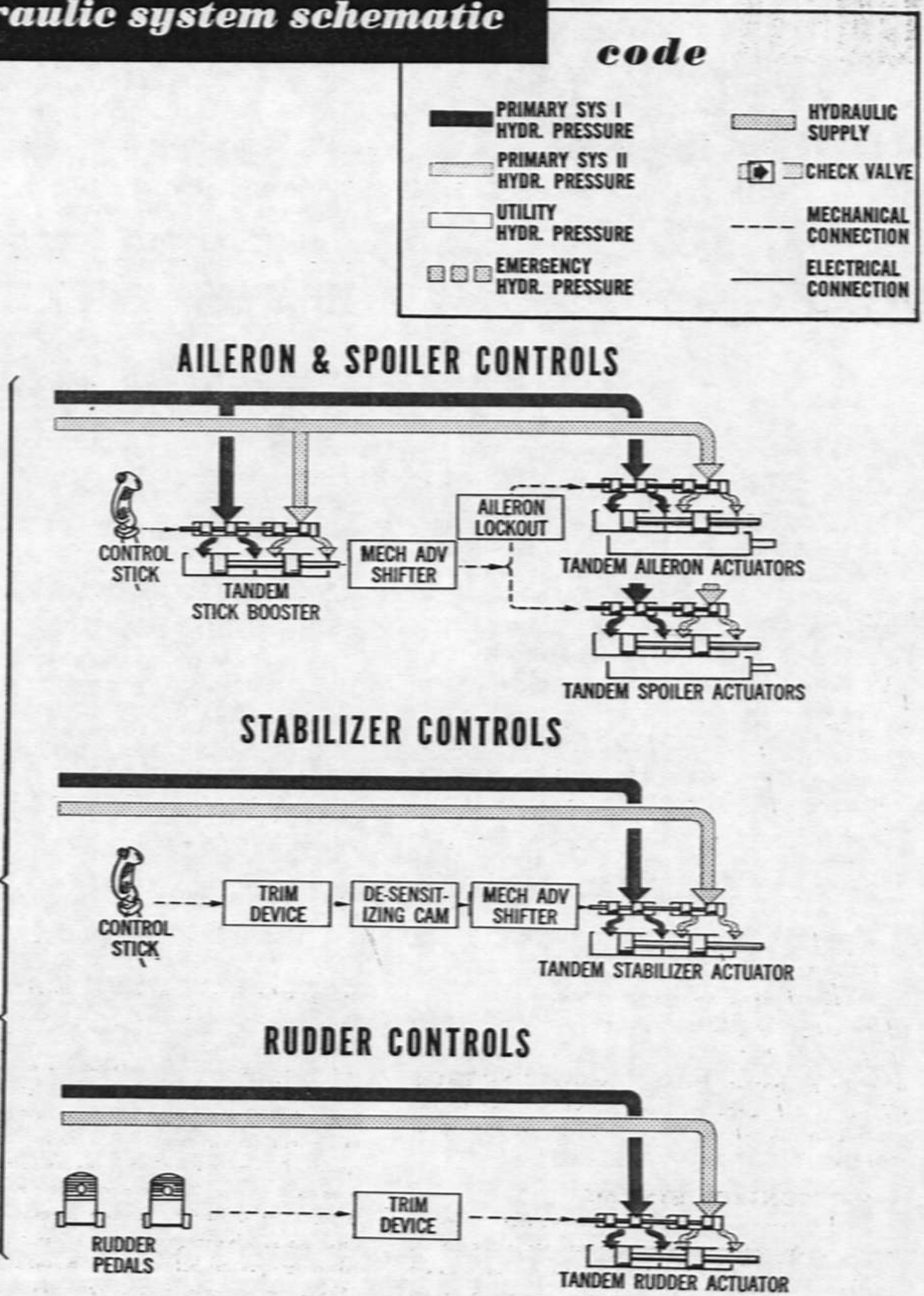


Figure 1–17 (Sheet 2 of 2)

Figure 1—18

AIR TURBINE SWITCH

The air turbine switch (18, figure 1—4) is a two position switch marked AIR TURBINE START and STOP. The START position supplies d-c primary bus power to open a valve in the compressed air line from the engine which supplies compressed air to operate the air turbine. The air turbine powers the utility hydraulic pump and the a-c generator. The OFF position stops the supply of air to the turbine making it inoperative. In the event of d-c primary bus failure the valve in the compressed air line will close and the turbine will be inoperative.

SYSTEM ONE PRESSURE TEST SWITCH

The system one pressure test switch (figure 1—18) is a push button type switch marked SYSTEM ONE PRESS TEST and powered from the d-c primary bus. When the switch is depressed a solenoid operated two position valve in the supply line from the utility to the emergency hydraulic system is closed. This allows the emergency hydraulic pump pressure output to be indicated on the system one pressure gage if the emergency hydraulic system control is in the TURBINE EXTENDED position. Releasing the switch de-energizes the solenoid and the spring-loaded feature of the valve again indexes the utility pressure to the emergency system.

HYDRAULIC PRESSURE GAGE

The hydraulic pressure gage (14, figure 1—3) is powered from the a-c primary bus and is marked HYD FRESS UTILITY. The gage indicates pressure output of the utility hydraulic pump. The scale limits are from 0 to 4000 psi.

FLIGHT CONTROL SYSTEMS

The operation of the primary control surfaces; aileron-spoiler, rudder, and maneuvering stabilizer is controlled by the control stick and rudder pedals in conjunction with cables and linkages. During normal operation, movement of the control stick or rudder pedals position hydraulic control valves which in turn actuate tandem hydraulic actuators which move the primary surfaces. In addition, a hydraulic booster is also provided to move the aileron-spoiler linkage. These tandem actuators and booster are normally powered by the primary hydraulic system with the utility and emergency systems acting as a standby source of power. The actuator systems are irreversible, that is, the air load at the surface will not produce a load at the control stick or rudder pedals. Control feel is simulated by spring-loaded (artificial feel) devices which vary the load felt by the pilot in proportion to the deflection of the control stick or rudder pedals. Roll trim is provided by a switch controlled electric actuator which positions a trim tab located on the left wing trailing edge flap. Trimming of the rudder and maneuvering stabilizer is accomplished by switch-controlled electric actuators which position the respective artificial feel device to a new neutral or "no load" position. A stick damper is provided to limit the rate of control stick deflection on the pitch axis. Each of the primary control systems will be described in its entirety under separate headings.

MANEUVERING STABILIZER

The conventional horizontal and elevators are combined into a one piece maneuvering stabilizer which is designed to give the pilot longitudinal control of the airplane at all altitudes and speeds. Fore and aft stick motion is transmitted to the rear of the airplane by two interconnected sets of cables, one on each side of the fuselage. The cable motion is then transmitted, through mechanical linkage, to a de-sensitizing cam, thru the mechanical advantage shifter to the control valve on the hydrauli actuator which in turn positions the maneuvering stabilizer. Either set of cables in the control linkage is capabl of controlling the stabilizer actuator. The de-sensitizin cam is designed so that during its initial rotation, fro neutral position, a relatively small output is applied the mechanical advantage shifter and transmitted to t stabilizer actuator. As the control stick travel is increase a relatively larger output is produced from the de-se sitizing cam. This occurs during travel in either directi away from the cam neutral point resulting in a less sen tive control stick when within the area immediately fo and aft of neutral. The mechanical advantage shifter provided so that the ratio between control stick trave and stabilizer movement can be varied to satisfy contro movement requirements for various altitudes and airspeeds. The mechanical shift occurs automatically or the automatic control can be overridden and any ratio may be selected manually. During automatic selection, the mechanical advantage ratio of 1:1 occurs at all altitudes when the speed is below 380 mph and at all speeds above 25,000 feet altitude. In this ratio the stabilizer travel is 8 degrees of leading edge up to 24 degrees of leading edge down from a neutral position of −1 degree. The mechanical advantage ratio of 3:1 occurs at all speeds above 380 mph below 13,000 feet altitude. Stabilizer travel is from 2¾ degrees leading edge up to 8½ degrees leading edge down. At all speeds above 380 mph between 13,000 and 25,000 feet altitude the mechanical advantage ratio is 1.7:1. Stabilizer travel is from 4¼ degrees leading edge up to 16¼ degrees leading edge down. A stick damper is provided to limit the rate of travel of the control stick in the pitch axis only. The damper is an independent hydraulic unit which contains its own reservoir and is not connected to the airplane's hydraulic system. The maneuvering stabilizer is normally powered by the number one and two primary hydraulic system. However, in the event of failure of primary system the emergency hydraulic system is manually selected to operate the stabilizer. The utility hydraulic system augments the emergency hydraulic system during emergency operation.

STICK GRIP OVERRIDE SWITCH

The stick grip override switch (figure 1—19) is provided so that trim about the pitch axis can be controlled in the event of failure of the trim switch on the control stick. The switch has four positions; two positions are GRIP TRIM PWR and off, the other PITCH TRIM positions are NOSE DN and NOSE UP. When the switch is positioned to the GRIP TRIM PWR position d-c primary bus power is supplied to the control stick trim switch for control of pitch and roll trim. In the event of failure of the trim switch, which may cause a runaway trim or no trim, the off position is provided so the d-c primary power can be disconnected from the trim switch circuit. In this position both roll and pitch trim circuits are inoperative. The NOSE DN and NOSE UP positions control the pitch trim actuator directly and are used in event of failure of the trim switch to trim the airplane about the pitch axis.

PITCH MECHANICAL ADVANTAGE SWITCH

The pitch mechanical advantage switch (figure 1—18) is provided so that the pilot can select automatic or manual pitch ratio settings as desired. The switch is marked PITCH MECH. ADV with four positions; AUTO, OFF, DEC and INC. With the switch in the AUTO position, mechanical advantage ratio settings of 1:1, 1.7:1 and 3:1 are automatically set in at predetermined conditions of airspeed and altitude. When positoned to OFF all d-c primary power is removed from the ratio changer and the mechanical advantage ratio will remain in any position selected by the pilot. The momentary contact INC or DEC positions are used to manually select any mechanical advantage ratio between the limits of 3:1 and 1:1.

TRIM SWITCH

Airplane trim about the roll and pitch axis is controlled by a five position switch which is spring-loaded to the center or off position. With the stick grip override switch in the GRIP TRIM PWR position the lateral positions of the trim switch adjust the position of the trim tab on the aft edge of the left trailing edge flap thereby trimming the airplane about the roll axis. The fore and aft positions of the switch adjusts the artificial feel unit in the stabilizer control to a no load position. Both trim circuits are energized from the d-c primary bus.

NEUTRAL PITCH TRIM INDICATOR

The neutral pitch trim indicator light (figure 1—18) illuminates when the actuator in the stabilizer artificial feel unit is in the neutral position. The light does not necessarily indicate the neutral position of the stabilizer. It will indicate neutral position of the stabilizer only if there is no fore or aft load on the control stick. The light is marked NEUTRAL PITCH TRIM and is powered from the d-c primary bus.

MECHANICAL ADVANTAGE RATIO INDICATOR

The mechanical advantage ratio indicator (8, figure 1—4) shows the mechanical ratio between the control stick movement and the maneuvering stabilizer travel within the limits of 3:1 and 1:1. The indicator is marked MECH ADVANTAGE SHIFTER RATIO with range markup from 1:1 to 4:1 and is powered by the d-c primary bus.

AILERONS AND SPOILERS

The aileron and spoilers act together to effect satisfactory roll characteristics at all speeds and altitudes. The surfaces are operated by the lateral movements of the control stick through a system of cables and push pull rods. Tandem hydraulic actuators at each surface provide the actuating force. The aileron control is locked out or becomes inoperative automatically at speeds above 475 mph. This system is employed as the combination of aileron and spoiler combination at high speed would over-control the airplane. High sensitivity of the aileron and spoilers at landing speeds established a requirement for a mechanical advantage changer in the system. This changer operates mechanically in conjunction with the extension of the trailing edge flaps. When the trailing edge flaps are fully retracted the mechanical advantage changer is in the 1:1 ratio. At 320-340 mph, 20 degrees of trailing edge flap deflection is available and the mechanical advantage changer shifts from 1:1 to some intermediate position. At 260 mph, the flaps extend fully and the mechanical changer goes to the 2:1 ratio position. Aileron surface travel is 20 degrees up and down from neutral and spoiler surface travel is 61 degrees up from neutral when operating in the 1:1 ratio. The spoilers and ailerons reach full deflection at the same time. A spring capsule is incorporated in the control linkage to act as an artificial feel unit. The ailerons and spoilers are normally powered by the number one and two primary hydraulic system. However, in the event of failure of the primary system the emergency hydraulic system is manually selected to operate the controls. The utility hydraulic system augments the emergency hydraulic system during emergency operation.

TRIM SWITCH

Airplane trim about the roll and pitch axes is controlled by a five position trim switch which is spring-loaded to the center or off position. With the stick grip override switch in the GRIP TRIM PWR position the lateral positions of the trim switch adjusts the position of the trim tab on the aft edge of the left trailing edge flap thereby trimming the airplane about the roll axis. The trim circuit is powered by the d-c primary bus.

RUDDER CONTROL

The rudder is controlled by a system of cables and push-pull rods which incorporate an artificial feel unit. A tandem hydraulic actuator at the rudder is the actuating force. The actuator is normally powered by the number one and number two primary hydraulic system. However, in the event of failure of the primary system the emergency hydraulic system is manually selected to operate the rudder. The utility hydraulic system augments the emergency hydraulic system during emergency operation. The rudder pedals are mounted to a carriage assembly which rides on rails in the cockpit. The carriages move along the rails as the pedals are actuated. A spring-loaded lever on the outboard side of each carriage incorporates a pawl, which engages in notches in a bar, which is an integral part of the rudder cable system. The various notches in the bar provide for individual adjustment of each pedal over a distance of approximately 6½ inches in nine steps. The pedals are suspended at the lower end to allow rotary motion for power brake actuation. An artificial feel unit is incorporated in the control system to give the pilot a sense of control feel by increasing the control forces as the rudder pedals are moved to their extreme positions.

YAW TRIM SWITCH

The yaw trim switch (figure 1–18) is marked YAW TRIM and has three positions; NOSE LEFT, off and NOSE RIGHT spring-loaded to the off position. When held in the NOSE LEFT or NOSE RIGHT position d-c primary bus power is applied to the rudder trim actuator to adjust the artificial feel unit so as to change the neutral position of the spring capsule thereby trimming the airplane about the vertical axis. When the switch is released, all power is removed from the rudder trim actuator and the artificial feel device remains in the selected position.

WING FLAPS

Leading and trailing edge flaps are installed in each wing. Each system will be described individually in the following paragraphs.

LEADING EDGE FLAPS

The leading edge flaps are full span extending from the outboard side of each air duct to the inboard side of the wing tip panels. The flaps are designed to reduce drag during and improve control during high speed maneuvering flight. The flaps are powered by the utility hydraulic power supply system. When hydraulic pressure is applied to the extend side of the actuating cylinder the resulting lateral piston movement is transmitted to three actuators. As the shafts of the actuators are moved outboard, the worm gear within the actuator assemblies convert the lateral movement to rotary movement of the arm assembly which rotates the attached flap downward. The flaps travel through an arc of 20 degrees.

LEADING EDGE FLAP CONTROL

The leading edge flap control (figure 1–8) is marked L.E. FLAPS with three positions UP, NEUTRAL and DOWN. When indexed to DOWN, a circuit to the solenoid control valve is completed to apply utility hydraulic pressure to the extend side of the actuating cylinders which will rotate the flaps down 20 degrees. When placed in NEUTRAL the ports of the solenoid valve are closed and the hydraulic fluid remains locked in the system thus maintaining a fixed flap position. When in the UP position a circuit is completed to apply hydraulic pressure to retract the flap actuators thereby retracting the flaps. The solenoid control valve is powered by the d-c primary bus.

LEADING EDGE FLAP POSITION INDICATOR

The leading edge flap position indicator (9, figure 1–4) shows the position of the leading edge flaps from the limits of travel of 0 to 20 degrees in terms of per cent of travel. The scale is marked UP and DOWN with 4 numbered intermediate positions. The indicator is powered by the d-c primary bus.

TRAILING EDGE FLAPS

The trailing edge flaps are partial span fowler type extending from the inboard end of each wing to the inboard end of the aileron. The flaps have a maximum downward travel of 46 degrees. Each flap is actuated by a d-c primary bus powered electric motor through a flexible shaft which, through a series of drive boxes, transmits the rotary motion to screw jack actuators in the wings. The two flexible drive shafts are interconnected by an additional shaft which connects between the gear boxes. This interconnection insures uniform flap movement, in that slow movement of one flap is transmitted through the shaft and retards the movement of the other flap. A differential of more than 5 degrees between flap travel will de-energize all circuits and prevent further flap movement. Complete stoppage of one flap will stop movement of the other flap. A blow-back feature is incorporated in the system through two pressure switches. When airspeed exceeds 250 mph a pressure switch energizes circuits which retract the flaps approximately half way. When airspeeds exceeds 450 mph a second pressure switch fully retracts the flaps. Incorporated as an integral part of each flap trailing edge is a movable trim tab. The tab on the right flap may be adjusted on installation and is not controllable in flight. The tab on the left wing, however, is electrically actuated by the control stick trim switch and has a maximum travel of 10 degrees above and below the flap chord line.

TRAILING EDGE FLAP CONTROL

The trailing edge flap control (figure 1–8) is marked T.E. FLAPS with three positions UP, NEUTRAL and DOWN. When placed in the DOWN position d-c primary bus power is applied to extend the flaps. The UP position energizes the retracting circuit. The NEUTRAL position de-energizes both circuits and halts flap movement at any intermediate position.

TRAILING EDGE FLAP POSITION INDICATOR

The trailing edge flap position indicator (5, figure 1–4) continually indicates the position of the trailing edge flaps in percentage of extension. The scale is marked UP and DOWN with 4 numbered intermediate positions. The indicator is powered by the d-c primary bus.

SPEED BRAKES

Each of the four hydraulically operated speed brakes consists of a greater segment attached at two hinge points at the aft end of the fuselage. In their closed position, the speed brakes are faired to each other and streamlined to the fuselage contour. Provisions are incorporated to automatically retract or prevent the vertical speed brakes from extending when the landing gear is extended. Mechanical interconnects are not provided for synchronization. The hydraulic system will allow the speed brakes to "blow-back" under excessive air loads. A ground test switch, installed in the aft end of the fuselage, is provided so that all speed brakes can be extended with the airplane in the static position. The speed brakes are powered by utility hydraulic power supply system pressure.

SPEED BRAKE SWITCH

The speed brakes are operated by actuating the three position speed brake switch (figure 1–8) on the throttle grip. The IN position of the switch disconnects d-c primary bus power from the solenoid control valves directing pressure to retract the speed brakes. The NEUT position energizes the control valves to block all ports, enabling the pilot to hold the brakes at any desired intermediate position. The OUT position energizes the solenoid control valves and directs hydraulic pressure to extend the speed brakes. Both control valves are operated simultaneously for actuating the speed brakes when the landing gear is in the retracted position. When the landing gear selector handle is placed in the DOWN position and the uplocks release, the control valve for the vertical pair of speed brakes is automatically de-energized, retracting the vertical brakes. The horizontal brakes hold the position dictated by the speed brake switch. If the landing gear is retracted again, the vertical speed brakes will extend or remain in the retracted position, depending upon the position of the control switch.

LANDING GEAR SYSTEM

The retractable tricycle landing gear consists of three air-oil shock struts together with fairing doors which enclose the struts when the landing gear is retracted. The fairing door remains open when the gear is extended. The main gears retract inboard into the lower surface of the wings while the nose gear retracts aft into the fuselage. The main gear is locked up by hooks which engage rollers on the inboard fairing doors. The nose gear is locked up by a hook engaging a lock on the nose gear strut. The uplock hooks are spring-loaded to the locked position and hydraulically released. In the event of hydraulic failure, the uplocks can be released manually. All gears are locked down by spring-loaded locks which are also unlocked by hydraulic pressure. Inadvertent retraction of the gear when the airplane is on the ground is prevented by a solenoid lock which automatically prevents moving the landing gear selector handle. A switch is provided to override this safety system in emergencies. An emergency system for lowering the gear, in the event of hydraulic failure, is also provided. A hydraulic steering mechanism is incorporated in the nose gear strut which also serves as a shimmy damper when the steering mechanism is not engaged. The main gear wheels are equipped with hydraulically operated disc type brakes. During normal operation the landing gear is retracted and extended with utility hydraulic system pressure. In an emergency, the landing gear uplocks are manually unlocked and all three struts drop to the locked down position by gravity. Landing gear position indicators and a warning horn are provided to inform the pilot of the position of the landing gear struts.

LANDING GROUND SAFETY LOCKS

Ground safety locks are provided for each of the landing gear struts. These locks consist of a pin and warning streamer. The pins, when inserted, prevent the downlocks on the struts from being unlocked. The pins must be removed before flight.

LANDING GEAR SELECTOR HANDLE

The landing gear selector handle (2, figure 1—4) controls the extension and retraction of the landing gear struts. The handle has two position, UP and DOWN, and must be pulled aft in order to move from one position to the other. With the handle in the DOWN position, the hydraulic selector valve is mechanically positioned to supply hydraulic pressure to unlock the landing gear uplocks, extend the gear and power the brakes. The downlocks are spring-loaded and automatically lock when each strut is fully extended. When the weight of the airplane is on the landing gear struts, and the selector handle is in the DOWN position, a locking solenoid is de-energized preventing the selector handle from being moved from the DOWN position. The solenoid is energized by d-c primary power when the airplane is airborne, if the main gear struts are fully extended or the emergency release switch is actuated. This allows the selector handle to be moved to the UP position. The UP position supplies hydraulic pressure to unlock to downlocks and retract the landing gear strut. The uplocks are spring-loaded and automatically lock when the struts are fully retracted. Hydraulic pressure is maintained in the landing gear system when extended or retracted.

EMERGENCY LANDING GEAR RELEASE SWITCH

The emergency landing gear release switch (1, figure 1—4) is provided so that the landing gear can be collapsed while the airplane is still on the ground during take-off or landing emergencies. The switch is marked EMER RELEASE OF L.G. HANDLE and is protected by a cover type guard. If it becomes necessary to retract the landing gear and the airplane is not airborne, the cover guard is lifted and the switch actuated. The landing gear selector handle can then be moved to the UP position.

CAUTION

If the main landing gear shock struts are overinflated so as to be fully extended it will be possible to move the landing gear selector handle to the UP position with the airplane on the ground.

LANDING GEAR EMERGENCY EXTENSION HANDLE

The landing gear emergency extension handle (4, figure 1—4) is used to extend the landing gear in the event of utility hydraulic pressure failure. The handle is marked PULL FOR EMER L.G. EXT. and when pulled aft, releases the main and nose gear uplocks by means of a cable control. Both the main and nose gear struts will extend by gravity and the spring-loaded downlocks will automatically engage.

LANDING GEAR POSITION INDICATORS

Three landing gear position indicators (11, figure 1—4) are marked LEFT, NOSE and RIGHT. When the respective gear is locked down the outline of a wheel appears on the indicator. If the gear is in any position between locked down or locked up, alternate red and white stripes appear and when the gear is retracted the word UP appears on the indicator. The indicators are powered from the d-c primary bus.

Note

The alternate red and white stripes appear on the indicators whenever the d-c primary bus is not energized.

LANDING GEAR WARNING LIGHT

The landing gear warning light incorporated with the landing gear selector handle is a red light illuminated by d-c primary bus power when any gear is in any unlocked condition. The light also illuminates if the landing gear is up and locked when the throttle is retarded below minimum cruising rpm.

LANDING GEAR WARNING HORN

The landing gear warning horn will sound if the throttle is retarded below minimum cruise rpm and the landing gear is not down and locked. The horn can be silenced by depressing the horn silence switch. However, if the throttle is advanced and then retarded the horn will blow again if the d-c primary bus is energized.

LANDING GEAR WARNING HORN SILENCE SWITCH

The landing gear warning horn silence (figure 1—8) is a push button type switch marked L.G. HORN SILENCE. The warning horn can be silenced by depressing the silence switch momentarily. If the horn has been silenced and the throttle is opened above the minimum cruise then closed again, the horn will start to sound again.

LANDING GEAR WARNING LIGHT AND HORN TEST SWITCH

The landing gear warning light and horn test switch (7 figure 1—4) is a push button type switch marked TEST L.G. HORN HANDLE LT. If the test switch is pushed and held, the red light in the landing gear selector handle will illuminate regardless of the position of the landing gear. This tests the electrical circuit and the lamp in the selector handle. The landing gear warning horn will also sound if the throttle is positioned below the minimum cruise rpm position.

NOSE WHEEL STEERING SYSTEM

A nose wheel steering system, provided for directional control while on the ground is electrically engaged, hydraulically powered and mechanically controlled by positioning the rudder pedals. When the nose wheel is extended the nose wheel steering mechanism is connected to the rudder pedal linkage. The pilot then moves the appropriate rudder pedal for the desired turn and at the same time depresses the switch on the control stick. When the desired turn is obtained, hydraulic pressure is cut off. Hydraulic pressure then assists in shimmy damping. Hydraulic pressure is supplied by the utility hydraulic power supply system while electric power is supplied by the d-c secondary bus.

NOSE WHEEL STEERING SWITCH (RADAR OUT SWITCH)

The nose wheel steering switch on the control stick serves a dual purpose. When the nose wheel shock strut is not in the fully extended position d-c secondary bus power is supplied to the nose wheel steering hydraulic control valve when the switch is depressed. When the weight is off the nose wheel strut the switch reverts to a radar out switch which is described in Section IV.

BRAKES

WHEEL BRAKES

Each main wheel is provided with a hydraulically operated dual disc type brake and a power brake valve when the landing gear selector handle is in the DOWN position, utility system hydraulic pressure is applied to the inlet port of the power brake valves. When the rudder pedals are depressed, the power brake valve supplies metered pressure to the dual disc brakes. The metered pressure is proportional to the rudder pedal deflection. An anti-wheel spin system is provided to prevent rotation of the wheels when the landing gear is retracted. This prevents any debris from cluttering the wheel wells. When the landing gear selector handle is placed in the UP position and the downlocks unlock utility system hydraulic pressure is applied to the brake cylinder to stop wheel rotation. When the uplocks engage, hydraulic pressure is shut-off and the brakes system is de-energized. In the event of hydraulic system failure, emergency brake pressure is supplied by an accumulator. This system is manually selected.

EMERGENCY BRAKE HANDLE

The emergency brake handle (6, figure 1—4) is a manual control marked EMER BRAKE. When the handle is pulled aft a selector valve is positioned to transfer the brake system hydraulic pressure supply from the down side of the landing gear system to the utility system accumulator. Brakes are then applied in the normal manner by positioning the rudder pedals. With the accumulator fully charged enough pressure is available for two to three full brake applications.

DRAG CHUTE

A 20 foot drag chute is installed in a compartment below the rudder to reduce landing roll, permitting the use of shorter runways and to serve as an added safety factor in the event of brake system failure or on wet, slippery runways. The drag chute and riser cable is packed in a canvas deployment bag. Deployment of the drag chute is effected by actuation of a switch which electrically releases the door locks and locks the riser cable to the airplane. Wind action pulls the pilot chute and drag chute free of the deployment bag and clear of the airplane. Jettisoning of the drag chute is accomplished by releasing the riser cable from its attaching fitting.

DRAG CHUTE SWITCH

The drag chute switch (figure 1—19) is a two position switch marked EXTEND and off and is guarded in the off position. When the cover guard is lifted and the switch is placed in the EXTEND position, d-c primary bus power is supplied to open the drag chute compartment. The pilot chute is released which, in turn, deploys the drag chute in an inflated condition.

Figure 1—19

DRAG CHUTE JETTISON SWITCH

The drag chute jettison switch (figure 1–19) is a push button type switch marked PUSH TO JETTISON. If depressed, when the drag chute is deployed, the d-c primary bus power is supplied to a release mechanism. The release mechanism uncouples the riser cable from the airplane structure.

INSTRUMENTS – FLIGHT

The turn and bank indicator, the radio compass indicator, the flap position indicators, the mechanical advantage ratio indicator, the d-c voltmeter, loadmeter and the pitot heater are operated from the d-c power supply. The engine tachometer and the exhaust gas temperature indicator are self generated electrical instruments which do not require power from the airplane's electrical system. The slaved gyro compass indicator, attitude gyro indicator, the fuel level indicator, the fuel flow indicator and the oil pressure indicator are powered from the a-c power circuit. The accelerometer and the machmeter do not require any electrical power. The pitot static pressures for flight instruments are obtained from a pitot static boom which extends from the nose of the fuselage. The pitot heater is described in Section IV.

AIRSPEED AND MACH NUMBER INDICATOR

The airspeed and mach number indicator (30, figure 1–3) replaces the conventional airspeed indicator and the mach meter. Indicated airspeed is read on the inner dial within a range of 100 to 750 knots. The same pointer is extended to read mach number on the outer dial. The outer dial rotates as altitude changes so that the mach number corresponds to the indicated airspeed reading at all altitudes. The mach number range is from .5 to 2.5. True airspeed is read on the face of the instrument on a counter. The limits of true airspeed indications run from 300 to 1299 knots. A knurled knob, located at the lower right corner of the instrument ring, allows setting the landing speed index pointer to the landing speed for various configurations. The adjustment of the landing speed index can be made in a range of 100 to 200 knots IAS.

SLAVED GYRO MAGNETIC COMPASS

Refer to Navigation Equipment in Section IV.

ATTITUDE GYRO

With the development of high-performance jet aircraft, pilot experience has shown that as speed increases, the available time for reference to flight instruments has been reduced nearly proportionately. The type MM-2 attitude gyro (6, figure 1–3) is provided to satisfy these needs. Pitch and roll attitudes are indicated by motions of a universally mounted sphere displayed as the background for a miniature refernce airplane. The Horizon is represented on the sphere as a white dashed line, with the sky indicated by a light grey area above the horizon line, and the earth by a dull black area below the horizon line. Horizontal markings of 5 degre separation on the face of the sphere indicate airplane attitudes up to 85 degrees of climb or dive. Similarly exact bank angles can be read on a semi-circular bank scale located on the upper half of the instrument. The spherical background of the instrument is free to move a full 360 degrees in roll without obstruction, and is capable of multiple rolls without error accumulation. The instrument is capable of errorless performance in all pitch altitudes up to ±85 degrees. It may be looped at rates normally encountered in jet aircraft with minimum errors.

STANDBY COMPASS

A conventional magnetic compass (31, figure 1–3) is furnished for navigation in the event of instrument or electrical system failure.

EMERGENCY EQUIPMENT

ENGINE FIRE WARNING SYSTEM

The engine fire warning system consists of a bank of nine normally open thermal switches, a warning light (32, figure 1–3) and a test switch. Seven of the thermal switches are located in the engine compartment and the other two are located in the cooling ducts on the engine. If temperatures reach the predetermined setting of the thermal switches the warning light will burn steadily if the d-c primary bus is energized. The warning light is used for both the fire warning and overheat systems and is marked STEADY-FIRE, and BLINKING-OVERHEAT. The press-to-test switch marked FIRE is actuated to check the fire warning thermal switch circuit.

ENGINE OVERHEAT WARNING SYSTEM

The engine overheat warning system consists of a bank of 15 normally open thermal switches, the fire warning light (32, figure 1–3) and a test switch. The thermal switches are located in the aft fuselage. When the temperature increases to the thermal-switch setting, the switches will close and the warning light will blink. The press-to-test switch, marked OVERHEAT, is actuated to check the overheat warning thermal switch circuit. The warning light will blink if the circuit is intact.

WINDSHIELD

The windshield consists of three transparent panels, set in rubber and mounted in an aluminum frame. The center panel is laminated bullet resistant glass with an electrical element incorporated between the laminations for de-icing and de-fogging. Each side panel is made up of two plastic panels with an air space between them for defogging. Heated and dried air is circulated between the two layers and is controllable from the cockpit. A defroster system, provided with a shut-off, directs hot air from the engine compressor to a perforated tube that directs the air over the inside of each side panel. For complete description of the defroster system see Section IV.

CANOPY

The clam shell type canopy is made up of two plastic panels with an air space between them. Heated and dried air, controllable from the cockpit is circulated between the two layers to defog the canopy. Power is not required to operate the canopy as it is manually opened and closed. The canopy is hinged to the fuselage and when opened moves up and aft in an arc so as to clear the pilot. It is counter-balanced with a spring and opening or closing is accomplished with a minimum of effort. The canopy is jettisoned for emergency exit by means of a thruster. The thruster is actuated by an initiator which is fired when the leg braces on the pilot's seat are raised or by actuation of the canopy ejection control. Canopy jettison is dependent upon the air flow for its separation from the airplane. In the closed position, the canopy is sealed to the cockpit structure by a rubber tube that is automatically inflated by air pressure from the engine compressor.

CAUTION

Do not rest arms on longerons while the canopy is open as injury will result if the canopy unexpectedly closes.

CANOPY CONTROL LEVER

The canopy control lever is spring-loaded to the aft position to prevent interference with the pilot. To open the canopy, the control lever is rotated forward then raised upward. This operation unlocks the canopy and pops it up and back with sufficient force to lock it in the open position. To close the canopy the uplock release latch is pulled back to release the canopy uplock and the canopy is then manually pulled down to the closed position. The canopy control lever is then rotated downward thus locking the canopy in the closed position. The canopy control lever will then automatically rotate to the aft position. When the canopy is jettisoned, the mechanical locks between the canopy and cockpit structure are automatically opened.

EXTERNAL CANOPY CONTROLS

The canopy can be opened or closed externally from either side of the fuselage. To open the canopy from the outside, the upper end of the flush mounted external canopy control on either side of the fuselage is pushed in until the control can be grasped. Pulling the handle up unlocks and opens the canopy. The canopy is closed in the reverse order.

CANOPY JETTISON CONTROL (SEAT LEG BRACE)

The right and left leg brace on the pilot's seat (figure 1–20) are interconnected and raise simultaneously. When the leg braces are raised the canopy jettison initiator is fired which unlocks the canopy and raises it into the airstream. The air flow rotates the canopy to the full open position and then separates it from the airplane. The seat ejection trigger on each leg brace is then exposed for seat ejection. A safety pin, with a red streamer attached, is inserted in the right leg brace to prevent inadvertent actuation of the leg brace during ground operaion.

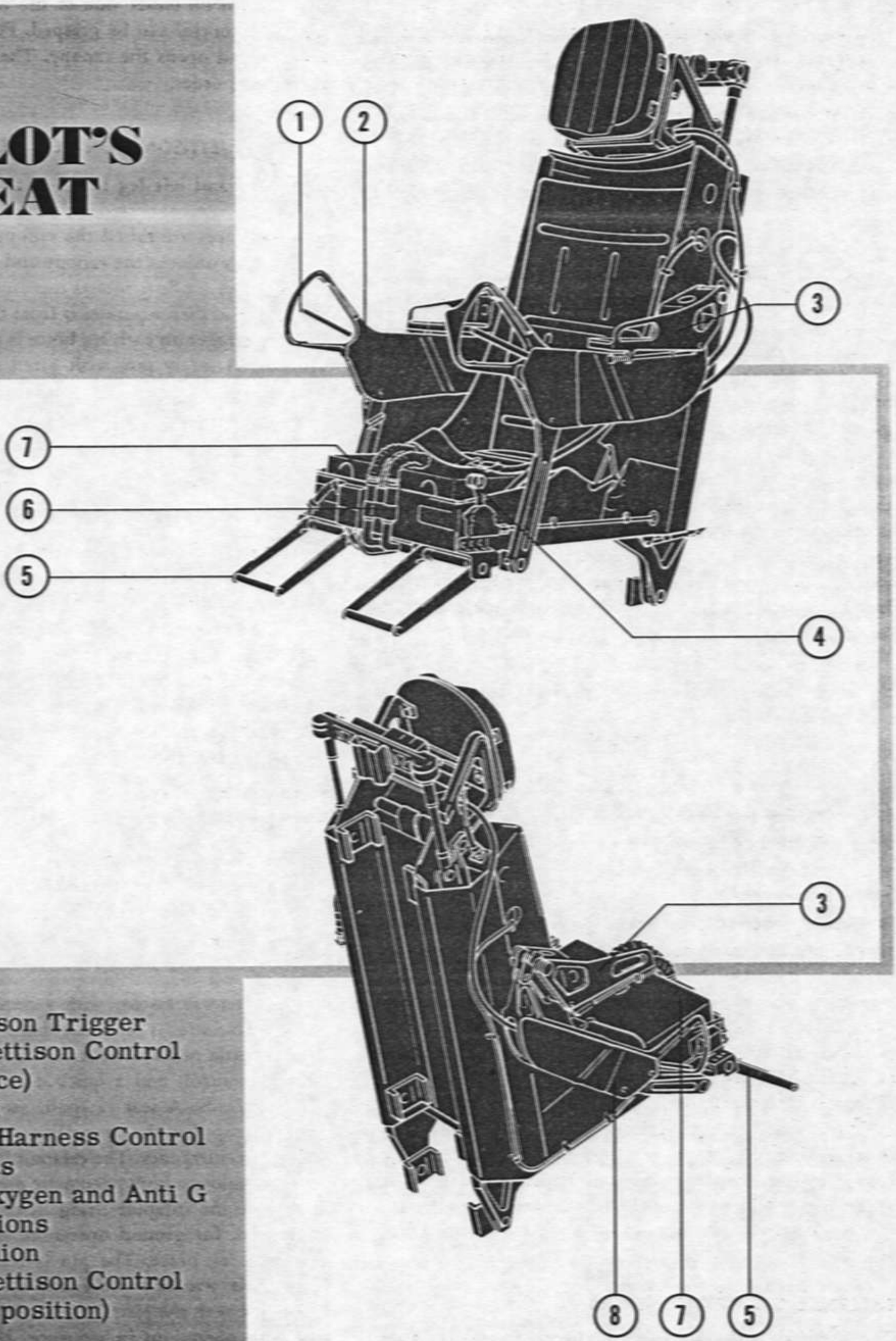
CANOPY JETTISON CONTROL (INTERNAL)

If the canopy is to be jettisoned without seat ejection, the canopy jettison control (5, figure 1–5) is pulled aft. This fires an initiator which unlocks the canopy and raises it into the air stream which separates it from the airplane. The canopy jettison control is installed in a recessed panel to prevent inadvertent actuation. Jettisoning the canopy in this manner eliminates the danger of ejecting the seat as the leg braces are still in the down position and the seat trigger is not exposed.

EJECTION SEAT

The pilot's seat (figure 1–20) is an upward ejection, catapult type, designed to eject the pilot clear of the aircraft in an emergency. The seat incorporates the following features; a false back, removable when a back pack parachute is used, a back pack parachute support, an armor plated headrest, provisions for electrical adjustment of seat height, adjustable armrests, an automatic lap belt release system with accommodations for automatic parachute, a shoulder harness with multi-directional inertia reel and controls, canopy jettison and seat ejection controls and a quick disconnect for the pilot's personal leads. A seat catapult, attached to the rear of the seat, supplies the propelling force to eject the seat during an emergency. The catapult is fired by an initiator which develops a high pneumatic pressure that is directed to actuate the catapult firing mechanism. The initiator is safetied, for ground operation, by a pin installed in the right leg brace. The pin is removed before flight. Since it is standard procedure to jettison the canopy before the seat, the ejection controls are designed to control both operations in sequence. However, in case the canopy fails to jettison, it is possible to eject the seat and pilot through the canopy without injury to the pilot.

