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CHANCE VOUGHT AIRCRAFT
INCORPORATED · DALLAS, TEXAS

PRELIMINARY
TITLE

61 1

Flight Guide for
Navy Model F8U-3 Airplane BuNo. 147085
SUBMITTED UNDER

REPORT NO	DATED
E8R-11546	14 Nov 1958
MODEL	CONTRACT NO.
F8U-3	NOa(s) 57-296
DATE ISSUED	SUPERSEDING

ENGINEERING DEPARTMENT

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PREPARED	CHECKED	APPROVED
T. S. Miller	W. McCormack	C. A. Lau

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INTRODUCTION

This Flight Guide is written to provide operating information for F8U-3 airplane BuNo. 147085. Operation of the airplane systems is covered and each pilot-operated control is illustrated. Information on equipment, limitations, and predicted flight characteristics is included. Both normal and emergency flight and ground procedures are discussed.

The organization of this report, by sections, is as follows:

Section I . . . . . Description
Section II. . . . . Normal Procedures
Section III . . . . . Emergency Procedures
Section IV. . . . . Auxiliary Equipment
Section V . . . . . Operating Limitations
Section VI. . . . . Flight Characteristics
Appendix I. . . . . Operating Data Charts

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Figure 1-1 The F8U-3 Airplane

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SECTION I

DESCRIPTION

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THE AIRPLANE.

The F8U-3 airplane (figure 1-1) is a single-place supersonic all-weather fighter designed for operation from aircraft carriers. The airplane is principally identified by its shoulder mounted sweptback two-position wing, low-mounted unit horizontal tail, two-position ventral fins, and its sharp "sugar-scoop" engine air inlet.

The wing, which is similar to that on the F8U-1, incorporates an aileron and flap boundary layer air control system and full-span double-hinged "droopable" leading edges. Spoilers on the wing upper surface supplement effectiveness of the ailerons. As in the F8U-1, the ailerons are drooped to act as flaps when the wing is raised for takeoff and landing. Raising and lowering of the wing also positions the ventral fins, which augment directional stability in high-speed flight.

The unit horizontal tail, vertical fin and rudder, speedbrake, and landing gear are similar to those on the F8U-1 in appearance and functioning. Like the F8U-1, the airplane has no battery, and all electrical power is normally supplied by an ac generator driven by the engine through a constant-speed drive. DC power is provided by rectifying ac power. The airplane has an emergency generator (ram-air turbine driven) on the left forward side of the fuselage.

Principal dimensions of the airplane are:

Wing span	39 ft 11.4 in
Length (overall, static ground position)	57 ft 9.84 in
Length (overall, waterline)	58 ft 8.88 in
Height (overall, static ground position)	16 ft 1.1 in
Design gross weight (takeoff)	37,500 lbs

(Overall height with wings folded does not exceed overall height, static ground position.)

ENGINE AND AFTERBURNER.

The airplane is powered by a J75-P5N twin spool, axial flow jet engine with integral afterburner. This engine has two compressor rotor sections which are driven by separate turbine stages to create low- and high-pressure levels of compression. Engine speed is measured on the basis of high-pressure (N_2) rotor operation. Air density and temperature are monitored by an engine-driven fuel control which meters fuel to the burner cans according to throttle setting, differential pressure and engine speed.

The afterburner has a separate fuel control and incorporates an iris-type 2-position nozzle operated by compressor bleed air. The afterburner can be throttle-modulated throughout its operating range.

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ENGINE FUEL SYSTEM.

The principal components of the engine fuel system are the engine fuel control, engine-driven pump, and afterburner fuel control unit.

ENGINE-DRIVEN FUEL PUMP. The engine-driven fuel pump consists of a centrifugal booster stage and three separate gear-type pump stages. One gear pump stage supplies fuel for the engine and the other two stages supply fuel to the afterburner. Failure of engine fuel pump will automatically operate a shuttle valve to reroute the output of an afterburner pump stage to the engine. The afterburner pump stage is capable of supplying fuel to the engine for normal operation.

ENGINE FUEL CONTROL. The hydromechanical fuel control meters fuel to the engine at a rate of flow that will produce thrust compatible with throttle setting under varying operating conditions. In addition to automatically compensating for altitude and temperature changes the fuel control also governs acceleration rates to prevent compressor stall.

The fuel control incorporates an emergency feature which permits the pilot to select a more direct throttle control of fuel flow in case of malfunction of normal control. Emergency operation is selected by positioning the FUEL CONTROL switch from "NORMAL" to "EMERG." When "EMERG" is selected, most of the automatic compensating features of the fuel control are bypassed and throttle setting changes have to be made slowly and smoothly in order to avoid fuel surges with accompanying erractic engine operation.

AFTERBURNER FUEL CONTROL. The afterburner fuel control receives fuel from the two afterburner pumps. It meters fuel to the afterburner as a function of engine burner-can pressure. Normal flow to afterburner is initiated when the throttle is advanced to afterburner operating range and then moved outboard. Afterburner operation is terminated by pulling the throttle inboard out of the afterburner detent. The outboard and inboard movement of the throttle controls an electrically operated afterburner shuttle valve. A cable-operated mechanical valve coupled to the fore and aft movement of the throttle provides for positive afterburner termination if the electrically operated valve fails open. If the electrical valve fails closed, no afterburner operation is possible.

Afterburner ignition is accomplished by an igniter valve that operates on fuel and air pressure differential. The igniter provides a squirt of fuel to the burner cans to provide a fire streak to ignite afterburner fuel. The igniter is self priming and will furnish successive fire streaks each time the afterburner shuttle valve is cycled.

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ENGINE CONTROLS AND INDICATORS.

THROTTLE. The throttle (34, figure 1-2) establishes the command requirements to be followed by the fuel controls to meter fuel to the engine and afterburner. The throttle quadrant also incorporates switches to control engine cranking and ignition and afterburner lightoff. The throttle grip incorporates a microphone button and speedbrake switch on the inboard side, as well as the cruise droop switch which is actuated by rotating the grip forward or aft. Inboard from the throttle handle there is a wheel which adjusts throttle friction. The throttle has six identified positions, as follows:

"OFF" . . . . . Fully retarded; no fuel flow to engine.

"CRANK" . . . . . Momentary position inboard from "OFF" to crank engine.

"IGNITE" . . . . . Momentary position outboard from "OFF" to start ignition timer.

"IDLE" . . . . . Position stop for engine idling speed. Detent prevents inadvertent retarding of throttle to "OFF."

"MILITARY" . . . . . Position for maximum engine thrust without afterburner. Throttle positioned inboard and advanced through transition range from "IDLE."

"MAX" . . . . . Position for maximum engine thrust with afterburning. Throttle positioned outboard from "MILITARY" to light-off afterburner. Variations of power may be obtained with afterburning by moving throttle fore and aft within transition range from afterburner detent (approximately 90% rpm position) to "MAX." An afterburner control switch is actuated as the throttle is retarded from "MAX." This opens an engine valve to bleed excess air overboard, preventing compressor deceleration stall.

FUEL CONTROL SWITCH. This guarded switch (15, figure 1-2) provides for selection of fuel control method. For all normal operations, this switch is left in "NORMAL." When the switch is placed in "EMERG", emergency dc power is routed to the engine fuel control to energize the solenoid that switches the unit to the emergency function. (See ENGINE FUEL SYSTEM.)

TACHOMETER. This instrument (11, figure 1-3) indicates percentage of rated maximum high-pressure compressor (N_2) rotor speed and receives its signal from the tachometer generator which is mechanically geared to the rotor.

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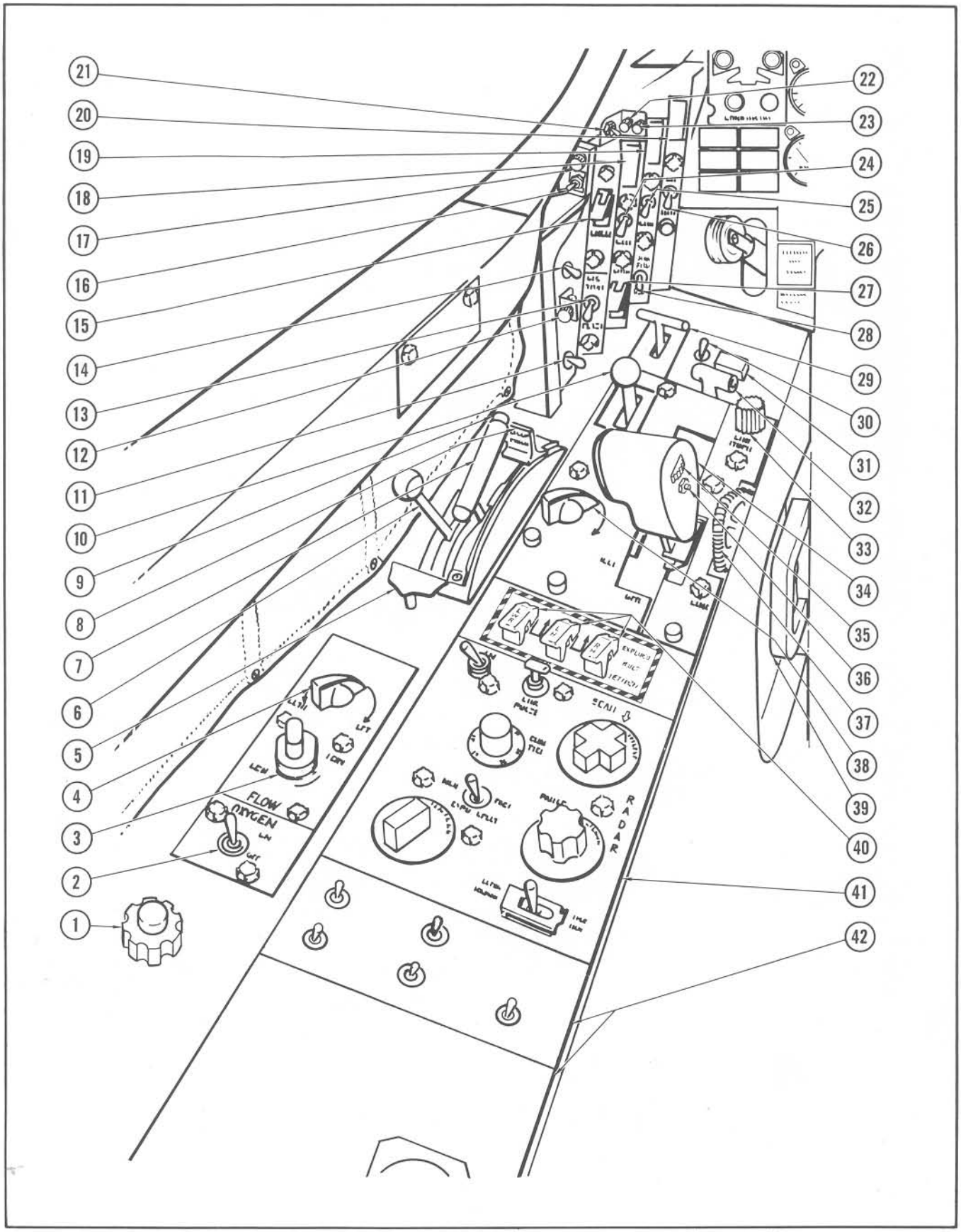


Figure 1-2. LH Console and Accordian Panels (Sheet 1)

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1. G VALVE
2. OXYGEN Switch
3. VENT AIR Control
4. PRESS SUIT TEMP Control
5. EMERG POWER Handle
6. Wing DN LOCK Handle
7. WING POSITION Handle
8. EMER DROOP & WING INC Guard
9. WING POSITION Handle Release Button
10. EMERG BRAKE Handle
11. RUDDER-AILERON INTERCONNECT Switch
12. Interconnect ACTUATOR LOCKED Light
13. EXT LIGHTS Switch
14. BEACON Switch*
15. FUEL CONTROL Switch
16. C.G. Variation Switch*
17. Transfer Pumps Warning Light*
18. YAW STAB OFF Light
19. ROLL STAB OFF Light
20. PITCH STAB OFF Light
21. MANEUVER Switch*
22. REC Light*
23. CAL Light*
24. YAW Stabilization Switch
25. ROLL Stabilization Switch
26. PITCH Stabilization Switch
27. ENGINE MASTER Switch
28. AUTO PILOT Switch
29. EMER NOSE TRIM Handle
30. DUCT BYPASS CONT Switch
31. DUCT BYP CONT OFF Light
32. Catapult Handgrip
33. UHF CHAN SELECT Knob
34. Throttle Grip (Droop Switch Incorporated)
35. SP BK Switch
36. MIKE Switch
37. Throttle Friction Control
38. RUDDER TRIM Control
39. EMER VENTRALS UP Handle
40. Missile EXPLOSIVE BOLT JETTISON Switches*
41. Radar Control Panel
42. Instrumentation Recording Control Panels*

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*Flight Test Installation

Figure 1-2. LH Console and Accordion Panels (Sheet 2)

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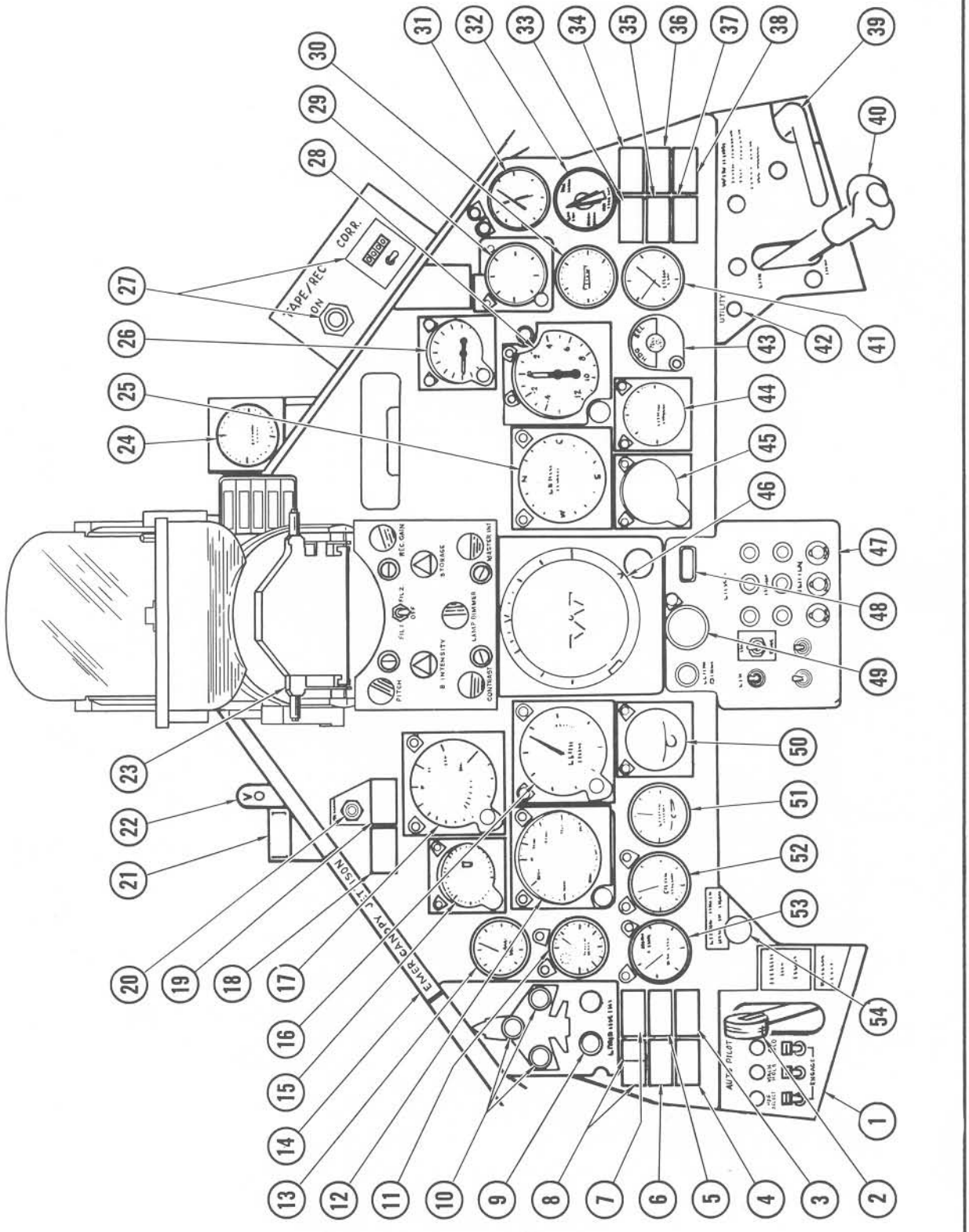


Figure 1-3. Instrument Board (Sheet 1)

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1. AUTO PILOT Control Panel
2. Landing Gear Handle
3. AUTO PILOT OFF Light
4. EMER FUEL CONTROL Light
5. SPEED BK OPEN Light
6. WING TANK ON Light
7. PROBE OUT Light (Inoperative)
8. Trim Neutral Lights
9. VENTRALS Position Indicator
10. LANDING GEAR Position Indicators
11. Tachometer
12. Radio Height Indicator
13. Engine Pressure Ratio Indicator
14. EMER CANOPY JETTISON Handle
15. Angle of Attack Indicator
16. Altimeter
17. Airspeed Indicator
18. OXYGEN Warning Light
19. FIRE Warning Light
20. Fire Warning TEST Switch
21. WHEELS Warning Light
22. Approach Indexer
23. Radar Indicator
24. Standby Compass
25. Bearing, Distance, & Magnetic Heading Indicator
26. Acceleration Indicator
27. Instrumentation Recording Light and Indicator*
28. Acceleration Indicator*
29. Clock
30. FUEL Quantity Indicator
31. FUEL FLOW Indicator
32. Fuel TANK QTY SELECTOR Switch
33. LOW HYD PRESSURE Warning Light
34. WING POSITION Light
35. ENGINE DEICING Light
36. MAIN TANK LOW LEVEL Warning Light
37. N_1 OVERSPEED Warning Light
38. BLC Light
39. WINGFOLD Handle
40. Arresting Gear Handle
41. HYD PRESS Indicator
42. UTILITY Hydraulic Pressure Check Switch
43. AUTO PILOT HDG SEL (Controller)
44. CABIN PRESSURE ALTITUDE Indicator
45. Rate of Climb Indicator
46. Attitude Indicator
47. Armament Control Panel*
48. UHF CHANNEL Indicator
49. NOSE TRIM Indicator
50. Turn-and-Bank Indicator
51. LIQUID OXYGEN Quantity Indicator
52. OIL PRESS Indicator
53. EXH TEMP Indicator
54. EMER DN LOCK Release Button

*Flight Test Installation

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Figure 1-3. Instrument Board (Sheet 2)

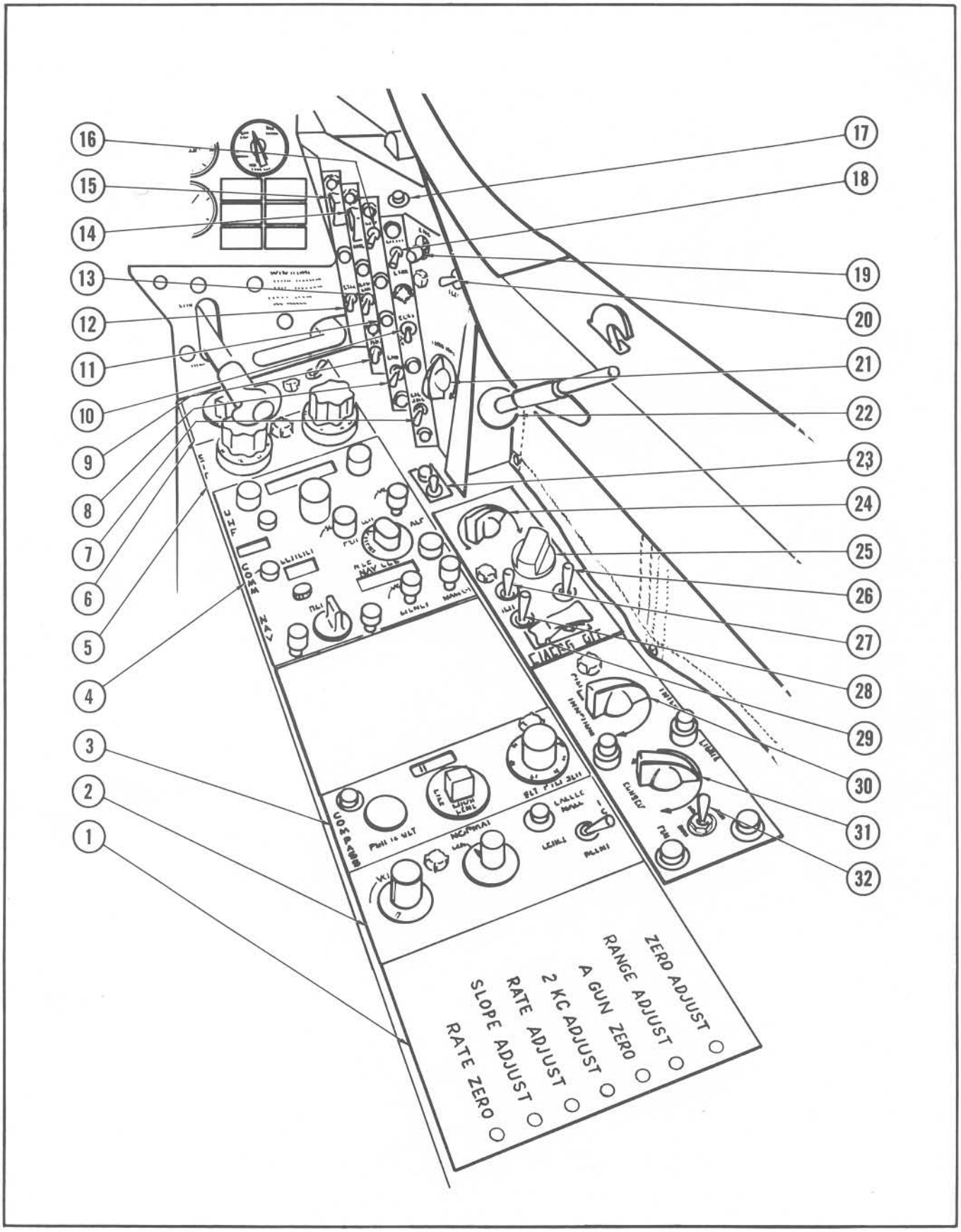


Figure 1-4 RH Console and Accordion Panels

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1. Radar Adjust Box
2. Mixing Amplifier (ICS) Panel
3. MA-1 COMPASS Control Panel
4. Communication and Navigation Control Panel
5. SIF Panel
6. IFF Panel
7. TAIL Lights Switch
8. EMER GEN Switch
9. TAXI LIGHTS Switch
10. WING Lights Switch
11. Emergency Generator ON Light
12. PITOT HEAT switch
13. STRIP LIGHTS Switch
14. FUEL DUMP Switch
15. ENGINE DEICER Switch
16. FUEL TRANS Switch
17. Exterior Lights MAN KEY Switch
18. FUS Lights Switch
19. MASTER GEN Switch
20. SEAT Adjust Switch
21. EMER LIGHTS - KNEEBOARD LTS Switch
22. Interior Canopy Release Handle
23. HIGH ALT LIGHTS Switch
24. Air Conditioning AUTO TEMP CONTROL
25. Air Conditioning MANUAL TEMP CONTROL
26. RAIN REMOVE Switch
27. DEFOG Switch
28. CABIN PRESSURE Switch
29. Air Conditioning EMERGENCY OFF Switch
30. INSTR LIGHTS Control
31. CONSOLE LIGHTS Control
32. FID LTS Switch

Figure 1-4. RH Console and Accordion Panels (Sheet 2)

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ENGINE MASTER SWITCH. This switch (27, figure 1-2) is guarded in the "ON" position and controls the fuel booster pumps and the engine fuel shutoff valve through the master fuel relay. The switch also supplies electrical power to the afterburner fuel control valve and the master fuel relay.

EXHAUST TEMPERATURE INDICATOR. This indicator (53, figure 1-3) indicates temperature of turbine exhaust gas in degrees centigrade. The indicator is graduated from 0° to 1,000°C.

ENGINE PRESSURE RATIO INDICATOR (P_{t7}/P_{t2}). This indicator (13, figure 1-3) presents a visual indication of the ratio of turbine-outlet total pressure to compressor-inlet total pressure. The indicator scale is graduated from 1.2 to 3.4. The indicator provides a reference to engine thrust for particular power settings. A two-digit window and movable index provide for accurate reading of intermediate values; and when pre-set to allowable value for takeoff, can be used as a "NO-GO" indication at T/O thrust check.

ENGINE FUEL FLOW INDICATOR. This indicator (31, figure 1-3) indicates the rate of fuel flow from the engine fuel control to the engine in pounds per hour. The indicator is graduated from 0 to 24,000 pph. Rate of fuel flow to the afterburner is not included.

ENGINE OIL PRESSURE. This indicator (52, figure 1-3) presents engine oil pressure as referenced to the breather. The dial registers from 0 to 100 psig. Normal engine operating pressure is between 40 and 50 psig.

FIRE WARNING LIGHT. This light (19, figure 1-3) is coupled to an electrical bridge-type controller which monitors resistance between continuous coaxial heat sensing elements in the engine compartment area. Overtemperature condition in the area of the sensing elements causes a change in element insulation resistance and results in the FIRE warning light being illuminated.

OIL COOLING.

The F8U-3 uses fuel to cool hydraulic oil, engine lubricating oil, and constant-speed-drive lubricating and operating oil. Since engine oil requires the most cooling, it is the only system to use two cooling stages. During most operating conditions, engine oil is cooled by an engine-mounted oil cooler that uses fuel from the engine fuel manifold down-stream of the engine-driven fuel pump. At higher operating temperatures, engine oil is routed through a section of the supplemental oil cooler. Fuel circulation through the supplemental cooler is accomplished by bypassing fuel from the main fuel manifold through the cooler then back to the main fuel cell. Supplemental cooling for engine oil automatically begins when temperature of the fuel leaving the engine oil cooler reaches 220°F and continues until temperature of the fuel drops below 205°F.

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Electrical power to operate the supplemental system is provided by the secondary dc bus. Failure of secondary dc power will leave system valves in the position held at the time of electrical failure.

The supplemental cooler is a five-section unit that has fuel flowing through it constantly. Since there is no control of this fuel flow, temperature regulation is accomplished by regulating oil flow through the cooler as in the case of engine oil. Hydraulic fluid, which flows through three sections of the cooler, requires no flow regulation to maintain temperature within operational limits. Constant-speed-drive oil, which flows through the remaining section, is temperature regulated by a valve to 225 ±10°F.

ENGINE ANTI-ICING.

Engine anti-icing is accomplished by bleeding hot air from the engine and routing it through the hollow inlet vanes. The hot air is taken from the high pressure compressor section and routed through a motor-operated valve to the vanes. The valve is operated by power from the emergency dc bus and is controlled by ENGINE DEICER switch (15, figure 1-4). When the switch is placed in "ON", the valve opens to let N_2 compressor bleed air through the inlet vanes. Below 90% rpm, sufficient engine heating may not be available to prevent formation of ice under severe conditions. This can be overcome by occasionally running the engine at higher thrust settings. The system should be turned on during periods of flight when icing conditions are anticipated. The ENGINE DEICING light (35, figure 1-3) on the instrument board is on when the deicing system is operating.

AUTOMATIC DUCT BYPASS CONTROL.

DESCRIPTION. This system provides optimum air mass flow in the inlet duct by controlling duct mach number during airplane flight above 1.4 MN. Cockpit control and indication consist of a DUCT BYPASS CONT switch and DUCT BYP CONT OFF light (30 and 31, figure 1-2) at the forward end of the left-hand console. The switch has two positions, "ON", and "OFF-RESET". With electrical power connected to the system, the light will be on when the switch is in "OFF-RESET", or when the switch is in "ON" and the monitor has shut the system off.

OPERATION. With electrical and utility system hydraulic power supplied to the system, the duct bypass doors will be open when the landing gear is down or when the wing is up. When the landing gear is up and the wing is down, the doors are closed and will remain closed up to approximately 1.4 airplane MN.

At or above 1.4 airplane MN, a free stream mach number transducer provides a reference signal of required duct mach number to a duct mach number transducer. The duct transducer compares actual duct mach number with the reference, and produces an electrical signal to the electro-hydraulic bypass door servos. Movement of the doors varies the amount of duct air bled overboard, controlling duct mach number.

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Synchronization of the doors is continuously compared by a monitor. If the doors are out of synchronization 30% or over, the system will shut down. If system electrical power is interrupted for 0.3 seconds or more, the system will shut down. If no continuous malfunction exists, the system may be reset by cycling the DUCT BYPASS CONT switch, The doors may be fully out of synchronization when a reset is attempted, as they could be if maintenance has been done on the system. In this case, the doors may be resynchronized by cycling the switch, but should not require more than four cycles.

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ENGINE START ACCESS.

The engine starting air access is located just above and aft of the left main gear upper door. The access is designed to accept a starting air adapter which connects the air output of a GTC-85 starting unit to the engine starting turbine mounted on the engine. The electrical leads for GTC control are plugged into a receptacle located against the rear bulkhead of left main wheel well near the air access.

FUEL SYSTEM.

GENERAL.

The F8U-3 fuel system consists of 12 separate fuel cells. Most of the fuel is carried in 11 fuselage bladder cells housed in sealed cavities around the engine and inlet duct. The fuselage cell cavities are vapor sealed and vented to prevent leaking fuel or vapors from accumulating in the structure or reaching the engine. The remaining cell is an integral "wet" cell in the wing center section. The wing cell is pressurized to provide for positive transfer of fuel into the main fuel cell or for rapid dumping to jettison excess wieght. Wing pressurization is controlled automatically and is pre-selected when landing gear handle is in the up position. Pressure is applied when fuel level in the main cell falls below a high-level thermistor or when the FUEL DUMP switch is placed in the dump position. (See FUEL DUMP.) Figure 1-5 is a simplified schematic of the system, and table 1-1 lists cells and capacities.

Table 1-1. Fuel Cells and Capacities

Cell	Capacity in Gallons	Approx Capacity in Pounds (JP-4)	(JP-5)
Upper Fwd Transf	82	533	549.4
Lower Fwd Transf	211	1,371.5	1,413.7
Center Mid Transf RH	68	442	455.6
Center Mid Transf LH	75	487.5	502.5
Lower Aft Transf RH	112	728	750.4
Lower Aft Transf LH	112	728	750.4
Upper Aft Transf RH	63	409.5	422.1
Upper Aft Transf LH	63	409.5	422.1
Wing	708	4,602	4,743.6
Forward Mid RH	56	364	375.2
Forward Mid LH	56	364	375.2
Main	498	3,237	3,336.6
Total Transf	1,494	9,711	10,009.8
Total Main	610	3,965	4,087
Total All	2,104	13,676	14,096.8

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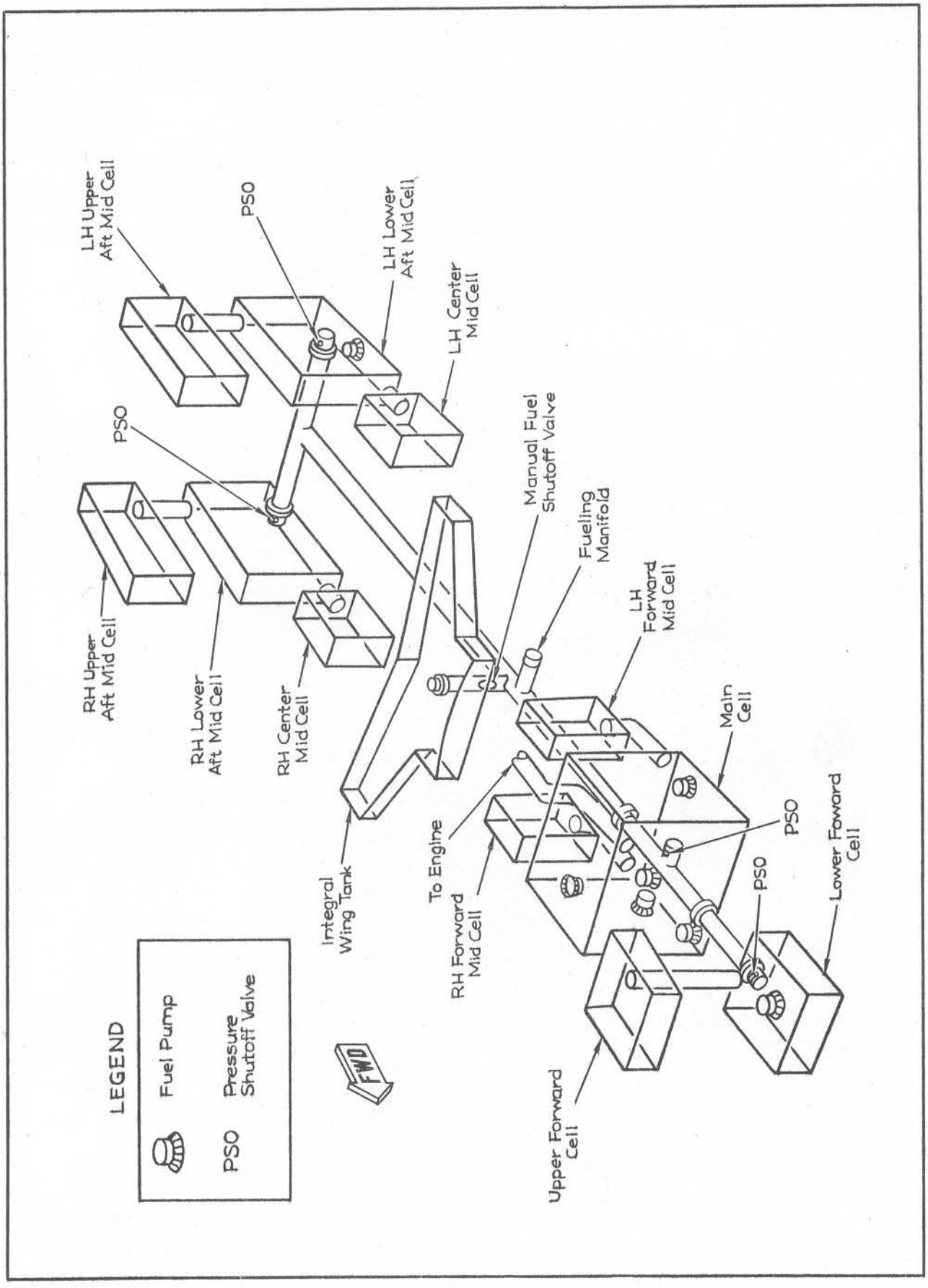


Figure 1-5 Fuel System Arrangement

FUEL TRANSFER SYSTEM.

All fuel to the engine is supplied from the fuselage main cell, and provisions are provided to resupply the main cell from all other fuel storage cells.

There is a fuel transfer booster pump in the lower forward cluster cell and another transfer pump in the lower left-hand aft cluster cell. These two pumps provide for transfer of fuselage fuel to the main cell and are controlled by the FUEL TRANS switch. Fuel flow between cluster cells is facilitated by interconnect plumbing and the pumps are located to drain their respective clusters in level flight or moderate climb attitudes.

Fuel flow from the clusters to the main cell is controlled by a pressure shutoff valve so that fuel flows into the main cell at approximately the same rate from both the forward and aft cells to maintain airplane balance and stability. Because of the difference in capacity of the forward and aft clusters the forward clusters will pump dry first. When the cluster cells are empty, the main cell level will drop slightly, uncovering a level sensing device which starts wing fuel transfer. A C.G. variation switch (16, figure 1-2) is installed in airplane BuNo. 147085 to allow either transfer pump to be shut off for flight testing. Placing the switch in "FWD" shuts off the forward pump; "AFT" shuts off the aft pump. A warning light (17, figure 1-2) above the switch is illuminated if either aft or forward cluster is pumped dry.

FUEL BOOST.

In addition to the transfer pumps in the forward cluster and aft cluster, there are five additional pumps located in the main fuel cell. Two of these operate any time the ENGINE MASTER switch is "ON". Of the three remaining pumps, two are operated only during inverted flight or when the airplane assumes a pitch attitude greater than 40° noseup or 10° nose down. The remaining fuel pump operates any time the throttle is advanced into the afterburner operating range.

One pump receives power from the primary ac bus while all the other pumps operate on secondary ac power. In case of failure of secondary ac power the one operating pump and the engine-driven fuel pump will supply sufficient fuel flow to satisfy all engine fuel requirements except maximum afterburning. In case of failure of all airplane booster pumps, gravity feed plug the engine-driven fuel pumps will supply minimum engine fuel requirements for normal flight attitudes at low altitudes (non afterburning below 30,000 feet, or with A/B below 10,000 feet).

FUEL CELL PRESSURIZATION AND VENTING.

The fuselage cells are pressurized to approximately 1/2 psi above cell cavity pressure by engine bleed air that is routed from the air conditioner. Positive cell pressure is maintained by a system of regulators

and sensing lines. The system incorporates an emergency vent that uses ram air to maintain positive cell pressures in normal flight attitudes if the engine flames out or the emergency air-conditioning valve is turned off.

Wing cell transfer pressure (approximately 7-1/2 psi above ambient) is applied only when fuel level in the main cell drops 200 gallons below full and wheels are in the up position. Wing pressurization system also incorporates emergency vent and relief valves that prevent vacuum buildup or overpressurization in case of failure of the regulating system.

All fuel cells are vented overboard at the right-hand wing tip through a common line. The cells also have common pressure relief lines to serve the separate pressure relief systems for wing and fuselage cells. The pressure/vent system allows the fuel cells to breathe during changes in altitude.

FUEL CELL CAVITY VENTING.

The fuselage fuel cell cavities are sealed to prevent leaking vapors or fuel from entering the airplane structure. The forward and rear fuel cavities are vented overboard on both sides of the fuselage in areas of prevailing negatives pressure.

FUEL DUMPING.

Fuel dumping is limited to jettisoning fuel from the wing integral cell. Compressed air for fuel dumping is provided through normal wing pressurization channels. In case of a dead or windmilling engine fuel cannot be dumped rapidly in as much as a partial vacuum exists in the high speed compressor. Fuel can be dumped at a reduced rate by gravity flow. Fuel is dumped from ports in the lower side of the wing outer panels near the wing tips.

FUEL SERVICING.

The fuel system is designed only for pressure fueling and is refueled through a single-point fueling manifold in the left main wheel well. Solenoid overrides on the pilot float valves of the pressure shutoff valves permit control of refueling through a fuel selector switch located near the fueling manifold. This switch provides for checking the fuel system valves, defueling the airplane, or fueling the airplane to total or partial capacity. When partial fueling is selected, only the fuselage cells are filled.

FUEL SYSTEM CONTROLS AND INDICATORS.

FUEL TRANSFER SWITCH. The FUEL TRANS switch (16, figure 1-4) controls the two fuselage transfer pumps and provides a manual control of wing pressurization. When the switch is placed in "AUTO", the transfer pumps are energized by secondary ac power and remain energized until the switch is placed in "PUMP OFF-WING ON". Fuel flow from the wing will

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start any time fuel level in the main fuel cell drops approximately 80 gallons below transfer level. Placing the switch in "PUMP OFF-WING ON" and back to "AUTO" will extinguish the WING TANK ON light and reset the wing pressurization system if transient flight conditions have started wing transfer. Placing the switch in "PUMP-OFF-WING ON" pressurizes the wing cell independently of operation of the automatic sequencing system.

MAIN TANK LOW LEVEL WARNING LIGHT. The main fuel cell is equipped with thermistors that sense fuel level and control wing pressurization of the wing and operation of the MAIN TANK LOW LEVEL warning light (36, figure 1-3). The light comes on when fuel in the main cell reaches less than 1,425 pounds in level flight. This light does not necessarily indicate quantity of total fuel remaining on board but does indicate fuel remaining available in the main cell.

WING TANK ON LIGHT. The WING TANK ON light (6, figure 1-3) comes on when the wing is automatically pressurized by lowering of fuel level in the main fuel cell. The light will be extinguished when the FUEL TRANS switch is placed in "PUMP OFF-WING ON".

FUEL DUMP SWITCH. The FUEL DUMP switch (14, figure 1-4) opens the fuel dump valves and pressurizes the wing so that wing fuel may be dumped overboard to lighten the airplane. Dumping may be stopped at any time by returning the switch to the off position.

FUEL QUANTITY INDICATING SYSTEM. The FUEL quantity indicator (30, figure 1-3) provides separate indications of total fuel remaining and of fuel remaining in groups of cells. The indicator window shows total fuel in pounds in all fuel cells and counts down as fuel is consumed. The indicator pointer indicates fuel quantity in groups of cells selected with the TANK QTY SELECTOR switch (32, figure 1-3). The positions of the selector switch and the cell groups selected are as follows:

"WING"	wing integral cell.
"AFT"	six aft fuselage cluster cells.
"FWD"	upper and lower forward cells.
"MAIN"	main and right and left mid fuselage cells.
"TEST"	zeros indicator for test

The system uses capacitance probes in all cells to determine amount of fuel remaining. The system is self referencing and the accuracy is not affected by changes in fuel or temperature. Indications are very accurate in level flight attitudes and reasonably reliable in all other flight attitudes.

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MAIN ELECTRICAL SYSTEM.

GENERAL.

Electrical power for the airplane is normally supplied by a 20-KVA, 400-cps, 115/200-volt, three- phase ac generator. There is no battery or dc generator and dc electrical requirements are provided by rectifying ac power. The generator is located inside the engine nose cone and is driven by a gear train and constant-speed drive that is mechanically coupled to the engine low-pressure (N_1) compressor. Any compressor rpm above "IDLE" power setting will furnish sufficient drive speed to operate the generator.

Note

Immediately after starting the engine (MASTER GEN switch in "MAIN") it may be necessary to advance throttle above "IDLE" momentarily to place the generator on the line.

The ac generating system incorporates an automatic control box, voltage regulator and speed switch. The speed switch senses overspeed or underspeed conditions of the constant-speed drive and furnishes this information to the control box. The control box also senses over- and undervoltage. Should the generator output fail to fall within power requirement limits, the control box overrides the MASTER GEN switch and disconnects electrical power.

Emergency electrical power is supplied by a 5- KVA, 400-cps, three-phase generator that is part of the emergency ram air power package located on the left side of the fuselage. This emergency power package is extended by pulling the EMERG POWER handle (5, figure 1-2) on the left-hand console. When the handle is pulled, compressed air extends the package, which cannot be retracted in flight.

Since the emergency power package contains only an ac generator, dc power is supplied by rectifying ac power. Power is distributed through emergency and primary ac and dc buses.

Operation of electrical equipment when using the emergency power package is subject to conditions under which the package is operating. If the package is supplying only electrical power, it will furnish adequate power at touchdown speeds with the EMER GEN switch in "ON." If, however, the unit is also being used to supply hydraulic power, the EMER GEN switch must be placed in "LAND" to decrease electrical load so that hydraulic pressure and flow rate at final approach speeds will be adequate to maintain control of the airplane.

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Table 1-2 and Table 1-3 are breakdowns of the three airplane electrical ac and dc buses showing the systems and major components each bus services.

