COVER PAGE

F5D-1 (Bu. No. 142350) NASA 213/802 AT THE ARMSTRONG AIR & SPACE MUSEUM 6 NOV 2024

DOUGLAS F5D-1 UTILITY FLIGHT MANUAL

This manual was used in the NASA Edwards AFB Facility pilot's office by research pilots including Neil Armstrong, Bill Dana, and Milt Thompson

This manual was donated by William H. Dana to the flight manual collection of Thomas R. Woodford in 1984 who created this scan for the Armstrong Air & Space Museum in November 2024

NOTES:

Manual regraded as UNCLASSIFIED in accordance with USN BUWEPS 5511.1

The manual contained two inserted documents (scans included at the end of this file):

- An "F5D-1 Questionnaire" written familiarization test filled out by Douglas test pilot Robert E. Drew which was stapled to the inside back cover
- A handwritten set of answers to the F5D-1 Questionnaire composed by an unknown pilot

THIS COVER PAGE ALLOWS VIEWING THE PDF DOCUMENT WITH THE PROPER FACING PAGES WHEN IN TWO-PAGE VIEW

COVER PAGE

CONFIDENTIAL

Utility Flight Handbook for *NAVY MODEL* F5D-1 AIRCRAFT

THIS PUBLICATION SUPERSEDES F5D-1 UTILITY FLIGHT HANDBOOK DATED 15 MARCH 1957

This publication shall not be carried in aircraft on combat missions or when there is a reasonable chance of its falling into the hands of an unfriendly nation, unless specifically authorized by the "Operational Commander."

PUBLISHED BY DIRECTION OF THE CHIEF OF THE BUREAU OF AERONAUTICS

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

CONFIDENTIAL [redacted]

REGRADED TO

UNCLASSIFIED

BUWEPS 5511.1

WARNING

THIS AIRPLANE HAS NOT COMPLETED ITS DEVELOPEMENT PROGRAM, AND THEREFORE EXHIBITS DEFICIENCIES OF WHICH THE PILOT MUST BE AWARE PRIOR TO FLIGHT.

PREDICTED FIN FLUTTER

Flutter is defined as a self-excited, divergent vibration of the airplane control surfaces and/or airplane structure. Due to its self-excited, divergent nature the pilot has no warning of the proximity of flutter and, upon encountering this phenomenon, the rate of divergence is usually such that structural failure results before corrective action is effective. Of a similar nature, although not divergent vibration, is the phenomenon of "buzz", wherein a control surface will oscillate rapidly at a constant small amplitude which may or may not be apparent to the pilot.

A region of vertical fin flutter is predicted within the level flight speed envelope of which the airplane is capable at altitudes of 20,000 feet and below. Flight within the flutter region (if the predictions are valid) would result in structural failure of the vertical fin. See figure 5-2 (sheet 2). A more rigid vertical fin has been designed, but contract considerations did not warrant its incorporation on the subject airplane. The region of rudder "buzz" is at airspeeds less than the predicted flutter airspeeds, and may become apparent to the pilot as a high frequency (30 cps) vibration that can be felt in the cockpit. Sustained flight in the region of control surface "buzz" is not recommended.

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CONFIDENTIAL

Utility

Flight Handbook

for

NAVY MODEL

F5D-1

AIRCRAFT

THIS PUBLICATION SUPERSEDES F5D-1 UTILITY FLIGHT HANDBOOK DATED 15 MARCH 1957

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1 July 1957

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TABLE OF CONTENTS

IMPORTANT

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

This is a utility handbook concerning the description and operation of airplanes BuNo. 139208, 139209, 142349, and 142350.

FOREWORD

This handbook is written as a text for the pilot for immediate study and later reference in order that he may gain complete familiarity with the airplane he is assigned to fly. Thus, as complete a picture as practicable of the basic structure and installations involved are included. It is not the function of this handbook to teach the pilot how to fly the aircraft, as it is assumed he is competent in this respect; however, the handbook contains information regarding behavior peculiar to the airplane in various conditions of flight and ground operation.

The handbook is divided into six sections. Sections I, II, and III are closely interrelated and contain complete information relative to the physical act of flying the airplane. Section I provides a complete description of the aircraft and its systems, instruments and controls. Emergency equipment that is not a part of an auxiliary system is also described.

Section II contains information for the normal operation of the airplane and describes all procedures to be accomplished by the pilot from the time the aircraft is approached prior to flight until it is left parked on the ramp after completing one non-tactical flight under ordinary conditions. Section III describes the procedures to be followed in meeting any emergency, except those in connection with auxiliary equipment, that could reasonably be expected to be encountered.

Section IV contains the description and operation of all auxiliary equipment which does not actually contribute to flight but enables the aircraft to perform specialized functions. All limitations and restrictions that must be observed during normal operation are discussed in Section V. Section VI attempts to evaluate any unusual flight characteristics, both favorable and unfavorable, which the aircraft may possess.

In order to make the text as specific as possible, the nomenclature used to identify controls and other equipment is identical, wherever possible, to that used in the airplane itself. Such nomenclature is capitalized. Also capitalized, and enclosed in quotation marks, are the positions of the controls as they are identified in the airplane. For example, "The SEAT switch is spring-loaded to the center position, and placing it momentarily to the 'UP' or 'DOWN' position will adjust the seat accordingly."

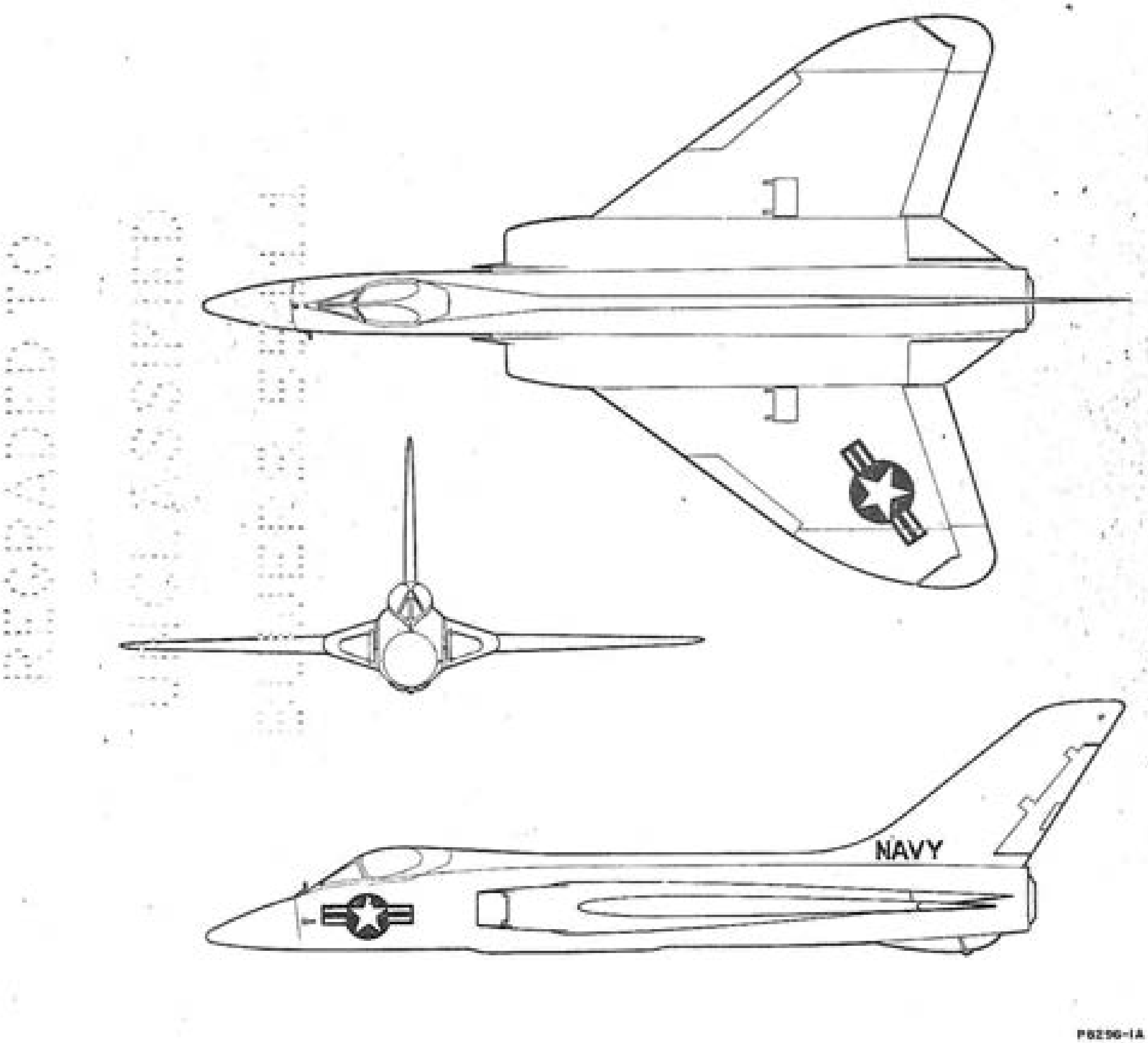


Figure 1-1. Model F5D-1 Airplane

SECTION I

DESCRIPTION

THE AIRCRAFT

The Navy Model F5D-1 airplane, manufactured by the El Segundo Division of the Douglas Aircraft Company, is a single place, single engine, jet propelled, high performance fighter-interceptor landplane capable of controlled supersonic operation in level flight. Propulsion is provided by a Pratt and Whitney J57-P-8 axial flow gas turbine engine with afterburner. The aircraft can operate from either a land base or a carrier with equal facility, having provisions for catapulted take-off and arrested landing. The tailless configuration of the aircraft has created the need for an unconventional control surface arrangement consisting of elevons, rudder, and slats. A tail bumper wheel is provided in addition to a conventionally arranged tricycle landing gear to prevent the tail cone from striking the deck or runway during landing at the high angles of attack inherent in the tailless configuration. Armament consists of seventy-two internally carried two-inch diameter, folding fin stabilized rockets, mounted in four retractable rocket doors. Alternate armament equipment, consisting of four 20 MM cannon or two air to air missiles, may be installed.

AIRPLANE DIMENSIONS

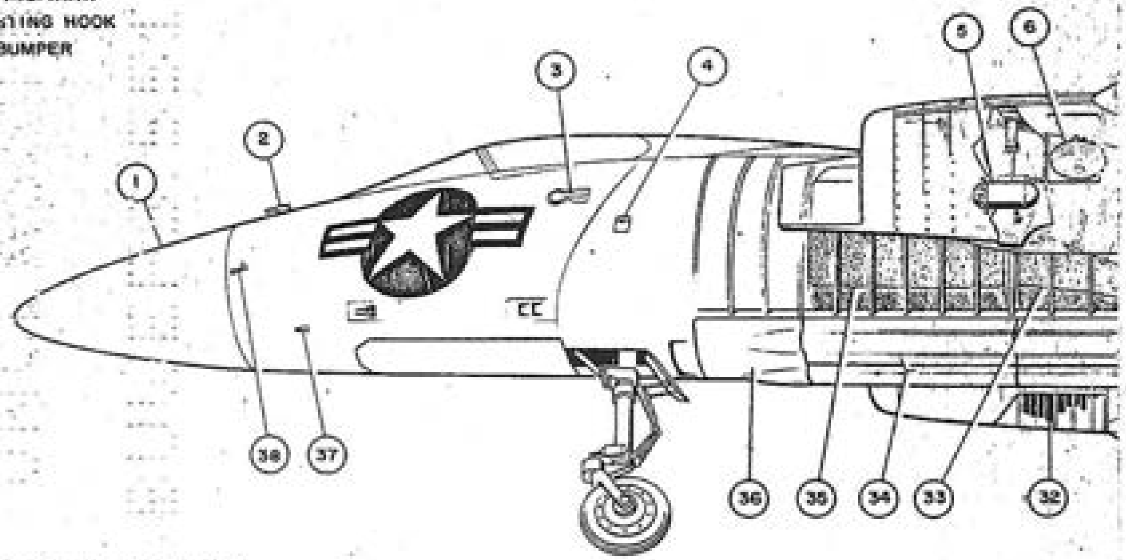
The principal dimensions of the airplane are as follows:

Length (3 point position)	53' 9 9/16"
Span (wings extended)	33' 6"
Span (wings folded)	27' 6"
Height (wings extended, 3 point position)	14' 10"
Height (wings folded, 3 point position)	14' 10"
Main wheel tread	12' 11"
Sweepback of wing leading edges	52. 5 degrees
Wing area (total)	557 sq. ft.

ENGINE

The Pratt and Whitney J57-P-8 "Turbo Wasp" engine is a continuous flow gas turbine engine consisting of two multi-stage axial flow compressors, eight combustion chambers, and a split, three stage turbine assembly. The engine is comprised of three main sections:

1. RADOME
2. PITOT TUBE
3. EXTERIOR CANOPY CONTROL
4. EXTERIOR CANOPY JETTISON CONTROL
5. DROP-OUT HYDRAULIC PUMP
6. LIQUID OXYGEN TANK
7. FIRE EXTINGUISHER DISCHARGE INDICATORS (1)
8. CATAPULT HOOK
9. FORWARD WING TANK
10. EMERGENCY GENERATOR
11. SLAT
12. PRESSURE FUELING RECEPTACLE (2)
13. SPEED BRAKE
14. NAVIGATION LIGHT
15. WING FOLD CONTROLS ACCESS
16. PORT MAIN ELEVON
17. TAIL LIGHT
18. RUDDER
19. TRIM TAP
20. PORT INBOARD ELEVON
21. AFT WING TANK
22. ARRESTING HOOK
23. TAIL BUMPER



24. STARBOARD INBOARD ELEVON
25. FUEL SYSTEM VENT EXIT
26. STARBOARD MAIN ELEVON
27. AFT ENGINE ACCESS DOOR
28. FORWARD ENGINE ACCESS DOOR
29. AIR STARTER ACCESS DOOR
30. EXTERNAL POWER RECEPTACLE AND SWITCH
31. AFT ROCKET DOORS
32. MAIN GENERATOR
33. AFT FUSELAGE SUMP TANK
34. FORWARD ROCKET DOORS
35. FORWARD FUSELAGE TANK
36. EQUIPMENT COMPARTMENT ACCESS DOOR
37. STATIC PORT
38. ANGLE OF ATTACK DETECTOR

(1) AIRPLANES BUNO. 139308 - 139309 ONLY.
(2) AIRPLANES BUNO. 142349 - 142357.