PILOT'S SEAT

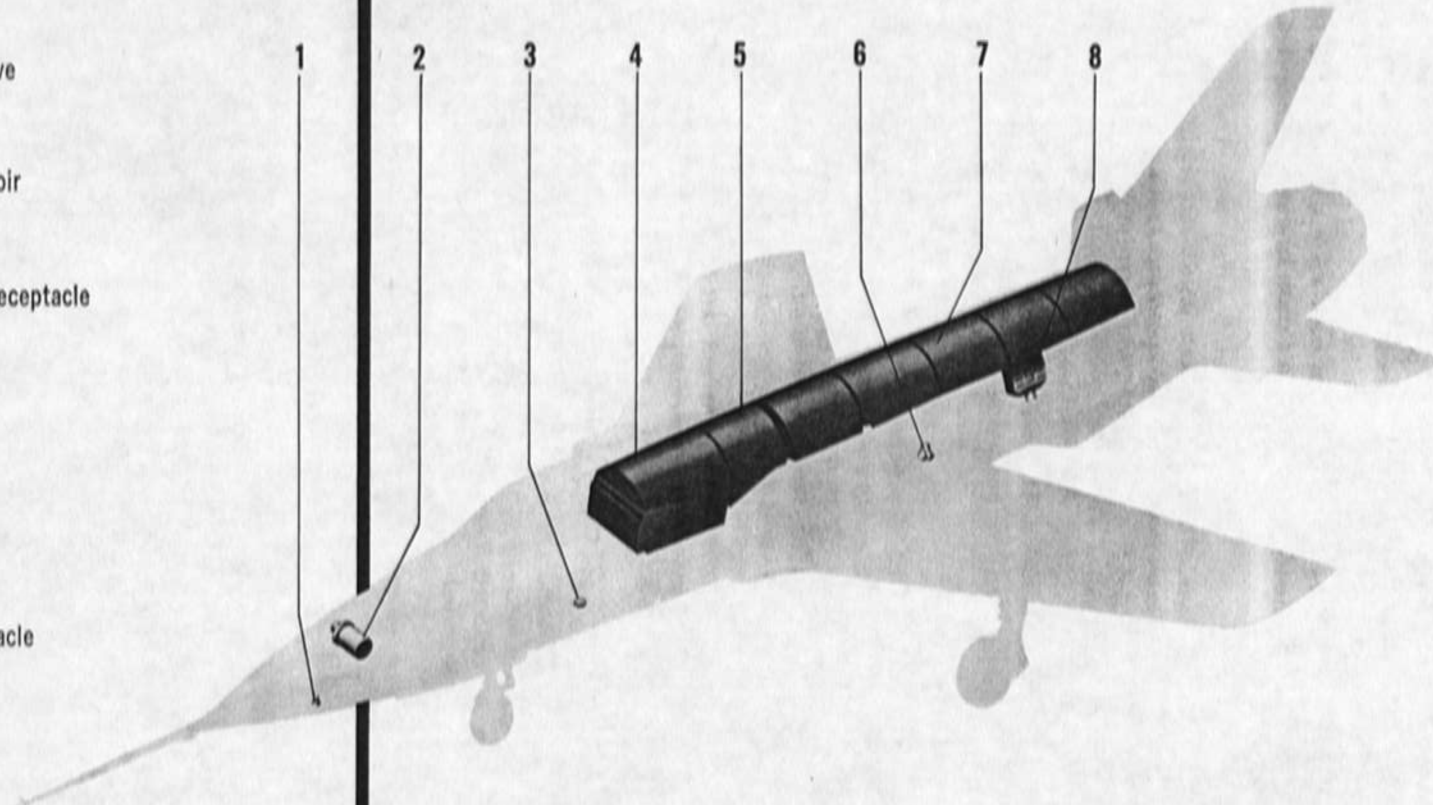


1 Seat Jettison Trigger
2 Canopy Jettison Control (leg brace)
3 Arm Rest
4 Shoulder Harness Control
5 Foot Rests
6 Radio, Oxygen and Anti G Connections
7 Seat Cushion
8 Canopy Jettison Control (stowed position)

Figure 1–20

SERVICING DIAGRAM

1. Oxygen Filler Valve
2. Hydraulic Reservoir
3. External Power Receptacle
4. Fwd. Fuel Tank
5. Main Fuel Tank
6. Refueling Receptacle
7. Aft Fuel Tanks
8. Oil Tank



specifications

fuel recommended	MIL-F-5624B Grade JP-4
oil	MIL-L-7808B-1
hydraulic fluid	MIL-O-5606
oxygen liquid	MIL-O-8069 Grade B Type II

Figure 1—21

AUTOMATIC SAFETY BELT

The ejection seat is equipped with a type M-1 automatic opening safety belt which facilitates pilot separation from the seat following ejection. Belt opening is accomplished as part of the ejection sequence and requires no additional effort on the part of the pilot. In operation, an initiator fires when the seat is ejected supplying gas pressure to the special belt disconnect thus releasing the belt and shoulder harness approximately 2 seconds after the seat fires. The time delay feature insures that the belt will not release until the pilot is entirely clear of the airplane. A key is inserted in the seat belt buckle automatic release mechanism when locking the buckle. This key is retained in the buckle when the automatic features operate, and pops out when the belt is opened manually. Parachutes incorporating automatic rip cord release mechanism may be used with the automatic seat belts. Manual operation of the system can override the automatic features at any time. For example, it is possible to manually open the lap belt even though the initiator action has started. However, it must be remembered that if the belt is opened manually, the parachute rip cord must be pulled manually. The parachute automatic feature may likewise be overridden by manual operation, even though the automatic parachute rip cord has been actuated.

VERTICAL ADJUSTMENT CONTROL

The vertical adjustment control (figure 1–20) is a three position switch spring-loaded to the off or center position. When the switch is moved to the up position d-c secondary bus power is supplied to actuate an electric motor and screwjack mechanism to raise the seat. The drive mechanism is self-locking and will remain in any selected position when the switch is released. Placing the switch in the down position lowers the seat. Over all travel of the seat is 5-1/4 inches.

SHOULDER HARNESS LOCK CONTROL

A two position (lock-unlocked) shoulder harness inertia reel lock control (4, figure 1–20) is located on the left side of the pilot's seat. When the control is in the unlocked position (full aft), the reel harness cable will extend to allow the pilot to lean forward in the cockpit; however, the reel harness cable will automatically lock whenever loads in excess of 2 to 3 g's are encountered. When the reel is locked in this manner, it will remain locked until the control is moved to the locked and then returned to the unlocked position. If the control is in the locked position (full forward) while the pilot is leaning forward, as he straightens up, the harness will retract with him, moving into successive locked positions as he moves back against the seat. The locked position is used when a crash landing is anticipated. This position provides an added safety precaution over and above that of the automatic safety lock. During the seat ejection sequence, a cam on the left leg brace indexes the control to the locked position as soon as either leg brace is raised to initiate canopy jettison.

ARMRESTS

The two armrests (3, figure 1–20) may be raised to the horizontal position or lowered to afford better access to the console controls. A release button is provided beneath the front edge of the armrest. When the release button is depressed the locking mechanism is released allowing the armrests to be moved. The armrests will both move to the raised position when either leg brace is raised for canopy jettisoning.

LEG BRACES

The right and left leg braces are interconnected to act simultaneously and each brace incorporates a seat trigger. During normal operation the leg braces are folded down and secured during ground operation only, by a safety pin inserted in the right leg brace. When either leg brace is raised, the canopy jettison initiator is fired which jettisons the canopy. At the same time the shoulder harness inertia reel is indexed to the locked position and the armrests are raised to the horizontal position. When the leg braces are in the up position a seat trigger is exposed in each leg brace. Squeezing either the right or left trigger fires the corresponding initiator which produces gas pressure to fire the seat catapult.

AUXILIARY EQUIPMENT

Section IV of this Handbook contains information on the following auxiliary equipment: oxygen, cockpit pressurization heating and ventilating, communications, air refueling, anti-g, de-icing, de-fogging, lighting and autopilot.

SECTION II

NORMAL PROCEDURES

TABLE OF CONTENTS

FLIGHT RESTRICTIONS

Refer to Section V of this Handbook for flight restrictions on the aircraft.

FLIGHT PLANNING

Determine cruise control data such as power settings, airspeeds, etc. from data in the Appendix of this Handbook.

Determine the turbine discharge pressure range for the engine under static conditions during the preflight engine check. Refer to figure.

Determine the turbine discharge pressure ratio for a constant thrust climb.

WEIGHT AND BALANCE

Check take-off and anticipated landing gross weight and balance. Consult Handbook of Weight and Balance Data, T.O. No. 1-1B-40 for loading procedure. Make sure weight and balance clearance (Form F) is satisfactory. Check that total weight of fuel, oil, armament, oxygen and special equipment carried is suitable to the mission to be performed. Refer to Section V for weight limitations for various configurations.

ENTRANCE TO AIRPLANE

1. Place utility ladder against either side of the airplane at the cockpit. No external grips or steps are provided.
2. Open canopy using the external canopy control.

Note

If the canopy is open the ladder may be hooked over the cockpit rail on either side of the airplane.

Exterior Inspection

1 Nose Section

b. Nose gear safety pin – Removed.

a. Nose wheel shock strut – Visible damage and proper inflation.

b. Nose gear safety pin – Removed.

c. Nose wheel tire – Proper inflation and evidence of slippage.

d. Nose compartment access doors – Secured.

2 Forward Fuselage and Right Wing

e. Landing gear safety lock – Removed.

a. All access doors – Installed.

b. Air duct plug – Removed.

c. Landing gear strut – Visible damage and proper inflation.

d. Landing gear tire – Blisters, grease or oil, proper inflation and evidence of leakage.

e. Landing gear safety lock – Removed.

f. Check primary hydraulic reservoir.

g. Wheel chock in place.

h. Leading edge flaps – Condition.

3 Right Wing Tip

a. Position lights.

b. Wing tip – Condition.

4 Right Trailing Edge and Aft Fuselage

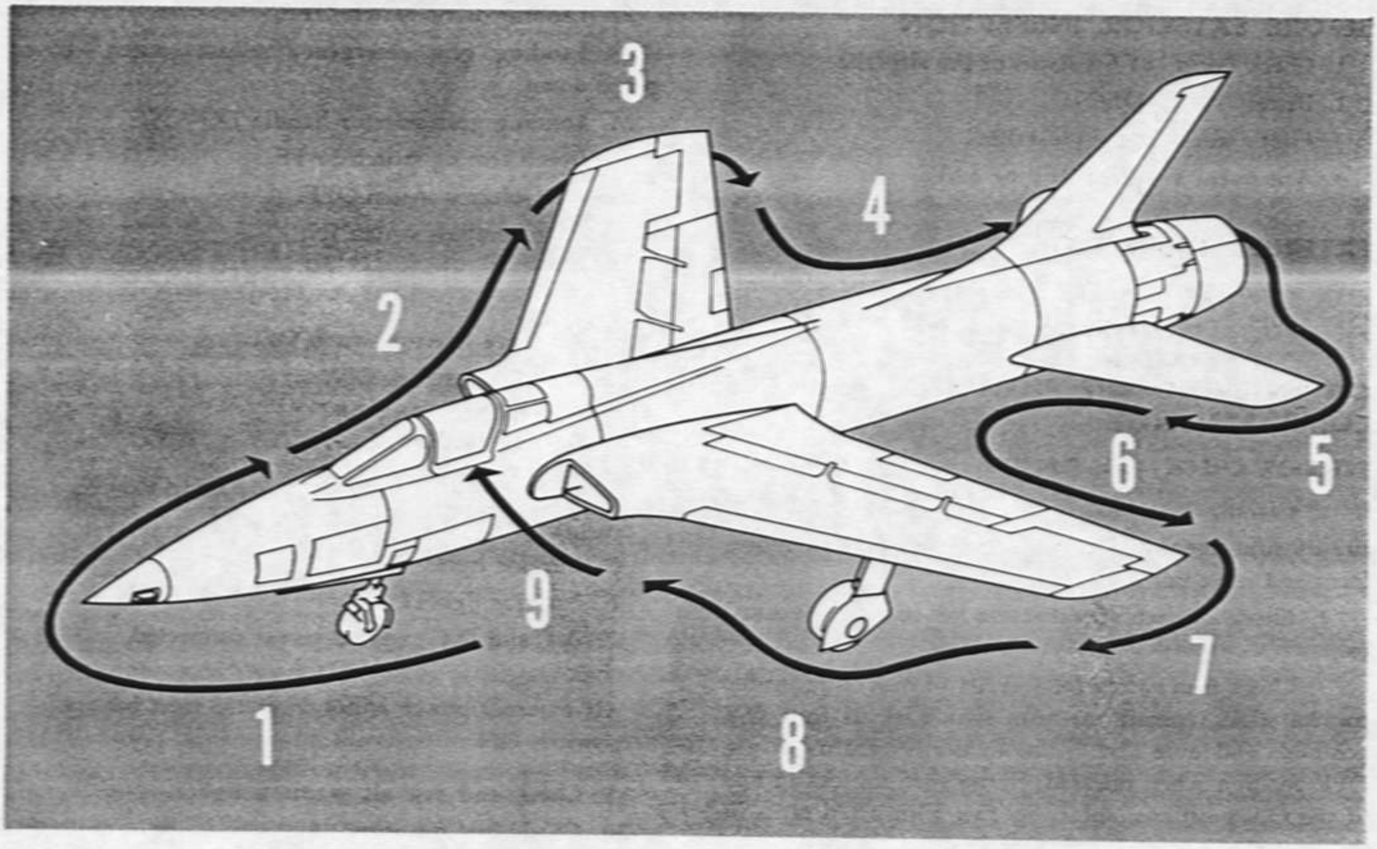
a. Ailerons – Condition.

b. Trailing edge flaps – Condition.

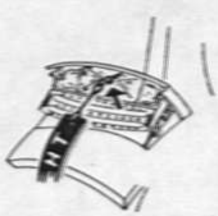
c. Engine access doors – Installed.

d. Bomb bay doors – Condition.

Figure 2–1 (Sheet 1 of 2)



5 *Empennage*



e. Drag chute safety pin removed and compartment secured.

a. Tail pipe dust plug – Removed.

b. Tailpipe for accumulation of fuel or oil.

c. Empennage for general condition.

d. Speed brakes – Condition.

e. Drag chute safety pin removed and compartment secured.

f. Evidence of hydraulic leaks.

g. Tail lights – Condition.

6 *Aft Fuselage and Left Wing Trailing Edge*

a. Make same inspection as in step 4 in reverse order.

7 *Left Wing Tip*

a. Make same inspections as in step 3 in reverse order.

b. Pitot cover – Removed.

8 *Left Wing and Forward Fuselage*

a. Make same inspections as in step 2 in reverse order.

9 *Aft Canopy*

a. Check engine oil supply.

b. Check hydraulic oil supply.

c. Canopy and canopy frame – Cleanliness and security.

d. Canopy for cracks, nicks, crazing and security.

Figure 2–1 (Sheet 2 of 2)

BEFORE EXTERIOR INSPECTION

1. Check Form 781 for status of the airplane.
2. Battery switch OFF.
3. Pilot's seat safety pin installed.
4. Armament switches OFF or SAFE.

EXTERIOR INSPECTION

Make a complete preflight inspection of the airplane. (See figure 2–1.)

ON ENTERING THE PILOT'S COMPARTMENT

INTERIOR CHECK – ALL FLIGHTS

1. Examine safety belt and shoulder harness for security of adjustment and proper operation of lock movement and leave unlocked.
2. Check all circuit breakers on left console – Pushed in.
3. Check trim switch for security of mounting on the control stick. Operate the trim switch in all four ON positions and note that it automatically returns to the OFF position when released. If the switch sticks in any of the ON positions, enter this fact with a red cross in the Form 781 and do not fly the airplane.

CAUTION

Do not twist the grip as such action may cause the grip to become less secure.

4. Windshield side panel defroster OFF.
5. Defogging switch OFF.
6. Temperature and pressure control OFF.
7. Cockpit pressure switch NORMAL PRESS.
8. Pitot heater switch OFF.
9. Jettison belly tank switch – Cover guard down.
10. Fuel system selector switch NORMAL.
11. Jettison wing tank switch – Cover guard down.
12. Fuel system selector OFF.
13. Emergency hydraulic system control – TURBINE RETRACTED.
14. Pitch mechanical advantage switch OFF.
15. Communications radio set OFF.
16. Trailing edge flap control UP.
17. Throttle control OFF.
18. Speed brake switch in same position as speed brakes.
19. Leading edge flap control UP.
20. Drag chute switch – cover guard down.
21. Taxi light switch OFF.
22. Landing light switch OFF.
23. Landing gear emergency release switch – cover guard down.
24. Landing gear selector handle DOWN.
25. Bomb arming switch SAFE.
26. Gun selector switch OFF.
27. Gyro compass switch IN.
28. Set clock, altimeter and accelerometer to correct setting.
29. Sequence selector switch OFF.
30. A-C power switch ON.
31. Generator switch ON.
32. Oxygen supply lever ON.
33. Oxygen diluter lever NORMAL OXYGEN.
34. Oxygen quantity – Check full.
35. Interior lights OFF.
36. Exterior lights OFF.
37. A-C and D-C external power connected.

NOTE

If external power is not available turn battery switch ON.

38. Check and test all warning lights.
39. Check fuel quantity.
40. Remove safety pin from seat handgrip and stow.
41. Connect oxygen, g-suit and radio connections.
42. Head airplane into the wind if practicable.

ENGINE OPERATION

BEFORE STARTING THE ENGINE

Before starting the engine, make sure that the wheels are securely chocked and that danger areas fore and aft of the airplane are clear of personnel, aircraft and vehicles.

WARNING

Suction at the intake duct is sufficient to kill or severely injure personnel drawn into or pulled suddenly against the duct. Danger aft of the airplane is created by the high exhaust temperature and blast from the tailpipe.

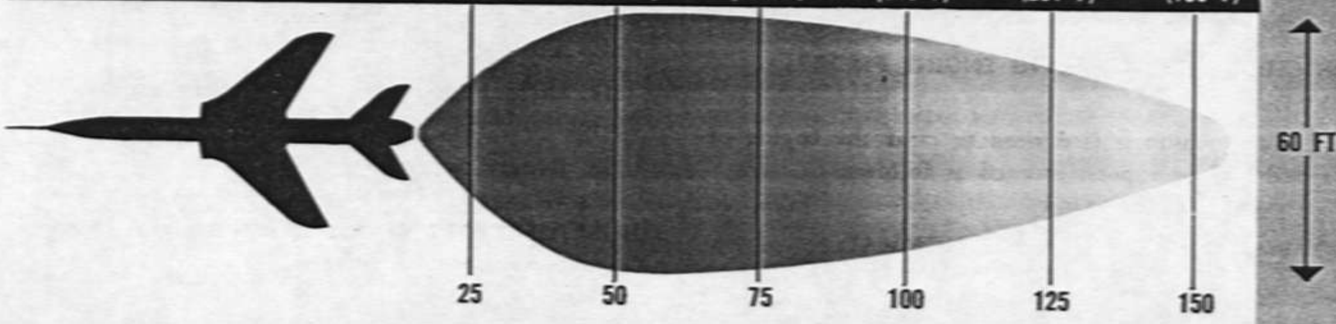
If the airplane is to be operated under possible conditions of carbon monoxide contamination, such as during runup or taxiing directly behind another operating jet aircraft or during runup with the tail into the wind the following procedure will be used:

1. Before starting engine, don oxygen mask, connect hose to oxygen regulator, and place diluter lever in the 100% OXYGEN position.

DANGER AREAS

ENGINE EXHAUST VELOCITY & TEMPERATURE

MILITARY POWER	VELOCITY TEMPERATURE	504 MPH 169°C (337°F)	252 MPH 102°C (216°F)	132 MPH 68°C (154°F)	65 MPH 41°C (106°F)	14 MPH 27°C (80°F)	0 MPH 15°C (59°F)
MAXIMUM POWER (Afterburner Take-off)	VELOCITY TEMPERATURE	990 MPH 630°C (1166°F)	450 MPH 347°C (657°F)	269 MPH 250°C (482°F)	188 MPH 176°C (349°F)	113 MPH 122°C (251°F)	55 MPH 74°C (165°F)



DISTANCE FROM NOZZLE — FEET

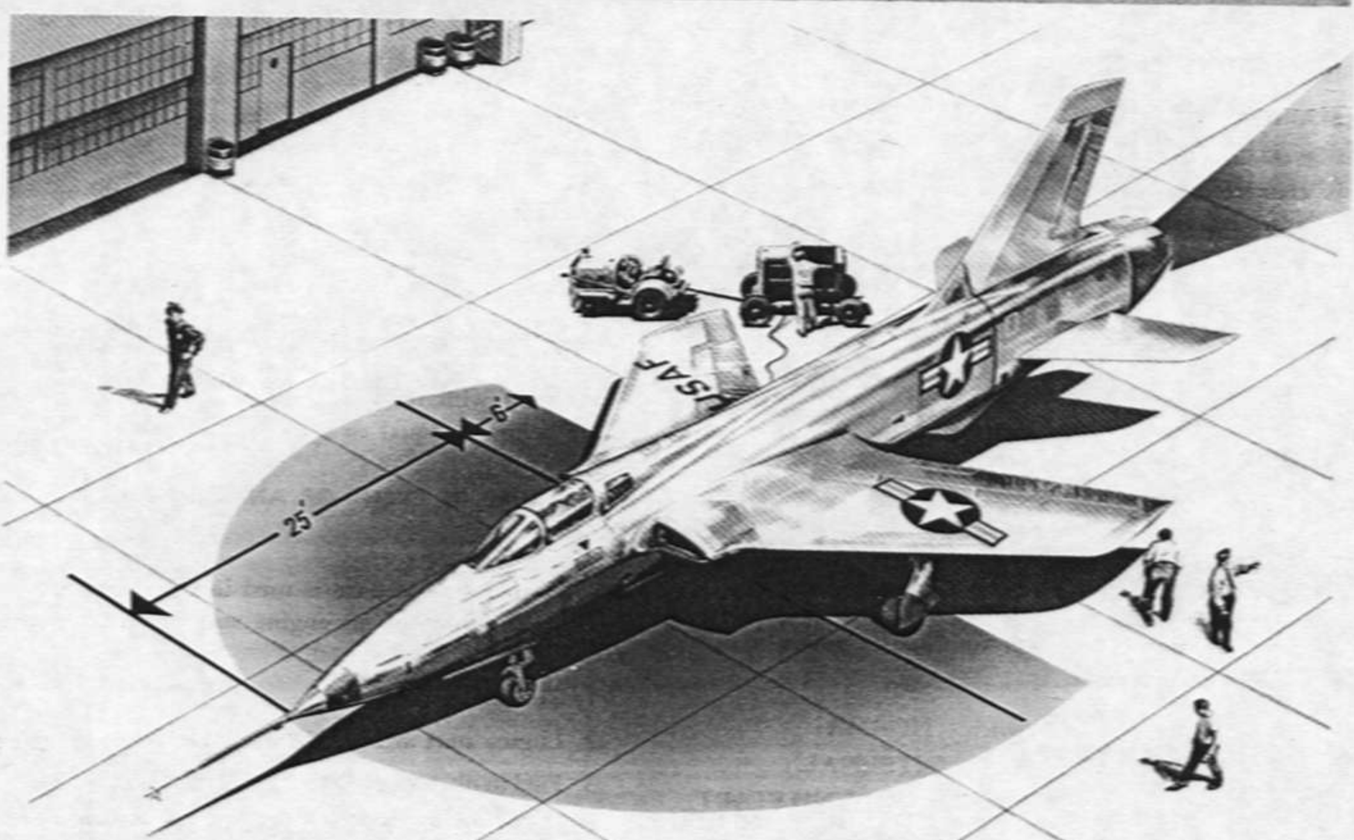


Figure 2–2

2. Whenever contamination is suspected 100 per cent oxygen will be used during ground operation and take-off.

3. After contamination is no longer suspected, place the oxygen regulator diluter lever in the NORMAL OXYGEN position.

WARNING

The oxygen diluter lever must be returned to the NORMAL OXYGEN position as soon as possible because the use of 100 per cent oxygen throughout a long mission will so deplete the oxygen supply as to be hazardous to the pilot.

PROCEDURE FOR CLEARING ENGINE PRIOR TO START

If for any reason it is desired to clear the engine of trapped fuel or vapors, proceed as follows:

1. Throttle control OFF.
2. Fuel system selector switch NORMAL.
3. Fuel tank selector switch – any feed position.
4. Engine start switch–ENGINE GROUND START.
5. Maintain cranking operation for 10 to 20 seconds.
6. Engine start switch STOP START.
7. Fuel tank selector switch to OFF after engine stops rotating.

STARTING ENGINE

1. Make sure proper external compressed air source is connected to starter system and that sufficient air pressure is available for a start.
2. Throttle control OFF.

CAUTION

Do not jockey the throttle control.

3. External power connected – A-C and D-C and battery switch OFF.

Note

If external power is not available place battery switch ON. Fuel pump pressure warning lights will illuminate and stay on until the engine rpm is high enough to supply A-C power to the booster pumps.

4. Fuel system selector switch – NORMAL.
5. Fuel tank selector switch – INTERNAL.
6. Ground start switch–ENGINE GROUND START. Check for oil pressure rise while cranking.
7. Move throttle control to IDLE when tachometer reaches 12-16 per cent rpm. Engine ignition is automatically supplied with throttle movement. Check fuel flow gage for at least 850 pounds per hour flow.
8. Exhaust gas temperature should not exceed 630°C during starting. After ignition, with the throttle at IDLE position, the rpm should increase steadily to 55 to 65 per cent and oil pressure should increase steadily to 30 psi min.
9. Engine light-up should take place within 20 seconds after throttle control is placed in IDLE. Exhaust gas temperature will reduce to or below 340°C at idle after idle rpm is obtained.
10. After idle rpm has been attained position the starter switch to STOP START.
11. Check engine instruments for proper indications, make sure warning lights are not illuminated.
12. Disconnect external power and starter compressed air sources.
13. Battery switch ON.

UNSATISFACTORY STARTS

1. Hot Start – Exhaust gas temperature exceeds starting temperature limit.

Note

A hot start can be anticipated by observing a greater initial starting fuel flow than normal for a given field elevation at the instant that the throttle control is placed in IDLE.

2. Aborted Start – The engine does not light-up within 20 seconds as evidenced by no rise in exhaust gas temperature, no increase in rpm, no sound indicating that fuel has ignited.

3. False Start – After light-up has occurred, the rpm does not increase to that of idle but remains at some lower rpm with the exhaust gas temperature at some value below or equal to the starting limit.

PROCEDURE IN EVENT OF AN UNSATISFACTORY START

The following procedure is used in the event that any of the provisions of an engine start or if an unsatisfactory start is experienced.

1. Throttle control OFF.
2. Engine start switch STOP START.
3. Fuel tank selector switch – OFF.
4. Investigate to find out reasons for difficulty.
5. Allow at least 30 seconds for fuel drainage before another start is attempted.

WARNING

If the throttle control is inadvertently retarded to the OFF position, do not reopen the throttle to regain the light during ground operation. Introducing unburned fuel into the engine may create a fire hazard.

ENGINE GROUND OPERATION

WARM UP

If a start is made with an engine which was cooled to an ambient temperature at −35°C or below it is recommended to leave the throttle control at IDLE for two minutes to warmup the engine.

ACCELERATION

The maximum permissible exhaust gas temperature during an acceleration is 670°C for 2 minutes. If this is exceeded during an acceleration, close the throttle slightly until the temperature has dropped below the steady state limit. The engine may then be accelerated by advancing the throttle control gradually so that the acceleration temperature limit is not exceeded. An immediate investigation should be conducted to determine the cause of exceeding the limiting temperature during the first acceleration and the cause should be eliminated before subsequent fast accelerations are made.

WARNING

If the engine rpm exceeds the acceleration limit the engine shall be immediately stopped and the necessary corrective actuation taken in accordance with the Inspection Requirements Handbook for the aircraft.

GROUND TEST

1. Air turbine switch START.
2. Actuate leading edge flaps through complete cycle. Flaps should extend in approximately 10 seconds and be synchronized within ±1 degree. Check position of flaps with indication on flap position indicator.
3. Actuate trailing edge flaps through complete range. Flaps should extend in approximately 6 seconds and be synchronized within ±1 degrees. Check position of flaps with indication on flap position indicator.
4. Actuate speed brakes through complete range. Speed brakes should open in approximately 3 seconds, but not necessarily synchronized. Position speed brakes in some intermediate position. The brakes should not creep.
5. Check utility hydraulic pressure – 2700 to 3000 psi.
6. Check primary hydraulic pressure gages – Primary system No. 1 and No. 2 – 2700 to 3000 psi.
7. Check aileron, rudder and stabilator for free and correct movement and full travel.
8. Emergency hydraulic system control – TURBINE EXTENDED, then depress system one pressure test switch. Primary system No. 1 pressure gage should register emergency accumulator pressure. This pressure will drop to zero if controls are actuated. Release test switch and return emergency hydraulic system control to TURBINE RETRACTED. Primary system No. 1 pressure gage should return to normal reading.
9. Pitch mechanical advantage switch AUTO. Mechanical advantage shift ratio indicator should read a ratio of 1:1.
10. Check charge on loadmeter.
11. Check D-C and A-C voltage.
12. Engine check.

a. Throttle control IDLE (55-65 per cent rpm).

Note

If a start is made with an engine which was cooled to an ambient temperature of −35°C or below it will be preferable to leave the throttle at IDLE for two minutes to warm up the engine.

b. Exhaust gas temperature 340°C Max.

c. Oil pressure – 30 psi min (40 to 50 psi normal).

d. Master warning light OFF.

e. Maximum permissible exhaust gas temperature during an acceleration is 670°C for two minutes.

Note

If the exhaust gas temperature is exceeded during acceleration, close the throttle slightly until the temperature has dropped below the steady state limit. The engine may then be accelerated by advancing the throttle control gradually so that the acceleration temperature limit is not exceeded. An immediate investigation should be conducted to determine the cause of exceeding the limiting temperature during the first acceleration and the cause should be eliminated before subsequent fast accelerations are made.

WARNING

If the engine rpm exceeds the acceleration limit the engine shall be immediately stopped and the necessary corrective action taken in accordance with the official Inspection Requirements Handbook for the aircraft.

ENGINE CHECK CHART

OPERATIONAL LIMITS

OPERATING CONDITION	ROTOR SPEED	MAXIMUM OBSERVED EXHAUST GAS TEMP. (°C)		TIME LIMITS	OIL PRESSURE	OIL TEMP. (°C)
	High Speed Rotor %	S.L. To 35,000 ft	Above 35,000 ft	(minutes)	(psig) NORMAL	RANGE
Maximum	(2)	630	660	5-15 (1)	45 ± 5	70-120
Military	(2)	620	650	30	45 ± 5	70-120
Normal Rated	(2)	580	610	Continuous	45 ± 5	70-120
Cruise 90% Normal Rated	(2)	540	570	Continuous	45 ± 5	70-120
80% Normal Rated	(2)	500	530	Continuous	45 ± 5	70-120
70% Normal Rated	(2)	470	500	Continuous	45 ± 5	70-120
Idle (3)	55-65	340			30 to 50	70-120
Starting		630	630			
Acceleration		670	670	2	45 ± 5	70-120

Notes

(1) 5 Minutes pertains to take-off and ground operation, 15 minutes are allowed in flight.

(2) For approximate per cent rpm for a given inlet temperature, see Temperature vs Rpm curve (figure 2–4).

(3) If a start is made with an engine which has cooled to an ambient temperature of −35°C or below it is preferable to leave the power lever at IDLE for two minutes to warm up the engine.

Figure 2–3

13. Check emergency system as follows:

a. Throttle control IDLE.

b. Fuel system selector switch EMERGENCY.

c. Emergency idle rpm should be compatible with normal system idle rpm.

d. Throttle control — Full forward or with pressure ratio variation whichever is attained first.

CAUTION

Do not exceed computed pressure ratio or military exhaust gas temperature.

e. If pressure ratio variation governs throttle control travel, sufficient thrust for the ambient conditions will be available.

f. If the throttle control is fully forward, the fuel flow gage should read between 6750 and 7400 pph under sea level static conditions without taking duct loss into consideration. Correct these valves as follows and compare with the observed fuel flow gage reading.

(1) Subtract 100 pph from the above values for each 1000 feet of field elevation above sea level.

(2) Subtract 33⅓ pph from the above valves for each one per cent of static duct loss.

g. Throttle control IDLE.

h. Fuel system selector switch NORMAL.

AFTERBURNER OPERATION

STARTING

1. Extreme caution must be exercised during ground operation of the afterburner to insure that all personnel are clear of the jet wake. The danger zone of the jet wake during afterburner operation is much greater than during normal operation.

2. Afterburner should not be attempted with the fuel pump warning light illuminated.

3. Throttle control 83 per cent military thrust minimum.

4. Actuate afterburner switch to ON by placing the throttle in the outboard position.

WARNING

If for any reason the exhaust nozzle fails to open as soon as afterburning takes place, there will be a rapid rise in exhaust gas temperature and an rpm reduction of approximately 4 per cent. If either of these indications is evident, afterburning should be terminated immediately.

STOPPING

1. Throttle control 83 per cent Military thrust minimum.

2. Afterburners switch to OFF by retarding the throttle. This will automatically de-energize the afterburner pump and close the exhaust nozzle.

Note

At any time, if the throttle control is retarded to any point below 83 per cent thrust position, afterburner will automatically be shut down.

3. Following ground operation of the afterburner, reard throttle control to IDLE position and check to see that proper drainage occurs at the drain valve. Hold at IDLE until drainage stops which may be as long as 2 minutes.

TAXIING INSTRUCTIONS

Remove chocks, close canopy, release brakes and increase power until airplane starts to move. Once the airplane is moving, taxi at lowest practicable rpm to conserve fuel and avoid damage from tailpipe blast. Maintain directional control through steerable nose wheel by use of rudder pedals. Nose wheel steering switch on the control stick grip must be depressed for nose wheel steering.

BEFORE TAKE-OFF

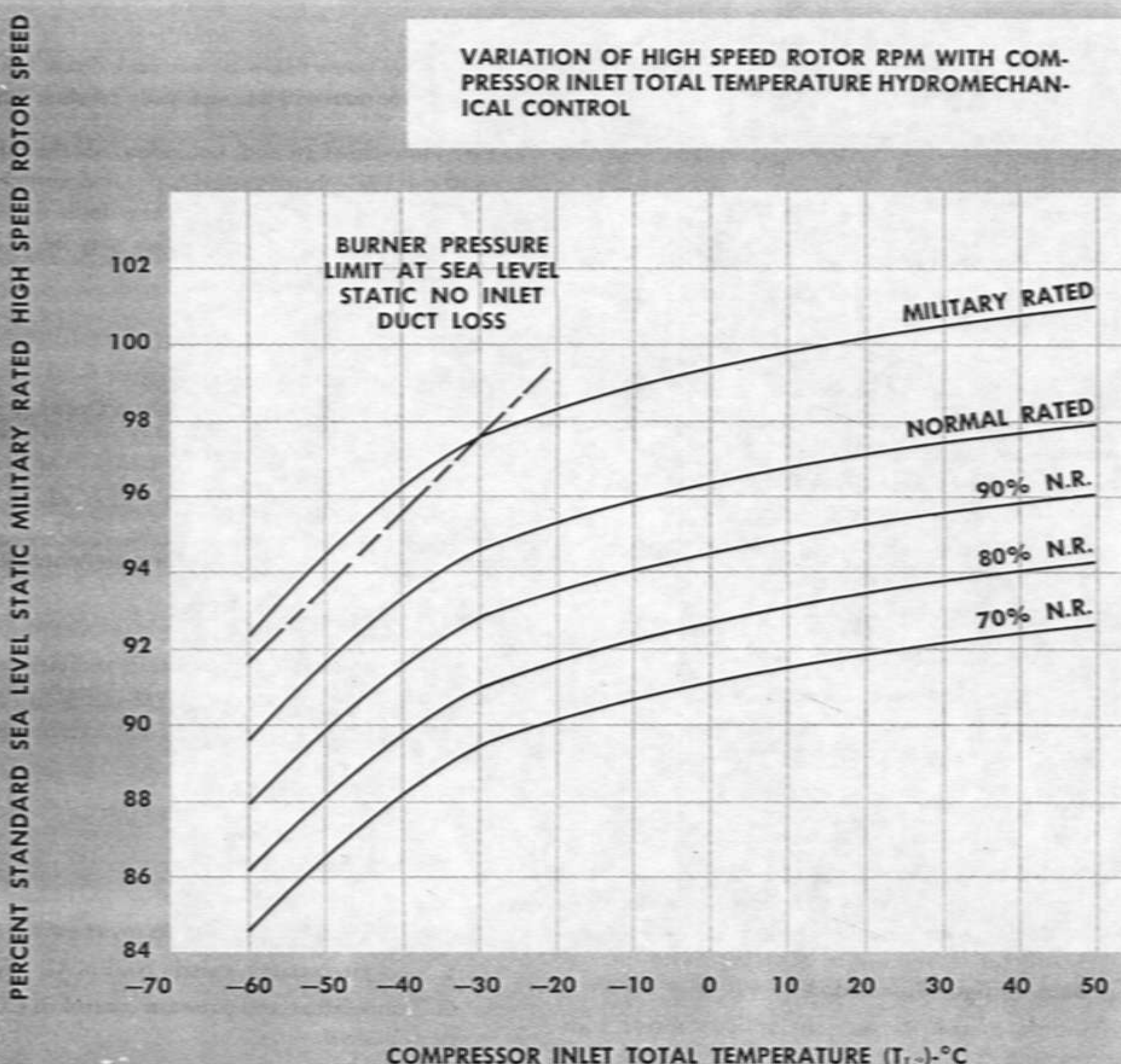
After taxiing into take-off position, complete the following checks.

PREFLIGHT AIRCRAFT CHECK

1. Safety belt and shoulder harness tightened, shoulder harness inertia reel handle unlocked.
2. Pitot heater switch OFF.
3. Windshield de-fogging switch OFF.
4. Windshield side panel defroster switch OFF.
5. Cockpit pressure switch NORMAL PRESS.
6. Temperature and pressure control in COLD HOT range as desired.
7. Emergency hydraulic system control TURBINE RETRACTED.
8. Pitch mechanical advantage switch AUTO.
9. Neutral pitch trim light illuminated.
10. Fuel tank selector switch — in last counter clockwise position depending on fuel loading.
11. Leading edge flaps UP.
12. Trailing edge flaps — Take-off position.
13. Speed brake switch IN.
14. Oxygen mask adjusted properly. Diluter lever NORMAL and supply lever ON.

TEMPERATURE *versus* RPM *curve*

VARIATION OF HIGH SPEED ROTOR RPM WITH COMPRESSOR INLET TOTAL TEMPERATURE HYDROMECHANICAL CONTROL



MAXIMUM HIGH SPEED ROTOR SPEED FOR STANDARD SEA LEVEL STATIC CONDITIONS MAY BE FOUND STAMPED ON THE ENGINE DATA PLATE FOR EACH INDIVIDUAL ENGINE

note

THESE CURVES ARE APPROXIMATE. ACTUAL VALUES MAY VARY SLIGHTLY BETWEEN INDIVIDUAL ENGINES

Figure 2–4

15. Communication equipment ON as required.

16. Check flight controls for free and correct movement.

PREFLIGHT ENGINE CHECK

1. Fuel system selector switch NORMAL.

2. Fuel tank selector switch in last counter clockwise position.

3. Advance throttle to full forward.

4. Oil pressure 40 psi min.

5. Exhaust gas temp 630°C Max if on afterburner or 620°C Max without afterburner.

Note

Exhaust temperatures up to 670°C is allowed for two minutes during the thrust overshoot period.

6. Tachometer – Should read below the red line.

7. Pressure ratio gage – Reading should be within or above the variation as computed under Flight Planning.

8. Return throttle control to IDLE.

TAKE-OFF

1. Refer to the Appendix for minimum take-off distances and speeds required for various combinations of gross weight, pressure altitude and air temperature, also for best climbing speeds, rate of an time to climb and fuel consumption.