Table 1-2. AC Power Distribution by Buses

	Primary	Secondary	Emergency
Phase A	Stabilization (Yaw Stab and Trim, Pitch Stab, and Aileron-Rudder Interconnect)	Transformer-Rectifier (DC Power)	Stabilization (Roll Stab and Roll Trim)
		Automatic Missile Control System	
	Intergrated Communications Package	Aft Transfer Pump	Attitude Indicator
	Communications Blower	A/P & Stabilization	
	MA-1 Compass	Forward Transfer Pump	Integrated Communications Package
	Fwd Main Boost Pump	Aft Main Boost Pump	
	Seat Control	#1 Inverted Flight Pump	
		#2 Inverted Flight Pump	Interior Lights
		A/B Fuel Boost Pump	
		Air Bottle Heaters	Liquid Oxygen
		Nose Gear Steering	
Phase B	Stabilization (Yaw Stab and Trim, Pitch Stab, and Aileron-Rudder Interconnect)	Transformer-Rectifier (DC Power)	Stabilization (Roll Stab & Roll Trim)
		Automatic Missile Control System	
	Integrated Communications Package	Aft Transfer Pump	Press Ratio Indicator
	Communications Blower	Forward Transfer Pump	
	MA-1 Compass	Aft Main Boost Pump	Attitude Indicator
	Fwd Main Boost Pump	#1 Inverted Flight Pump	Integrated Communications Package
	Aileron-Rudder Interconnect	#2 Inverted Flight Pump	Instrument Transformer
	Radar Altimeter	A/B Fuel Boost Pump	Fuel Quantity Indicating
		A/P & Stabilization	Pitot Heat
Phase C	Stabilization (Yaw Stab & Trim, Pitch Stab, & Aileron-Rudder Interconnect	Transformer-Rectifier (DC Power)	Stabilization (Roll Stab & Roll Trim)
		Automatic Missile Control System	
		Aft Transfer Pump	Attitude Indicator
	Integrated Communications Package	Forward Transfer Pump	
	Communications Blower	A/P & Stabilization	Integrated Communications Package
	MA-1 Compass	Aft Main Boost Pump	
	Fwd Main Boost Pump	#1 Inverted Flight Pump	
	Pitot Heater	#2 Inverted Flight Pump	
	Duct By-Pass	A/B Fuel Boost Pump	
		Vane Heater	
		Seat Control	

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Table 1-3. DC Power Distribution by Buses

Primary	Secondary	Emergency	
Integrated Communications Package	Boost Pump	Indicator Lights	Boundary Layer Control
MA-1 Compass	Trim Neut & Wingfold	Fuel Dump	Wing Incidence
Speed Brake	Landing Gear Safety Solenoid	Leading Edge Droop	Floodlight
Ail-Rudder Interconnect	Nose Gear Steering	Liquid Oxygen Warning	Integrated Elec Package
Duct By-Pass	Arresting Hook Solenoid	Fuel Control	Fire Detection
#1 Fueling & IFR*Relay	Taxi Light	Landing Gear Position	Attitude Indicator (V.G.I.)
#2 Fueling & IFR*Relay	Exterior Lights	Fuel Transfer	A/P & Stabilization
Radar Altimeter	Supplemental Oil Cooler	Wing Tip LIghts	N_1 Overspeed
A/P & Stabilization	Inflight Refueling*	Engine Ignite	Afterburner Control
Static System Transfer Valve	A/P & Stabilization	Approach Lights	Missile Explosive Bolt Jettison
Forward Firing Missile Jettison		Master Switch	
		Engine Anti-Ice	
		Ignition Timer	
		Ventrals Positions	

*Inflight Refueling not installed in BuNo. 147085 in flight test configuration.

EXTERNAL POWER RECEPTACLE.

Airplane BuNo. 147085 has an external electrical power receptacle on the left side of the fuselage below and slightly aft of the cockpit. There are two power receptacles for instrumentation on the lower door of the right main gear. Normal airplane electrical system requires 115/200-volt, 400-cps power with phase rotation ABC.

ELECTRICAL SYSTEM CONTROLS AND INDICATORS.

MASTER GENERATOR SWITCH. The MASTER GEN switch (19, figure 1-4) controls normal electrical power and has three positions labeled "MAIN", "OFF" and "EXT-RESET". The "MAIN" position connects power to the airplane electrical distribution system and is the normal position for flight. The "EXT-RESET" position is used for engine starting, which requires an external source of 400-cps three-phase 115/200-volt ac power. This switch position is also used to reset the power relays that control switching of generator power to airplane buses if the relays have dropped out due to generating system control box override of the MASTER GEN switch or momentary power interruption. In the "OFF" position, power from both the airplane generator and an external source is interrupted. The switch has no effect on emergency power distribution.

EMERGENCY GENERATOR SWITCH. Emergency electrical power is controlled by the EMER GEN switch (8,figure 1-4). This switch has three positions "OFF", "ON" and "LAND". In the "ON" position, power is supplied to the emergency and primary buses. In the "LAND" position, power is supplied only to the emergency buses to relieve electrical loading of the

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emergency generator when its ram air turbine is also being used to supply emergency hydraulic power.

EMERGENCY GENERATOR ON LIGHT. This light (11, figure 1-4) indicates that the generator is extended and is providing electrical power to the emergency transformer-rectifier.

UTILITY HYDRAULIC SYSTEM (See figure 1-6).

The utility hydraulic system provides pressure for operating the following equipment:

Ventral Fins	Speedbrake
Wheel Brakes	Wingfold
Flaps	Two-Position Wing
Leading Edges	Inlet Duct Bypass Doors
Arresting Gear	Nose Gear Steering
Missile Launchers	Landing Gear
	Inflight Refueling Probe*

The system uses MIL-O-5606 fluid, which is cooled by a flowing-fuel cooler, and the normal system pressure is 3,000 psi. Pressure is supplied by a variable-volume engine driven pump. Supply fluid to replace normal system leakage is provided by a hydraulic reservoir. The reservoir also serves to provide differential oil volumes as required by circuit cycling and to maintain a constant supply pressure to the engine-driven pump to prevent cavitation. The reservoir is pressurized by system pressure which is applied to the reservoir piston.

The UTILITY hydraulic pressure test switch (42, figure 1-3) causes pointer "2" of the HYD PRESS indicator (41, figure 1-3) to indicate utility hydraulic system pressure.

PNEUMATIC SYSTEM (See figure 1-7).

The pneumatic system provides emergency power to actuate systems that are normally powered by the utility hydraulic system. It also provides power to extend the emergency power package.

Emergency pneumatic pressure operates the following equipment:

Emergency Power Package (extend)
Ventral Fins (raise for landing)
Two-Position Wing (raise for landing)
Wing Leading Edges (droop)
Landing Gear (lower)
Wheel Brakes (give emergency braking)
Fuel Dump Valves (open for dumping wing fuel)

*Not installed on airplane BuNo. 147085 at present.

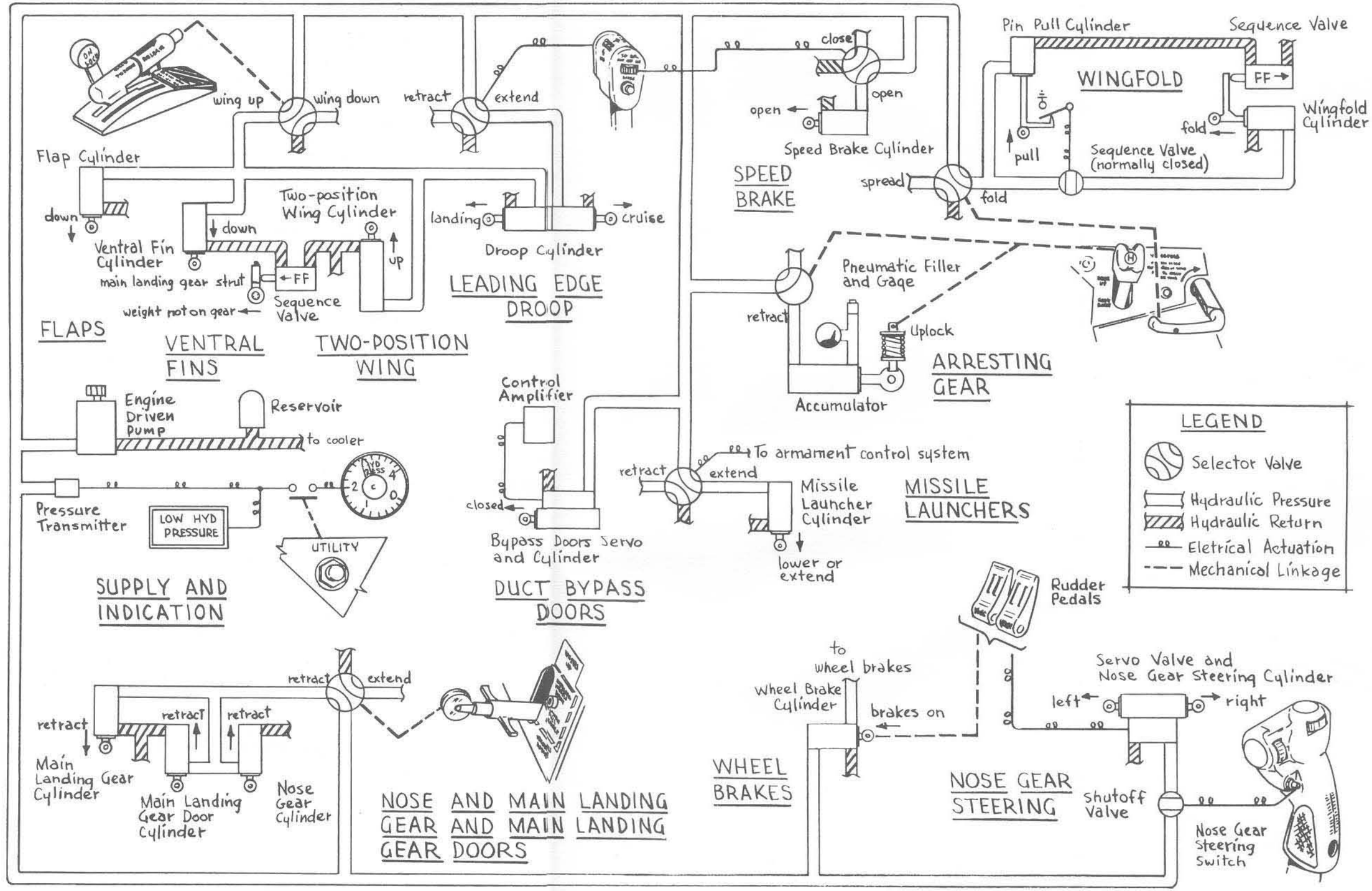


Figure 1-6 Utility Hydraulic System

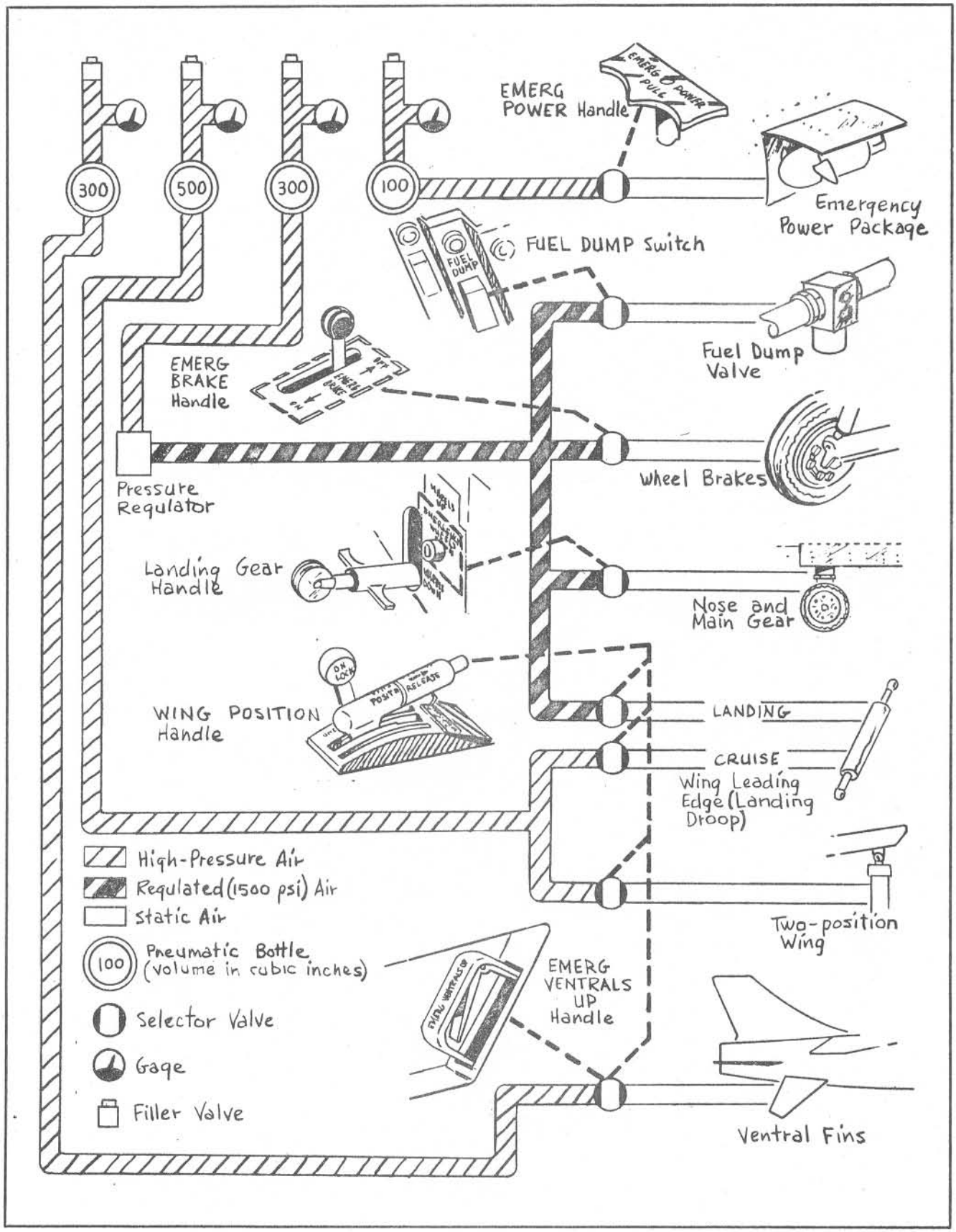


Figure 1-7 Pneumatic System

The four air bottles of the system are rated at 3,000 psi and are designed to be charged with dry nitrogen. The bottles are charged to between 2,350 and 2,450 psi, depending on ambient temperature, and are then heated to approximately 100°F to raise pressure to approximately 3,000 psi.

Note

Actuation of controls to obtain emergency operation is discussed in connection with the systems involved.

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POWER CONTROL HYDRAULIC SYSTEM (See figure 1-8).

GENERAL.

The F8U-3 has two full power control systems, PC 1 and PC 2, which supply hydraulic fluid under pressure to operate the flight control surfaces and the stabilization and auto pilot system actuators. To provide positive reliability, the two power control systems are completely independent and hydraulic fluid cannot be transferred from one system to the other. The systems are identical except that PC 1 includes the emergency hydraulic system and the roll stabilization and auto pilot actuator and PC 2 includes the yaw stabilization actuator, the aileron-rudder interconnect actuator, the pitch stabilization actuator, and the pitch auto pilot actuator. Each of the control surfaces is operated by hydraulic power control packages which contain tandem power control cylinders with a common piston. One half of each cylinder is supplied by PC 1 and the other half is supplied by PC 2. In the event of the failure of one system, the other system has sufficient pressure and flow to position the control surfaces for all normal flight loads. Accelerated flight maneuvers that impose high hinge moment and surface loads on the controls may cause the actuators to stall and result in slow or partial response to directed surface deflection.

Both of the systems use MIL-O-5606 fluid, which is cooled by a flowing-fuel cooler, and the normal operated pressure is 3,000 psi. This pressure is supplied by variable-volume engine-driven pumps. Each system incorporates an accumulator to supplement the pumps, to absorb pressure surges in the system, and to maintain pressure in the system momentarily while the emergency power package is extended in the event of failure of the engine-driven pumps. The accumulators are located in the main landing gear tension struts.

Supply fluid to replace normal system leakage is provided by a hydraulic reservoir. The reservoir also serves to provide differential oil volumes as required to maintain fluid demands occasioned by control movement and to maintain a constant supply pressure to the engine-driven pump to prevent cavitation. PC 2 reservoir is pressurized by system pressure and PC 1 reservoir is pressurized by a 25-cubic-inch air bottle. This bottle provides sufficient supply pressure to insure operation of the emergency pump except under the most adverse operating conditions.

The emergency power control circuit consists of a ram air turbine which drives a generator and a constant-displacement hydraulic pump. This circuit will provide sufficient hydraulic power to PC 1 to operate the control surfaces and the roll stabilization circuit, assuming that hydraulic fluid has not been lost from PC 1. The circuit is put into operation when the EMERG POWER handle (5, figure 1-2)

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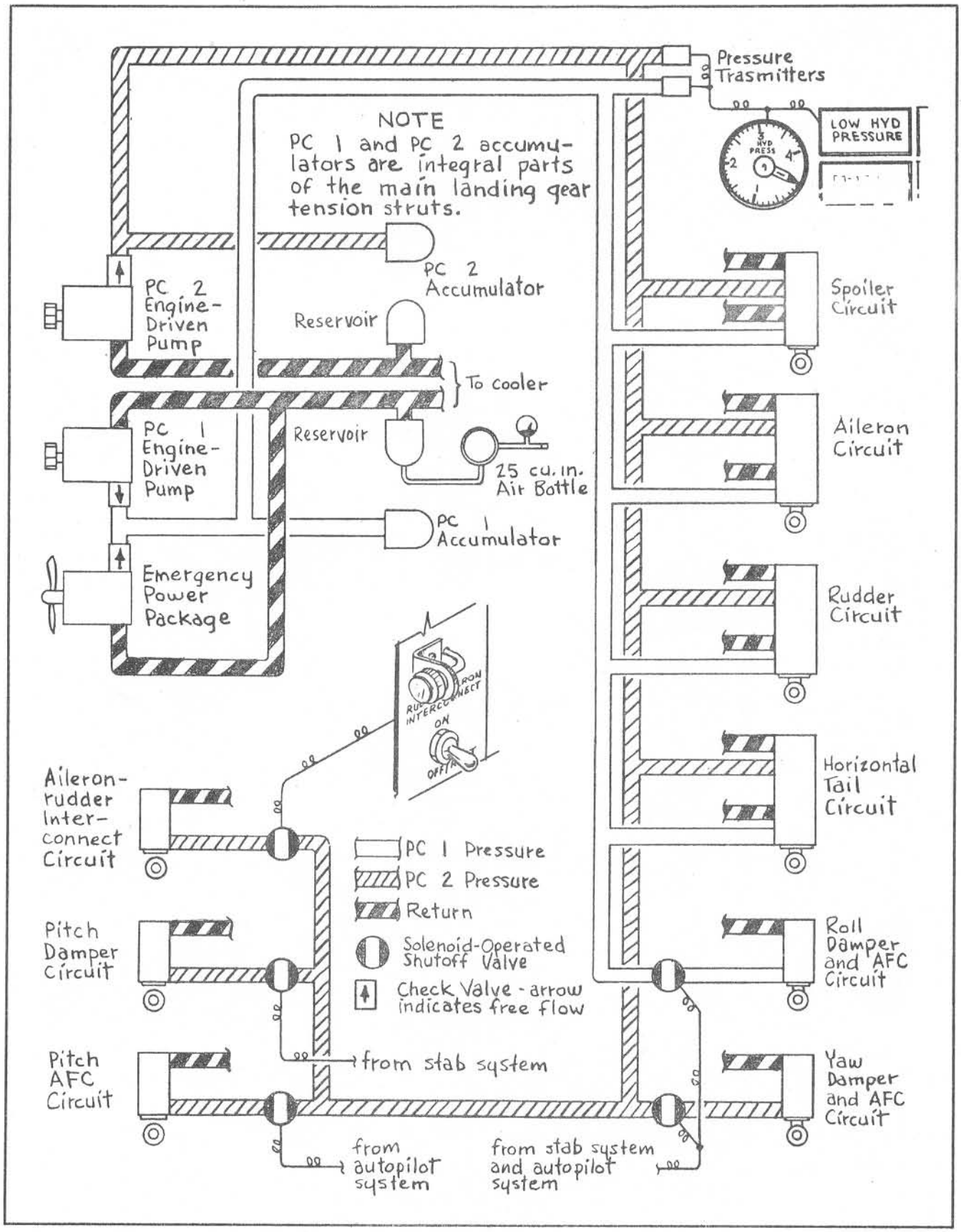


Figure 1-8. Power Control Hydraulic System

is pulled to extend the ram air turbine into the airstream. Rapid, repeated, full-control cycling at low airspeeds will put excessive flow demand on the pump and result in slow and/or partial surface response. Once the emergency power package has been extended it can be retracted only on the ground.

HYDRAULIC PRESSURE INDICATOR.

The HYD PRESS indicator (41, figure 1-4) contains two pressure indication pointers, one for PC 1 and one for PC 2, marked "1" and "2." When the emergency power package is extended emergency pressure is read from pointer"1." Utility hydraulic pressure may also be read on the indicator by depressing the "UTILITY" switch (42, figure 1-4). Utility hydraulic pressure is indicated by pointer "2."

LOW HYD PRESSURE WARNING LIGHT.

The LOW HYD PRESSURE warning light (33, figure 1-3) will be illuminated whenever pressure in either of the PC systems drops below 1,500 psi or that in the utility system drops below 700 psi. The HYD PRESS indicator must be used to determine which system is malfunctioning. It should be noted that the light is operative when the emergency power package is extended and emergency hydraulic power is being supplied to PC 1.

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FLIGHT CONTROL SYSTEMS.

GENERAL.

Primary flight control is achieved by a full hydraulic power system, using a conventional stick and rudder pedals. Longitudinal and lateral control systems use pushrods and mechanical linkage from the control stick to the hydraulic servo systems which drive the surfaces. The directional control system uses cables from the rudder pedals to a control rod system which drives the rudder surface through a hydraulic servo system. All systems have induced artificial feel to give the pilot a force reference comparable to control position and movement. Funk spring struts in all control rod systems protect the system from excessive pilot-applied and external loads in the power-off condition. All control rod systems have preload springs between feel spring ground and the power control hydraulic valves in order to remove linkage slop and to increase the stiffness of the system. Each primary control surface is trimmable and responds to stabilization or auto pilot system signals. (Refer to AUTOMATIC FLIGHT CONTROL SYSTEM).

LONGITUDINAL CONTROL.

GENERAL. The control surface is a unit horizontal tail which also functions as a trimmable stabilizer. The two independent surfaces are mechanically snychronized by a control linkage which drives the separate power control cylinders. Movement of the tail is controlled by stick movement, trim signals, stabilization system signals, and auto pilot system signals. Stick travel (with no pitch trim) is approximately 2.5 inches forward and 8.5 inches aft. Stick breakout force at neutral is approximately 1.5 pounds. Tail surface throw with the wing down is from 32° trailing edge up to 8° trailing edge down. With the wing up, the surface throw is from 24°24' trailing edge up to 8° trailing edge down.

ARTIFICIAL FEEL. Since the irreversible power control system prevents air load feedback, artificial feel is provided from three sources. Feel forces proportional to the degree of surface deflection are provided by springs. Feel forces proportional to constant acceleration and/or to pitching acceleration are provided by dual bobweights, mounted parallel to the primary control rod system. The forward bobweight is connected to the control stick, and the aft bobweight is connected to the variable gain linkage. The bobweight forces act in opposition to each other under steady acceleration, but due to the effect of the aft bobweight being lighter at the stick, the net result at the stick is a force additive to the feel spring force. During pitching accelerations, the bobweight forces work together to oppose any pilot effect to increase the magnitude of the pitching acceleration. Feel forces proportional to rate of surface deflection are provided by a viscous damper and the autopilot actuator.

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VARIABLE GAIN. A variable gain linkage downstream from the feel spring matches pitch sensitivity of the stick to the aerodynamic parameters of the airplane by changing the ratio of stick travel to surface deflection.

NORMAL PITCH TRIM. Pitch trim signals from the switch on the stick actuate a motor-driven actuator upstream from the variable gain linkage. This pitch trim actuator is mounted parallel to the primary control rod system and in series with the feel spring. Feel neutral will trim to match a new tail trim position, and the stick neutral will reposition to match a new trim setting. This allows the variable gain linkage to maintain the same pitch sensitivity about various stick neutral positions. Pitch trim throw is from 0°30' to 17°45' trailing edge up when the wing is down. With the wing up, trim throw is from 10° trailing edge up to 7°45' trailing edge down.

EMERGENCY PITCH TRIM. Emergency pitch trim signals from the EMER PITCH TRIM handle replace the normal pitch trim system if it malfunctions.

AUTOMATIC PITCH TRIM. Automatic (nose-down) pitch trim is mechanically introduced downstream from the variable gain linkage when the wing is raised. A cable attached to the wing positions a preloaded spring strut to feed the trim signal to the tail. Since this trim signal will not reposition the control stick, the optimum sensitivity position of the control stick is maintained during landing and takeoff. The spring strut contains a viscous damper to control the rate of automatic trim signal removal if the cable fails with the wing in the up position.

LATERAL CONTROL.

GENERAL. The control surfaces are conventional ailerons, supplemented by spoilers to increase the rate of roll. Stick travel is approximately 5.3 inches from full left to full right. Stick breakout force at neutral is approximately 1.5 pounds. Aileron throw with the wing down is from 20° trailing edge up to 20° trailing edge down. With the wing up, the ailerons are automatically drooped for flap action to a 40° trailing edge down position from contour neutral. Available aileron throw is then from 45° trailing edge up to 15° trailing edge down about the drooped position. (5° trailing edge up to 55° trailing edge down about wing contour neutral). When the wing is down, stops limit aileron travel to ±10°, in order to prevent inadvertent exceeding of normal roll restrictions. An increase of 10 pounds in stick force will overcome these stops to gain full aileron travel. Full lateral stick throw requires approximately 19 pounds force. Full-travel stick force required with the wing up is approximately the same as with the wing down.

The spoilers are slaved to the aileron power control package, and are raised in proportion to aileron deflection from 2° trailing edge up to 16° trailing edge up. Maximum spoiler throw is 52° and is reached at the aileron-up 16° position.

ARTIFICIAL FEEL. Since the irreversible power control system prevents airload feedback, artificial feel is provided by a spring, to give the pilot a force reference at the stick proportional to surface deflection.

ROLL TRIM. Trim signals from the roll trim knob on the control stick grip operate an electro-hydraulic actuator mounted downstream of the feel spring, and in series with the primary control rod system. This actuator also serves the flight stabilization and auto pilot systems. Stick neutral position will not change with trim or stabilization changes. Available roll trim is from 5°30' trailing edge up to 5°30' trailing edge down. No emergency trim is provided.

DIRECTIONAL CONTROL.

GENERAL. The control surface is a conventional rudder. Rudder pedal position is adjustable fore and aft by a handcrank mounted between the pedals. Pedal movement, rudder trim signals, stabilization system and auto pilot signals control rudder movement. Rudder throw with the wing up is from 25° left to 25° right. With the wing down, throw is limited by stops to from 7°30' right to 7°30' left.

ARTIFICIAL FEEL. Since the irreversible power control system prevents airload feedback, artificial feel is provided by springs. With the wing up, two springs in parallel with the primary control system provide feel forces proportional to rudder deflection. With the wing down, an additional spring is engaged by disengagement of the 7°30' stops. Pedal forces are approximately doubled when the wing is down.

RUDDER TRIM. Trim signals from the RUDDER TRIM knob operate an electro-hydraulic actuator mounted downstream of the feel springs and in series with the primary control rod system. This actuator also serves the flight stabilization and auto pilot systems. Rudder pedal position will not change with trim or stabilization changes. Available rudder trim is from 3°30' left to 3°30' right. No emergency trim is provided.

BOUNDARY LAYER CONTROL (BLC) SYSTEM.

This system consists of ducting, pipes, and valves used to direct engine high-pressure bleed air over the upper surface of the flaps and ailerons when the airplane is in the landing configuration. The BLC control valve is mechanically coupled to open as the wing is raised. With the wing up, the BLC warning light (38, figure 1-3) burns if the system is unbalanced, or the pressure is too low, or when rpm is below 80%. The blowing air over the ailerons and flaps increases the effectiveness of these surfaces, giving added stability and control when the wing is up.

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AUTOMATIC FLIGHT CONTROL SYSTEM.

GENERAL.

Airplane BuNo. 147085 is equipped with a prototype automatic flight control system (AFC system). This system provides trim, augmented flight stabilization and automatic flight control with selected functions including stick force guidance. The automatic flight control system can perform all normal flight maneuvers subject to limitations of 5 "G" positive or -1 "G" negative acceleration and can be overridden at any time by applying firm control force.

MECHANICAL OPERATION.

Trim, stabilization, and autopilot control are accomplished by electro-hydraulic servo systems using actuators powered by the PC-1 and PC-2 hydraulic systems. The actuators are used to change mechanically the relative length of the control linkage. Actuators in the aileron and rudder linkage are of the limited authority type and are used for trim, stabilization and autopilot control. The unit horizontal tail (UHT) control linkage incorporates three separate actuators, a hydraulic autopilot control actuator, a hydraulic stabilization actuator, and an electro-mechanical pitch trim actuator (refer to figure 1-9 for simplified schematic of pitch control linkage). The pitch trim actuator is in series with the feel spring and moves the control stick and surfaces simultaneously. When the autopilot "hold" modes are engaged, the pitch trim actuator is controlled by switches on the feel spring that positions the stick to follow the autopilot actuator. The autopilot actuator is connected in parallel with the normal linkage so that is acts as an hydraulic viscous damper when the autopilot is not engaged.

The flight stabilization pitch damper actuator is isolated from the normal pitch control linkage and moves the surfaces by repositioning a bellcrank on the surface side of the variable-gain linkage. The movement of this actuator does not affect control column position althrough actuator movement may be felt through the linkage in rough air. All of the stabilization actuators are of the "fail to neutral" type and incorporate cams that lock them in neutral when they are turned off or when there is failure of either electrical or hydraulic power.

AUGMENTED FLIGHT STABILIZATION.

Augmentation of the inherent stability of the airplane is provided by electronically controlled damper actuators in the UHT and rudder control linkage under all flight conditions and in the aileron control linkage with the wing raised. There is also a separate stabilization system (aileron-rudder interconnect) in the rudder control linkage to eliminate the adverse yaw characteristic associated with roll maneuvers.

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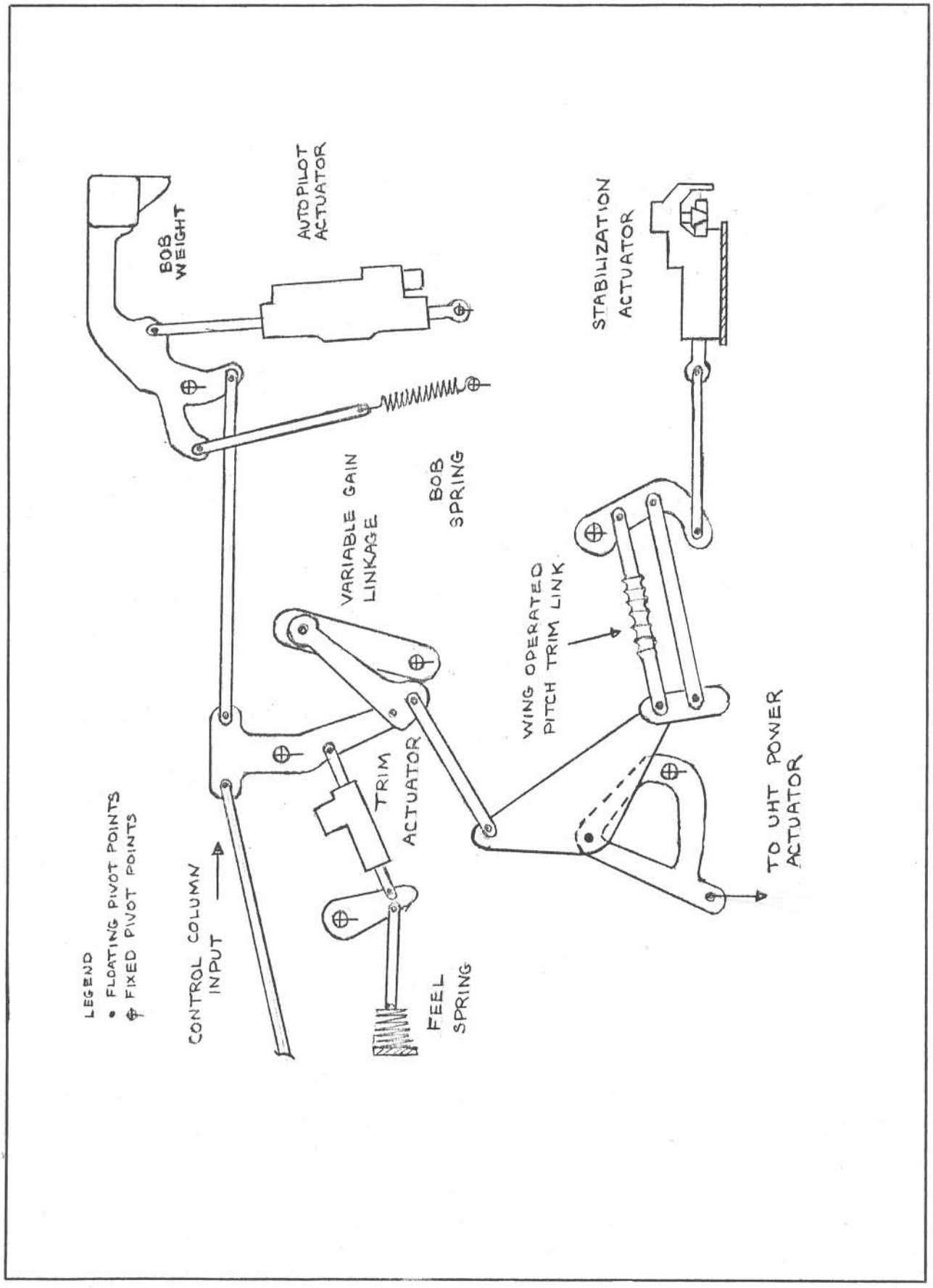


Figure 1-9. Mechanical Pitch Control Linkage

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Stabilization is provided so that fullest capabilities of the airframe can be exploited over the complete range of airspeeds with a minimum of discomfort and restriction (refer to section V for flight restrictions).

Stabilization consists of sensing rate of displacement of the aircraft about its flight axes and then automatically positioning flight surfaces to resist and dissipate the displacing force. The aircraft is returned to stable flight althrough not necessarily to the same attitude held before encountering the displacing force.

The stabilization system in the F8U-3 BuNo. 147085 uses rate gyros aligned with the flight axes to sense rate of displacement. Yaw stabilization gyro output goes through a parameter control transducer which varies the output (gain) to compensate for altitude effects. The output of the roll gyro is controlled by a transducer that is mechanically coupled to the aileron control linkage so as to change gain inversely as the stick is displaced laterally. This gain change system prevents the system from fighting stick commands. Roll stabilization is controlled by wing actuation through the wing incidence down switch and is activated as soon as the wing starts transition up and is deactivated just before the wing snugs down to the fuselage. Figure 1-10 is a simplified block diagram of the flight stabilization system showing basic signal paths. As can be seen from the diagram, the signals originate at the gyros and accelerometers, go through the parameter controls (gain changers) to the stabilization amplifier, then go to the stabilization actuator with another signal to the autopilot computer for the associated axis. The signal to the actuator positions the actuator and the signal to the computer is used to keep the autopilot director gyros aligned with the airplane. The roll and pitch stabilization subsystems are dual channel systems. One channel is a working channel that furnishes a signal to the actuator to position the surfaces. The second channel furnishes a signal to the simulator channel which controls the system monitor. The monitor is used to shut off the associated stabilization function when operation of the working channel does not agree with the simulator channel within the system limits. The yaw stabilization subsystem differs from the roll and pitch subsystems in having only a working channel and limit switches mechanically coupled to the rudder linkage. The yaw stabilization actuator is limited in travel and any malfunction that would cause the actuator to go hard over will trip a limit switch and disengage the system, causing the actuator to lock in neutral. The aileron-rudder interconnect subsystem is separate from the other stabilization systems and has its own amplifier and actuator. The signals for this system are produced by a transducer that is mechanically coupled to the aileron linkage. The amount of rudder deflection produced is proportional to the amount of aileron deflection. The interconnect subsystem, like the roll and pitch stabilization systems, is a dual channel system employing a monitor to shut the system down in case of malfunction or disagreement between the working channel and the simulator channel.

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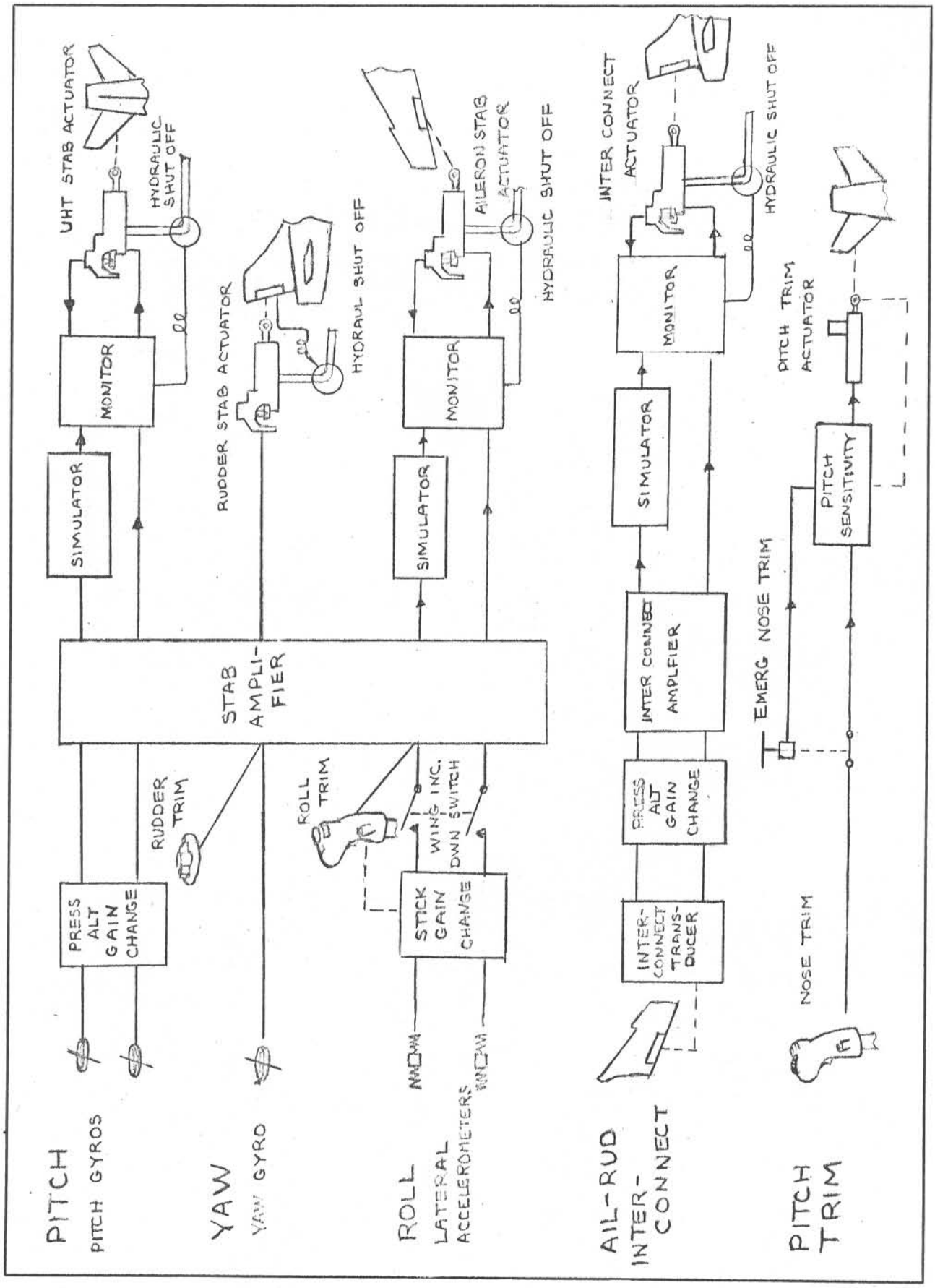


Figure 1-10. Stabilization and Trim Block Diagram

STABILIZATION CONTROLS AND INDICATORS.

Augmented flight stabilization is controled by the YAW, ROLL and PITCH switches (24, 25, and 26, figure 1-2) and the AIL-RUD INTERCONN switch (11, figure 1-2). These switches control electrical and hydraulic power to the stabilization subsystems through interlocking relays. In normal operation, the switches must be placed in "OFF-RESET" before being placed in "ON" to engage the associated stabilization function. After engine starting, it may be necessary to cycle the stabilization control switches several times in order to engage the system.

Note

If a stabilization function disengages in flight, the appropriate stabilization switch should be cycled at least three times to determine if the reason for disengagement is system malfunction or transient error. Repeated disengagement in flight will indicate system unbalance or malfunctioning.

Pitch and interconnect stabilization functions are powered from the PC-2 hydraulic system and the primary a-c and d-c electrical buses. Yaw stabilization control receives hydraulic power from PC-1 and electrical power from the emergency electrical buses. Roll stabilization is powered from the PC-1 hydraulic system and the emergency a-c and d-c electrical buses. There is a red indicator light above each stabilization switch which lights when the switch is in "ON" position but the actuator is locked. The lights are controlled by the locking cams on the actuators and are provided to indicate when a stabilization function is engaged but not operating.

TRIM.

Manual trim control is provided for all three flight axes. The roll trim thumb wheel on the stick grip controls a dual potentiometer in the roll stabilization servo loop and the serrated pitch trim button on the top of the grip is in the pitch trim electrical circuit. The ROLL thumb wheel is indexed to indicate aileron deflection in degrees from neutral. When the thumb wheel is rotated right or left an unbalance signal is introduced in the roll servo loop causing the actuator to reposition the surface without moving the stick or affecting the feel spring.

The pitch trim button is labeled NOSE UP and DN and is spring loaded in the center position. When the botton is pushed forward or aft, selected polarity d-c electrical power is applied to the motor of the actuator, for as long as the button is deflected, causing it to reposition the stick column lonitudual feel spring. The stretching and

compression of the feel spring moves the stick and deflects the surfaces through the normal control linkage. The pitch (NOSE) trim circuit incorporates, in series with the motor, a rheostat that is mechanically coupled to the actuator shaft to decrease the speed of travel of the actuator near trim neutral position. The slowing of the actuator at the beginning of its travel decreases trim sensitivity to permit easier trimming of the airplane in normal flight attitudes. The pitch trim circuit also incorporates provisions for obtaining emergency trim in case of malfunction of the stick grip system. When the EMERG NOSE TRIM "T" handle on the left-hand console is pulled up, the stick switch and associated wiring are removed from the circuit. The spring-loaded "T" handle can then be pushed forward and aft to run the actuator motor and effect the desired trim.

The RUDDER TRIM (yaw) control knob (38, figure 1-2) controls yaw trim by introducing an unbalance in the yaw stabilization servo loop to cause the yaw stabilization actuator to deflect the rudder surface. There are no emergency provisions for aileron or rudder trim and the roll and yaw stabilization systems must be operating before aileron and rudder trim can be operated. Failure of the roll or yaw stabilization systems will de-activate the associated aileron or rudder trim system.

TRIM NEUTRAL INDICATORS.

The aircraft is equipped with a trim neutral light (8, figure 1-3) and a NOSE TRIM indicator (49, figure 1-3) to indicate trim position of the surfaces for field takeoff or catapult launching. The neutral trim light is controlled by magnetic switches on the rudder and aileron linkage. The switches are arranged so that lateral center position of the control column and aileron linkage plus the center position of the rudder pedals and rudder control linkage will energize both halves, to indicate "AIL NEUT" and "RUD NEUT," when the wing is up and weight is on the landing gear. The NOSE TRIM indicator, which is controlled by a potentiometer coupled to the pitch control linkage, shows position of the UHT in degrees from neutral with the wing in the landing configuration.