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Figure 1-2. General Arrangement (Sheet 1)

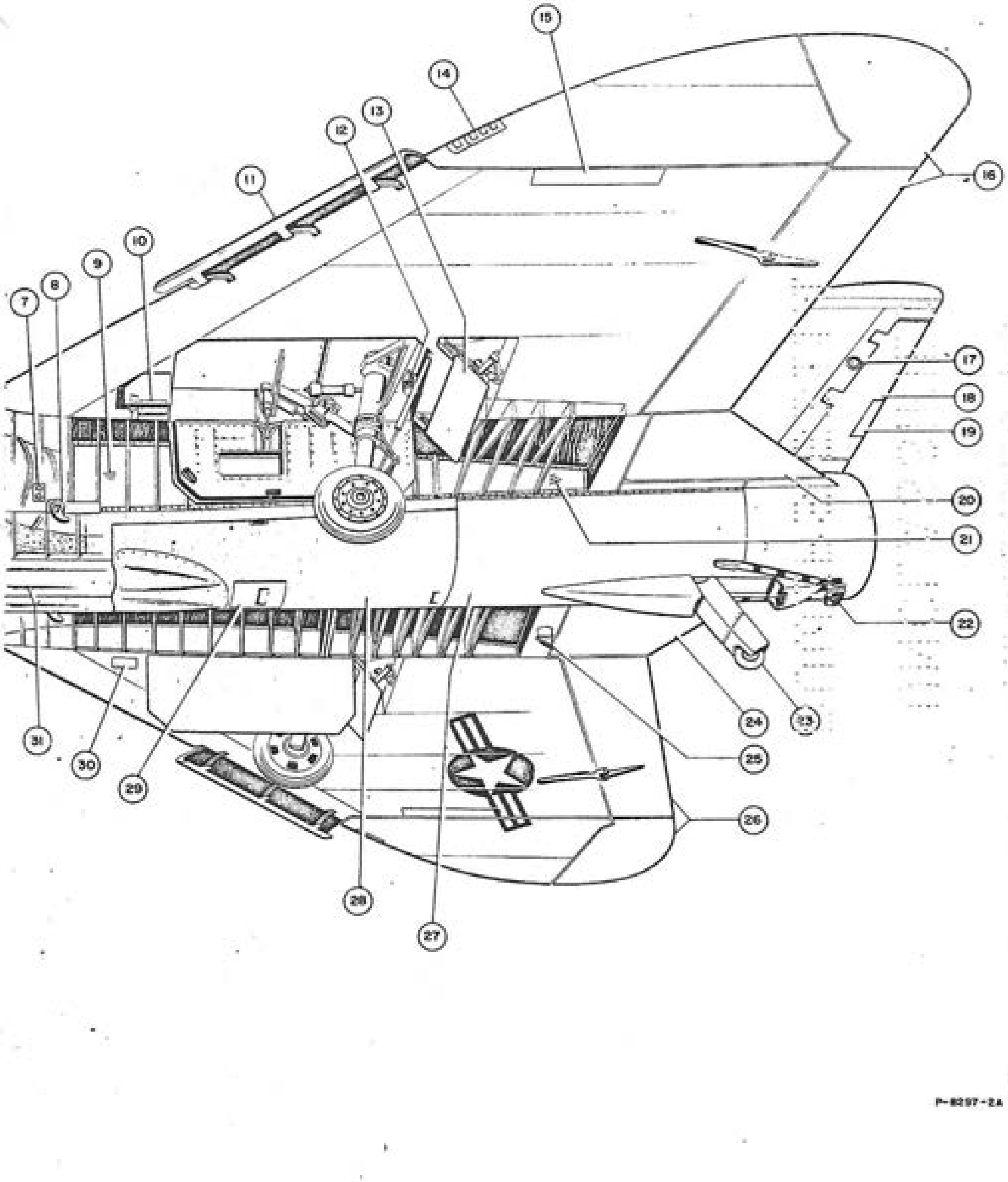


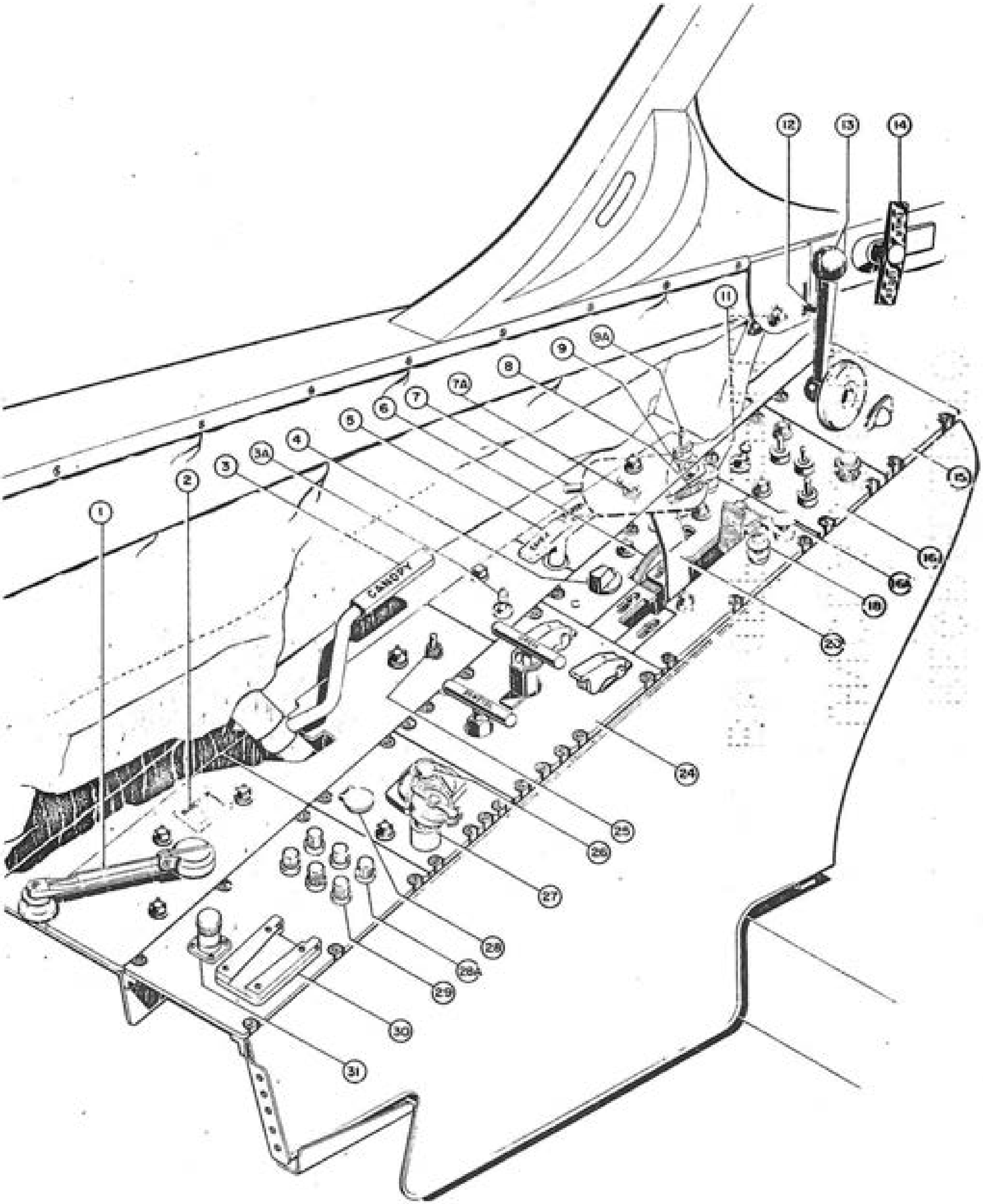
Figure 1-2. General Arrangement (sheet 2)

Key to figure 1-3, (sheet 1)

1. Mechanical advantage changer control
2. Mechanical advantage changer indicator window
3. Interior canopy control handle
3A. Transonic trim switch
4. Rudder trim control
5. Emergency stores release handle
6. Throttle friction and lock control
7. Master exterior lights switch (temporarily: wire recorder switch)
7A. Automatic systems switch
8. Speedbrake switch
9. Microphone switch
9A. Vertical gyro erection switch (in autopilot) (inoperative)
10. (deleted)
11. Engine starter switch
12. Landing gear retraction release control
13. Landing gear control handle
14. Emergency landing gear release handle
15. Face mask temperature control
16. Engine control panel
16A. Pitch damper engage switch
17. (deleted)
18. Yaw damper engage switch
19. (deleted)
20. Throttle lever
21. (deleted)
22. (deleted)
23. (deleted)
24. Spin and drag chute control panel
25. Oxygen control
26. Anti-blackout control valve
27. Anti-blackout suit hose receptacle
28. Personnel gear adapter
28A. Fuses for yaw damper, ignition and fuel flow
29. Spare fuses
30. Stowage receptacle (interim) (for pressure suit oxygen regulator)
31. Face mask heater connection

- -

the compressor section; the accessory section; and the combustion chamber, turbine, and exhaust section. The compressor section consists of a low pressure, nine stage unit driven by the second and third stages of the turbine assembly, and a high pressure, seven stage unit driven by the first stage of the turbine assembly. The combustion chamber, turbine and exhaust section houses the eight combustion chambers which are interconnected by cross-over tubes to allow flame passage and continuous burning in all chambers,



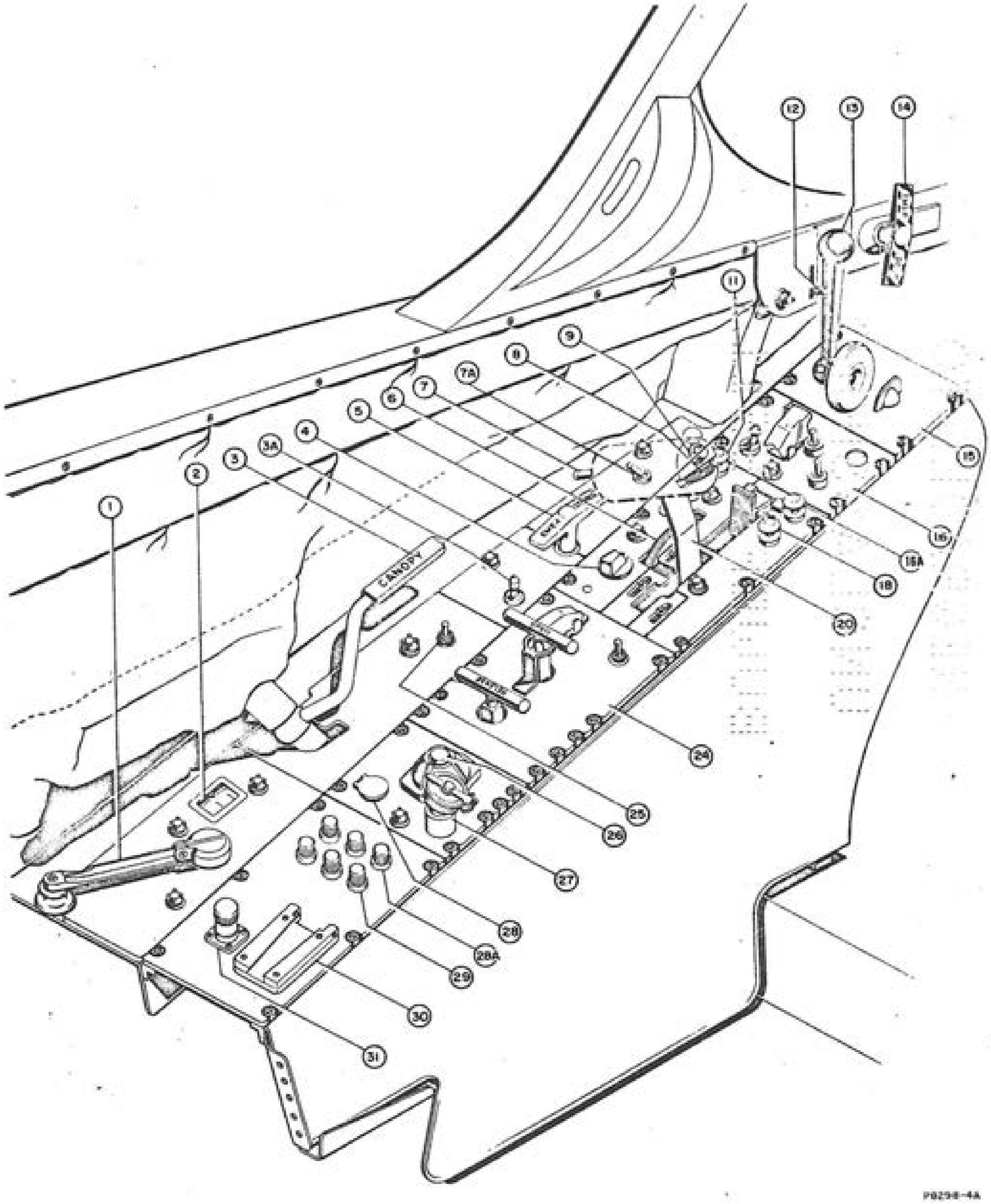
Airplanes BuNo. 142349-142350
Figure 1-3. Cockpit - Left Side (Sheet 1)

Key to figure 1-3, (sheet 2)

1. Mechanical advantage changer control
2. Mechanical advantage changer indicator window
3. Interior canopy control handle
3A. Transonic trim switch
4. Rudder trim control
5. Emergency stores release handle (inoperative)
6. Throttle friction and lock control
7. Master exterior lights switch (temporarily: wire recorder switch)
7A. Automatic systems switch
8. Speedbrake switch
9. Microphone switch
10. (deleted)
11. Engine starter switch
12. Landing gear retraction release control
13. Landing gear control handle
14. Emergency landing gear release handle
15. Face mask temperature control panel
16. Engine control panel
16A. Pitch damper engage switch
17. (deleted)
18. Yaw damper engage switch
19. (deleted)
20. Throttle lever
21. (deleted)
22. (deleted)
23. (deleted)
24. Spin and drag chute control panel
25. Oxygen control
26. Anti-blackout control valve
27. Anti-blackout suit hose receptacle
28. Personnel gear adapter
28A. Fuses for yaw damper, ignition and fuel flow
29. Spare fuses
30. Stowage receptacle (interim) (for pressure suit oxygen regulator)
31. Face mask heater connection