2. Advance throttle to full forward position.

3. Release brakes and begin take-off run.

4. If afterburner operation is desired place throttle control in afterburner position.

Note

The pressure ratio gage reading may increase or decrease slightly when the afterburner is turned on.

5. Check exhaust gas temperature gage to remain within limits immediately after afterburner initiation. If the temperature is over 670°C for two minutes or is over 630°C steady state abort take-off.

AFTER TAKE-OFF

1. Landing gear selector handle UP when definitely airborne. Check landing gear unsafe light and position indicators.

2. Trailing edge flaps UP.

3. Before 7000 feet altitude is reached, throttle control must be retarded beyond the point at which the fail safe stops in the fuel control engage. This action allows the fuel control to compensate for altitude increases and also prevents overtemperature and overspeeds.

4. As soon as added thrust is no longer needed, shut-off afterburner.

Note

Maximum time allowance for afterburner operation in flight is 15 minutes and for Military power operation 30 minutes.

5. Adjust throttle and speed for best climb.

CLIMB

If a constant percentage thrust climb is desired, adjust throttle for computed turbine discharge pressure. Once this thrust condition is set up, the fuel control will maintain an approximate constant per cent thrust output with a fixed throttle position.

AFTERBURNER OPERATION IN FLIGHT

STARTING

CAUTION

Afterburning should not be attempted with the fuel pump warning light illuminated.

1. Throttle control – 83 per cent Military thrust minimum.

2. Turn afterburner on by moving throttle control outboard.

WARNING

If for any reason, the exhaust and nozzle fails to open as soon as afterburning takes place, there will be a rapid rise in exhaust gas temperature and rpm reduction of approximately 4 per cent if either of these indications is evident, afterburning should be terminated immediately.

STOPPING

1. Throttle control 83 per cent military thrust minimum.

2. Deactuate afterburner by returning throttle control to the inboard position. This will automatically deenergize the afterburner pump stage and close the exhaust nozzle.

Note

If the throttle control is retarded to a point below 83 per cent military thrust position at any time the afterburner will automatically be shut down.

DURING FLIGHT

1. Determine the compressor discharge pressure ratio for the desired thrust condition for cruising.

2. A maximum exhaust gas temperature of 670°C is permitted for acceleration and is limited for two minutes. If temperatures above 670°C are experienced, a slower throttle lever movement should be used.

3. For a given throttle control position, exhaust gas temperature will increase with altitude.

4. If over temperature cannot be controlled, an immediate landing should be made using the minimum thrust necessary to sustain flight.

5. Throttle control movement must be smooth and continuous. A minimum of one second should be used for transition from IDLE to MILITARY.

FLIGHT CHARACTERISTICS

Refer to Section VI for detailed information on the airplane flight characteristics.

PRE-TRAFFIC PATTERN CHECK LIST

1. Gun selector switch OFF.
2. Sequence selector switch OFF.
3. Fuel selector switch NORMAL (unless EMERG position is necessary for flight.)
4. Check hydraulic pressures – Normal.
5. Safety belt and shoulder harness tightened, shoulder harness inertia reel UNLOCKED.

LANDING APPROACH

1. Enter pattern at knots IAS.
2. Speed brake switch OUT.
3. Landing gear selector handle DOWN when speed has reduced below 243 knots IAS.
4. Leading edge flaps.
5. Trailing edge flaps.

LANDING

NORMAL LANDING

CROSS WIND LANDING

LANDING WITH EXTERNAL LOAD

GO-AROUND

AFTER LANDING

1. Maintain directional control by differential braking.
2. Speed brake switch IN and flaps UP when taxiing speed is reached.
3. Engage nose wheel steering after runway is cleared and when slow taxiing is necessary.

STOPPING THE ENGINE

1. Flight controls neutral.
2. The engine must be allowed to idle for five minutes for cooling purposes prior to shut-down.
3. Throttle control OFF.
4. Turn fuel tank slector switch OFF, after the engine stops rotating.
5. Check that engine decelerates freely.
6. Battery switch OFF.
7. All switches off except the generator switches.

BEFORE LEAVING THE AIRPLANE

1. Chocks in place and landing gear ground safety locks installed.
2. Safety pin installed in pilots seat.
3. Fill out Form 781
4. Close canopy.

SECTION III

EMERGENCY PROCEDURES

TABLE OF CONTENTS

ENGINE FAILURE

Complete engine failure, due to damage within the engine, rarely occurs. In these cases air starts should not be attempted as fire may result. However, engine flame-outs in jet engines can occur and are generally caused by improper fuel scheduling caused by a malfunction of the fuel control system or incorrect operating techniques. Engine instruments often provide indications of fuel control system failures before the engine flame-out. If a flame-out occurs due to fuel control system failure or improper operating techniques an air start can usually be accomplished provided time and attitude permit.

PROCEDURE OF ENCOUNTERING ENGINE FLAME-OUT

1. Throttle control – OFF.
2. Wait 30 seconds then relight as follows.

ENGINE AIR START

Flight relights may be consistently made subject to the conditions of the Flight Relight Chart figure 3–1. This chart defines the conditions of altitude and indicated airspeed which would give the approximate rpm as shown by the lines of constant engine rpm.

1. Throttle control OFF.

2. Tachometer – 12 to 30 per cent rpm as shown in figure 3–1.

3. Fuel control selector switch NORMAL (EMERGENCY if primary fuel system failure is suspected).

4. Fuel tank selector – Positioned to tank with fuel.

5. Air start switch – AIR START.

6. Throttle control to IDLE.

Note

Fuel flow at high altitude will be relatively high even when the throttle control is in the IDLE position. By retarding the throttle below IDLE to an angle between 9 and 5 degrees, the fuel flow can be reduced to a maximum of 300 pph below the normal minimum flow to facilitate high altitude relights.

7. If a satisfactory start is accomplished, the rpm should advance to idle rpm which, at altitude, may be as high as 80 per cent rpm.

8. Set throttle control to desired thrust condition.

9. During the start observe that the exhaust gas temperature stays below the maximum starting limit of 630°C. After the start is attained, idle exhaust gas temperature should settle out to or below 340°C. The oil pressure should be at least 30 psi after idle rpm is attained.

10. In the event that light up does not occur within 20 seconds after the throttle control is advanced to IDLE or if the engine fails to accelerate to idle within approximately 45 seconds after light up, retard the throttle to the OFF position and discontinue the attempted air start. The engine should be allowed to windmill for at least 30 seconds before another start is attempted.

RELIGHT AFTERBURNER

1. Afterburner switch OFF (throttle inboard).
2. Wait 5 seconds.
3. Afterburner switch ON (throttle outboard).

ENGINE FAILURE DURING TAKE-OFF

BEFORE FLYING SPEED IS REACHED

1. Throttle control OFF.
2. Fuel tank selector OFF.
3. Jettison external stores if necessary.
4. Deploy drag chute and brake to a stop on runway.
5. If impossible to stop on runway: Jettison canopy and retract landing gear.
6. Battery switch OFF after gear is retracted.

AFTER LEAVING THE GROUND

Prepare for an emergency landing or if sufficient altitude has been obtained before engine failure, follow emergency bail-out procedure.

1. Landing gear selector handle UP.
2. Jettison external stores if necessary.
3. Throttle OFF.
4. Fuel tank selector OFF.
5. Jettison canopy.
6. Shoulder harness LOCKED.
7. Land straight ahead, changing course only enough to miss obstacles.
8. Deploy drag chute.
9. Battery switch off after touchdown.

ENGINE FAILURE DURING FLIGHT

1. Throttle control OFF.
2. Wait 30 seconds then attempt an air start.
3. If an air start is impossible, make a forced landing or abandon the airplane as conditions dictate.

MAXIMUM GLIDE

Maximum glide distances with a windmilling engine can be obtained by the following procedure.

1. Throttle OFF.
2. Fuel tank selector OFF.
3. Leading edge flaps UP.
4. Trailing edge flaps UP.
5. Speed brakes IN.
6. Landing gear UP.
7. Jettison external stores.
8. Trim airplane to maintain recommended glide speed. (See figure 3–2).

LANDING WITH DEAD ENGINE

FIRE

FIRE WARNING AND OVERHEAT LIGHT BLINKING

Blinking of the overheat warning light indicates an overheat condition or possible fire in the aft section necessitating the following action.

1. Reduce thrust immediately.
2. If light goes out, continue flight at reduced power and land as soon as possible.
3. If light does not go out, indicating a possible fire rather than overheat, check for indications of fire such as smoke in cockpit, engine noise, abnormal exhaust gas temperature or verification from another airplane.

FLIGHT RELIGHT CHART

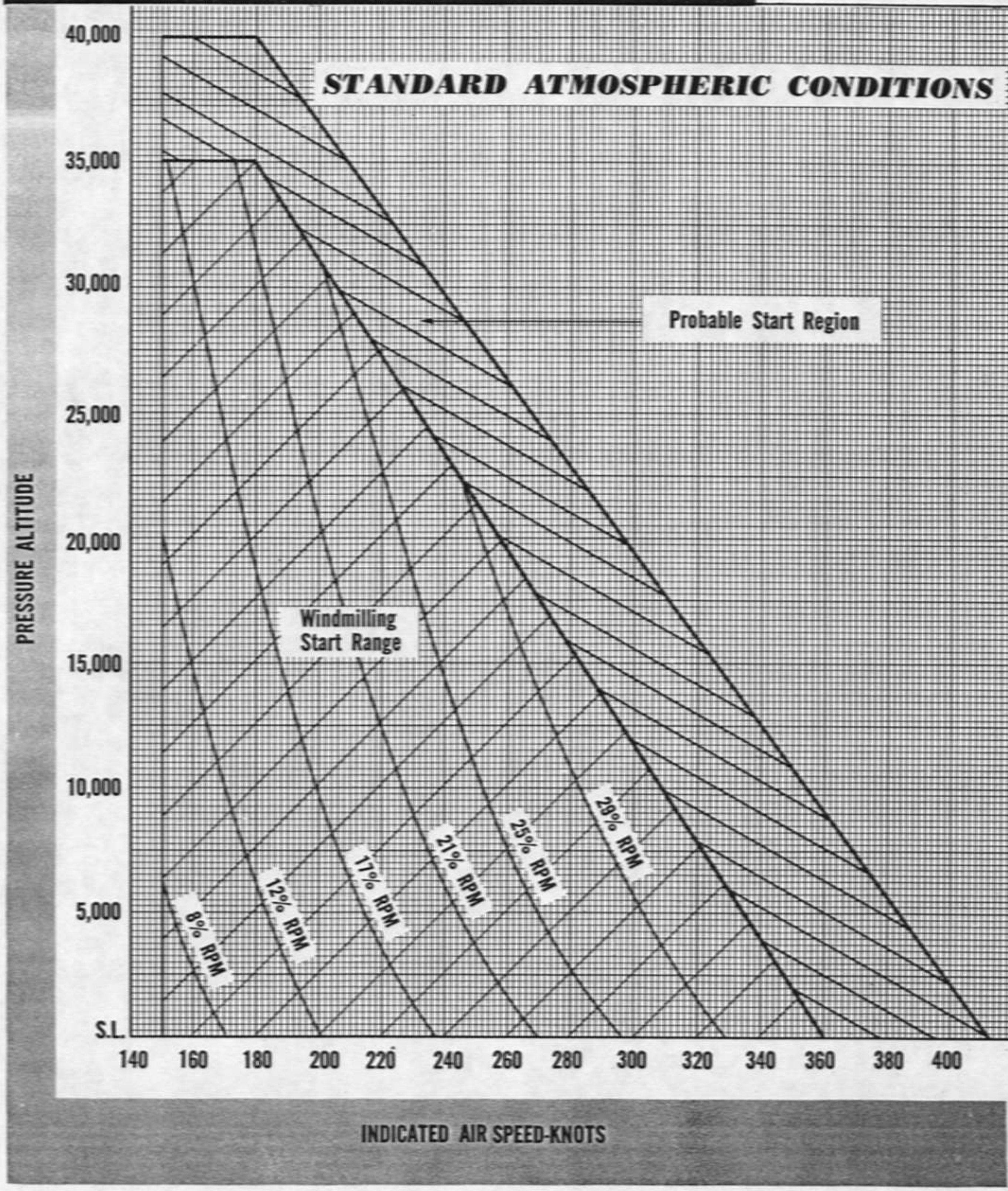


Figure 3–1

4. If no fire is apparent, continue flight at minimum power and land as soon as possible.

5. If positive indications of fire exist, close throttle, turn fuel tank selector OFF and make decision to land or bail-out.

FIRE WARNING AND OVERHEAT LIGHT ON STEADY

Illumination of the fire warning light indicates a fire in the engine section necessitating immediate action as follows:

1. Throttle control OFF.
2. Fuel tank selector OFF.
3. Check for other indications of fire such as smoke in cockpit, engine noise, abnormal exhaust gas temperature or verification from another airplane.
4. If positive indications of fire exist, make decision to land or bail-out.

FIRE WHILE STARTING THE ENGINE

1. Throttle control OFF.
2. Fuel tank selector OFF.
3. Battery switch OFF or disconnect external power supply.
4. Leave cockpit immediately.

FIRE DURING TAKE-OFF

IF NOT AIRBORNE

1. Throttle control OFF.
2. Fuel tank selector OFF.
3. Trailing edge flaps DOWN.
4. Speed brake OUT.
5. Deploy drag chute.
6. Brake to a stop on runway if possible.
7. Jettison external stores if necessary.
8. If impossible to stop on runway: Jettison canopy, retract landing gear then turn battery switch OFF.
9. Leave cockpit as soon as possible.

IF AIRBORNE

1. Throttle control OFF.
2. Fuel tank selector OFF.
3. Jettison external stores.
4. Trailing edge flaps DOWN.
5. Land straight ahead on runway and brake to a stop if possible.
6. If impossible to stop on runway: Jettison canopy, retract landing gear then turn battery switch OFF.
7. Leave cockpit as soon as possible.

BEYOND RUNWAY

1. If power is not required to clean obstacles, close throttle and turn fuel tank selector OFF.
2. Make decision to either bail-out or make an emergency landing.

INTERNAL ENGINE FIRE AFTER SHUTDOWN ON THE GROUND

When a fire is indicated within the engine after shutdown it is imperative that the engine be cooled in the vicinity of the fire by rotating the compressor.

1. Connect external compressed air supply.
2. Throttle control OFF.
3. Fuel tank selector OFF.
4. Engine ground start switch GROUND START.
5. Maintain cranking until all evidence of fire has disappeared.
6. Engine ground start switch STOP START.

AFTERBURNER CUT-OFF FAILURE

If the afterburner remains lit after the throttle control is moved inboard the following procedure is recommended.

1. Throttle control – Reduce smartly to some point below 80 per cent rpm to mechanically terminate afterburning; then advance slowly to the minimum thrust which will sustain flight and land as soon as possible.

CAUTION

Do not pause at the afterburner actuation point; move the throttle control smartly through this range.

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 critical flight condition by advancing the throttle control smartly above 80 per cent military power.

SMOKE ELIMINATION

LANDING EMERGENCIES

LANDING WITH WHEELS RETRACTED

1. Jettison external stores.
2. Jettison canopy.
3. Safety belt and shoulder harness tightened and inertia reel LOCKED.

CAUTION

The pilot is prevented from bending forward when the reel lock control is in the locked position. Therefore, all switches not readily accessible should be "cut" before moving the control to the locked position.

4. Leading edge flaps.
5. Trailing edge flaps –
6. Speed brakes –

LANDING WITH ANY ONE GEAR UP OR UNLOCKED

BRAKE FAILURE

1. Use up fuel until minimum fuel remains.
2. Close throttle on touchdown.
3. Deploy drag chute.
4. Pull emergency brake handle and apply brakes in normal manner with rudder pedals.

EMERGENCY ENTRANCE

If the pilot cannot open the canopy by action of the internal canopy control the following procedure is recommended:

1. Actuate the external canopy control and lift the canopy up.
2. If this fails the pilot can actuate the canopy jettison switch on the left console to unlock the canopy.
3. If canopy cannot be opened, break plexiglas aft of seat to gain entrance.
4. When access to the cockpit is gained, check the position of the ejection seat armrests.
5. If armrests are down in the normal position, insert safety pin if possible and be careful not to foul or raise the armrests.
6. If the armrests are up, extreme caution must be observed to assure that the seat triggers are not actuated.

DITCHING

BAIL-OUT PROCEDURE

In all cases of emergency exit in flight, it is recommended that escape be accomplished by means of seat ejection. This is the safest method of escape in either high speed or low speed flight since it precludes the possibility of pilot injury through collision with the tail surfaces. Because of the increasing incidents of vertebrae injuries occurring to pilots during forced landings of high performance aircraft, more consideration should be given to more frequent use of the ejection seats in preference to forced landings. Ejection should be made whenever possible above 1500 feet (although with the use of the automatic belt and parachute successful ejections may be accomplished at altitudes as low as 300 feet). Whenever the terrain is unknown or unsuited for forced landings the ejection seat should be utilized. For bail-out procedure: (see figure 3–2.)

FUEL SYSTEM FAILURE

ENGINE FUEL CONTROL SYSTEM FAILURE INFLIGHT

1. Throttle control IDLE.

CAUTION

When safety of flight necessitates, transfer to the emergency fuel system may be made at any throttle control setting. At all other times, the transfer should be made with the throttle control in the IDLE position.

2. Fuel system selector switch EMERGENCY.
3. Move throttle control to desired thrust.

WARNING

In emergency, fuel flow is manually selected and controlled (except for altitude compensation up to approximately 30,000 feet); therefore, throttle manipulation must be cautious with due regard for maximum exhaust gas temperatures. Do not transfer back to normal for duration of flight.

MINIMUM SAFE EJECTION ALTITUDES

Manually actuated parachute ______ 1500 ft.

Automatic-opening parachute if parachute lanyard key is inserted into safety belt buckle ______ 300 ft.

EJECTION PROCEDURE

Before Ejection

IF TIME AND CONDITIONS PERMIT

1. Unclamp the oxygen hose from the shoulder harness and clip to parachute harness prior to ejection, to avoid being strapped in the seat with the oxygen hose.

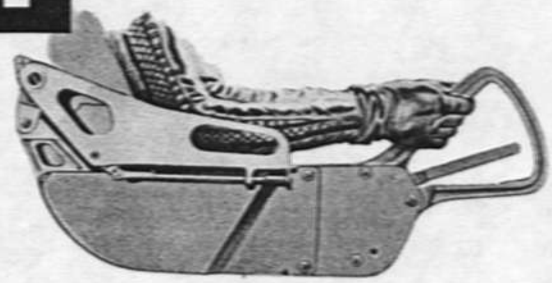
2. Before ejection stow all loose equipment.

3. At altitude, pull ball handle on bail-out bottle.

4. Unfasten safety belt before ejection below 2,000 feet without automatic-opening parachute to aid in separating from seat.

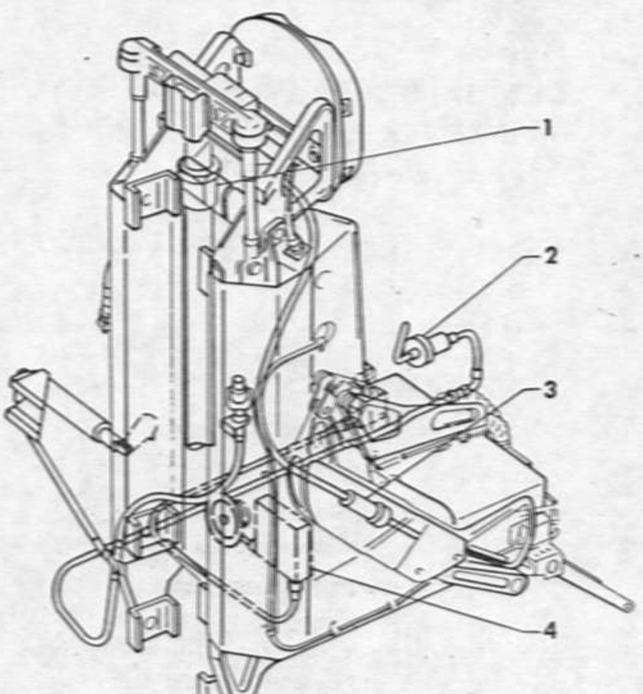
BAIL-OUT

1 **Jettison canopy by pulling either leg brace all the way up until it locks. *Shoulder harness locks automatically when leg braces are raised.***



If Canopy Fails to Jettison

If canopy fails to jettison when the armrests are raised, actuate the canopy jettison switch on the left console or attempt to open the canopy by releasing the internal canopy control. If all efforts fail to release the canopy the seat can be ejected through the canopy.



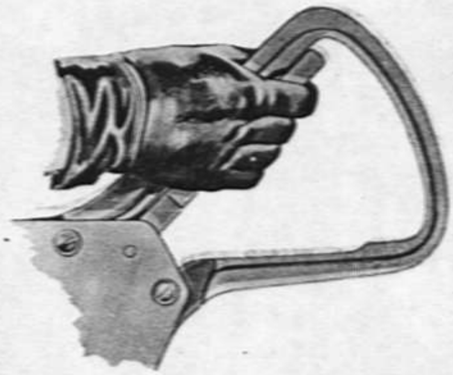
1. Seat Catapult
2. Canopy Jettison Iniator (left side)
3. Seat Ejection Iniator (left & right side)
4. Shoulder Harness Inertia Reel

Figure 3—2. (Sheet 1 of 2)

PROCEDURE

2 Place feet firmly on footrests and brace arms on armrests. Sit erect with head hard against headrest and chin tucked in.

3 Squeeze either seat jettison trigger to eject seat.



AFTER SEAT EJECTS

1 Manually unfasten safety belt and kick free of seat if safety belt fails to open automatically after 2 seconds. Then pull parachute arming lanyard.

2 Kick free of seat and pull parachute arming lanyard if wearing automatic-opening aneroid type parachute *without* arming lanyard key inserted into safety belt buckle. If wearing automatic parachute *with* lanyard key inserted into safety belt buckle, parachute opens at a preset altitude after pilot kicks free from seat. Parachute opens after a preset time interval if below preset altitude.

3 If wearing a manually operated parachute, kick free of seat, then pull "D" ring when altitude is reached where normal breathing is possible.

Figure 3—2 (Sheet 2 of 2)

Note

In the case of procedural practice during pilot transition and with a properly functioning fuel control, the transfer back to NORMAL should be made with the throttle control at the position which would give the approximate idle rpm for the existing altitude under normal operation.

ENGINE STAGE FUEL PUMP FAILURE

In the event of failure of the engine stage fuel pump, indicated by illumination of the engine boost pump warning light, the afterburner stage fuel pump will automatically supply fuel pressure to the engine fuel control and engine operation will be normal. If the engine fuel control should also fail the emergency fuel system can be selected.

MAIN TANK BOOSTER PUMP FAILURE

In the event of failure of the main tank booster pump, fuel will automatically be supplied to the engine from the forward and aft tanks. External fuel can be transferred to the forward and aft tank by positioning the fuel tank selector as in normal operation. When all other fuel is consumed the fuel remaining in the main tank can be recovered by direct suction from the engine driven booster pump at altitudes below feet.

FORWARD AND AFT TANK BOOSTER PUMP FAILURE

If the forward and aft tank booster pumps should fail, operation will be normal until all other fuel is used. The remaining fuel in the forward and aft tanks can be recovered by operating below feet altitude.

ELECTRICAL POWER SUPPLY SYSTEM FAILURE

D-C GENERATOR FAILURE

Equipment powered by d-c secondary bus will be inoperative. Turn off all unnecessary equipment to conserve battery.

A-C GENERATOR FAILURE

With failure of the a-c generator the standby inverter will automatically energize essential a-c equipment if the inverter switch is in the STANDBY position.

HYDRAULIC POWER SUPPLY SYSTEM FAILURE

UTILITY SYSTEM PRESSURE FAILURE

In the event of failure of the utility system pressure systems actuated by utility pressure will be inoperative. However, the landing gear can be extended by manually releasing the uplocks.

PRIMARY SYSTEM FAILURE

If either primary system number one or number two should fail the primary control surfaces will be actuated by the operating system. If both primary systems fail, position the emergency hydraulic system control to the TURBINE EXTENDED position. This extends the air driven emergency pump and transfers the primary flight control systems to the emergency hydraulic system. If utility pressure is still available, systems operated by the utility system will still function normally.

LANDING GEAR SYSTEM FAILURE

LANDING GEAR RETRACTION WHILE ON THE GROUND

If it becomes necessary to retract the landing gear while the weight of the airplane is on the main struts proceed as follows:

1. Actuate the landing gear emergency release switch.
2. Position the landing gear control to UP.

NOTE

The landing gear will retract only if utility hydraulic pressure and primary bus power are available.

LANDING GEAR EXTENSION WHILE IN THE AIR

If the landing gear will not extend by normal procedures, indicated by the landing gear position indicators or low hydraulic pressure proceed as follows:

1. Reduce speed to 243 knots IAS or below.
2. Pull landing gear emergency release handle all the way back.
3. If necessary, yaw the airplane to lock the main gears.

Note

Observe the landing gear warning light and the position indicators when yawing the airplane to determine when the spring-loaded downlocks are engaged.

SECTION IV

DESCRIPTION AND OPERATION OF AUXILIARY EQUIPMENT

TABLE OF CONTENTS

HEATING, PRESSURIZATION AND VENTILATING SYSTEM

Cabin pressurization, heating and ventilating are combined into an air conditioning system (figure 4–1). Compressed air is diverted from the final stage of the high pressure compressor of the engine, passed through a heat exchanger then used for cabin air conditioning, windshield sidepanel defrosting, defrosting, de-fogging dry air system pressurization for the anti-g suit, the canopy seal and pressurization of the external fuel tanks. When the canopy is closed and locked, the cabin is sealed by an automatically inflated rubber seal. Cabin temperature is controlled by diverting a portion of hot air from the engine compressor through a heat exchanger where it is partially cooled. This partially cooled air is then directed through a cooling turbine where it is brought to a low temperature for cockpit cooling. A source of hot air, drawn directly from the engine compressor is combined with the cold air for temperature variations in the cockpit. Varying the amount of hot air by means of a control provides any desired cockpit air temperature. The temperature controlled air enters the cockpit through two cabin exit tubes, one at each side aft of the pilot's seat, and through two foot warmers. The foot warmers are hollow tubes on which the rudder pedal assemblies ride. These tubes are provided with a series of holes through which the temperature controlled air is directed onto the pilot's feet. Pressurization is maintained automatically by the pressure regulator which controls the outflow of air to maintain the cockpit pressure in the prescribed range. Two pressure differential ranges may be selected. The cabin altimeter (23, figure 1–4) indicates the equivalent cabin altitude. A safety valve is incorporated which performs, pressure-relief, vacuum relief and dump-valve functions. The pressure-relief and vacuum relief functions are automatic and protect the cockpit

AIR CONDITIONING SYSTEM

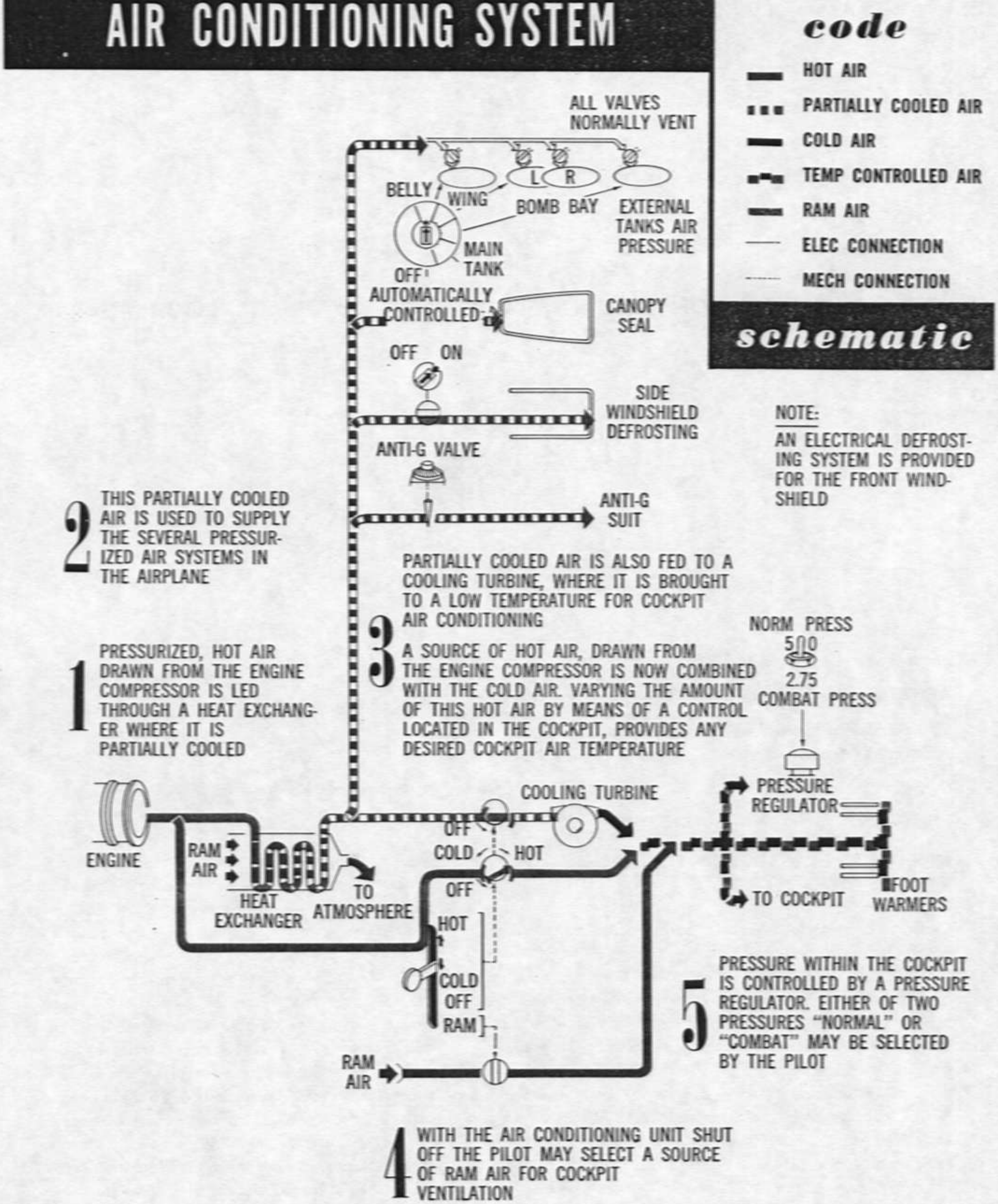


Figure 4—1

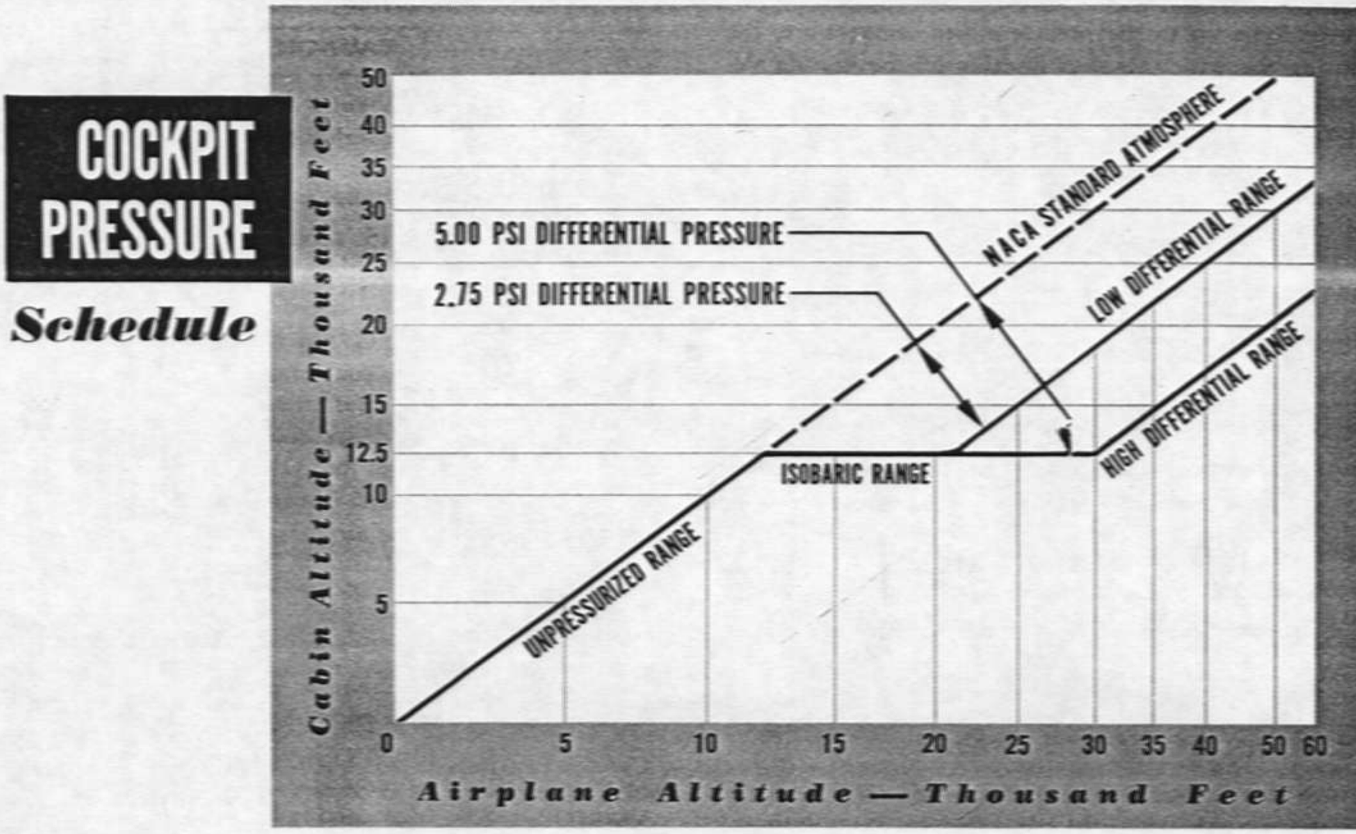


Figure 4–2

from high or low pressures. The dump valve function can be selected to dump cabin pressure, if necessary. At the same time, the ram air valve is opened to allow ram air circulation through the cockpit.

TEMPERATURE AND PRESSURE CONTROL

The temperature and pressure control (figure 4–3) manually selects cabin temperature or cabin ventilation as desired. The control has four positions; HOT, COLD, OFF and RAM. When in the COLD position all of the hot air from the engine is routed through the cooling turbine for cooling. As the control is moved toward the HOT position, hot air is allowed to flow around the cooling turbine and mix with the cooled air. At the same time, the amount of air directed to the cooling turbine is decreased. Successive increments on movement of the control to HOT increases the cockpit temperature proportionally until the control is in the HOT position where all the air entering the cockpit will be directly from the engine compressor. With the temperature and pressure control in the OFF position all valves are closed and no air enters the cockpit. The RAM position opens the ram inlet valve and the dump valve. This allows a circulation of ram air through the cockpit.

COCKPIT PRESSURE SWITCH

The cockpit pressure switch (figure 4–3) is a two position switch marked NORMAL PRESS 5.00 and COMBAT PRESS 2.75. In the NORMAL PRESS 5.00 position, the pressure regulator maintains cabin altitude equivalent to 12,500 feet up to 30,000 feet. Above 30,000 feet a maximum pressure differential of 5.00 psi is maintained between the cockpit and the outside atmosphere. In the COMBAT PRESS 2.75 position, the pressure regulator maintains cabin altitude equivalent to 12,500 feet up to 21,200 feet. Above 21,200 feet a maximum pressure differential of 2.75 psi is maintained between the cockpit and the outside atmosphere. This position is provided for pilot safety to minimize danger resulting from sudden decompression during combat. The cockpit pressure switch is powered from the d-c secondary bus. The temperature and pressure control must be in the COLD HOT range for the cockpit pressure switch to be active.

ANTI-ICING AND DE-ICING SYSTEMS

WINDSHIELD DE-FOGGING SYSTEM

The center windshield panel is designed and built so that an electrical current, passing through an electrical conductive coating between laminations, keeps the windshield panel at a predetermined temperature. The system incorporates a temperature sensing element molded into the panel. When the sensing element opens, the heating element heats the windshield. As the windshield heats up to the predetermined temperature, the sensing element closes thus shutting off the heating element until it is needed again. The electrical circuit can also be turned off for ground operation as the windshield would become too hot with eventual damage to the glass panel. Power for the de-fogging system is supplied by the a-c secondary bus.

DE-FOGGING SWITCH

The windshield de-fogging switch (figure 4–3) is an ON, OFF switch which controls a-c secondary bus power supply to the windshield de-fogging system.

WINDSHIELD SIDE PANEL DEFROSTING SYSTEM

Perforated tubes, clamped to brackets on the windshield side panels are connected by tubing and flexible hoses to a manually operated hot air mixing valve. The mixing valve is supplied with air which has passed through the heat exchanger. The amount of air directed to the perforated tubes can be manually controlled.

SIDE PANEL DEFROSTER CONTROL

The amount of air being supplied to the side panel defroster tubes is manually controlled by the side panel defroster control (figure 4–3). The control rotates 90 degrees between the ON and OFF positions with detents of quarterly increments for proportional flow of air.

ANTI-FOGGING SYSTEM

The anti-fogging system provides a flow of dry air between the double glass panes of the side windshield and canopy windows to clear any condensed moisture from the inter-pane surfaces. The system consists of a dessicator and the necessary tubing and connections to the window panels. The pressurized cockpit air is forced into the dessicator, where it is dried, and then passed into the areas between the panels. The drying agent in the dessicator must be renewed preiodically. The system is never inoperative because any air that enters normally will always pass through the dessicator.

PITOT HEATER

The pitot tube, located on a boom on the nose of the fuselage, is electrically heated to keep it ice free. The heater is powered from the d-c primary bus.

PITOT HEAT SWITCH

The pitot heat switch (figure 4–3) has two positions; OFF and PITOT ON. The PITOT ON position turns on the heater element which heats the pitot tube and keeps it free of ice.

CAUTION

The pitot heater should not be used on the ground as serious overheating will occur without sufficient air flow.

COMMUNICATIONS AND ASSOCIATED ELECTRONIC EQUIPMENT

Communication and associated electronic equipment consists of radio sets to provide airplane-to-airplane and airplane-to-ground communication, radar sets for ground support and aircraft identification and navigation equipment for guidance during flight. All sets are remotely controlled from the cockpit with visual indicators on the consoles and instrument panel. Antennas are consealed within the airplane and therefore are protected from air loads during high speed flight and are also kept free of ice and dirt.

INTERPHONE

The AN/AIC-10 interphone set is provided so that type M-32/AIC oxygen mask microphone and type H-75/AIC headset or their equivalents can be used. This microphone and headset combination makes possible voice communication of high intelligibility in jet aircraft at all altitudes and under severe ambient noise conditions. Units of the interphone control are used to connect the headset and microphone to the communications equipment. An interphone station is provided for use of ground maintenance personnel.