AUTOMATIC PILOT.

GENERAL. Airplane BuNo. 147085 incorporates a full maneuvering autopilot. This automatic system is specificially designed to relieve the pilot of all normal flight duties and enable him to more easily accomplish an intercept mission using the full capabilities of the airborne missile control system. The prototype system is capable of maintaining climb, cruise, dive or orbit attitudes. In addition, the system provides airspeed, altitude, and heading "hold" functions that can be selected by operation of switches or controls in the cockpit. The system also

provides simple one-handed control over flight maneuvers without the necessity for constant monitoring of surface controls in the cockpit. The autopilot system operates by maintaining tight control of the aircraft around its longitudinal axes. In the automatic pilot mode of flight, all control surface movement is controlled by director gyros whose spin axes are aligned with the flight axes of the airplane. When the aircraft flight path deviates from gyro alignment, an error signal is immediately produced to return the aircraft to the desired flight path. The error signal is generated instaneously with the deviation and is maintained until the aircraft is realigned with the directed flight path. Coupling the control surfaces directly to gyro movement in this fashion results in zero velocity error (the correct amount of effective surface deflection to produce instantaneous realignment at all airspeeds and altitudes). The autopilot includes an input command acceleration ("g") limiting circuit and relay interlock circuits that restrict autopilot commands to safe acceleration limits and switch control functions to permit flexibility. Before the autopilot can be engaged, pitch, roll and yaw stabilization must be engaged and the wing must be in the clean condition (down and locked).

Note

Neither the autopilot nor any autopilot function may be engaged when the airplane is in a bank angle is excess of 5°.

Figure 1-11 is a simplified block signal diagram of the autopilot system and figure 1-12 is an illustration of autopilot function controls and indicators.

The autopilot uses a director gyro aligned with the yaw and pitch flight axes for direct control of the airplane and then provides precessing signals to regulate the gyros to the desired flight path and selected autopilot function. Thus, with flight stabilization engaged, the director gyros follow the aircraft; with automatic pilot engaged, the aircraft is "rigidly" coupled to the director gyros and follows them. Signals to precess the director gyros are originated by the air data computer, MA-1 compass, heading select transformer, and the stick force sensors. The air data computer contains a series of pressure-sensitive motor-driven synchro transformers. The synchros furnish direct outputs that are proportional to Mach number and pressure altitude. The air data computer also furnishes rate of change information (altitude change and mach number change) that is used in the autopilot pitch computer to add rate information to computer output and to furnish rate of change of altitude information to the system interlocks.

Note

The system interlocks will prevent engagement of ALT HOLD or MACH HOLD if the control stick grip is not in longitudinal detent. The interlocks

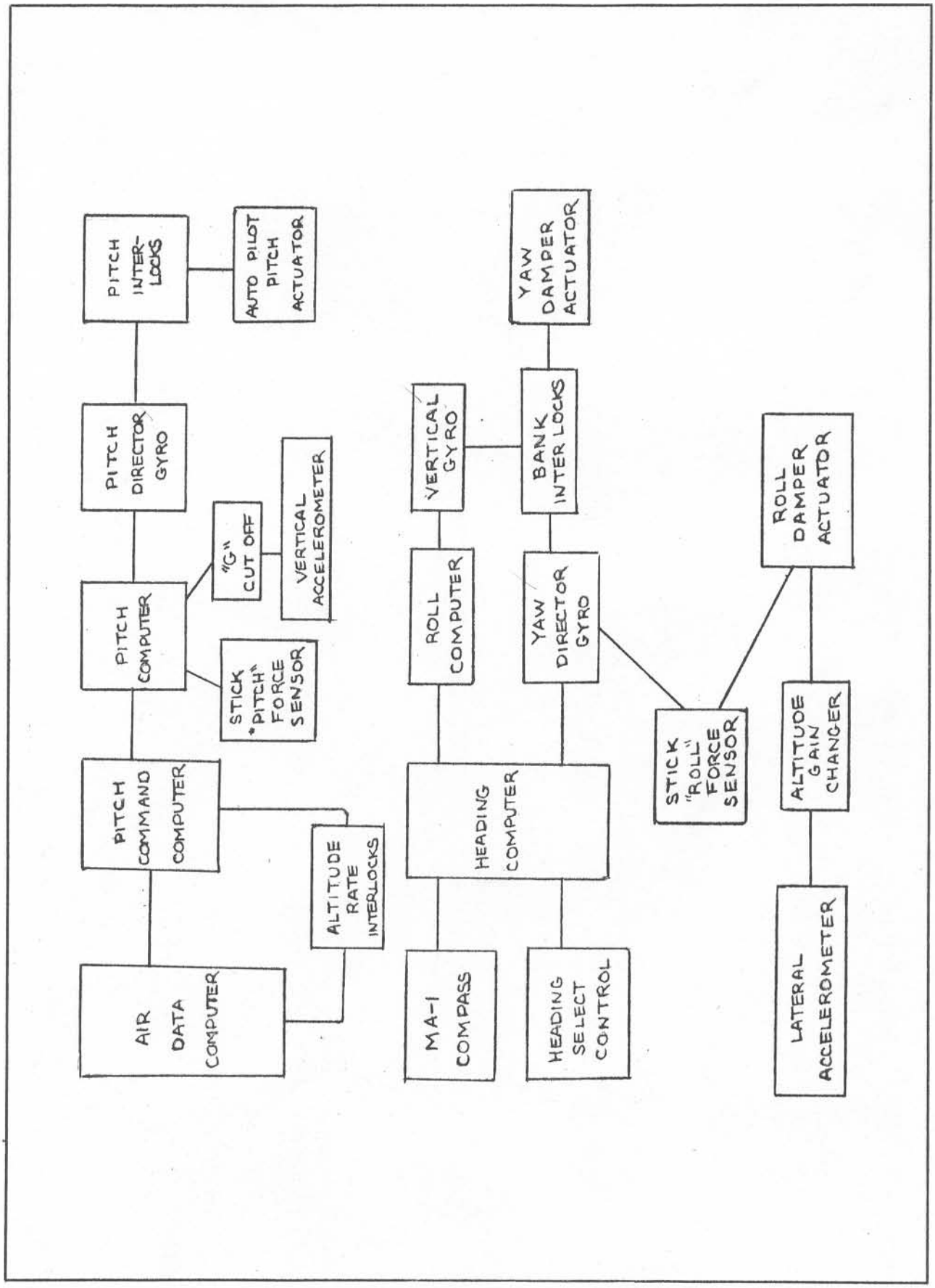


Figure 1-11. Autopilot Simplified Block Diagram

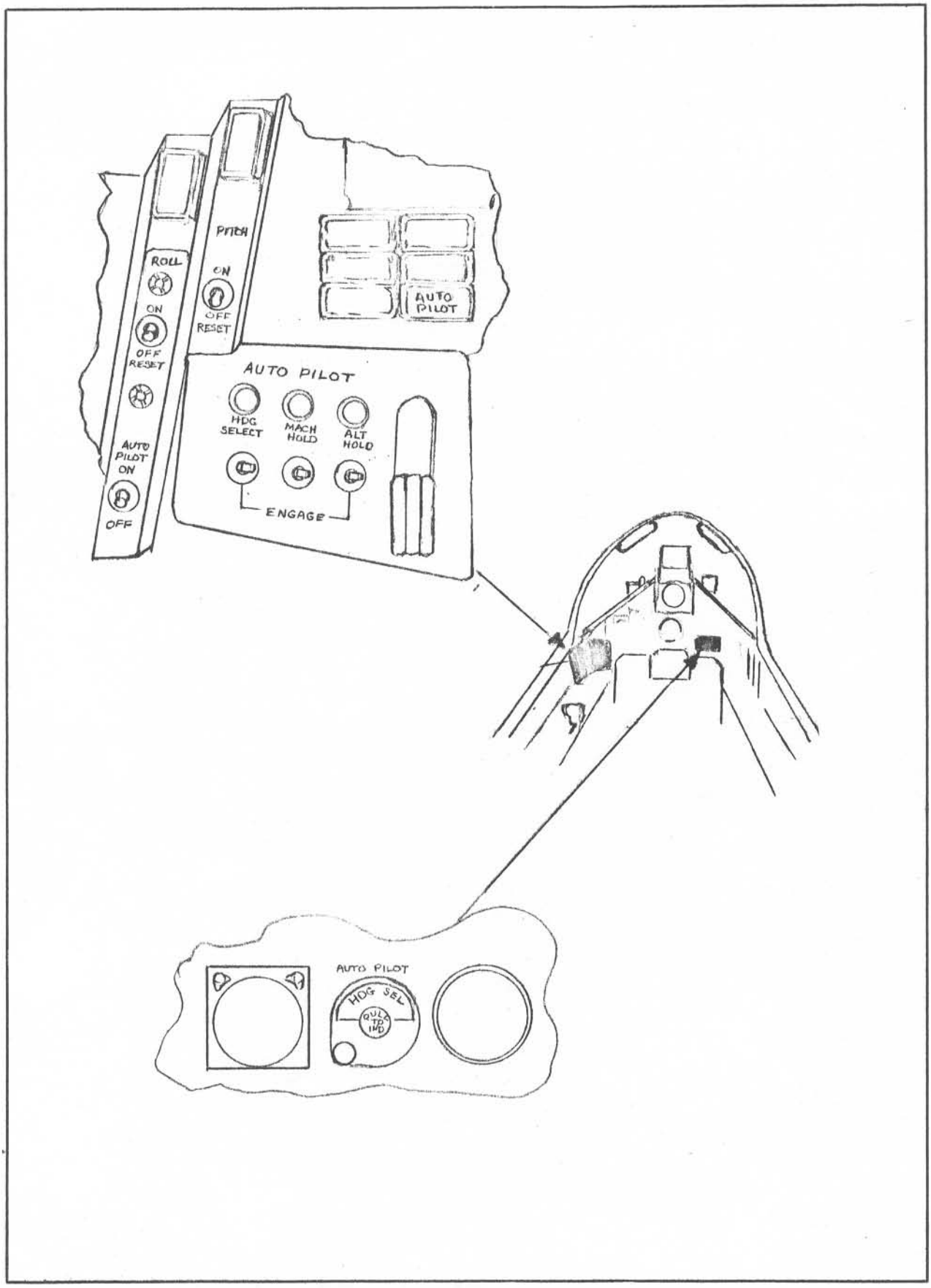


Figure 1-12. Autopilot Controls and Indicators

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will also prevent engagement of ALT HOLD when the rate of altitude change exceeds 2,000 feet per minute.

The selected outputs of the air data computer are altitude hold and mach hold. These autopilot functions are controlled by the ALT HOLD and MACH HOLD switches on the AUTO PILOT function selector panel on the lower left side of the instrument board. In addition to these selected functions, the air data computer also furnishes mach feel. The mach feel output of the air data computer is used as an autopilot pitch reference. When autopilot is engaged a constant mach feel signal is fed to the pitch computer to improve speed stability in supersonic flight. If climb to a new altitude is desired, back pressure on the stick grip may be held until the desired altitude is reached. When altitude is reached, releasing back pressure will cause the aircraft to level off and assume the speed stability attitude held at the time the climb was started. Once the autopilot has been engaged, mach feel reference may be changed by changing pitch trim or by engaging either MACH HOLD or ALT HOLD. Interlock relays prevent both MACH HOLD and ALT HOLD from being engaged simultaneously.

Note

The function switches on the AUTO PILOT control panel will be automatically repositioned to the OFF position when the associated function is disengaged by the system interlock.

MACH HOLD FUNCTION. The MACH HOLD function of the autopilot may be engaged at any mach number consistant with maintenance of a reasonable flight path and attitude, providing the bank angle is less than 5° and pitch angle is less than 50°.

Note

The interlock relays will prevent engagement of MACH HOLD if bank angle is greater than 5° and will disengage MACH HOLD if pitch angle exceeds 50° or if the stick grip is moved out of the longitudinal detent.

The MACH HOLD error signal for maintaining pitch attitude compatible with selected Mach number originates in the air data computer. This error signal is routed through the pitch computer to the pitch director gyro which controls aircraft pitch angle. The pitch computer incorporates a limiting circuit that limits the error output signals to a value that will hold "g" limits to 5g positive and 1g negative acceleration. The computer also incorporates a dual series "g" cutoff system that receives acceleration rate signals from an accelerometer aligned with the aircraft

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pitch axis. This "g" cutoff system disengages the autopilot when the rate signal from the accelerometer indicates a rate of acceleration in excess of 5g or minus 1g. The pitch computer also furnishes a signal to the system interlocks so as to disengage MACH HOLD or ALT HOLD if either function produces a command signal that results in a sustained acceleration in excess of 0.5g positive or negative. When MACH HOLD is selected, a clutch in the air data computer engages the mach synchro transformer at the mach number held by the aircraft at the instant of engagement. Variation in mach number will then result in pitch angle changes of the aircraft to maintain the mach number. Bank angles in excess of 45° will also disengage MACH HOLD.

Note

Sudden variations in mach number may exceed 0.1 mach without producing sufficient vertical acceleration to cause the interlocks to disengage the autopilot or MACH HOLD function. Such variations may cause the air data synchro engagement clutch to slip. Slippage of the clutch will result in a change of the MACH HOLD reference.

ALTITUDE HOLD FUNCTION. ALT HOLD operates in the same manner as MACH HOLD. ALT HOLD may be engaged at less than 5° bank angle and with an altitude rate of change of less than 2,000 feet per minute. When ALT HOLD is selected, a clutch in the air data computer engages the altitude synchro. Any change above or below engage altitude will cause the air data computer to produce an error signal. This error signal is routed to the pitch computer and pitch director gyro to bring the aircraft back to the desired altitude.

Note

Sudden altitude (pressure) changes in excess of 500 feet may not produce vertical accelerations sufficient to cause the system interlocks or "g" cutoff to disengage the system. Such changes may cause the air data synchro engagement clutch to slip. Slippage of the clutch will result in a change of the reference altitude.

The ALT HOLD function of the autopilot is subject to the same acceleration limits as is the MACH HOLD function. Error signals in excess of plus 5g or -1g, or acceleration rates that will produce accelerations is excess of plus 5g or -1g, will disengage the entire autopilot system. ALT HOLD command signals that will produce sustained acceleration in excess of 0.5g will disengage the ALT HOLD function. Bank angles of less than 45° do not affect this function.

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HEADING CONTROL FUNCTIONS. The autopilot system provides an automatic heading "hold" function and a manually selected heading "select" function. The holding function is automatically engaged whenever autopilot is engaged, unless HDG SELECT mode has been selected. In this case, the yaw director gyro is coupled to the MA-1 compass system through the heading computer and the roll computer. As long as airplane bank angle is less than 5°, the output of the MA-1 compass system is used as a reference for maintaining, through the yaw director gyro, the heading that was being steered when the autopilot was engaged. Corrections to maintain heading are made initially by deflecting the rudder. Any skidding that results from rudder movement will be sensed by the lateral accelerometer, which signals the roll system to bank the airplane to achieve coordinated realignment. Any external force that causes a bank angle in excess of 5° or any lateral stick movement will temporarily disconnect the automatic heading reference. When the airplane again assumes wings-level attitude, the automatic reference will reengage with a new reference heading that reflects MA-1 compass heading after the maneuver.

The heading "select" function is engaged through use of the AUTO PILOT HEDG SEL knob and the HDG SELECT switch. For normal operation the HEDG SEL knob should be pulled and rotated until the number 2 pointer on the DRMI indicates the desired course. The HDG SELECT switch should then be engaged. The airplane will then turn to the selected heading. When the HEDG SELECT knob is rotated, an error signal is generated by the heading select transformer. This signal will cause the heading computer to establish a new heading reference and produce an error signal to the roll computer and the yaw director gyro. The rudder will deflect and the roll computer will furnish a signal to the roll actuator to bank the aircraft. The airplane will perform a coordinated turn to the selected heading and will then roll out on course. The rate of any turn commanded by rotation of the HEDG SEL control will not exceed 45° bank angle.

STICK FORCE CONTROL FUNCTIONS. The stick grip in airplanes equipped with automatic flight control is gimbal mounted and is held centered by springs. The grip is mounted to allow slight lateral and longitudinal movement when the centering springs are compressed. Approximately two pounds longitudinal or one pound lateral force is required to overcome the centering springs and move the grip out its center detent position. When the stick grip moves out of detent, it trips one of the detent switches. The switches are located in opposed positions at the base of the stick grip on the longitudinal and lateral movement axes so that movement of the grip on either axis will trip at least one switch. Movement diagonal to the axes will trip two switches. Actuation of switches prepares the autopilot for stick command guidance. When the stick grip is moved longitudinally (forward or aft) a detent switch is tripped to disengage ALT HOLD or MACH HOLD (if either function is engaged) and to connect the output of the stick force sensor to the pitch director gyro.

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The stick force sensor is a transducer, located in the base of the stick grip, that produces a signal to the pitch director gyro proportional to the force applied to the stick grip. The gyro precesses and causes the autopilot pitch actuator to move the UHT. The autopilot actuator moves the pitch control linkage and compresses or stretches the feel spring, causing the control column to follow actuator movement. Thus, the autopilot actuator repositions the control column and at the same time allows feel force to increase, creating the illusion that force applied to the stick grip is moving the control column. In the ALT HOLD or MACH HOLD modes, switches on the feel spring run the pitch trim actuator to reposition the feel spring so as to maintain neutral feel forces as the autopilot actuator moves the control column. If longitudinal stick pressure is released, the autopilot will immediately revert to the automatic mach feel mode. This will cause the aircraft to immediately level off to mach reference attitude and will cause the stick to return to the trimmed position. The transition will be made within the plus 5g and -1g acceleration error limits.

When the stick grip is moved laterally out of detent (center position), one of the two lateral detent switches is tripped, which connects the output of lateral force sensor transducer in the base of the grip to the roll damper actuator and the yaw director gyro. The yaw director gyro provides a signal to the yaw stabilization and autopilot actuator to position the rudder surface and the lateral stick force sensor provides a signal directly to the roll stabilization and damper actuator to position the ailerons. The airplane will start to bank and the bank angle will continue to increase as long as lateral stick force is applied. The rate of roll is limited by the roll rate gyro to 150° per second (roll rate may be increased through the normal control linkage by overpowering the autopilot). If lateral pressure is released, the roll computer will maintain the bank attitude held at the time lateral stick pressure is released. The airplane may then be rolled to any desired bank angle including a complete slow roll. Recovery from a roll is accomplished by applying lateral pressure to the stick grip in the direction desired for roll recovery. In addition to being used to roll the aircraft in the autopilot mode, lateral stick movement may be used for steering. When the aircraft is banked in autopilot using lateral stick force the lateral accelerometer will sense any slipping or skidding. The output of the accelerometer is routed to the yaw director gyro causing it to precess and command the rudder surface deflection required to keep aircraft from skidding and slipping. The result is that the aircraft will perform a coordinated turn at a rate determined by the angle of bank.

Flexibility and maneuverability during autopilot flight using stick control is maintained by system interlocks. As soon as the stick grip is moved out of the longitudinal detent, the MACH HOLD and ALT HOLD functions are disengaged. This prevents function error signals from opposing stick commands. When the stick grip is moved laterally out

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of detent, the system interlocks disengage heading hold and HEDG SELECT to prevent the heading computer from furnishing a signal in opposition to lateral stick commands.

SUMMARY OF AUTOPILOT OPERATION. Autopilot system operation is summarized in table 1-4.

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Table 1-4. AUTOPILOT SYSTEM OPERATION - SUMMARY

Note

Autopilot controls and indicators are illustrated in figure 1-12.

Function	Selection	Description	Operating Limits	Engagement Limits
AUTO PILOT	Manual, with all stabilization channels engaged, by actuation of AUTO PILOT switch; loss of function indicated by lighting of AUTO PILOT light.	Places control surfaces under control of automatic system subject to heading hold and mach feel functions. Arms stick force guidance.	0.25g pitch rate, +5 or -1g's vertical acceleration, 28° per second change in attitude in pitch and yaw. Requires between 50 and 75 pounds manual force for override. Operates in clean condition only.	Less than 5° bank angle.
Mach Feel	Automatic with engagement of autopilot.	Provides speed stability augmentation for hands-off flight when MACH HOLD and ALT HOLD are not engaged.	0.25g pitch rate and +5g or -1g resultant acceleration.	Same as for auto pilot.
MACH HOLD	Manual (autopilot engaged), by holding MACH HOLD switch in "ENGAGE" until MACH HOLD light comes on.	Engages mach reference to maintain pitch attitude required for constant mach number.	0.25g pitch rate, 0.5g command acceleration, 45° bank angle and 50° pitch angle.	Less than 5° bank angle; less than 50° pitch attitude.
ALT HOLD	Manual (autopilot engaged), by holding ALT HOLD switch in "ENGAGE" until ALT HOLD light comes on.	Engages altitude reference to maintain constant altitude.	0.25g pitch rate, 0.5g command acceleration and 45° bank angle.	Less than 5° bank angle; less than 2, 2,000 f/p/m rate of altitude change.

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Table 1-4. AUTOPILOT SYSTEM OPERATION - SUMMARY (cont)

Function	Selection	Description	Operating Limits	Engagement Limits
Heading Hold	Automatic with engagement of autopilot.	Engages heading reference held by MA-1 compass at time A/P is engaged to keep airplane on cardinal heading.	5° bank angle.	Less than 5° bank angle.
HEDG SELECT	Manaul (autopilot engaged), by holding HEDG SELECT switch in "ENGAGE" until HEDG SELECT light comes on.	Engages heading select transformer, controlled by HDG SEL control, to allow airplane to be turned automatically to selected heading.	45° bank angle and 30° pitch angle	Less than 45° bank angle and 30° pitch angle.
Stick Force Control	Manual with engagement of autopilot.	Offers complete airplane control through use of one hand to perform all coordinated maneuvers including coordinated acrobatics within the system acceleration limits.	5g positive acceleration and lg negative acceleration.	Same as for autopilot

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TWO-POSITION WING.

WING POSITION.

Wing position is selected by the WING POSITION handle (7, figure 1-2) to raise the wing for takeoff or landing and lower the wing for high speed flight. The WING POSITION handle release button (9, figure 1-2) must be depressed to move the handle to "UP" or "DN." An actuating cylinder and a snubber cylinder raise or lower the wing, using utility hydraulic system pressure. Hydraulic pressure holds both cylinders in the wing-up position. Hydraulic failure causes the actuating cylinder to lock, and stalls the snubber cylinder. Both cylinders lock internally in the wing-down position. Both cylinders are mechanically locked in the wing-down position by moving the wing DN LOCK handle (6, figure 1-2) to "LOCK." The handle will not go into "LOCK" until both cylinders are fully retracted. The handle must be moved to "UNLOCK" to allow the WING POSITION handle to be moved from "DN" to "UP."

To raise wing if hydraulic power fails, the WING POSITION handle is moved fully forward. Then, with the EMER DROOP & WING INC guard (8, figure 1-2) raised, the handle is placed in "EMERG UP." Emergency pneumatic pressure raises the wing.

The WING POSITION warning light (34, figure 1-3) will come on when wing is up with landing gear up, or wing is down and gear is down. The light will come on if the mechanical wing-down lock is not locked.

WING LEADING EDGE DROOP.

GENERAL. The droop surfaces are full-span, double-hinged, operated by utility system hydraulic pressure. An emergency air system is provided to actuate full droop in case of hydraulic system failure. A three-position indicator on the instrument panel indicates droop position as follows:

"UP"	Droops retracted (clean condition)
"DN"	Cruise (partial) or landing (full) droop extension
Sawtooth (black and white)	Droops improperly positioned in relation to each other, or no electrical power to indicator

CRUISE DROOP. Rotating the throttle grip aft actuates the cruise droop switch to "OUT," partially extending the droop surfaces, to increase

high angle-of-attack efficiency and give higher lift for maneuvering flight. Rotating the grip forward retracts the surfaces. Raising the wing bypasses the cruise drop switch, and fully extends the surfaces. If the wing is lowered, the surfaces retract fully, or if cruise droop is still selected, the surfaces retract to the cruise position.

LANDING DROOP. Placing the WING POSITION handle in "WING UP" fully extends the droop surfaces to provide maximum lift at takeoff or landing. Placing the handle in "WING DOWN" causes the surfaces to retract fully or partially (cruise droop selected). Raising the EMER DROOP & WING INC guard and placing the WING POSITION handle forward extends the surfaces to full droop. Continued handle movement inboard and aft to "EMERG UP" raises the wing.

FLAPS.

Plain flaps with blowing boundary layer control are provided on the wing center section trailing edge, next to the fuselage. Flap position is a function of wing position, controlled by the WING POSITION handle, and operated by utility hydraulic system pressure. With the wing up, the flaps lower 40°. The flaps raise as the wing is lowered. No emergency air is provided for flap operation. There is no cockpit indication of flap position, but the WING POSITION warning light will come on if either flap fails to operate.

WINGFOLD.

The wing outer panels are folded and spread by utility system hydraulic pressure and are mechanically locked in the spread position by hinge pins. The hinge pins are also mechanically locked in place. Red warning flags, on top and on bottom of the wing, extend when the hinge pins are not locked. The outer panels may be folded or spread with the wing in the up or down position, however, the ailerons must be in neutral prior to folding the wings to prevent damage. Pulling the WING FOLD handle (39, figure 1-3) aft unlocks the hinge pins and extends the warning flags. Turning the handle clockwise pulls the hinge pins and folds the outer panels. Reversing the sequence spreads and locks the panels.

VENTRAL FINS.

The fins are pivoted downward from the fuselage just forward of the horizontal tail and are lowered as the wing is lowered in order to increase directional stability. Operation is by utility hydraulic system pressure, and is controlled by the WING POSITION handle (7, figure 1-2). Placing the handle in "DN" lowers the fins and in "UP" raises the fins to a horizontal position. The ventrals extend before

the wing is fully lowered, and raise before full wing raising, to maintain stability while the wing is in transit.

A sequence valve operated by airplane weight on the landing gear permits wing cycling on the ground without repositioning the fins. The VENTRALS position indicator (9, figure 1-3) shows "UP" or "DN" to correspond with ventral fin position. A black and white sawtooth indication appears if circuit continuity is broken, electrical power fails, or if fin positions disagree with each other.

Emergency raising of the wing by air pressure will also raise the ventral fins. Pulling the EMER VENTRALS UP handle (39, figure 1-2) raises only the fins to permit clean condition landings after failure of the utility-hydraulic system or when the wing position systems fail to raise the ventrals.

SPEEDBRAKE.

A conventional speedbrake on the fuselage underside provides control of deceleration in flight. The speedbrake is operated by utility hydraulic system pressure and is controlled by the "SP BK" switch on the throttle grip. Placing the switch in "OUT" extends the speedbrake, and in "IN," retracts it. Center position of the switch stops actuation and holds the speedbrake in position. The switch is inoperative with the wing up and/or when the airplane weight is on the landing gear. Raising the wing in flight will close the speedbrake regardless of switch position. The SPEED BK OPEN light (5, figure 1-3) comes on with speedbrake opening and stays on until full closing. Excessive air loads in high-speed flight will partially close the speedbrake, or allow it to open partially from the closed position. After reducing speed, the switch must be recycled to fully open the speedbrake.

ARRESTING GEAR.

The conventional arresting gear hook is controlled by the arresting gear handle (40, figure 1-3). In the "HOOK UP" position, the hook is held retracted by utility hydraulic system pressure and by a mechanical up-lock. The up-lock will retain the hook if electrical power of hydraulic power fails. Placing the handle in "HOOK DOWN" mechanically releases the up-lock, electrically releases utility pressure, and allows air pressure from a dashpot accumulator to extend the hook and hold it in the extended position. The hook will also extend if hydraulic or electrical power fails. Placing the handle again in "HOOK UP" allows utility hydraulic system pressure to raise the hook and recharge the dashpot accumulator for the next "down" cycle. The mechanical up-lock engages as the hook fully retracts. A warning light in the handle comes on if the hook is down or up but not locked. Hydraulic or electrical failure with the hook down will prevent retraction.

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LANDING GEAR.

GENERAL.

The fuselage mounted tricycle-type landing gear is normally extended and retracted by utility hydraulic system pressure, with a pneumatic air system provided for emergencies. The nose gear is mounted on the right of the airplane centerline to provide armament clearance.

The landing gear is extended or retracted by the landing gear handle (2, figure 1-3). The wheel-shaped handle is spring loaded into the "WHEELS UP" and the "WHEELS DOWN" detents and contains an internal warning light. A down-lock solenoid prevents placing the handle in the "WHEELS UP" position when weight is on the main landing gear. Placing the landing gear handle in "WHEELS UP," retracts the landing gear and closes the wheel well doors. The nose gear doors are mechanically linked to the nose gear and are opened or closed with the actuation of the nose gear. Two of the main gear doors are hydraulically actuated and the third is mechanically linked to the strut. When the gear is fully retracted, the doors are mechanically locked in the up position and the landing gear indicators (10, figure 1-3) will indicate "UP." A "sawtooth" indication will appear on the indicators when the gear is in an intermediate position.

The gear is extended by placing the landing gear handle in "WHEELS DOWN" to unlock the gear, hydraulically open the main gear doors and extend the gear. When the gear is fully extended, the nose gear is mechanically locked and the main gear is locked by internal locks in the cylinder. The landing gear indicators will indicate "DN" when the gear is locked down.

In the event the gear cannot be extended hydraulically, pneumatic pressure may be used by pushing the landing gear handle in, rotating it and pulling it aft.

An EMER DN LOCK release button (54, figure 1-3) is provided to permit retracting the gear on the ground in case of an emergency.

The WHEELS warning light (21, figure 1-3) will flash on and off if the throttle is retarded when the wing is up and the landing gear is not down and locked.

WHEEL BRAKES.

The power-boosted wheel brakes are normally actuated by utility hydraulic system pressure. The force applied at the rudder pedal tips governs the amount of braking action. Manual (no boost) operation of the brakes is possible when utility hydraulic pressure

is not available. A pneumatic braking system is provided for emergency operation by the EMERG BRAKE handle (10, figure 1-2). The amount of braking action is proportional to the movement of the handle towards "ON." Differential braking is not possible when using the emergency system.

NOSE GEAR STEERING.

The nose gear steering is operated by utility hydraulic system pressure, with electrical power supplied by the secondary ac and dc buses. Depressing the nose gear steering switch on the control stick and pushing the appropriate rudder pedal will rotate the nose gear in the desired direction. Nose gear steering is possible only when weight is on the landing gear and is limited to 60° right or left of the trail position. A mechanical centering mechanism centers the gear as retraction starts.

INSTRUMENTS.

GENERAL.

The airplane is equipped with conventional powerplant, flight, and navigation instruments. The instruments associated with specific systems of the airplane are described in the system descriptions, and instruments installed for flight test pruposes are covered in general under FLIGHT TEST INSTRUMENTATION.

The airplane has two static pressure systems serving the pilot's instruments. At low speeds, static pressure is taken from ports on the side of the fuselage. At approximately MN=1.05 to 1.10, a signal from the air data computer actuates a static transfer valve to switch the static pressure source to ports in the nose boom. The static pressure systems serve the airspeed indicator, altimeter, and rate-of-climb indicator.

AIRSPEED INDICATOR (17, figure 1-3).

The conventional combination airspeed and Mach Number indicator indicates airspeeds from 80 to 850 knots and Mach Number from 0.4 to 2.5.

ALTIMETER (16, figure 1-3).

The standard single-pointer type altimeter indicates pressure altitudes from sea level to 80,000 feet. The pointer indicates over a scale in increments of 100 feet up to 1,000 feet. Altitude in increments of 1,000 feet is indicated in a window on the instrument.

RADAR ALTIMETER (Radio Height Indicator-12, figure 1-3).

This indicator shows terrain clearance of the aircraft, providing reliable information up to 10,000 feet over land and to 20,000 feet over water.

Accuracy is ±2 feet to 40 feet, and ±5% of indicated altitude to 20,000 feet. The movable limit index may be set to a desired clearance altitude. If the airplane drops below this altitude, the limit light on the indicator will come on (within tolerance limits of ±3% of the pre-set altitude).

TURN & BANK INDICATOR (50, figure 1-3).

This instrument is a miniature air-driven type that obtains operating pressure from a regulator that uses engine compressor bleed air.

ACCELERATION INDICATOR (26, figure 1-3).

Airplane BuNo. 147085 is equipped with a 3-1/2 inch "lock displaced" acceleration indicator with a reset knob on the lower left corner.

ATTITUDE INDICATOR (46, figure 1-3).

The airplane is equipped with a Lear Model 4005 (U.S.A.F. type MM-4) 5-inch remote attitude indicator (figure 1-4). The indicator employs a fixed airplane-moving horizon presentation and is integrally lighted. Continuous attitude indication is provided through 360° of roll but indication is limited to 85° climb or dive. A pitch adjustment knob on the lower right-hand corner provides for adjusting the horizon to zero-climb attitude. The instrument and system operate from the emergency ac bus. An "OFF" flag appears when electrical power is lost. The system requires approximately one minute for the gyros to reach speed and the "OFF" flag to disappear.

WARNING

Disappearance of the "OFF" flag does not necessarily indicate that instrument is operating properly. The indicator should be periodically checked against other flight instruments to insure reliability.

ANGLE-OF-ATTACK INDICATING SYSTEM.

The F8U-3 is equipped with an angle of attack indicating system consisting of a vane operated transducer on the right side of fuselage near the cockpit, a bridge amplifier, an indicator, an approach indexer, and the approach lights. The system provides the pilot and LSO with visual indications of airplane angle of attack for convenience in controlling airspeed during landing approaches.

Signals received from the vane operated transducer are presented on the ANGLE OF ATTACK indicator (15, figure 1-3) on a scale reading from 0 to 30. Each unit on the scale is equal to approximately 1.5° angle of attack and 5 knots airspeed. The indicator controls the operation of the angle-of-attack approach indexer (22, figure 1-3) and the approach lights. The indexer and approach lights are operated with reference to the index marker at the 3 o'clock position of the indicator. The approach indexer and approach lights function only when the landing gear handle is in the "WHEELS DOWN" position. If the pointer on the indicator is on the center bar of the reference marker, the center "O" light on the indexer and the amber approach light will be on. If the pointer is above the center bar on the reference marker but still within the limits of the reference marker, both the center "O" light and the top "V" light on the indexer will be on. The green approach light will be on. If the pointer on the indicator is completely above the reference mark, only the upper "V" on the indexer and the green approach light will be on. The same lighting sequence occurs for pointer positions below the center bar of the reference marker except that the lower inverted "V" of the indexer and the red approach light will be lighted.

The reference marker must be set to accomodate various configurations and airplane gross landing weights. Information for setting will be furnished at a later date.

The system is flown by coordinating throttle and stick movement to establish the desired approach path at optimum angle of attack. The stick is used to establish the angle of attack and the throttle is used to obtain the desired rate of descent.

If the indexer light or lights fail, the indicator may still be used to fly the approach. Operation of the system should be checked by porpoising the aircraft slightly during initial approach and checking instrument and light response to the maneuver.

The angle of attack indicator can be used under all flight conditions to check aircraft angle of attack for stalls, maximum endurance flight or other desired operations.

FLIGHT TEST INSTRUMENTATION.

Various switches, lights and control panels provide special controls and indicators for normal aircraft systems and flight test instrumentation. Because of the number of changes during the flight test program, switches and controls may be marked inoperative or safety wired in the desired operating position.

Operation of strictly flight test controls will be covered by briefing priot to each flight.

The BEACON switch (14, figure 1-2) turns on the radar tracking beacon to allow more effective tracking by ground control stations during flight testing.

The C.G. variation switch (16, figure 1-2) allows the pilot to shut off either the forward or the aft transfer fuel booster pump for flight testing of C.G. travel. The neutral position of the switch allows normal transfer fuel sequencing. (Refer to FUEL MANAGEMENT.) A warning light (17, figure 1-2) above the switch is illuminated if either the forward or the aft cell cluster is pumped dry.

A large acceleration indicator (28, figure 1-2) gives the needed accuracy of measurement of "g" forces during flight testing.

The MANEUVER switch (21, figure 1-2), REC light (22, figure 1-2), CAL light (23, figure 1-2), Recording Control Panels (42, figure 1-2), and the Recording Light and Indicator (27, figure 1-3) permit control of flight test instrumentation.

EJECTION SEAT.

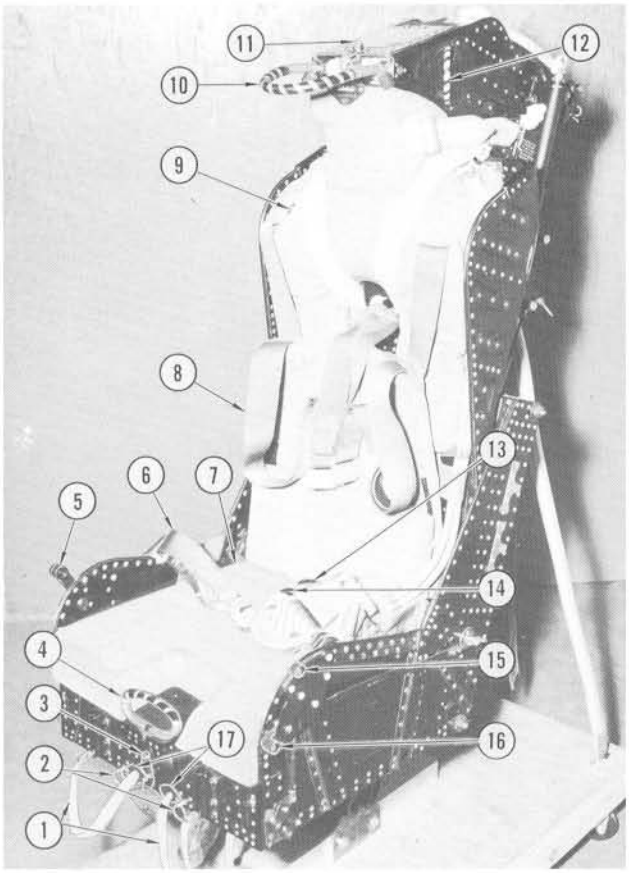
GENERAL.

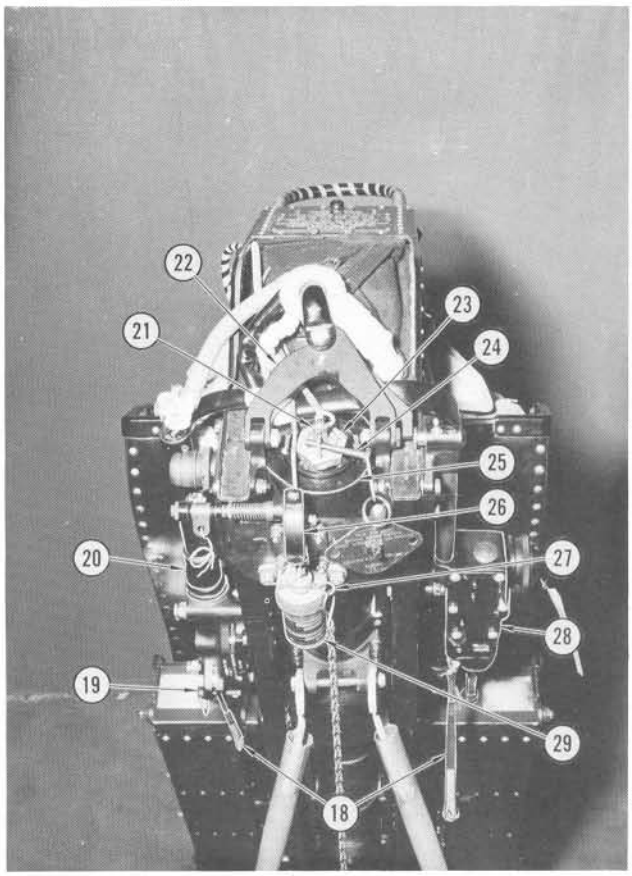
A Martin-Baker ejection seat (figure 1-13) is installed in airplane BuNo. 147085. While this seat is commonly referred to as providing runway- or ground-level escape, it does not have a graound-level capability as presently installed in the F8U-3. The seat will provide escape at very low altitudes, as determined by functions of flight path and speed, and throughout the flight envelope up to approximately 600 knots. Ejection at very low altitudes is practical as a result of the high ejection velocity attained through use of a long-stroke catapult gun and because of the capability of the parachute for rapid deployment. At high altitudes, separation of the pilot from the seat and deployment of the parachute are automatically delayed to permit rapid descent to an altitude at which the use of oxygen is not required. The seat and pilot are stabilized during the descent by two drogues that are deployed immediately after ejection. At very high ejection speeds, separation of pilot from the seat and deployment of the parachute is automatically delayed until a safe speed for deployment is reached. The ejection trajectory made possible by use of a telescopic catapult gun provides adequate clearance over the vertical tail at the highest probable ejection speeds.

The seat is fitted with a face curtain handle as the primary means of firing, although it may also be fired by means of an alternate firing handle located on the leading edge of the seat pan.

EJECTION SEQUENCE.

The canopy initiator is fired to jettison the canopy when the face-curtain handle is pulled through the first part of its travel. Separation of the canopy from the airplane releases the canopy interlock mechanism to allow





NOTE

The seat requires the use of PK-2 pararaft, modified FIRE-F3262000 seat cushion assembly, and either full pressure suit or coveralls with integrated harness vest.

1. Leg Restraint Cords
2. Snubbing Unit Release Rings
3. Alternate Firing Handle Safety Pin
4. Alternate Firing Handle
5. Harness Emergency Release Lever
6. Lap Belt
7. Seat Cushion Assembly
8. Shoulder Harness
9. Personnel Parachute
10. Face Curtain Handle
11. Face Curtain Safety Pin
12. Seat Emergency Arming Handle
13. Parachute Forward "D" Ring
14. Parachute Aft "D" Ring
15. Shoulder Harness Lock Lever
16. Leg Restraint Release Lever
17. Leg Line Lock
18. Trip Rods
19. Drogue Gun Safety Pin
20. Drogue Gun
21. Catapult Firing Cam
22. Scissor Shackle
23. Catapult Firing Head
24. Catapult Firing Cam Safety Pin
25. Catapult Gun
26. Canopy Interlock Mechanism
27. Canopy Initiator Safety Pin
28. Time Delay Mechanism
29. Canopy Initiator

Figure 1-13 Ejection Seat

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the face-curtain to be pulled through to the full extent of its travel. This action withdraws the firing cam from the firing head of the catapult gun and ejects the seat and pilot from the airplane.

If the canopy fails to be jettisoned, the face curtain or alternate firing handle will be stopped after approximately eight inches of travel. The seat emergency arming handle on the left side of the face curtain housing permits the canopy interlock mechanism to be bypassed so that the face-curtain handle can be pulled to the full extent of its travel for ejection of the seat and pilot through the canopy (refer to "EJECTION PROCEDURE" in section III for procedure and precautions for ejection through the canopy).

As the seat begins to move upward, the leg-restraint cords tighten up between the snubbing unit and the leg-line lock on the seat pan, automatically securing the pilots legs to the front face of the seat pan, where they are firmly held until the harness is automatically released and the pilot separates from the seat. Also, as the seat moves upward, two trip-rods attached to brackets on the seat bulkhead withdraw firing cams to fire the drogue gun and initiate action of the altitude-time delay mechanism.

As the seat moves upward, the pressure suit vent line, g-suit line (if used), and oxygen hose are disconnected by action of the pilot's services disconnect mechanism on the left side of the seat. At the same time, the emergency oxygen lanyard, which is attached to the airplane, is pulled to activate the bailout oxygen system housed in the seat cushion assembly.