- -

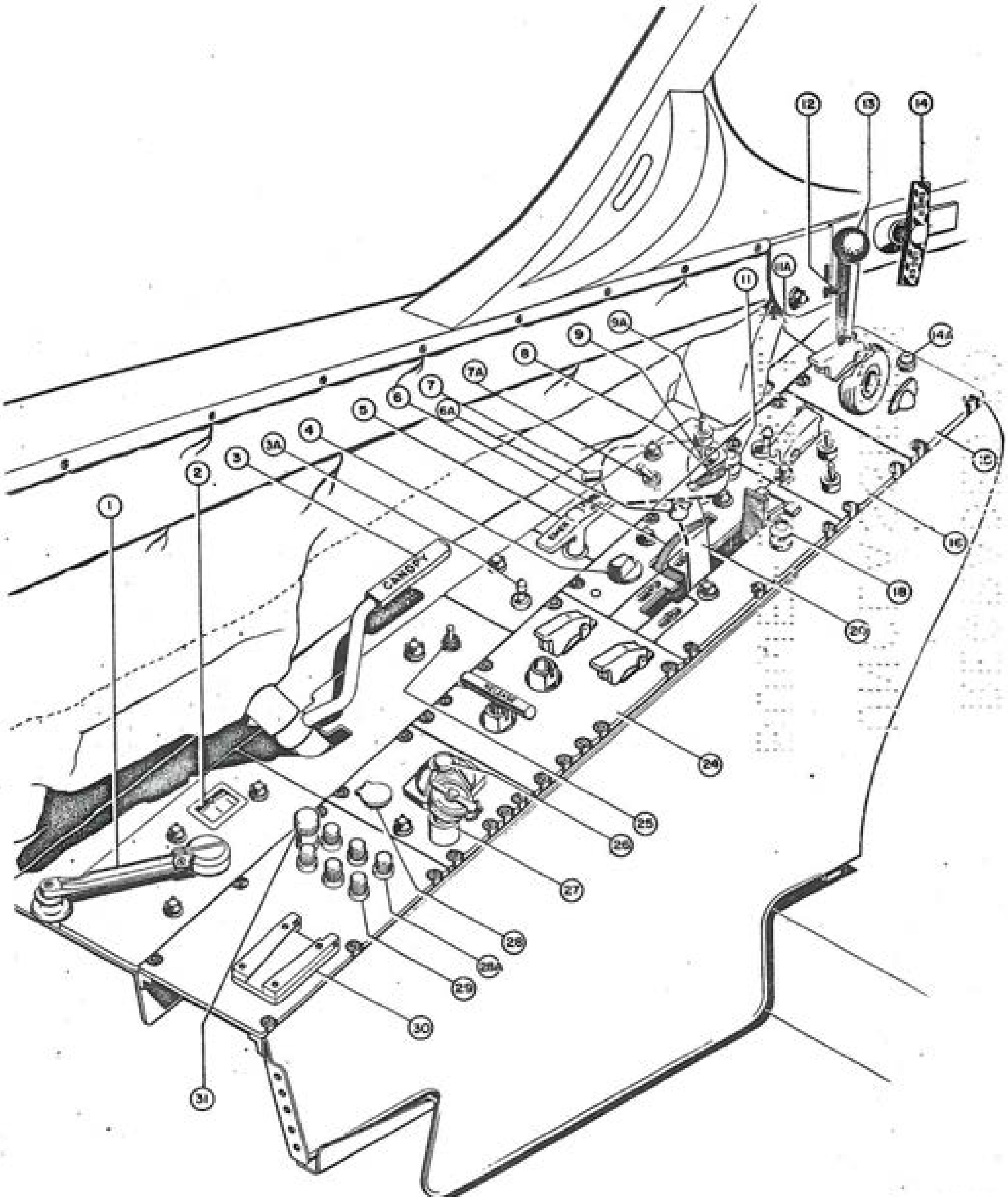
the three stage turbine which drives the compressor units, and the exhaust duct which channels the exhaust gases out of the engine. The main accessory section is located on the engine at the point of smallest diameter, and contains those components necessary for proper operation of the engine. These components include the fuel filter, engine combination fuel pump, fuel control unit, pressurizing and dump valve, oil system pressure and scavenging pumps, oil cooler and reservoir, compressor air bleed valves, spark igniters, and air turbine starter. Power to drive the fuel control unit, fuel and oil pumps, and connection to the air turbine starter is by means of an accessory drive shaft geared to the high pressure compressor rotor.



Airplane BuNo. 139208
Figure 1-3. Cockpit - Left Side (Sheet 2)

Key to figure 1-3, (sheet 3)

1. Mechanical advantage changer control
2. Mechanical advantage changer indicator window
3. Interior canopy control handle
3A. Transonic trim switch
4. Rudder trim control
5. Emergency stores release handle (inoperative)
6. Throttle friction and lock control
6A. Rudder trim control (hydraulic)
7. Master exterior lights switch (temporarily: wire recorder switch)
7A. Automatic systems switch
8. Speedbrake switch
9. Microphone switch
9A. Vertical gyro erection switch (in autopilot) (inoperative)
10. (deleted)
11. Engine starter switch
11A. Rocket doors switch
12. Landing gear retraction release control
13. Landing gear control handle
14. Emergency landing gear release handle
14A. Canopy open light
15. Face mask temperature control panel
16. Engine control panel
17. (deleted)
18. Yaw damper engage switch
19. (deleted)
20. Throttle lever
21. (deleted)
22. (deleted)
23. (deleted)
24. Spin and drag chute controls panel
25. Oxygen control
26. Anti-blackout control valve
27. Anti-blackout suit hose receptacle
28. Personnel gear adapter
28A. Fuses for yaw damper, ignition and fuel flow
29. Spare fuses
30. Stowage receptacle (interim) (for pressure suit oxygen regulator)
31. Face mask heater connection



Airplane BuNo. 139209
Figure 1-3. Cockpit - Left Side (sheet 3)

Key to figure 1-4, (sheet 1)

1. Master warning light
2. Master warning test switch
3. Slats position indicators
4. Gear door gapping indicators
5. Fire warning light
6. Airspeed indicator
7. Remote attitude indicator
8. Accelerometer (standard)
9. Turn and bank indicator
10. Speed brakes indicator
11. Tachometer
12. Altimeter
13. Mach number indicator
14. Accelerometer (flight test)
15. Sump tank fuel quantity gage
16. Liquid oxygen gage
17. Wheels position indicator
18. ARC-27 switch (U.H.F.)
19. Tail pipe temperature indicator
20. Rudder, elevon and M.A.C. indicators
21. ID-250/ARN course indicator
22. Force indicator
23. Total fuel quantity indicator
24. Cabin differential pressure gage
25. Fuel quantity test switch
26. Canopy unlocked light
27. Landing gear position indicator selector
28. Cine switch
29. Pressure ratio gage
30. Oil and fuel pressure gage
31. Oil temperature gage
32. Elevon and utility hydraulic pressure gage
33. Counter
34. (Top row)
    Force gage selector switch
    Remote attitude caging switch
    (Bottom row)
    Ignition on light
    Exhaust nozzle position lights
35. (Top row)
    FM CALIB indicator light
    INSTR FAIL indicator
    OSCIL FAIL indicator
    CAM BLINK indicator
    (Bottom row)
    FM-FM switch
    INSTR switch
    OSCIL switch
    CAM switch
36. Film remaining indicator
37. Stick trim switch
38. Bombs ("B") button
39. Guns-rockets trigger switch
40. Auto control release handle
41. Canopy jettison handle

Also shown: standby compass and mirrors

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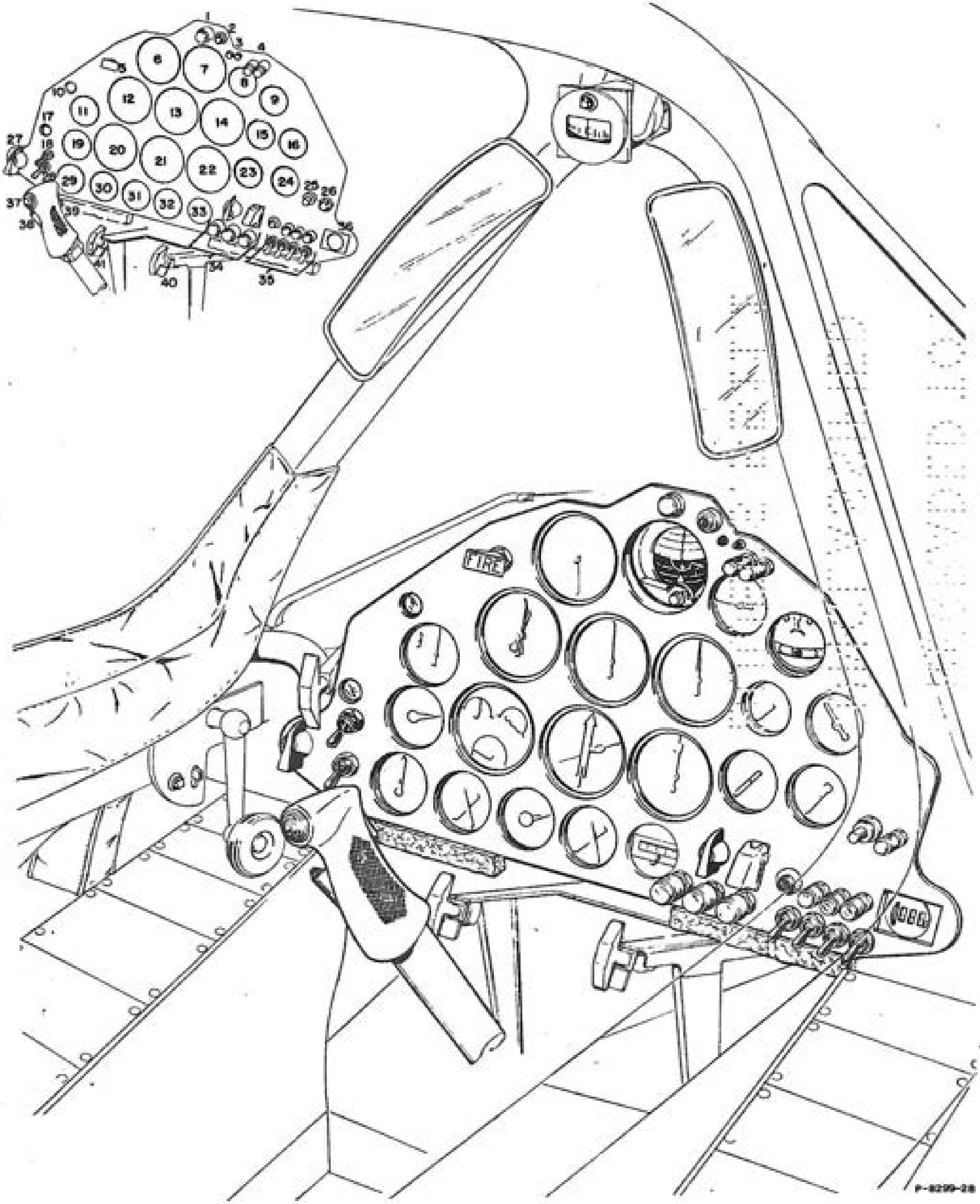
FUEL CONTROL SYSTEM

The basic engine fuel control system consists of a three-stage combination fuel pump and pump stage emergency transfer valve, and a hydro-mechanical fuel control unit. The function of the system is to maintain, automatically, a fixed percentage of trimmed engine speed for any given throttle lever setting.

COMBINATION FUEL PUMP. The combination fuel pump consists of a centrifugal boost stage supplying fuel to two separate gear-type pump stages, one providing fuel for the engine, the other for the afterburner. The emergency transfer valve assembly senses any pressure loss due to failure of the engine fuel supply gear stage and diverts sufficient fuel flow from the afterburner gear stage to the engine fuel control unit to meet engine operating requirements.



Airplanes BuNo. 142349-142350
Figure 1-4. Instrument Panel (sheet 1)

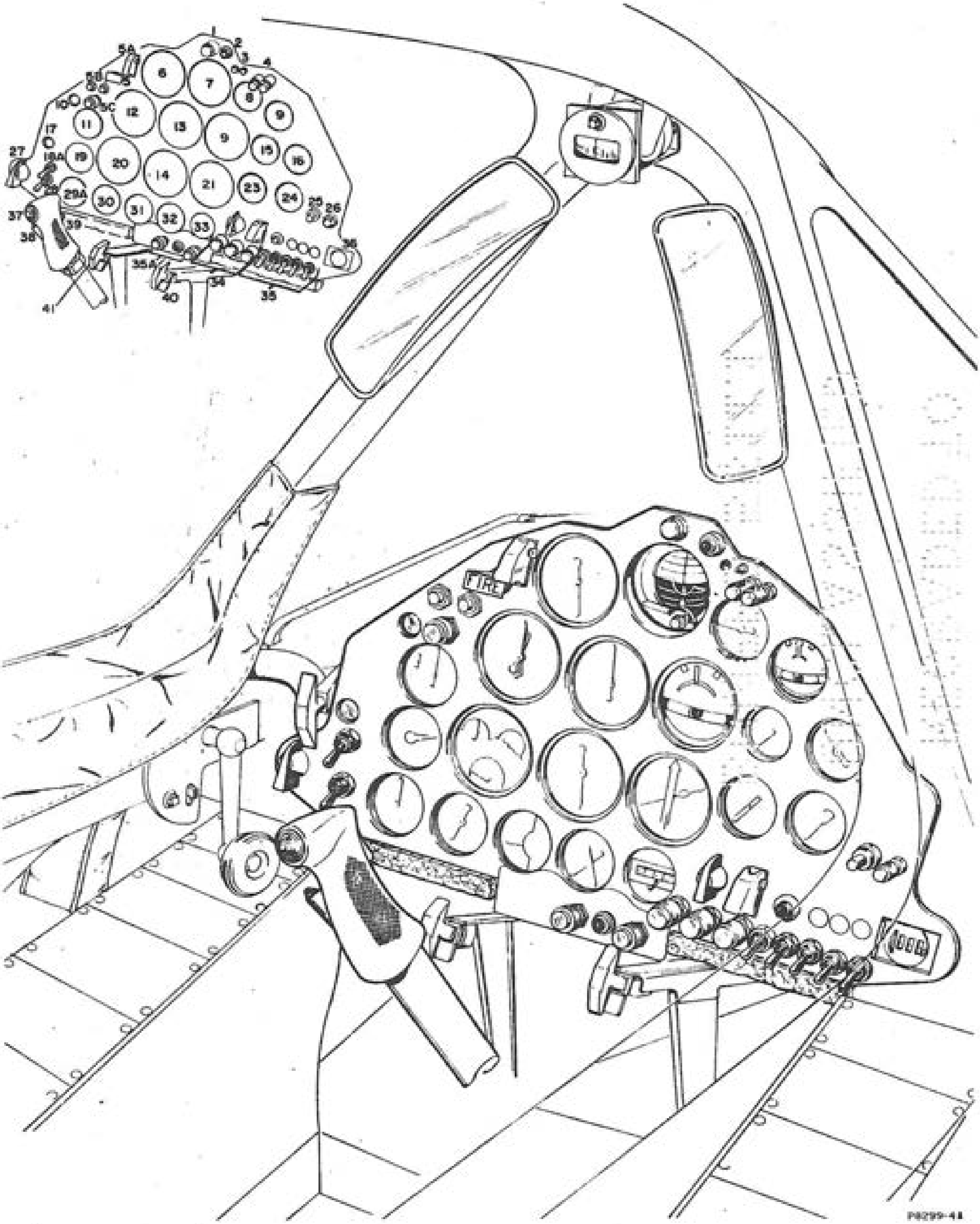
Key to figure 1-4, (sheet 2)