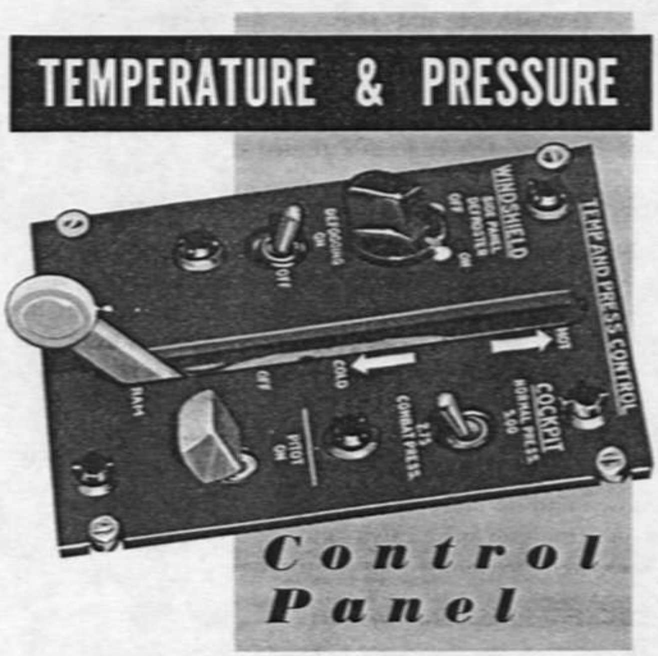


Figure 4–3

COMMAND SET AN/ARC-34

Radio set AN/ARC-34 (7 figure 1–5) is a remotely controlled receiver-transmitter operating in the 225.0 to 399.9 mc band. A total of 1,750 crystal controlled receive-transmit channels in increments of one-tenth of a megacycle, are within tuning range of the radio set and may be manually selected by the pilot. Any 20 channels within the radio's total frequency range may be preset in any order for quick selection by the rotary-type preset channel selector switch on the radio control panel. Four knobs are provided for manual selection of any operating frequency, so that manual operation does not disturb the preset channel arrangement. Receiver and transmitter tuning is automatically completed after a channel change. Two receivers, a main, and a guard receiver, which is constantly tuned to the guard frequency, are provided. The guard receiver may be operated along with the main receiver, thus making it possible to monitor the guard channel for command or emergency communication, while still carrying on communication on another channel. Receiver operation is manually selected by a rotary-type function switch. Continuous tone transmission is also provided through a tone button which may be used for voice transmission, or as a key for coded signals. All radio operating controls for the AN/ARC-34 radio are located on the radio control panel on the left console, with the exception of the microphone press-to-talk button, which is located on the throttle. The pilot can monitor his own transmitted signals through a side-tone circuit which feeds from the transmitter into his headset. The transmitter and the main receiver are tuned to the same frequency. The radio will operate if the battery switch is ON and the d-c primary bus is energized.

Figure 4–4

FUNCTION SWITCH

The function switch (figure 4–4) a rotary, four-position selector switch marked OFF-MAIN-BOTH-ADF is used to turn the AN/ARC-34 radio on or off, or select the type of receiver operation desired. The OFF position shuts off power to the transmitter, main and guard receivers. In the MAIN position the radio receives on the main receiver and both the main receiver and transmitter are tuned to the preset or manually set up channel selected, while the guard receiver remains inoperative. When the function switch is in the BOTH position, the radio will receive signals simultaneously from the guard receiver, which is constantly tuned to the guard channel, and from the main receiver while the transmitter will operate on either a preset or manually selected frequency depending upon the type of channel selection used. The ADF position is for the automatic direction finder.

MANUAL-PRESET-GUARD SELECTOR SWITCH

The lever-type, three-position selector switch (figure 4–4) marked MANUAL-PRESET-GUARD is used to select the method of channel selection, or to switch both the main receiver and the transmitter to the guard channel frequency. A MANUAL-PRESET-GUARD switch indicator window is provided and the type of selection set up by the switch is indicated, while the other positions are visible, but covered by a green shutter. In the MANUAL position, the four tuning knobs at the top of the control panel are used, permitting manual selection of anyone of the 1,750 frequencies in the radio's turning range for transmit-receive operation. The PRESET and GUARD positions on the MANUAL-PRESET-GUARD switch indicator window will be covered, as will the preset channel indicator window. The indicator windows above each of the four manual tuning knobs are open and provide a direct reading in megacycles and tenths of a megacycle of each frequency manually selected by the manual knobs. In the PRESET position, the preset channel selector switch is used, and transmit-receive operation may be carried on any one of the 20 preset channels, as selected by the preset channel selector switch and indicated in the preset channel indicator. The manual tuning knob windows and the MANUAL and GUARD positions on the MANUAL-PRESET-GUARD switch indicator windows will be blanked out. In the GUARD position, the main receiver and transmitter are switched to the guard channel frequency. However, if the function switch is in the BOTH position the main and guard receivers will operate simultaneously on the guard frequency. The manual tuning indicator window, the preset channel indicator window and the PRESET and MANUAL positions in the MANUAL-PRESET-GUARD indicator window will be covered.

TONE PUSH BUTTON

The spring-loaded push button marked TONE (figure 4–4) switches the radio over from receive to tone transmission on the manually selected or preset frequency that the radio is operating on, as long as the depressed position is held. A 1,020 cycle tone is continuous during voice transmission as long as the tone button is depressed. The tone button may be used as a key for code transmission. The microphone press-to-talk button, regardless of its position, has no effect on the operation or function of the tone button.

Note

> Tone transmission is usually perceptible over a slightly greater distance and through greater interference than voice transmission. Therefore it is especially adaptable in an emergency for code key operation, when interference or jamming conditions make voice transmission impractical, or to serve as a radio marker signal for direction finding.

VOLUME CONTROL

The volume control (figure 4–4) marked VOLUME regulates the headset volume for signals received on both the main and guard receivers. Volume control range is predetermined so that the signal volume may not be reduced below a preset level.

MICROPHONE PRESS-TO-TALK BUTTON

The microphone press-to-talk button (figure 1–8), located on the engine throttle control, switches the radio set from receive to transmit operation for voice modulation when held in the depressed position. When the button is released the radio returns to receive operation.

STARTING

1. Place the function switch in the BOTH position.
2. Turn MANUAL-PRESET-GUARD selector switch to PRESET position.
3. Rotate preset channel selector knob until desired channel number appears in preset channel indicator window. Allow approximately one minute for equipment warm-up and automatic channel tuning adjustment cycle. At the end of the warm-up period the equipment will be in the standby condition ready to receive signals on the preset command and fixed guard frequencies simultaneously. During transmission periods the pilot should receive his own signals on a side-tone received by his headset. A little receiver noise may

IDENTIFICATION RADAR

Figure 4–5

or may not be heard during non-transmission periods, depending on the squelch adjustment in the receiver transmitter. This control is not available to the pilot.

Note

> When selecting a new preset channel there will be a delay of four seconds before the automatic tuning cycle adjusts the set for operation on the newly selected frequency.

4. Adjust the volume control for a comfortable signal level in the headset.
5. Before transmission, check that equipment is functioning and tune-up is complete by listening for voice side-tone, or tone signal when the tone button is depressed.

STOPPING

1. Place the function switch in the OFF position.

Note

> Once the radio set has been tuned off the 60-second warmup period must be completed before radio transmission can be resumed.

OPERATING PRECAUTIONS

1. Transmit only when the channel is clear to prevent confusion and loss of messages. The guard frequency

should not be used for transmission unless the message is urgent.

2. Use both receivers for general reception, unless signals from one make the desired signals from the other unusable.

3. It is possible to set the preset channel buttons or manual frequency selector knobs for frequencies below 225.0 megacycles. Since this is below the operating frequency range, the automatic tuning mechanism cannot accomplish tune-up. It will operate for approximately 120-seconds and will then be turned off automatically by a protective relay. Under these conditions there will be no side-tone if transmission is attempted and operation of the frequency selection knobs will have no effect. To restore operation, select a channel within the 225.0 to 399.9 megacycle range. Turn the function switch to the OFF position and then to BOTH. After approximately one minute the radio equipment will operate in a normal manner.

IDENTIFICATION RADAR – AN/APX-6A

The AN/APX-6A radar identification (6, figure 1–6) set is used to automatically identify the airplane whenever it is properly challenged by suitably equipped friendly air or ground forces. The set can also identify the airplane, in which it is installed, as a friendly airplane within a group of specific friendly airplanes and has means for transmitting a special distress code. Functionally, the AN/APX-6A set receives challenges and transmits replies to the source of the challenges, where the replies are displayed, together with the associated radar targets, on radar indicators. When a radar target is accompanied by a proper reply from the IFF set, the target is considered friendly. The IFF set is powered from the d-c and a-c secondary busses.

OPERATION OF IFF RADAR

CAUTION

Before take-off, check that AN/APX-6A frequency counters have been set to proper frequency channels.

1. Rotate master switch to NORM position unless otherwise instructed.

2. Rotate master switch to STDBY position to maintain equipment inoperative but ready for instant use.

3. Set Mode 2 and Mode 3 switches to OUT unless otherwise instructed.

4. If in distress, press dial stop and, at same time, rotate master switch to EMERGENCY position for emergency operation. Set automatically transmits distress signals when interrogated.

STOPPING

1. Rotate master switch to OFF position.

RADAR SET – AN/APW-11

Information and operating instructions on the ground support radar set, AN/APW-11, (7, figure 1–6) will be incorporated at a later date.

RADIO RECEIVING SET – AN/ARN-14D

The AN/ARN-14D omnidirectional receiver is a VHF navigation receiver, with associated indicators, that provides reception of all VHF omnirange, localizer, and voice facilities (which include both military and commercial communication) in 108 to 135.9 megacycle channels. Reception of VHF omnirange facilities provides a display of on-course information, indication of the magnetic bearing of the VHF omnirange from the airplane, indication of angular difference between selected course and the magnetic heading of the airplane, and to-station, from-station indications. Reception of localizer facilities provides on-course information for the instrument landing approach system. Voice facilities are received on communication channels or are superimposed on navigational signals in localizer or omnirange bands for station identification. The AN/ARN-14D control panel (figure 4–7) is marked VHF NAV and contains a power switch, an audio volume control and a frequency selector switch. Display of the informa-

Figure 4–6

RADIO RECEIVING SET

Figure 4–7

tion selected by the positioning of the frequency selector will appear on either the course indicator (22, figure 1–3) or the course bearing indicator (16, figure 1–3) or both. The radio set is powered from the d-c secondary bus and the a-c primary bus.

OPERATION OF RADIO SET AN/ARN-14D
VHF OMNI-RANGE FLYING

1. Turn power switch to ON position.

2. Set desired VHF omni-range into the vertical window of the control panel. Identify station from coded signal in headset.

CAUTION

Check to see that the flag alarm OFF indicator for the vertical bar disappears. The flag alarm must be out of sight before indications of the vertical bar can be relied upon.

3. Using the set control on the course indicator set the magnetic course through the station, on which it is desired to fly, into the course window.

VHF OMNI-RANGE HOMING

1. Turn power switch to ON position.

2. Set the frequency of the VHF omni-range on which it is desired to home, into the vertical window of the control panel. Identify station from coded signal in headset.

3. Turn set control until the vertical bar centers and TO appears in the TO-FROM window.

4. Turn aircraft until the white circle of the relative heading indicator is approximately under the upper position of the vertical bar.

5. Cross check the equipment by observing that pointer number 2 of course indicator is at the fixed index at the top of the instrument. The heading indicated on the course indicator should correspond approximately to the course indication in the course window.

VHF OMNI-RANGE FLYING ON PRE-DETERMINED TRACK

1. Turn power switch to ON position.

2. Set frequency of the VHF omni-range, toward or away from which it is desired to fly, into the vertical window of the control panel. Identify station from coded signal in headset.

3. Using the SET control on the course indicator, set the desired magnetic course through the station into the COURSE window. If this course is the course toward the VHF Omni-range, the TO-FROM indicator will indicate TO. If this course is the course away from the VHF Omni-range the TO-FROM indicator will indicate FROM.

4. Turn the aircraft until the white circle of the relative heading indicator is underneath to upper half of the vertical bar.

5. Continue to steer the aircraft in such a manner as to keep the white circle of the relative heading indicator underneath the vertical bar. As the desired track is approached the vertical bar will move toward center and it will be necessary to turn the aircraft to keep the white circle underneath the vertical bar. When the vertical bar reaches center the aircraft is on the desired track.

6. If the aircraft passes over the station through the cone of confusion, the vertical bar will swing from side to side and the TO in the TO-FROM window will change to indicate FROM. Pointer number 2 of the course indicator will reverse and point toward the station which is now directly behind the aircraft.

PROCEDURE TURN

1. While over the station turn the SET control until the desired outbound course appears in the COURSE window. The TO-FROM indication will be FROM.

2. Turn aircraft until the heading as read on the course indicator corresponds to the COURSE indication on the COURSE window.

3. Turn the aircraft until the white circle of the relative heading indicator is approximately underneath the upper portion of the vertical bar.

4. Fly to the procedure turn starting point by reference to the vertical bar.

5. Make a 45 degree right turn. The relative heading indicator will indicate approximately 45 degrees.

6. After holding this heading for one minute start standard rate (needle width) turn to left.

7. While in left turn, turn SET control 180 degrees until the reciprocal of the outbound course appears in the COURSE window.

8. After 180 degrees of turning turn to keep the white circle of the relative heading indicator underneath the vertical bar until the bar and circle are centered at the top.

LOCALIZER APPROACH

1. Turn power switch to ON position.

2. Set the frequency of the localizer into the vertical window of the control panel.

3. Turn the SET control on the course indicator until the magnetic course of the final approach to the runway appears in the course window. This step will allow the use of the relative heading indicator in approaching the course and thus prevent bracketing.

4. Turn the aircraft until the white circle of the relative heading indicator is underneath the upper half of the vertical bar.

5. As the course is approached and the vertical bar moves toward center turn the aircraft in the direction that the vertical bar is moving so as to keep the white circle underneath the vertical bar. This will cause the aircraft to turn onto the course without overshooting or bracketing.

6. Steer aircraft to keep vertical bar centered using normal sensing.

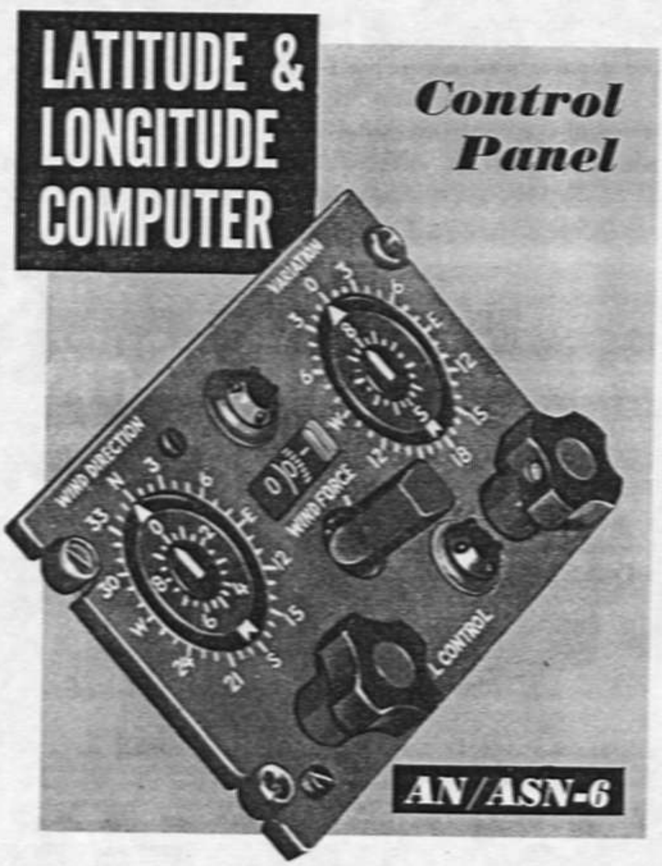


Figure 4–8

LATITUDE AND LONGITUDE COMPUTER – AN/ASN-6

The AN/ASN-6 lattitude and longitude computer system, consisting of a remotely controlled computer, a computer control, an amplifier and a latitude and longitude indicator is provided in the aircraft. The computer provides the pilot with a constant, automatically computed, direct reading indication of the aircrafts position changes. The aircraft's position appears on direct reading counters of the latitude and longitude indicators in degrees of longitude and latitude. The known position of a ground reference point is set in by the pilot as the aircraft flys over it. Factors effecting the aircraft's position such as wind velocity, wind direction and magnetic variation are manually set into the computer system by the pilot. The true airspeed indicator and the J-2 slaved gyro magnetic compass provide automatic inputs of true airspeed and aircraft heading to the computer system. Both manual and automatic inputs are integrated and computed and the aircraft position is transmitted to the indicator counters. A departure switch is provided to disconnect the latitude and longitude counters on the indicator so they may be reset for ground reference position. The computer will be turned on whenever the d-c and a-c secondary busses are energized. Approximately five minutes are required to allow the amplifier to warm-up after electrical power is applied to the system.

COMPUTER CONTROL UNIT

The computer control unit (9 figure 1–6) is provided to set the manual inputs of wind direction, wind force and magnetic deviation into the computer system. The three rotary control knobs are marked as follows: WD for wind direction, WF for wind force and VAR for magnetic heading deviation. Wind direction is shown on a course scale of 10 degree increments and a fine scale in increments of ½ degree. The wind force is shown on a rotary drum-type counter in a scale range from 0 to 200 knots. Magnetic heading deviation is shown on a course scale of 10 degree increments and a fine scale of ½ of a degree measurement of degrees east or west of true north. These inputs are all set on the computer control panel introducing the information into the computer system before arrival at the

Figure 4–9

ground reference point or before the departure switch is actuated to set the latitude and longitude indicator counters in operation.

LATITUDE AND LONGITUDE INDICATOR

A latitude and longitude indicator (1 figure 1–6) marked GROUND POSITION INDICATOR provides a continuous direct reading of the aircraft position in degrees of longitude, east or west and degrees of latitude, north or south on two counter windows. A lever type switch is provided for each counter window to set the latitude and longitude of the aircraft at the ground reference point. When the latitude switch lever is raised upward, the latitude counters indicate increasing values of north latitude, or decreasing values of south latitude. If the latitude switch lever is actuated downward, the values of north latitude decrease, while the values of south latitude increase. The switch lever for setting longitude, operates in a similar manner. When the switch is operated to the left the values of east latitude decrease while the values of west longitude increase. If the longitude switch lever is operated to the right, the values of east longitude increase while the values of west longitude decrease. Safety latches are provided to lock the switch levers. The departure switch must be in the STANDBY position before the counters can be set by means of the switch levers.

AN ASN-6 DEPARTURE SWITCH

The AN/ASN-6 departure switch is a two position switch marked RUN and STANDBY. The STANDBY position cuts off the signals from the computer system to the ground position indicator, locking the counters so that they may be reset. When the airplanes position has been set on the indicator counters, the departure switch is placed in the RUN position which unlocks the indicator counters permitting them to register the changes in the airplanes position.

OPERATION OF THE AN/ASN-6 COMPUTER

Note

There is no power switch on the latitude and longitude computer set, therefore, the set will be energized whenever the d-c and a-c secondary busses are energized. A minimum of five minutes should be allowed after power is applied for the amplifier to warm up.

1. The departure switch is left in the STANDBY position until arrival at the ground reference point. The aircraft position is then set manually on the ground position indicator counters. The point of departure is not used, as inaccuracies are introduced into the computer by a steep angle of climb.

CAUTION

The counters will continue to operate until the switch levers are returned to the center position. Set safeties on switches.

2. Set in wind direction, wind force and magnetic variation on the computer panel.

3. Upon arrival at ground reference point, position the departure switch in the RUN position. The latitude and longitude indicator counters will then operate showing constant indication of the airplane's changing position.

Figure 4–10

OXYGEN SYSTEM

A liquid oxygen system, consisting of five liter capacity vacuum insulated container, build-up coils, check valves, relief valves and quantity gage is provided. Liquid oxygen is stored in the vacuum container and passes into the build-up coils. Here it evaporates into gaseous oxygen and passes into the oxygen regulator at approximately 70 psi. The oxygen supply from the regulator to the pilot's pressure breathing demand oxygen mask passes through a quick disconnect on the front of the pilot's seat. Excessive pressures in the system between the vacuum container to the regulator are relieved through the relief valves and vented overboard.

Note

> The liquid oxygen quantity gage should read between 4 and 4½ liters when the system is fully charged. Do not be alarmed that the gage does not read 5 liters, since it is impossible to change the liquid oxygen converter to 5 liters. Use the oxygen duration chart to determine your oxygen duration for the indicated supply.

OXYGEN REGULATOR

Figure 4–11

REGULATOR

A type D-2 automatic pressure breathing diluter-demand oxygen regulator (figure 4–11) is installed on the right console and includes a low pressure gage and flow indicator. The regulator automatically supplies the proper mixture of oxygen and air at all altitudes with provisions for positive pressure breathing at high altitudes and also indicates the proper functioning of the oxygen system by means of a warning light.

REGULATOR DILUTER LEVER

The regulator diluter lever (figure 4–11) has two positions NORMAL OXYGEN and 100% OXYGEN. When the lever is in the NORMAL OXYGEN position, the regulator unit will function to provide automatic mixing of air and oxygen in sea level concentration at all altitudes. In the event that the regulator malfunctions, a pressure relief valve in the regulator unit will relieve to protect the pilot from excessive pressure. When the lever is in the 100% OXYGEN position, the automatic air-oxygen mixing feature is by-passed and 100 per cent oxygen is supplied regardless of altitude.

OXYGEN SUPPLY SHUT-OFF LEVER

An ON-OFF oxygen supply shut-off lever (figure 4–11) is located on the aft end of the regulator. When the lever is in the ON position, system oxygen is supplied to the regulator unit. When the lever is in the OFF position the oxygen supply to the regulator is shut-off to prevent any flow of oxygen from that unit when not in use.

OXYGEN EMERGENCY TOGGLE LEVER

The oxygen emergency toggle lever (figure 4–11) is provided for emergency operation or to supply maximum pressure for leakage test of the oxygen mask. When the lever is in the EMERGENCY position (either to the right or left from the normal off position), oxygen is supplied to the mask at a fixed pressure increase over that being supplied automatically. This will provide an increase flow of oxygen in an emergency. When the lever is pushed straight into the test mask position, maximum pressure is delivered at the outlet regardless of altitude, for leakage test of the oxygen mask. The lever will return to the off position when released. In the off position, oxygen flow is controlled automatically by the regulator unit.

OXYGEN WARNING SYSTEM SWITCH.

An ON-OFF switch (figure 4–11) is provided on the regulator to control the oxygen warning system. When the switch is in the ON position, power is supplied from the primary bus to the warning light circuit. In the OFF position the warning light circuit is de-energized.

Note

> The oxygen system warning light is not installed at this time.

LOW PRESSURE GAGE.

The low pressure gage (figure 4–11) incorporated in the oxygen regulator unit records the oxygen pressure being supplied to the regulator. The gage scale reads from 0 to 500 psi. The maximum operating pressure is approximately 70 psi for the liquid oxygen system.

OXYGEN QUANTITY GAGE

An oxygen quantity gage (figure 4–11) is installed to

CREW MEMBER OXYGEN DURATION HOURS

BLACK FIGURES INDICATE DILUTER LEVEL NORMAL OXYGEN

RED FIGURES INDICATE DILUTER LEVEL 100% OXYGEN

1 CREW MEMBER
1 TYPE A-3 CONVERTER

CABIN ALTITUDE FEET	GAGE QUANTITY—LITERS 5	4	3	2	1	BELOW 1
40,000	24.0 24.0	19.2 19.2	14.4 14.4	9.6 9.6	4.8 4.8	EMERGENCY DESCEND TO ALTITUDE NOT REQUIRING OXYGEN
35,000	24.0 24.0	19.2 19.2	14.4 14.4	9.6 9.6	4.8 4.8	
30,000	17.3 17.8	13.8 14.2	10.4 10.6	6.9 7.1	3.5 3.5	
25,000	13.3 16.8	10.6 13.4	8.0 10.0	5.3 6.7	2.7 3.4	
20,000	10.1 19.0	8.1 15.2	6.1 11.4	4.0 7.6	2.0 3.8	
15,000	7.8 23.0	6.2 18.4	4.6 13.8	3.1 9.2	1.5 4.6	
10,000	6.6 23.0	5.3 18.4	4.0 13.8	2.6 9.2	1.3 4.6	

Figure 4—12

record the amount of liquid oxygen remaining in the vacuum container. The gage is calibrated to read from 0 to 5 liters.

OXYGEN FLOW INDICATOR

A blinker type oxygen flow indicator (figure 4—11) is incorporated in the pressure demand regulator unit. Black and luminescent segments alternately appear through four slots in the indicator face with each breath taken through the oxygen mask.

PRESSURE DEMAND OXYGEN MASKS

Only the type A-13, A-13A or MS22001 pressure demand oxygen mask will be used with the automatic pressure demand oxygen regulator. These masks can be identified by the presence of a gray anodized aluminum exhalation valve (pressure-compensating), which is located in the mask directly below the chin position. Pressure demand masks, when used at altitude, will occasionally produce a distinct vibration in the mask that can be identified by a "wheezing" sound. This condition may be overlooked, in that operation qualities are not disturbed in any manner. If a blocking condition (cannot exhale) occurs during flight, a "sharp" exhalation will usually correct the difficulty. In the event a "sharp" exhalation does not relieve the blocking condition, the mask may be lifted off the face momentarily at the chin section. Extreme caution must be exercised in using this procedure, since the danger of hypoxia increases rapidly above 30,000 feet cabin altitude. Oxygen masks other than those specified above will not be used with automatic pressure demand oxygen regulators. Use of unauthorized oxygen masks results in rapid depletion of the aircraft oxygen supply, and pressure breathing required at altitude will be lost.

USE OF AUTOMATIC PRESSURE DEMAND REGULATOR

1. The diluter lever will always be set at the NORMAL OXYGEN position, except in cases where noxious gases are suspected, or pre-breathing of oxygen is deemed necessary. These exceptions are rare, and if the diluter lever is placed in the 100% OXYGEN position, extreme care must be exercised in monitoring the oxygen supply.

2. Turn oxygen supply shut-off lever to the ON position.

3. Press oxygen emergency toggle lever straight in to test mask for leakage at any altitude. Place lever to right or left of the normal off position to provide an increased flow of oxygen only in case of an emergency. If emergency use is necessary, however, extreme caution must be used to prevent rapid loss of system pressure through the emergency valve.

4. As breathing through the mask is started, the warning light should blink brightly a few seconds and then change to a dim continuous glow, and the oxygen flow indicator should start functioning. The proper flow of oxygen will be automatically maintained by the regulator unit.

OXYGEN REGULATOR CHECK

Check the oxygen regulator prior to take-off with the diluter valve first at the NORMAL OXYGEN position and then at the 100% OXYGEN position as follows.

1. Remove mask and blow gently into the end of the oxygen regulator hose as during normal exhalation. If there is a resistance to blowing, the system is satisfactory. Little or no resistance to blowing indicates a faulty demand diaphragm or diluter air valve, a leaking mask-to-regulator tubing, or a faulty ejection seat quick disconnect.

2. With oxygen supply shut-off lever in the ON position, oxygen mask connected to regulator, diluter lever in 100% OXYGEN position, and normal breathing, conduct the following check:

a. Deflect emergency toggle lever to right or left. A positive pressure should be supplied to mask. Return emergency toggle lever to center position.

b. Depress emergency toggle lever straight in. A positive pressure should be applied to the mask. Hold breath to determine if there is leakage around mask. Release emergency toggle lever, positive pressure should cease.

3. Return diluter lever to NORMAL OXYGEN.

EMERGENCY OPERATION

1. With symptoms of anoxia, or if smoke or fumes should enter the cabin, immediately place the regulator diluter lever in the 100% OXYGEN position and place the emergency lever in the EMERGENCY position.

2. In the event of accidental loss of cabin pressure. no action is required if oxygen is being used, as the regulator unit will automatically compensate for the increased cabin altitude.

3. If the oxygen regulator should become inoperative, pull the cord of the H-2 emergency oxygen cylinder, and descend to a cabin altitude not requiring oxygen.

AIR REFUELING SYSTEM

The air refueling system (figure 4–13) is intended to operate in conjunction with a tanker which is equipped with a drogue. Transfer of fuel from the tanker is accomplished when the receptacle in the drogue is engaged with the probe on the extended probe boom on the forward left side of the fuselage. Fuel transfer through the probe is automatically distributed through the manifold to all internal tanks and to all installed external tanks. With the tanker hose and drogue in position, and the probe boom extended, the pilot flies the probe into the cone of the drogue until the nozzle on the probe engages the receptacle in the drogue. When engagement is completed, the transfer of fuel is initiated by the tanker engineer. As each internal tank becomes full, float-operated refueling shut-off valves within the tanks close preventing further flow from the manifold. Float-operated switches in the bomb bay auxiliary and external tanks provide the means of controlling solenoid operated valves in the bleed lines of externally mounted shut-off valves. As each external tank becomes full the float-operated switch opens to de-energize the bleed solenoid, closing the bleed line, and shutting off further fuel flow from the manifold. Upon completion of the refueling operation, the probe-drogue connection can be broken by the tanker pulling away or the receiver dropping back. The breakaway force in the disconnect in the drogue is adjusted so that undue strain on the connection will cause an automatic disconnect. Self-sealing disconnects on both probe and drogue prevent fuel spillage.

REFUEL PROBE CONTROL

The refuel probe control (17, figure 1–3) manually positions a hydraulic selector valve to control the movement of the refueling probe. When the control is pulled aft as marked PULL-EXTEND a cable unlocks the forward door, indexes a hydraulic selector valve to direct utility hydraulic pressure to extend the probe. The door on the forward end of the probe compartment opens to allow passage of the probe boom assembly. When extended the forward door returns to its faired position and circuits are energized to illuminate the probe light and open the fuel transfer and refueling valve in each external tank. The probe lamp shines through the probe compartment to illuminate the extended probe. Hydraulic pressure holds the probe in the extended position until the fuel transfer is accomplished and a disconnect is initiated. The pilot then pushes the refuel probe control to PUSH-RETRACT position. This actuates the hydraulic selector valve to retract the probe. The initial movement opens the forward door and allows the probe to retract into the compartment. As the linkage is fully retracted, the forward door is faired to the aircraft and the lock secures the probe in the retracted position, de-energizes the refueling circuits and extinguishes the probe light. If hydraulic pressure is not available, a spring-loaded bellcrank on the latch assembly will extend the probe sufficiently to allow complete extension by ram air pressure. After refueling cycle is completed the probe cannot be retracted.

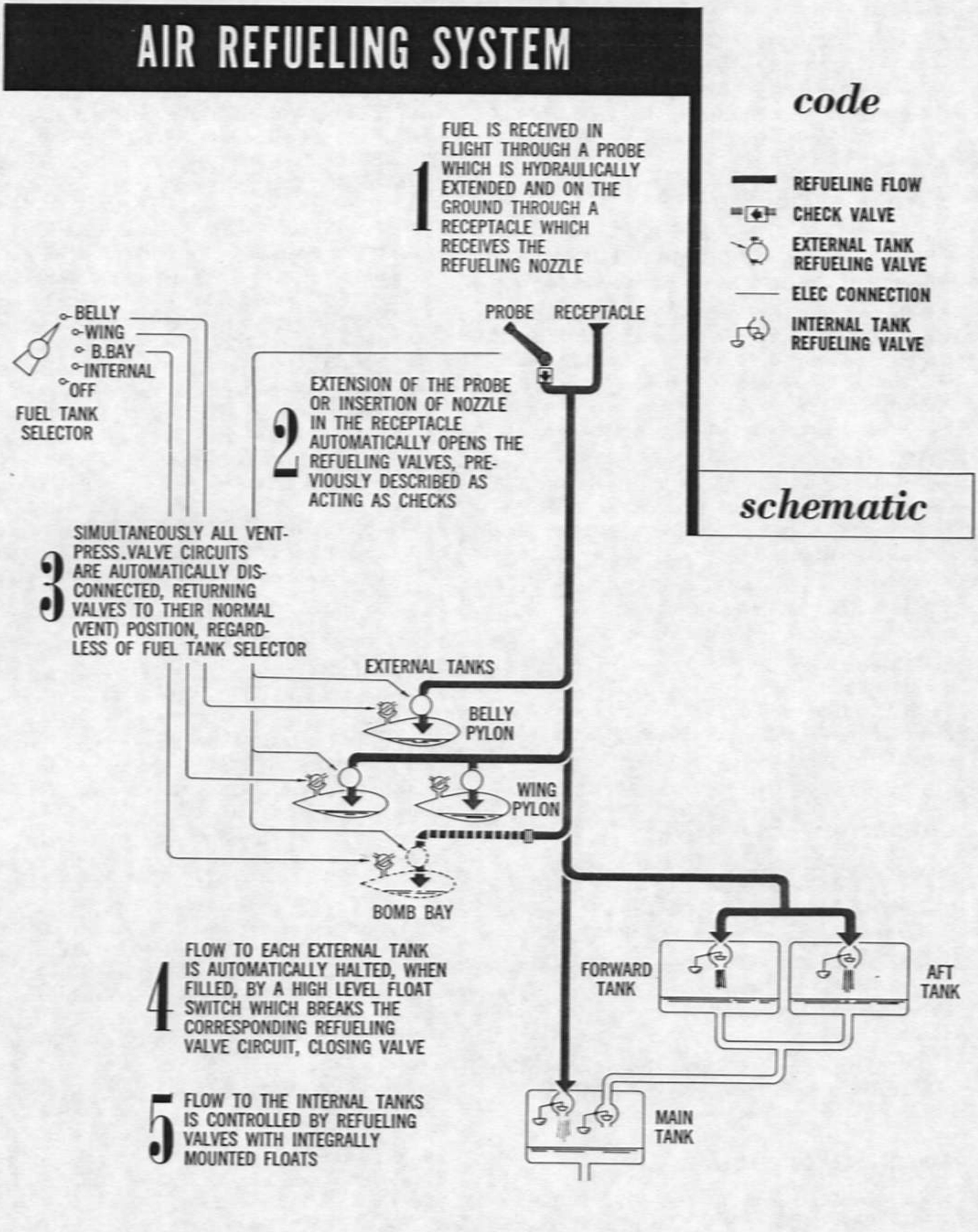

AIR REFUELING SYSTEM
code
REFUELING FLOW
CHECK VALVE
EXTERNAL TANK REFUELING VALVE
ELEC CONNECTION
INTERNAL TANK REFUELING VALVE
schematic
1 FUEL IS RECEIVED IN FLIGHT THROUGH A PROBE WHICH IS HYDRAULICALLY EXTENDED AND ON THE GROUND THROUGH A RECEPTACLE WHICH RECEIVES THE REFUELING NOZZLE
PROBE
RECEPTACLE
BELLY
WING
B.BAY
INTERNAL
OFF
FUEL TANK SELECTOR
2 EXTENSION OF THE PROBE OR INSERTION OF NOZZLE IN THE RECEPTACLE AUTOMATICALLY OPENS THE REFUELING VALVES, PREVIOUSLY DESCRIBED AS ACTING AS CHECKS
3 SIMULTANEOUSLY ALL VENT-PRESS. VALVE CIRCUITS ARE AUTOMATICALLY DISCONNECTED, RETURNING VALVES TO THEIR NORMAL (VENT) POSITION, REGARDLESS OF FUEL TANK SELECTOR
EXTERNAL TANKS
BELLY PYLON
WING PYLON
BOMB BAY
4 FLOW TO EACH EXTERNAL TANK IS AUTOMATICALLY HALTED, WHEN FILLED, BY A HIGH LEVEL FLOAT SWITCH WHICH BREAKS THE CORRESPONDING REFUELING VALVE CIRCUIT, CLOSING VALVE
FORWARD TANK
AFT TANK
5 FLOW TO THE INTERNAL TANKS IS CONTROLLED BY REFUELING VALVES WITH INTEGRALLY MOUNTED FLOATS
MAIN TANK

Figure 4–13

AUTOPILOT

The FC5-R1 autopilot system provides the pilot with several different degrees or modes of automatic flight control while imposing no restrictions on maneuvering with the manual controls. If desired, maneuvers may be accomplished without manually operating the rudder. In this event, the autopilot will provide coordination automatically. The pilot may operate the rudder manually to skid the airplane at any time, however, without opposition from the autopilot system. A force switch, which forms part of the control stick, is provided to allow the pilot to disengage the autopilot for maneuvers without the necessity of throwing a separate switch in addition to maneuvering the control stick or applying a force to overpower the autopilot. The force switch consists of micro switches wired into the control system so that force on the control stick actuates them and drops out autopilot operation for the duration of the maneuver. When the force on the control stick is removed the micro switches drop back in and the autopilot automatically picks up to function as set by the control panel. In any of the autopilot modes of operation maneuvering may be accomplished by the pilot by use of the manual controls in the normal manner. In addition, if the rudder is not operated manually, the autopilot operates to coordinate all maneuvers. When the pilot releases the control stick the autopilot system normally reverts automatically to the control method set previous to the maneuver. However, if, at the time the stick is released, the airplane bank angle exceeds 30 degrees, the existing attitude is maintained, heading control is not re-engaged, and a coordinated turn results. Similarly, if, at the time tht stick is released, the rate of climb or dive exceeds 500 feet per minute, altitude control is not re-engaged, and the airplane is restrained to the altitude existing at the time of release. Automatic trim is provided in roll and pitch in all automatic pilot modes. Another mode of operation is the three axis stability augmentation. In this mode, the autopilot system operates to increase the damping of the airplane in roll, pitch and yaw. The action of the system is not apparent to the pilot except as evidenced by increased stability of the airplane. In this mode, airplane altitude and path are not controlled by the autopilot system but must be controlled by the pilot. All modes of operation are selected at the flight control panel. Operation of the autopilot system is dependent on d-c secondary and a-c secondary power.

FLIGHT CONTROL PANEL

The flight control panel (11, figure 1–6) consists of two push-pull, ON-OFF type switches with solenoids to hold them in the ON position and one four position mode selector switch DAMPER SWITCH. One push-pull switch is marked DAMPERS and PULL OFF. When depressed, the autopilot system operates to increase the damping of the airplane in roll, pitch and yaw. In this mode, airplane altitude and path are not controlled by the autopilot but must be controlled by the pilot. The dampers switch may be engaged 30 seconds after application of electrical power to the system.

Figure 4–14

AUTOPILOT SWITCH

The autopilot switch is a push-pull switch (figure 4–14) marked AUTOPILOT and PULL OFF. When depressed, the autopilot has complete control of the flight of the airplane. Any one of four methods of flight control may be selected by means of the mode selector switch. Before the autopilot is engaged the damper switch must be engaged and then the desired method of control is selected by using the mode selector switch. The autopilot may be engagea two to two and a half minutes after application of electrical power to the system.

MODE SELECTOR SWITCH

The mode selector switch (figure 4–14) is a four position rotary switch. The markings are APW-11, ALT, MACH and ARR-39. The autopilot switch must be depressed for operation of any of the four modes of control. When in the ALT position, the pilot manually flies the airplane to the commanded altitude and the autopilot then maintains this altitude by command signals transmitted from the ground support radar APW-11 data link. The altitude and heading of the airplane are held to the values existing at time of engagement of this autopilot function. The MACH position holds the mach number and heading of the airplane to values existing at the time of engagement of this autopilot function. When positioned to APW-11 position the airplane heading is controlled by ground command signals. When positioned to ARR-39 position the airplane flight path, both course and altitude, are controlled by ground command AGCA signals transmitted through the ARR-39 data link.

AUTOPILOT OPERATION

Power is applied to the autopilot system when the d-c secondary and a-c secondary buses are energized. This starts the gyro motors, warms up the amplifiers and puts the system in a standby condition. The stability augmentation mode (DAMPERS switch) may be engaged 30 seconds after application of power to the autopilot system. Anyone of the four methods of autopilot control may be engaged two to two and a half minutes after application of power to the system. To engage the autopilot the dampers switch must previously have been engaged, selection of the desired method of control by means of the mode selector switch then depressing the autopilot switch. Changes in the mode of control selected may be made after autopilot engagement by use of the mode selector switch. Failure of either push-pull switch to lock in the down (or engaged) position is indicative of a loss of power or other malfunction of the autopilot system or that the attitude is outside the specified limits, and automatically prevents engagement of the particular control involved. Similarly, after engagement, loss of power or certain malfunctions of the system will result in automatic disengagement of the system and return of the push-pull switch to the up (disengaged) position. To completely disengage the autopilot in an emergency the emergency disengage switch, mounted on the forward side of the control stick, is actuated. This switch may be operated by the pilot without removing his hand from the control stick. To re-engage the damper and autopilot functions after emergency disengagement, the 30 second and two to two and one half minute time delays must still be observed.