One-half second after the seat and pilot are ejected, the drogue gun, located on the seat guide rail, fires a piston which extracts the controller drogue from its housing. This drogue, in turn, extracts the main drogue which draws with it a nylon cord attached to a scissor shackle on top of the seat. When deployed, the drogues stabilize the seat and aid in deployment of the personnel parachute.

After a time interval determined by either the "g" limiter or the barostat, the time delay mechanism removes the restraint imposed on the scissor shackle by a locking plunger, thereby allowing the shackle to open and free the drogues from the seat. At the same time, the action of the mechanism frees the leg lines and parachute from the seat. On release from the seat, the drogues pull on a withdrawal line which disconnects the face curtain and deploys the personnel parachute. Two straps clipped to the seat pan prevent the pilot from leaving the seat until the parachute lifts him clear. If the ejection occurs at an altitude above 10,000 feet, the barostat interrupts operation of the time-delay mechanism until the pilot and seat are below this altitude. In ejections at very high speeds, the "g" limits interrupts operation of the time-delay mechanism until the pilot and seat have slowed down to a safe speed for parachute deployment.

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SHOULDER HARNESS LOCK LEVER. The shoulder harness lock lever, which is located on the left side of the seat (figure 1-13), controls the shoulder harness snubber reel. When the lever is pulled aft, the harness reel is unlocked, permitting the pilot to lean forward. When the lever is released to return forward, the harness reel is locked against any forward movement of the pilot. If the pilot is leaning forward when he locks the harness, the reel will take up slack as the pilot leans back.

LEG RESTRAINT RELEASE LEVER. Pulling upward on the leg restraint release lever, which is located on the left side of the seat (figure 1-13), releases the leg restraint cords from the seat, thereby releasing the pilot's legs from the seat pan.

SNUBBING UNIT RELEASE RINGS. The leg restraint cord snubbing unit release rings, which are located on the front face of the seat pan (figure 1-13), provide a means of releasing the snubber unit to allow the pilot to adjust the length of each leg restraining cord for comfort when his feet are on the rudder pedals.

HARNESS EMERGENCY RELEASE LEVER. The harness emergency release lever, which is located on the right side of the seat (figure 1-13), permits the pilot to release the shoulder restraint, leg restraint and lap belt harness straps from the seat in cases where rapid exit from the airplane, as in ditching, is desired. Releasing the harness from the seat permits the pilot to take with him the pararaft kit, personnel parachute, and seat cushion. To obtain complete separation from the seat, the forward parachute "D" ring must also be pulled to release the parachute from the drogue chute system withdrawal line.

PARACHUTE "D" RINGS. The forward and the aft "D" rings are located on the left side of the lap belt (figure 1-13). The forward "D" ring, when pulled out to the full extent of its travel, disconnects the personnel parachute from the seat, and thus from the automatic mechanism, and also uncovers the aft "D" ring. The aft "D" ring, when pulled, deploys the personnel parachute.

SEAT ADJUSTMENT SWITCH.

The SEAT adjust switch (20, figure 1-4) controls electrical power to the seat actuator to raise or lower the seat within limits established by upper and lower limit switches.

CANOPY.

The canopy is a one-piece clamshell-type pivoted at the aft end. The edge of the canopy incorporates a pressure seal diaphragm, and the lip of the cockpit and windshield has a vinyl-tape-covered striker which depresses the diaphragm to make the pressure seal.

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CAUTION

Care should be taken when entering or leaving the cockpit to prevent gear or feet from damaging the striker.

The canopy is counterbalanced to allow it to be raised or lowered easily. It will remain open in tailwinds up to 15 knots and is stressed to withstand headwinds of 60 knots. Headwinds in excess of 60 knots may cause the pivot bolts to shear and allow the canopy to be blown off or fall back over the cockpit. The canopy employs four rotating hook latches, two on each side of the canopy rail, to seal and lock it in the closed position. The interior canopy release handle on the forward right hand cockpit rail is pushed forward to release the canopy locks and moved aft to lock the canopy. An indicator on the left side of the canopy shows position of canopy locks and should be checked each time the canopy is closed.

WARNING

Failure to fully seat canopy locks may result in loss of canopy at altitude with resulting explosive decompression.

Normal external release of the canopy locks is accomplished by the CANOPY RELEASE handle, which is located on the left side of the airplane flush with the skin just below the canopy rail. The handle is operated by pushing inward on point labeled "PUSH" and then pulling the handle out and forward.

The canopy is equipped with a cartridge-operated latch thruster and a cartridge-operated actuator. These actuators are triggered automatically as part of the ejection procedure, and manually by pulling the EMER CANOPY JETTISON handle (14, figure 1-3). The thruster opens the latches and bypasses gas to cause firing of the canopy actuator which forces the canopy upward into the slipstream so that air loads can aid in shearing the pivot bolts and allow the canopy to be carried away. An EMERGENCY CANOPY JETTISON handle is located on each side of fuselage outside of the canopy for use in pilot rescue operations. Ground safety pins on the canopy initiators prevent inadvertent firing of the ejection system during ground operations.

WARNING

Stand clear of canopy when pulling the external EMERGENCY CANOPY JETTISON handle. The canopy will open violently upward, then fall back to one side of the fuselage. Firing the canopy on the ground will do extensive damage to the fuselage area above canopy pivot point.

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SECTION II

NORMAL PROCEDURES

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BEFORE ENTERING AIRPLANE.

Refer to sections V and VI for flight restrictions and a description of flight characteristics; refer to Appendix I for operating data.

Refer to "WEIGHT LIMITATIONS," section V, and the Handbook of Weight and Balance (AN 01-1B-40). Check Form F (Weight and Balance Clearance) of latter handbook for proper loading.

Consult thrust check chart in CVA Specification 654-9 to determine the correct engine PRESSURE RATIO indicator reading for performing the "MILITARY THRUST CHECK."

Check yellow sheet for status.

EXTERIOR VISUAL INSPECTION.

Proceed as follows:

NOSE SECTION

External electrical power unit and compressor starter unit	Standing by
Access doors and panels	Secured
Nose boom	Cover removed; static ports clear
Nose cone	Secured
Intake duct	Clear

NOSE WHEEL WELL

Check general condition; no hydraulic leaks	
Air bottle gages	Proper pressure
Nose gear	Check strut extension and tire inflation
Nose gear ground-handling down lock	Removed

RIGHT FORWARD FUSELAGE

Static ports	Clear
Fuselage lights	Undamaged
Underside of fuselage	Check for fluid leaks
Access doors and panels	Secured
Speed brake	Check condition; no hydraulic leaks
Missiles	Check for security and proper attachment

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RIGHT MAIN WHEEL WELL

Check general condition; no hydraulic leaks	
Main gear	Check strut extension and tire inflation
PC hydraulic accumulator (tension strut)	Check for leaks
Main gear ground-handling down lock	Removed

RIGHT WING

Check general condition; no hydraulic or fuel leaks	
Access doors and panels	Secured
Leading edge and droop hinge area	Check condition; no hydraulic leaks
Position light	Undamaged
Auxiliary approach light	Undamaged
Aileron and wing spoiler	Check condition; no hydraulic leaks
Flap	Check condition and security

MID AND AFT FUSELAGE AND EMPENNAGE

Access doors and panels	Secured
Fuel cell cavity vent ports	Clear
Underside of fuselage	Check for fluid leaks
Horizontal tail	Check condition and security
Vertical tail	Check condition
Position light	Undamaged
Tailpipe cover	Removed
Afterburner exhaust nozzle flaps	Check condition

LEFT WING

Repeat checks for right wing

LEFT MAIN WHEEL WELL

Repeat checks for right main wheel well	
FUEL SELECTOR switch	"OFF"
PRESSURE FUEL CAP	Secured
JET STARTING receptacle	Lead from starter cart connected
Oil filling and overflow ports	Capped

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LEFT FORWARD FUSELAGE

Access doors and panels	Secured
Static ports	Clear
Missiles	Check for security and proper attachment

ENTRANCE.

Release lower, middle and top steps and handgrip door. Release bottom step and pull to extended position. Grasp handgrip and ascend steps, starting with left foot. The canopy is opened manually by depressing the forward part of the CANOPY RELEASE handle, marked "PUSH," to release the handle arm. Grasp the handle arm and pull forward to unlock the canopy.

COCKPIT CHECKS (ALL FLIGHTS).

Ejection system safety pins	Six pins removed
Canopy interlock mechanism pin	Installed
Face curtain handle	Stowed
Seat alternate firing handle	Stowed
Seat emergency arming handle	Stowed
Bailout bottle lanyard	Secured to airframe
Pilot's services quick disconnect lanyard	Secured to airframe
Harness and parachute	Properly secured to seat
Seat	Adjusted to comfortable height
Leg restraint release lever	Up
Leg restraint straps	Secured to seat
Antiblackout and pressure suit lines	Connected (as required)
Rudder pedals	Adjusted
External electrical power	Connected
MASTER GEN switch	"EXT & RESET"
Position indicators	"Sawtooth" not showing
Attitude indicator	"OFF" not showing
Seat	Adjusted
G-VALVE knob	As desired
VENT AIR knob	As desired
Pressure suit TEMP knob	As desired
EMERG POWER handle	Stowed
EMERG DROOP & WING INC guard	Down
WING POSITION handle	Match wing position
RUDDER TRIM knob	Neutral
Throttle	"OFF"
Throttle friction wheel	Adjusted
CRUISE DROOP switch	Match droop position
SP BK switch	"IN"

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EMERG BRAKE handle	"OFF"
FUEL CONTROL switch	"NORMAL"
YAW switch	"OFF-RESET"
ROLL switch	"OFF-RESET"
PITCH switch	"OFF-RESET"
ENGINE MASTER switch	"OFF"
EMER PITCH TRIM handle	Down
DUCT BYPASS CONT switch	"OFF-RESET"

Check and set all flight, flight test, and navigation instruments. Press all indicating lights to test operation. Press "TEST" switch to check fire detection circuit continuity. ("FIRE" warning light will come on).

Check FUEL quantity indicator operation by turning TANK QTY SELECTOR switch to "TEST." Turn TANK QTY SELECTOR switch to "MAIN," "FWD," "AFT," and "WING" and check TANK pointer on FUEL quantity indicator for proper quantity. Check TOTAL reading for proper quantity.

LOW HYD PRESSURE warning light	Off
ENGINE DEICER switch	"OFF"
PITOT HEAT switch	"OFF"
FUEL TRANS switch	"AUTO"
EMERG GEN switch	"OFF"
Interior and exterior lights controls	As desired
Arresting gear handle	"HOOK UP"
Emergency ram air handle	"CLOSED"
Air-conditioning AUTO TEMP CONTROL	As desired
Air-conditioning MANUAL TEMP CONTROL	"OFF"
DEFOG switch	"OFF"
RAIN REMOVE switch	"OFF"
Oxygen system	Gage showing 3-1/2 liters minimum; warning light out with hoses connected and system on.
CABIN PRESSURE switch	"ON"
Air-conditioning EMERGENCY OFF switch	Guard down
Radar POWER switch	"OFF"
AMCS SYSTEM POWER switch	"OFF"
EXPLOSIVE BOLT JETTISON switches	Guards down
JETTISON switches (armament panel)	"OFF"

BEFORE STARTING ENGINE.

Check operation of communication equipment (section IV) and return switches to "OFF" positions.

CAUTION

Do not operate electronic equipment longer than 10 minutes without external cooling air.

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Check that all personnel and equipment are clear of air intake duct and tailpipe, and that fire guard is posted with fire-fighting equipment. (Refer to "FIRE," section III, for procedure to be used if fire occurs during starting.)

External electrical power	Connected
MASTER GEN switch	"EXT & RESET"
Attitude indicator (ac power indication)	"OFF" not showing
Position indicators (dc power indication)	"Sawtooth" not showing
Throttle	"OFF"
FUEL CONTROL switch	"NORMAL"
DUCT BYPASS CONT switch	"ON"

STARTING ENGINE.

ENGINE MASTER switch. . . . . "ON"

Signal ground crew to connect starting air source. At signal from ground crew, proceed as follows:

Throttle. . . . . "CRANK" momentarily
At 10% rpm, "IGNITE"
At 12% rpm, "IDLE"

Note

Ignition should take place within 20 seconds, as indicated by rapid rise in rpm and exhaust temperature. Allow engine to accelerate to idle range rpm (57% to 65%).

EXH TEMP indicator. . . . . 630° C maximum.

Increase engine speed to 75% rpm to actuate CSD control, then return throttle to "IDLE." Place MASTER GENERATOR switch in "MAIN" and check attitude indicator and position indicators for indications of ac and dc power. If there is no indication of power, shut down engine.

Check that DUCT BYP CONT OFF light is off. If light is on, cycle switch to "OFF-RESET" and back to "ON." If light remains on, shut down engine.

EXH TEMP indicator	340°C, maximum (after engine stabilizes at "IDLE)
OIL PRESS indicator	35 psi minimum
LOW HYD PRESSURE warning light	Off

After engine stabilizes at idle rpm, signal ground crew to remove starting air source and external electrical power.

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CAUTION

If throttle is inadvertently retarded to "OFF," do not advance throttle to regain light because a hot start may result. Allow a 30-second fuel drainage period and repeat "STARTING ENGINE" procedure.

UNSATISFACTORY ENGINE STARTS.

HOT START.

If exhaust temperature exceeds 630°C during starting, proceed as follows:

Throttle	"OFF"
ENGINE MASTER switch	"OFF"
MASTER GEN switch	"OFF"

Investigate for damage and cause of difficulty.
Perform "PURGING ENGINE" procedure before attempting another start.

FAILURE TO START.

If engine does not start within 20 seconds after throttle is placed in "IDLE," or if after ignition, engine does not stabilize at idle conditions; proceed as in "HOT START" above.

PURGING ENGINE.

To clear engine of trapped fuel or vapors, proceed as follows:

External electrical power	Connected
MASTER GEN switch	"EXT & RESET"
Attitude indicator	"OFF" not showing
Position indicators	"Sawtooth" not showing
Throttle	"OFF"
FUEL CONTROL switch	"NORMAL"
ENGINE MASTER switch	"ON"
External air source	Connected
Throttle	"CRANK" momentarily
MASTER GEN switch	"OFF" after 15 to 20 seconds
ENGINE MASTER switch	"OFF"

Allow a 30-second fuel drainage period before starting engine.

ENGINE GROUND OPERATION.

Normally, no engine warmup period is required. After engine has stabilized at idle conditions, throttle may be advanced to full power.

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GROUND TESTS.

Position indicators	"Sawtooth" not showing
Attitude indicator	"OFF" not showing
Hydraulic pressure indicator	3,000 (±200) psi, all systems
Flight controls	Neutral
Wings	Spread and locked (red warning flags not showing)
YAW STAB OFF light	On
YAW switch	"OFF-RESET" momentarily, then "ON"
YAW STAB OFF light	Off
ROLL STAB OFF light	On
ROLL switch	"OFF-RESET" momentarily, then "ON"
ROLL STAB OFF light	Off
PITCH STAB OFF light	On
PITCH switch	"OFF-RESET" momentarily, then "ON"
PITCH STAB OFF light	Off

Raise wing and check droop operation of leading edge and ailerons, lowering of flaps, and trim change of horizontal tail.

Rotate all trim controls through full range and check for proper control surface response. Place RUDDER TRIM and ROLL trim knobs to full left or right, then place MASTER GEN switch in "OFF" and check that rudder and ailerons return to neutral. Place MASTER GEN switch in "MAIN," return RUDDER TRIM and ROLL knobs to neutral, and reset yaw and roll systems.

Pull EMER PITCH TRIM handle, move to NOSE DN and NOSE UP positions, and check for proper horizontal tail response. Return handle to stowed position. Re-engage pitch system.

Move flight controls and check for proper response.

Lower wing and check operation of leading-edge, ailerons, spoilers, flaps, and horizontal tail.

Rotate all trim controls through full range and check for proper control response, then return knobs to neutral.

Move flight controls and check for proper response. Check cruise droop operation and leave CRUISE DROOP switch in "OUT." Turn on communcation and navigation equipment as desired.

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CAUTION

Do not operate electronic equipment longer than 10 minutes without external cooling air.

Place arresting gear handle in "HOOK DOWN" and check hook operation. Return handle to "HOOK UP."

Raise wing and check:

AIL NEUT light	On
RUD NEUT light	On
WING POSITION light	Off
LE DROOP indicator	"DN"
VENTRALS position indicator	"UP"

VISCOUS DAMPER CHECK.

To test operation of the viscous damper, push stick forward rapidly and check for slow, smooth return of stick to neutral. If stick snaps back toward neutral, the airplane should not be flown until the trouble is corrected.

WARNING

Incorrectly serviced or malfunctioning viscous damper can allow inadvertent pilot-induced violent or uncontrollable pitch oscillations in flight.

TAXIING.

Signal ground crew to remove chocks and close cabin entry steps. To obtain full nose gear steering range, taxi with the wing up. Advance throttle if necessary after releasing brakes. As airplane moves forward, retard throttle to "IDLE" and use nose gear steering for turn.

Check operation of instruments.

Note

Maximum wind velocity for safe operation of the canopy is 60 knots.

BEFORE TAKEOFF.

MILITARY THRUST CHECK.

Set PRESSURE RATIO indicator counter to the low end of tolerance band shown on thrust check chart in CVA Specification 654-9. Advance throttle smoothly and rapidly to "MILITARY" and obtain peak reading of engine pressure ratio.

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If indicator pointer does not pass movable index, the engine is not acceptable for flight.

OIL PRESS indicator	40 to 50 psi
EXH TEMP indicator	630°C maximum

Note

During acceleration to "MILITARY," EXH TEMP may go as high as 635°C.

MANUAL FUEL CHECK.

Throttle	"IDLE"
FUEL FLOW indicator	1,150 to 1,550 pph
Tachometer	53 to 65% rpm

Note

When FUEL CONTROL switch is placed in "EMERG," rpm may drop. To ensure electrical power output, maintain a minimum of 57% rpm.

Cycle FUEL CONTROL switch 3 times to "EMERG" to bleed system. Advance throttle slowly to "MILITARY."

CAUTION

Do not exceed 638 EXH TEMP indicator reading and engine PRESSURE RATIO indicator limits determined for thrust check.

Proper engine pressure ratio should be reached before throttle is in "MILITARY."

Throttle	57% rpm
FUEL CONTROL switch	"NORMAL"

AIRCRAFT CHECK.

A brief TAKEOFF check list appears on the right side of the instrument board.

Wing	Raised
Canopy	Closed

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CRUISE DROOP switch	"OUT"
FUEL CONTROL switch	"NORMAL"
Pilot's OXYGEN switch	"ON"
FUEL TRANS switch	"AUTO"
PITOT HEAT switch	As desired
Instruments	Check and set
Rudder trim	Neutral trim (RUD NEUT) light on
Roll trim	Neutral trim (AIL NEUT) light on
Pitch trim	Takeoff value
Flight controls	Checked
Bypass duct doors	Fully open
Engine PRESSURE RATIO indicator index	Set to takeoff value

FIELD TAKEOFF.

Refer to Appendix I for takeoff ground roll distance. Refer to section III for takeoff emergencies. Maximum overshoot exhaust temperature (cold engine) is 635°C, maximum overshoot oil pressure is 80 psi.

CAUTION

Release brakes before igniting afterburner. Brakes will not hold airplane with throttle in "MAX."

Throttle	Advance to "MILITARY" or "MAX"

CAUTION

If, after igniting afterburner, a rapid rise in exhaust temperature and a decrease in rpm are observed, stop afterburning immediately as exhaust nozzle flaps failed to open.

OIL PRESS indicator	40 to 50 psi
EXH TEMP indicator	635°C maximum
Tachometer	105.5% rpm, maximum

CAUTION

If tachometer indication exceeds 107%, reduce power to the minimum required for flight and land as soon as possible.

Engine PRESSURE RATIO indicator . . . . . Takeoff value from "MILITARY THRUST CHECK" (may be slightly higher or lower in afterburner.

During takeoff roll, maintain directional control by use of brakes and nose gear steering.

AFTER TAKEOFF.

Airspeed restructions require wing lowering and locking before reaching 210 knots IAS and landing gear retraction before reaching 240 knots IAS. (Refer to section V for operation limitations.) Because of rapid acceleration when the afterburner is used, the time available for completion of gear and wing positioning is greatly reduced. Retraction of the gear will take about 8 seconds and cause almost no trim change. Lowering of the wing will take approximately 6 seconds and should be started at approximately 190 knots.

CAUTION

Airspeeds above 210 knots IAS are not permitted until the wing is down and locked (DN LOCK handle in "LOCK" and WING POSITION light out).

WING POSITION handle release button	Depress
WING POSITION handle	"DN" (below 210 knots)
DN LOCK handle	"LOCK" (below 210 knots)
WING POSITION LIGHT	Off
VENTRALS position indicator	"DN"
FUEL TRANS switch	"AUTO"
CRUISE DROOP switch	"IN"
LE DROOP indicator	"UP"

If the DN LOCK handle cannot be placed in "LOCK", or if the WING POSITION light remains on, recycle wing.

Landing gear handle	"WHEELS UP"
Landing gear indicators	"UP"

CLIMB.

Refer to Appendix I for climb data.

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FUEL MANAGEMENT.

Since fuel transfer is automatically regulated to provide center-of-gravity control, pilot action normally is required only for fuel quantity monitoring and pump turnoff functions.

Normal fuel transfer control is selected by positioning the FUEL TRANS switch to "AUTO" before the engine is started. After a fuel cell quantity check is performed, the TANK QTY SELECTOR switch is positioned to "MAIN" to permit monitoring main cell quantity during flight.

Note

With the TANK QTY SELECTOR switch in "MAIN," it is possible to check for proper fuel transfer operation by observing that the FUEL quantity indicator "TANK" indication remains nearly constant while the "TOTAL" bar shows decreasing quantity. It should be noted, however, that during afterburning, engine demands may exceed transfer rates and the main cell level may drop temporarily.

The FUEL TRANS switch is left in "AUTO" until the WING TANK ON light comes on. The TANK QTY SELECTOR switch should then be positioned to "FWD" and to "AFT" and the FUEL quantity indicator "TANK" indication should be read in each position to ascertain if forward and aft fuselage transfer cells are empty. If they are, the FUEL TRANS switch should be positioned to "PUMP OFF - WING ON" to turn off the transfer fuel booster pumps. Wing fuel transfer, which begins automatically when the fuselage transfer system is no longer able to meet engine demands, will continue to meet engine demands until the wing is empty. Return the TANK QTY SELECTOR switch to "MAIN."

If wing fuel transfer begins because of maneuvering or high fuel demand during afterburning, and a quantity check reveals that there is still fuselage transfer fuel available, it is possible to reset the system when the main cell "recovers" to the transfer level. When cruise flight conditions are restored and the FUEL quantity indicator shows that the main cell has recovered, position the FUEL TRANS switch to "PUMP OFF - WING ON" momentarily, then back to "AUTO." The WING TANK ON light will go out and the wing fuel transfer circuit is reset to begin transferring fuel automatically when the fuselage cells are pumped out.

The MAIN TANK LOW LEVEL warning light comes on as an indication that only reserve fuel (approximately 1,425 pounds) remain in the main fuel

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system. When the light comes on, ascertain that the transfer system is empty by positioning the TANK QTY SELECTOR switch to "FWD", "AFT", and "WING" and reading the FUEL quantity indicator "TANK" indication for each position.

Note

Refer to section III, EMERGENCY PROCEDURES, for action required in the event of a fuel system failure.

Since fuel system reliability is based upon the use of fuselage transfer fuel before wing and main system fuel is used, no attempt should be made (by positioning the FUEL TRANS switch in "PUMP OFF - WING ON" early in the flight plan) to use fuselage transfer fuel for reserve.

DESCENT.

Refer to Appendix I for data on maximum range descent with engine at "IDLE" thrust. Refer to section III for data on maximum range descent with dead engine. Refer to section VI for dive recovery characteristics and maximum rates of descent.

Note

Do not exceed a rate of descent of 10,000 feet per minute with the engine windmilling and with the air-conditioning system inoperative (EMERGENCY OFF switch actuated).

BEFORE ENTERING TRAFFIC PATTERN.

HYD PRESS indicator	3,000 (±200) psi all systems
Position indicators	"Sawtooth" not showing
Attitude indicator	"OFF" not showing
Instruments	Set and checked
FUEL quantity indicator	Sufficient quantity on TOTAL counter
FUEL FLOW indicator	Checked
FUEL TRANS switch	"PUMPS OFF - WING ON"
OIL PRESSURE indicator	40 to 50 psi
DEFOG switch	As necessary
RAIN REMOVE switch	As necessary
EMERG GEN swtich	"OFF"

Shoulder harness . . . . . Locked

TRAFFIC PATTERN.

See figure 2-1. Refer to section V for maximum allowable landing gross weights. A brief checklist appears on the left side of the instrument board.

CAUTION

Maximum speed for gear extension is 240 knots IAS; maximum speed for raising the wing is 210 knots IAS.

CRUISE DROOP switch	"OUT"
Landing gear handle	"WHEELS DOWN"
LANDING GEAR indicators	"DN"
DN LOCK handle	"UNLOCK"
WING POSITION handle release button	Depressed
WING POSITION handle	"UP"
WING POSITION light	Off
VENTRALS position indicator	"UP"
LE DROOP indicator	"DN"
SP BK switch	"IN"
DUCT BYPASS DOORS switches	"OPEN"

FIELD LANDING.

Refer to Appendix I for landing ground roll distances. Refer to section III for landing emergencies. See figure 2-1 for landing pattern.

Note

Enter the downwind leg of the landing pattern with a minimum of 1,700 pounds of fuel. This allows for a complete approach, one wave off with 5 seconds of afterburning, a go-around, and a normal landing plus a reserve of 800 pounds.

BEFORE SHUTDOWN.

WING POSITION handle release button	Depressed
WING POSITION handle	"DN"
DN LOCK handle	"LOCK"
WING POSITION light	On
CRUISE DROOP switch	"IN"
AIL NEUT light	On
Wings	Folded

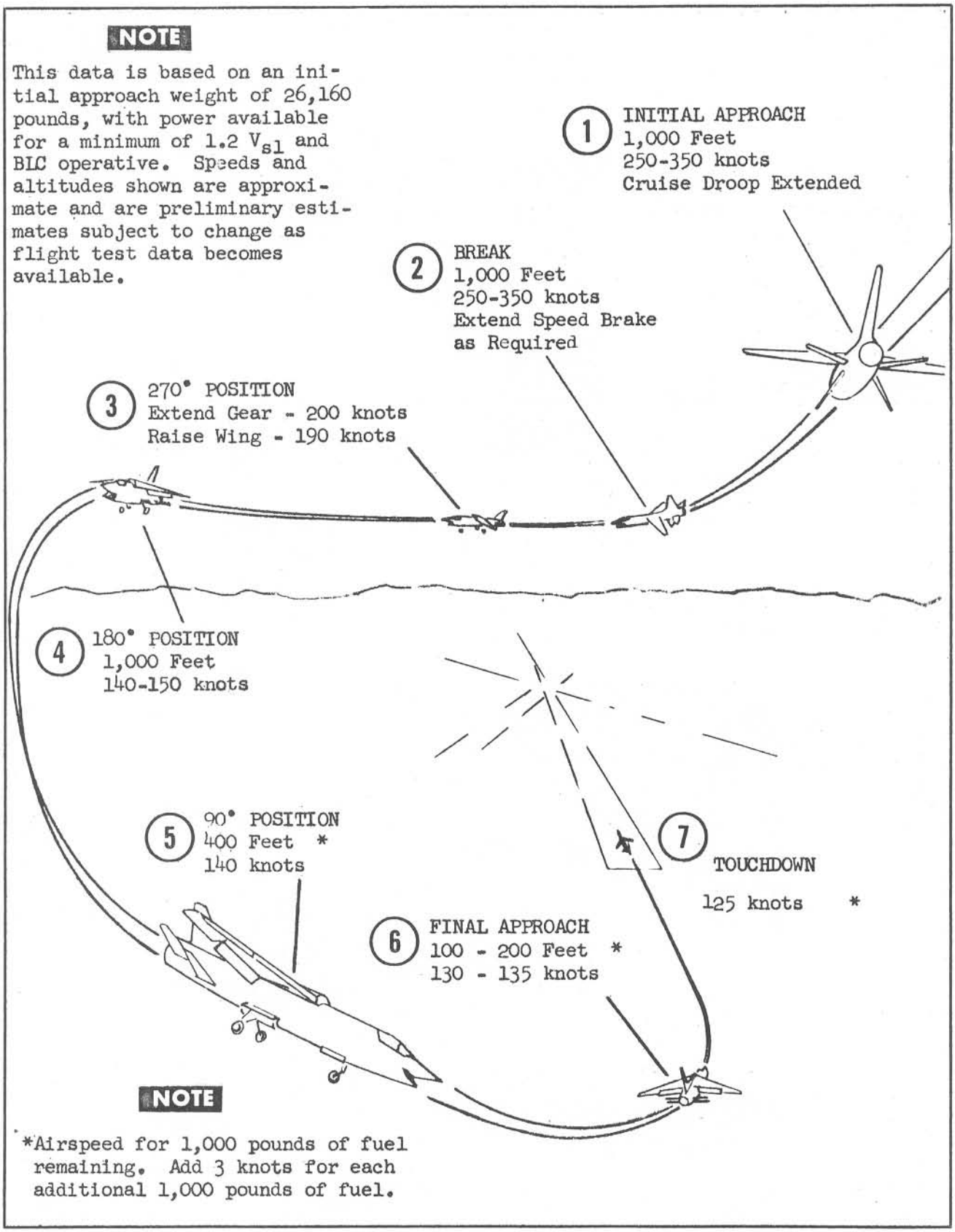


Figure 2-1. Field Landing

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CAUTION

Ailerons must be in neutral before folding wings to prevent damage to ailerons.

Communication and navigation switches	Off
Trim knobs	Neutral
DEFOG switch	"OFF"
RAIN REMOVE switch	"OFF"
PITOT HEAT switch	"OFF"
YAW switch	"OFF-RESET"
ROLL switch	"OFF-RESET"
PITCH switch	"OFF-RESET"

STOPPING ENGINE.

When the engine has been operated at high power settings for an appreciable length of time, operate the engine at "IDLE" for 3 to 5 minutes to allow it to cool and prevent seizure of the rotors.

Before stopping engine, advance throttle to operate engine at 75% rpm for 30 seconds (when conditions permit), then retard throttle of "OFF." This will scavenge residual oil to provide a true oil level for servicing and to reduce oil leakage from engine bearing cavities.

After placing throttle in "OFF," check the tachometer for free engine deceleration indication.

CAUTION

Do not shut down engine by use of the ENGINE MASTER switch, except in an emergency, because damage to engine-driven fuel pump will result from cavitation.

BEFORE LEAVING AIRPLANE.

Interior and exterior light switches	Off
EMERG GEN switch	"OFF"
MASTER GEN switch	"OFF"
SP BK switch	"OFF"
ENGINE MASTER switch	"OFF"
All other switches	Off
Wheels	Checked
Face curtain safety pin	Inserted in face curtain housing
Alternate firing handle safety pin	Inserted in seat pan

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SECTION III

EMERGENCY PROCEDURES

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ENGINE FAILURES.

An unexpected drop in engine speed will probably be observed as the first indication of an engine failure. An accompanying drop in engine pressure ratio and in exhaust temperature will occur and indications of electrical power failure will follow. If a turbine failure has occurred, exhaust temperature may rise until power is reduced.

Erratic engine operating indications, possibly accompanied by compressor stall, indicate unstable engine operation which may immediately precede an engine failure.

ENGINE FAILURE DURING TAKEOFF.

If the engine fails when there is sufficient runway remaining to permit stopping, retard the throttle to "OFF" and apply braking as required. The EMERG BRAKE handle should be employed if necessary. If the engine fails when there is inadequate runway remaining to permit stopping on the runway, and it is considered desirable to collapse the landing gear, proceed as follows:

EMER DN LOCK release button	Held depressed
Landing gear handle	"WHEELS UP"
Throttle	"OFF"
ENGINE MASTER switch	"OFF"

If the engine fails after the airplane becomes airborne, but airspeed is not high enough to permit a "zoom" to safe ejection altitude, land straight ahead. Proceed as follows:

Two-position wing	As is
EMERG POWER handle	Pulled
EMER CANOPY JETTISON handle	Pulled
Landing gear handle	"WHEELS DOWN"
EMERG VENTRALS UP handle	Pulled (If ventral fins are down)

Note

With emergency power package supplying only hydraulic power (EMERG GEN switch "OFF"), the minimum airspeed at which adequate control response will be maintained is 130 knots IAS (160 knots IAS with EMERG GEN switch in "LAND" to supply electrical power).

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If safe ejection altitude can be reached, follow "EJECTION PROCEDURE".

ENGINE FAILURE DURING FLIGHT.

If the engine fails in flight and sufficient altitude is available, perform "FLAMEOUT PROCEDURE" and then attempt an airstart.

WARNING

To prevent entry of air into the engine fuel system, always place the throttle in "OFF" immediately upon recognizing a flameout. Advance the throttle only when "AIRSTART PROCEDURE" is established.

FLAMEOUT PROCEDURE.

Throttle	"OFF"
EMERG POWER handle	Pulled
EMERG GEN switch	"OFF"

WARNING

Do not extend emergency power package with the EMERG GEN switch in "ON" or "LAND." Extending the package at airspeeds below 200 knots IAS with a load on the generator may preclude obtaining electrical power for an airstart.

EMERG GEN switch	"ON"
Emergency power indicator light	On
Position indicators	"Sawtooth" not visible
Attitude indicator	"OFF" flag not visible

AIRSTART PROCEDURE.

If there are no indications of fire or explosion and there is no severe engine vibration, accomplish "FLAMEOUT PROCEDURE" and proceed with an airstart attempt as follows:

Airspeed	180 knots IAS to 0.70 IMN
FUEL CONTROL switch	"NORMAL or "EMERG," as situation dictates
Throttle	"IGNITE" momentarily, then "IDLE"

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WARNING

To ensure availability of fuel when attempting an airstart with less than 1,000 pounds of fuel remaining in the main system and with the EMERG GEN switch in "LAND," attain a level or nose-up attitude as the throttle is advanced to "IDLE."

EXH TEMP indicator	638° C, maximum
OIL PRESS indicator	35 psi, minimum
Tachometer	Engine speed rising to correspond with throttle setting
Engine PRESSURE RATIO indicator	Rising indication
Throttle	Advance to desired setting after engine stabilizes at "IDLE"

Note

Engine idling speed increases with altitude; idling speeds as high as 75% rpm are not unusual at high altitudes.

Master generator switch	"EXT & RESET" momentarily, then "MAIN"
EMERG GEN switch	"OFF"
Position indicators	"Sawtooth" not visible
Attitude indicator	"OFF" flag not visible
LOW HYD PRESSURE light	Off
HYD PRESS indicator	Check for proper pressures
Stabilization circuits	Reset

UNSATISFACTORY AIRSTART.

An airstart attempt is regarded as unsatisfactory under the following conditions:

Ignition, as indicated by simultaneous rise in engine speed, exhaust temperature, engine pressure ratio, and engine oil pressure, does not occur within 20 seconds after the throttle is advanced to "IDLE."

Engine does not accelerate to idling speed within 45 seconds after ignition.

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Exhaust temperature exceeds starting temperature limit of 638° C.

Note

If oil pressure does not return to normal operating value of 40- to 50-psi, maintain lowest thrust settings compatible with operating conditions and return to base.

If an unsatisfactory airstart attemp occurs, proceed as follows:

Throttle. . . . . "OFF"
FUEL CONTROL switch . . . . . "EMERG"
Repeat "AIRSTART PROCEDURE" after ignition timer has completed its 30-second cycle.

Note

The ignition timer runs for 30 seconds after the throttle is placed in "IGNITE." Selecting "IGNITE" again before the 30-second period expires will not recycle the timer.

If a start is not obtained in second attempt, manually depress the "IGNITE" switch in the throttle quadrant and repeat "AIRSTART PROCEDURE."

ENGINE OVERSPEEDING AND OVERHEATING.

If engine speed exceeds 106.7% rpm but does not exceed 108% rpm, monitor throttle setting to avoid further overspeeding and report overspeed condition. If speed exceeds 108% rpm, monitor throttle setting and return to base.

If engine exhaust temperature exceeds 638° C but remains below 700° C for 2 minutes or less, or if temperature remains between 700° C and 725° C for 5 seconds or less, monitor throttle setting and report overtemperature. If these limits are exceeded, monitor throttle setting and return to base.

AFTERBURNER FLAMEOUT.

If an afterburner flameout occurs, immediately withdraw the throttle from the afterburner range to stop fuel flow and close the exhaust nozzle. When making repeated attempts to get a relight, delay momentarily in non-afterburning range to allow the afterburner igniter valve to recycle.

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AFTERBURNER CUTOFF FAILURE.

If afterburning continues after the throttle is withdrawn from the afterburner range, retard the throttle smartly past the afterburner aft detent stop. This action will terminate afterburning mechanically and it will not be possible to re-ignite the afterburner.

Note

If cutoff failure occurs because of loss of main electrical system power, normal operation will be restored when emergency power is available.

MAXIMUM GLIDE.

Maximum gliding distance with a dead engine is attained with the wing down and the leading edges in the cruise droop position.

Information on glide speeds and distances will be furnished at a later date.

FIRE.

ENGINE FIRE ON THE GROUND.

If the FIRE warning light is lit or fire is observed in the aft section while external power and the starter services are still connected to the airplane, proceed as follows:

Throttle	"OFF"
ENGINE MASTER switch	"ON"
MASTER GEN switch	"EXT & RESET"
Throttle	"CRANK" (momentarily)

Allow cranking to continue until fire has been extinguished or it is necessary to abandon the airplane, then:

ENGINE MASTER switch	"OFF"
MASTER GEN switch	"OFF"

If external power and the starter services have been disconnected and it is not possible to reconnect them, proceed as follows:

Throttle	"OFF"
ENGINE MASTER switch	"OFF"
MASTER GEN switch	"OFF"

Abandon the airplane.

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ENGINE FIRE IN FLIGHT.

If the FIRE warning light comes on, make the following check (altitude permitting):

Throttle. . . . . "IDLE"

Push TEST switch to check operation of fire detection system. If the FIRE warning light goes out, advance the throttle to resume normal operation, being careful to avoid further overheating.

If the light comes on again or if there are other indications of an engine or aft section fire, proceed as follows:

Throttle. . . . . "OFF"
ENGINE MASTER switch. . . . . "OFF"
EMER POWER handle . . . . . Pulled
EMER GEN switch . . . . . "ON"

If indications of a fire still exist after the engine is shut down, perform "EJECTION PROCEDURE." If no indications of a fire exist, perform "DEAD ENGINE LANDING" procedure (terrain and other conditions permitting).

ELECTRICAL FIRE.

If an electrical fire occurs, attempt to locate and de-energize the circuits involved by shutting down individual systems. If this fails, place MASTER GEN switch in "OFF" and follow procedure for "COMPLETE ELECTRICAL FAILURE."

COCKPIT SMOKE AND FUMES.

When smoke or fumes are detected in the cockpit, proceed as follows:

Air-Conditioning EMERG OFF switch . . . . . "CLOSED"
CABIN PRESSURE switch . . . . . "OFF"
DEFOG switch . . . . . "CLOSED"
Emergency cockpit ventilation handle. . . . "OPEN"

DEAD ENGING LANDING PROCEDURE. (Refer to figure 3-3.)

LANDING EMERGENCIES.

In all landing emergencies, wing cell fuel should be dumped before the final approach and landing. Proceed as follows:

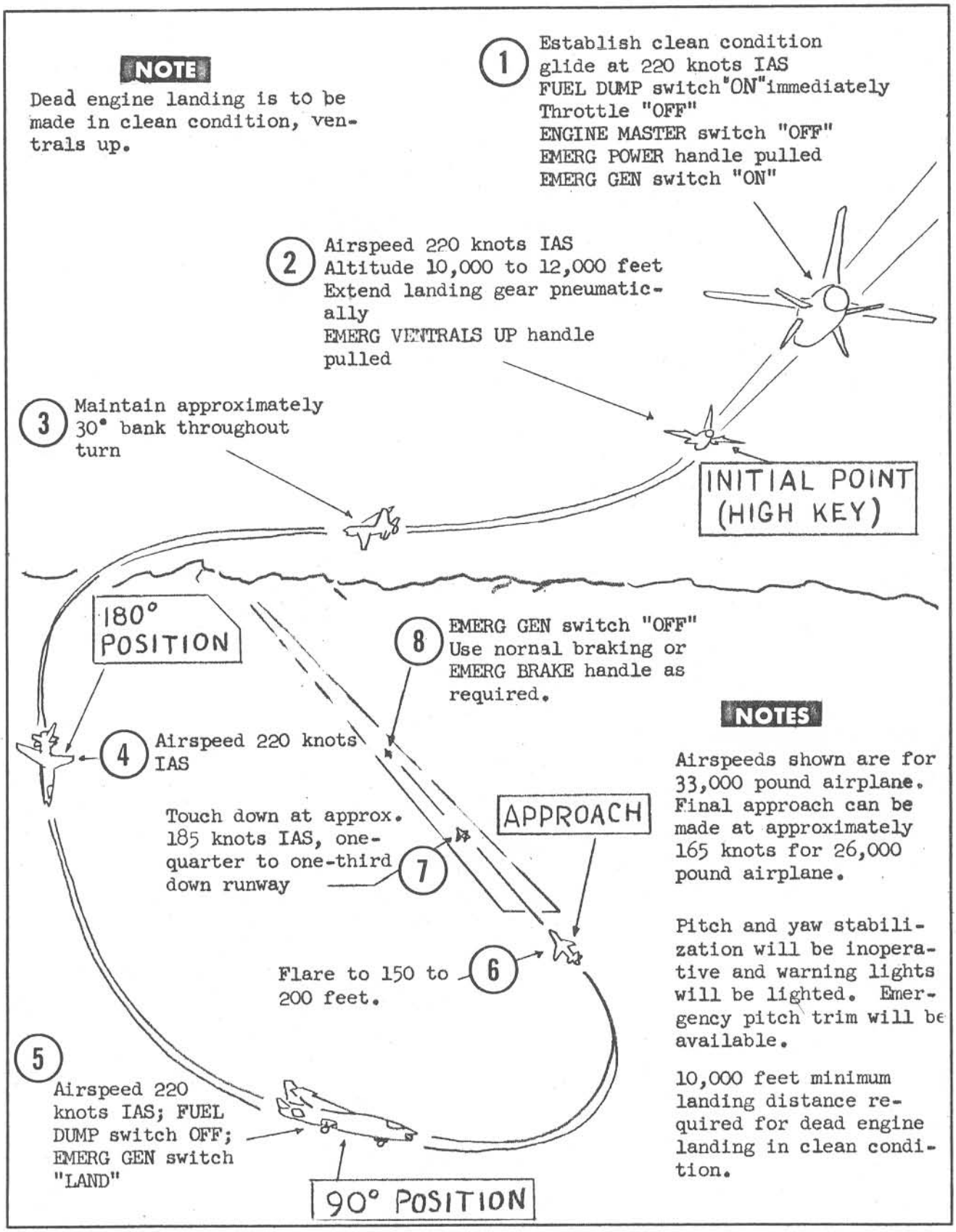


Figure 3-1. Dead Engine Landing

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FUEL DUMP switch. . . . . Actuated

Maximum fuel dumping is obtained by maintaining clean condition wings-level flight. The full wing cell fuel load can be dumped in approximately 2.5 minutes at sea level. Altitude should not be permitted to increase more than 7,000 feet during fuel dumping.

Note

Complete fuel dumping is not possible in nose-down attitudes. To ensure complete dumping, maintain slight nose-up attitude.

When dumping is complete, or just before touch-down if dumping is not complete, reposition the FUEL DUMP switch to "OFF."