1. Master warning light
2. Master warning test switch
3. Slats position indicators
4. Gear door gapping indicators
5. Fire warning light
5A. Fire extinguisher switch
5B. Fire extinguisher bottles test switches
5C. Fire extinguisher bottles test light
6. Airspeed indicator
7. Remote attitude indicator
8. Accelerometer (standard)
9. Turn and bank indicators
10. Speedbrakes indicator
11. Tachometer
12. Altimeter
13. Mach number indicator
14. Accelerometer (flight-test)
15. Sump tank fuel quantity gage
16. Liquid oxygen gage
17. Wheels position indicator
18. (deleted)
18A. Oscillator correlate switch
19. Tail pipe temperature indicator
20. Rudder, elevon and M.A.C. indicators
21. ID-250/ARN course indicator
22. (deleted)
23. Total fuel quantity indicator
24. Cabin differential pressure gage
25. Fuel quantity test switch
26. Canopy unlocked light
27. Landing gear position indicator selector
28. Cine switch
29. (deleted)
29A. Turbine discharge pressure gage
30. Oil and fuel pressure gage
31. Oil temperature gage
32. Elevon and utility hydraulic pressure gage
33. Counter
34. (Top row)
    Force gage selector switch
    Remote attitude caging switch
    (Bottom row)
    Ignition on light
    Exhaust nozzle position lights
35. (Top row)
    FM CALIB indicator light
    (Bottom row)
    FM-FM switch
    HSR switch (inoperative)
    INSTR switch
    OSCIL switch
    CAM switch
35A. INSTR FAIL indicator
    OSCIL FAIL indicator
    CAM BLINK indicator
36. Film remaining indicator
37. Stick trim switch
38. Bombs ("B") button
39. Guns-rockets trigger switch
40. Auto control release handle
41. Canopy jettison handle

Also shown: standby compass and mirrors

- -

HYDRO-MECHANICAL CONTROL UNIT. The fuel control unit meters fuel in proportion to throttle lever position through a power lever fuel valve within a fuel metering orifice. A positive minimum flow adjustment is provided in this valve. The percent of trimmed engine speed selected by throttle positioning is closely regulated for all operating conditions through the action of a fly-ball governor which governs fuel flow through a servo valve. During acceleration, the hydro-mechanical fuel control senses burner static pressure, engine inlet temperature, and engine speed, combining these variables to limit fuel flow to the maximum allowable for a given engine to prevent



Airplane BuNo. 139208
Figure 1-4. Instrument Panel (sheet 2)

Key to figure 1-4, (sheet 3)

1. Master warning light
1A. Master warning reset button
2. Master warning test switch
3. Slats position indicators
4. Gear door gapping indicators
5. Fire warning light
5A. Fire extinguisher switch
5B. Fire extinguisher bottles test switches
5C. Fire extinguisher bottles test light
6. Airspeed indicator
7. Remote attitude indicator
8. Accelerometer (standard)
9. Turn and bank indicator
10. Speedbrakes indicator
11. Tachometer
12. Altimeter
13. Mach number indicator
14. Accelerometer (flight-test)
15. Sump tank fuel quantity gage
16. Liquid oxygen gage
16A. LOX 'push-for-aux-bottle' switch
17. Wheels position indicator
18. (deleted)
18A. Oscillator correlate switch
19. Tail pipe temperature indicator
20. Rudder, elevon and M.A.C. indicators
21. Rate-of roll indicator (or) ID-250/ARN course indicator
22. Force indicator
23. Total fuel quantity indicator
24. Cabin differential pressure gage
25. Fuel quantity test switch
26. (deleted)
27. Landing gear position indicator selector
28. Cine switch
29. (deleted)
29A. Turbine discharge pressure gage
30. Oil and fuel pressure gage
31. Oil temperature gage
32. Elevon and utility hydraulic pressure gage
33. Counter
34. (Top row)
    Force gage selector switch
    Remote attitude caging switch
    (Bottom row)
    Ignition on light
    Exhaust nozzle position lights
35. (Top row)
    FM CALIB indicator light
    (Bottom row)
    FM-FM switch
    HSR switch (inoperative)
    INSTR switch
    OSCIL switch
    CAM switch
35A. INSTR FAIL indicator
    OSCIL FAIL indicator
    CAM BLINK indicator
36. Film remaining indicator
37. Stick trim switch
38. Bombs ("B") button
39. Guns-rockets trigger switch
40. Auto control release handle
41. Canopy jettison handle
42. "GSAP" camera switch

Also shown: standby compass and mirrors

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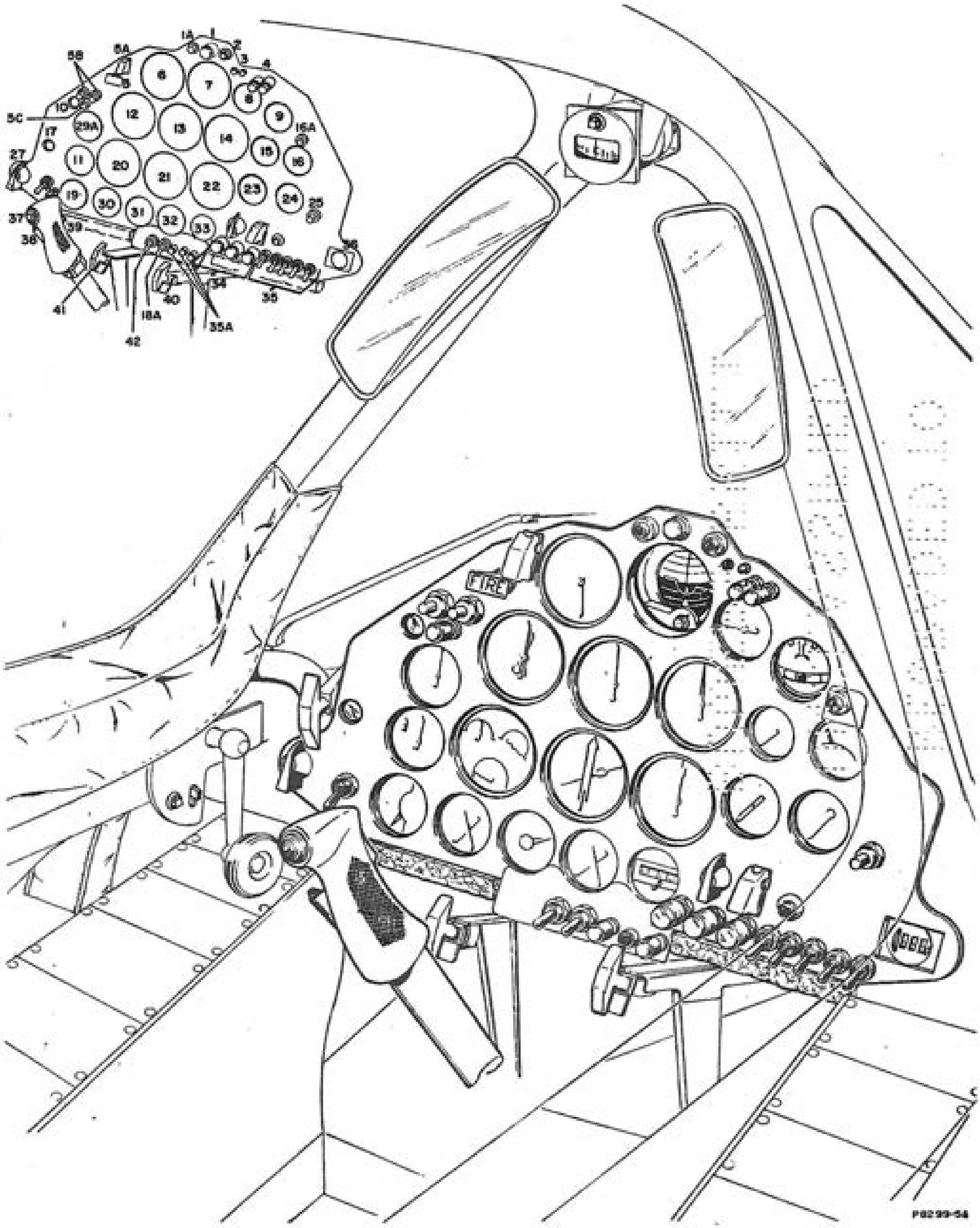
compressor surge and over-temperaturing. When operating at a steady state for a given throttle setting, the hydro-mechanical unit compensates automatically for variations in altitude, airspeed, compressor inlet temperature, engine speed, and burner pressure to prevent overspeeding, over-temperaturing, and compressor surge. During deceleration the control unit schedules a minimum fuel flow as a function of burner pressure to prevent lean "blow-out" by limiting the rate of reduction of fuel flow to a flow sufficient to sustain



Airplane BuNo. 139209
Figure 1-4. Instrument Panel (sheet 3)

Key to figure 1-5, (sheet 1)

1. Arresting hook control
2. Seat switch
3. Emergency generator and hydraulic pump release handle
4. Air conditioning switch
5. Master warning annunciator
6. Emergency electric hydraulic pumps switch
7. Spare fuses
8. Spare instrument lamps
9. Spare console lamps
10. Manual fuel valve control
11. Interphone outlets
12. MA-1 compass controller
13. (deleted)
14. Impulse generator control panel
15. (deleted)
16. Air conditioning control panel
17. ARC-27A (UHF) control panel
18. Auxiliary fuel selector switch
19. (deleted)
20. Auxiliary fuel quantity indicators
21. Fuel quantity push-to-test switch

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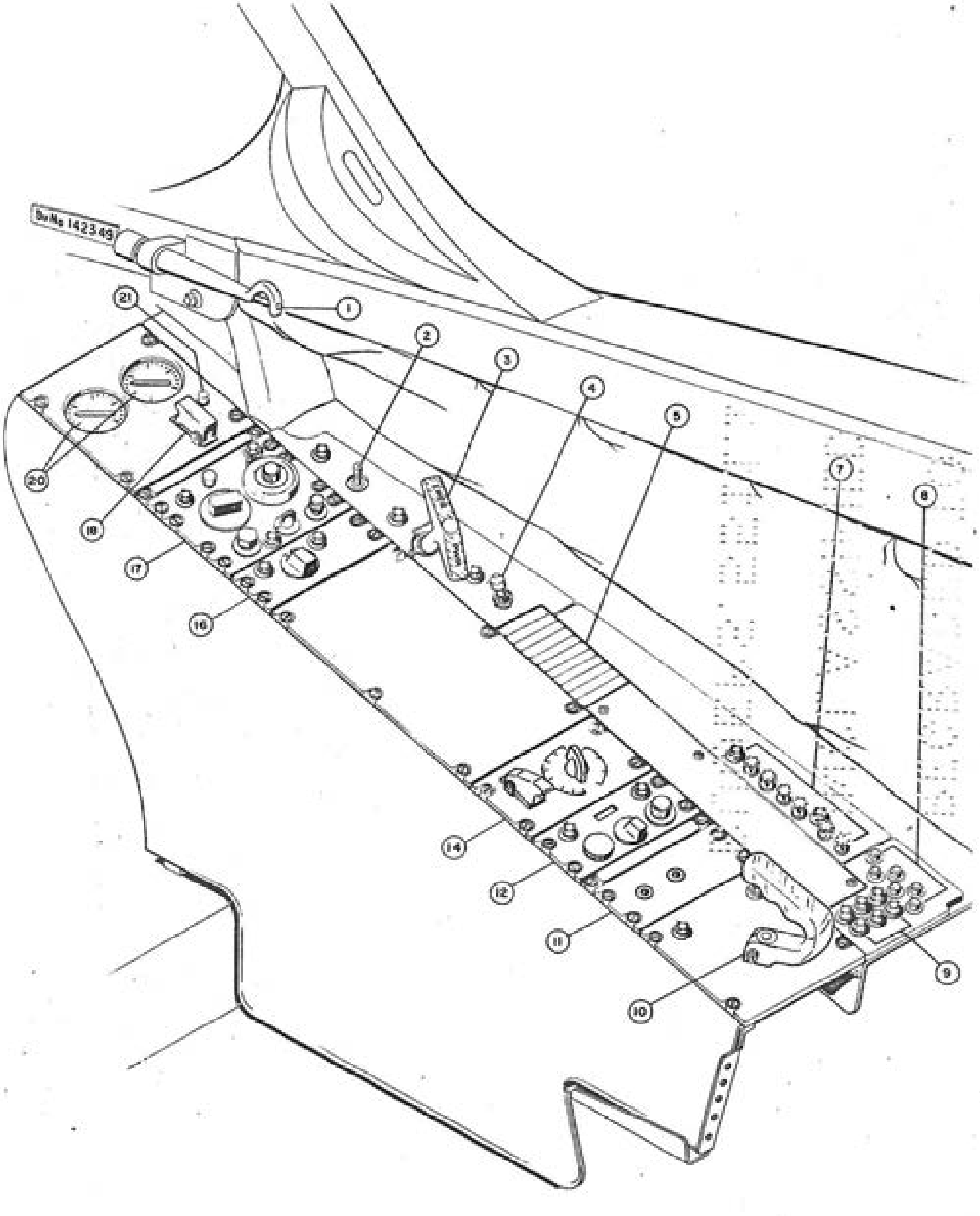
combustion. This minimum flow schedule also provides a low limit of fuel flow for altitude idle and the starting operation. A manual fuel control system is incorporated into the fuel control unit as a safety feature in the event of automatic fuel control malfunction. The manual system may be electrically selected by the pilot through means of a switch in the cockpit. When the manual fuel control system is selected, an emergency actuating solenoid on the fuel control unit isolates the automatic fuel metering unit and all fuel metering is then controlled manually by a throttling valve mechanically linked to the throttle in the cockpit. This throttling valve also acts as a fuel shutoff during engine shut-down. The hydro-mechanical fuel control unit employed on J57-P-8 engines does not incorporate either an automatic emergency fuel control transfer system, or minimum fuel flow locks armed by the "TAKE OFF" position of the throttle.

FUEL CONTROL SYSTEM SELECTOR SWITCH. The engine fuel control system selector switch, labeled ENG CONT (16, figure 1-3), controls selection of either the automatic or the manual fuel system. In the "PRIMARY" position, the switch selects the automatic fuel control system which governs the percent of trimmed engine speed selected by the pilot. When placed in the "MANUAL" position, the switch activates an emergency actuating solenoid which isolates the automatic fuel control system and places the pilot in manual control of fuel metering through a mechanical linkage from the throttle to the emergency throttling valve in the fuel control unit. An EMERG FUEL



Airplanes BuNo. 142349-142350
Figure 1-5. Cockpit - Right Side (sheet 1)

Key to figure 1-5, (sheet 2)

1. Arresting hook control
2. Seat switch
2A. Cabin air temperature switch
3. Emergency generator and hydraulic pump release handle
4. Air conditioning switch
5. Master warning annunciator
6A. (blank)
7. Spare fuses
8. Spare instrument lamps
9. Spare console lamps
10. (deleted)
10A. (blank)
11. (deleted)
12. MA-1 compass controller
13. (deleted)
13A. External lights control panel (inoperative)
14. Impulse generator control panel
15. (deleted)
16. Air conditioning control panel
17. ARC-27A (UHF) control panel
18. Auxiliary fuel selector switch
19. (deleted)
20. Auxiliary fuel quantity indicators
21. Fuel quantity push-to-test switch

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control warning light (5, figure 1-5) is provided on the right-hand console. This light illuminates whenever the ENG CONT switch is in the "MANUAL" position.