NAVIGATION EQUIPMENT

SLAVED GYRO MAGNETIC COMPASS

A type J-2 slaved gyro magnetic compass is installed in the airplane which provides visual indication of the magnetic heading of the airplane. The indication is read on as indicator (4, figure 1–3) whose operation is governed by a gyro whose spin axis is stabilized in a horizontal plane by means of a leveling device and whose orientation in azimuth is slaved to the earth's magnetic meridian by a direction-sensing component, located in the left stabilizer. The compass requires both a-c and d-c power. The d-c power is supplied from the primary bus and the a-c power is supplied by the secondary a-c bus. The gyro is free to operate within 85 degrees from level flight in dive and climb, and in right and left bank. At the limits, it strikes mechanical stops, which render the indications on the directional gyro control and the settable dial indicator inaccurate. After return to level flight, errors up to 5 degrees in heading may be introduced; but the gyro will recover its erect and slaved positions automatically, in a period of 5 minutes or less, and thereafter will again resume correct indications until the limits are again exceeded. The flux valve unit of the remote compass transmitter remains pendulous through 30 degrees on both sides of the vertical, in pitch and roll. When these limits are exceeded, or a coordinated turn is being executed, the vertical components of the earth's field are picked up which results in flash signals. Restoration of the airplane to an attitude within these limits renders the flux valve unit pendulous again, and it automatically resumes correct sensing. A thermal switch in the amplifier provides fast slaving and leveling of the directional gyro, during the initial operation of the compass.

GYRO COMPASS SLAVING SWITCH

The gyro compass slaving switch has two positions: IN and OUT. The IN position supplies power to the heating, leveling and slaving systems. The OUT position cuts off the power supply to the control field of the slaving torque motor, and is used when the horizontal lines of magnetic force dip at 84 degrees or more.

GYRO COMPASS FAST SLAVING SWITCH

The gyro compass fast slaving switch marked PUSH FOR FAST SLAVING, is pushed in momentarily to shorten the time required to restore the gyro to its erect and slaved position, after level flight is resumed, following maneuvers in which the gyro has hit the mechanical stops. Approximately three minutes of fast slaving is obtained by depressing the fast slaving switch. The fast slaving switch is also used during initial starting of the compass to assure both a-c and d-c power are supplied to the system simultaneously.

Note

After the fast slaving switch is pressed, a time delay circuit maintains the fast slave action for approximately three minutes. During this three minute interval, any maneuvering of the airplane can induce errors into the equipment. At the completion of the three minutes, the system normally reverts to slow slave and the large errors which have been introduced will remain for a considerable time. Therefore, the fast slaving switch should not be used during flight, except when the airplane can be maintained in straight flight for at least three minutes after the fast slaving switch is depressed.

STARTING

The compass will operate when the d-c primary and a-c secondary busses are energized. Allow 3 minutes to elapse so that the gyro in the directional gyro control comes up to operating speed, levels and aligns the indication on the settable dial indicator with that sensed by the remote compass transmitter.

Note

It is necessary, for proper operation of the J-2 compass, that the a-c and d-c power supplies to the system be turned on simultaneously. To assure this, depress the fast slaving switch momentarily after the engine is running. This action turns the d-c power (to the compass) off and on again with the inverter running.

OPERATION

SETTING INDICATOR

By means of the SET COURSE knob on the indicator, set the dial index for the heading it is desired to fly. It is preferable to set the dial index against the zero bezel index of the indicator, although any index may be chosen.

USING THE COMPASS

STRAIGHT FLIGHT

After the airplane becomes airborne, the indicator is referred to in the same manner as a magnetic compass.

IN TURNS

Perfect 45, 90 and 180 degree turns can be executed by setting the dial index, with the overlapping pointer against the zero bezel index, then flying the aircraft to align the pointer with the 45 and 90 degree bezel indices on both sides of the zero index, or with the index at 180 degrees. The final heading may be set against the zero bezel index by means of the SET COURSE knob. Another method is to set the dial index for the new heading against any bezel index, then flying the aircraft to align the pointer with that bezel index.

STANDBY COMPASS

Refer to INSTRUMENTS in Section I.

RADIO COMPASS

Refer to Communications and Associated Electronic Equipment in this Section.

LATITUDE AND LONGITUDE COMPUTER SET

Refer to Communications and Associated Electronic Equipment in this Section.

MISCELLANEOUS EQUIPMENT

ANTI-G SUIT PROVISIONS

An air pressure outlet connection on the front of the pilot's seat provides for the attachment of the air pressure intake tube of the pilot's anti-g suit. Air pressure for inflation of the anti-g suit bladder is conducted from the engine compressor through a pressure regulating valve (10, figure 1–5) located on the left console. The valve opens under a g loading of 1.5 to 2.0 g and will allow pressures to be applied to the suit with increases in g-load approximately as follows: at 2g from 0 to 1.2 psi; at 4g from 2.9 to 4.2 psi; at 6G from 5.85 to 7.3 psi; and at 8g from 8.7 to 10.4 psi. When the acceleration decreases below the valve opening g setting, the valve closes and exhausts the suit pressure into the cockpit. The relief valve will prevent the pressure in the suit from exceeding 11 psi.

PILOT'S LIQUID SUPPLY

A one quart thermos-type liquid container (12, figure 1–5) is stowed at the aft end of the left hand console. The container is held in position by a rubber-lined plate operated by a spring-loaded-to-open linkage. The liquid container is inserted through the aft hole in the panel where guides seat the container on the rubber base and clips mounted on the cockpit floor. When the links are pulled upward from their locked position, the springs pull the rubber-lined plate upward to release the container from its stowed position.

PILOT'S RELIEF PROVISIONS

The pilot's relief provisions (11, figure 1–5) consist of a horn terminating in a removable pint container. The horn contains a spring-loaded double action valve. When the valve lever is actuated, it provides for pressure equalization, usage and air escape. In the normal position of the valve, the air escape and usage valve sections will be closed. The valve and bottle assembly is mounted on the left console forward of the liquid container. It is retained in position by the pressure of a sponge rubber pad attached to the cockpit floor after the lugs on the valve are inserted in the slots in the mounting hole and the assembly turned counterclockwise to engage the lugs underneath the panel.

SECTION V
OPERATING LIMITATIONS

TABLE OF CONTENTS

BALANCE REQUIREMENTS

1. It is possible to exceed the center of gravity range of the aircraft with various store configurations due to the great variety of configurations. Therefore, each configuration should be examined separately before flight.

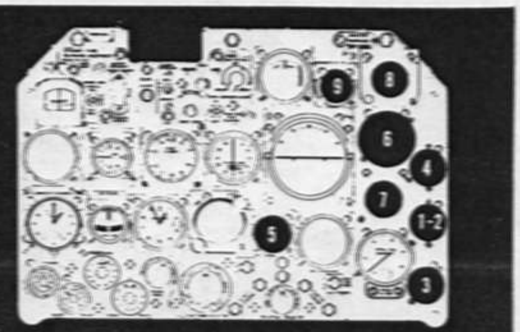
2. There is no way of determining the fuel distribution when partially filling the fuel tanks, due to single point refueling. Therefore, all internal fuel tanks, exclusive of the bomb bay tank, must be full prior to take-off.

3. In the event the stores are dropped and ammunition expanded and the 60 gallons reserve fuel is required for landing, the external fuel tanks must be released from the aircraft prior to landing.

4. With failure of both aft fuel tank pumps, an aft center of gravity beyond the permissible range for landing may be experienced. Therefore *EXTREME CAUTION SHOULD BE EXERCISED ON SUCH LANDINGS.*

5. When ammunition is not carried in the aircraft, such as a ferry mission, 460 pounds of ballast will be required in the ammunition boxes.

INSTRUMENT MARKINGS



Hydraulic Pressure

Controls System No. One Pressure

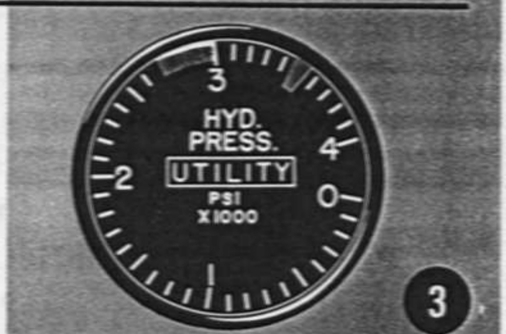
2700-3000 psi *Normal*
3500 psi *Maximum*

Controls System No. Two Pressure

2700-3000 psi *Normal*
3500 psi *Maximum*

Utility System Pressure

2700-3000 psi *Normal*
3500 psi *Maximum*



Oil Pressure

30 psi *Min during idle*
40-50 psi *Normal*
50 psi *Maximum*

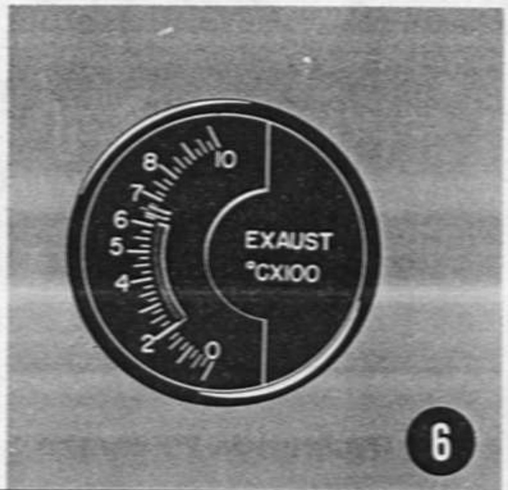
Tachometer

104% *Maximum permissible*

Figure 5–1 (Sheet 1 of 2).

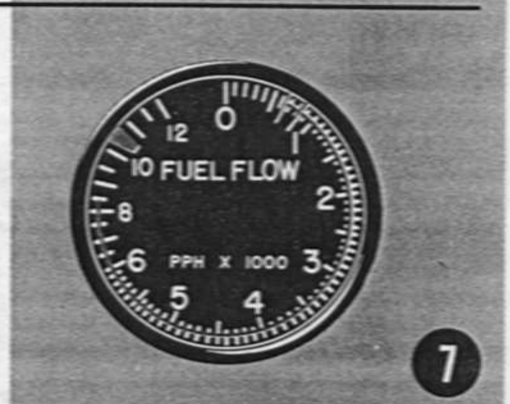
Exhaust Gas Temperature

200°C	Minimum
200°C-610°C	Continuous above 35,000 ft
580°C	Continuous below 35,000 ft
630°C	Max thrust limit below 35,000 ft — 5 min ground limit 15 min flight limit and starting.
620°C	Military limit below 35,000 ft 30 minutes
660°C	Max afterburner limit above 35,000 ft 15 minute flight limit
650°C	Military limit above 35,000 ft 30 minute limit
670°C	Max during acceleration Max 2 minute duration

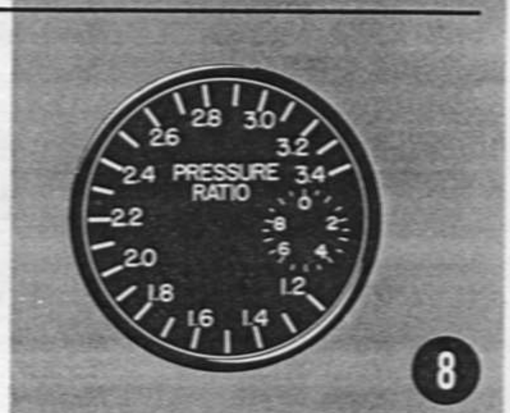


Fuel Flow Indicator

650#/hr	Idle (for air relight fuel flow may be 300 lb/hr below 650
650-8500#/hr	Continuous operation
850#/hr	Minimum during start
10,500#/hr	Maximum and military power limit



Pressure Ratio Indicator



Accelerometer

7.0g	Maximum subsonic
4.0g	Maximum supersonic
—2.0g	Maximum

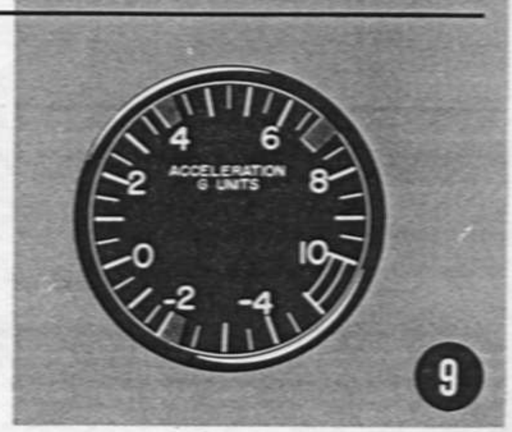


Figure 5—1 (Sheet 2 of 2)

ACCELERATION LIMITATIONS

The following limit load factors are 80 per cent of Design limits. These limits will be adhered to until completion of static and structural integrity tests have been completed.

	"g" limit POSITIVE	"g" limit NEGATIVE
Design gross wt 31,392 lb		
Subsonic	7.5	–3.0
Supersonic	5.9	–2.4
Design gross wt 33,070 lb		
Subsonic	5.9	–2.4
Supersonic	5.9	–2.4
Maximum Internal Bomb Load + Maximum External Fuel Load	3.2	–1.6
Maximum Internal and External Fuel	3.2	–1.6
Maximum Internal Fuel + Maximum External GP Bombs	3.2	–1.6

limit ground load factors

Maximum Take-off Gross Weight	44,612 lb
Maximum Landing Gross Weight	29,227 lb
Maximum Landing Gross Weight after aborted take-off	37,964 lb

Figure 5–2

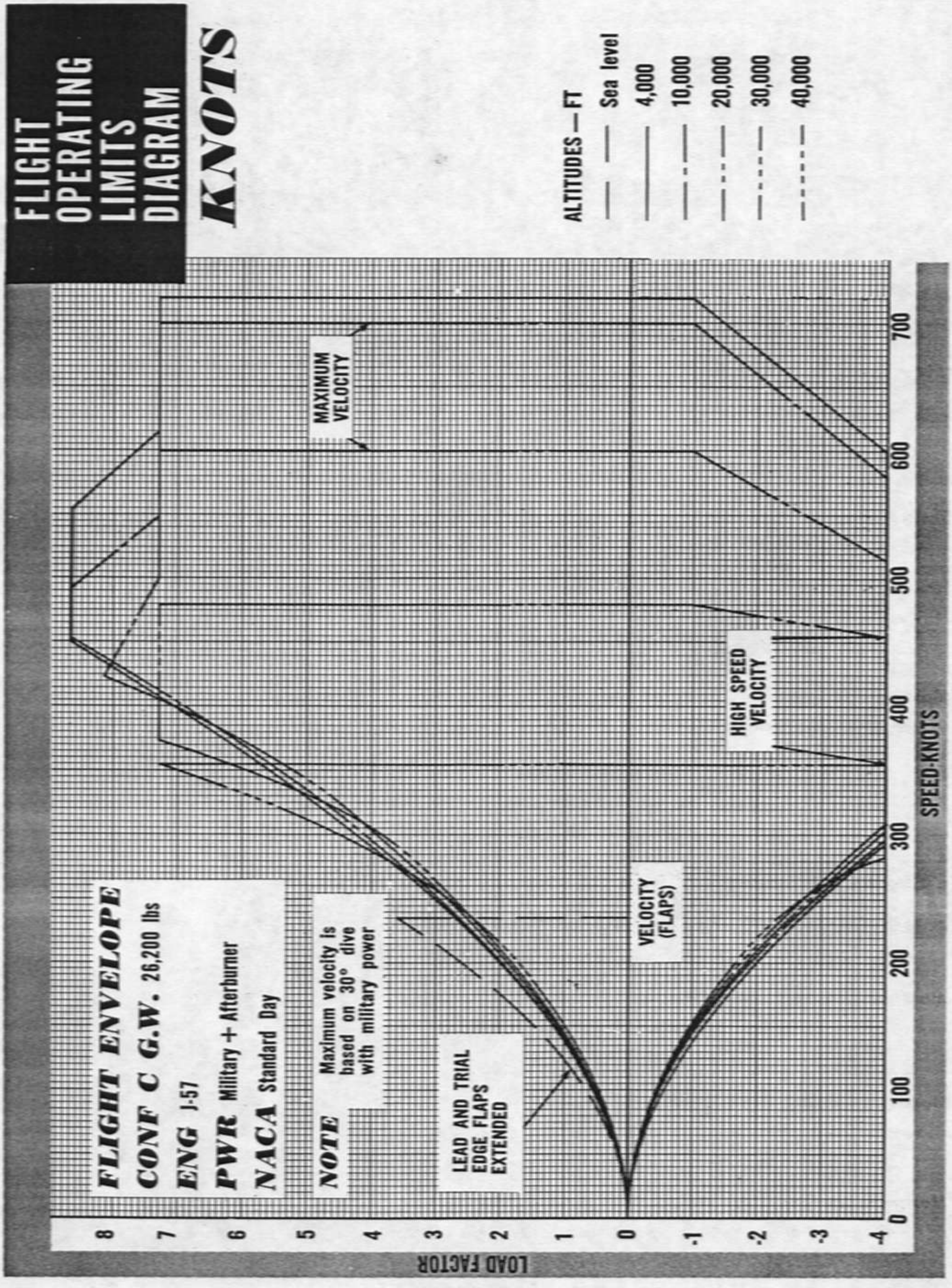

FLIGHT OPERATING LIMITS DIAGRAM
KNOTS
ALTITUDES — FT
Sea level
4,000
10,000
20,000
30,000
40,000
FLIGHT ENVELOPE
CONF C G.W. 26,200 lbs
ENG J-57
PWR Military + Afterburner
NACA Standard Day
NOTE
Maximum velocity is based on 30° dive with military power
LEAD AND TRIAL EDGE FLAPS EXTENDED
MAXIMUM VELOCITY
HIGH SPEED VELOCITY
VELOCITY (FLAPS)
LOAD FACTOR
8
7
6
5
4
3
2
1
0
-1
-2
-3
-4
SPEED-KNOTS
0
100
200
300
400
500
600
700

Figure 5—3

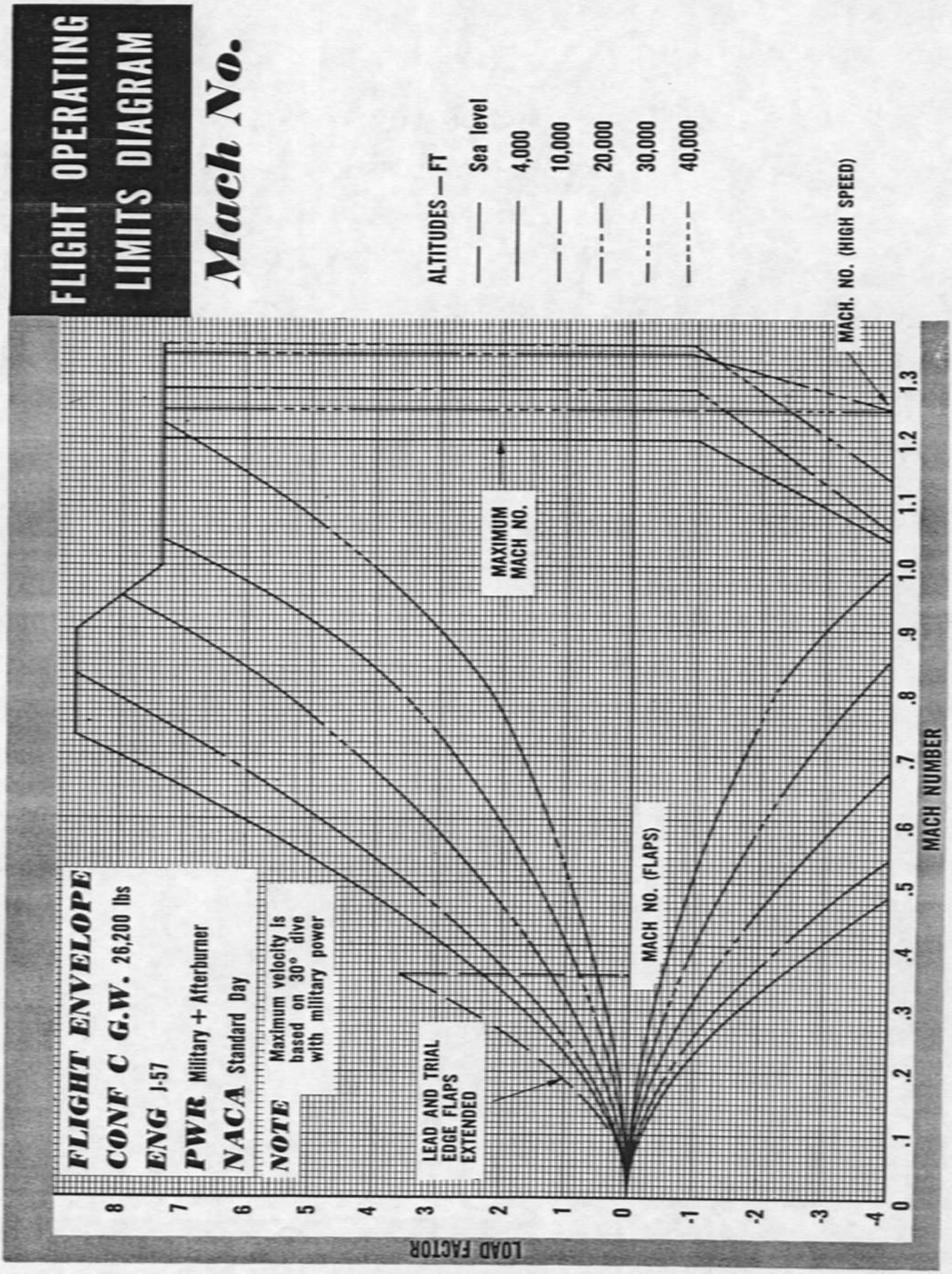

FLIGHT OPERATING LIMITS DIAGRAM
Mach No.
ALTITUDES — FT
Sea level
4,000
10,000
20,000
30,000
40,000
FLIGHT ENVELOPE
CONF C G.W. 26,200 lbs
ENG J-57
PWR Military + Afterburner
NACA Standard Day
NOTE
Maximum velocity is based on 30° dive with military power
LEAD AND TRIAL EDGE FLAPS EXTENDED
MACH NO. (FLAPS)
MAXIMUM MACH NO.
MACH. NO. (HIGH SPEED)
LOAD FACTOR
8
7
6
5
4
3
2
1
0
-1
-2
-3
-4
MACH NUMBER
0
.1
.2
.3
.4
.5
.6
.7
.8
.9
1.0
1.1
1.2
1.3

Figure 5—4

SECTION VI

FLIGHT CHARACTERISTICS

(WILL BE SUPPLIED WHEN AVAILABLE.)

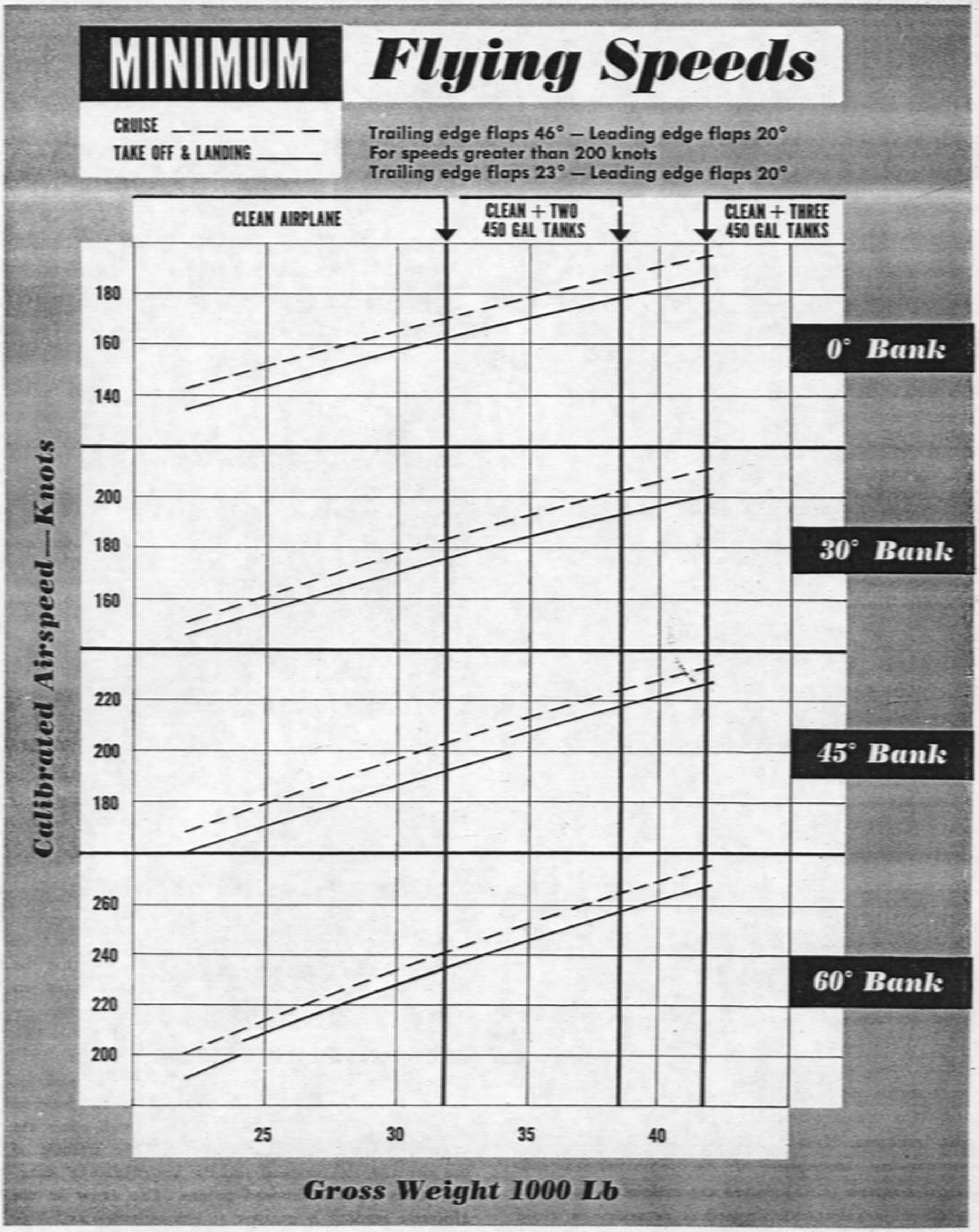

MINIMUM Flying Speeds
CRUISE
TAKE OFF & LANDING
Trailing edge flaps 46° — Leading edge flaps 20°
For speeds greater than 200 knots
Trailing edge flaps 23° — Leading edge flaps 20°
CLEAN AIRPLANE
CLEAN + TWO 450 GAL TANKS
CLEAN + THREE 450 GAL TANKS
0° Bank
30° Bank
45° Bank
60° Bank
Calibrated Airspeed—Knots
Gross Weight 1000 Lb
140
160
180
200
220
240
260
25
30
35
40

Figure 6–1

APPENDIX I

PERFORMANCE DATA

TABLE OF CONTENTS

INTRODUCTION.

To facilitate preflight and in-flight mission planning, two standard types of operating data are presented in this Section. The first type provides altimeter and airspeed position error corrections. The second type provides performance during normal operation, with various weight configurations and the instructions necessary to obtain this performance. All data pertain to NACA standard ambient temperatures unless otherwise indicated. Most of the charts are applicable to non-standard atmosphere if the recommended calibrated airspeed (CAS) valves are maintained unless a deviation in calibrated airspeed is necessary to avoid violating engine limits. This rule is necessary because performance is greatly dependent on Mach number which, at each pressure altitude, is dependent on CAS alone. Fuel quantities are given in pounds so that the charts can be used when the engine is operated on either the recommended or emergency alternate fuel.

ALTIMETER CORRECTION CHART.

Static pressure, which affects the altimeter readings, is not always accurately measured due to the location of the static vent. Additional factors which effect the degree of error in the altimeter are the attitude of the airplane, the airspeed and the installation of external tanks on the inboard pylons. The error in the altimeter reading is greatest at low altitudes and high speeds making it more critical during ground support missions. The altimeter position error correction chart (figure A–1) is provided so that for a given altitude, airplane loading configuration and airspeed the altimeter error can be determined. The error is added algebraically to the altimeter reading to obtain the true pressure altitude.

AIRSPEED INSTALLATION CORRECTION CHARTS.

In order to obtain true airplane speeds several corrections must be applied to the airspeed indicator reading. The first correction is made for the error in the individual instrument. This value is noted in the instrument calibration card and when applied to the instrument reading provides indicated airspeed (IAS). The second correction is for airspeed installation error. This correction is taken from figure A–2, and when applied to the indicated airspeed (IAS) provides calibrated airspeed (CAS). The third correction is taken from figure A–3 and when applied to calibrated airspeed provides equivalent airspeed (EAS). Dividing the equivalent airspeed by the square root of relative density (ratio of ambient to standard sea level density) provides true airspeed (TAS). Vectorially adding wind velocity to true airspeed provides ground speed (GS).

MACH NUMBER POSITION ERROR CORRECTION CHART.

Due to the inaccuracies of the static pressure position errors will be carried into the Machmeter. The Mach number position error correction chart (figure A–4) indicates the amount of error for various speeds in terms of Mach No.

TAKE-OFF CHARTS.

Take-off chart (figure A–6) is provided for no assist, with afterburner in the take-off configuration. The chart takes into account ambient temperature, pressure altitude and gross weight. Take-off speed, ground roll and distance to clear a 50 ft obstacle with or without a headwind are obtained from the charts.

EXAMPLE: WITH AFTERBURNER NO ASSIST.

Determine take-off speed, ground roll and distance to clear 50 ft obstacle for the following configuration:

Airplane gross weight	29,000 lbs
Pressure altitude	2,000 feet
Air temperature	35°C
Headwind	0 and 30 knots

Procedure:

Enter figure A–6 at air temperature (A) and move up to pressure altitude curve (B). Continue horizontally across to airplane gross weight (C) and vertically to ground roll distance (E). Drop to 0 wind curve (F) and horizontal to distance to clear 50 ft obstacle (G). For a 30 knot headwind, drop from airplane gross weight (C) to (D) then follow curve to headwind (H) then drop to ground roll distance (J). Continue from (J) vertically to 30 knots headwind curve (K) and horizontal to distance to clear 50 ft obstacle (L).

Conclusion:

Take-off speed (C) 150 knots CAS

Ground roll distance (E) 3900 feet (zero wind)

Distance to clear 50 ft obstacle (G) 5000 ft (zero wind)

Ground roll distance (J) 2600 feet (30 knot headwind)

Distance to clear 50 ft obstacle (L) 3500 ft (30 knot headwind)

CLIMB GRAPHS.

Climb charts for afterburner power, military power and normal power, based on a recommended climb speed schedule are shown for each configuration. Time and distance are plotted against gross weight with guide lines to show the reduction in gross weight during climb due to the fuel used.

USE.

To obtain the climb data desired, enter the proper climb chart at the gross weight and altitude at start of climb. Note the time and distance at this point. From this initial altitude point, trace a curve parallel to the guide lines until it intersects the desired altitube at end of climb. Note the time, distance, and gross weight at this intersection. The difference between the initial and final time is the time required to climb. The difference between initial and final values for distance and for gross weight gives, respectively, the distance traveled and fuel used to climb. Since time and distance are zero at level, the time required and distance traveled may be read directly for climbs starting at sea level. Fuel used, however, must still be determined by the difference in gross weights.

OUTBOUND PROFILE CHARTS.

These charts give the relationship of time, fuel, distance, and altitude to maximum range for no-wind conditions. This relationship is based on a mission sequence of take-off, military thrust climb, and maximum range cruise. The fuel curves include an allow-

(WILL BE SUPPLIED WHEN AVAILABLE.)

Figure A—1. Altimeter Position Error Correction Chart

(WILL BE SUPPLIED WHEN AVAILABLE.)

Figure A—2. Airspeed Position Error Correction Chart

COMPRESSIBILITY

correction table

Subtract Correction From Calibrated Airspeed To Obtain Equivalent Airspeed

PRESSURE ALTITUDE	CAS-KNOTS 150	200	250	300	350	400	450	500	550	600	650	700
5,000	0	0	1	2	2	3	5	6	8	10	12	14
10,000	0	1	2	3	5	7	10	13	17	21	24	26
15,000	1	2	3	5	8	12	17	23	31	32	36	40
20,000	1	3	5	8	12	17	23	31	37	43	47	50
25,000	2	4	7	11	17	24	32	41	48	54	57	59
30,000	2	5	9	15	23	32	41	50	58	63	66	68
35,000	3	7	12	20	29	40	50	59	67	72	75	76
40,000	4	9	16	25	37	48	59	68	75	80	82	
45,000	5	11	20	32	44	56	66	74	82	88		
50,000	7	14	25	38	51	62	72	81	89			

example:

During flight at 25,000 feet and 250 knots CAS, correction factor is 7 knots. Subtract 7 from 250 knots CAS to obtain 243 knots equivalent airspeed (EAS).

Figure A–3

(WILL BE SUPPLIED WHEN AVAILABLE.)

Figure A–4. Mach Number Position Error Correction Chart

STANDARD ALTITUDE TABLE

Standard Sea Level Air:
T = 15° C. W = .07651 lb/cu. ft. ρ_o = .002378 slugs/cu. ft.
P = 29.921 in. of Hg. 1" of Hg. = 70.732 lb/sq. ft. = 0.4912 lb/sq. in.
This table is based on NACA Technical Report No. 218 a_o - 1116 ft./sec.

Altitude feet	Density Ratio ρ/ρ_o	$\frac{1}{\sqrt{\sigma}}$	Temperature Deg. C	Temperature Deg. F	Speed of Sound Ratio a/a_o	Pressure In. of Hg.	Pressure Ratio P/Po
0	1.0000	1.0000	15.000	59.000	1.0000	29.92	1.0000
1000	.9710	1.0148	13.019	55.434	.997	28.86	.9644
2000	.9428	1.0299	11.038	51.868	.993	27.82	.9298
3000	.9151	1.0454	9.056	48.301	.990	26.81	.8962
4000	.8881	1.0611	7.075	44.735	.986	25.84	.8636
5000	.8616	1.0773	5.094	41.169	.983	24.89	.8320
6000	.8358	1.0938	3.113	37.603	.979	23.98	.8013
7000	.8106	1.1107	1.132	34.037	.976	23.09	.7716
8000	.7859	1.1280	-0.850	30.471	.972	22.22	.7427
9000	.7619	1.1456	-2.831	26.904	.968	21.38	.7147
10000	.7384	1.1637	-4.812	23.338	.965	20.58	.6876
11000	.7154	1.1822	-6.793	19.772	.962	19.79	.6614
12000	.6931	1.2012	-8.774	16.206	.958	19.03	.6359
13000	.6712	1.2206	-10.756	12.640	.954	18.29	.6112
14000	.6499	1.2404	-12.737	9.074	.950	17.57	.5873
15000	.6291	1.2608	-14.718	5.507	.947	16.88	.5642
16000	.6088	1.2816	-16.699	1.941	.943	16.21	.5418
17000	.5891	1.3029	-18.680	-1.625	.940	15.56	.5202
18000	.5698	1.3247	-20.662	-5.191	.936	14.94	.4992
19000	.5509	1.3473	-22.643	-8.757	.932	14.33	.4790
20000	.5327	1.3701	-24.624	-12.323	.929	13.75	.4594
21000	.5148	1.3937	-26.605	-15.890	.925	13.18	.4405
22000	.4974	1.4179	-28.586	-19.456	.922	12.63	.4222
23000	.4805	1.4426	-30.568	-23.022	.917	12.10	.4045
24000	.4640	1.4681	-32.549	-26.588	.914	11.59	.3874
25000	.4480	1.4940	-34.530	-30.154	.910	11.10	.3709
26000	.4323	1.5209	-36.511	-33.720	.906	10.62	.3550
27000	.4171	1.5484	-38.493	-37.287	.903	10.16	.3397
28000	.4023	1.5768	-40.474	-40.853	.899	9.720	.3248
29000	.3879	1.6056	-42.455	-44.419	.895	9.293	.3106
30000	.3740	1.6352	-44.436	-47.985	.891	8.880	.2968
31000	.3603	1.6659	-46.417	-51.551	.887	8.483	.2834
32000	.3472	1.6971	-48.399	-55.117	.883	8.101	.2707
33000	.3343	1.7295	-50.379	-58.684	.879	7.732	.2583
34000	.3218	1.7628	-52.361	-62.250	.875	7.377	.2465
35000	.3098	1.7966	-54.342	-65.816	.871	7.036	.2352
36000	.2962	1.8374	-55.000	-67.000	.870	6.708	.2242
37000	.2824	1.8818	-55.000	-67.000	.870	6.395	.2137
38000	.2692	1.9273	-55.000	-67.000	.870	6.096	.2037
39000	.2566	1.9738	-55.000	-67.000	.870	5.812	.1943
40000	.2447	2.0215	-55.000	-67.000	.870	5.541	.1852
41000	.2332	2.0707	-55.000	-67.000	.870	5.283	.1765
42000	.2224	2.1207	-55.000	-67.000	.870	5.036	.1683
43000	.2120	2.1719	-55.000	-67.000	.870	4.802	.1605
44000	.2021	2.2244	-55.000	-67.000	.870	4.578	.1530
45000	.1926	2.2785	-55.000	-67.000	.870	4.364	.1458
46000	.1837	2.3332	-55.000	-67.000	.870	4.160	.1391
47000	.1751	2.3893	-55.000	-67.000	.870	3.966	.1325
48000	.1669	2.4478	-55.000	-67.000	.870	3.781	.1264
49000	.1591	2.5071	-55.000	-67.000	.870	3.604	.1205
50000	.1517	2.5675	-55.000	-67.000	.870	3.436	.1149

Figure A–5. Standard Altitude Table

ance for start, taxi and take-off, and the fuel used in climb for each altitude, as well as the fuel required for maximum range cruise. The time lines include the time required to climb to cruise altitude but do not include the time to start, taxi, or take-off. The initial climb path shows the distance traveled during the military thrust climb from sea level to cruising altitude, using the climb speed schedule shown on the Military Thrust Climb graph. The continuation of the initial climb path is the optimum cruise flight path. Level off at the altitude shown for maximum range. Maintain constant Mach number reducing power setting as required. After sufficient fuel has been consumed, climb at the recommended military thrust climb schedule to the next higher 5000 foot level, set recommended cruise Mach number and cruise out. Repeat climb and cruise as shown. For a cruise at a constant altitude, set up the recommended Mach number at the intersection of the climb path and the cruise altitude. As the flight progresses, the power setting must be decreased gradually as fuel is consumed, to maintain the recommended Mach number. As an aid to preflight planning, a line of best range for constant-altitude flight appears on the chart (white dashed line). This curve is not a flight path, but a plot of best cruise altitude against distance. For distances greater than those covered by the curve, use step-climb procedure for maximum range. A cruise table gives recommended Mach numbers and approximate operating conditions for both cruise-climb procedure and for cruise at constant altitude. Cruise-at-constant-altitude data is given for each 5000 feet.

OPTIMUM RETURN PROFILE CHARTS.