In all landing emergencies, jettison missiles in a safe area before attempting a landing.

GEAR UP LANDING.

If a gear-up landing must be made, use normal landing procedure until just before touch-down, then proceed as follows:

EMER CANOPY JETTISON handle. . . . . Pulled
Throttle . . . . . "OFF"
ENGINE MASTER switch . . . . . "OFF"
MASTER GEN switch. . . . . "OFF"

Abandon the airplane as soon as it comes to rest.

LANDING WITH NOSE GEAR UP.

If a landing must be made with the nose gear up, or down but not locked, proceed as follows:

EMER CANOPY JETTISON handle . . . . . Pulled

Use normal landing procedures until the nose begins to drop during the landing rollout then:

Throttle. . . . . "OFF"
ENGINE MASTER switch. . . . . "OFF"
MASTER GEN switch . . . . . "OFF"

Ease the nose down before losing pitch control. Use brakes only if necessary to avoid obstacles.

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LANDING WITH ONLY ONE MAIN GEAR DOWN.

If a landing must be made with only one main gear down, proceed as follows:

EMER CANOPY JETTISON handle . . . . . Pulled

Use normal landing procedure in the approach. Touch down on the side of the runway corresponding to the extended gear. When the wing opposite the extended gear begins to drop following touch-down, accomplish the following:

Throttle. . . . . "OFF"
ENGINE MASTER switch. . . . . "OFF"
MASTER GEN switch . . . . . "OFF"

Ease the wing down before losing roll control.

LANDING WITH BLOWN MAIN GEAR TIRE.

When a landing is to be made with a blown main gear tire, land in the center of the runway and maintain directional control by using nose gear steering. Do not use brakes unless absolutely necessary in order to avoid obstacles.

CAUTION

To minimize skidding, neutralize rudder pedals before engaging nose gear steering.

If arresting gear is available, make an arrested landing.

EMERGENCY ENTRANCE.

To gain access to the cockpit in an emergency, employ the EMERGENCY CANOPY JETTISON handle (identified by a "RESCUE" arrow on both sides of the fuselage) as follows:

Press the release tab in until the handle is freed, then grasp the handle and pull it away from the fuselage to full travel of the lanyard. A sharp pull will then fire the canopy actuator. After the lanyard is pulled, continue to move away from the airplane so as to avoid being hit by the canopy.

EJECTION PROCEDURE.

Note

Ejection at airspeeds from stall speed to 525 knots results in relatively minor forces being exerted on the body.

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Ejection at airspeeds from 525 knots to 600 knots results in appreciable forces and escape is hazardous. Ejection at airspeeds above 600 knots is extremely hazardous.

Safe ejection at low altitude is facilitated by a "zoom" maneuver which increases time available for separation from the seat and for parachute deployment.

The Martin-Baker ejection seat, althought commonly referred to as providing runway- or ground-level escape, does not have a ground-level capability as presently installed in the F8U-3. The seat will provide escape at very low altitudes, as determined by functions of flight path and speed, and throughout the flight envelope up to approximately 600 knots.

NORMAL SEQUENCE.

1 If circumstances permit, reduce airspeed to 400 knots or less by extending the speedbrake, retarding throttle, and/or performing a "zoom" maneuver.

2 Sit erect with head braced firmly against parachute pack and legs pulled in against face of seat pan.

3 Grasp face curtain handle with both hands, elbows in and thumbs outward, pull firmly forward and downward with a continuous movement. If, for any reason, the face curtain handle cannot be reached, grasp the alternate firing handle, positioned in the front face of the seat pan, with either hand and pull handle firmly upward.

WARNING

Use the alternate firing handle only if it is not possible to use the face curtain handle.

4 If the canopy fails to be jettisoned when either the face curtain handle or the alternate firing handle is pulled, pull the EMER CANOPY JETTISON handle. If this fails, attempt to jettison the canopy by actuating the normal canopy release handle. If the canopy cannot be jettisoned by these means, pull the seat emergency arming handle and again pull the face curtain handle or the alternate firing handle to eject the seat through the canopy.

WARNING

Ejection through the canopy is hazardous and should be employed only as a last resort.

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MANUAL SEAT SEPARATION SEQUENCE.

In the event of damage to the seat mechanism which prevents normal operation of the automatic mechanism after ejection, provision is made for manual separation from the seat and manual deployment of the personnel parachute.

1 With the right hand, pull the forward "D" ring on the lap belt away from the body to full length of its travel. The parachute and automatic mechanism will be disconnected and the aft "D" ring will be uncovered.

2 Pull back the harness emergency lever on the right side of the seat. The seat harness will unlock.

3 Roll out of seat, and when clear, pull the aft "D" ring on the lap belt. The personnel parachute will deploy and descent will be normal.

Note

Considerably more altitude is required for manual operation of the escape system than for the normal sequence.

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FUEL SYSTEM FAILURES.

FUEL CONTROL UNIT FAILURE.

Failures of the engine fuel control system will be indicated by erratic engine instrument readings at a constant throttle setting, or by unusual response (stalling, overspeeding, underspeeding) to throttle movement. When such a failure is suspected, select emergency manual fuel control as follows:

Throttle. . . . . "IDLE" (conditions permitting)
FUEL CONTROL switch . . . . . "EMERG"

WARNING

Throttle movements must be made slowly to prevent flameout, overtemperature or overspeed. Do not return the FUEL CONTROL switch to "NORMAL" for duration of flight, since doing so may result in a flameout.

In an emergency, the afterburner can be used with "EMERG" fuel control selected.

Land as soon as possible.

FUEL PUMP FAILURE.

There is no direct indication of failure of the engine stage of the engine-driven fuel pump. Since the afterburner stages may not supply adequate fuel for maximum engine performance (afterburning) in all operating regimes, a sudden drop in thrust may indicate such a failure. If fuel pump failure is suspected, discontinue afterburning and return to base.

TRANSFER FUEL SYSTEM FAILURE.

Failure of the transfer fuel system forward or aft cluster booster pumps will be indicated by the WING TANK ON light coming on when fuel remains in the forward or aft transfer cells (refer to "FUEL MANAGEMENT" for normal procedure when light comes on).

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Note

The WING TANK ON light may come on when high fuel demand during afterburner operation decreases main fuel cell level. If the forward and aft cells still contain fuel in this case, place FUEL TRANS switch in "PUMP OFF-WING ON" momentarily when afterburning is terminated, then reposition to "AUTO" until transfer is complete.

If it is suspected that both pumps have failed, place FUEL TRANS switch in "PUMPS OFF-WING ON" and return to base. If it is suspected that either pump has failed, but not the other, allow transfer to continue only if the total of main fuel and wing fuel is inadequate to permit return to base. If fuel is adequate, position FUEL TRANS switch to "PUMPS OFF-WING ON" and return to base.

Lighting of the MAIN TANK LOW LEVEL light before the WING TANK ON light comes on may indicate failure of the automatic sequencing system. If this occurs, check wing cell fuel level and if it appears that fuel is not transferring, place FUEL TRANS switch in "PUMP OFF-WING ON" to bypass the automatic sequencing system and pressurize the wing.

ELECTRICAL SYSTEM FAILURES.

MAIN POWER SYSTEM FAILURES.

A failure of the main generator will result in loss of both ac and dc power as indicated by simultaneous loss of all ac-operated indicating systems and appearance of "sawtooth" indication on all position indicators.

Failure of the main transformer-rectifier will result in loss of only dc power as indicated by appearance of "sawtooth" indication on all position indicators. AC-operated indicating systems will continue to function.

WARNING

If afterburner is in use at time of electrical failure, stop afterburning by retarding throttle smartly past afterburner aft detent stop. The throttle can then be advanced again but afterburning cannot be resumed until power is obtained from the emergency power package.

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When a main power system failure occurs, proceed as follows:

EMERG GEN switch . . . . . "OFF"
EMERG POWER handle . . . . . Pulled

WARNING

Do not extend emergency power package with the EMERG GEN switch in "ON" or "LAND." Extending the package below 200 knots IAS with a load on the generator may preclude obtaining electrical power.

EMERG GEN switch . . . . . "ON"
Emergency power indicator light. . . . . On
Position indicators. . . . . "Sawtooth" not visible
Attitude indicator . . . . . "OFF" flag not visible
ROLL, PITCH, and YAW stabilization switches. . . . . "OFF-RESET" momentarily, then "ON"
STAB OFF lights. . . . . Off

Note

To ensure availability of fuel with less than 1,000 pounds of fuel remaining in the main system and with EMERG GEN switch in "LAND," maintain a level or nose-up attitude.

Do not extend the speed brake above 250 knots IAS with less than 1,000 pounds fuel remaining.

Do not fly at negative g loads or remain between +0.5g and +0.2g or -0.2g and -0.5g for more than five seconds. Flight from +0.2g to -0.2g should be accomplished as a rapid transient condition.

Attempt afterburning only in an emergency. Afterburner operation will be reliable only below 6,000 feet.

COMPLETE ELECTRICAL FAILURE.

In case of a complete electrical failure, proceed as follows:

a. Reduce altitude and retard throttle as necessary to maintain engine operation under reduced fuel flow resulting from loss of all fuel booster pumps.

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b. Do not exceed restrictions listed in "Note" under "MAIN POWER SYSTEM FAILURES."

WARNING

If afterburner is in use when emergency power system fails, retard throttle smartly past afterburner aft detent stop to terminate afterburning. The throttle may then be advanced to resume normal engine operation, but afterburning will not be available.

UTILITY HYDRAULIC SYSTEM FAILURES.

Failure of the utility hydraulic system will be indicated by the LOW HYD PRESSURE light coming on and pointer "2" of the HYD PRESS indicator showing low pressure when the UTILITY button is depressed. The emergency pneumatic system permits emergency actuation of the following components that are normally powered by the utility hydraulic system: two-position wing, wing leading edges, ventral fins, wheel brakes and landing gear.

CAUTION

If a system is actuated pneumatically, do not subsequently reposition controls in an attempt to actuate the system hydraulically until necessary maintenance has been performed.

VENTRAL FINS, WING, AND LEADING EDGES.

To raise the ventral fins and the wing and to extend the leading edges to landing droop position, proceed as follows:

Airspeed	210 knots IAS, maximum
EMERG VENTRALS UP handle	Pulled
DN LOCK handle	"UNLOCK"
EMERG DROOP & WIND INC guard	Raised
WING POSITION handle release switch	Depressed
WING POSITION handle	Moved fully forward, then inboard and aft to "EMERG UP"

WARNING

The WING POSITION light will be on regardless of the relative positions of wing and landing gear, since the flaps remain up when pneumatic wing actuation is employed.

LE DROOP indicator . . . . . "DN"
VENTRALS indicator . . . . . "UP"

LEADING EDGES.

The leading edges can be extended to the landing droop position, without raising the wing, by proceeding as follows:

EMER DROOP & WING INC guard. . . . . Raised
WING POSITION handle release switch. . . . . Depressed
WING POSITION handle . . . . . Moved fully forward

VENTRAL FINS.

Emergency retraction of the ventral fins occurs automatically when emergency wing actuation is selected. Retraction also may be obtained separately by pulling the EMER VENTRALS UP handle.

WARNING

If emergency ventral retraction is performed while the wing is down, the wing may be raised hydraulically but should not be lowered until the ventral system has been serviced.

LANDING GEAR.

Airspeed . . . . . 240 knots IAS, maximum; 185 knots IAS, minimum
Landing gear handle. . . . . Position to "WHEELS DOWN," push in, rotate clockwise and pull aft.

Note

Although the gear will be extended whenever the handle is rotated and pulled aft, nose gear down lock and lock indication will be available only if the handle is also positioned to "WHEELS DOWN."

LANDING GEAR position indicators . . . . . "DN"
WHEELS warning light . . . . . Not flashing when throttle is retarded to approach power settings

If the WHEELS warning light is flashing or any of the position indicators do not show "DN," rock the airplane or dive and climb in an attempt to force the gear to the down-and-locked position.

WHEEL BRAKES.

Emergency wheel-braking is obtained by moving the EMERG BRAKE handle toward "ON." Initial movement of the handle should be made slowly to avoid locking the brakes. Braking can be applied intermittently by manipulating the handle.

POWER CONTROL HYDRAULIC FAILURE.

If pressure is lost in power control system No. 1, system No. 2 will provide adequate control except that roll stabilization will not be available. To obtain roll stabilization in that case, extend the emergency power package by pulling the EMERG POWER handle. Emergency pump pressure will then be read on pointer "1" of the HYD PRESS indicator.

An hydraulic pump in the emergency power package will provide sufficient pressure and flow for operating the flight controls upon loss of output from both engine-driven pumps. The emergency pump pressurizes only power control system No. 1 and will not be effective if fluid has been lost from that system.

Note

The position of the EMER GEN switch affects hydraulic output of the emergency power package. With the switch in "LAND," electrical load on the emergency power package is reduced; the "OFF" position (used when there is no electrical failure) eliminates operating electrical loads altogether. Minimum airspeed for obtaining adequate hydraulic pressure with no electrical load is 130 knots IAS.

TRIM AND STABILIZATION FAILURES.

Failure of any or all of the trim and stabilization circuits will be indicated by inability to trim, undesirable oscillations, or illumination of one or more STAB OFF warning lights.

Note

The autopilot will disengage automatically when any or all of the stabilization circuits disengage .

YAW CIRCUIT FAILURE.

If the YAW STAB OFF warning light comes on, position the YAW stabilization switch to "OFF-RESET" momentarily, then return it to "ON." If the warning light remains lighted or goes off but immediately comes on again, place the switch in "OFF-RESET." The yaw trim and stabilization circuit is then inoperative and the rudder will automatically return to neutral trim condition.

Note

The YAW STAB OFF warning light will come on if power control hydraulic system No. 2 fails. In this event, resetting is not possible and the YAW stabilization switch should be positioned to "OFF-RESET."

ROLL CIRCUIT FAILURE.

If the ROLL STAB OFF warning light comes on, position the ROLL stabilization switch to "OFF-RESET" momentarily, then return it to "ON." If the warning light remains lighted or goes off but immediately comes on again, place the switch in "OFF-RESET." The roll trim and stabilization circuit is then inoperative and the ailerons will automatically return to neutral trim condition.

Note

The ROLL STAB OFF warning light will come on if power control hydraulic system No. 1 fails. When pressure is available from the emergency power package, the stabilization system can be reset. If both sources of pressure have failed, resetting is not possible and the ROLL stabilization switch should be positioned to "OFF-RESET."

PITCH CIRCUIT FAILURE.

If the PITCH STAB OFF warning light comes on, position the PITCH stabilization switch to "OFF-RESET" momentarily, then return it to "ON." If the warning light remains lighted or goes off but immediately comes on again, place the switch in "OFF-RESET." The pitch stabilization circuit is then inoperative but pitch trim may still operate normally.

Emergency pitch trimming is selected by pulling the EMER PITCH TRIM handle and rotating it toward "NOSE UP" or "NOSE DN" as desired.

Note

The PITCH STAB OFF warning light will come on if power control hydraulic system No. 2 fails. In this event, resetting is not possible and the PITCH stabilization switch should be positioned to "OFF-RESET." Normal pitch trim will still be available.

AILERON-RUDDER INTERCONNECT FAILURE.

Failure of aileron-rudder interconnect system on airplane BuNo. 147085 is indicated by a warning light mounted on a bracket above the left-hand console. If the light comes on, cycle the INTERCONNECT switch on the panel below the light. If the light does not go off and stay off, the system will not operate. Failure of the interconnect system does not impose any operating restrictions.

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SECTION IV

AUXILIARY EQUIPMENT

CONTENTS

Air-Conditioning System. 4-2
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Integrated Electronics (CNI) Package . . 4-5
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AIR CONDITIONING SYSTEM. (See figure 4-1).

Cooling, heating, and pressurization services are provided by a pneumatically controlled air-conditioning system that employs a heat exchanger and an expansion-turbine cooling unit to condition engine bleed air. Additional cooling capacity for very high speed flight is provided by a water boiler which absorbs heat energy from the engine bleed air. The system pressurizes the cockpit, pressure suit, fire control radar, and components of the communications, navigation and identification equipment. The system cools the cockpit, radio equipment, pressure suit, radar altimeter, and MA-1 compass gyro and amplifier. In addition, the system controls rain removal, windshield defogging and supplies cooled air to pressurize the fuel cells.

When the system is in operation (engine running), hot compressor bleed air flows through the heat exchanger, where it loses some of its heat to ram air taken from the inlet duct. Some of the air from the heat exchanger is routed to the fuel cell pressurization system and to the rain removal lines. Partially cooled air flows from the heat exchanger to the turbine.

Expansion of the air as it flows through the turbine provides cold air for the cooling services. Energy given up by the air passing through the turbine drives a fan which boosts ram air flow through the heat exchanger.

Both automatic and manually governed pneumatic controls are provided to select and maintain desired cockpit cooling or heating, which is achieved by controlled mixing of hot air from the heat exchanger and cold air from the turbine. Temperature control for pressure suit ventilating air is achieved in a similar manner. Cockpit pressurization is automatically scheduled by the cockpit air outflow valve. Cooling and pressurization of electronic systems components is automatically controlled and is effective at all times when the air-conditioning system is operating.

WARNING

Operation of this airplane at maximum performance without water in the air-conditioning system boiler will produce extremely high cockpit temperatures. Make sure boiler is serviced before each flight.

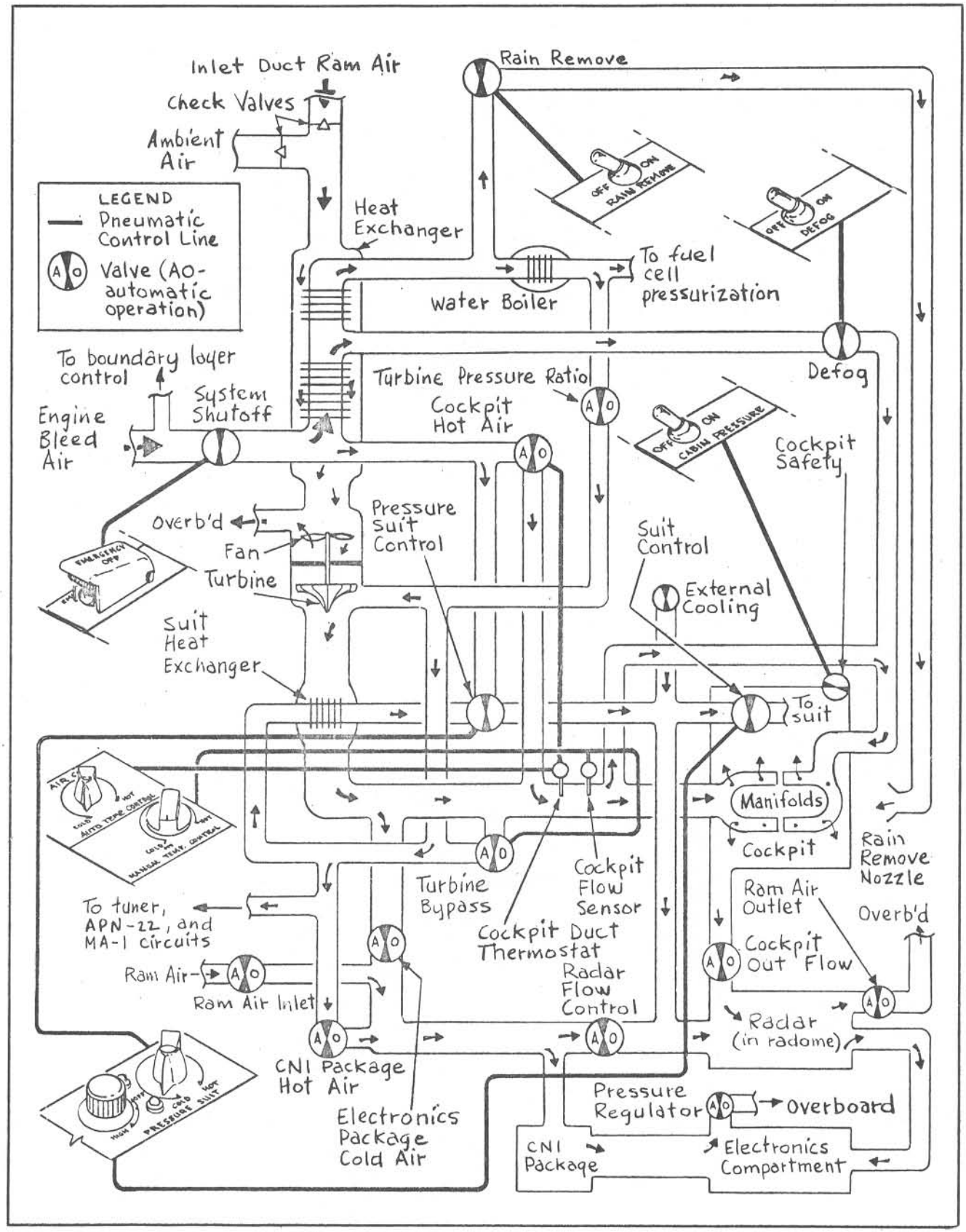


Figure 4-1 Air-Conditioning System

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AIR CONDITIONING CONTROLS.

The control panel for the air conditioning system is located on the right-hand console. It mounts two control knobs and four switches. The AUTO TEMP CONTROL knob (24, figure 1-4) provides automatic regulation of cockpit inlet temperature to maintain constant temperature during changes in flight conditions and power settings. The MANUAL TEMP CONTROL knob (25, figure 1-4) provides control of cockpit inlet temperature when AUTO TEMP CONTROL knob is in full cold position. The manual control is provided mainly to afford pilot temperature control in case of malfunction of the automatic system.

The DEFOG switch (27, figure 1-4) controls flow of heated defogging air across the inside of the windshield. The RAIN REMOVE switch (26, figure 1-4) controls the flow of heated air across the outside of the windshield. Cockpit pressurization is controlled by the CABIN PRESSURE switch (28, figure 1-4) which may be used to release cockpit pressure overboard at the pilot's discretion. The EMERGENCY OFF switch (29, figure 1-4) provides a positive means of shutting down the system by cutting off engine bleed air. When this switch is actuated, all cooling and pressurizing functions are lost except that ram air is available to cool the electronic equipment if ram air temperature is less than 120° F.

WARNING

DEFOG switch must be in the "OFF" position at speeds in excess of Mach 1.4 to avoid overheating the windshield.

PRESSURE SUIT CONTROLS.

The PRESS SUIT TEMP control knob (4, figure 1-2) controls the temperature of the ventilating air supplied to the pilot's pressure suit by the air conditioning system. The rate of ventilating air flow is controlled by the VENT AIR knob (3, figure 1-2).

EMERGENCY COCKPIT VENTILATION VALVE.

To provide an emergency means of ventilating the cockpit, a manually operated valve is located just above the air conditioning control panel on the right-hand side wall of the cockpit. This valve, when opened, extends a small scoop into the airstream so as to supply ram air in the cockpit. In order to ventilate the cockpit with this valve at low airspeeds, the CABIN PRESSURE switch must be in the "OFF" position.

PITOT HEAT.

The pitot heads are electrically heated to prevent icing. Pitot heat is controlled by the PITOT HEAT switch (12, figure 1-4). Power to energize the dual pitot head heaters is supplied from the emergency ac bus.

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INTEGRATED ELECTRONICS (CNI) PACKAGE.

GENERAL.

The F8U-3 communications, navigation and identification equipment is combined into one system called the Integrated Electronics Central AN/ASQ-19(XN-2). This equipment consists of 5 separate main modular units plus indicators, control panels, and antennas. The five main units are located in two compartments directly aft of the canopy. Four units are mounted on a rack that serves as a plenum chamber to supply air for cooling. The remaining unit, which incorporates its own external blower, is in a compartment forward of the rack. The system consists of an AN/ARC-52 UHF receiver, a TACAN navigation unit, an ADF navigation unit, and an IFF coder-receiver-transmitter. The system also incorporates intercom facilities with a station in the left main gear well to provide voice link to the cockpit. All modules of the system that employ high voltage are pressurized to prevent arcing at high altitudes and all units in the compartment aft of the cockpit are cooled by the air-conditioning system.

The system incorporates its own emergency power supply switching system whereby the power supply will switch from airplane primary to airplane emergency ac power in case of failure of the primary system. This function is automatic and serves to insure communications when the EMER GEN switch is placed in the "LAND" position.

In addition to switching between sources of airplane power, there is an emergency switching function within the unit itself that insures operation of all units except TACAN and ADF in case of failure of portions of the system's own power supply. If the 450-volt or 130-volt sections of the system power supply should fail, the system will automatically switch to its own emergency power system. The system will automatically switch back to the normal power supply if it is restored to operation. Operation of the system on emergency power is indicated by a light on the bottom of the communication control panel. In addition to loss of ADF and TACAN, a reduction of transmitter output power and, therefore, of transmitter range will be experienced. None of the UHF receiver operations will be affected.

CAUTION

The system requires cooling for all ground operation of over 10 minutes duration if the air conditioning system is not operating (engine not running). To permit ground operation of other systems for extended periods, a switch in the nose wheel well permits the system to be turned off.

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The UHF and TACAN function switches should be placed in "STBY" and IFF should be turned off when external power is applied to airplane on the ground unless external cooling is provided for the integrated electronics package. External power should not be left on for periods in excess of 10 minutes unless system circuit breakers are pulled. UHF transmitter should never be keyed on the ground unless cooling is provided.

Note

The TACAN system is inoperative on airplane BuNo. 147085.

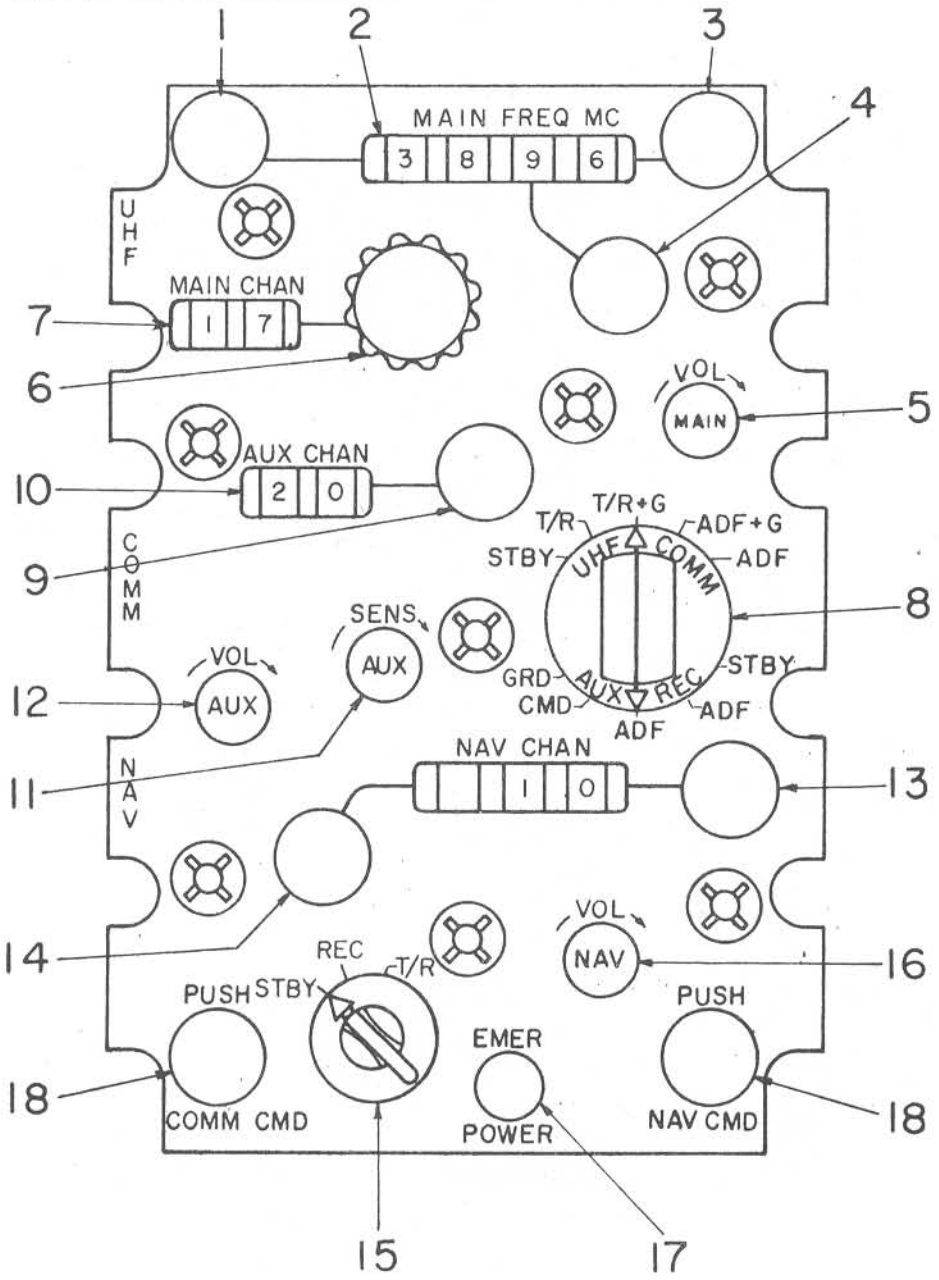
ELECTRONIC EQUIPMENT CONTROLS.

The primary controls for communications, ADF, and TACAN equipment are incorporated into a single control panel (figure 4-2) located on the right-hand console. The UHF section of the panel provides for selection of any of 19 preset channels plus guard frequency or manual selection of any of 1,750 channels between 225 and 400 megacycles. The guard frequency is 243.0 megacycles and the UHF unit incorporates a separate guard receiver so that placing the function switch in TR+G position and selecting 243.0 by manual control has the effect of placing both receivers in parallel. Manual setting of UHF channels is accomplished by selecting the letter "M" with the channel selector knob and then using the three UHF frequency control knobs to dial the selected frequency. Turning the knobs has no effect on frequency until the channel selector control is in the "M" position. The only other controls for the UHF system are the volume control and the function selector switch. The center portion of the panel is labeled COMM and includes controls for the auxiliary UHF receiver. The auxiliary receiver has 20 preset channels between 265.0 and 284.9 megacycles but provides no method for manual tuning.

The lower part of the panel incorporates TACAN controls. The controls provide for manual selection of any of 126 preset channels between 962 and 1213 megacycles. The preset channels are divided into two ranges consisting of every even frequency from 1151 through 1213 megacycles. The first range is navigation channels 1-63 inclusive and the second range is channels 64-126. Channels 127, 128 and 129 are not operational. TACAN bearing and range are displayed on the number two pointer and in the window of the Bearing Distance & Magnetic Heading Indicator.

Controls for the IFF system and are on standard panels mounted on the right-hand console. The intercom provisions are controlled from the ICS

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1. UHF frequency control (1st two digits)
2. UHF MAIN FREQ MC window
3. UHF frequency control (last digit)
4. UHF frequency control (3rd digit)
5. UHF MAIN receiver VOL control
6. UHF channel selector control
7. UHF MAIN CHAN window
8. UHF and AUX receiver function control
9. AUX receiver channel selector control
10. AUX CHAN window
11. AUX receiver SENS control
12. AUX receiver VOL control
13. Tacan channel selector control (last digit)
14. Tacan channel selector control (1st two digits)
15. Tacan function control switch
16. Tacan NAV VOL control
17. EMER power supply light
18. "Not operational"

Figure 4-2. Communication and Navigation Controls

panel mounted on the right-hand console. The intercom function switch should remain in "NORMAL" position at all times. Operation is controlled by placing toggle switch in either "HOT MIC" or "COLD MIC" position. When switch is placed in HOT MIC position, the cockpit microphone is hot and the wheel well station can hear voice signals from the cockpit. The MIKE switch on the throttle grip should not be depressed for intercom communications.

To provide the pilot with a convenient means of selecting UHF radio channels, an alternate channel selector switch and an indicator are installed in the cockpit. The UHF CHANNEL selector knob (33, figure 1-2) mounted on the left-hand console permits channel selection when the pilot's right hand must remain on the flight controls. The UHF CHANNEL indicator window (48, figure 1-3) on the instrument board places more within the pilot's line of vision an indication of the channel selected.

COURSE INDICATOR.

Note

The COURSE indicator is replaced in airplane BuNo. 147085 by a flight test accelerometer.

Course information received by the TACAN receiver is displayed on the ID-387/ARN Course Indicator. The instrument incorporates a selector knob that is used to set in selected course on the window on the top of the indicator. In normal operation, TACAN is tuned to a selected ground station. If a satisfactory ground signal is received, the blank covering the to-from window on the left upper side of the indicator will show "TO" or "FROM" depending on whether the ground station is in a compass guadrant forward or aft of the line of flight. The vertical bar is referenced to the course set in the indicator window and will indicate airplane position left or right of the selected course. The heading pointer shows the angular relationship of airplane heading to the course selected. The horizontal bar, which is designed to present glide slope information, is not operative in this airplane. The vertical bar also incorporates an OFF flag which shows when the system has lost electrical power or electrical power to the indicator has been interrupted.

BEARING, DISTANCE AND MAGNETIC HEADING INDICATOR (DRMI).

This instrument (25, figure 1-3) is a combined indicator that incorporates the functions of the conventional ID-310 Distance Indicator, the ID-307 Azimuth Indicator and the ID-250 Radio Magnetic Indicator. The indicator presents TACAN and ADF information on two pointers and a window while the compass card is referenced to the MA-1 compass system. When the

instrument is used to display only ADF information, the number two pointer is slaved to the number one pointer. ADF bearing information is displayed on the number one pointer any time the communications function switch is placed in "UHF-ADF" or "AUX REC-ADF" and the receiver is tuned to a ground station. TACAN range information is presented in a window on the dial face. This window will display an "OFF" flag when TACAN is not operating or in case of failure of electrical power to the indicator. TACAN bearing information is displayed on the number two pointer and distance information is displayed in the range window any time the TACAN function switch is placed in "REC" or "T/R" and the receiver is tuned to TACAN. (Refer to figure 4-3 for a typical example of the use of the indicator for navigation and to figure 4-2 for illustration of radio and navigation controls.)

MA-1 COMPASS SYSTEM.

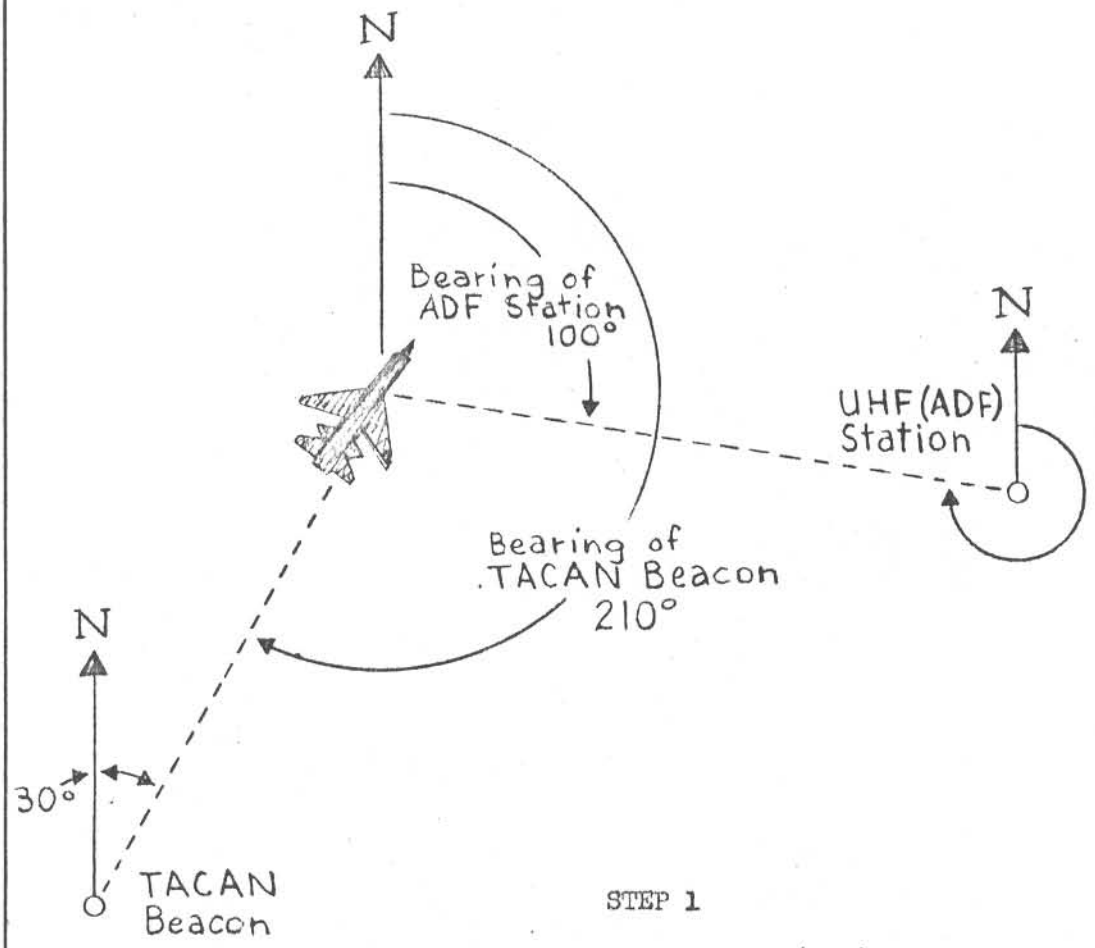
The MA-1 compass system provides flux-referenced magnetic or gyro-corrected heading information which is presented on the compass card of the DRMI. The system consists of a control panel, a gyro, an amplifier, a remote transmitter and the indicator (DRMI). The control panel (3, figure 1-4) mounts the system mode control which allows selection of FREE N LAT, FREE S LAT, or SLAVED methods of operation. In either of the free gyro positions, drift correction may be selected by the latitude switch and is applied to the gyro to correct for drift induced by the earth's rotation. In the slaved position, the gyro is slaved to corrections from the flux transmitter. In order to ensure correct operation in the SLAVED position, the PULL TO SET adjust control on the panel is used to align the compass card with approximate magetic heading, then the white bar in the window of the control panel is synchronized to the index pointers by further adjustment of the knob.

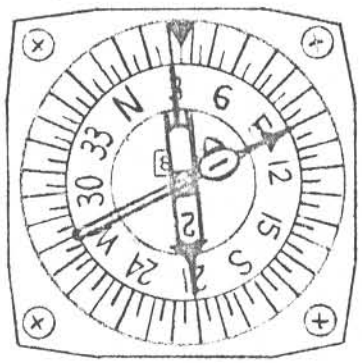
CAUTION

If the white bar does not move to the right with clockwise rotation of the PULL TO SET knob, the compass card on the indicator will not be correctly synchronized. Continue rotation of the knob until white bar moves in correct direction and is centered.

In the SLAVED mode of operation, the system has a correction rate of approximately 2° per minute so that an error of 3° induced by a sustained turn or by acrobatics would require 1.5 minutes for correction.

In the FREE N LAT mode of operation, which is the normal setting for gyro operation in the northern hemisphere, the compass card may be set to any desired magnetic heading. When the system is operating on





STEP 1

TUNE IN UHF (ADF) STATION ON AUX RECEIVER. READ BEARING WHEN RELIABLE ON NO. 1 POINTER OF BEARING DISTANCE HEADING INDICATOR. DRAW LINE FROM ADF STATION AT BEARING INDICATED BY BLUNT END OF NO. 1 POINTER.

STEP 2

TUNE IN TACAN BEACON. OBTAIN BEACON IDENT. TONE. READ BEARING OF TACAN BEACON WHEN RELIABLE ON DOUBLE NO. 2 POINTER OF INDICATOR. DRAW LINE FROM TACAN BEACON AT BEARING INDICATED BY BLUNT END OF NO. 2 POINTER.

STEP 3

AIRCRAFT POSITION IS INDICATED BY INTERSECTION OF LINES.

Figure 4-3. Typical Use of DRMI

free gyro, it should be checked and reset when an accurate heading can be obtained, inasmuch as the system can have an uncorrected drift error of 4° per hour.

The system requires approximately 2 minutes to warm up and stabilize after electrical power is applied. The synchronizing window of the control panel is equipped with a masking flag which shows "OFF" if electrical power or the system gyro should fail.

OXYGEN SYSTEM.

The F8U-3 incorporates a conventional breathing oxygen system consisting of a liquid oxygen converter, a quantity indicator, a low-pressure and low-level warning light and a shutoff switch. The converter, which serves as a storage tank and converts the liquid oxygen to gaseous oxygen, is located in a compartment below and slightly aft of the cockpit on the left side of the fuselage. The system can be serviced with standard liquid oxygen equipment and purged with gaseous oxygen. Access to the servicing connection and to the vent and buildup valve is attained through a compartment access panel. The system, when fully serviced, contains 5 liters of liquid oxygen and supplies gaseous oxygen at 70 psi to the service outlet. The system is designed to be used with a breathing regulator that is part of the pilot's equipment. Oxygen from the service outlet in the cockpit is piped to the pilot's services quick disconnect. The oxygen quantity indicating system indicates quantity in increments of one liter. The indicator is located on the lower left-hand side of the instrument panel (51, figure 1-3). The oxygen warning light is located on the upper left side of the instrument board (18, figure 1-3) and is lighted when pressure at the service outlet drops below 50 psi or when quantity drops below 1/2 liter as monitored by the quantity indicating system.

Refer to table 4-1 for oxygen duration.

WARNING

If oxygen quantity indicator shows more than 1/2 liter but oxygen warning light is illuminated, descend below 10,000 feet altitude immediately. System operation at low pressure will not supply adequate oxygen.

In the event of failure of the main oxygen system, oxygen may be obtained from the emergency oxygen (bailout) supply by pulling the "green apple" lanyard which is located at the left front corner of the seat cushion. The bailout oxygen supply is of short duration and a descent to low altitude should be made immediately. The bailout supply is automatically activated when the ejection seat is fired.

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Table 4-1. Oxygen Duration

Duration in hours and minutes with pilot's oxygen valve in "ON"

Cabin Altitude (Feet)	GAGE INDICATOR - LITERS						
	5	4	3	2	1	1/2	Below 1/2
40,000 & Above	30:18	24:12	18:12	12:06	6:00		
35,000	18:30	14:48	11:06	7:24	3:42	1:48	
30,000	13:36	10:54	9:12	5:24	2:42	1:24	
25,000	10:12	8:12	6:12	4:06	2:00	1:00	
20,000	8:00	6:24	4:48	3:12	1:36	:48	
15,000	6:24	5:06	3:48	2:36	1:18	:36	
10,000	5:00	4:00	3:00	2:00	1:00	:30	
5,000	4:12	3:18	2:30	1:36	:48	:24	
Sea Level	3:30	2:48	2:06	1:24	:42	:18	

Descend below 10,000 feet

BASED ON:
1. One 5-liter liquid oxygen converter.
2. 800 liters of gaseous oxygen per liter of liquid oxygen.
3. Consumption data per Specification MIL-I-19326 (AER) taken from NAVAER 03-50-517.

STANDBY COMPASS.

The standby compass (24, figure 1-3) is located in a bracket on the cowl above the right side of the instrument board. The standard aperiodic aircraft-type compass is used to verify and set compass card of the DRMI. The compass correction card is located on the forward part of the right-hand canopy rail.

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ARMAMENT.

GENERAL.

The airplane is equipped with a completely integrated armament system comprising Airborne Missile Control System (AMCS) Aero 1B and provisions for carrying and launching three AAM-N-6a (Sparrow III) air-to-air missiles. AMCS Aero 1B permits detection of targets under all-weather operating conditions, provides radar tracking of targets to aid the pilot in making an intercept, and supplies radar illumination of targets for radar-guided Sparrow III missiles. An air data computer supplies information on flight conditions to the AMCS and to the automatic flight control system, which relieves the pilot from some of the flying tasks required to perform an intercept (refer to "Automatic Flight Control System").