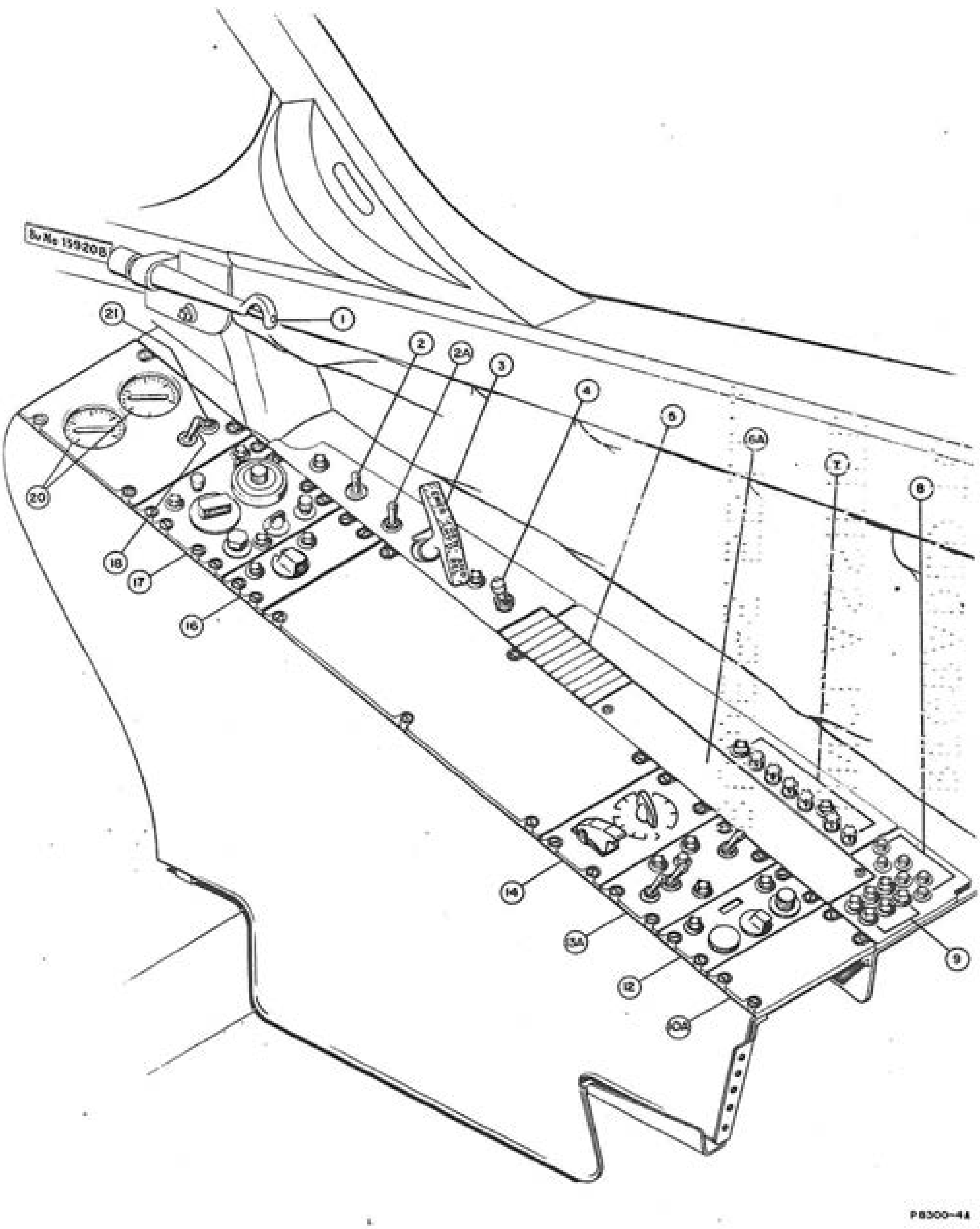
CAUTION

> When operating on the manual fuel control system, the throttle must be moved slowly and smoothly, observing engine instruments, to prevent "lean-out," "rich-out," over speeding, or over-temperaturing.

PUMP FAILURE WARNING LIGHT. The FUEL PUMP failure warning light (5, figure 1-5) located on the right-hand console, illuminates to indicate failure of the engine stage of the engine combination fuel pump. The illumination of this warning light will be the only indication of such failure since transfer to the afterburner stage is accomplished automatically by the pump-stage emergency transfer valve.

THROTTLE

The throttle (20, figure 1-3), located on the left-hand console, establishes the power level for which the fuel control unit meters



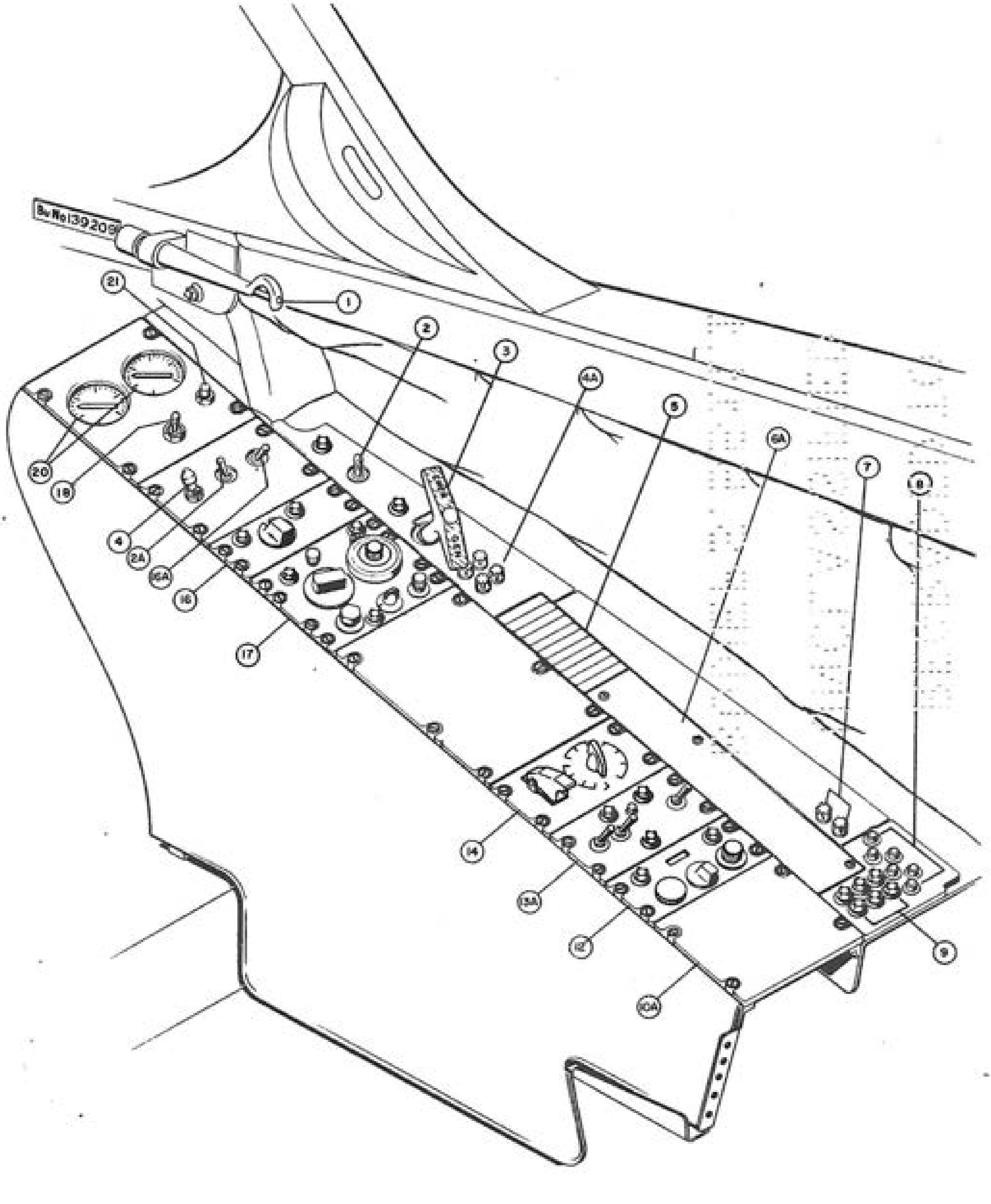
Airplane BuNo. 139208
Figure 1-5. Cockpit - Right Side (sheet 2)

Key to figure 1-5, (sheet 3)

1. Arresting hook control
2. Seat switch
2A. Cabin air temperature switch
3. EMERgency GENerator release handle
4. Air conditioning switch
4A. Fuses
5. Master warning annunciator
6A. (blank)
7. Spare fuses
8. Spare instrument lamps
9. Spare console lamps
10. (deleted)
10A. (blank)
11. (deleted)
12. MA-1 compass controller
13. (deleted)
13A. External lights control panel (inoperative)
14. Impulse generator control panel
15. (deleted)
16. Air conditioning control panel
16A. Windshield defog switch
17. ARC-27A (UHF) control panel
18. Auxiliary fuel selector switch
19. (deleted)
20. Auxiliary fuel quantity indicators
21. Fuel quantity push-to-test switch

- -

fuel to the engine. The throttle mechanism incorporates an "off" position which closes the emergency throttling and shut-off valve, depriving the engine of all fuel and opening the engine fuel manifold drain valve; an "IGNITE" position which is entered by moving the throttle outboard from the "off" position, thus closing a switch which energizes the engine ignition system; an "idle" position, guarded by a stop device to prevent the throttle from being moved inadvertantly to "off"; and a transitional range from idle rpm to maximum rpm. To activate the afterburner, the throttle is moved outboard after passing forward of the afterburner gate, approximately at the mid-point of total throttle lever travel. This action actuates the afterburner switch and simultaneously engages the afterburner detent which holds the throttle in the outboard position without requiring the attention of the pilot during take-off. At the extreme forward end of the throttle quadrant is the "TAKE OFF" position. The afterburner detent may be engaged or disengaged at any point between the afterburner gate and the "TAKE OFF" position. A limit switch located in the throttle linkage is actuated whenever the throttle is fully back in the "idle" position. This switch serves to energize a solenoid valve, causing the afterburner nozzle to "pop-open" to reduce thrust during ground operations. Incorporated into the throttle handgrip are three switches: a two position



P8300-5A

Airplane BuNo. 139209
Figure 1-5. Cockpit - Right Side (sheet 3)

SPEEDBRAKE switch; a radio transmit MIC switch located on the inboard end of the grip; and a master EXT LIGHTS switch located on the outboard end of the grip.

FRICTION WHEEL. A throttle lever friction control wheel labeled THROTTLE FRICTION & LOCK (6, figure 1-3), controls the force required to move the throttle lever from one position to another. When the throttle lever friction device is fully engaged, full pilot effort will not move the throttle.

MASTER ENGINE SWITCH

The ENGINE MASTER switch (16, figure 1-3) is located on the ENGINE control panel on the left-hand console. The switch has two positions; "ON" and "OFF." The "ON" position of the ENGINE MASTER switch energizes the engine fuel control circuit through the ENG CONT switch, the a-c electrical fuel boost pump if a-c electrical power is available, and arms the FUEL PUMP FAILURE warning light circuit, PITOT & ENG ANTI-ICING switch, afterburner switch, ignition switch, and ignition timer. The ENGINE MASTER switch receives power from the primary d-c electrical bus through the ENGINE CONTROL fuse in the equipment compartment.

ENGINE ANTI-ICING SWITCH

A two position toggle switch, labeled PITOT & ENG ANTI-ICING (16, figure 1-3), controls operation of the engine anti-icing system. Placing the switch in the "ON" position actuates a valve which allows high temperature compressor discharge air to flow through external tubes on each side of the engine to the compressor case and inlet guide vanes. An anti-icing air regulator and air valve are incorporated in each tube system. The regulator automatically controls the volume of air flowing through the tubes, permitting maximum flow at 70°F bleed air temperature and minimum at 560°F. The "ON" position of this switch also energizes the electrical heating elements in the pitot static pressure tube head.

ENGINE INSTRUMENTS

PRESSURE RATIO INDICATOR. (1) A pressure ratio indicator (29, figure 1-4), located on the instrument panel, is provided to indicate the ratio of tailpipe (Pt 7) pressure to pressure at the intake (Pt 0), as a means of checking take-off thrust at military power. The

(1) Airplanes BuNo. 142349-142350

instrument is graduated from 1.2 to 3.4. A knob on the lower left-hand side of the instrument operates a counter dial and simultaneously moves an index pointer which travels around the perimeter of the instrument face. The knob is turned until the minimum acceptable take-off pressure ratio for ambient conditions is displayed on the dial by the index pointer. When take-off power is applied, the needle should advance around the dial to coincide with the index pointer at the moment 97% of military power is delivered. APPROX

TAIL PIPE PRESSURE GAGE. (1) A turbine discharge out pressure gage (29A, figure 1-4) marked "TAIL PIPE PRESSURE" indicates the turbine discharge out pressure in inches of mercury when the reading is multiplied by ten.

TURBINE OUTLET TEMPERATURE INDICATOR. The turbine outlet temperature indicator (19, figure 1-4) is mounted on the instrument panel. This instrument indicates in hundreds of degrees centrigrade the temperature of engine exhaust gases immediately downstream of the turbine assembly.

TACHOMETER. An engine tachometer (11, figure 1-4) is located on the instrument panel. This instrument reflects the speed of the high pressure compressor rotor expressed as a percentage of 9,976 rpm. The maximum rated rpm of any individual engine is stamped on the engine data plate.

ENGINE COOLING

Air flowing from the front and rear compressers through the turbine and exhaust sections serves to cool the engine internally. Ambient air is directed through the engine compartment and vented overboard through three exit ducts at the vapor seal between the compressor and turbine sections of the engine to cool and ventilate the accessory section. Air from the engine oil cooler intake is bled off to ventilate and cool the engine compartment aft of the vapor seal in the burner, turbine, and afterburner section. This air is ducted overboard through the tailcone. In addition, air bled from the air intake ducts is passed between the accessory compartment insulating blankets and surrounding structure, by-passing the vapor seal, to ventilate the vapor compartment between the insulating blankets and structure of the burner, turbine, and afterburner compartment. This air is ducted overboard through two exit ramps just forward of the tail cone. For afterburner cooling provisions, refer to AFTER-BURNER in this section.

(1) Airplanes BuNo. 139208-139209.