These charts show the minimum fuel required for maximum distance (no wind) based on an optimum flight path from any starting point within the range of the airplane configuration. The flight path required is indicated by the different shaded areas and the notes relative to them. The fuel curves are based on a military thrust climb to, and recommended cruise at the optimum altitude. The military thrust climb speed schedule and recommended cruise settings are tabulated on each chart. No reserve for descent and landing has been included. The time shown at the optimum altitude is cruise time only; it does not include the time required for the climb to optimum altitude or any allowance for descent, loiter and landing. The chart may be entered at the initial altitude with either the fuel on board (to determine the distance available) or with the distance to be flown (to determine the fuel required). The shaded area in which the initial point falls, establishes the cruising procedure to be used, as stated in the note relative to the area. The time required to fly the distance is the time at cruise altitude (obtain from profile), plus the time required to climb (obtained from either profile or graphical military thrust climb chart). The effect of wind must be applied to obtain the actual fuel and time to fly the distance. A close approximation can be obtained by considering the head or tail wind for the time it requires to complete the flight (neglecting the difference in wind at the lower altitudes since comparatively little time is spent during the climb phase).

MAXIMUM ENDURANCE PROFILE CHART.

These charts show the maximum time available for the fuel on board when loitering at a constant altitude. The recommended calibrated airspeed (CAS) and the approximate operating conditions are tabulated on each chart for weights covered by fuel quantities shown. To determine the time available for a given amount of fuel: Enter the chart at the amount of fuel on board at the start of loiter and the flight altitude; note the initial time. Re-enter the chart at the amount of fuel on board at the end of the endurance flight (initial fuel on board less fuel to be used) and read the final time. The difference between the initial and final time is the time available to loiter at constant altitude. To obtain the fuel required to loiter a given time: Enter the chart at the amount of fuel on board at the start of loiter and flight altitude; note the initial time. Re-enter the chart at the time at the end of loiter (initial time less time to loiter) and read final fuel on board. The difference between the initial and final fuel on board is the fuel required to loiter.

DESCENTS.

Two types of descents are shown for two configurations of the airplane; the recommended descent (with idle power), the maximum range descent (with idle power). Distance, time, fuel consumed and rate of descent are shown on the graphs.

LANDING DISTANCES.

Distances, ground and total to clear a 50 ft obstacle, with or without the drag chute inflated are shown in the landing distance graph. In both graphs, the distances are computed for the speed brake closed.

COMBAT ALLOWANCE CHART.

The Combat Allowance Chart shows the relationship between time and fuel with changes in altitude at military power and normal power. Combat time or fuel may be determined from this chart for a given thrust setting. The time limitations for military thrust operation are shown. Normal thrust does not have a time limitation.

NAUTICAL MILES PER POUND OF FUEL GRAPHS.

Cruise data (zero wind) throughout the speed range from maximum endurance to military thrust are shown on the Nautical Miles per pound of Fuel Graphs. Several weights for each configuration are given at altitudes of sea level, 15000, 25000, 35000, and 40000 feet. Each graph includes specific range (nautical miles per pound) fuel flow, and power settings. Also included are curves of recommended cruise true Mach number, maximum endurance, and normal and military thrust. Specific range is plotted against true Mach number. Cruising range is the product of specific range multiplied by fuel amount. Large fuel amounts should be divided into several smaller amounts. The smaller the amount of fuel used in the calculation, the greater the accuracy of the range. To obtain the cruising range for a given amount of fuel, use the following steps. (If several fuel amounts are being used, repeat the steps for each. The sum of the individual ranges is the total cruising range.)

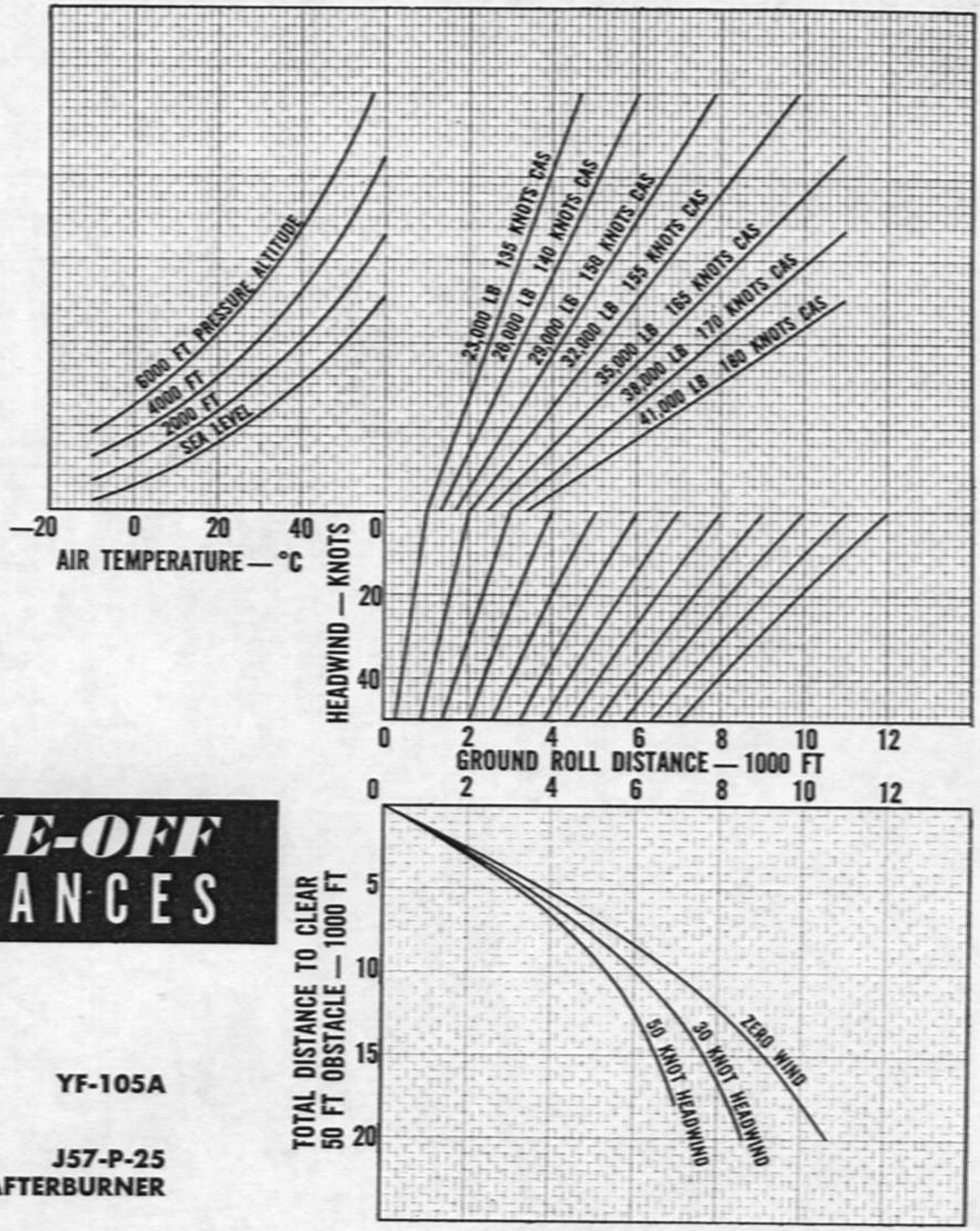
a. Select the proper graph for the airplane configuration and altitude.

b. Determine the average weight of the airplane for the amount of fuel being considered.

c. Enter the graph at this average weight and the desired true Mach number, or desired power setting, to obtain specific range (nautical miles per pound of fuel).

d. The specific range multiplied by the amount of fuel (pounds) equals the cruising range.

e. Interpolate the approximate fuel flow and power setting at the true Mach number and average weight. When there is a wind to be considered, multiply the specific range found in step c by the range factor (ground speed divided by true airspeed) to obtain the specific range for wind. Proceed with steps d and e to complete the problem.



TAKE-OFF DISTANCES

Model: YF-105A

Engine: J57-P-25 WITH AFTERBURNER

NO ASSIST

HARD SURFACE RUNWAY

Figure A–6

AFTERBURNER POWER *Climb*

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN

CLIMB SPEEDS		
ALT. FT.	CAS—KNOTS	MACH NO.
SEA LEVEL	600	.91
10,000	515	.91
20,000	435	.92
30,000	360	.93
35,000	325	.94
40,000	290	.94

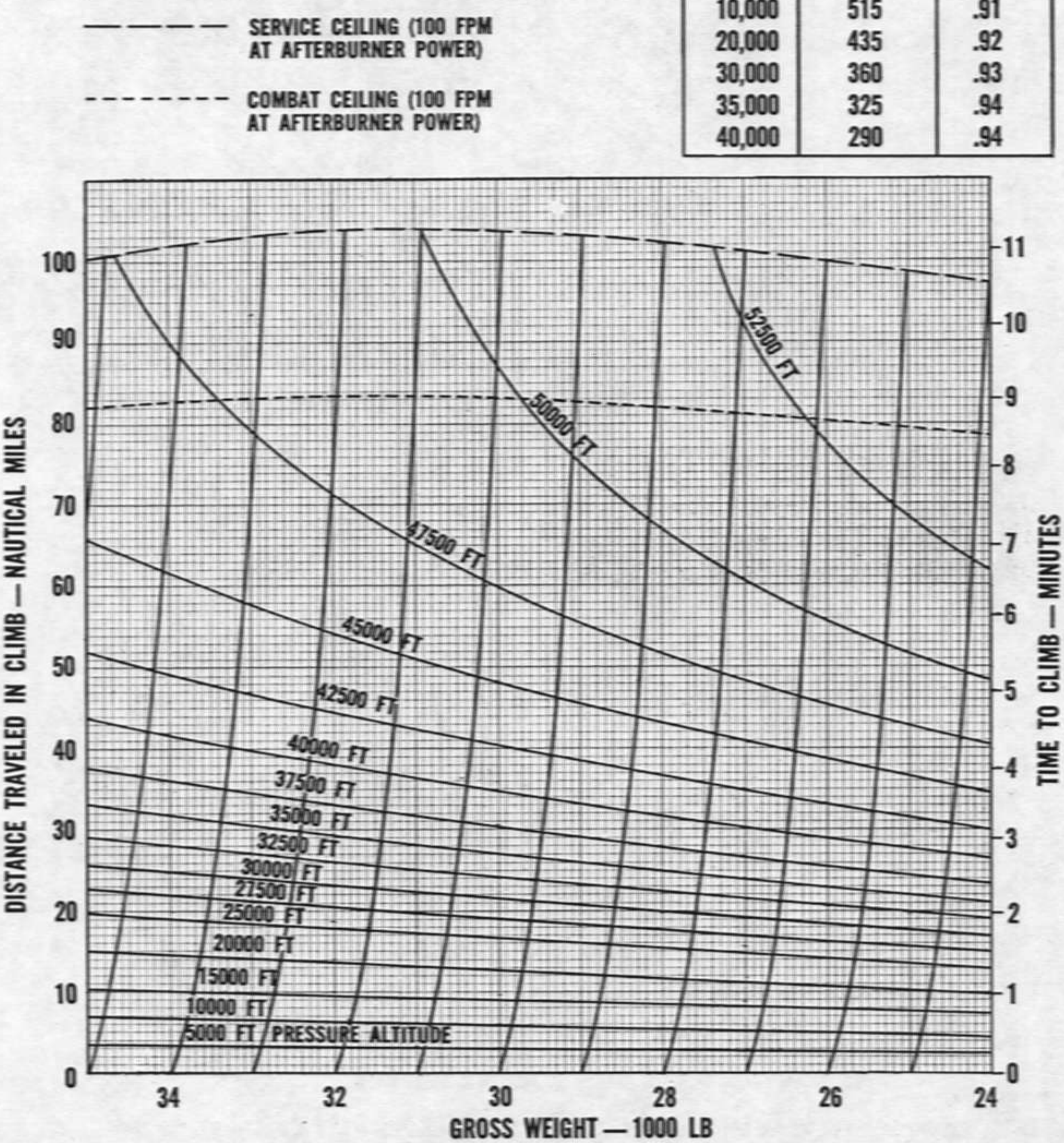


Figure A—7

MILITARY POWER *Climb*

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN

CLIMB SPEEDS		
ALT. FT.	CAS—KNOTS	MACH NO.
SEA LEVEL	375	.57
10,000	355	.64
20,000	330	.72
30,000	305	.81
35,000	280	.82
40,000	250	.83

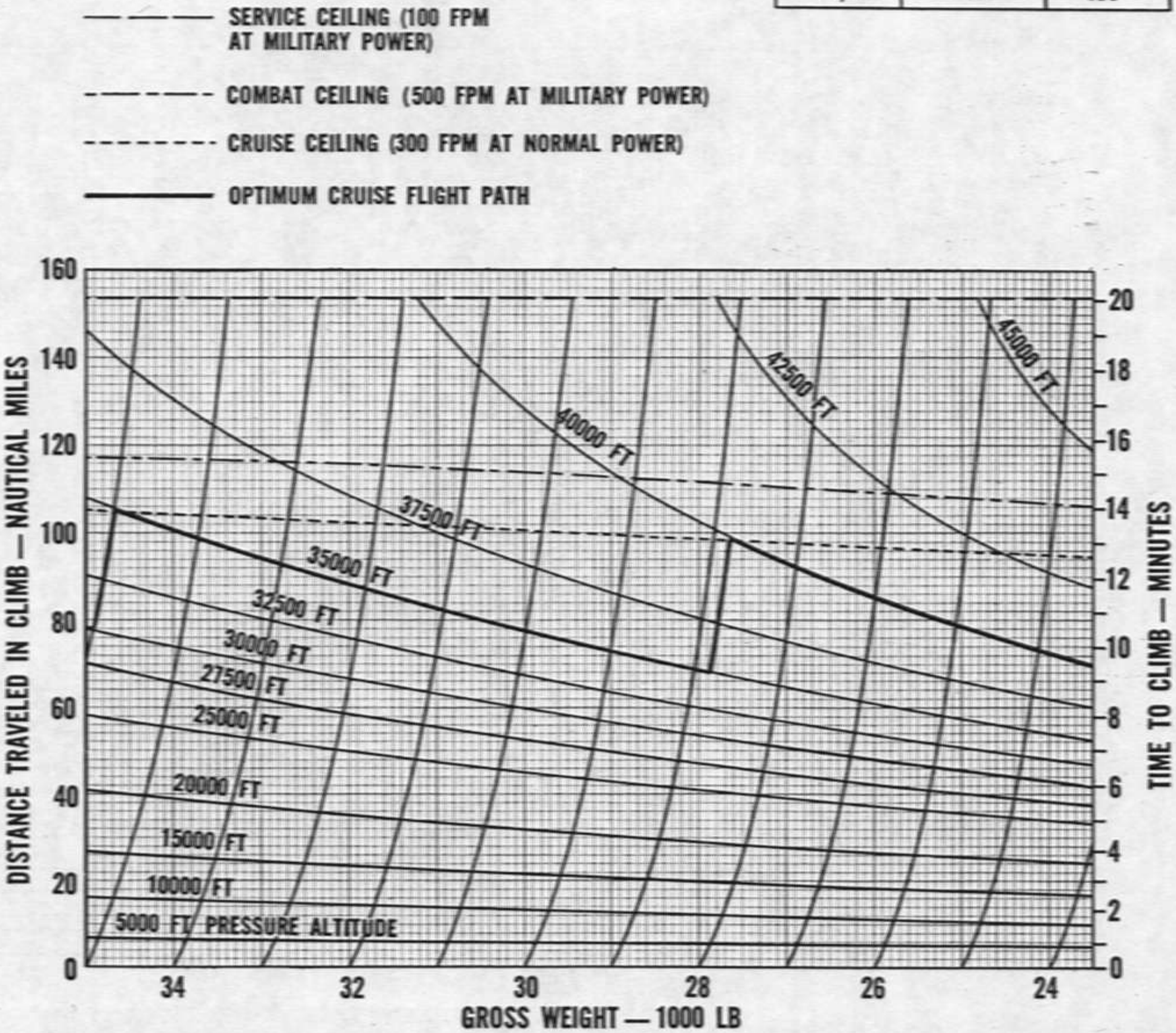


Figure A–8

NORMAL POWER Climb

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN

CLIMB SPEEDS		
ALT. FT.	CAS—KNOTS	MACH NO.
SEA LEVEL	325	.49
10,000	315	.57
20,000	305	.67
30,000	290	.76
35,000	265	.78
40,000	235	.78

— — — — SERVICE CEILING (100 FPM AT NORMAL POWER)

- - - - - - CRUISE CEILING (300 FPM AT NORMAL POWER)

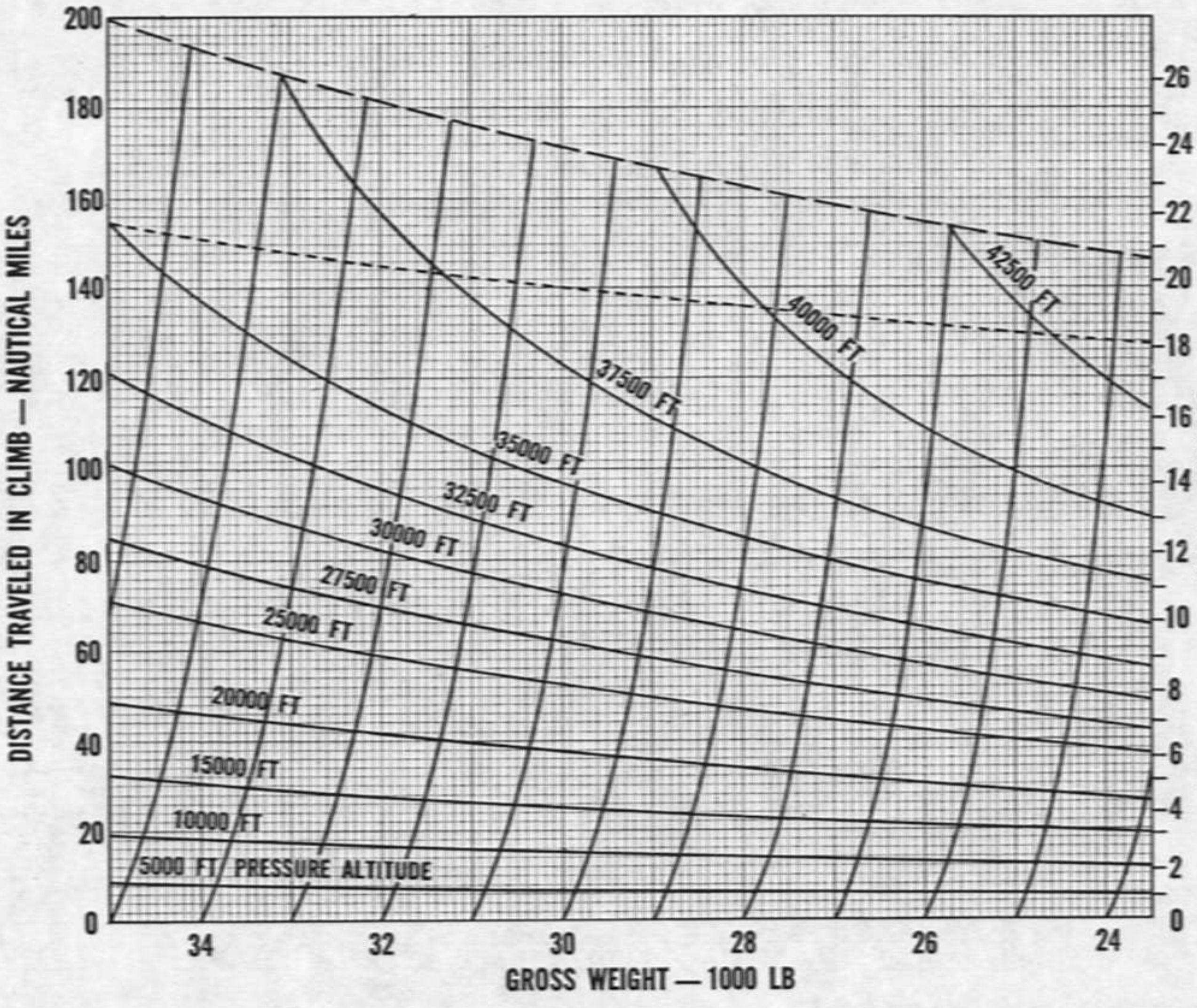


Figure A—9

AFTERBURNER POWER *Climb*

Model: YF-105A

Engine: J57-P-25

Configuration: CLEAN + TWO 450 GAL TYPE I TANKS

CLIMB SPEEDS		
ALT. FT.	CAS—KNOTS	MACH NO.
SEA LEVEL	515	.78
10,000	475	.85
20,000	425	.90
30,000	350	.92
35,000	315	.91
40,000	275	.89
45,000	240	.89
50,000	215	.89

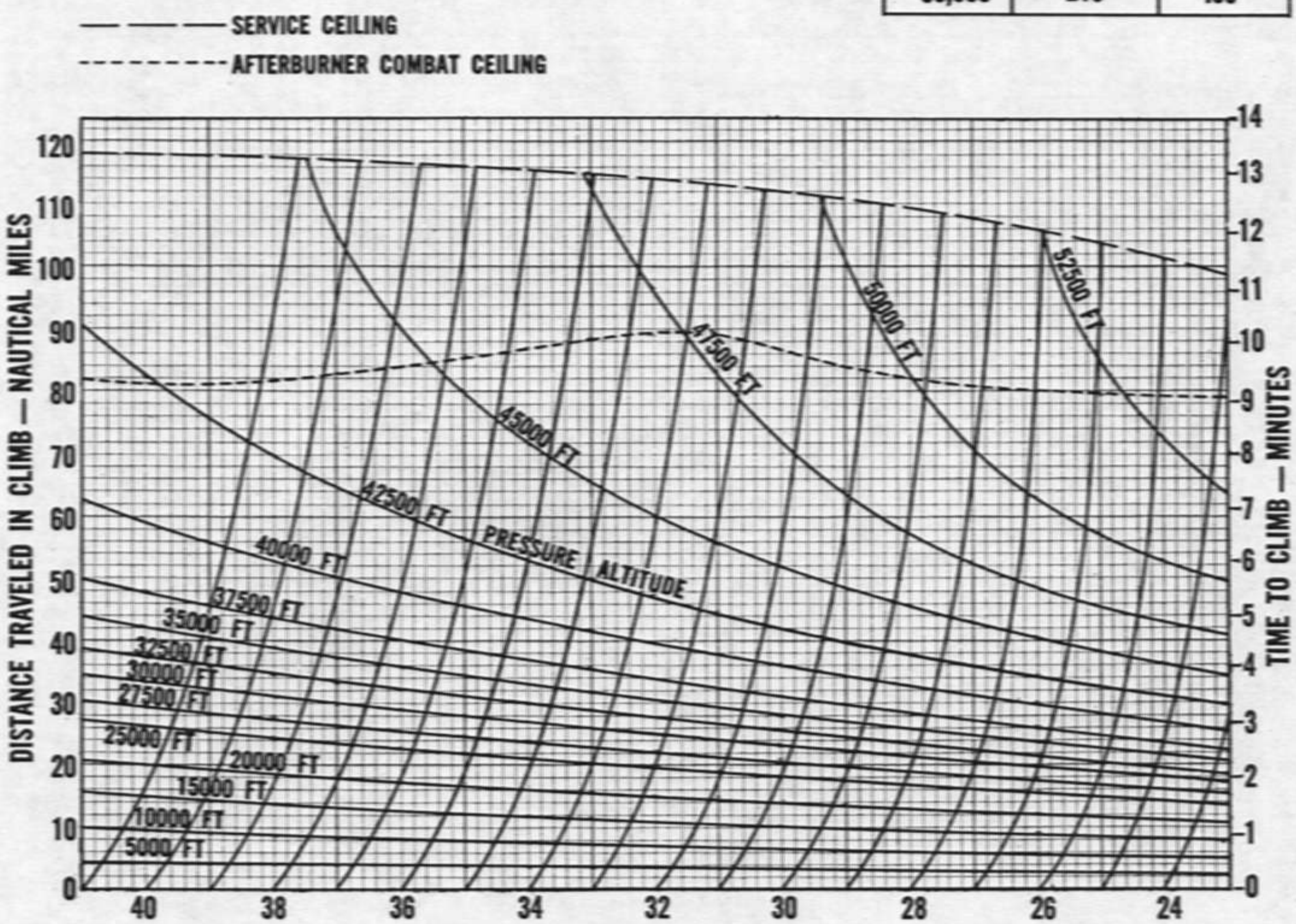


Figure A–10

MILITARY POWER *Climb*

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN + TWO 450 GAL TYPE I TANKS

CLIMB SPEEDS		
ALT. FT.	CAS—KNOTS	MACH NO.
SEA LEVEL	325	.49
10,000	310	.56
20,000	300	.66
30,000	290	.76
35,000	265	.78
40,000	235	.78

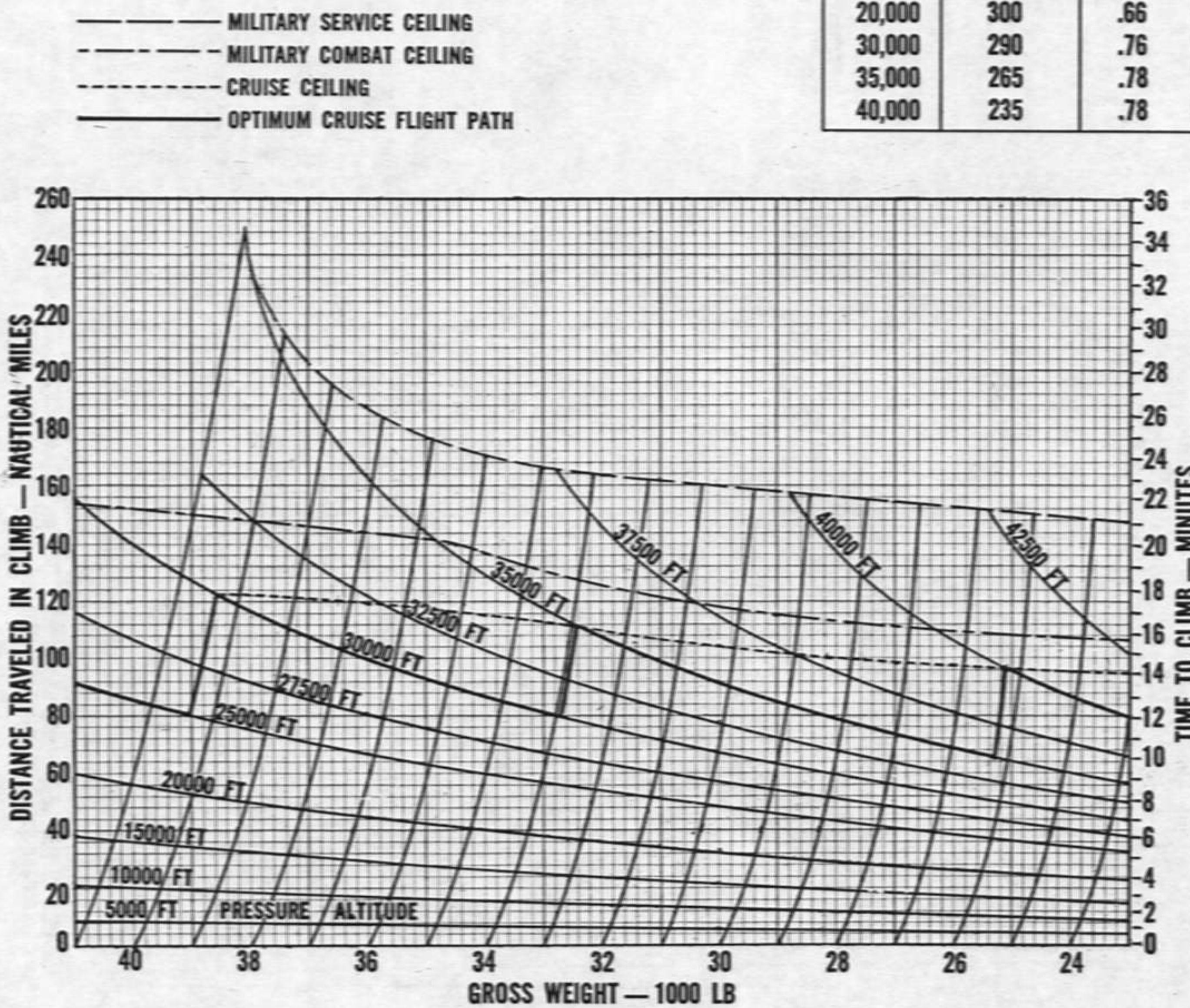


Figure A—11

NORMAL POWER Climb

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN + TWO 450 GAL TYPE I TANKS

CLIMB SPEEDS		
ALT. FT.	CAS—KNOTS	MACH NO.
SEA LEVEL	305	.46
10,000	285	.52
20,000	275	.60
30,000	270	.72
35,000	255	.76
40,000	230	.76

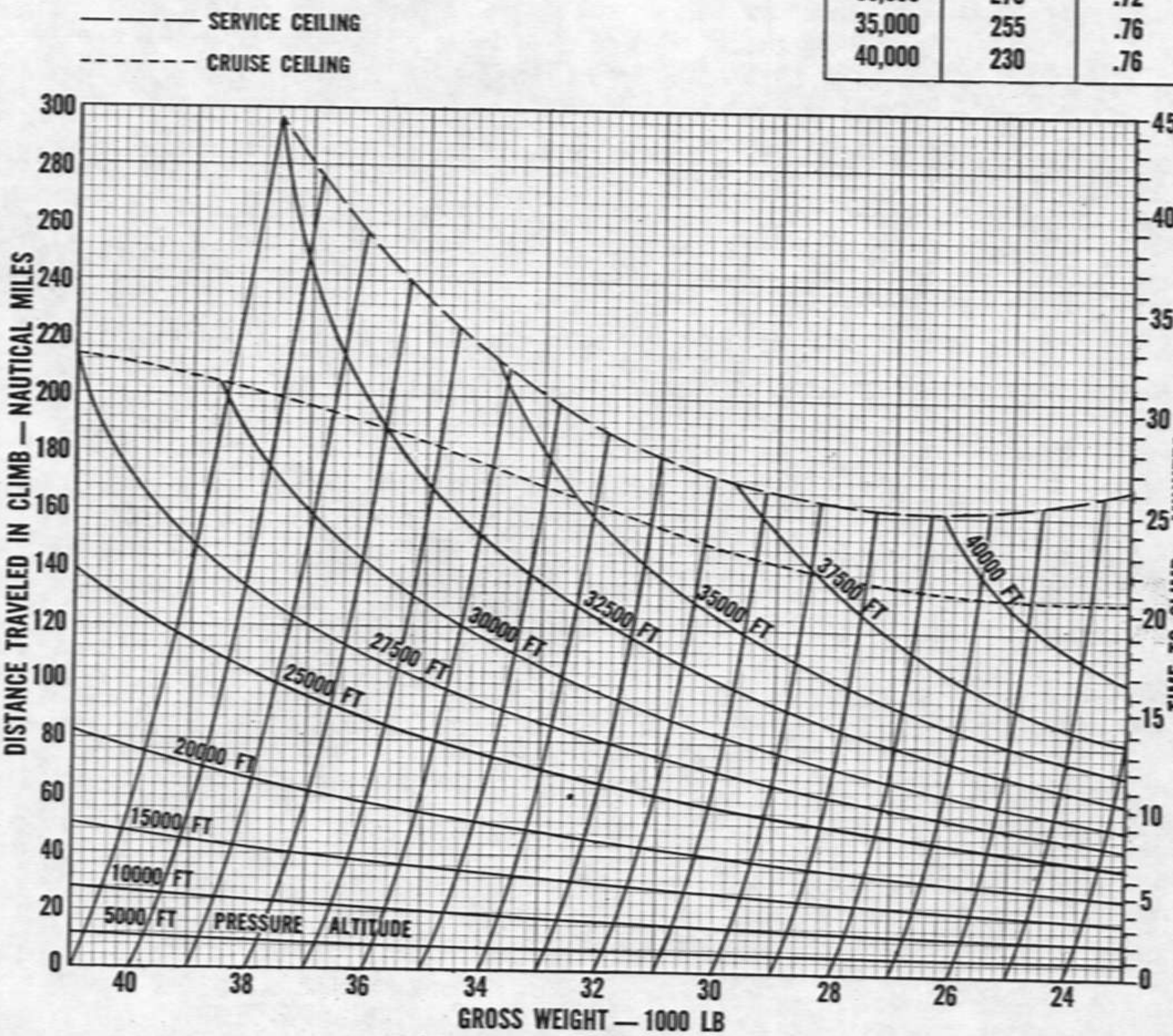


Figure A—12

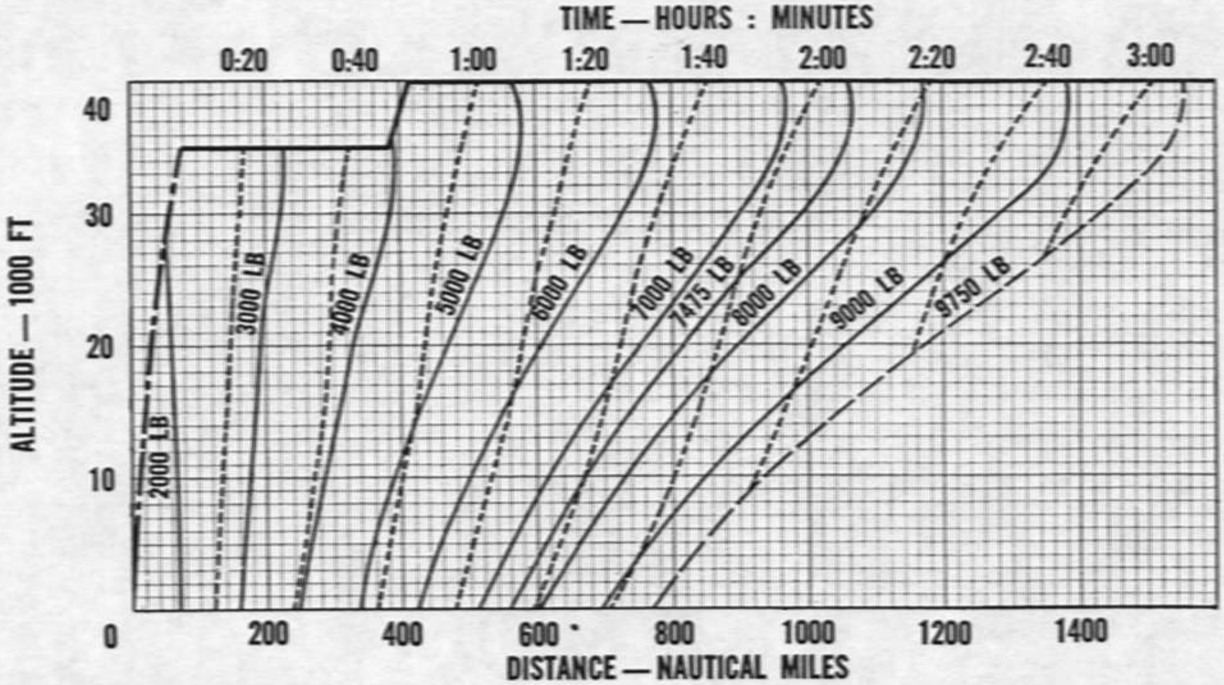
OUTBOUND Profile

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN
Gross Weight: 31,912 LB

ALT. FT.	MACH NO.	APPROXIMATE		
		CAS	LB/HR	TAS
SEA LEVEL	.54	355	3975	355
15,000	.67	340	3295	420
25,000	.75	310	2770	450
35,000	.84	285	2510	485
40,000	.87	265	2450	500

LEGEND

- INITIAL CLIMB PATH
- OPTIMUM CRUISE FLIGHT PATH
- TIME (START, TAXI & TAKE-OFF NOT INCLUDED)
- FUEL CONSUMED
- 9750 LB FUEL CONSUMED — ZERO FUEL REMAINING



Outbound Profile Notes —

1. Fuel allowance for start, taxi and take-off — 1,173 lb — included.
2. No allowance for reserves made for loiter, descent or landing.
3. Use Military Power for climb (See Military Power Climb chart for detail information).
4. Cruise at recommended Mach No.

Figure A—13

OPTIMUM RETURN Profile

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN
Gross Weight: 31,912 LB — 22,162 LB

ALT. FT.	MACH NO.	APPROXIMATE		
		CAS	LB/HR	TAS
SEA LEVEL	.54	355	3975	355
15,000	.67	340	3295	420
25,000	.75	310	2770	450
35,000	.84	285	2510	485
40,000	.87	265	2450	500

NOTES

1. Fuel required at any point includes Military Power climb to optimum cruise 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 lines and optimum cruise altitude line or line of best range.
4. Cruise at recommended Mach No.