Note

Airplane BuNo. 147085 incorporates special controls for use in the Contractor's flight test program. In the following paragraphs, the words "flight test" and "production" will be employed to point out differences in the two configurations of the missile control system. Any information that is not otherwise identified is applicable to both configurations.

To permit structural evaluation, airplane BuNo. 147085 is equipped to carry four AAM-N-7 (Sidewinder IC) missiles. Two missile launcher-pylon combinations may be attached to each side of the fuselage forward of the wing. There are no provisions in this airplane for launching or jettisoning the Sidewinder missiles.

MISSILES.

Three AAM-N-6a missiles are carried on Aero 2A rail launchers on the left side, right side, and underside of the fuselage forward of the wing. For flight, the launchers are retracted and rotated so that the missiles are partly submerged into the fuselage to provide a reduced-drag aerodynamic configuration. The launchers are extended for firing as part of the automatic firing sequence. Any or all of the missiles may be jettisoned in emergencies by means of a jettisoning system that is activated by switches in the cockpit. The launcher actuators are powered by the utility hydraulic system. If that system fails, the launchers will remain locked in the position held at the time of failure.

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Sparrow III missiles are provided with two types of guidance units, semi-active continuous wave (CW) radar guidance or passive infrared (IR) homing guidance. Either type of missile may be loaded at any of the three launching stations; and with a mixed load the pilot may make a selection between firing either type or all missiles. The CW missile requires its target to be "illuminated" by continuous wave energy transmitted by the launching aircraft. The guidance unit in the missile receives the reflected energy from the target and employs it to steer an intercept course to the target. Sparrow III IR missiles employ a self-contained seeker to track radiant energy from a target.

AMCS AERO 1B.

The Airborne Missile Control System Aero 1B incorporates all of the equipment necessary for the detection and interception of airborne targets and for the target illumination required by CW Sparrow III missiles. The major components of the system are Airborne Intercept Radar AN/APQ-74, Radar Set Group AN/APA-128, Indicator Group AN/APA-138 and three Aero 2A rail launchers. To provide a completely integrated system for launching Sparrow III missiles, the airplane incorporates an air data computer, an angle-of-attack detector system, a radar control stick, and the switching and indicator circuits required to control missile launching.

AIRBORNE INTERCEPT RADAR AN/APQ-74. The airborne intercept radar provides modes of operation for radar navigation (mapping), target searching, and target tracking. The radar is operated through use of the controls on the radar control panel (figure 4-4) and the radar control stick (figure 4-5). In the mapping and search mode of operation, the pilot may select radar operating ranges of 100 or 200 nautical miles. For target range- and angle-tracking, radar operation at ranges of 24 or 12 nautical miles may be selected. Search and tracking information originated by the intercept radar is presented to the pilot on the cockpit radar indicator. During target tracking, the radar supplies range, closing velocity (range rate), angular rates, and position information to the computer section of the AN/APA-128 radar set group.

RADAR SET GROUP AN/APA-128. The radar set group accepts data from the airborne intercept radar and from the air data computer for use in computing lead pursuit course information that is presented to the pilot as the positioning of an aiming dot on the radar indicator display. The radar set group also supplies continuous wave energy for prelaunch tuning and for guidance of CW Sparrow III missiles.

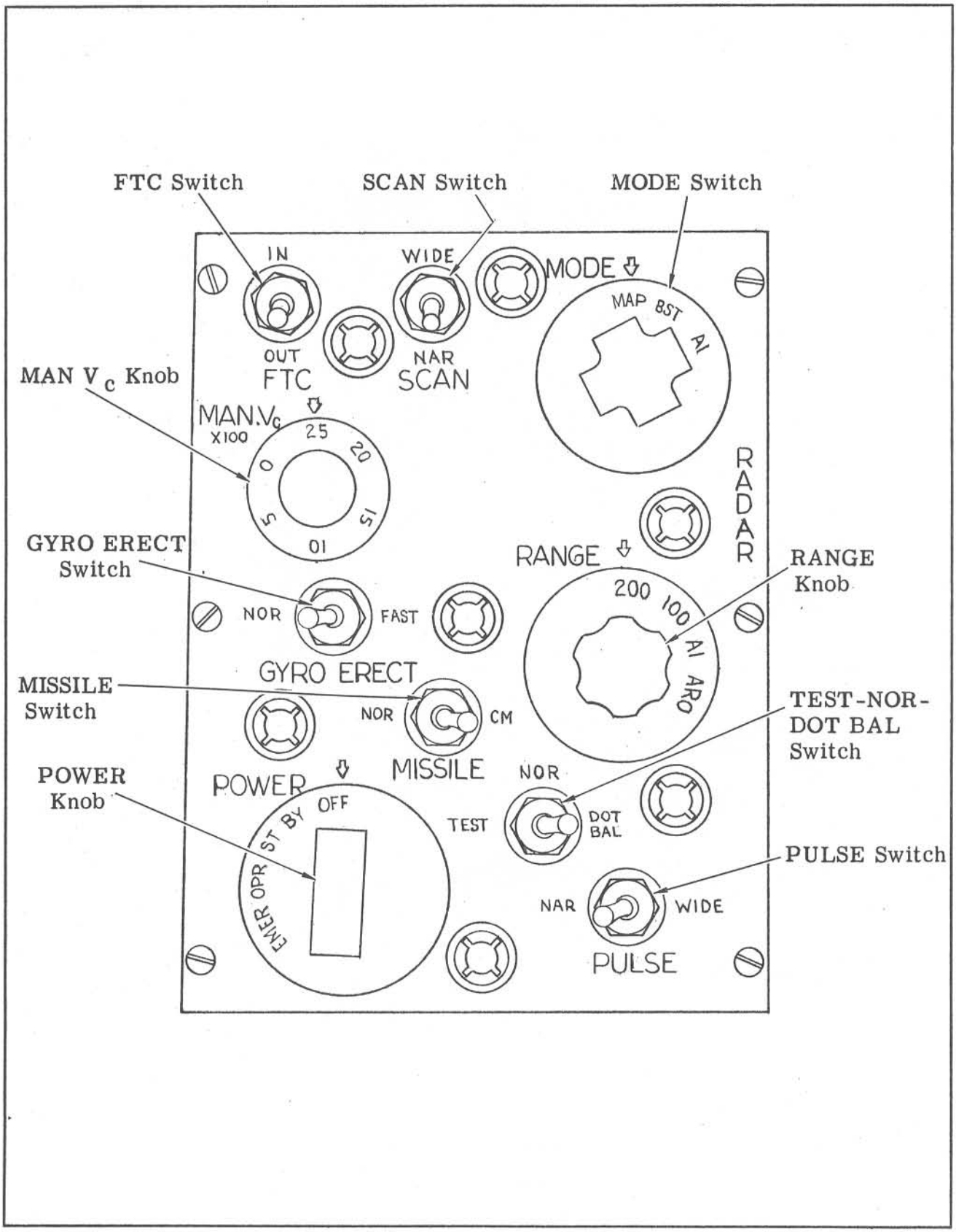


Figure 4-4 Radar Control Panel

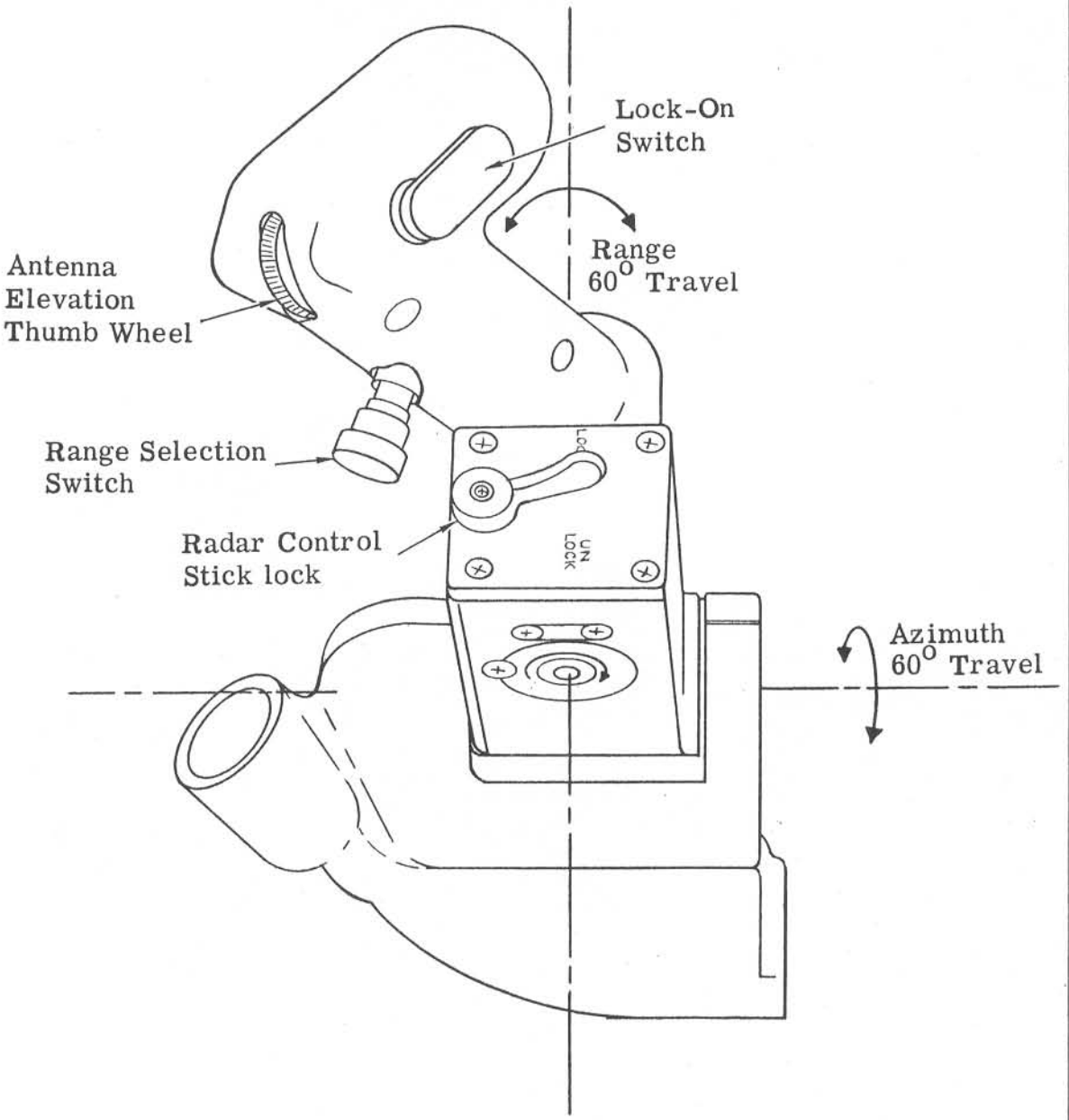


Figure 4-5 Radar Control Stick

Prelaunch tuning is required to ensure that the missiles will be captured and guided after they are launched. The radar set group transmitter illuminates the target and supplies a reference signal to the missile by radiating continuous wave energy through the antenna of the airborne intercept radar, which simultaneously tracks the target in range and azimuth.

INDICATOR GROUP AN/APA-138. The cockpit radar indicator provides the pilot with a visual presentation of search and tracking information from the airborne intercept radar and of steering information from the radar set group computer. The display, which includes a superimposed fixed reticle, is presented in two ways, directly through a magnifying lens in the lower portion of the indicator unit, or as a projected and further magnified image on a combining glass in the upper portion of the unit. The storage tube used in the radar indicator provides images of sufficient brightness and duration as to permit tracking without the use of a hood even in full sunlight. There are display modes for search and mapping, for attack, and for break-away.

Search Presentation. The search presentation consists of a boresight dot, an artificial horizon, an elevation strobe marker, an acquisition circle, and a B-scope display (figure 4-6). The B-scope display provides a radar "picture" of the sector being scanned by the radar and is produced on the screen by the B-sweep, which moves laterally across the presentation as the radar searches in azimuth. The vertical dimension of the "B" display represents range and the horizontal dimension represents azimuth. The artifical horizon provides the pilot with a continuous reference to the roll and pitch attitude of the airplane. The elevation strobe indicates the elevation position of the radar antenna and provides a reference for putting the antenna on-target in elevation. The acquisition circle represents radar azimuth and range and is used to acquire a target for lock-on from the search mode of operation. The radar control stick is used to "steer" the acquisition circle over a target echo, in range and azimuth, while the radar is still searching. Horizontal movement of the acquisition circle represents azimuth; vertical movement represents range. A thumb wheel on the radar control stick positions the elevation strobe.

Attack Presentation. The attack presentation, which appears automatically when lock-on occurs, consists of the artificial horizon, elevation marker, an azimuth strobe (collapsed B-sweep) with a range gate, an aiming dot, a boresight dot, and a range circle (figure 4-6). The aiming dot provides the pilot with a reference for steering the lead pursuit course computed by the radar set group for CW Sparrow III firing. The pilot flies the airplane toward the aiming dot by making corrections so as to bring the boresight dot over the aiming dot. The

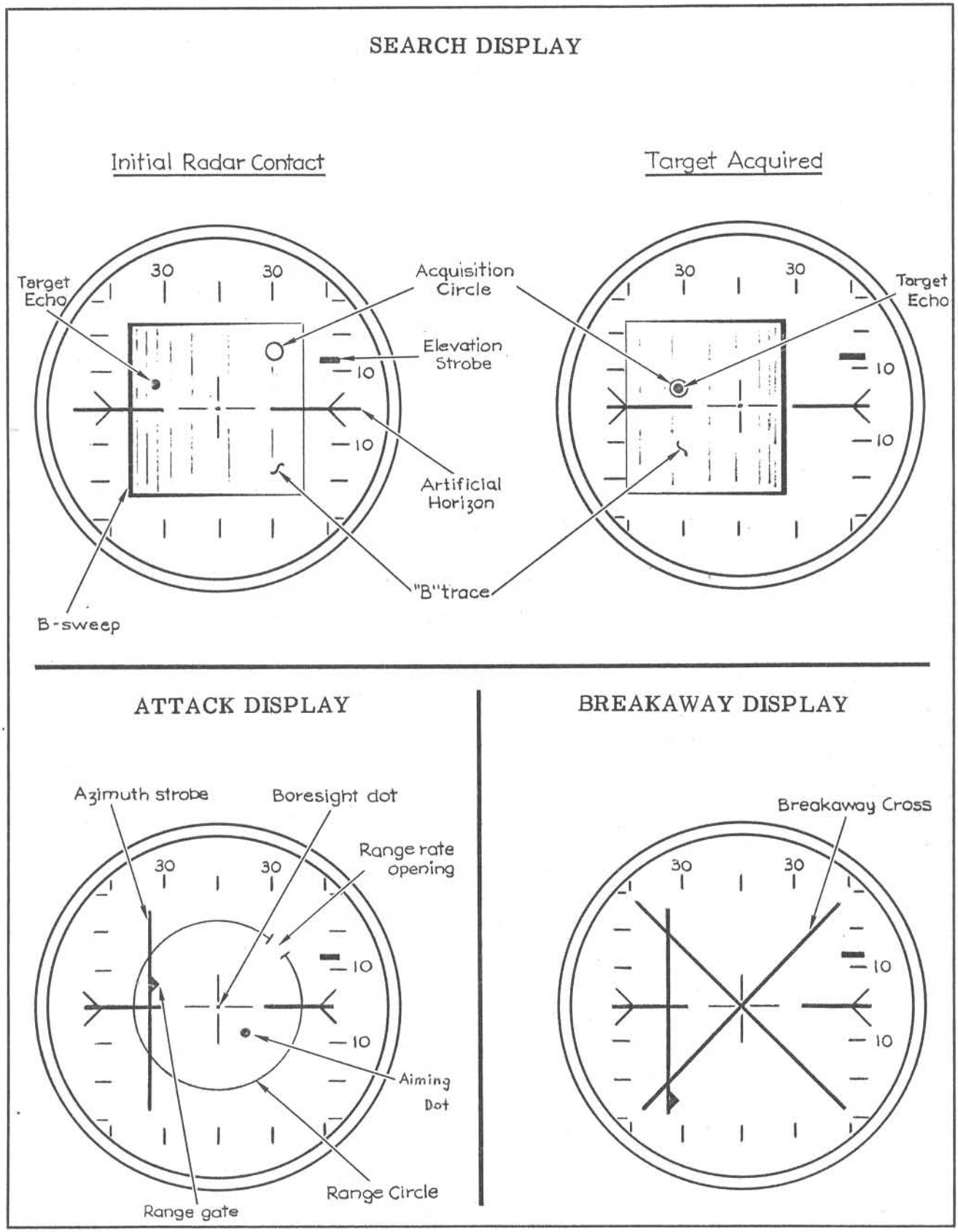


Figure 4-6 Radar Indicator Displays

range circle indicates range-to-go and range rate. The diameter of the circle decreases with decreasing range down to the minimum firing range. At minimum firing range, the circle is approximately 1/4 inch in diameter. The position of the opening in the range circle is an indication of range rate. When the opening is stationary at the twelve o'clock position, there is neither an opening nor a closing rate between the attacking airplane and the target. When the opening is positioned clockwise from twelve o'clock, there is a closing rate proportional to the displacement of the opening toward six o'clock, which represents a closing rate of 2,500 knots. Positions counterclockwise from twelve o'clock indicate opening rates up to 900 knots at the nine o'clock position. Movement of the opening indicates a change in the closing or opening rate of the two aircraft.

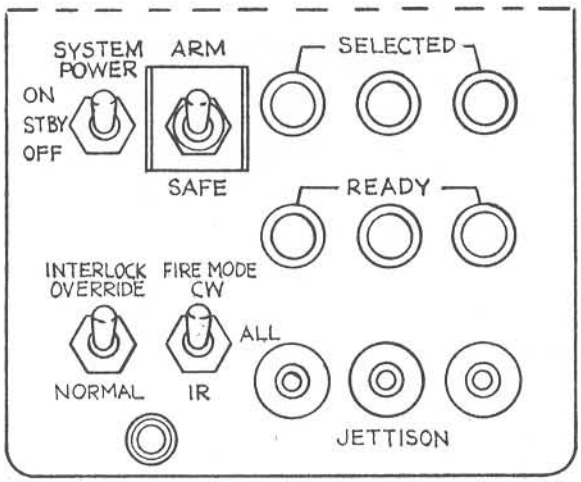
Break-Away Presentation. The break-away presentation, which appears when the airborne intercept radar is tracking a target but the range is less than the minimum missile firing range, consists of the artificial horizon, elevation marker, azimuth strobe, and a large "X" across the whole presentation (figure 4-6).

AIR DATA COMPUTER. The air data computer processes indicated angle of attack, static pressure, pitot pressure and total temperature to obtain data required for operation of the missile control system, duct bypass door control, and the automatic flight control system. The missile control system receives true airspeed, true angle of attack, altitude data functions, and altitude switching signals from the air data computer. True airspeed is required by the range interlock computer in the AN/APA-128 radar set group to determine maximum and minimum ranges for launching Sparrow III missiles. True airspeed, true angle of attack, and the altitude data functions are required by the lead angle computer in the radar set group to compute the proper lead angle for a missile intercept. The altitude switching signals automatically change the autopilot gain in the Sparrow missiles at 17,000, 32,000 and 46,000 feet. The air data computer receives indicated angle of attack data from a relative wind transducer on the right side of the fuselage. This information is used in the computer for computation of true angle of attack as a function of mach number.

CONTROLS AND INDICATORS.

SYSTEM POWER SWITCH. The SYSTEM POWER switch, which is located on the armament panel (figure 4-7), controls electrical power for the missiles and the missile control circuits. When the switch is in "OFF," the missiles receive no electrical power and are not tuned to the continuous wave (CW) radar frequency of the launching aircraft. The CW radar transmitter (radar set group) will be in standby condition if the airborne intercept radar is turned on. With the switch in "STBY," power is supplied to the missiles and to the firing circuits to permit the system to distinguish between IR and CW missiles. The missiles will not be tuned to the control radar frequency until the switch has been

PRODUCTION PANEL



FLIGHT TEST PANEL

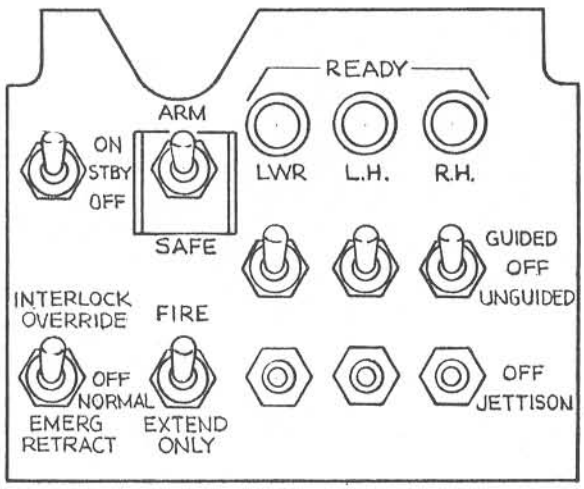


Figure 4-7 Armament Control Panels

placed in "ON." Placing the switch in "ON" permits the CW radar transmitter to operate and enables the guidance systems of the CW Sparrow III missiles to tune to the radar transmitter frequency. The missile guidance systems will remain tuned if the SYSTEM POWER switch is repositioned to "STBY," but the tuned condition may be lost if the switch is placed in "OFF."

ARMING SWITCH. The arming switch, which is located on the armament panel (figure 4-7), interrupts the trigger circuit to safeguard against inadvertent firing of the missiles. With the SYSTEM POWER switch in "STBY" or "ON" and the arming switch in "ARM," power is supplied through the landing gear door safety switches to the launching circuits of the selected missiles.

RANGE INTERLOCK OVERRIDE SWITCH (production). The range interlock override switch, which is located on the armament panel (figure 4-7), provides the pilot with a means for overriding the range interlock circuits. When the switch is in "NORMAL," the interlocks will permit the selected missiles to be launched only within their maximum and minimum range capabilities for the existing attack conditions. If the pilot squeezes the trigger switch while he is still outside the range limits, the missile will be launched only when the range comes within limits (IN RANGE light on). If the tactical situation warrants launching of missiles before the in-range signal is obtained, the interlock switch may be placed in "INTERLOCK OVERRIDE" and the missile may be launched at any time.

RANGE INTERLOCK OVERRIDE SWITCH (flight test). The flight test range interlock override switch, which is located on the flight test armament panel (figure 4-7), provides the same functions in the "OFF-NORMAL" and "INTERLOCK OVERRIDE" positions as does the interlock switch in the production configuration. However, the switch incorporates a third position, "EMERG RETRACT," which permits the pilot to retract any missile that has been extended but fails to be launched. In such a case, the missile could also be retracted by placing the SYSTEM POWER switch in "OFF," but there would be a lengthy delay before retraction.

FIRE MODE SWITCH (production). The FIRE MODE switch, which is located on the armament panel (figure 4-7), controls the selection of Sparrow III missiles to be launched. Electrical power is supplied to the switch when the SYSTEM POWER switch is positioned in either "STBY" or "ON." It is then possible to select for launching only CW missiles, only IR missiles or all missiles by positioning the switch in "CW," "IR," or "ALL." When "ALL" is selected, the launching sequence is (1) underside missile, (2) left missile, (3) right missile.

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FIRE - EXTEND ONLY SWITCH (flight test). In the flight test configuration, the FIRE - EXTEND ONLY switch, which is located on the flight test armament panel (figure 4-7), replaces the FIRE MODE switch. When the switch is placed in "FIRE," all three missiles are selected for launching. Placing the switch in "EXTEND ONLY" permits the missiles to be extended as for launching but with the launching circuits opened.

JETTISON SWITCHES. The three JETTISON switches located on the armament panel (figure 4-7), permit individual forward firing jettisoning of each missile in the event of a hangfire or other emergency. The switches are normally positioned in "OFF." When any of the switches is positioned to "JETTISON," power is routed through the landing gear safety relay (gear up) to extend the launcher and, when the launcher is fully extended, to launch the missile in an unarmed and unguided condition. The missilees cannot be jettisoned when operating on emergency electrical power.

EXPLOSIVE BOLT JETTISON SWITCHES (flight test). The EXPLOSIVE BOLT JETTISON switches on the left-hand console (figure 1-2) permit individual missiles to be jettisoned by means of an explosive bolt separation system that expels the missile outward from the fuselage. With the wing up and the landing gear up, a missile may be jettisoned at any time regardless of launcher position. With the wing down and the landing gear up, a missile may be jettisoned only when in the extended position.

STATION SELECTOR SWITCHES (flight test). The station selector switches, which are located on the flight test armament panel (figure 4-7), permit selection of missiles to be launched in either a guided or unguided condition. With the FIRE - EXTEND ONLY switch in "FIRE" and the arming switch in "ARM," placing the station selector switch for any station in "GUIDED" will provide a READY light indication if that station is loaded with an IR missile or with a CW missile that is tuned. When the READY light is on, the missile may be launched. Placing the switch in "UNGUIDED" will bypass all timing circuits. The READY light will come ON and the missile can be launched with no delay other than actual launcher extension time. The "OFF" position for any station causes that station to be bypassed in the launching sequence. With the FIRE - EXTEND ONLY switch in "EXTEND ONLY" and the arming switch in "ARM," placing the station selector switch for any station in either "GUIDED" or "UNGUIDED" will illuminate the READY light for that station and will permit the launcher at that station to be extended when the trigger is squeezed. The missiles will be extended in the normal firing sequence if all of the stations are selected. None of the missiles will retract until the interlock override switch is placed in "EMERG RETRACT" or system power is turned off.

READY AND SELECTED LIGHTS (production). The three green READY lights and the three red SELECTED lights, which are located on the armament panel (figure 4-7), provide indications of readiness for launching

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for each launching station. Combinations of light indications for each station indicate that missiles are ready and selected for launching as shown in table 4-2 .

TRIGGER SWITCH. The trigger switch, which is located on the flight control stick grip, provides the means for initiating the automatic launching sequence. Holding the trigger switch depressed with the arming switch in "ARM," closes the missile master firing relay (through the range interlock circuit). The selected launcher extends, the missile power unit starts, and a 1.1-second time delay relay begins clocking down. When the launcher is extended and the time delay relay actuates, the launcher lockpin is pulled and a voltage is applied to the firing squib to launch the missile. When the trigger switch is released, the firing circuit automatically switches to the next selected missile, which can be launched by again depressing the trigger switch.

RADAR CONTROL STICK. The radar control stick, which is mounted on the flight control stick housing (figure 4-5), provides a means for controlling airborne intercept radar antenna positioning. The stick is used to steer the acquisition circle in the radar indicator search display onto specific target echoes so that the antenna may be slewed on-target to obtain a lock-on. Fore and aft movements of the stick control the radar range gate position as reflected by vertical movement of the acquisition circle. Moving the stick forward increases the range and causes the acquisition circle to move upward. Lateral movements of the stick control the radar antenna azimuth position as reflected by horizontal movement of the acquisition circle. The elevation angle of the antenna is controlled by the elevation thumb wheel on the stick and is reflected by vertical movements of the elevation strobe on the radar indicator. The stick mounts a lock-on switch which controls antenna slewing. When the stick is moved with the lock-on switch not depressed, the acquisition circle on the radar indicator is repositioned, but the antenna continues to search. When the lock-on button is depressed, the antenna will slew to the position indicated by the acquisition circle and the radar will lock on the target as evidenced by the appearance of the attack display. The switch may be depressed and held while the stick is used to position the antenna to scan any desired sector. The range switch on the stick permits the pilot to select long- or short-range presentation in both the search and attack modes of operation. A manually operated lock is provided for locking the radar control stick in the non-operating position.

RADAR INDICATOR CONTROLS. The display on the cockpit radar indicator (figure 4-8) is regulated by the pilot through use of controls mounted at the lower edges of the indicator. The visual quality of the display is adjusted by the BRIGHTNESS knob, CONTRAST knob, REC GAIN (receiver gain) knob, B INT ("B" trace intensity) knob, reticle LAMP DIMMER knob, and STORAGE knob. The STORAGE knob permits the persistence of the image on the radar indicator screen to be varied and may be pushed in to obtain

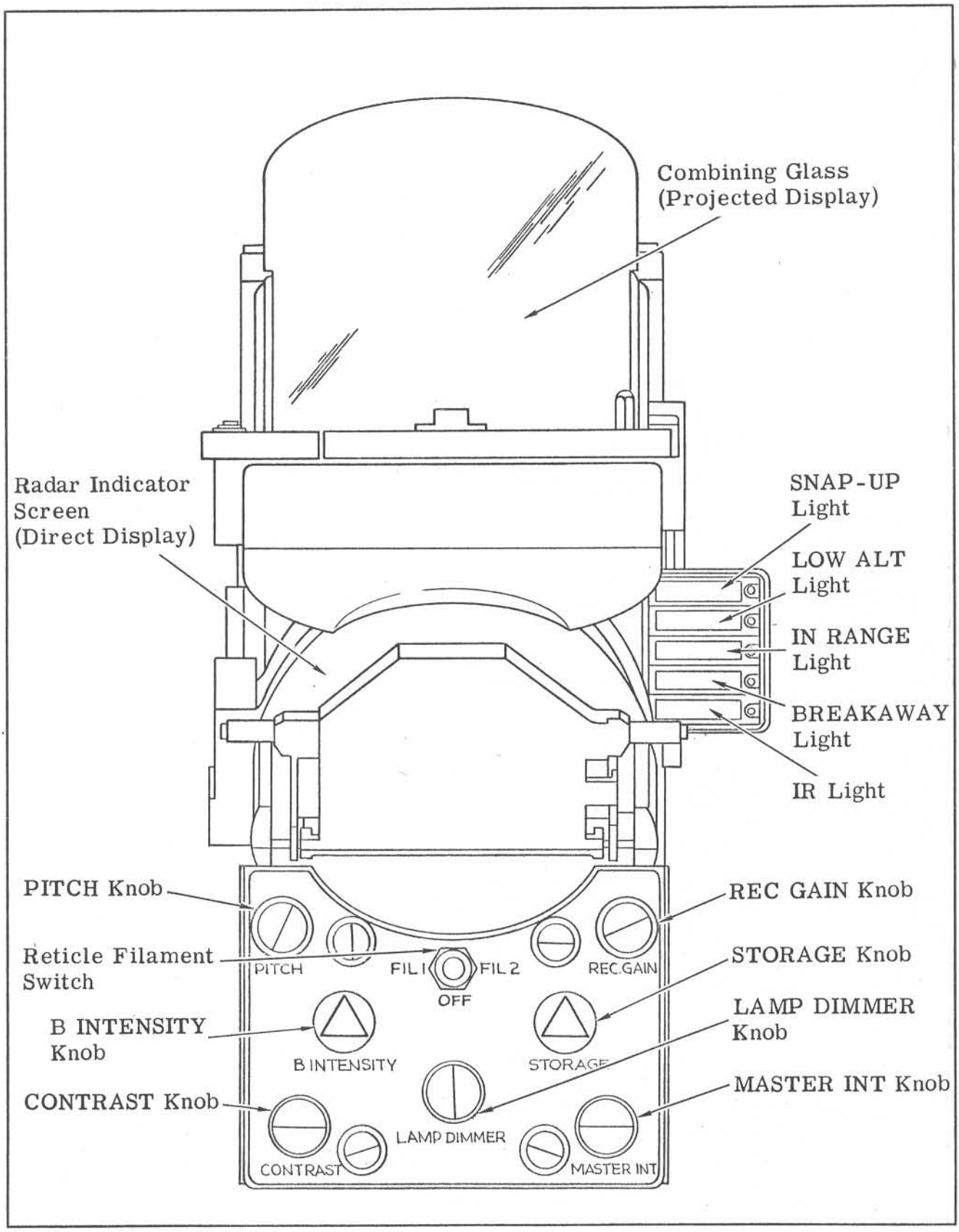


Figure 4-8 Cockpit Radar Indicator

Table 4- 2 . Missile READY and SELECTED Light Indications (PRODUCTION).

Note: Information applies with SYSTEM POWER switch in "ON."

Missile Arming Switch Position	FIRE MODE Switch Position	READY Light Indication	SELECTED Light Indication	Missile Status
	ANY STATION WITH CW SPARROW III MISSILE LOADED			
Off	"CW" or "ALL"	Off	On	Missile ready (tuned to frequency of missile control radar) and selected for launching.
Off	"IR"	On	Off	Missile ready but not selected for launching.
"ARM"	"CW" or "ALL"	On	On	Missile ready and selected for launching.
"ARM"	"IR"	On	Off	Missile ready, but not selected.
	ANY STATION WITH IR SPARROW III MISSILE LOADED			
Off	"IR" or "ALL"	Off	On	Missile ready to launch and selected.
Off	"CW"	On	Off	Missile ready but not selected.
"ARM"	"IR" or "ALL"	On	On	Missile ready to launch and selected.
"ARM"	"CW"	On	Off	Missile ready but not selected

Note

Indications in the table apply for CW missiles when the SYSTEM POWER switch is in "STBY," providing that the switch has previously been placed in "ON" to permit the missile to tune to the proper radar frequency.

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instantaneous erasure of the image. A PITCH knob is provided for adjusting the artificial horizon reference for changes in angle of attack. A switch is provided for selecting alternate filaments of the reticle lamp or for turning the lamp off.

Five indicator lights are located on the right side of the cockpit radar indicator. The IN RANGE light comes on to indicate to the pilot that he is within the range envelope for the type of missile (CW or IR) that is selected and ready. The SNAP UP light comes on to indicate when the proper range is reached (for the type of missile that is selected and ready) for making a snap-up launching. The BREAKAWAY light comes on to indicate when to initiate a breakaway maneuver. The LOW ALT light indicates when the airplane is operating at a dangerously low altitude. The IR light is not used.

INTERCEPT RADAR CONTROLS. A panel on the left-hand console (figure) provides controls for governing the operation of airborne intercept radar AN/APQ-74. For details on operation of the airborne intercept radar refer to Westinghouse Electric Corporation's Brief Operating Instructions for Radar Set AN/APQ-74.

NORMAL PROCEDURES.

PREFLIGHT CHECK. The following functional check should be performed before flight to ensure proper operation of the airborne missile control system. Proceed as follows:

Radar POWER switch	"OFF"
SYSTEM POWER switch	"OFF"
Radar indicator controls	Turned fully counterclockwise

CAUTION

Failure to set indicator controls counterclockwise before supplying power to the system may damage the indicator screen when power is applied.

Check that external electrical power and cooling air are connected to the airplane and verify that cooling air is flowing through radar equipment.

MODE switch	"AI"
FTC switch	"OUT"
MAN Vc knob	Zero
GYRO ERECT switch	"NOR"
SCAN switch	"WIDE"
PULSE switch	"NAR"

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TEST switch . . . . . "NOR"
MISSILE switch. . . . . "NOR"
Missile arming switch . . . . . "OFF"
INTERLOCK OVERRIDE switch . . . . . "NORMAL"
Radar POWER switch. . . . . "OPR"
SYSTEM POWER switch . . . . . "ON"

Allow the system to warm up for approximately 5 minutes, then adjust indicator controls for desired visual quality of the search presentation.

WARNING

Ensure that personnel remain well clear of the radome area while the radar transmitter is energized. If possible, maintain radar antenna elevation at highest setting. Serious injury can result from prolonged exposure to rf energy.

Radar control stick lock. . . . . "UNLOCK"

Move stick forward and aftward and verify that indicator acquisition circle tracks in range.

Move stick right and left and verify that acquisition circle tracks in azimuth.

Rotate elevation thumb wheel and check that elevation strobe tracks properly.

If a suitable moving target is available, depress the radar stick lock-on switch and track target with acquisition circle. When acquisition circle is on-target, release lock-on switch and verify that radar locks on and tracks target. Check for proper indicator presentation (attack). Vary the settings of the FIRE MODE switch with the missile arming switch in both the off and "ARM" positions and check for proper missile status indications according to table 4-2.

Note

The missile launching circuits are disabled when the airplane is on the ground. The missile arming switch may be placed in "ARM" to check missiles without energizing the launching circuits.

INFLIGHT CHECK. The missile control system should be rechecked following takeoff by repeating the procedure employed in the preflight check. The arming switch may be placed in "ARM" to check the READY and SELECTED indications as in the preflight check.

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WARNING

If another aircraft is tracked to check radar lock-on, the arming switch must be placed in "SAFE" to guard against an inadvertent launching should the in-range condition exist and the trigger switch be depressed.

Note

The arming switch controls the launching system. The missile warheads are armed only after the missiles are launched and are closing on-target.

After the system is checked, readiness for firing may be maintained, with the radars off the air, by positioning controls as follows:

Arming switch	"ARM"
Radar POWER switch	"STBY"
SYSTEM POWER switch	"STBY"
FIRE MODE switch	Set for type of missile desired

INTERCEPT AND ATTACK.

When it is desired to begin searching for a target, the missile control system can be brought from standby to full operation immediately by positioning switches as follows:

Radar POWER switch	"OPR"
SYSTEM POWER switch	"ON"

When the B-sweep "paints" a target trace on the radar indicator, position the acquisition circle over the target and continue to track the target while adjusting the elevation wheel for the best target return. When the target is within lock-on range, depress the lock-on switch. When the lock-on switch is released the indicator should show an attack presentation, indicating that the radar has locked on and is tracking the target.

Note

If it is desired to search a sector other than directly ahead, the lock-on switch may be depressed and the radar antenna may be positioned manually with the radar stick control to search the desired sector.

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Maneuver the airplane to bring the boresight dot over the aiming dot. When the IN RANGE light comes on, the first selected and ready missile may be launched by depressing the trigger switch and holding it depressed for approximately 1 second. The second and third selected missiles may be launched by depressing and releasing the trigger switch to fire each missile. When a CW missile has been launched, the radar must continue to track the target until the missile reaches the target. The pilot may break away from the lead pursuit course he established by flying onto the aiming dot, but he must fly within the tracking limits of the radar antenna. When an IR missile has been launched, it is not necessary to continue tracking the target.

INTERIOR LIGHTS.

GENERAL.

The interior light system comprises conventional instrument lights and console floodlights, emergency cockpit floodlights, and high-altitude cockpit floodlights. The instrument light and console floodlight controls are located on the INTERIOR LIGHTS panel on the right-hand console. The INSTR LIGHTS knob (30, figure 1-4) and the CONSOLE LIGHTS knob (31, figure 1-4) permit the normal interior lights to be turned on and lighting intensity to be varied. The FLD LTS switch (32, figure 1-4) permits different intensities of console flood lighting to be selected.

Dimming of the warning lights is controlled by INSTR LIGHTS and CONSOLE LIGHTS control knobs. When both the instrument and the console lights are turned on, the warning light dimming relays are energized and all warning and indicator lights are dimmed.

KNEEBOARD AND EMERGENCY LIGHTS.

The EMER LIGHTS - KNEEBOARD LTS switch (21, figure 1-4) controls both the kneeboard lights and the emergency instrument board floodlights. In the EMER LIGHTS position, the emergency floodlights are turned on. Positioned in the "OFF - BRT" range, the switch permits the kneeboard lights to be turned on and intensity to be varied. Both the kneeboard lights and the emergency lights are powered directly from the emergency dc bus.

HIGH-ALTITUDE LIGHTS.

Two high-altitude floodlights are mounted to project diffused white light over both shoulders of the pilot. These lights are controlled by the HIGH ALT LIGHTS switch and are powered directly from the emergency dc bus.

Note

The high-altitude lights can be used in an emergency after failure of normal interior lights. However, use of high-altitude lights will cause canopy and windshield glare.

EXTERIOR LIGHTS.

GENERAL.

This airplane is equipped with conventional position lights and with formation lights (strip lights) on both sides of the fuselage and on each wing. Exterior lights are controlled by the EXT LIGHTS switch (13, figure 1-2): the FUS, TAIL and WING light switches (18, 7, 10, figure 1-4), the STRIP LIGHTS switch (13, figure 1-4), and the MAN KEY button on the rail above the right-hand switch panel.

The EXT LIGHTS switch has positions labeled "STEADY," "OFF," and "FLASH." When this switch is in "STEADY," all lights selected with the FUS, WING, TAIL and STRIP LIGHTS switches burn steadily. When the master switch is in "FLASH," all lights selected will flash. When the master switch is in "OFF," the fuselage position lights will burn when keyed by the MAN KEY button.

The FUS, TAIL, WING and STRIP LIGHTS switches control selection and intensity of the wing and tail lights and the fuselage lights. The fuselage lights burn any time the EXT LIGHTS switch is in a position other than off. All exterior lights except the wing tip lights are powered from the secondary dc bus. The wing tip lights are powered from the emergency dc bus and will operate on emergency electrical power.

TAXI LIGHT.

This airplane is designed to be equipped with a taxi light mounted on the nose gear. This light will be controlled by the TAXI LIGHTS switch (9, figure 1-4). The taxi light is inoperative on airplane BuNo. 147085.

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APPROACH LIGHTS.

The three approach lights mounted on the nose gear flipper door provide the landing signal officer with indications of airplane pitch attitude in the landing approach. The green, red and amber lights are switched on by the angle-of-attack indicating system in the proper sequence to indicate high, low, and desired angle of attack. For carrier landings, the approach lights will be lighted when the landing gear and the arresting hook are extended. A switch in the nose wheel well permits approach light operation with the hook retracted for field carrier landing practice. (Refer to "Angle-of-Attack Indicating System" in section I for additional information on approach light operation.)

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SECTION V

OPERATING LIMITATIONS

Note

The latest revision of Enclosure 1 to Engineering Memorandum E8M-1796, comprising operating limitations and flight restrictions, will be supplied at a later date and will be supplemented and revised as necessary during the flight test program.

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SECTION VI

FLIGHT CHARACTERISTICS

CONTENTS

General 6-2
Stalls 6-2
Spins 6-4
Flight Controls 6-4

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GENERAL.

The F8U-3 is a high-performance, carrier- or land-based all-weather fighter airplane. The two-position (variable incidence) wing makes possible low fuselage attitudes with maximum visibility for carrier operation. Low stalling speed and a high level of controllability in slow flight are obtained by use of double-hinged leading edge droops, and blowing boundary-layer control over the aileron and flaps. The ailerons droop along with the flaps in the wing-up position to provide high lift. The wing also incorporates spoilers to increase the rate of roll in the clean configuration. A speed brake is provided on the belly of the fuselage, operable in all clean flight conditions. Flight control surfaces are operated by dual independent hydraulic power control systems. Pitch, yaw, and roll damping and trimming, aileron-rudder interconnect, and auto pilot are provided by the automatic flight control system.

STALLS. (Refer to figure 6-1 for stalling speeds.)

GENERAL.

There are no design indications that unusual or dangerous flight characteristics will be present when the airplane stalls. It is expected that the F8U-3 will have the same type of stall characteristics as earlier F8U series airplanes. Positive control of the airplane is expected to be available under all stall conditions and there are no forces apparent which will cause an abrupt loss of control. Stall recovery is effected by conventional methods, similar to the F8U-1 airplane.

LANDING CONFIGURATION STALLS.

The airplane will remain reasonably stable at the stall with positive horizontal tail effectiveness. Response to aileron movement will be reduced, but lateral control can be maintained by rudder movement. The stall may be somewhat sharper than in the F8U-1 but should be similar to the F8U-1 BLC airplane.

CLEAN AND CRUISE CONFIGURATION STALLS.

The clean and cruise configuration stalls are expected to be similar to those of the F8U-1 airplane. Stalls should be preceded by moderate buffet at approximately the same lift as the F8U-1 airplane. Care should be taken not to make abrupt or excessive aileron movements at the stall.

ACCELERATED STALLS.