Key to figure 1-6

1. Firing mechanism for ejection seat catapult
2. Safety pin lanyard (shown schematic)
3. Safety pin
4. Ejection control (face curtain) handle
5. Headrest
6. Ripcord handle (manual "D-ring")
7. Left suit shoulder strap connection
8. Parachute
9. Oxygen and radio hose (mask connection)
10. Oxygen and radio hose (console connection)
11. Shoulder harness inertia reel control
12. Left seat belt connection
13. Auxiliary front handle ejection control ("D-ring")
14. Emergency oxygen bottle actuating handle ("green-apple")
15. Emergency oxygen bottle pressure gage
16. Personnel Emergency equipment (seat pan and pararaft kit)
17. Harness release handle
18. Barometric opening parachute arming lanyard
19. Right seat belt connection
20. Right suit shoulder strap connection
21. Inertia reel connection

IGNITION SYSTEM

The engine ignition system consists of two spark igniters and an ignition timer. The spark igniters are located in the lower two combustion chambers in the burner section of the engine. The ignition timer energizes the igniters for a firing cycle of 30 seconds duration when activated by the ignition switch.

IGNITION SWITCH. The ignition switch is incorporated in the throttle lever assembly. This is a momentary contact limit switch and is actuated by outboard movement of the throttle lever when the throttle is in the "off" position. The ignition switch completes a circuit from the primary d-c electrical bus, through the ENGINE MASTER switch, to the ignition timer.

ENGINE STARTER SYSTEM

An air turbine starter is located on the engine accessories section. The starter is powered by air pressure from an external GTC starter unit, and motors the high pressure compressor rotor through the accessories drive shaft. Control of the external starter unit is accomplished by a shut-off valve in the GTC starter unit and a limit switch in the engine starter itself. The shut-off valve is controlled by the START switch in the aircraft. This push-button type switch opens the shut-off valve in the starter unit through the limit switch, which then holds the shut-off valve open until the engine reaches approximately 50% rpm. At this speed the limit switch opens, the shut off valve closes, and the START switch disengages.

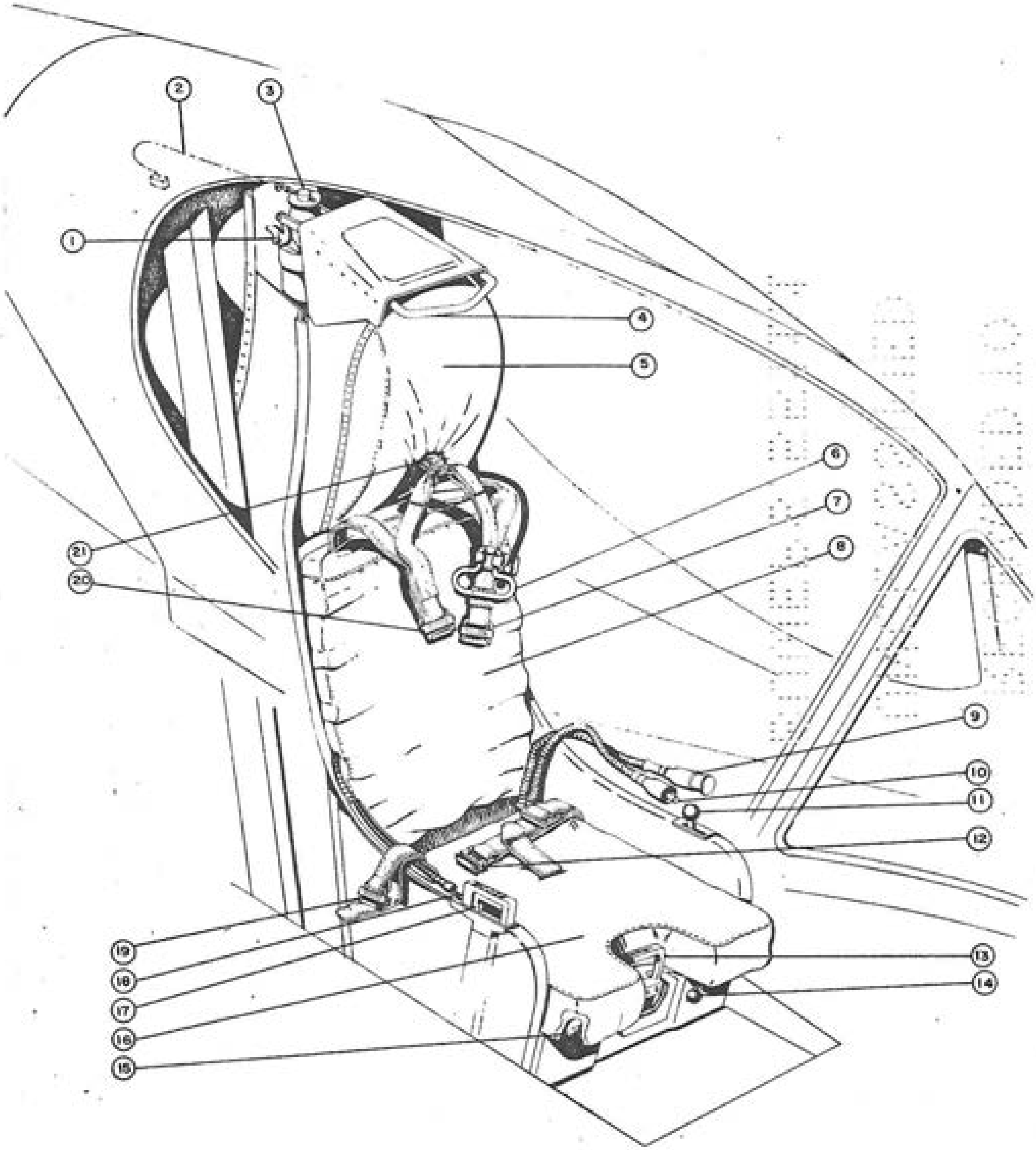


Figure 1-6. Cockpit - Ejection Seat

STARTER SWITCH. The push-button type START switch (11, figure 1-3) on the left-hand console controls the starting operation. Depressing the START switch with the external GTC starter electrical cable connected to the airplane locks the switch down and opens the starter air shut-off valve. When the engine reaches approximately 50% rpm the START switch automatically pops up. The starting operation may be discontinued at any time by manually pulling up on the START switch.

CAUTION

> To avoid damage to the airplane starter, do not use external GTC starting units which provide air at temperatures in excess of 375°F.

AFTERBURNER

An afterburner is attached to the rear of the turbine case of the engine to provide thrust augmentation. An inner liner in the afterburner is perforated at the forward end to allow engine exhaust gases to pass between the inner liner and outer shell of the afterburner for cooling purposes. A two position, variable-area exhaust nozzle is attached to the rear of the afterburner exhaust duct to provide the proper exhaust orifice size for either normal or afterburning engine operation. The afterburner system incorporates a separate fuel control and igniter system.

AFTERBURNER FUEL CONTROL

The afterburner fuel control system becomes separate from the engine fuel control system at a point in the combination fuel pump. In addition to the afterburner gear stage of the combination fuel pump, the system includes the afterburner fuel control unit, fuel nozzles, and afterburner switch. When the throttle is moved outboard, engaging the afterburner detent, a switch is actuated which causes the afterburner gear stage of the engine combination fuel pump to deliver fuel to the afterburner fuel control unit which then meters fuel as necessary to the afterburner fuel nozzles as a function of compressor discharge pressure.

AFTERBURNER SWITCH. The afterburner switch is located in the throttle quadrant at the outboard side of the throttle lever slot between the afterburner gate and the "TAKE OFF" position. To activate this switch, the throttle lever is moved outboard to the afterburner position at any desired point forward of the afterburner gate. This switch energizes a valve in the engine combination fuel pump, allowing fuel to be delivered to the afterburner fuel control unit.

AFTERBURNER IGNITER CONTROL

The afterburner igniter control is actuated by fuel pressure from the afterburner fuel control unit. The igniter injects a stream of raw fuel into the engine upstream of the turbine assembly where it ignites and passes into the afterburner to ignite the fuel issuing from the afterburner fuel nozzles.

EXHAUST NOZZLE CONTROL

The exhaust nozzle control is actuated by fuel pressure from the afterburner stage of the engine combination fuel pump. Fuel pressure arriving at the nozzle control moves a piston, porting high pressure compressor bleed air to the exhaust nozzle actuating cylinders, causing the nozzle to open or close as required. A "pop-open" nozzle control, operated by a limit switch in the throttle linkage and a solenoid on the exhaust nozzle control, is installed to cause the afterburner nozzle to open without lighting-off the afterburner. This system ports high pressure compressor bleed air to the nozzle actuating cylinders when the throttle is in the "idle" position by mechanical and electrical means, rather than by afterburner fuel pressure, in order to reduce thrust during ground operations.

<u>OIL SYSTEM</u>

The engine is lubricated and partially cooled by an automatic oil system. An oil tank is located on the upper left side of the engine at the point of smallest diameter. The tank has a total capacity of 5.5 U. S. gallons, of which 3.0 gallons are usable, 1.0 gallon is reserve, and 1.5 gallons remain in the system when the engine is not operating. Expansion space for 1.6 gallons is provided. Refer to Table I at the end of this section for engine oil grade and specification. An engine driven oil pressure pump and five gear-type scavenger pumps provide constant circulation of oil to lubricate the six main bearings and other components of the engine. Scavenged oil is filtered and routed through both a ram air cooler and fuel-oil cooler before re-entering the system. Maximum oil consumption is estimated to be approximately four pounds per hour at normal rated thrust.

OIL PRESSURE INDICATOR. An oil pressure indicator (30, figure 1-4) on the instrument panel reflects the differential pressure between system oil pump discharge pressure and engine accessory compartment pressure. When the system is operating normally, a pressure relief valve maintains a relatively constant 45 psi differential oil pressure at the bearing orifices. The indicator is a dual instrument labeled PRESSURE, which shows both oil and fuel pressure in pounds per square inch. Oil pressure is read opposite the needle displaying the letter "O".

OIL TEMPERATURE INDICATOR. An OIL temperature indicator (31, figure 1-4) on the instrument panel reflects the temperature of engine oil in degrees centigrade.

FUEL SYSTEM

The engine fuel supply is carried in six fuel tanks. These tanks may be serviced through a two point pressure fueling system, (1) or by means of six gravity fuel tank filler caps, (2) one located at the top of each tank. Later aircraft (1) incorporate provisions for carrying two 300 gallon external fuel tanks, one on each side just inboard of the main landing gear doors. A fuel transfer system is provided, allowing fuel to be transferred through a piping system by means of tank pressurization and air driven fuel pumps. All fuel is transferred into the aft fuselage sump tank containing an electrically driven fuel boost pump which delivers fuel under pressure to the engine fuel pump. A manual fuel supply shut-off valve control is provided in the cockpit. Refer to Table I at the end of this section for fuel grades and specifications of recommended, alternate, and emergency fuels.

FUEL QUANTITY DATA

GALLONS

TANKS	NUMBER		USABLE FUEL (EACH)	UNUSABLE FUEL-LEVEL FLIGHT (EACH)	EXPANSION SPACE (EACH)	TOTAL VOLUME (EACH)
AFT FUSELAGE SUMP	1	G	185.0	0.0	4.0	189.0
		P	184.5	0.0	4.5	189.0
FORWARD WING	2	G	281.5	0.3	1.2	283.0
		P	278.2	0.3	4.5	283.0
AFT WING AUXILIARY	2	G	168.0	1.0	2.0	171.0
		P	165.0	1.0	5.0	171.0
FORWARD FUSELAGE AUXILIARY	1	G	265.0	0.5	6.5	272.0
		P	264.5	0.5	7.0	272.0
DROP	2	G	300.0	*	*	300.0
		P	300.0	*	*	300.0

G = GRAVITY FUELING
P = PRESSURE FUELING

USABLE FUEL TOTALS	G	P
FORWARD WING AND AFT FUSELAGE SUMP TANKS	748.0	740.9
FORWARD WING, AFT FUSELAGE SUMP, AFT WING, AND FORWARD FUSELAGE TANKS	1349.0	1335.4
FORWARD WING, AFT FUSELAGE SUMP, AFT WING, FORWARD FUSELAGE, AND DROP TANKS	1949.0	1935.4

* INFORMATION TO BE ADDED
DATA BASIS: CALCULATIONS

P6302-1A

Figure 1-7. Fuel Quantity Data Table

(1) Airplanes BuNo. 142349-142350.
(2) Airplanes BuNo. 139208-139209.

INTERNAL FUEL TANKS

The six fuel tanks provided in the airplane are comprised of four integral tanks in the center wing, two on each side of the engine bay, and two bladder-type tanks in the fuselage between the equipment compartment and engine bay. The four integral center wing tanks are comprised of a left and right forward wing tank and a left and right aft wing tank. The two forward wing tanks are intended to carry fuel on all flights, while the two aft wing tanks are considered as auxiliary tanks and will not normally be fueled except for fighter or ferry missions. The two bladder-type fuselage tanks consist of the aft fuselage sump tank and the forward fuselage tank. The aft fuselage sump tank contains the fuel level control valves regulating transfer of fuel from all other tanks, and the fuel boost pump which delivers fuel to the engine. All fuel aboard the airplane is transferred to this tank before delivery to the engine. The forward fuselage fuel tank is considered as an auxiliary tank and will not normally be fueled except for fighter or ferry missions. All six tanks incorporate provisions for fuel transfer, pressure fueling and de-fueling, (1) gravity filling, (2) and water and sediment drainage. For information concerning total and usable fuel capacities of each tank, refer to figure 1-7.

FUEL TANK PRESSURIZING AND VENTING

All fuel tanks are vented, and the forward fuselage and aft wing tanks pressurized, through an inter-connected pressurizing and venting system. Pressurization of the forward fuselage and aft wing tanks (auxiliary tanks) is accomplished by piping low pressure engine compressor bleed air to the pressurizing and venting system air valve (view "A", figure 1-8) which distributes the air to the tanks to effect fuel transfer. A solenoid-operated internal transfer three-way air valve controls the flow of air to the distributing air valve in response to signals from the pilot-operated AUX FUEL TRANS switch. When in operation, the system delivers air at six psi to the auxiliary tanks. The distributing air valve incorporates a relief valve to relieve any pressure in excess of 7 1/2 psi which may occur during a climb. A ground compressed air supply connection is provided in the system to facilitate pressure defueling of the auxiliary fuel tanks. The aft fuselage sump and forward wing tanks are vented at all times, and the forward fuselage and aft wing tanks when fuel is not being transferred, through the distributing air valve and vent system overboard pipe. The vent system exit (25, figure 1-2), located on the bottom of the center wing just forward of the right-hand inboard elevon, is heated by high pressure compressor bleed air to prevent the formation of ice or frost. The vent exit is designed to provide a small positive differential in fuel vent system pressure over ambient pressure to preclude the possibility of the bladder-type fuselage fuel tanks collapsing as a result of negative internal pressures. The fuselage fuel tanks cavity (surrounding support structure) is ventilated by ambient air

(1) Airplanes BuNo. 142349-142350.
(2) Airplanes BuNo. 139208-139209.

entering the fuselage behind each intake scoop boundary layer separator blade, and leaving at the left-hand catapult hook. This venting system is designed to maintain a slight negative pressure around the fuselage fuel tanks to assist in preventing collapse of the tanks when partially full or empty. In addition, a flapper valve is incorporated in the distributing air valve to provide rapid pressure differential relief for the forward fuselage fuel tank during high speed dives.