LEGEND

- - - - - FUEL REQUIRED
— — — LINE OF BEST RANGE FOR CONSTANT ALTITUDE
—— OPTIMUM RANGE — FLIGHT PROFILE
—— CLIMB PATH — GUIDE LINES
• • • TIME AT CRUISING ALTITUDE

IN THIS AREA CRUISE AT INITIAL ALTITUDE
CLIMB TO OPTIMUM ALTITUDE AND CRUISE AT THAT ALTITUDE
IN THIS AREA CLIMB TO OPTIMUM STEP CLIMB ALTITUDE AND USE STEP CLIMB PROCEDURE

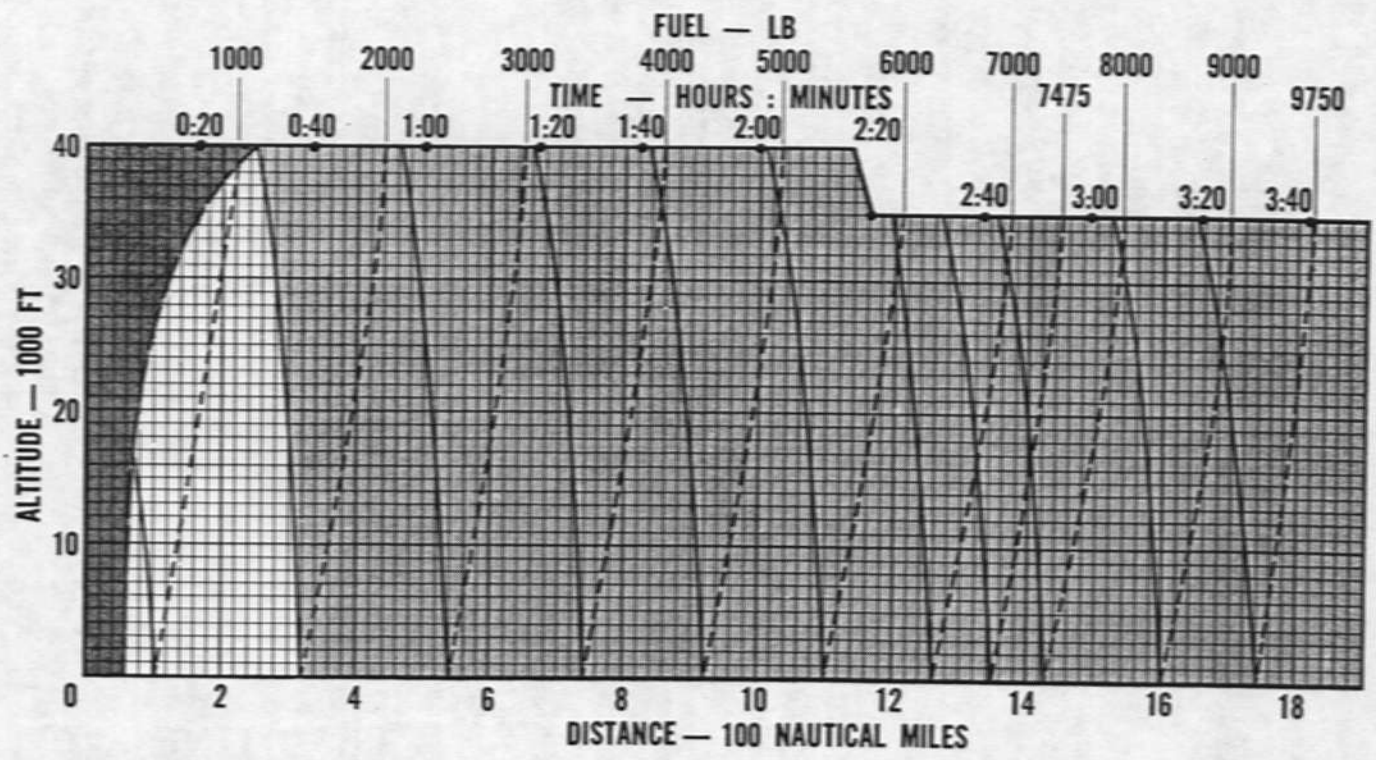
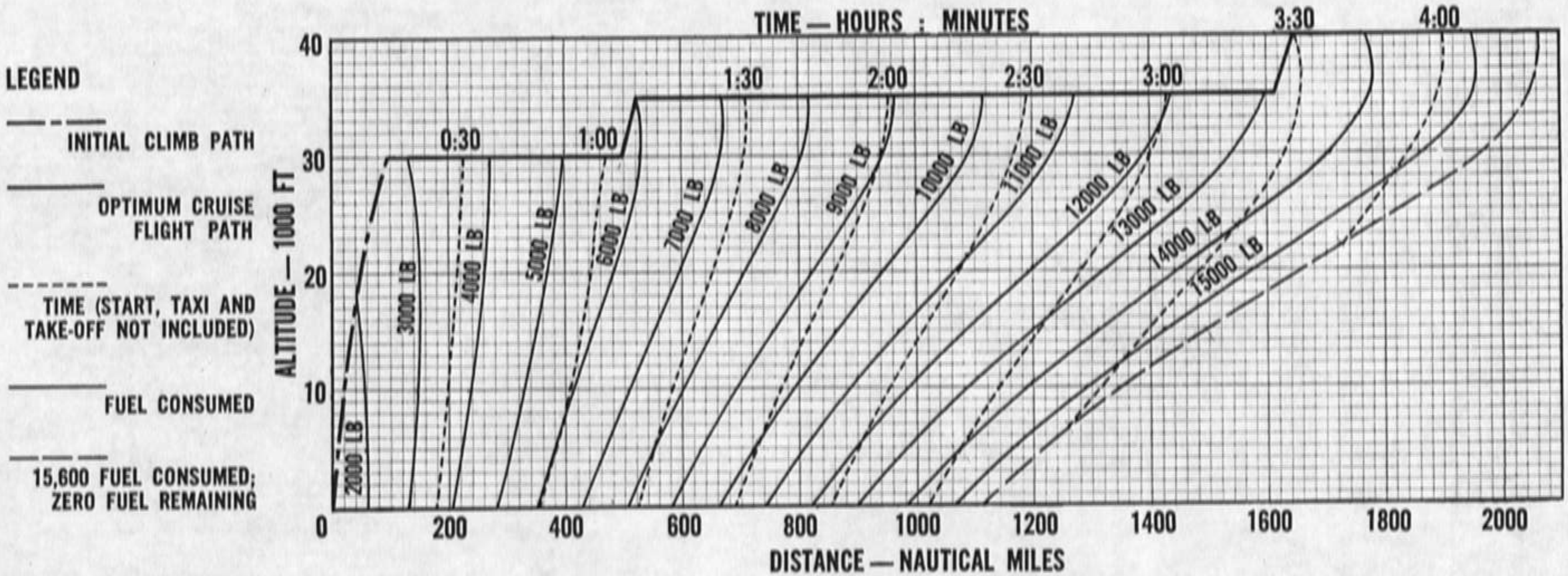


Figure A–14

OUTBOUND *Profile*

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN + TWO 450 GAL TYPE I TANKS
Take-Off G.W.: 38,453 LB

ALT. FT.	MACH NO.	APPROXIMATE		
		CAS	LB/HR	TAS
SEA LEVEL	.51	335	4365	335
15,000	.64	325	3770	440
25,000	.74	310	3365	445
30,000	.79	300	3220	465
35,000	.82	280	2965	475
40,000	.86	265	2680	490



Outbound Profile Notes —

1. Fuel allowance for start, taxi and take-off — 1,173 lb — included.
2. No allowance or reserves made for loiter, descent or landing.
3. Use Military Power for climb (See Military Power Climb chart for detail information).
4. Cruise at recommended Mach No.

Figure A–15

OPTIMUM RETURN Profile

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN + TWO 450 GAL TYPE I TANKS
Gross Weight: 38,453 LB — 22,853 LB

ALT. FT.	MACH NO.	APPROXIMATE		
		CAS	LB/HR	TAS
SEA LEVEL	.51	335	4365	335
15,000	.64	325	3770	440
25,000	.74	310	3365	445
30,000	.79	300	3220	465
35,000	.82	280	2965	475
40,000	.86	265	2680	490

NOTES
1. Fuel required at any point includes Military Power climb to optimum cruise 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 lines and optimum cruise altitude line or line of best range.
4. Cruise at recommended Mach No.

LEGEND

IN THIS AREA, CRUISE AT INITIAL ALTITUDE.

CLIMB TO OPTIMUM ALTITUDE AND CRUISE AT THAT ALTITUDE.

IN THIS AREA, CLIMB TO OPTIMUM STEP CLIMB ALTITUDE AND USE STEP CLIMB PROCEDURE

- - - - - - - FUEL REQUIRED
— — — LINE OF BEST RANGE FOR CONSTANT ALTITUDE
———— OPTIMUM RANGE — FLIGHT PROFILE
———— CLIMB PATH — GUIDE LINES
• • • TIME AT CRUISING ALTITUDES

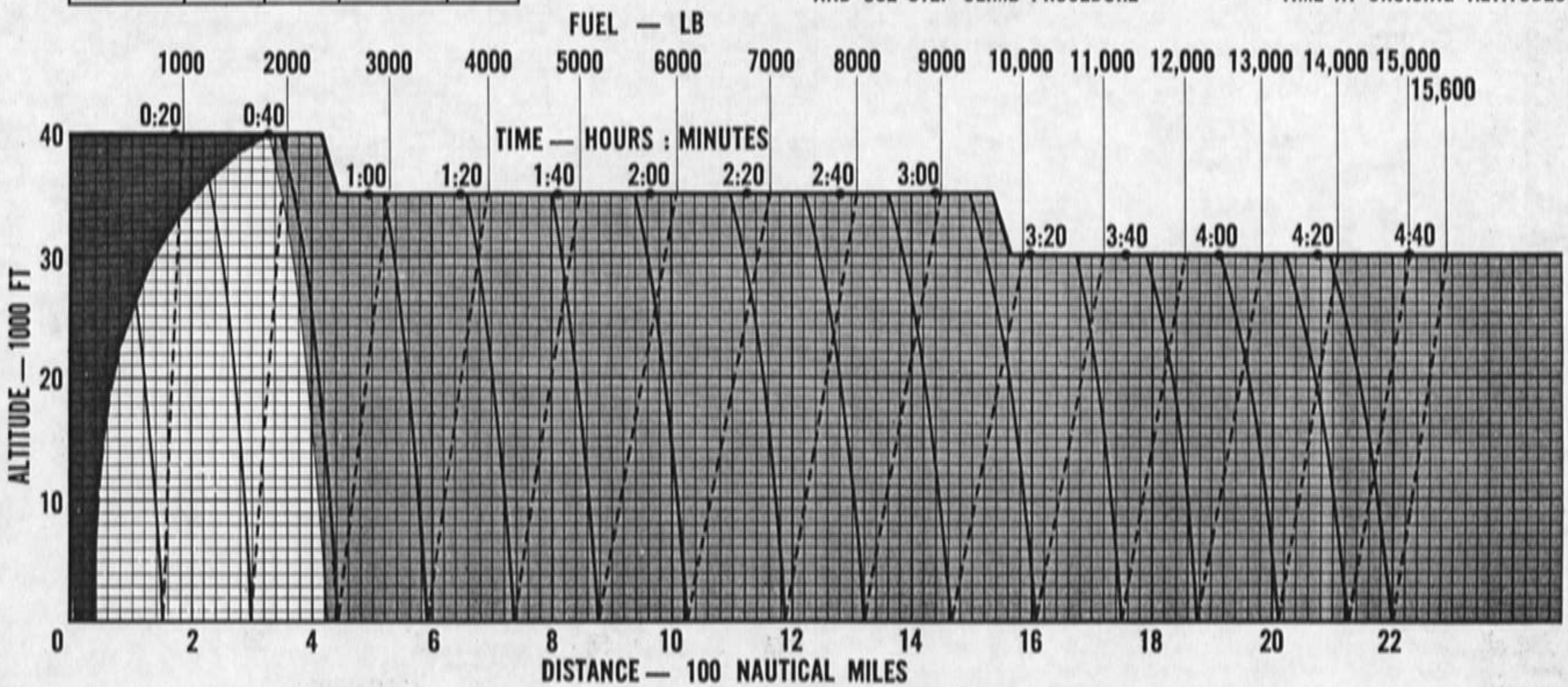


Figure A—16

MAXIMUM ENDURANCE Chart

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN

ALT. FT.	CAS	TAS	MACH NO.	LB/HR
SEA LEVEL	200	200	.30	2525
15,000	220	275	.44	2395
25,000	220	325	.54	2130
35,000	220	385	.67	2065
40,000	215	415	.72	2150

NOTES:
1. Loiter at recommended CAS.
2. Maintain constant altitude.

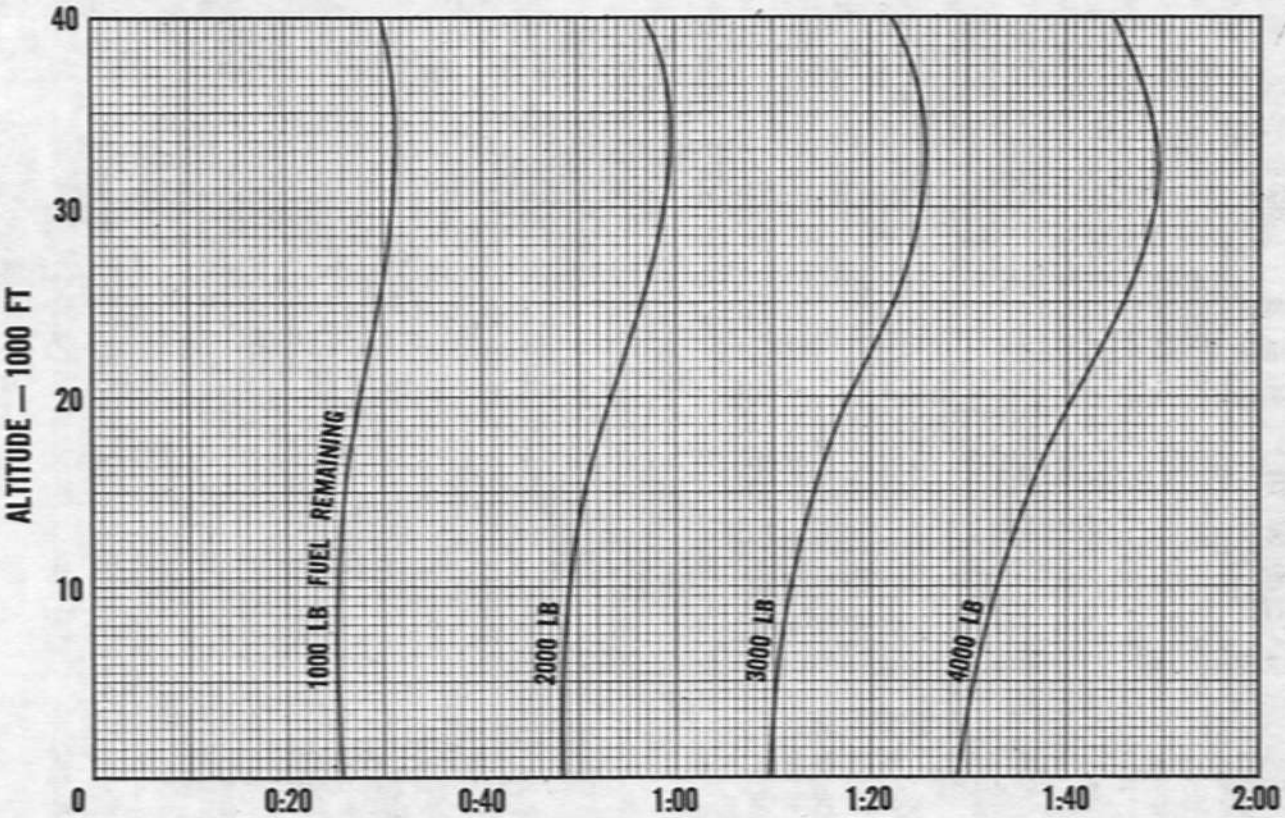


Figure A–17

MAXIMUM ENDURANCE *Profile*

Model: YF-105A

Engine: J57-P-25

Configuration: CLEAN + TWO 450 GAL TYPE I TANKS

ALT. FT.	CAS	TAS	MACH NO.	LB/HR
SEA LEVEL	210	210	.32	2910
15,000	210	260	.42	2635
25,000	210	310	.51	2420
35,000	225	390	.67	2420
40,000	215	410	.71	2395

NOTES:

1. Loiter at recommended CAS.
2. Maintain constant altitude.

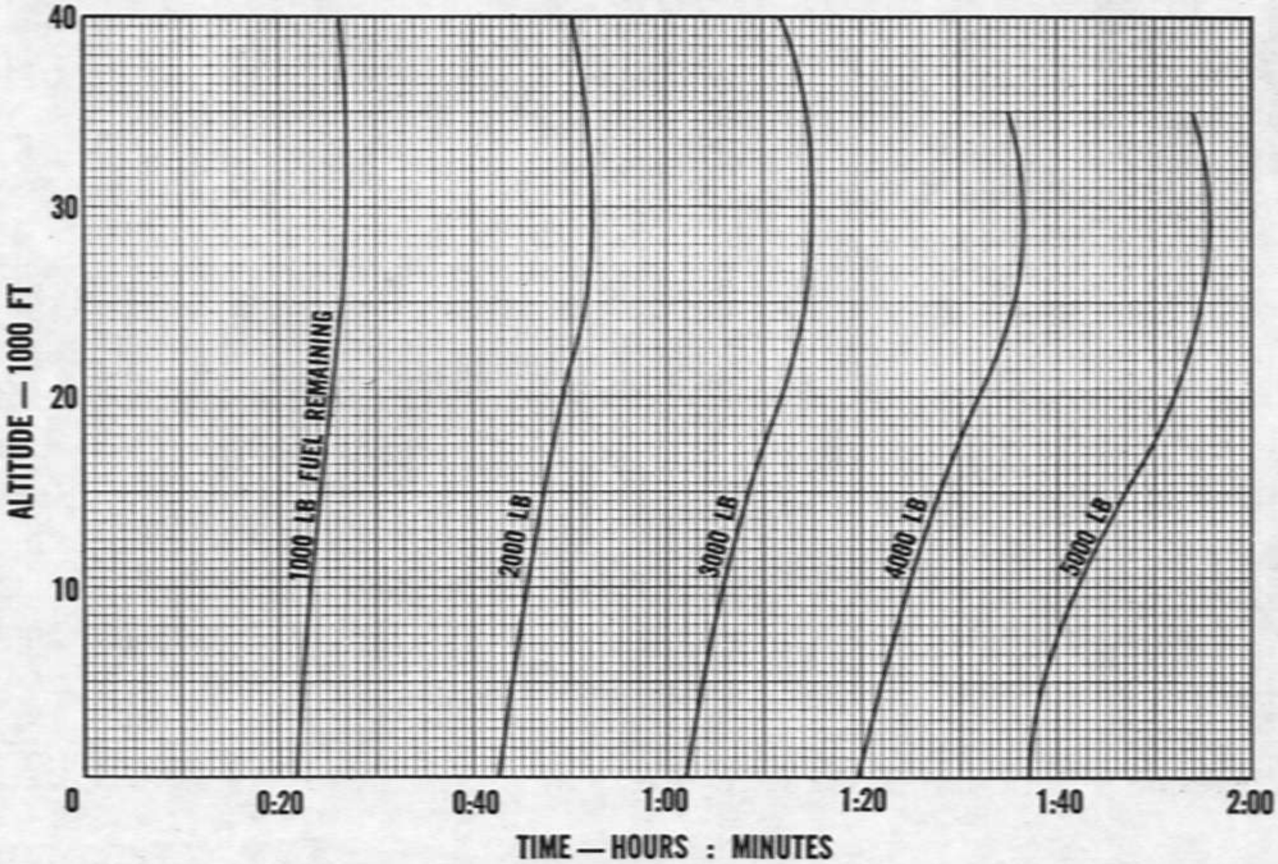


Figure A–18

COMBAT ALLOWANCE CHART

Model: YF-105A **Engine: J57-P-25**

Configuration: CLEAN

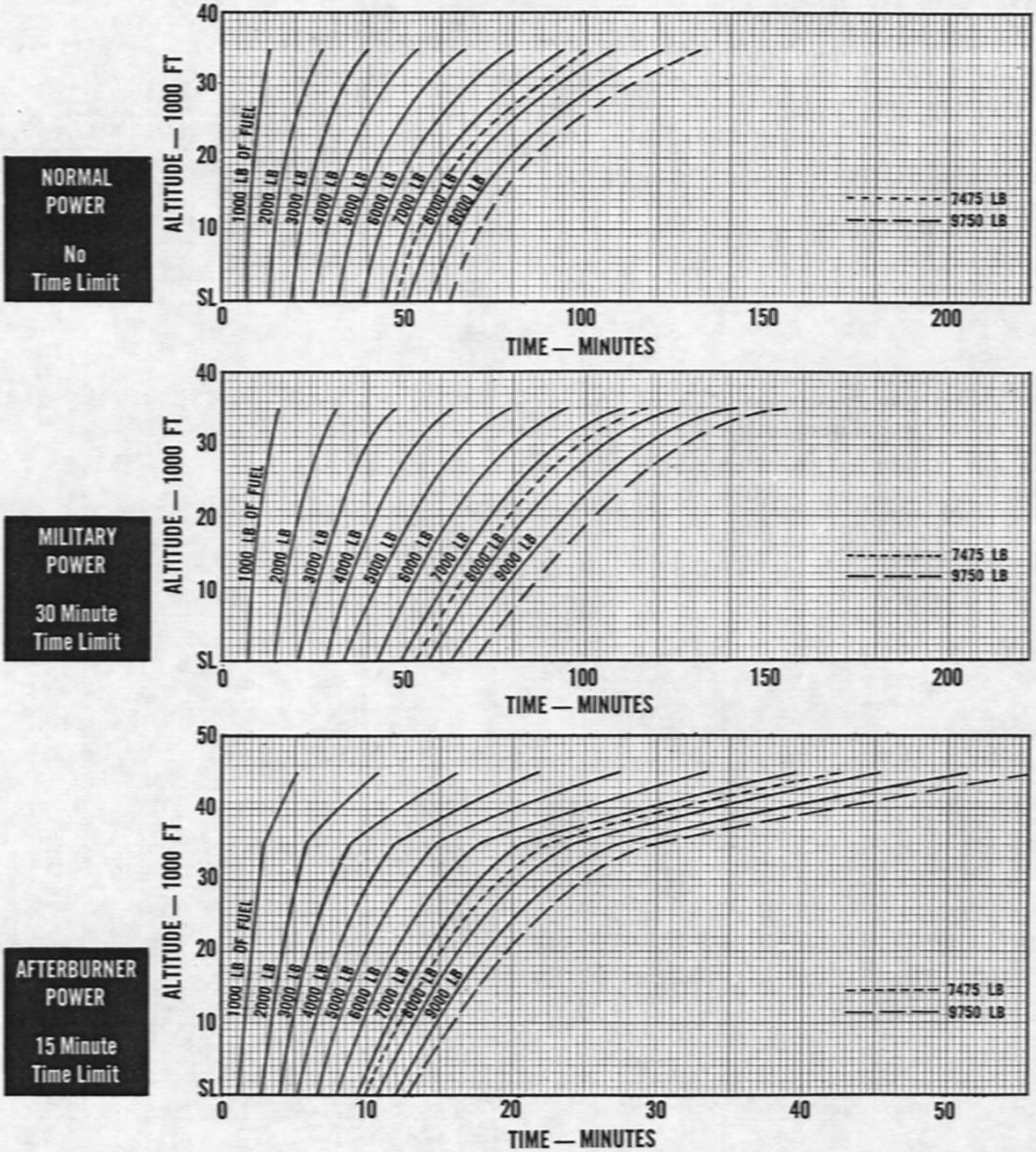


Figure A–19

COMBAT ALLOWANCE CHART

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN + TWO 450 GAL. TYPE I TANKS

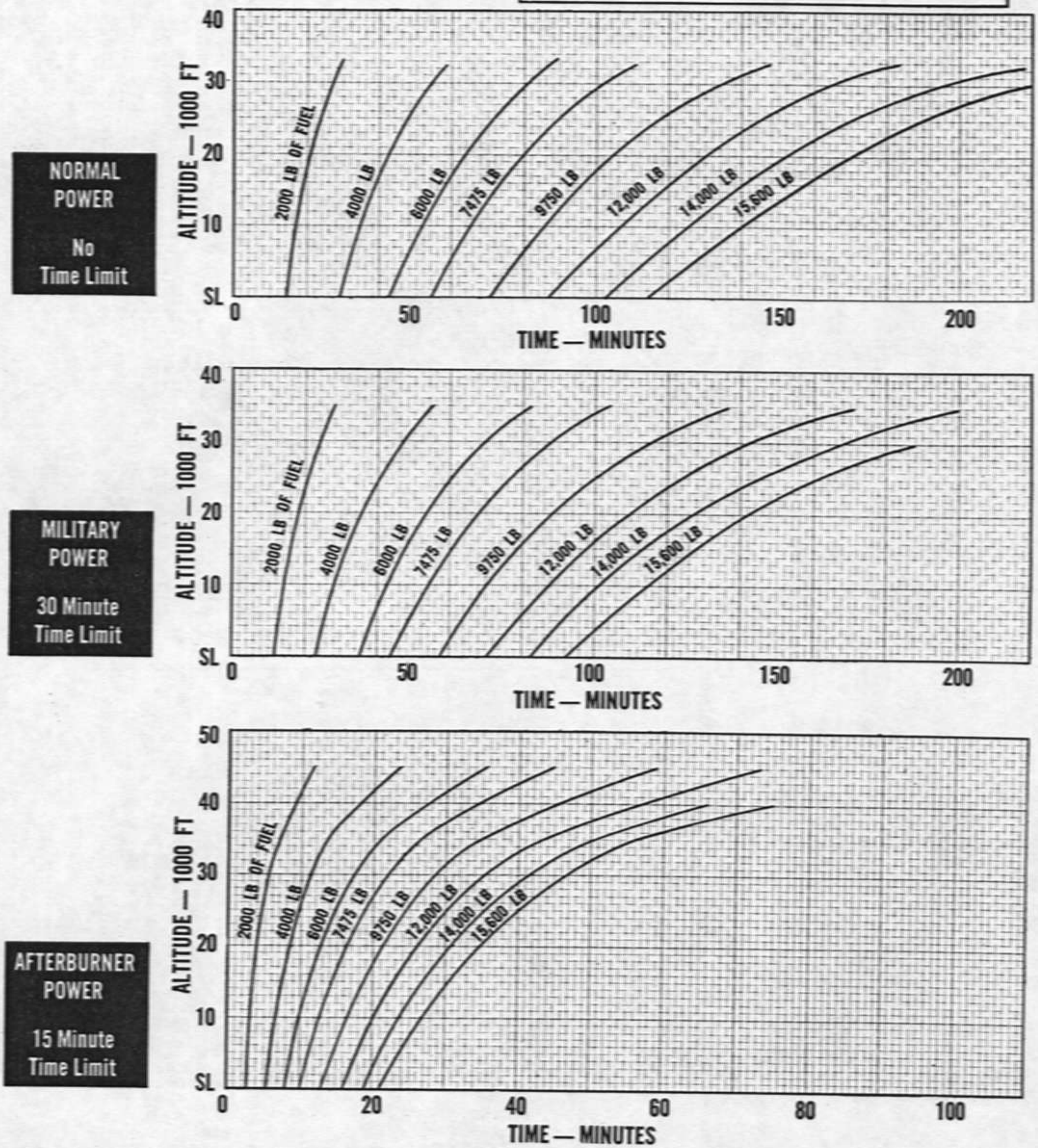
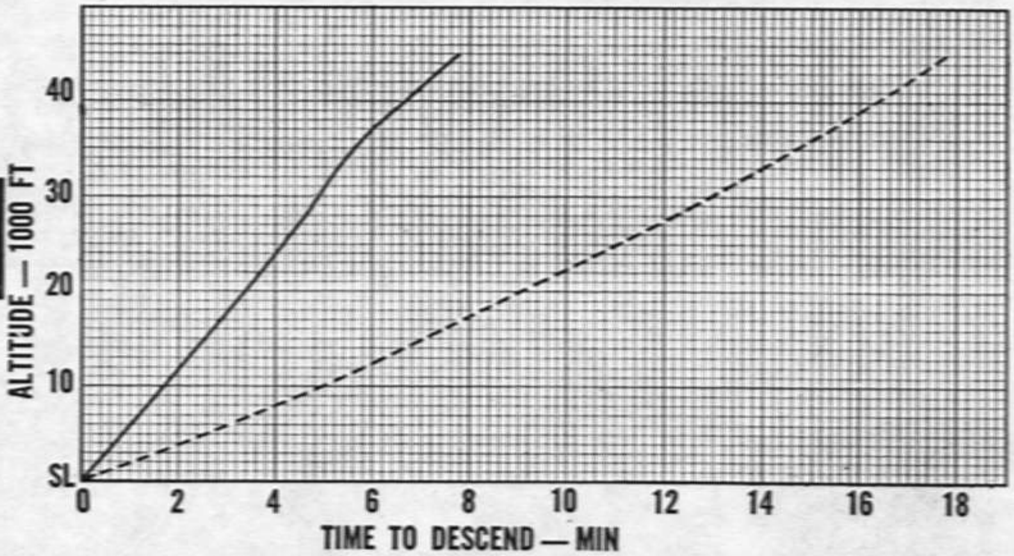
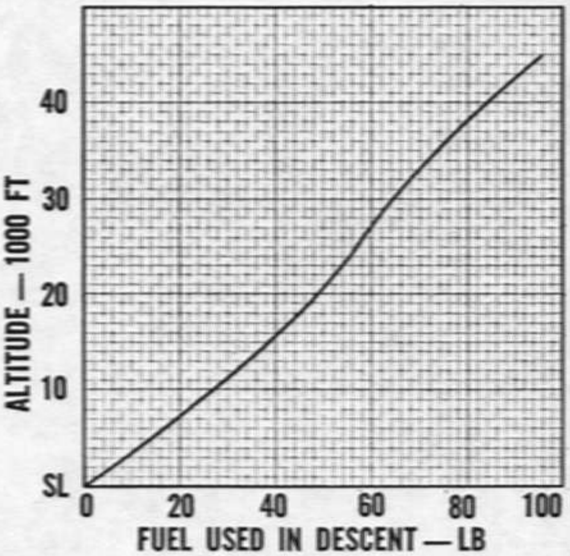


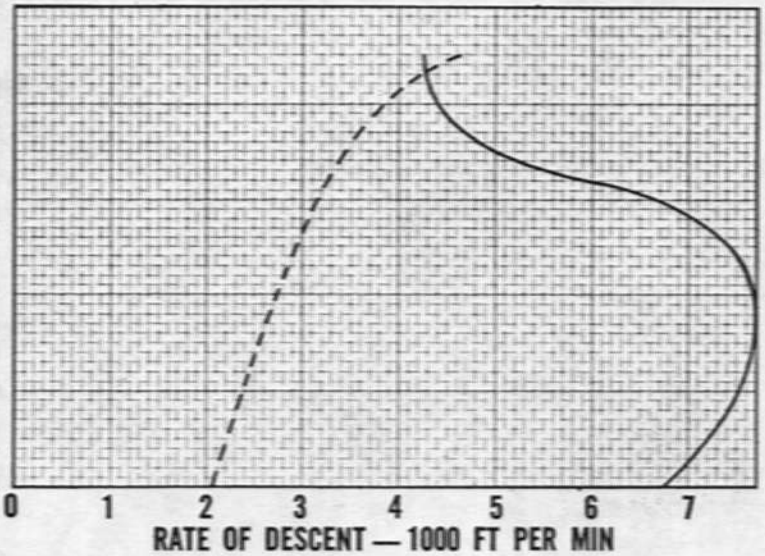
Figure A–20

DESCENTS

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN
Gross Weight: 23,625 LB







———— RECOMMENDED DESCENT

Use Idle Power, Speed Brakes closed—Descend at .90 Mach Number or 400 Knots CAS, whichever is less.

- - - - - - POWER OFF MAXIMUM RANGE

Use Power-off, Speed Brakes closed — Descend at 220 Knots CAS.

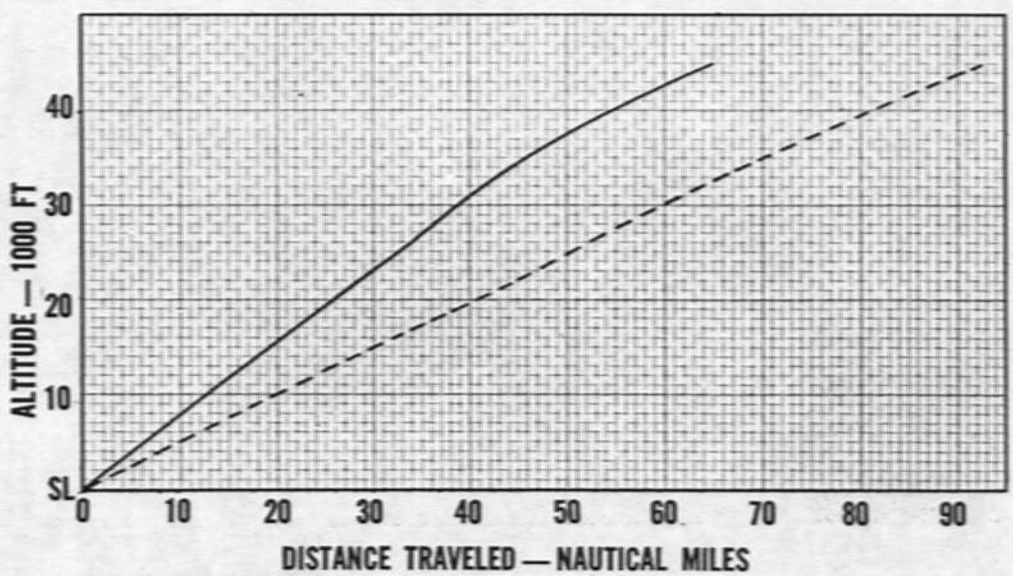
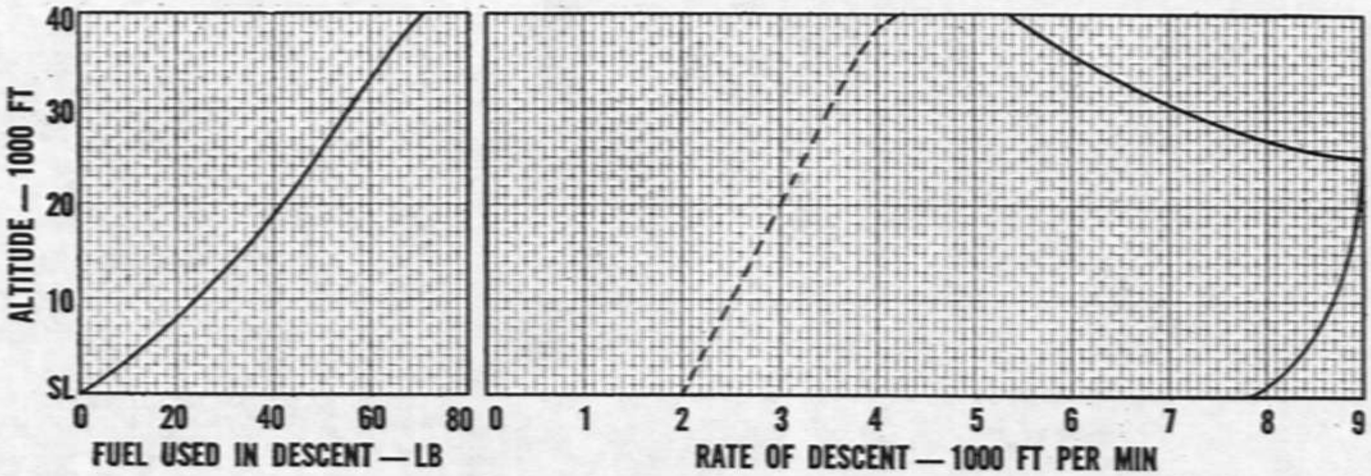
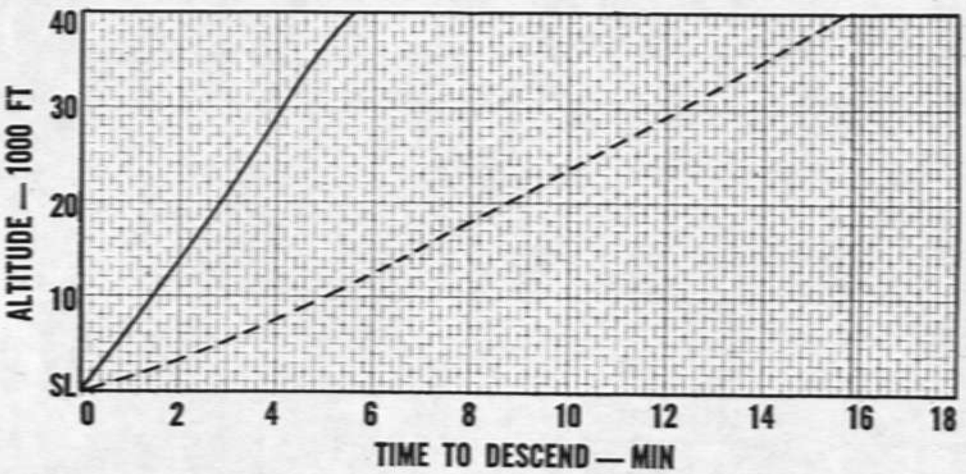


Figure A–21

DESCENTS

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN + TWO 450 GAL. TYPE I TANKS
Gross Weight: 25,193 LB





——————— RECOMMENDED DESCENT

Use Idle Power, Speed Brakes closed — Descend at .90 Mach Number or 400 Knots CAS, whichever is less.

– – – – – – – POWER OFF MAXIMUM RANGE

Use Power-off, Speed Brakes Closed — Descend at 220 Knots CAS.