Accelerated flight stalls are expected to be preceded by moderate buffet. No change in flight characteristics from the F8U-1 airplane is expected during accelerated stalls. Care should be taken not to make abrupt or excessive aileron movements.

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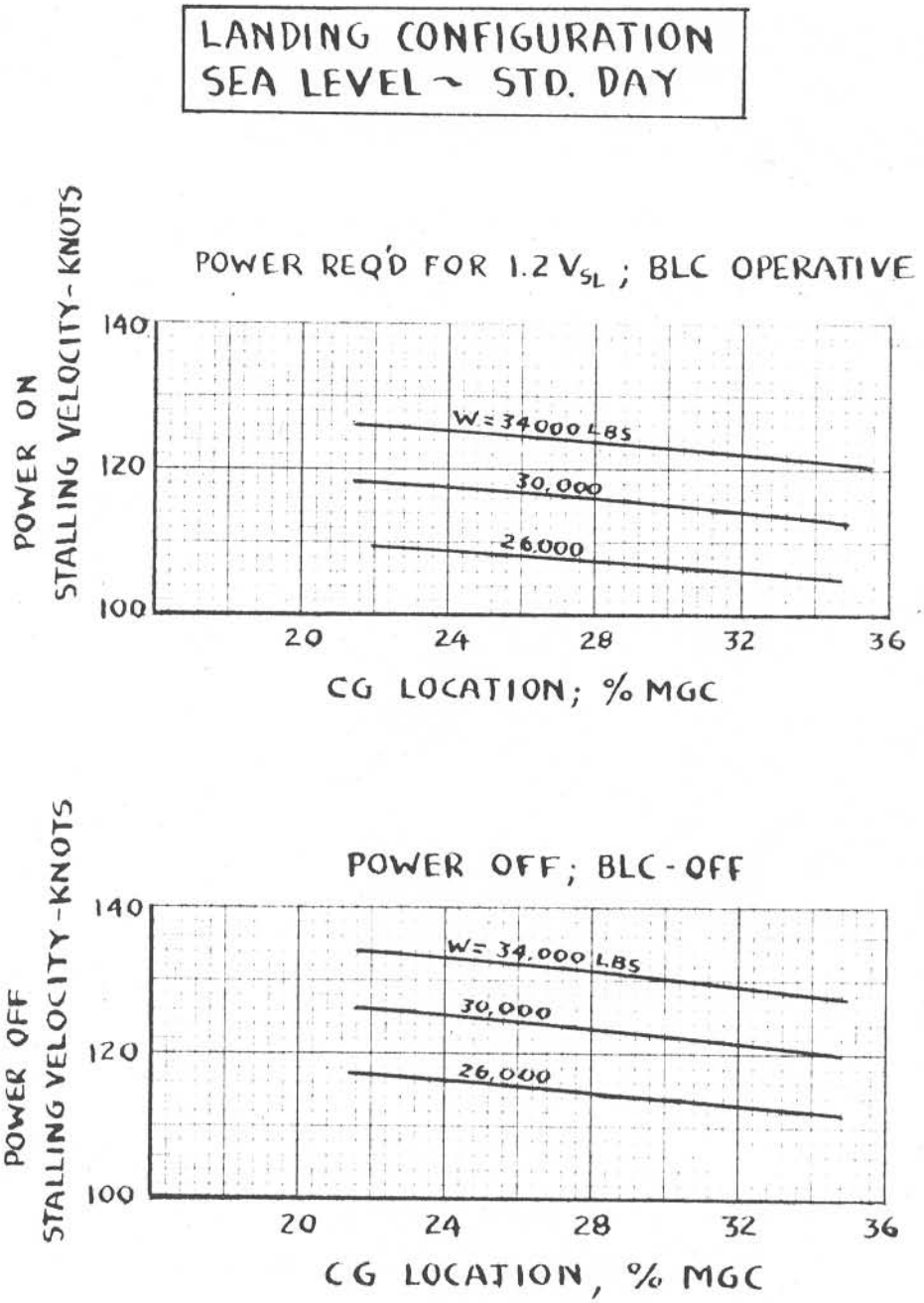


Figure 6-1. Stalling Speeds

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SPINS.

GENERAL.

WARNING

Intentional spinning of the airplane for other than flight test purposes is prohibited.

Wind tunnel model test results show no tendency of the airplane to spin. However, because of the low wind tunnel test Reynolds numbers, the wind tunnel test results may not give final indications of spin characteristics. It is felt that if the airplane were to spin, it would have an oscillatory-type spin similar to the F8U-1 airplane. It is important to recognize and stop pre-spin indications before they are fully developed.

If the recovery procedure is followed quickly at the early indications of a spin, rapid recovery will be possible.

SPIN PREVENTION.

To prevent developing spin flight characteristics if a wing drop occurs at the stall, the ailerons should be deflected with the wing drop. Aileron deflection should be followed by forward stick movement to reduce the angle of attack.

SPIN RECOVERY.

If the airplane is entering a spin, or is in a spin, rapidly deflect the ailerons with the spin while holding approximately 2/3 aft stick travel, and holding full opposite rudder. As rotation ceases, quickly neutralize ailerons and rudder and ease stick forward. This procedure should be effective for either the landing or clean configuration.

INVERTED SPIN.

Model tests show no tendency of the airplane to spin from inverted flight. Should such a spin occur, it is expected that recovery can be made by applying opposite rudder.

FLIGHT CONTROLS.

AILERONS AND SPOILERS.

Lateral control is provided through ailerons and spoilers. The spoilers are directly slaved to the aileron control system, and are actuated by aileron travel above clean condition neutral to increase the rate of roll. The ailerons also are automatically drooped to act as additional flaps when the wing is raised for takeoff or landing. Stick forces for full aileron displacement are approximately the same in the landing condition as in the clean condition. Aileron travel with the airplane in the

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clean condition is restricted to 50 per cent of full displacement by stops which can be overridden by applying 4 pounds additional stick force. The control stick will remain in the neutral position for all roll trim settings. In the event of malfunction of the roll trim and stabilization system, the roll actuator will return to neutral and lock. Aileron control remains normal, with the actuator serving as a simple idler in the control system.

WARNING

With airplane in the landing condition, lateral stability is marginal at higher airspeeds without yaw and roll damping. If roll and yaw damping fail under these conditions, abrupt maneuvers may result in the airplane becoming uncontrollable. If lateral instability is noted, correct the condition by reducing airspeed, increasing angle of attack, or returning to clean condition.

NOTE

Refer to AUTOMATIC FLIGHT CONTROL SYSTEM, section I, for flight characteristics when autopilot is engaged.

HORIZONTAL TAIL.

Longitudinal control of the airplane is provided by a unit horizontal tail. Control in the takeoff and landing speed ranges is excellent and stick forces are relatively light. Stick movements are positive for all normal flight operation. Pitch trim adjustment trims out the stick force with no change in stick position. Runaway of a trim actuator will not result in reduction of the available horizontal tail deflections. The emergency trim system may be used to neutralize horizontal tail trim when a trim actuator runaway occurs. However, if the pitch trim actuator fails, emergency pitch trim will be ineffective. If the stick grip electrical trim system fails, emergency pitch trim may be used to neutralize horizontal tail trim. When wing position is changed, the horizontal tail is automatically repositioned by mechanical linkage to compensate for changed trim requirements. In this case, the control stick neutral position will not change. In event of malfunction of the pitch stabilization system, the pitch damping actuator will return to neutral and lock. Longitudinal control remains normal, with the actuator serving as a simple idler in the system.

WARNING

Avoid abrupt horizontal tail movements with low fuel aboard and aft cg location, at 0.8 to 0.95 IMN at altitudes of 25,000 feet or less, to prevent over control. Do not exceed negative g loading in excess of -2g's above 30,000 feet, or -1g above 40,000 feet at 0.9 to 1.0 MN to avoid abrupt pitch-down tendency.

NOTE

Refer to AUTOMATIC FLIGHT CONTROL SYSTEM, section I, for flight characteristics when auto pilot is engaged.

RUDDER.

Directional control is provided by a conventional rudder. When the airplane is in the clean condition, rudder stops limit the travel. The stops are disengaged and full rudder travel is available when the wing is raised for landing. Rudder pedal force with the airplane in the clean condition is approximately double that for the landing condition for the same rudder pedal travel. The rudder pedals will remain in the neutral position for all trim conditions. In the event of malfunction of the yaw trim and stabilization system, the yaw actuator will return to neutral and lock. Rudder control remains normal, with the actuator serving as a simple idler in the control system.

NOTE

Refer to AUTOMATIC FLIGHT CONTROL SYSTEM, section I, for flight characteristics when auto pilot is engaged.

FLAPS.

The flaps provide additional lift at takeoff and landing. When the wing is raised for landing or takeoff, the flaps are automatically lowered.

VENTRAL FINS.

In high-speed flight, the ventral fins are down to provide additional directional stability, and on takeoff or landing, the fins are rotated to a horizontal position. The ventral fins operate in conjunction with the wing. When the wing is raised, the fins rotate to horizontal position, and when the wing is lowered, the fins are rotated to the vertical position. Emergency raising of the wing will also raise the fins, and for flight test clean condition landings, a separate pneumatic emergency system allows the pilot to raise the ventral fins alone.

SPEED BRAKE.

The speed brake is used to control airplane speed in dives and maneuvering flight. Trim changes resulting directly from speed brake deflection are expected to be moderate at all airspeeds. A relief valve in the hydraulic supply system prevents excessive airloads from being exerted on the extended brake. When the wing is raised, the speed brake is automatically closed to prevent landing with an open brake. On the ground, aircraft weight on the landing gear actuates a switch to prevent brake opening with the wing down.

LEADING EDGE DROOP.

Cruise droop position of the wing leading edge is used for additional stability and control during climbs and maneuvers, and is used to increase the cruising range and buffet onset boundary of the airplane. A slight nose-down tendency will result from extending to cruise droop. Cruise droop shall not be extended below 15,000 feet in excess of 550 knots EAS. Landing droop is automatic with raising of the wing and is used only for takeoff or landing.

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APPENDIX I

OPERATING DATA CHARTS

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Description of Charts ---------- A-2
Operating Data Charts
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Takeoff Distance - Military Thrust ---------- A-5
Climb - Maximum Thrust ---------- A-6
Climb - Military Thrust ---------- A-7
Climb - Optimum Airspeed ---------- A-8
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Nautical Miles per 1,000 Pounds Fuel ------- A-11

NOTE

Charts in this appendix are labeled J-75-P5A (same as J75-P5N) engine. Operations with the J-75-P6 engine involve only minor differences in fuel consumption.

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GENERAL.

This appendix contains data from which performance of the airplane can be predicted. The information is not only useful for flight planning but also serves to illustrate the effects of many variables which affect the airplane's performance. The landing gear, wing, and wing leading edge configurations used to qualify various charts are listed below:

Configuration	Description
Clean	Landing gear retracted, wing down, leading edge up
Cruise	Landing gear retracted, wing down, leading edge up in cruise droop position
Landing	Landing gear extended, wing up, leading edge in landing droop position

Dashed-arrow notations on the charts illustrate the procedure for using the graphs.

DESCRIPTION OF CHARTS.

The following paragraphs describe the charts that cover takeoff, climb, landing, and cruise control. Cruise control charts are in graphical form to facilitate flight planning.

TAKEOFF DISTANCE.

These charts (figures A-1 and A-2) provide ground-roll distance and airspeed required for takeoff at various gross weights with maximum or military thrust. Separate charts are provided for military thrust and maximum thrust. Information is based on the airplane in the takeoff condition with takeoff performed on a hard surface runway. The distance to clear a 50-foot obstacle is approximately 155% of ground roll ("MILITARY" thrust) or 135% of ground roll ("MAXIMUM" thrust). To determine takeoff ground roll, enter the temperature - altitude correction chart at the left with field temperature. Read upward to the field altitude, then across to the gross weight of the airplane. Read downward to the bottom of the gross weight chart and to the crosswind correction chart. Read downward along the ground roll curve to the existing headwind, then read downward to the take-off ground roll scale.

CLIMB.

The climb charts (figures A-3 through A-5) give calibrated airspeeds, time, distance traveled, and fuel used for climbs to various altitudes with military or maximum thrust. To use the time - distance - fuel chart, enter the bottom of the chart with the gross weight of the airplane, making allowance for difference of temperature from standard day. Read along the guide

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line to desired altitude. To determine time and distance traveled, read straight across to the scales on the left. To determine fuel used, read vertically downward to obtain a new gross weight figure. Subtract the new gross weight from the gross weight used to enter the table; the difference is the fuel used in pounds. To use the optimum speed climb chart enter at the left with pressure altitude, read across to the appropriate power curve, then read vertically downward to determine CAS.

LANDING GROUND ROLL.

This chart (figure A-6) gives the ground distance used in landing roll-out for various gross weights, density altitudes, and headwinds. In addition, sheet 2 gives the touch-down airspeed and the maximum speed at which brakes are to be applied to make good the calculations of sheet 1. For example, enter the chart at the temperature - altitude correction scale and project vertically, field elevation. Project horizontally to the gross weight curve, downward to the headwind curves, along the curves to the observed headwind line, then downward to the ground-roll scale. To find the distance needed to clear a 50-foot obstacle, add 1,000 feet to the ground-roll distance. The touch-down airspeed is obtained by entering the top chart of sheet 2 at the assumed gross weight, projecting vertically to the curve, and then horizontally to the left to read airspeed. To obtain the maximum speed for application of brakes, enter the lower chart of sheet 2 at the density altitude (obtained from sheet 1), project horizontally to the landing-gross-weight curve, then downward to the headwind curve. Projecting downward along the headwind curve to the headwind value, then vertically downward to the speed scale, gives the speed at which the brakes first may be applied without damage to the brakes or tires.

NAUTICAL MILES PER 1,000 POUNDS FUEL.

The nautical miles per 1,000 pounds fuel charts (figures A-7 through A-12) give the cruise distance obtained per 1,000 pounds of fuel and the rate of consumption for various gross weights, altitudes, and airspeeds. The charts also give the throttle setting necessary to make good a desired airspeed. To obtain cruise conditions for maximum range, select the chart for the desired cruise altitude and enter at the recommended airspeed line. Where this line intersects with the airplane gross weight, read horizontally across to the distance scale at the left to obtain nautical miles per 1,000 pounds fuel consumed; read downward from the intersection to obtain calibrated airspeed, true airspeed, and Mach number. To find the rate of fuel consumption, interpolate between the consumption rate curves.

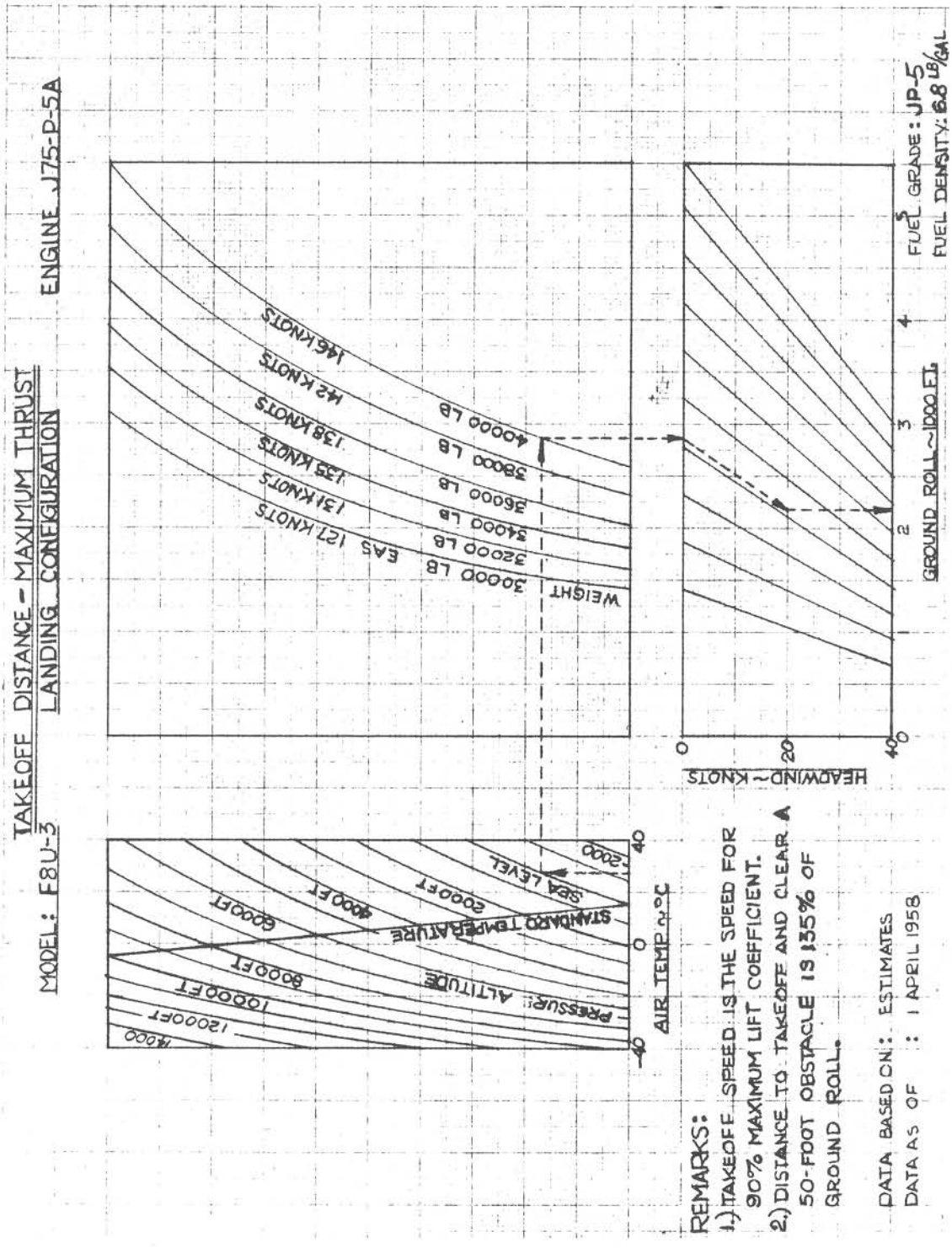


Figure A-1. Takeoff Distance - Maximum Thrust

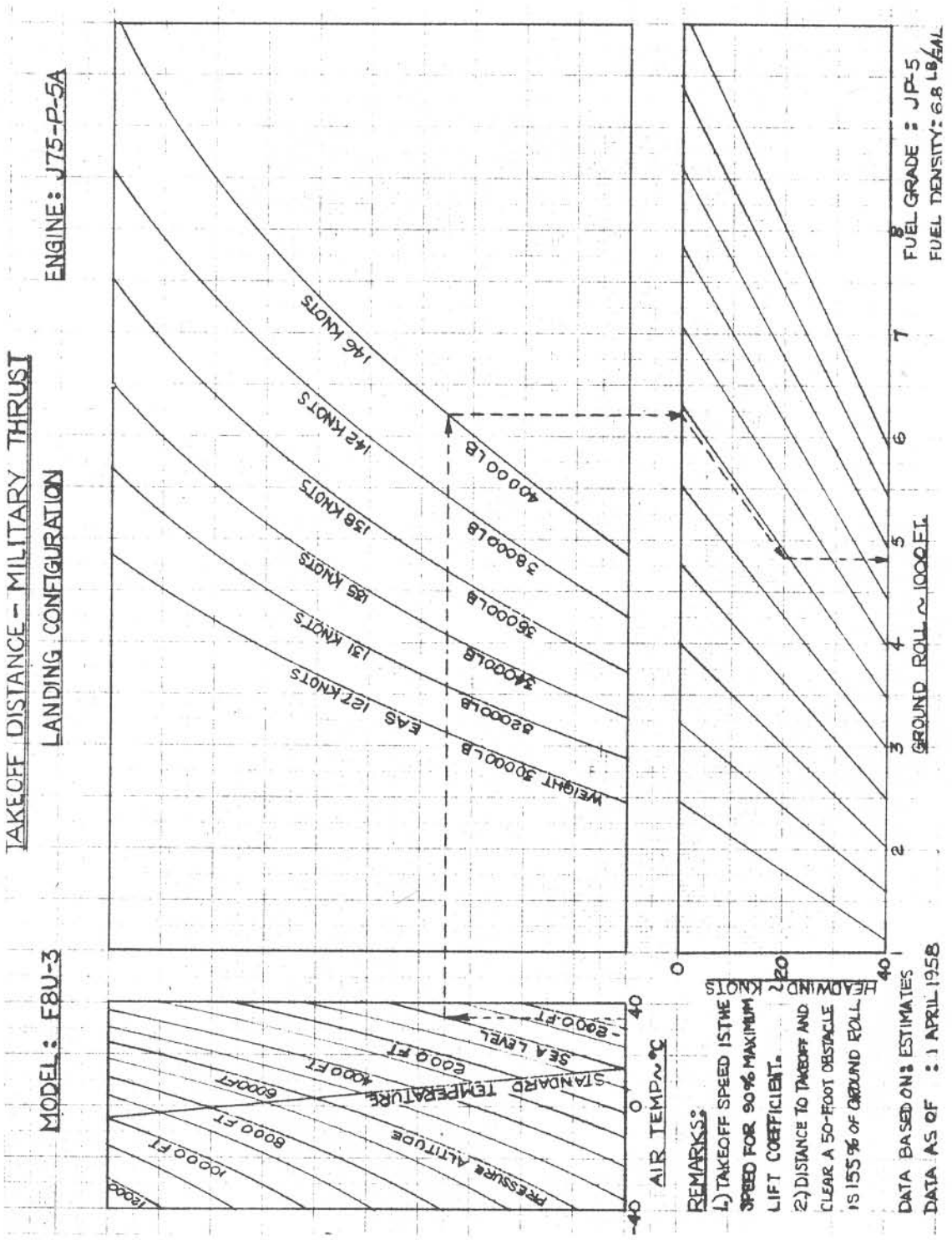


Figure A-2. Takeoff Distance - Military Thrust

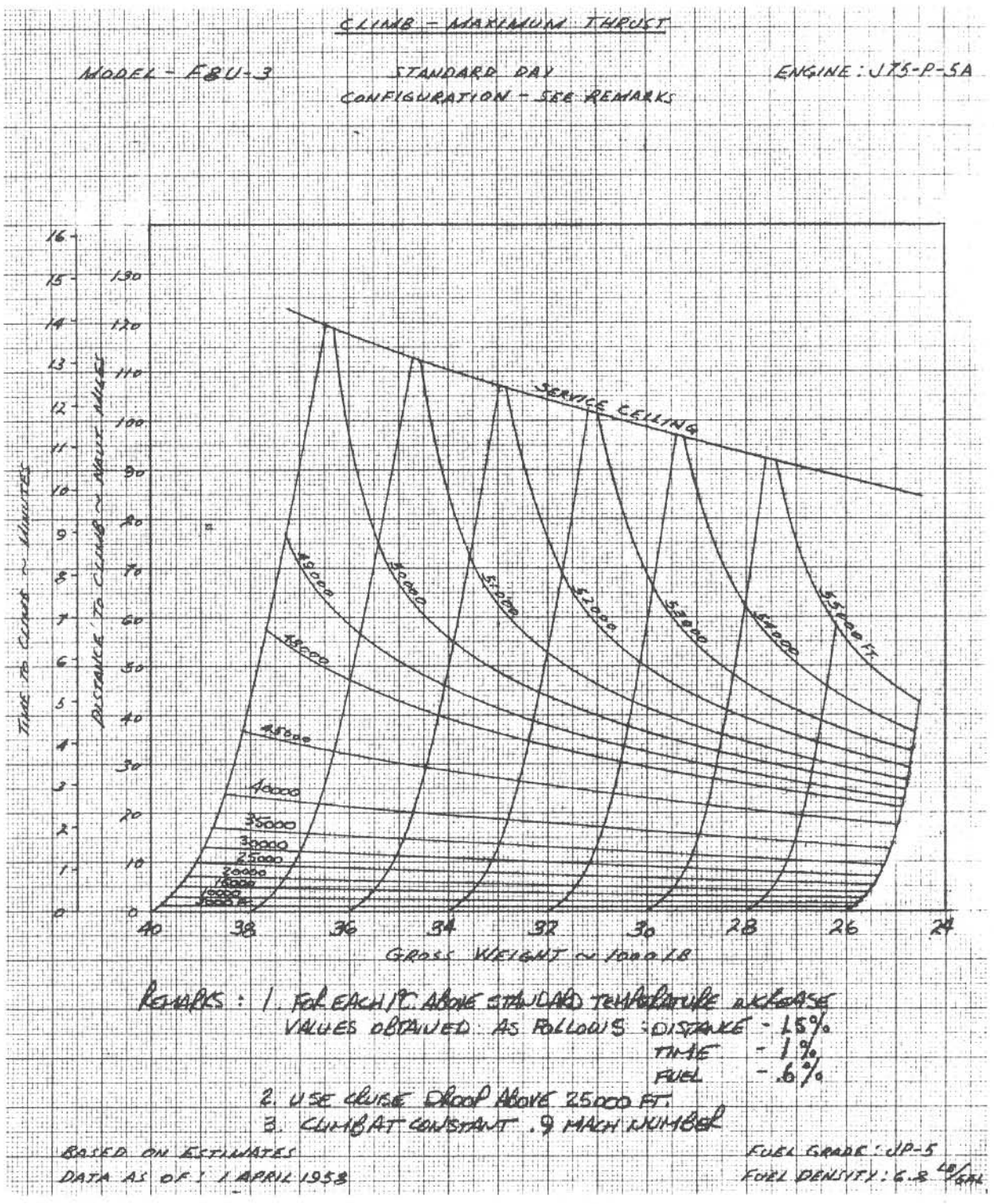
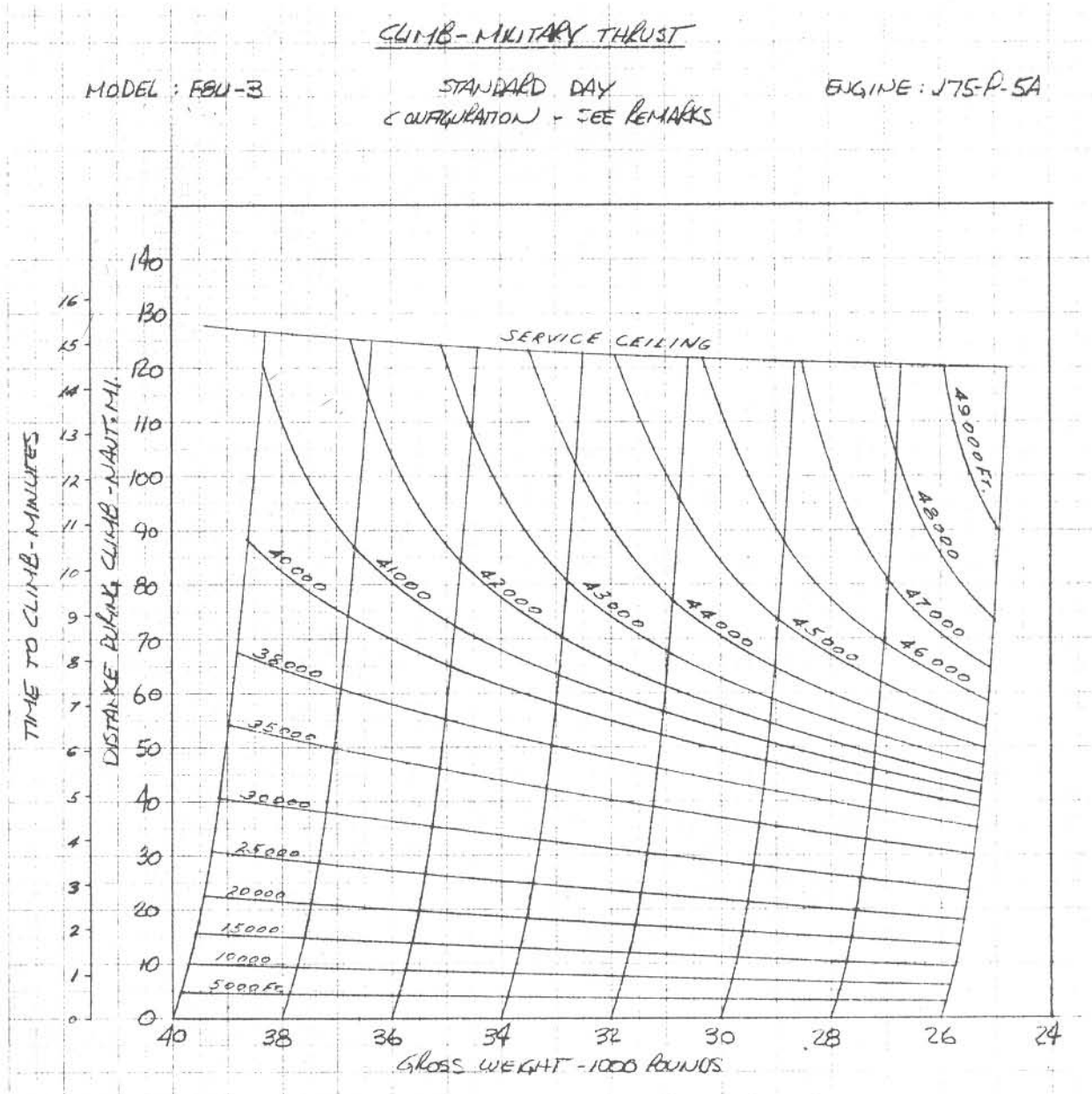


Figure A-3. Climb - Maximum Thrust



REMARKS : 1. FOR EACH 1°C ABOVE STANDARD DAY TEMPERATURE INCREASE VALUES OBTAINED AS FOLLOWS : DISTANCE : 2%
TIME : 2%
FUEL : 1%

2. USE CRUISE DROOP ABOVE 25000 FT.
3. CLIMB AT CONSTANT 500 KNOTS TRUE AIRSPEED

BASED ON : ESTIMATES
DATA AS OF : 1 APRIL 1958

FUEL GRADE : JP-5
FUEL DENSITY : 6.8 LB/GAL.

Figure A-4. Climb - Military Thrust

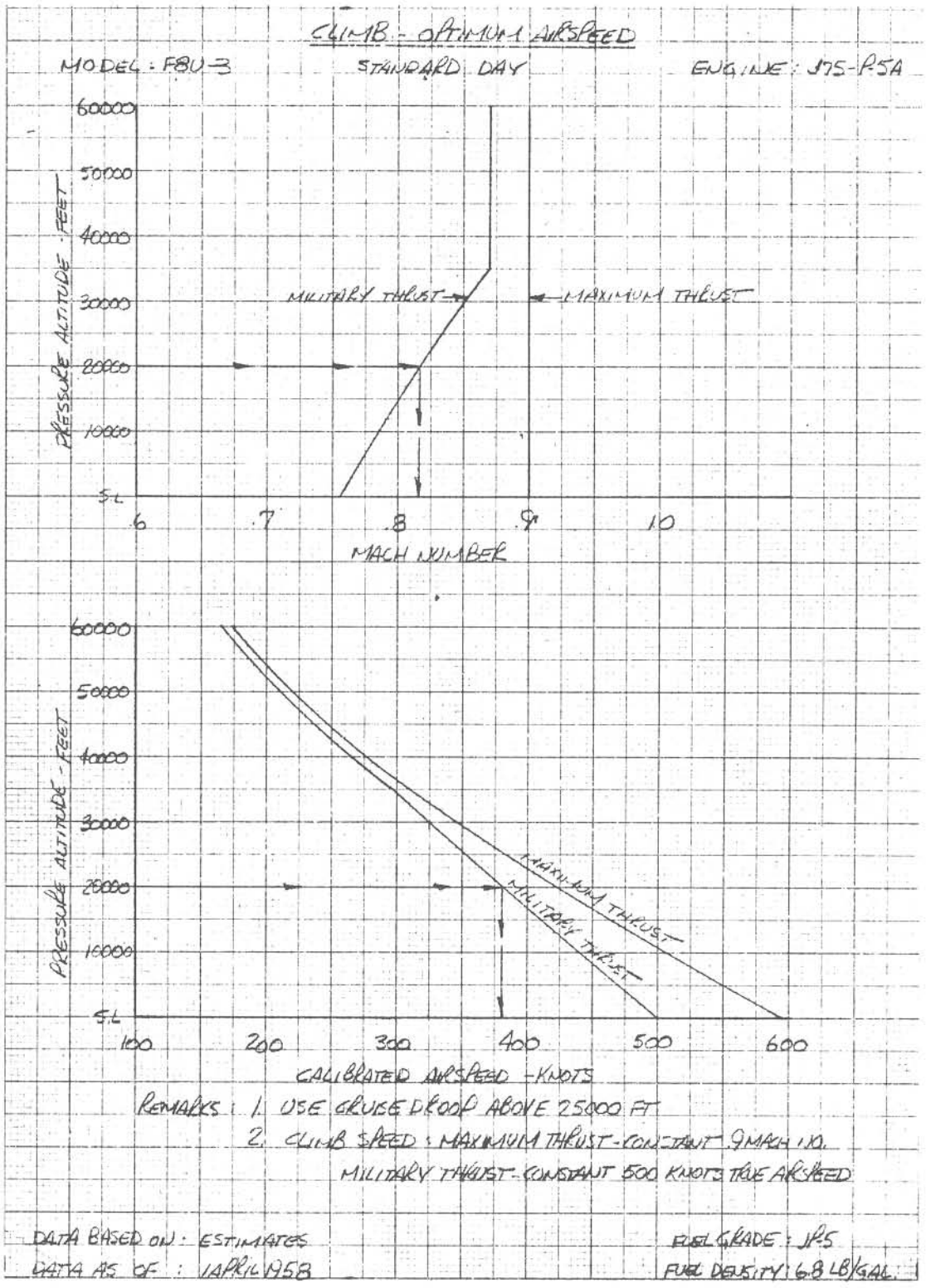


Figure A-5. Climb - Optimum Airspeed

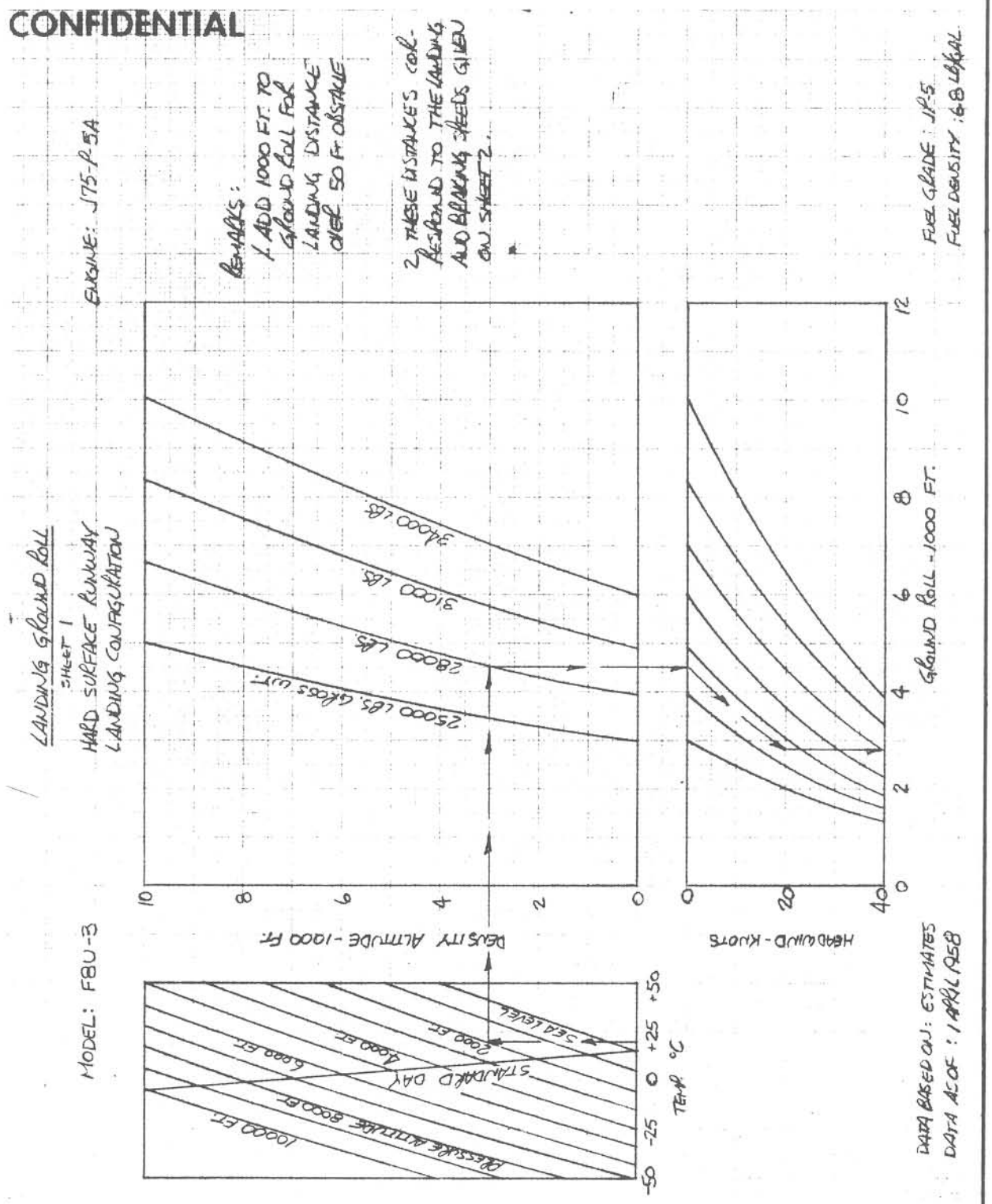


Figure A-6.. Landing Ground Roll (Sheet 1)

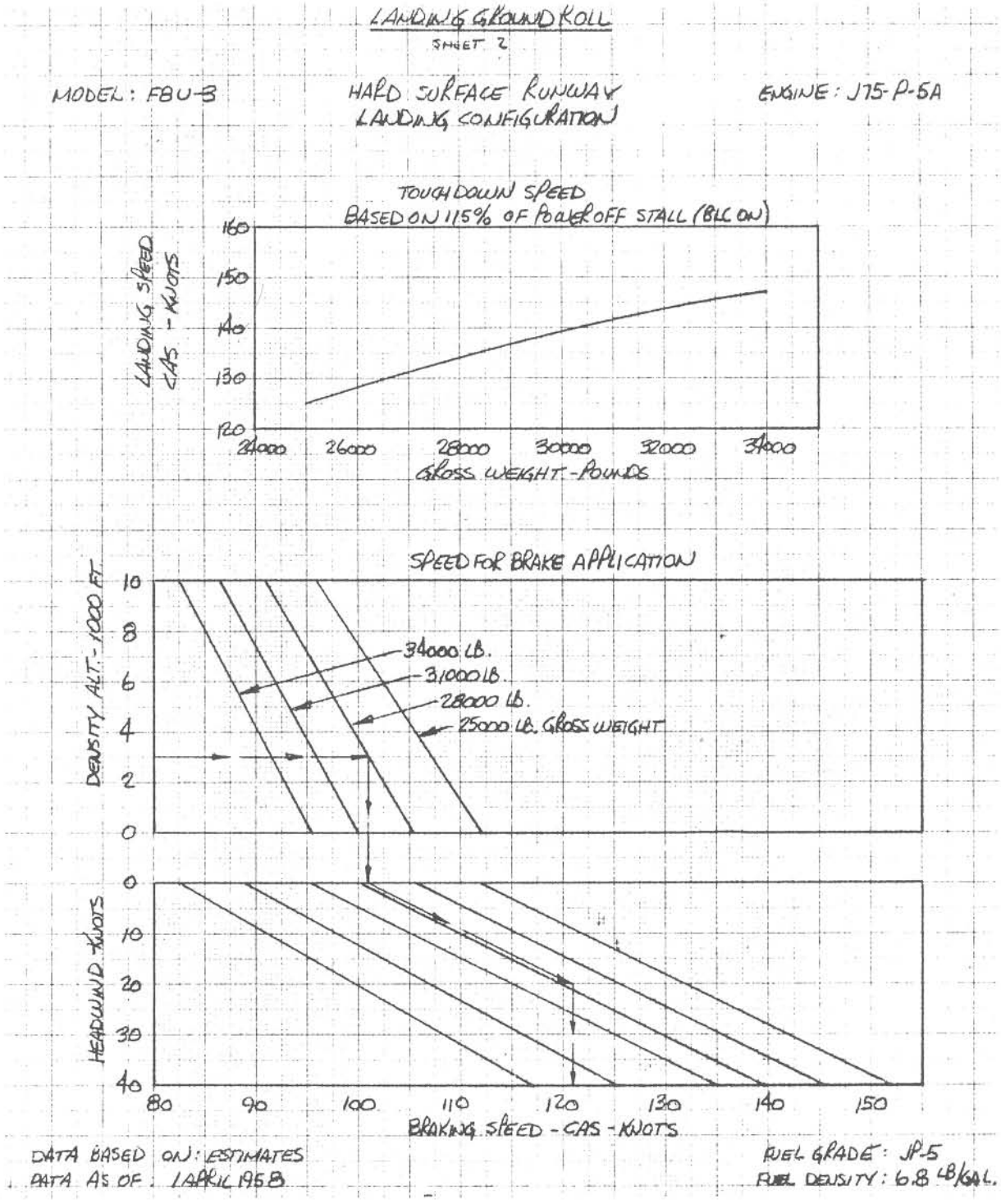


Figure A-6. Landing Ground Roll (Sheet 2)

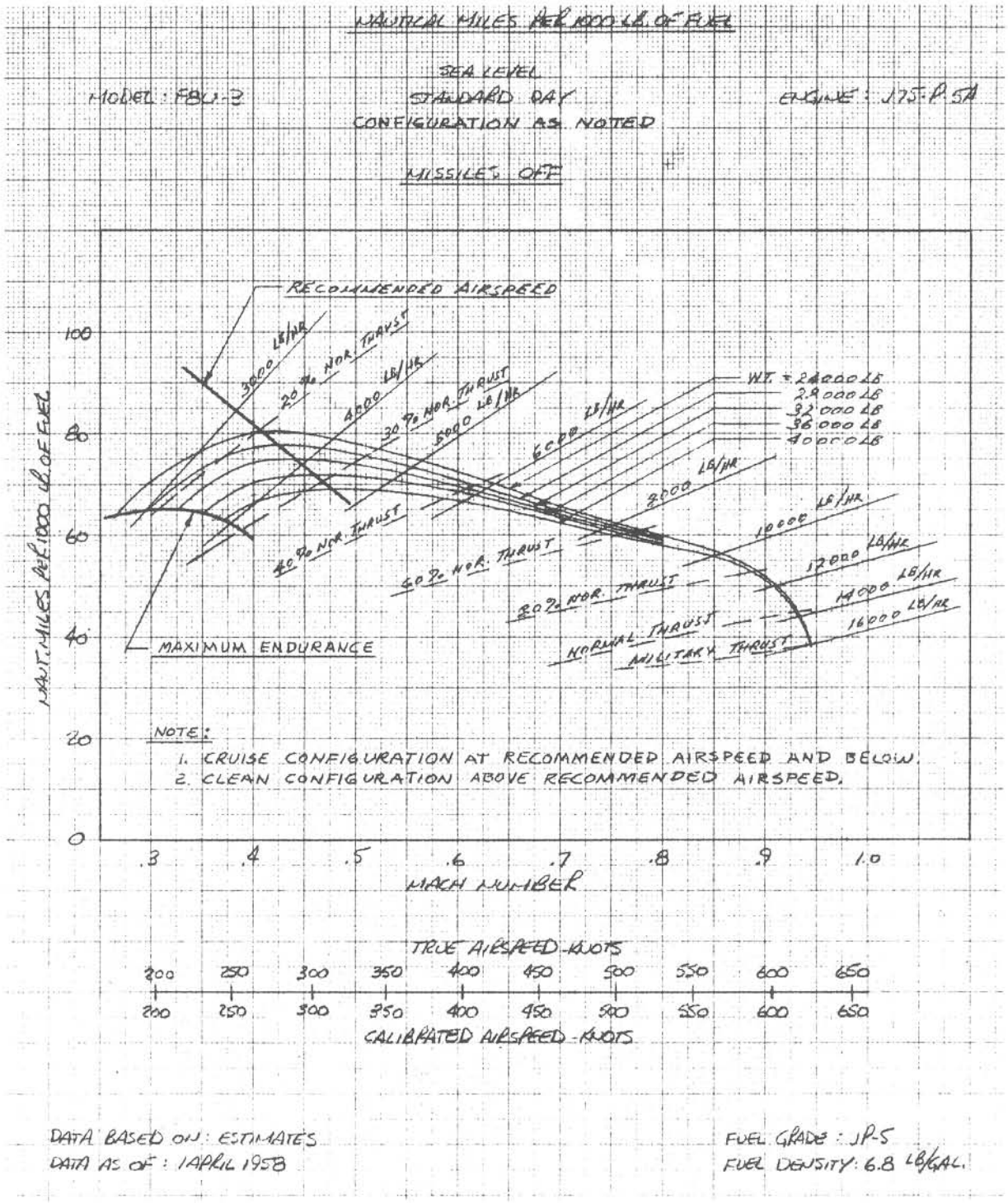


Figure A-7. Nautical Miles per 1,000 Pounds Fuel - Sea Level (Sheet 1)