FUEL TRANSFER

Fuel transfer from the auxiliary tanks and the forward wing tanks to the aft fuselage sump tank is accomplished by engine bleed air tank pressurization and engine bleed air driven fuel transfer pumps respectively. Pressurization of the forward fuselage and aft wing tanks is controlled by the three-way internal transfer air valve. This valve is a solenoid operated valve controlled by the AUX FUEL TRANS switch, which opens to pressurize the auxiliary tanks and closes to de-pressurize the auxiliary tanks. When pressurized, the auxiliary tanks fuel supply is transferred through piping and check valves (to prevent inter-transfer between auxiliary tanks) to the aft fuselage sump tank. Two fuel level control valves in the auxiliary fuel transfer system control the level of fuel in the aft fuselage sump tank during the transfer operation to prevent transfer fuel from being pumped overboard through the aft fuselage sump tank vent. These two valves also serve to permit pilot selected transfer of auxiliary fuel from the forward fuselage tank or the aft wing tanks, or both. Transfer of fuel from the forward wing tanks to the aft fuselage sump tank is controlled by a third fuel level control valve, located slightly lower in the sump tank than the auxiliary fuel transfer control valves. At any time that the fuel level in the sump tank falls from the auxiliary fuel transfer level to the forward wing tank transfer level, the control valve opens to admit forward wing tanks transfer fuel. The forward wing tanks air driven transfer pumps operate at all times when the engine is running, therefore the transfer of forward wing tanks fuel is automatic, and no control of this portion of the fuel transfer system by the pilot is possible. Like the auxiliary fuel transfer system, the forward wing tanks transfer system incorporates check valves to prevent inter-transfer of fuel and resultant undesirable asymmetrical loadings. Transfer of fuel from all tanks to the aft fuselage sump tank is at a rate sufficient to meet or exceed engine fuel flow requirements at military power plus afterburning.

AUXILIARY FUEL TRANSFER CONTROL. The AUX FUEL TRANS switch (16, figure 1-3) on the pilot's left-hand console controls the transfer of auxiliary fuel when carried. The switch has three positions; "OFF," "INTERNAL," and "DROP." In the "OFF" position of the switch the solenoid of the internal three way valve is energized, holding the normally-open valve in the closed position, thus depriving the forward fuselage and aft wing tanks of the pressurized air required to effect fuel transfer. The "INTERNAL" position of the AUX FUEL TRANS switch breaks the electrical circuit from the d-c primary bus to the internal transfer three way air valve, permitting the valve to open and the auxiliary fuel tanks to become pressurized. The "DROP" position of the AUX FUEL TRANS switch breaks the electrical circuit from the d-c primary bus to the drop tanks transfer three way air valve, (1) permitting the valve to open and the drop tanks to become pressurized to effect drop tansks fuel transfer (refer to DROP TANKS, in this section). The AUX FUEL TRANS switch is also tied into the fuel quantity indicating system. For a description of this function, refer to FUEL QUANTITY INDICATING SYSTEM in this section.

The AUX FUEL SELECTOR switch (18, figure 1-5) on the right-hand console controls selection of fuel transfer from the forward fuselage tank or aft wing tanks, or both, when the AUX FUEL TRANS switch is in the "INTERNAL" position.

The AUX FUEL SELECTOR switch has three positions; "FWD ONLY", "BOTH", and "AFT ONLY." In the "BOTH" position of the switch, two solenoid-controlled float valves contained in one of a pair of dual float valves in the aft fuselage sump tank (which control the auxiliary fuel transfer control valves) are de-energized; permitting forward fuselage and aft wing tanks auxiliary fuel to transfer to the sump tank. In the "FWD ONLY" position of the switch, the solenoid-controlled float valve which shuts off the aft wing tanks auxiliary fuel control valve is energized, thereby stopping the flow of fuel from the aft wing tanks to the aft fuselage sump tank. In the "AFT ONLY" position of the switch, the solenoid-controlled float valve which shuts off the forward fuselage tank auxiliary fuel control valve is energized, thereby stopping the flow of fuel from the forward fuselage tank to the aft fuselage sump tank. The other dual float valve prevents auxiliary fuel from being pumped overboard through the aft fuselage sump tank vent in the event of failure of the solenoid-controlled dual float valve.

Note

> The AUX FUEL SELECTOR switch is inoperative when the AUX FUEL TRANS switch is in either the "OFF" or the "DROP" position.

FUEL BOOST PUMP

An electrically driven fuel boost pump, powered by the a-c monitored bus, is submerged in the aft fuselage sump tank. The fuel boost pump incorporates both a normal and an inverted flight fuel inlet and will deliver fuel to the engine combination fuel pump at a boost pressure of approximately 30-40 psi. Operation of the fuel boost pump is controlled by the ENGINE MASTER switch.

(1) Airplanes BuNo. 142349-142350.

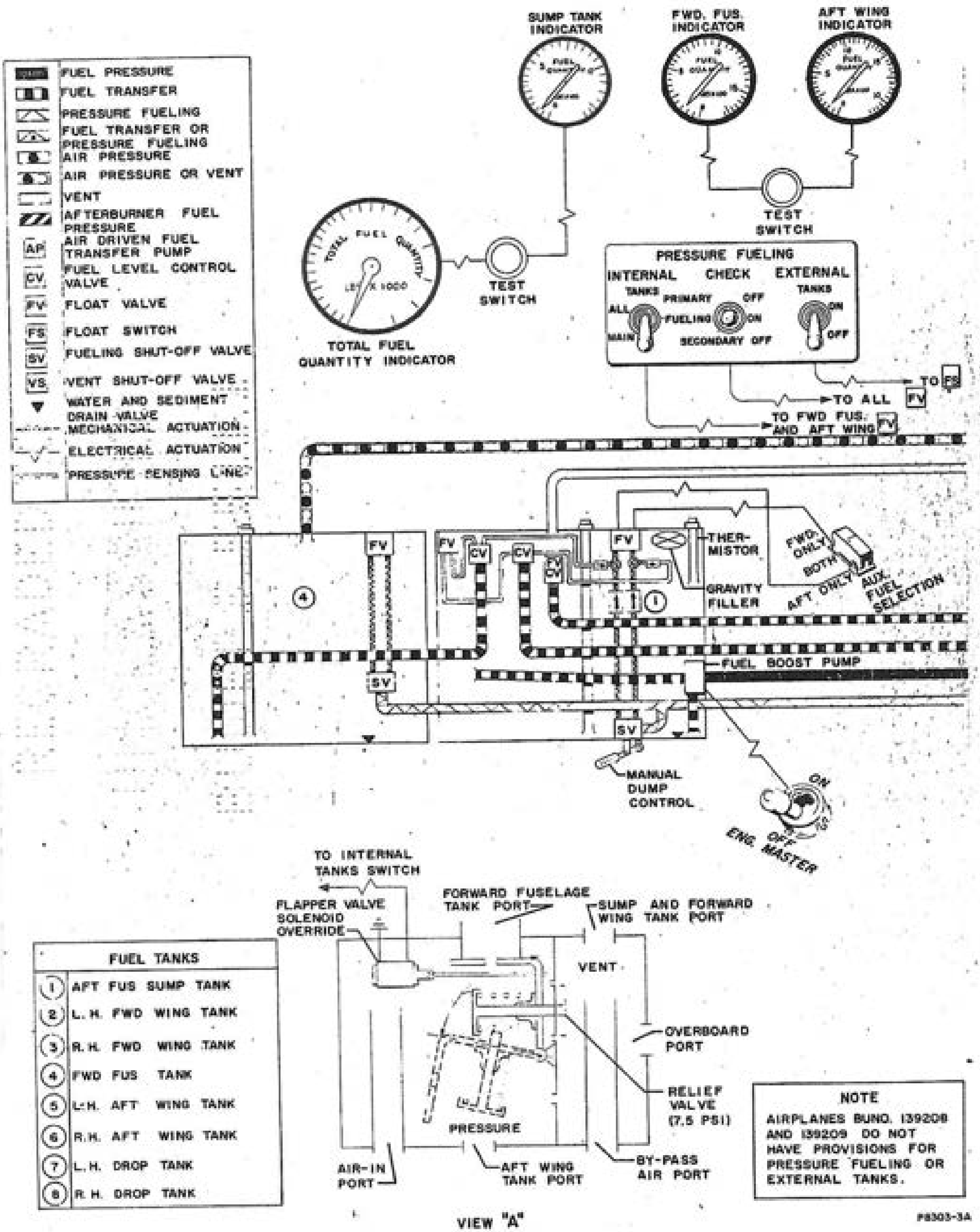


Figure 1-8. Fuel System Schematic (sheet 1)

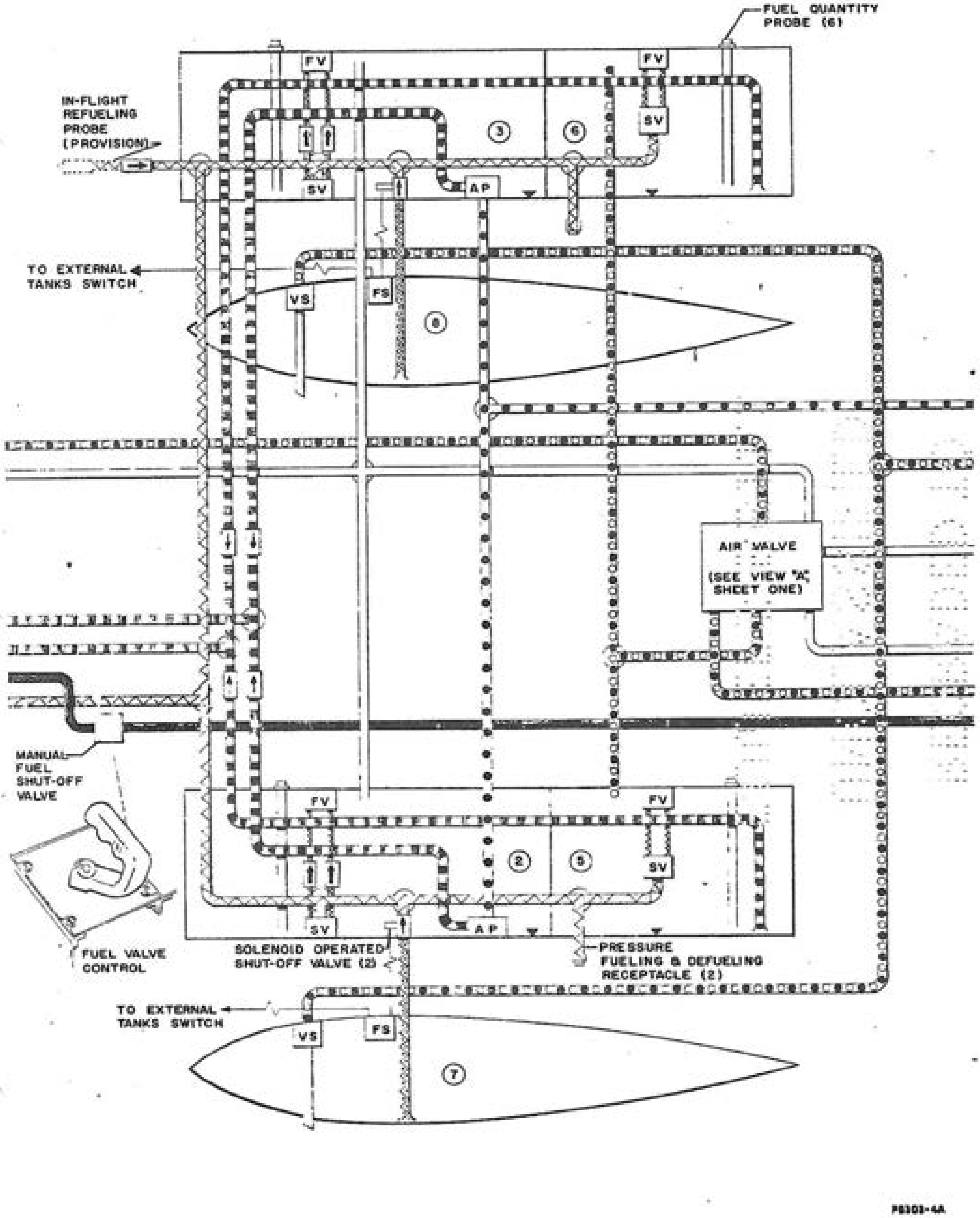


Figure 1-8. Fuel System Schematic (sheet 2)

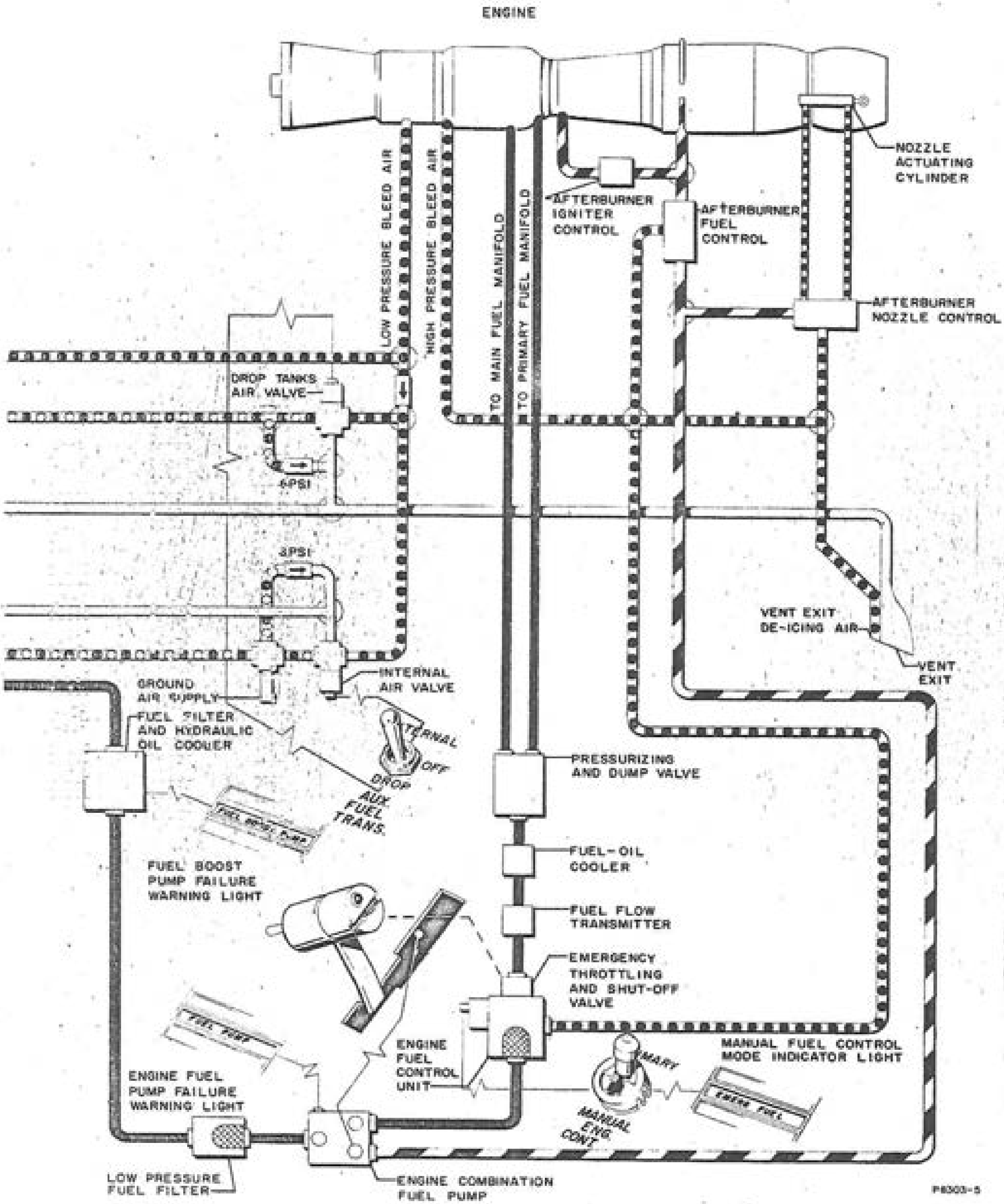


Figure 1-8. Fuel System Schematic (sheet 3)

FUEL BOOST PRESSURE WARNING LIGHT. A FUEL BOOST PUMP pressure warning light (5, figure 1-5) is provided on the right-hand console. When the warning light is "out" it indicates that fuel boost pump pressure is above 5 ± 1 psi. Illumination of the warning light indicates that fuel boost pressure is below this value.