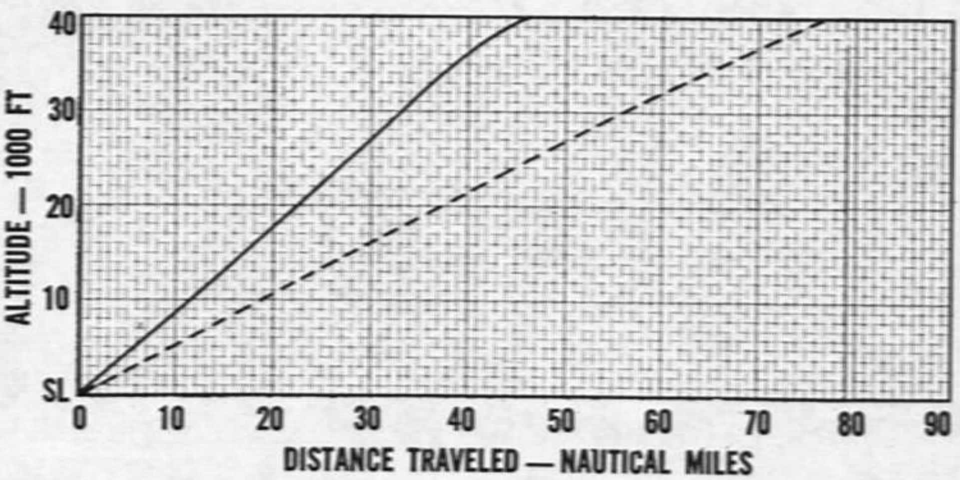
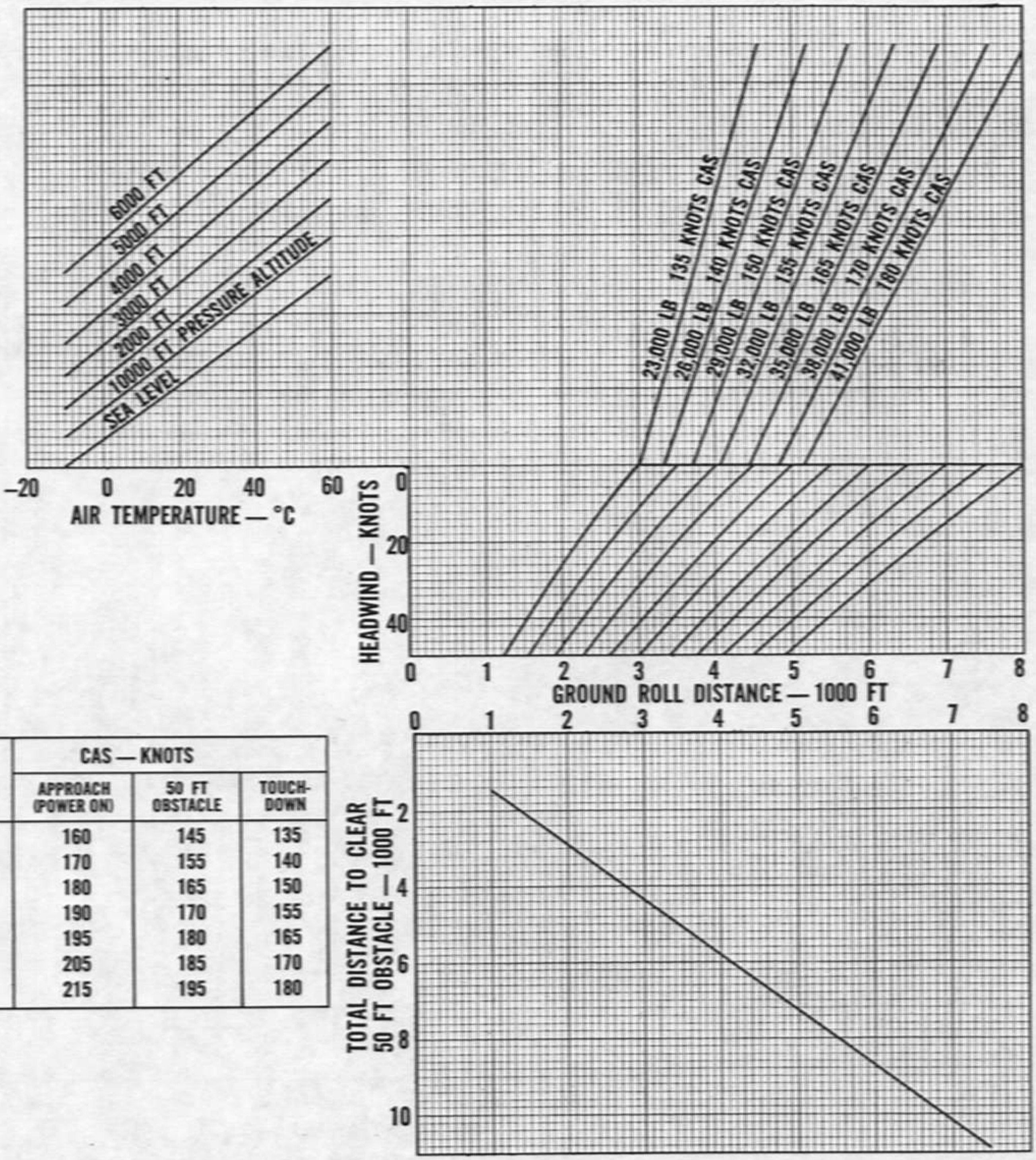


Figure A—22

LANDING *Distances*

Model: YF-105A — **IDLE POWER**
Engine: J57-P-25 — **HARD SURFACE RUNWAY**



G.W. 1000 LB	CAS — KNOTS APPROACH (POWER ON)	50 FT OBSTACLE	TOUCH-DOWN
23	160	145	135
26	170	155	140
29	180	165	150
32	190	170	155
35	195	180	165
38	205	185	170
41	215	195	180

Figure A—23

CRUISE CONDITIONS *Sea Level*

Model: YF-105A

Engine: J57-P-25

Configuration: CLEAN

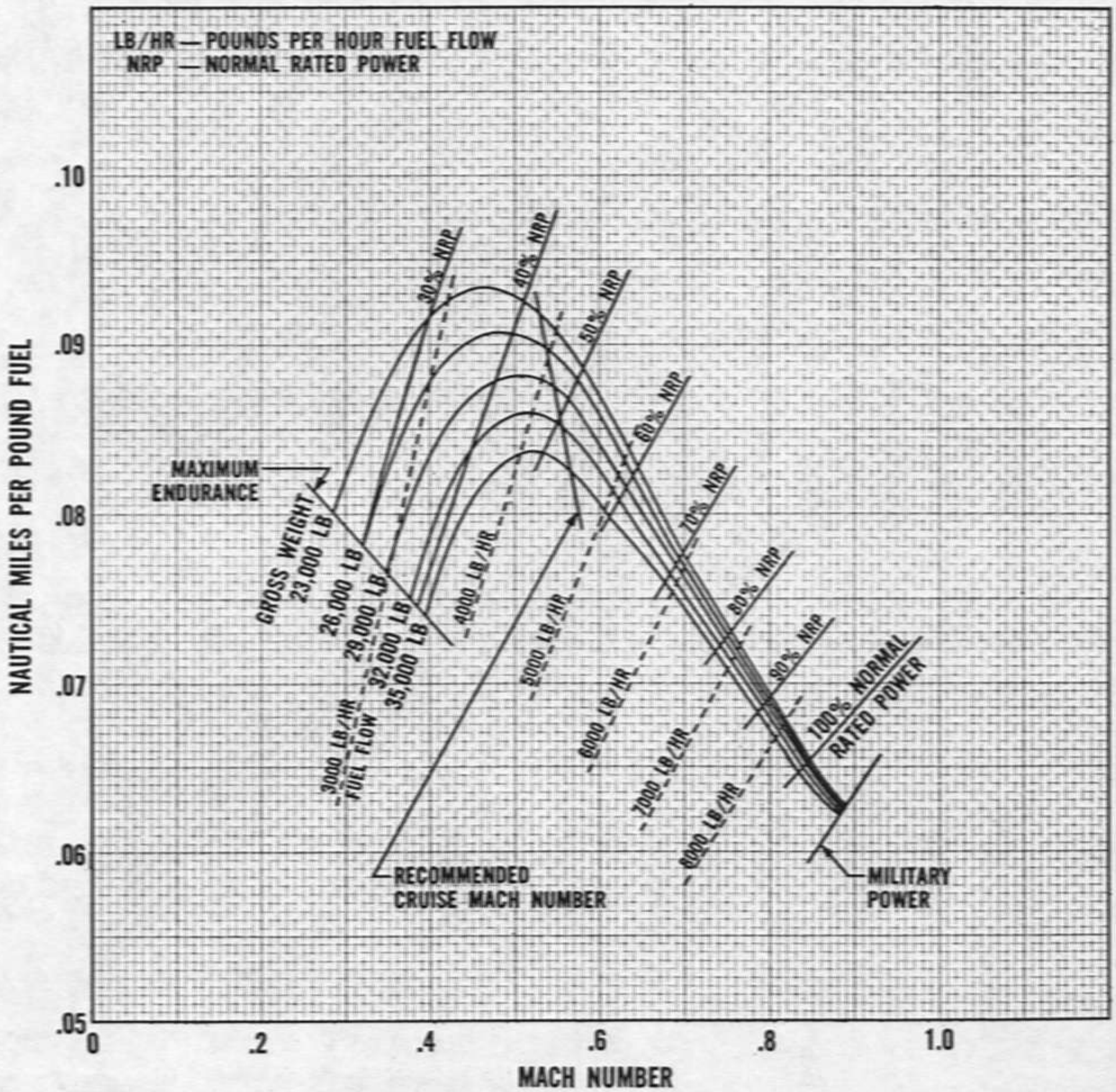


Figure A—24

CRUISE CONDITIONS 15,000 ft

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN

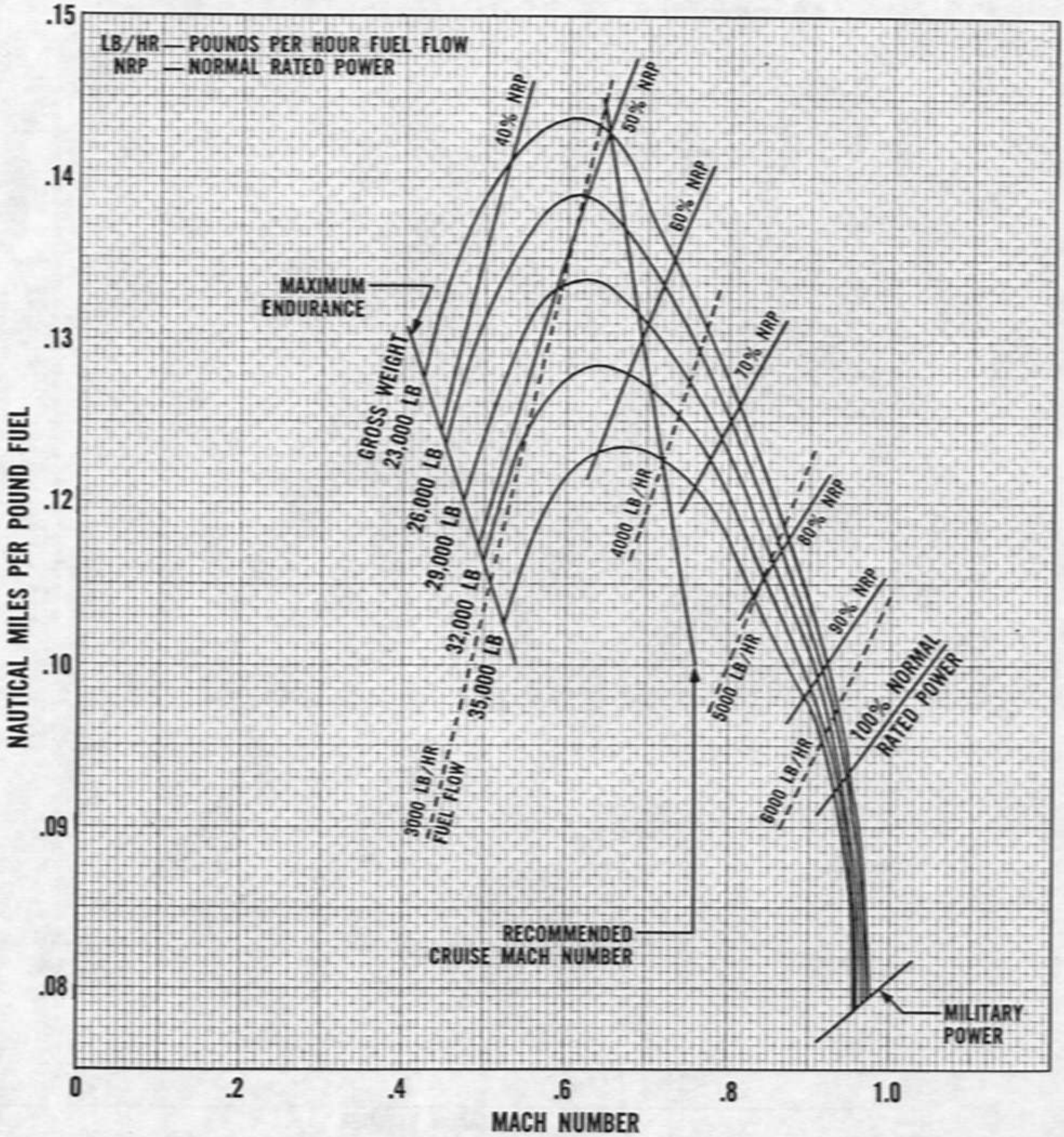


Figure A—25

CRUISE CONDITIONS 25,000 ft

Model: YF-105A

Engine: J57-P-25

Configuration: CLEAN

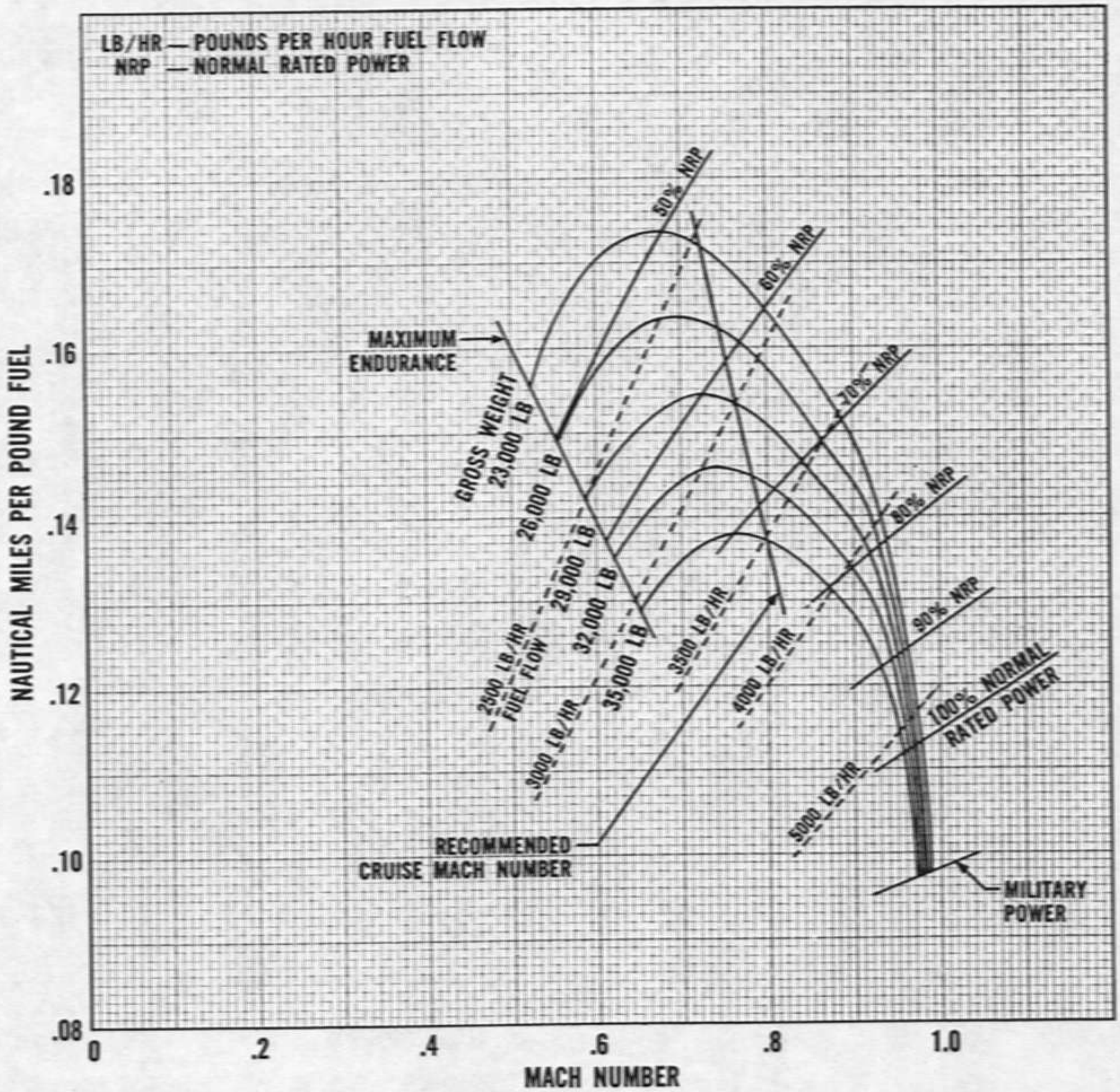


Figure A—26

CRUISE CONDITIONS 35,000 ft

Model:	**YF-105A**
Engine:	**J57-P-25**
Configuration:	**CLEAN**

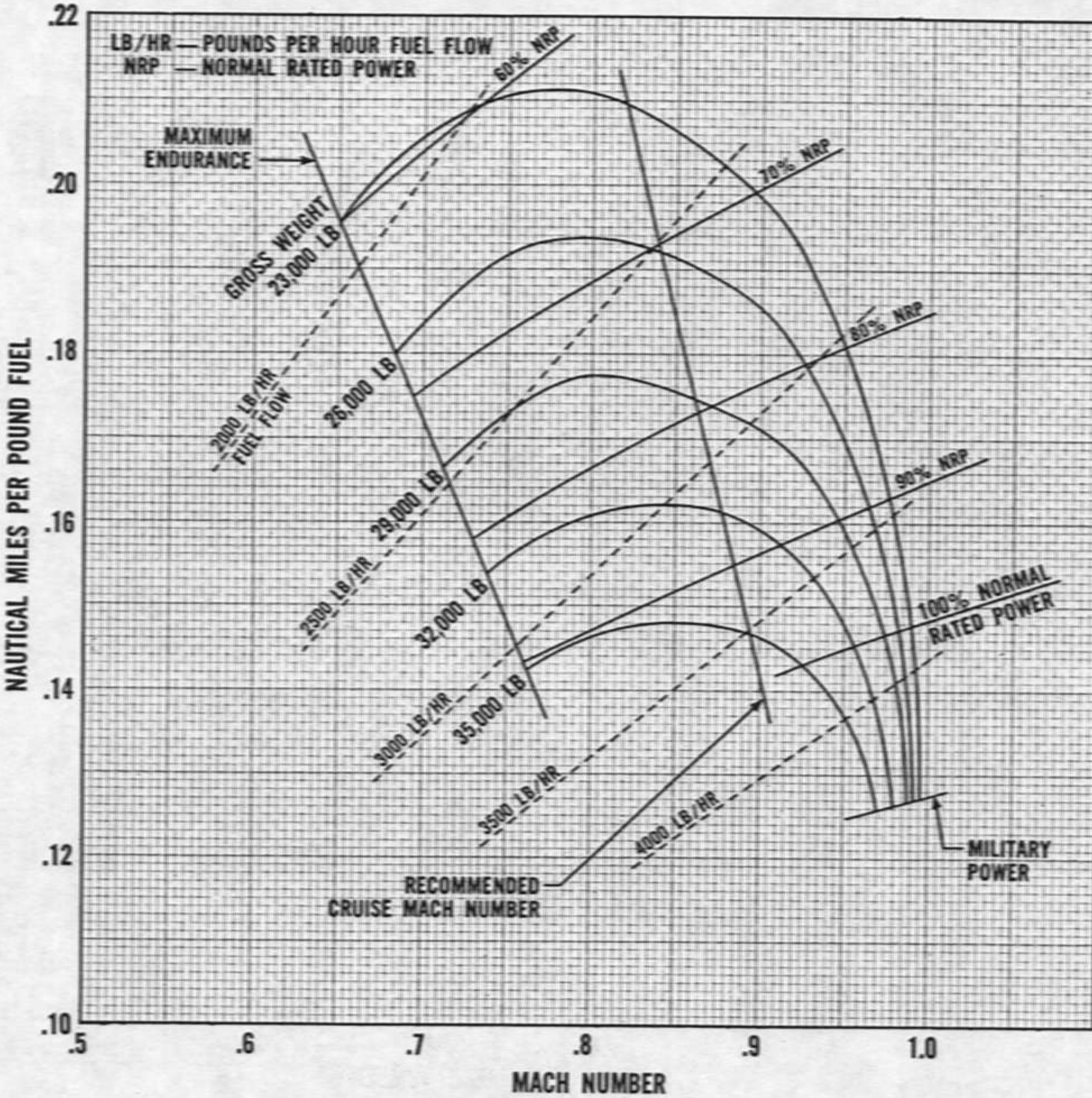


Figure A—27

CRUISE CONDITIONS 40,000 ft

Model: YF-105A

Engine: J57-P-25

Configuration: CLEAN

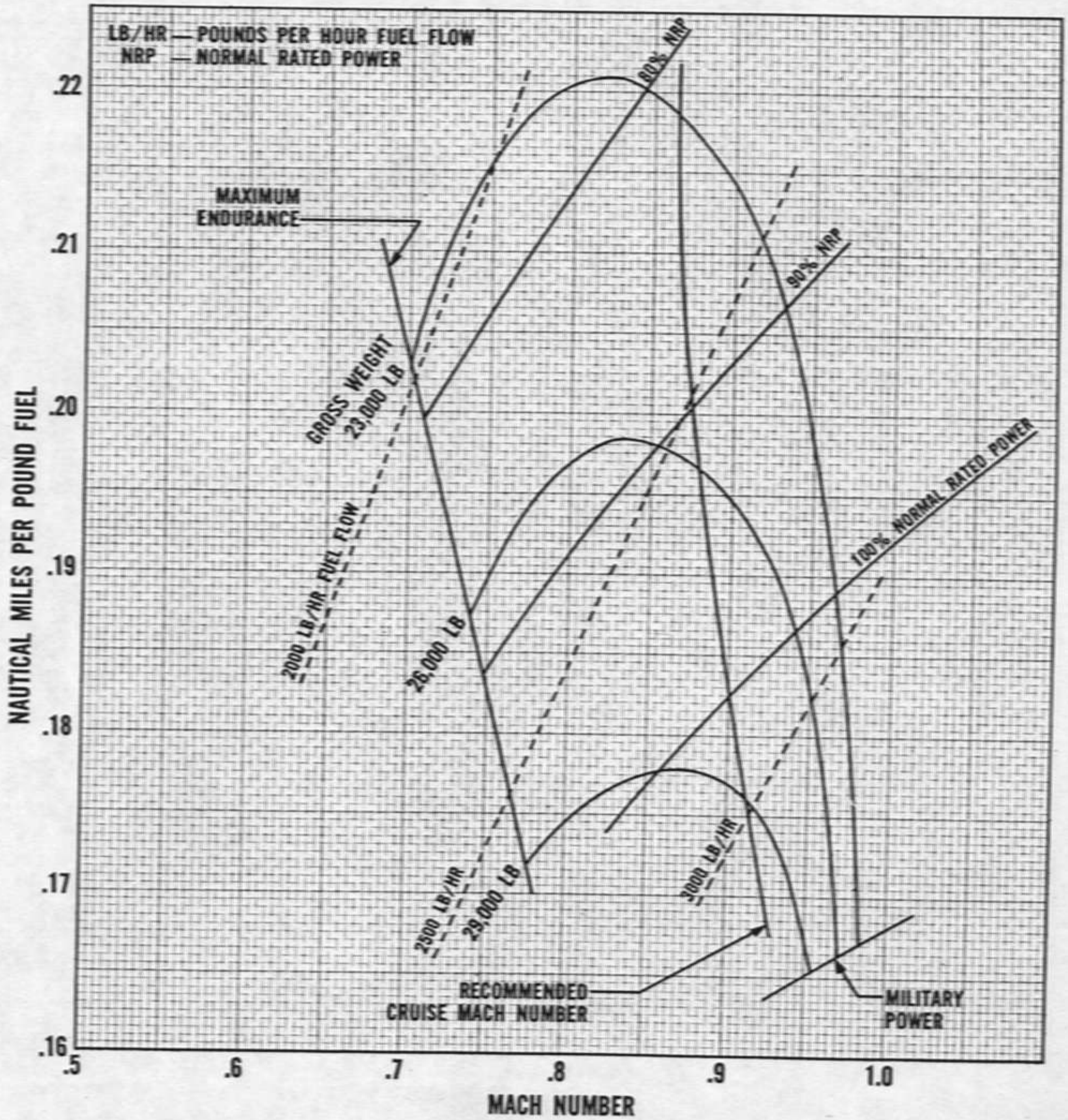


Figure A–28

CRUISE CONDITIONS *Sea Level*

Model: YF-105A

Engine: J57-P-25

Configuration: CLEAN + TWO 450 GAL. TYPE I TANKS

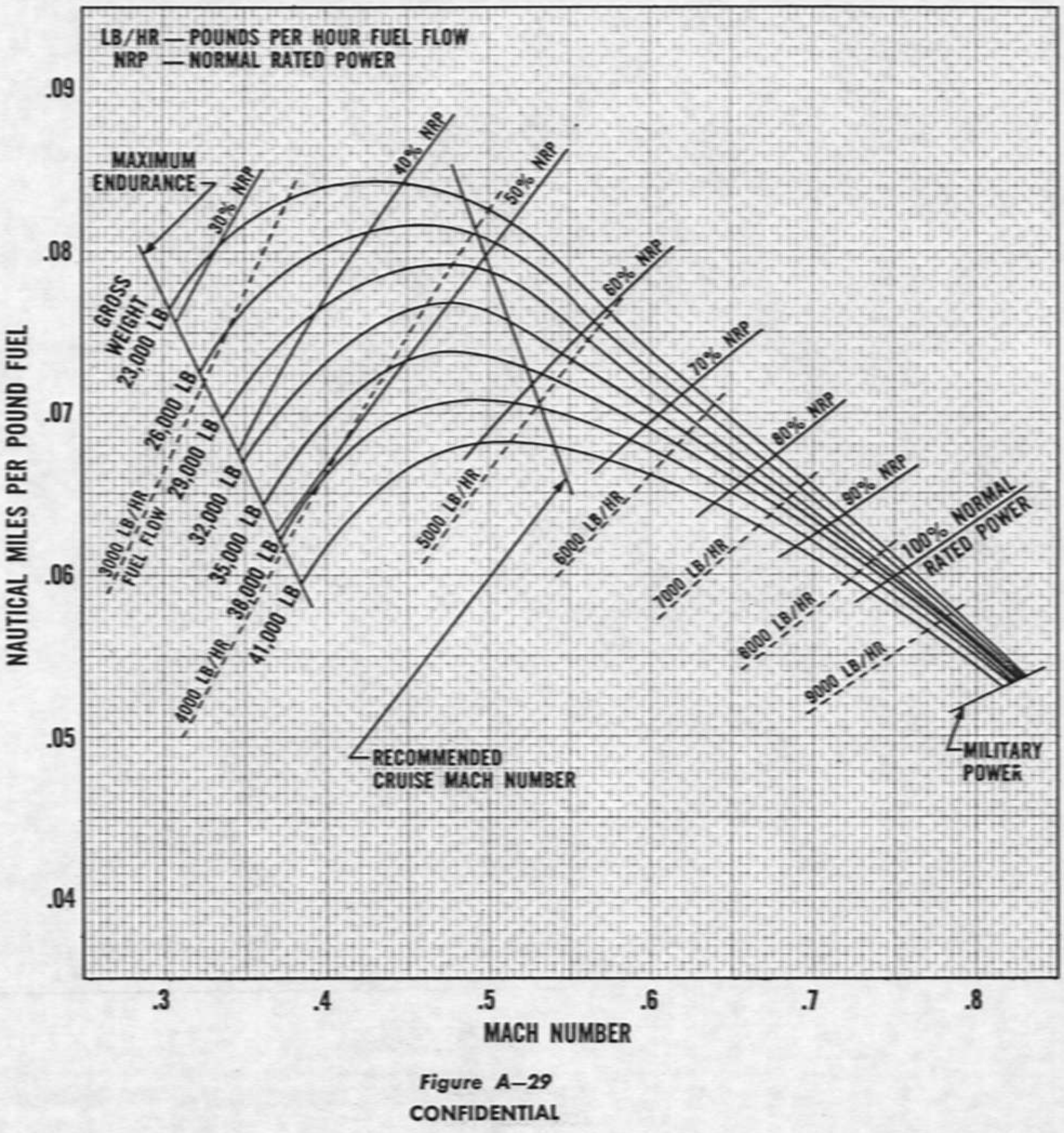


Figure A–29

CRUISE CONDITIONS 15,000 ft

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN + TWO 450 GAL. TYPE I TANKS

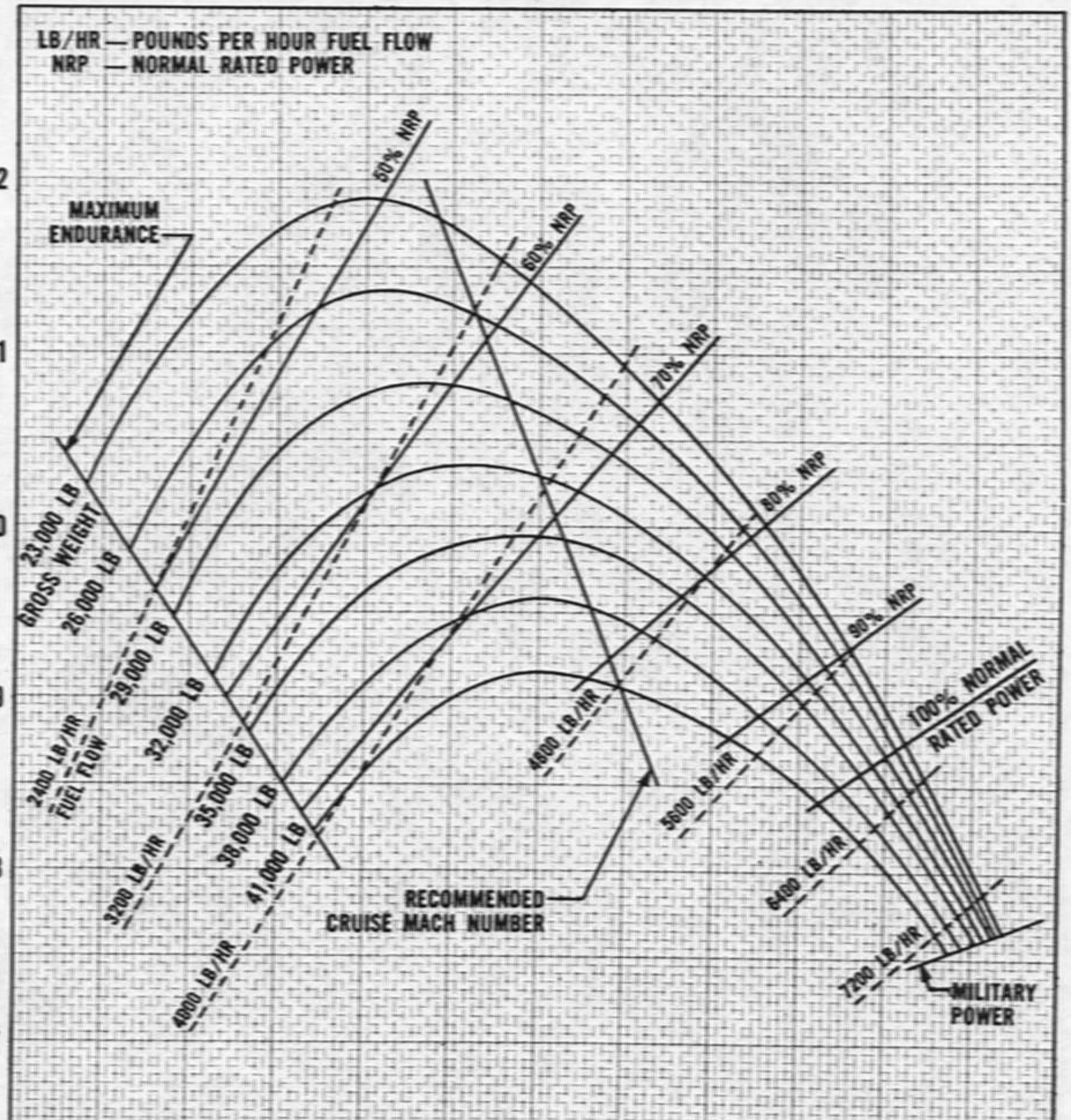


Figure A—30

CRUISE CONDITIONS 25,000 ft

Model: YF-105A
Engine: J57-P-25
Configuration: CLEAN + TWO 450 GAL. TYPE I TANKS

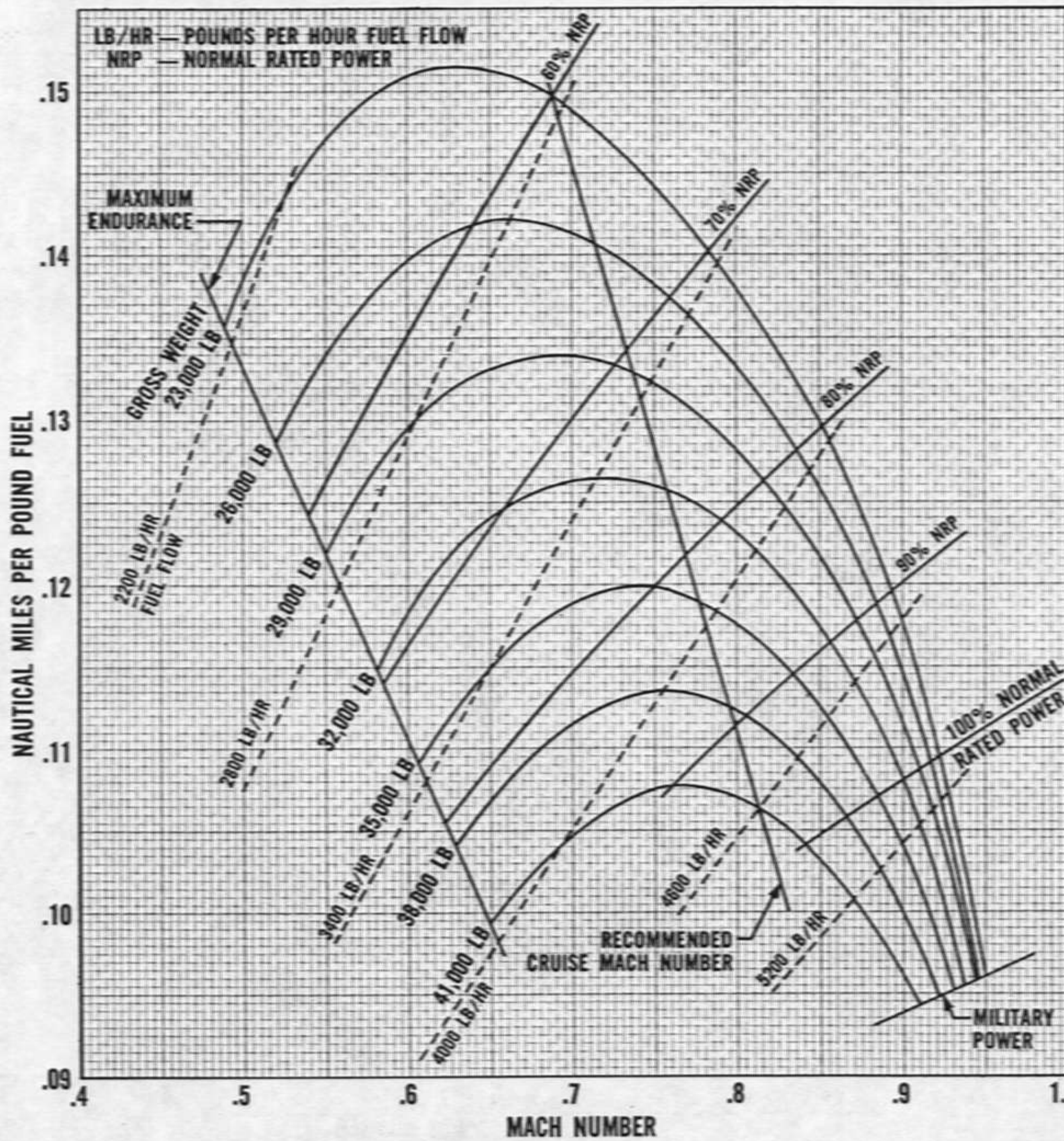


Figure A—31

CRUISE CONDITIONS 35,000 ft

Model: YF-105A

Engine: J57-P-25

Configuration: CLEAN + TWO 450 GAL. TYPE I TANKS

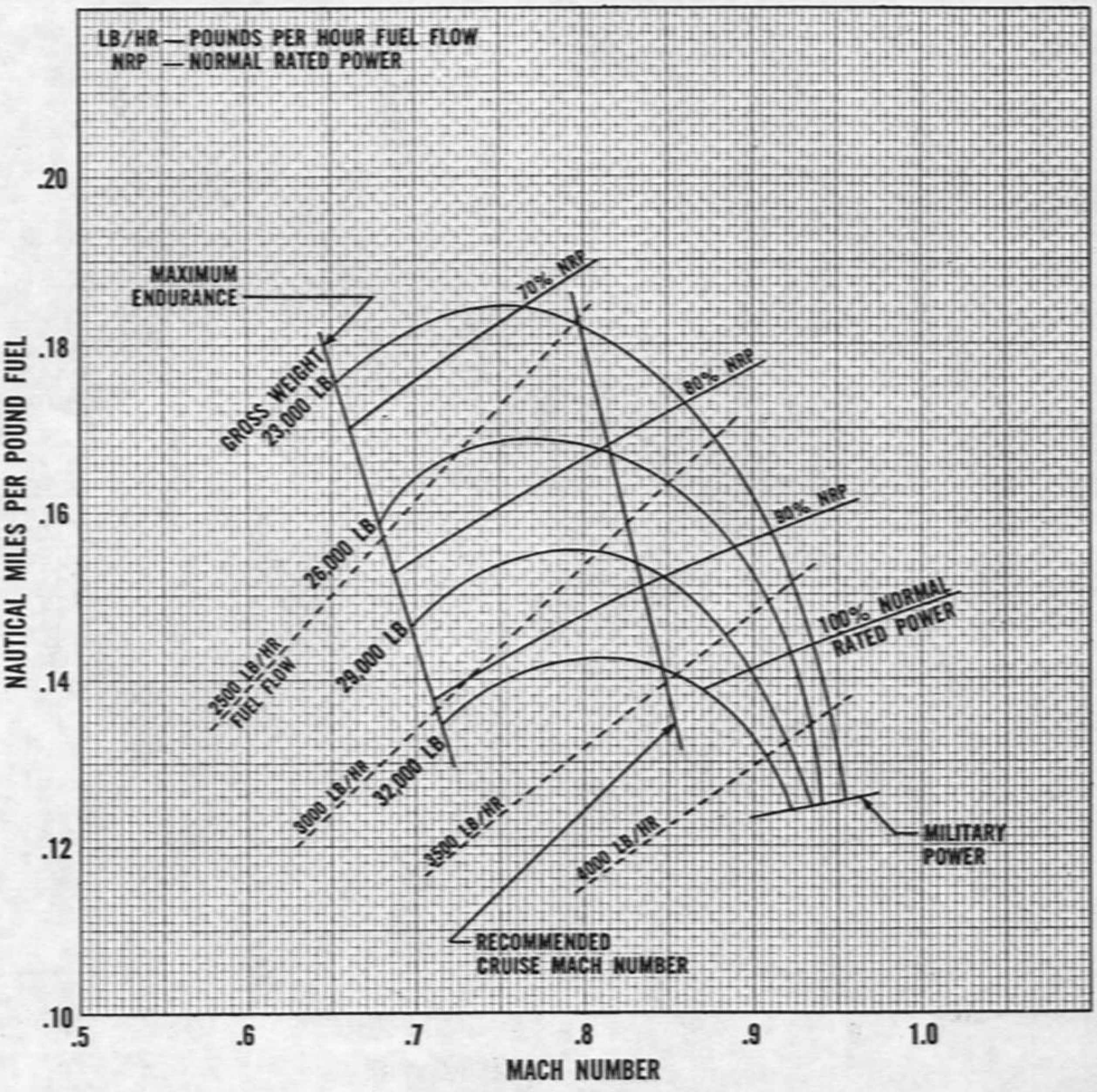


Figure A—32

CRUISE CONDITIONS

40,000 ft

Model: YF-105A

Engine: J57-P-25

Configuration: CLEAN + TWO 450 GAL. TYPE I TANKS

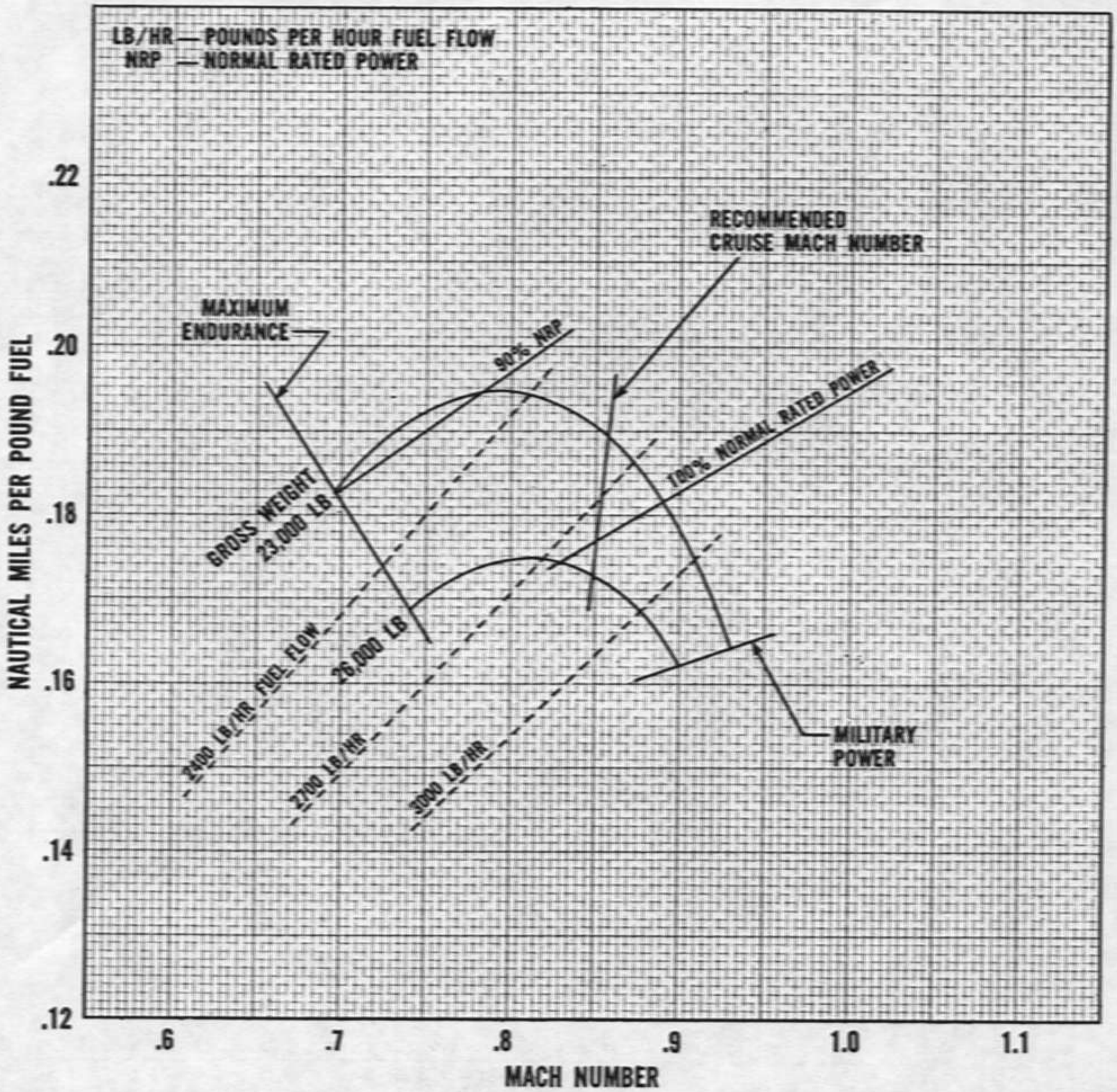


Figure A—33

ALPHABETICAL INDEX