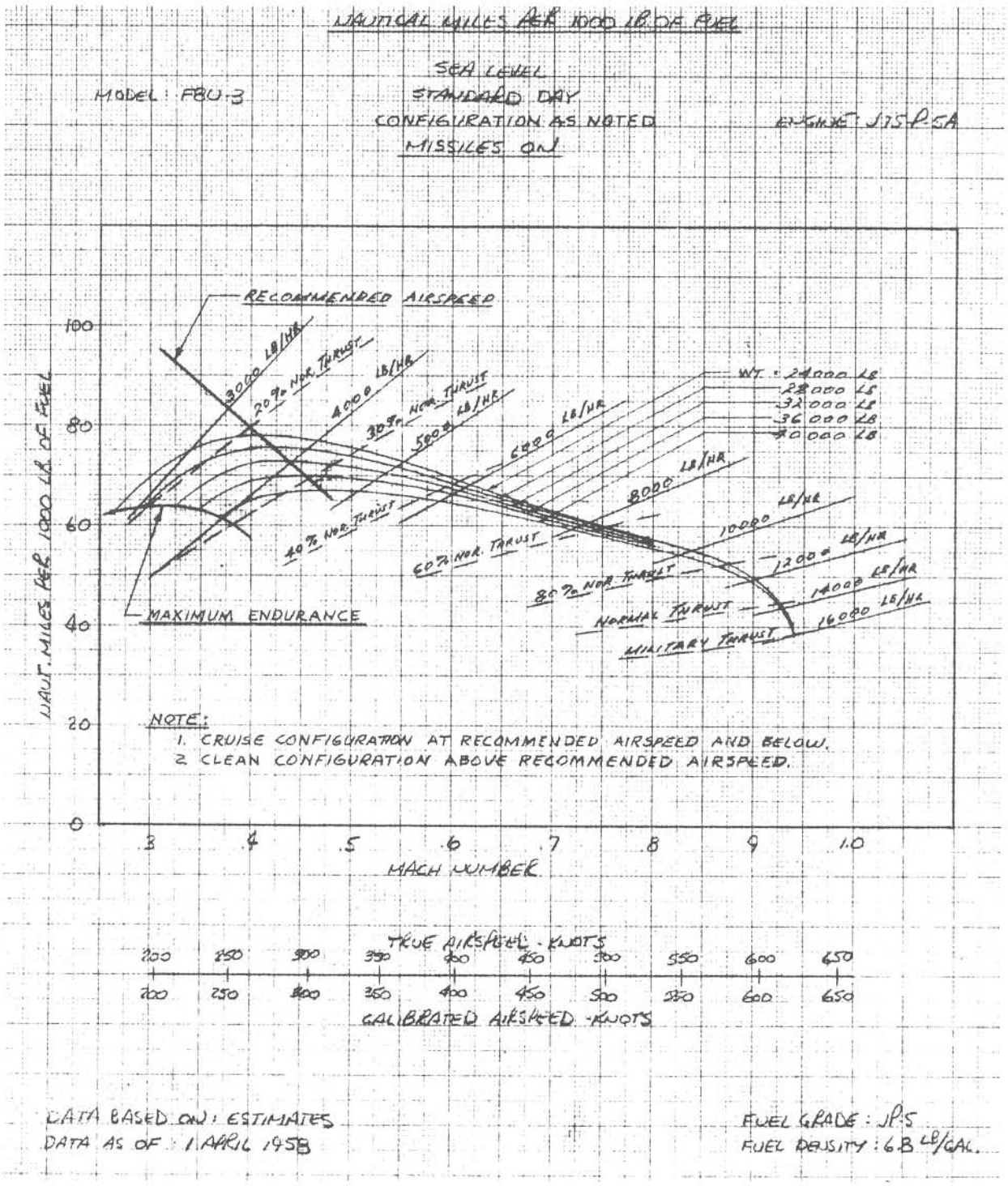


Figure A-7. Nautical Miles per 1,000 Pounds Fuel - Sea Level (Sheet 2)

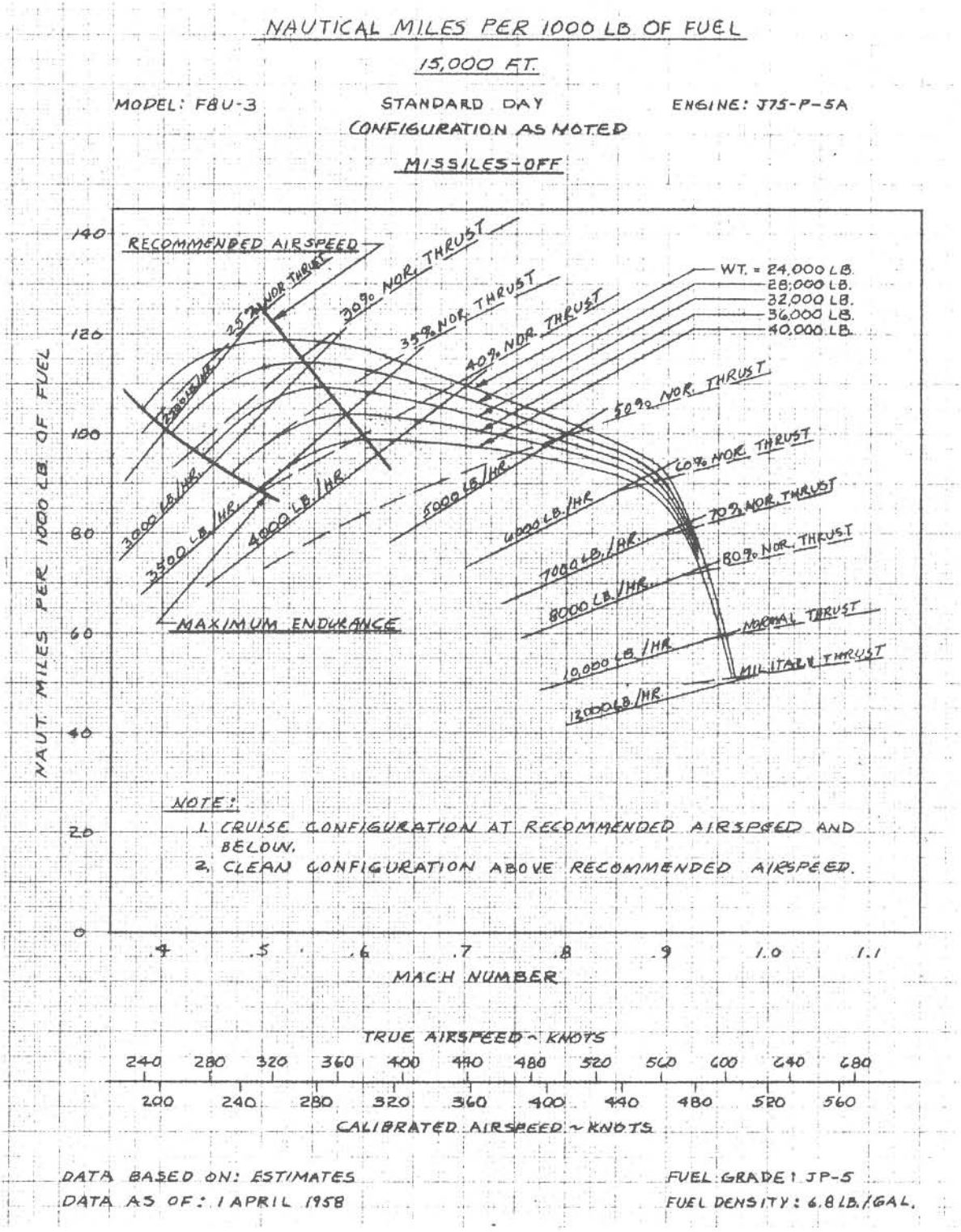


Figure A-8. Nautical Miles per 1,000 Pounds Fuel - 15,000 Feet (Sheet 1)

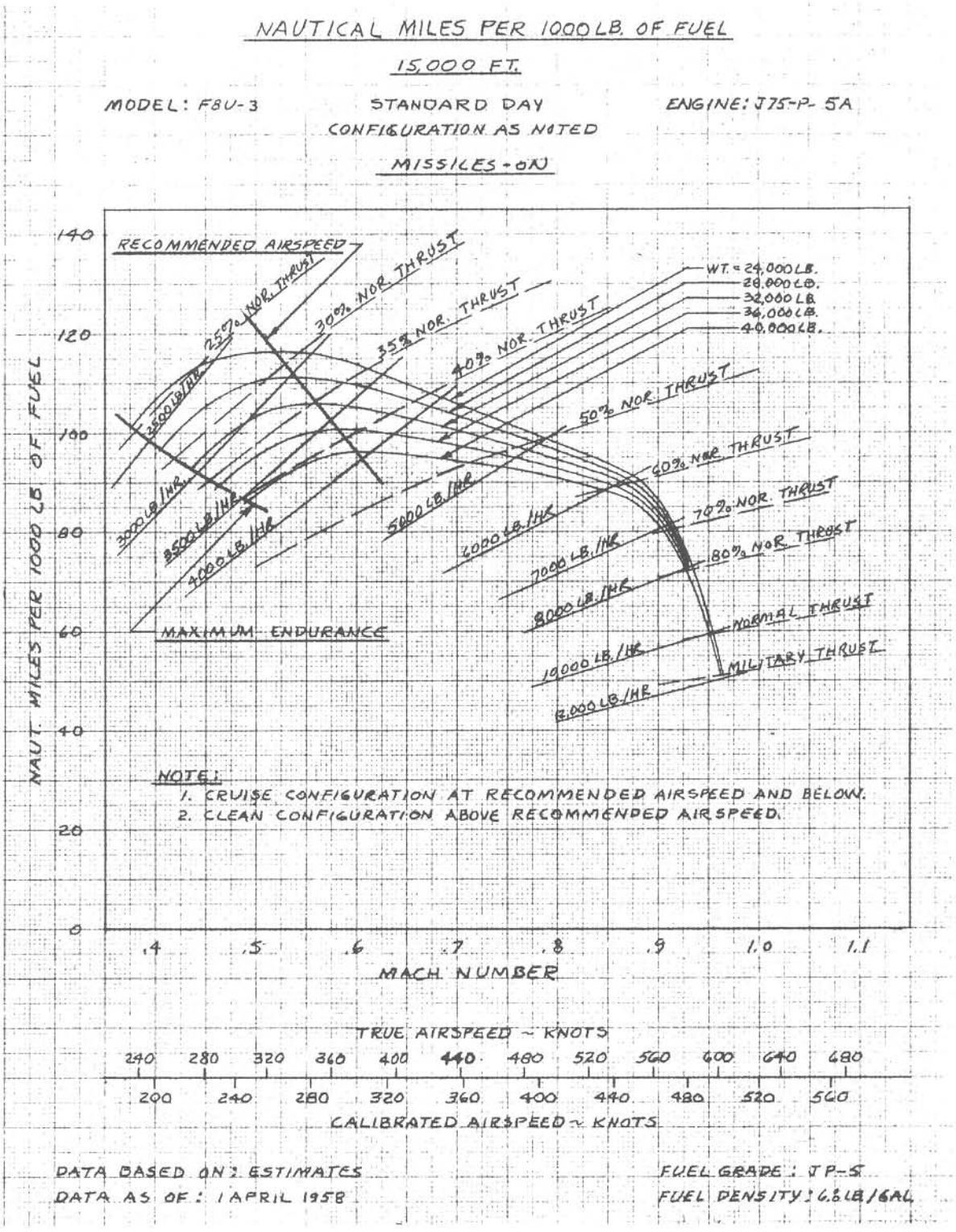


Figure A-8. Nautical Miles per 1,000 Pounds Fuel - 15,000 Feet (Sheet 2)

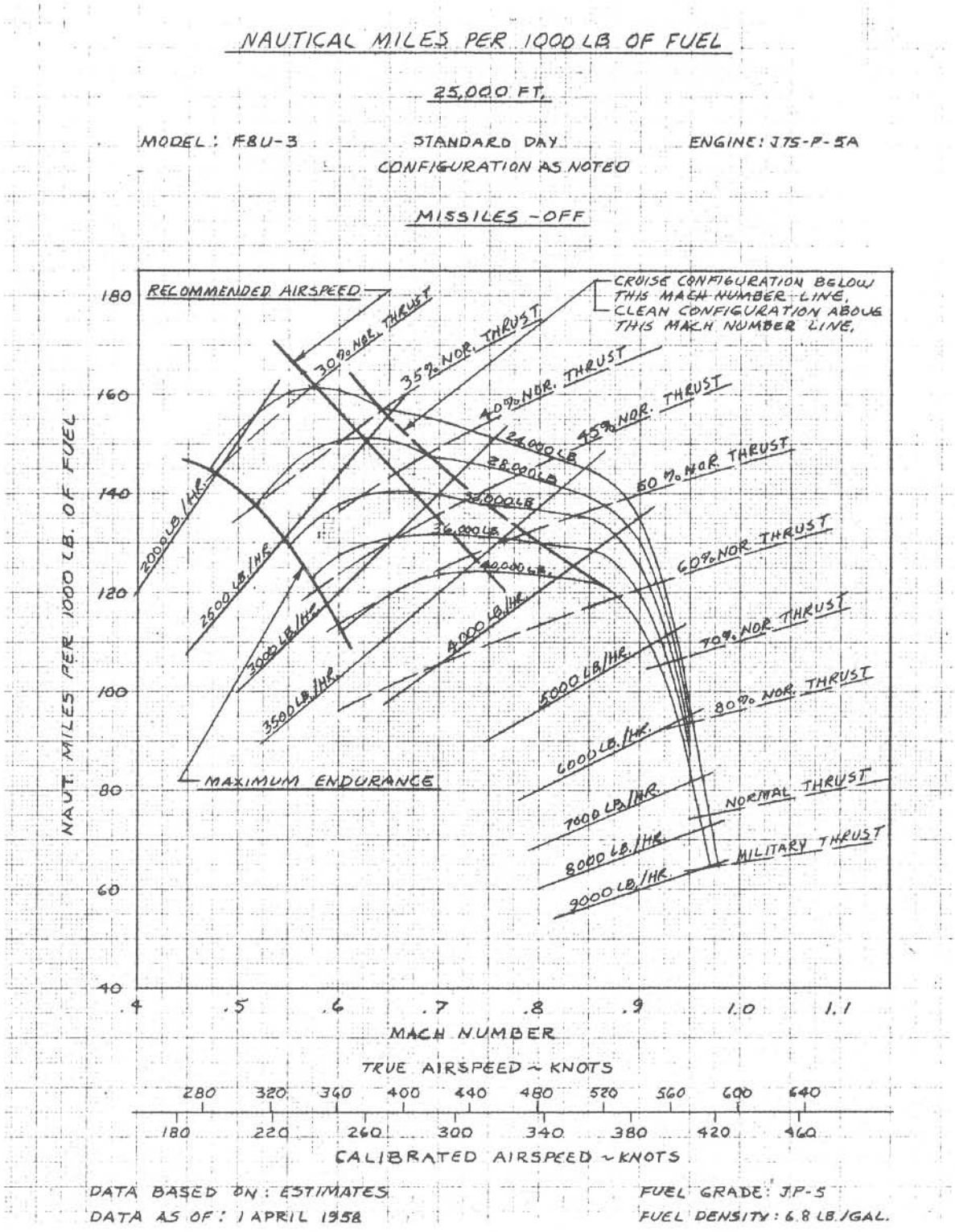


Figure A-9. Nautical Miles per 1,000 Pounds Fuel-25,000 Feet (Sheet 1)

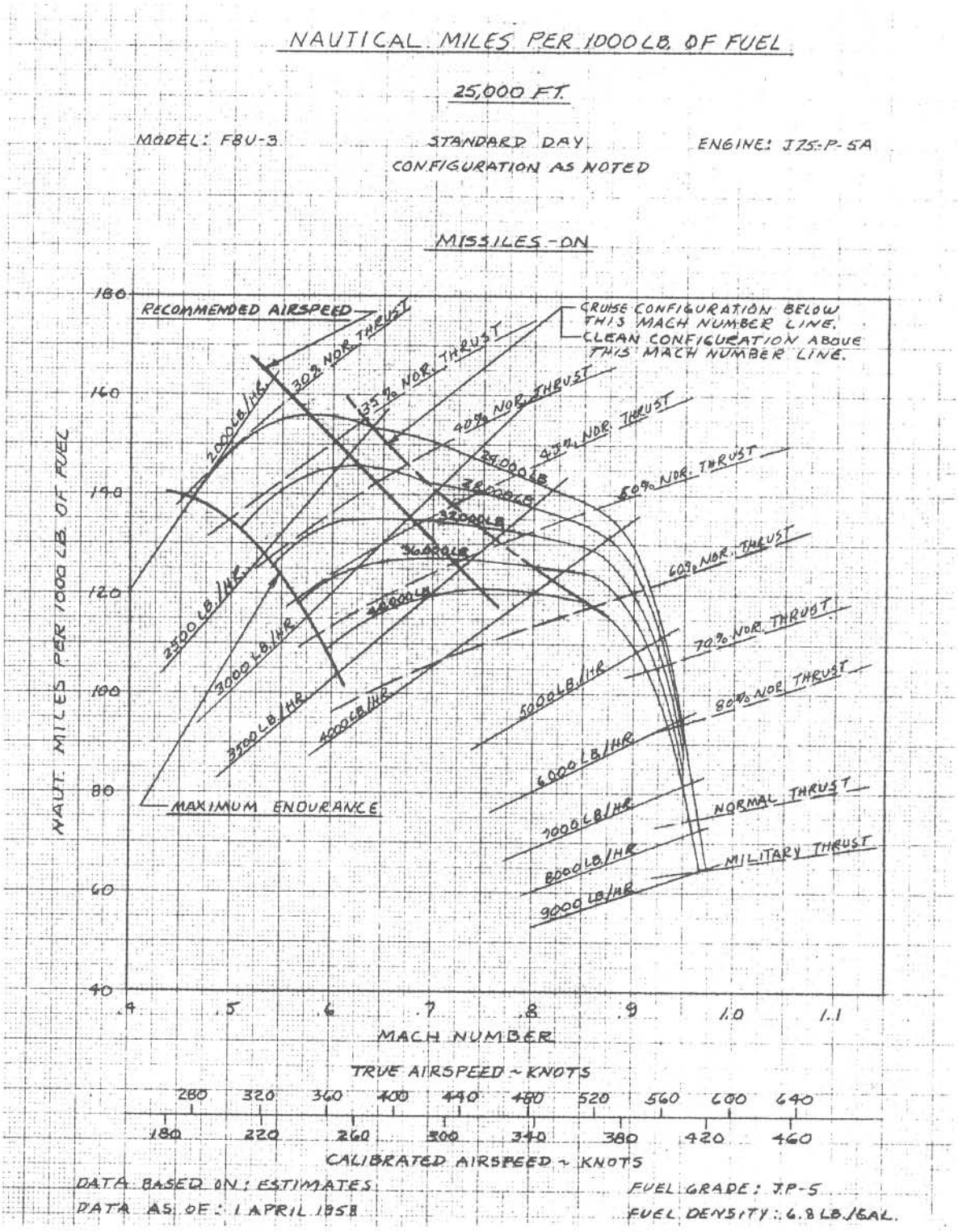


Figure A-9. Nautical Miles per 1,000 Pounds Fuel - 25,000 Feet (Sheet 2)

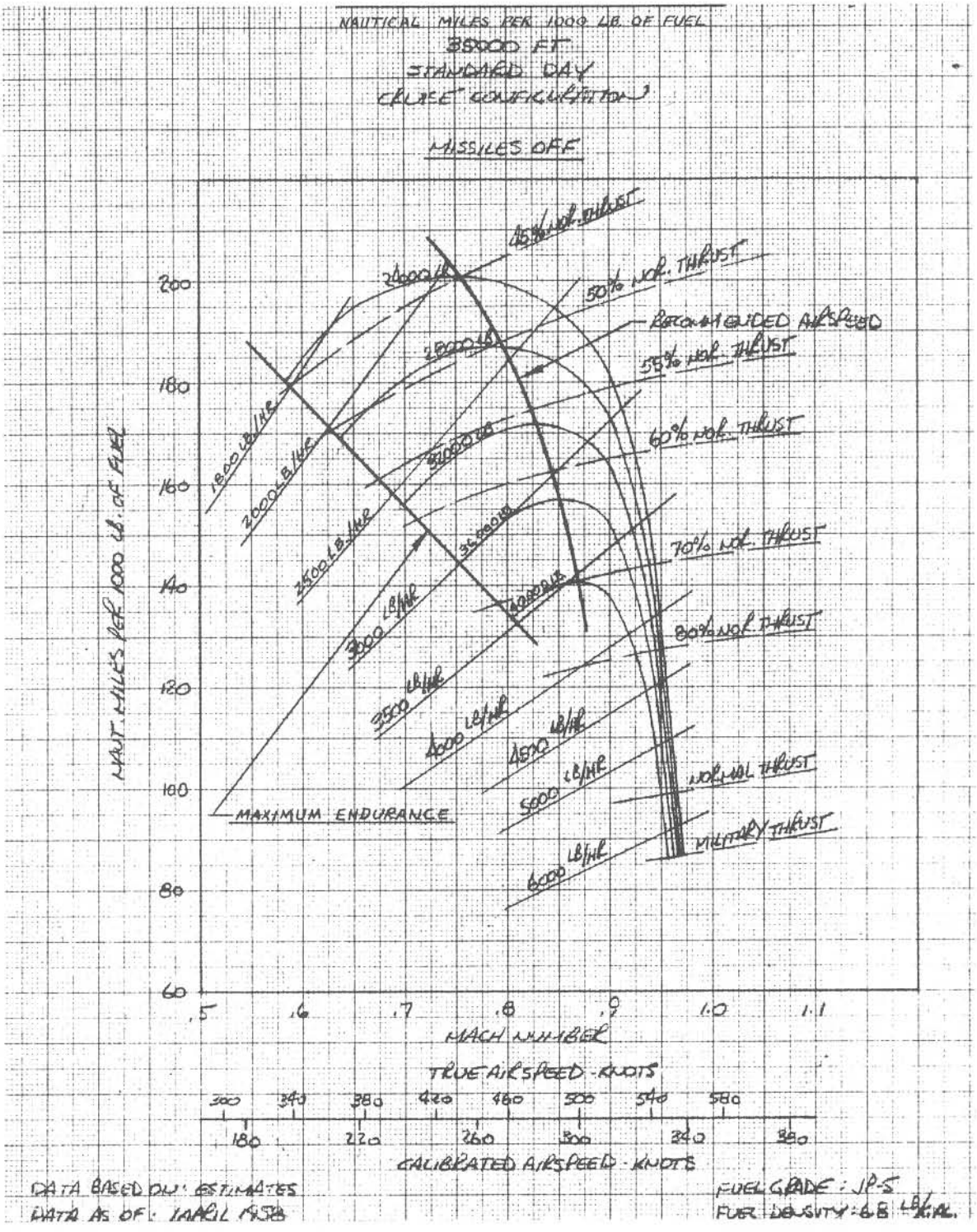


Figure A-10. Nautical Miles per 1,000 Pounds Fuel - 35,000 Feet (Sheet 1)

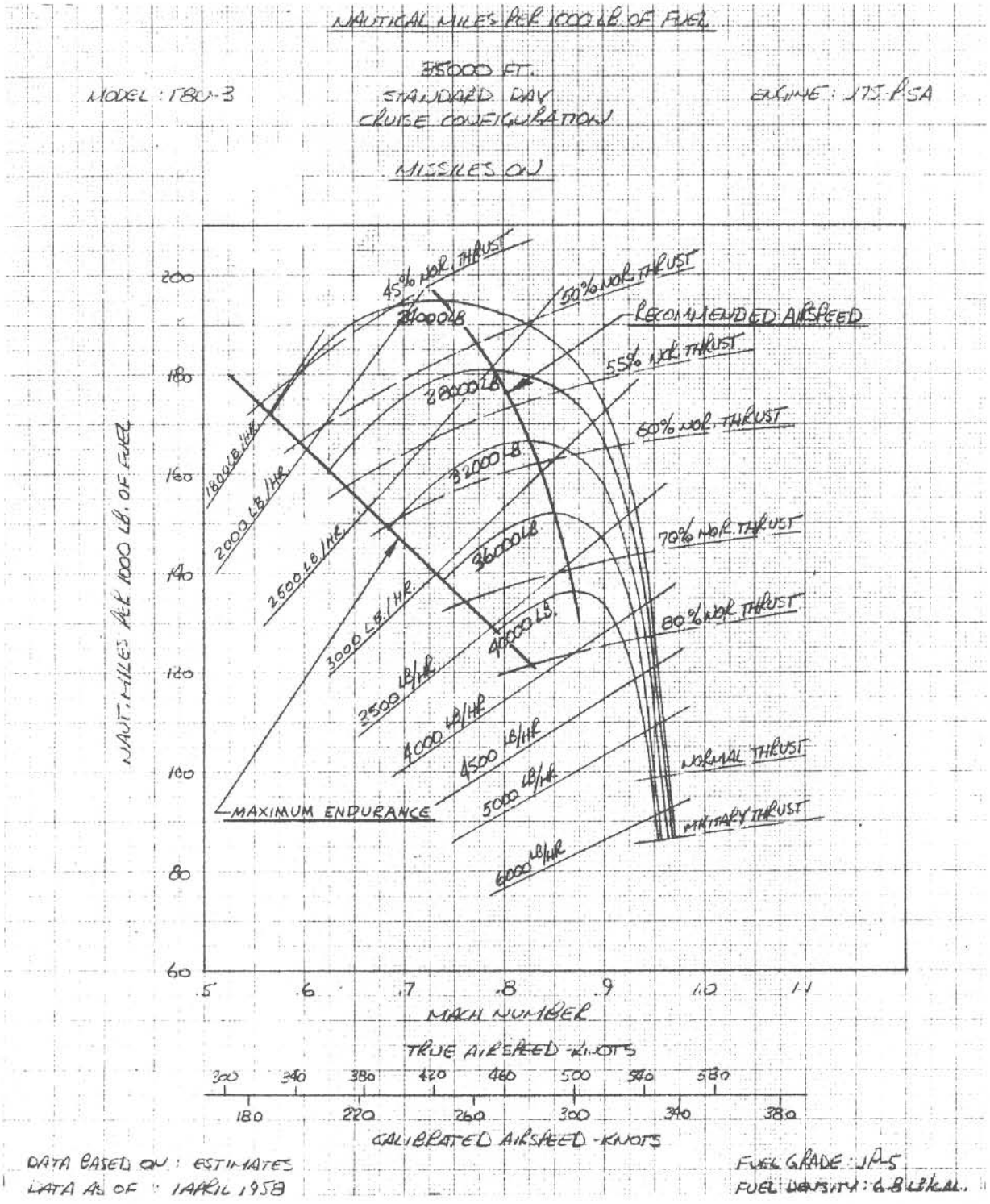


Figure A-10. Nautical Miles per 1,000 Pounds Fuel - 35,000 Feet (Sheet 2)

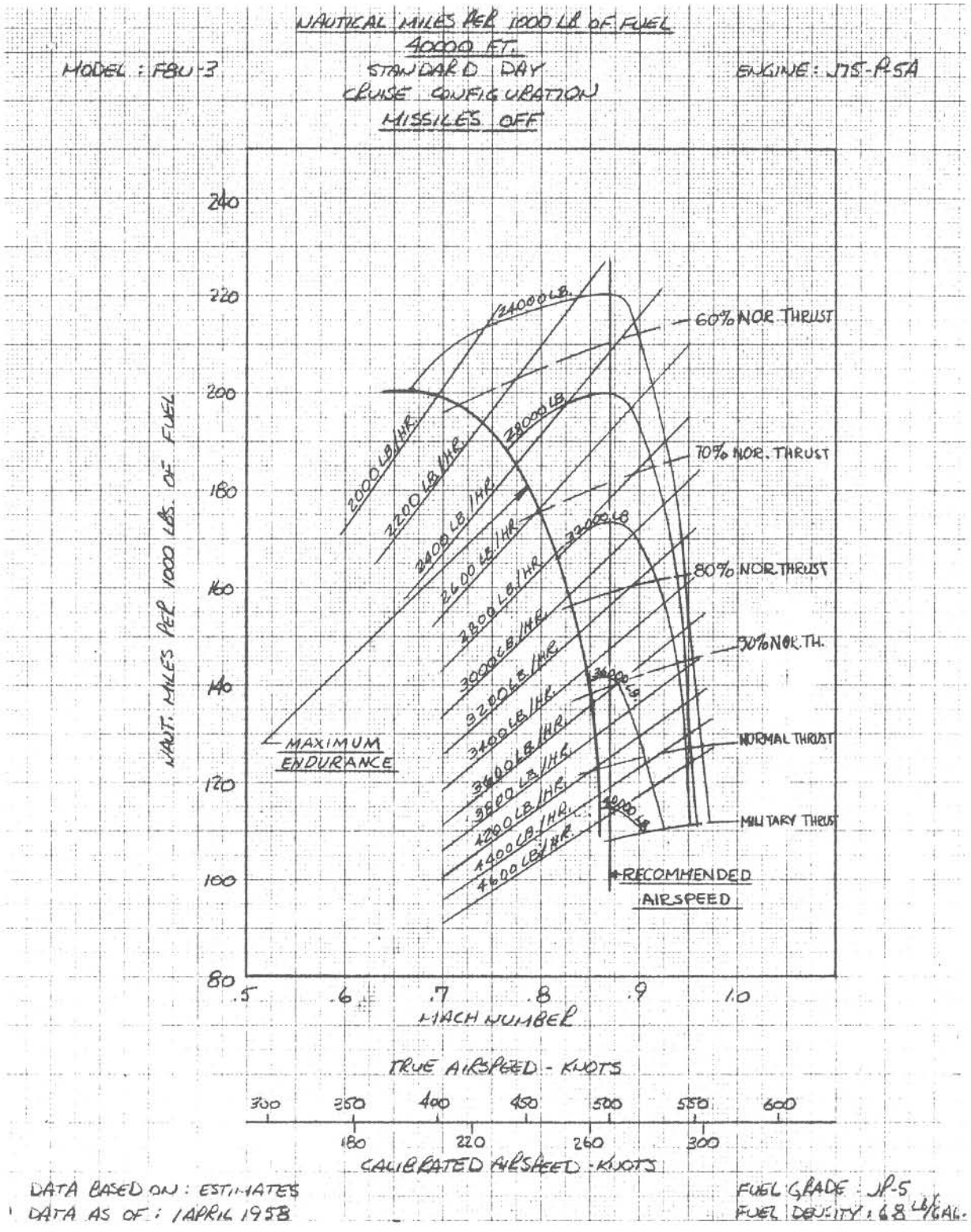


Figure A-11. Nautical Miles per 1,000 Pounds Fuel - 40,000 Feet (Sheet 1)

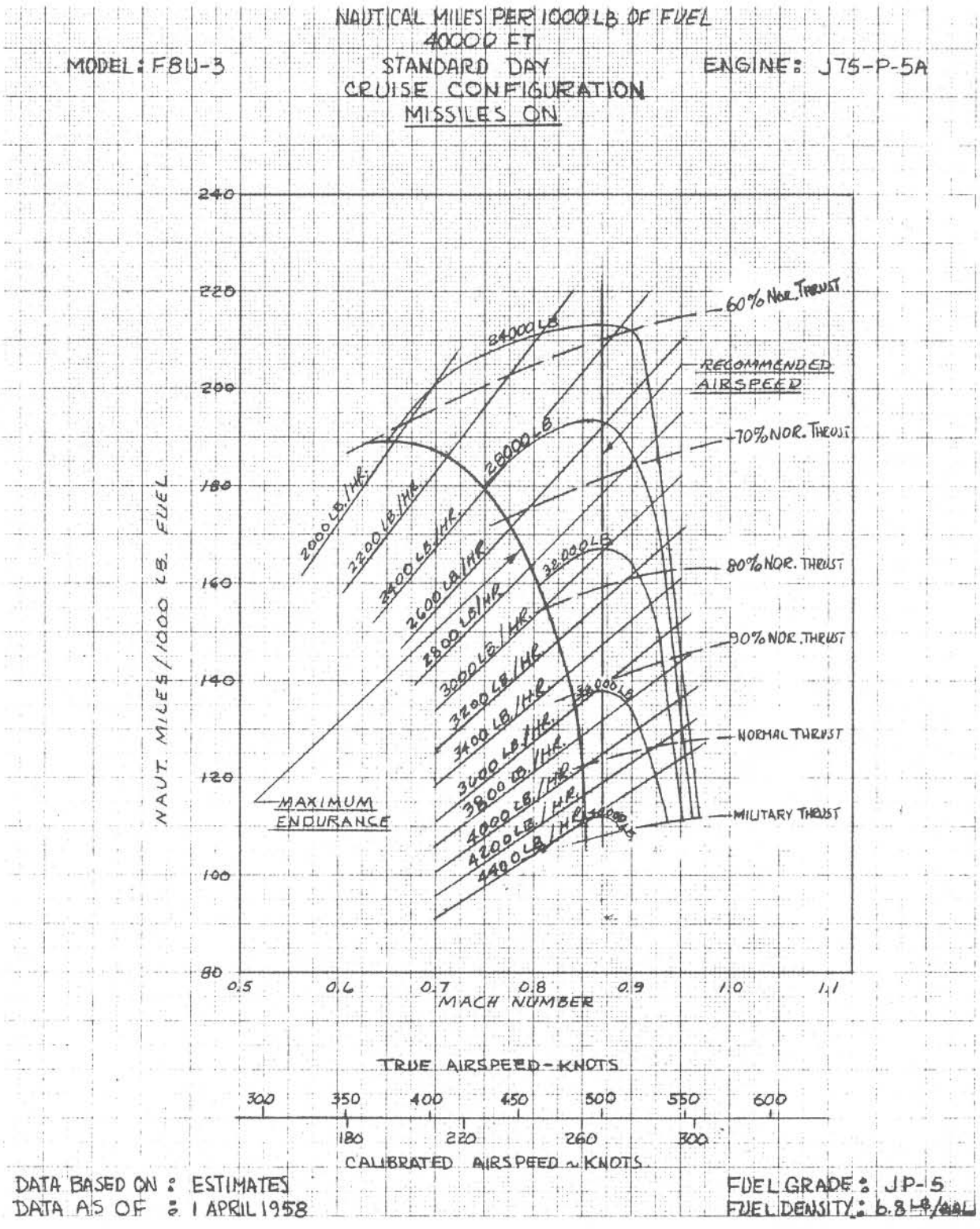


Figure A-11. Nautical Miles per 1,000 Pounds Fuel - 40,000 Feet (Sheet 2)

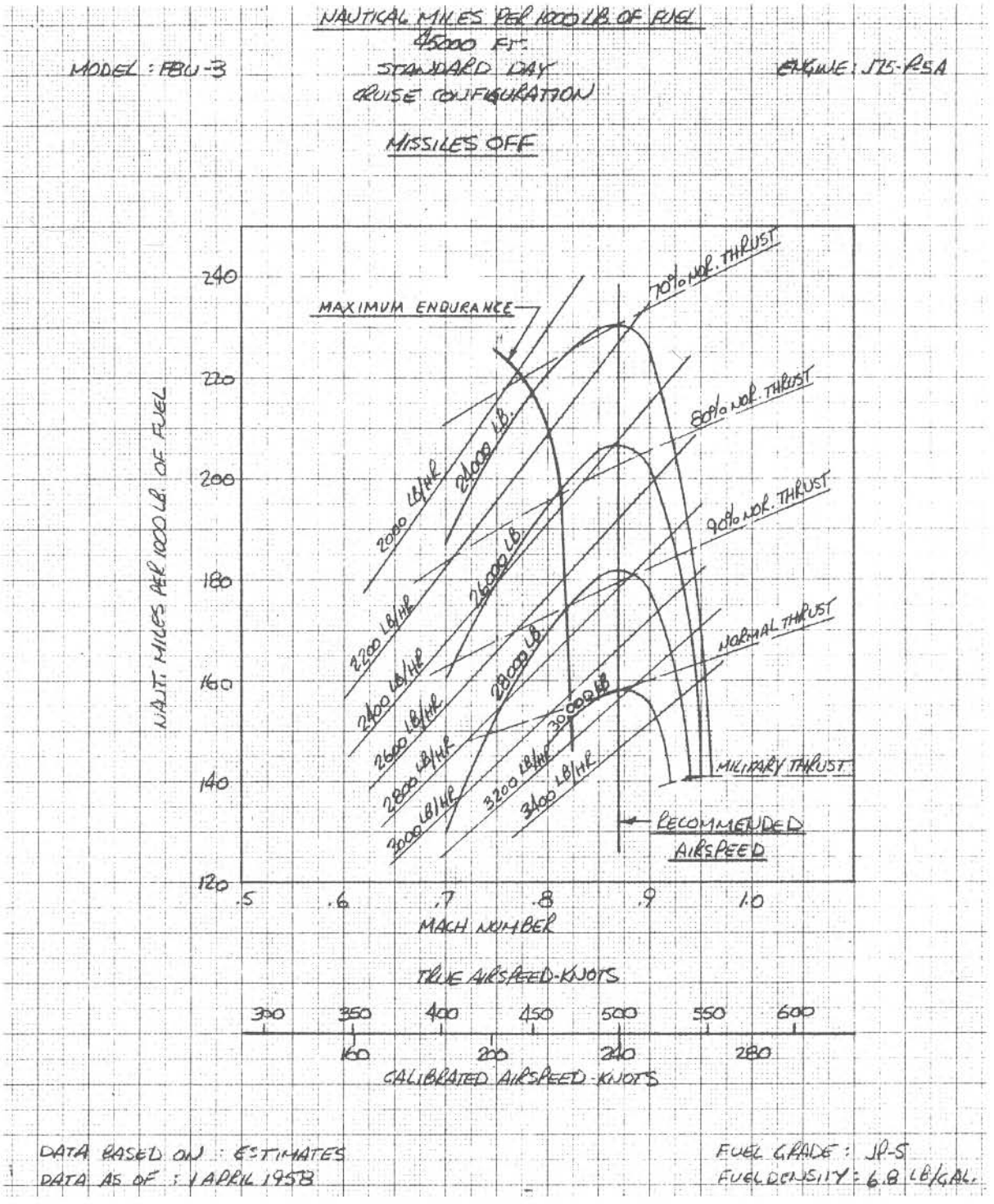


Figure A-12. Nautical Miles per 1,000 Pounds Fuel - 45,000 Feet (Sheet 1)

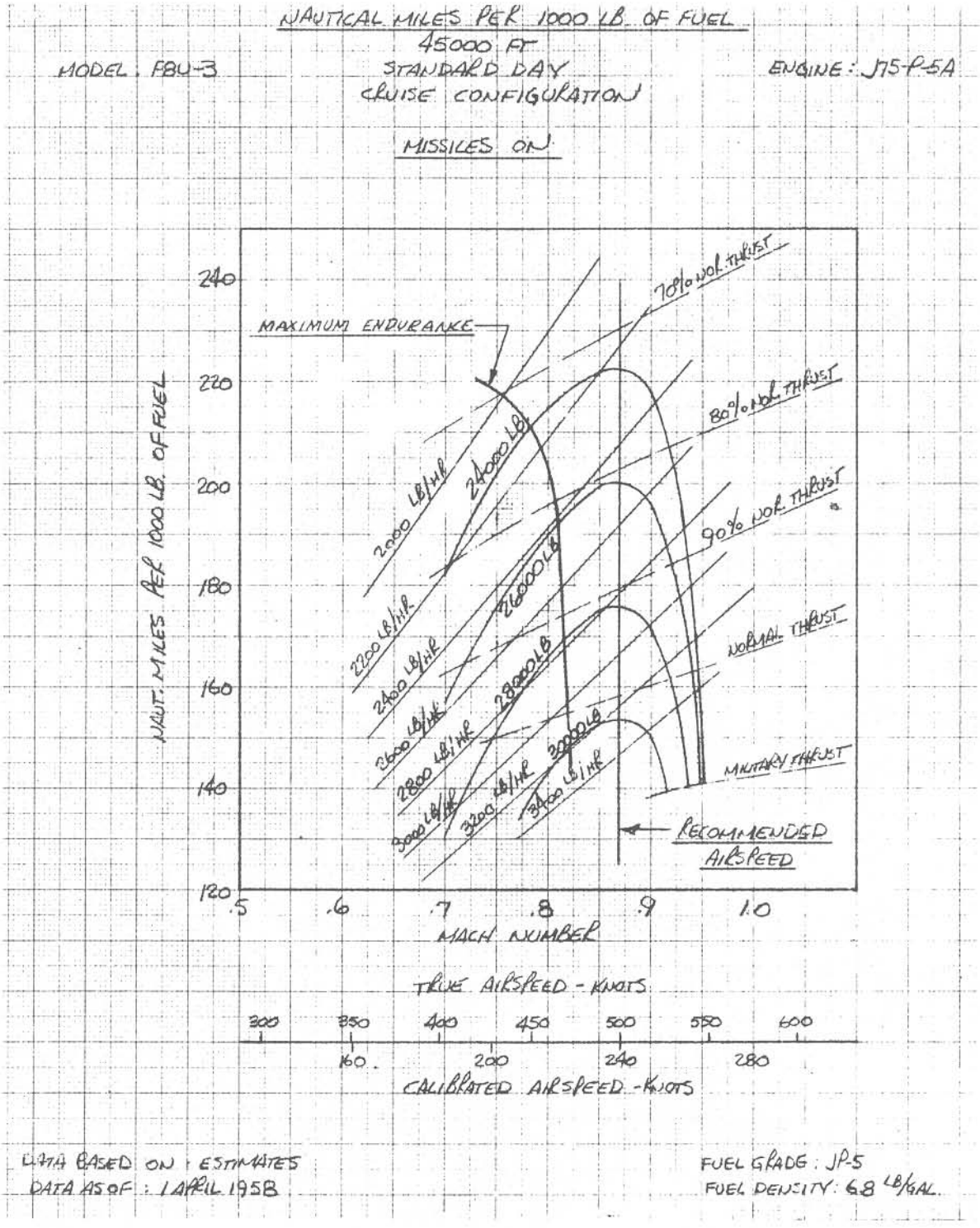


Figure A-12. Nautical Miles per 1,000 Pounds Fuel - 45,000 Feet (Sheet 2)

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