FUEL SHUT-OFF VALVE. The fuel system incorporates a manually operated FUEL VALVE control (10, figure 1-5) located on the right-hand console,[1] or on the canted bulkhead [2] adjacent to the pilot's seat. The FUEL VALVE control handle has two positions, "OPEN" and "CLOSE." The "CLOSE" position of the control stops all fuel flow from the airplane fuel system to the engine fuel control system.

FUEL QUANTITY INDICATING SYSTEM [3]

The fuel quantity indicating system is comprised of six dielectric-type fuel quantity probes, two relay control units, four fuel quantity indicators, a thermistor unit, two fuel quantity test switches, and associated wiring. Each tank contains one fuel quantity probe. The six probes are wired through the relay control units in such a manner as to indicate, on the TOTAL FUEL QUANTITY indicator (23, figure 1-4), the total fuel on board in all six tanks (if serviced), the aft fuselage sump and forward wing tanks, or the sump tank only, depending on the functional status of the fuel transfer system and resultant aft fuselage sump tank fuel levels. The thermistor unit, mounted in the aft fuselage sump tank, is the determining element as to whether the indicating system displays total fuel carried, or actual fuel available for consumption as a result of either malfunction or mismanagement of the fuel transfer system.

The thermistor unit is comprised of four thermistors arranged in two pairs, one above the other. The upper pair of thermistors is situated at the level of the auxiliary fuel transfer level control valves, and serves to signal to the upper level relay control unit the presence or absence of auxiliary transfer fuel in the sump tank. If forward fuselage and aft wing tanks auxiliary fuel is carried, and the pilot does not place the AUX FUEL TRANS switch at "INTERNAL," or if the tanks pressurizing system fails, then the fuel level in the aft fuselage sump tank will fall below the upper pair of thermistors as a result of engine fuel consumption. The thermistors will then signal the upper level relay control-unit to drop out the indication of any remaining auxiliary fuel on the TOTAL FUEL QUANTITY indicator, thereby signaling to the pilot that only aft fuselage sump and forward wing tanks fuel is available for consumption. The lower pair of thermistors is situated at the level of the forward wing tanks fuel transfer level control

(1) Airplanes BuNo. 142349-142350.
(2) Airplanes BuNo. 139208-139209.
(3) Airplanes BuNo. 139208, 142349-142350.

valves, and serves to signal to the lower level relay control unit the presence or absence of forward wing transfer fuel in the sump tank. When forward wing tanks transfer fuel is exhausted, or in the event of failure of the air-driven pumps, the fuel level in the aft fuselage sump tank will fall below the lower pair of thermistors as a result of engine fuel consumption. The thermistors will then signal the lower level relay control unit to drop out the indication of all remaining fuel (except sump tank fuel) on the TOTAL FUEL QUANTITY indicator. At the same time that all fuel readings except sump tank fuel quantity are dropped from the TOTAL FUEL QUANTITY indicator, the lower level relay control unit will cause the FUEL QUANTITY warning light (5, figure 1-5) on the right-hand console to illuminate, indicating to the pilot that 142 gallons (925 pounds), or less, of usable fuel remain.

FUEL QUANTITY INDICATING SYSTEM (1)

The fuel quantity indicating system is comprised of six dielectric-type fuel quantity probes, relay control unit, four fuel quantity indicators, thermistor, two fuel quantity test switches, and associated wiring. Each fuel tank contains one fuel quantity probe. The six probes are wired through the relay control unit in such a manner as to indicate, on the TOTAL FUEL QUANTITY indicator, the quantity of fuel remaining in either the three main fuel tanks or the three main plus the three auxiliary fuel tanks, depending on the position of the AUX FUEL TRANS switch. With the AUX FUEL TRANS switch set at the "INTERNAL" position, the fuel quantity indicating system will reflect, on the TOTAL FUEL QUANTITY indicator (23, figure 1-4), the total fuel quantity in all six tanks. With the AUX FUEL TRANS switch set at "OFF," the system will indicate, on the TOTAL FUEL QUANTITY indicator, only the fuel quantity of the aft fuselage sump and forward wing tanks, regardless of fuel remaining in the forward fuselage and aft wing auxiliary tanks. The aft fuselage sump tank contains the thermistor unit. The unit is located approximately half way down the length of the fuel quantity probe, placing it below the fuel levels normally maintained by the main and auxiliary fuel level control valves during fuel transfer. If the fuel supply in the aft fuselage sump tank falls below the 100-gallon (650 pounds) level, due either to exhaustion of transfer fuel or to malfunction or mismanagement of the fuel transfer system, the thermistor unit causes the reading of any remaining transfer fuel to be dropped out. At the same time that all fuel readings except sump tank fuel quantity are dropped from the TOTAL FUEL QUANTITY indicator, the relay control unit will cause the FUEL QUANTITY warning light (5, figure 1-5) on the right-hand console to illuminate, indicating to the pilot that 100 gallons, or less, of usable fuel remain.

FUEL QUANTITY INDICATORS. Four fuel quantity indicators are provided in the aircraft: a TOTAL FUEL QUANTITY indicator (23, figure 1-4), functionally described in the preceding paragraphs; a sump tank FUEL QUANTITY indicator (15, figure 1-4) on the instrument panel; a FWD. FUS. TANK FUEL QUANTITY indicator (20, figure 1-5) on the right-hand console; and an AFT WING TANK FUEL QUANTITY indicator (20, figure 1-5) on the right-hand console. The sump tank, FWD. FUS. TANK, and AFT

(1) Airplane BuNo. 139209.

WING TANK indicators are installed for flight test purposes, and show fuel quantity in their respective tanks regardless of fuel system functional status or aft fuselage sump tank fuel levels.

FUEL QUANTITY TEST SWITCHES. Two fuel quantity test switches are provided. The FUEL QUA. TEST switch (25, figure 1-4) on the instrument panel is used to test the TOTAL FUEL QUANTITY and sump tank FUEL QUANTITY indicators. The PUSH TO TEST switch (21, figure 1-5) on the right-hand console is used to test the FWD. FUS. TANK and AFT WING TANK FUEL QUANTITY indicator. Depressing either switch causes the pointers of the respective fuel quantity indicators to rotate counter-clockwise toward the zero reading on the dial. When the switch is released, the fuel quantity indicator pointers should return to their previous indication, if functioning properly.

NOTE

- A given volume of fuel will vary in weight, depending on its density, and although the system partially compensates for this variation, the indication of fuel quantity in pounds will vary when the tanks are full if standard conditions do not prevail.

- To obtain the most nearly accurate fuel quantity indication, the aircraft should be in a ground or flight attitude of 5 degrees nose up.

DROP TANKS (1)

The aircraft is equipped with provisions for carrying two 300-gallon drop tanks on the external stores racks. The drop tanks are vented, and have provisions for gravity fueling, pressure fueling, and pressurization to effect fuel transfer at the option of the pilot. The drop tanks may be jettisoned electrically by pulling the EMER STORES REL handle (5, figure 1-3) on the left-hand console.

Fuel transfer from the drop tanks to the forward wing tanks is effected by means of drop tank pressurization. Placing the AUX FUEL TRANS switch at "DROP" opens a solenoid-operated drop tanks transfer three-way air valve which directs low pressure engine compressor bleed air to the drop tanks. Once pressurized, the flow of fuel from the drop tanks to the forward wing tanks is controlled by the forward wing tanks pressure fueling dual solenoid float valves, which

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stop the transfer of fuel when the wing tank is full or allow it to continue when space is available. Placing the AUX FUEL TRANS switch at "OFF" energizes the drop tank transfer three-way air valve solenoid, therby closing the valve and discontinuing transfer of fuel from the drop tanks. If electrical failure occurs, the drop tanks transfer three-way air valve is automatically opened, providing automatic transfer of drop tank fuel whenever wing tank space permits. To prevent drop tanks pressurizing air from being exhausted overboard through the drop tanks vents, each drop tank is equipped with a combination float and diaphragm vent shut-off valve. This valve acts to close the drop tank vent whenever the tank is full or when pressurizing air is introduced.

PRESSURE FUELING AND DE-FUELING SYSTEM (1)

The pressure fueling system is designed to permit fueling at 200 gallons per minute through either of two pressure fueling receptacles, or at 300-400 gallons per minute utilizing both pressure fueling receptacles. The fuel system may be de-fueled through either of the fueling receptacles at a rate of 100 gallons per minute, or through both receptacles at a rate approaching 200 gallons per minute. Each fuel tank contains a dual float shut-off valve to control the fueling operation; a primary float which is the pilot for the shut-off valve, and a secondary float which is a standby for the shut-off valve. A three-position momentary-contack CHECK SWITCH is provided on the pressure fueling control panel in the left-hand wheel well to check the operation of the dual float shut-off valves. Moving the switch to either the "PRIMARY OFF" or "SECONDARY OFF" position causes solenoids to raise the floats to simulate the normal shut-off valve action at the maximum fuel capacity level. This check can be made only after the pressure fueling operation has begun. It is necessary to plug in external electrical power to fuel the airplane using this system.

To de-fuel the aft fuselage sump and forward wing tanks through the pressure fueling receptacles, it is necessary to hold "up" the handle of the fuselage sump tank manual shut-off valve, located at the rear of the armament bay, during removal of the last 346 gallons. To de-fuel the forward fuselage, aft wing, and drop tanks it is necessary to supply air pressure to the GROUND SUPPLY connection, accessible through the oil filler door on the left-hand side of the airplane, and pressurize the auxiliary and drop tanks. This procedure will begin transfer of auxiliary fuel to the aft fuselage sump tank and forward wing tanks which in turn are defueled by manually holding "up" the handle of the aft fuselage sump tank shut-off valve. At the end of the de-fueling operation, the handle of the aft fuselage sump tank shut-off valve should be placed in the "down" position.

PRESSURE FUELING SWITCH PANEL. The PRESSURE FUELING switch panel is located at the rear of the left-hand wheel well. This panel has three switches: the INTERNAL TANKS switch, CHECK switch, and EXTERNAL TANKS switch. The INTERNAL TANK switch is used to select the

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tanks to be fueled. The switch has two positions, "MAIN", and "ALL". The CHECK SWITCH is used to test the operation of the dual float pressure fueling shut-off valves, and has three positions, "PRIMARY OFF", "FUELING ON", and "SECONDARY OFF." The CHECK switch will test the shut-off valves of those tanks selected by the position of the INTERNAL TANK switch only. The EXTERNAL TANKS switch has two positions, "ON" and "OFF". The pressure fueling switch panel receives power from the monitored d-c bus.

ELECTRICAL SYSTEM

Electrical power is normally supplied by an air turbine driven main a-c generator which delivers 115/200 volt, 3 phase, 400 cycle constant frequency a-c power. The aircraft is not equipped with a battery or d-c generator. D-c power is provided by conversion of a portion of available 115/200 volt a-c current into 28 volt d-c current by means of a transformer-rectifier. Additional transformers convert the 115/200 volt, 3 phase, a-c current into 115 volt, 3 phase current and 26 volt a-c current to operate certain electrical equipment requiring these voltages. Electrical power is distributed to the electrically operated aircraft components through a distribution network comprised of the a-c primary, a-c monitored, d-c primary, and d-c monitored busses. An air stream driven emergency generator is provided to furnish electrical power to essential equipment in the event of main a-c generator failure during flight. The aircraft electrical system may be energized when the aircraft is on the ground (engine inoperative) by an external electrical power source when connected to the external power receptacle located in the right wing just forward of the landing gear door. Operation of the electrical system is completely automatic, and no electrical controls are furnished in the cockpit except the emergency generator release handle. For a schematic presentation of a-c and d-c electrical power supply and distribution, see figures 1-9 and 1-10. Refer to Section III for information concerning emergency operation of the electrical system.

MAIN GENERATOR

The main generator is a 115/200 volt, 3 phase ABC rotation, 400 cycle unit rated at 20KVA. The three phases are Y connected with the common grounded, resulting in the voltage across any line (phase) and ground being 115 volts, and across any two lines being 200 volts. The generator is driven by a constant speed air turbine drive unit operated by engine compressor bleed air at the speed necessary to generate a-c current at 400 ± 20 cycles per second (cps). The generator and drive unit (32, figure 1-2) are located in the right-hand air scoop structure and are cooled by ram air bled from the inside of the right-hand air intake duct. Full electrical power requirements are met by the generator and drive unit at engine idling speed. An A.C. GEN. failure warning light (5, figure 1-5), provided on the right-hand console, will illuminate to indicate main a-c generator failure.

EXTERNAL POWER

The aircraft electrical system may be energized on the ground with the engine inoperative through means of an external power receptacle (30, figure 1-2) and switch. Any external electrical power supply capable of delivering 115/200 volt, 3 phase ABC rotation, 400 cycle a-c current, and rated at 20KVA may be utilized. The external power receptacle is located in the right-hand wing forward and inboard of the main landing gear door, and is connected to the a-c monitored bus when the external power switch is in the proper position.

EXTERNAL POWER SWITCH. The external power switch, located adjacent to the external power receptacle, must be placed in the "INTERNAL" position by the ground crew after the engine is operating before the main generator can energize the electrical system. When the switch is placed in the "EXTERNAL" position, the main generator is disconnected and external power is supplied to the aircraft electrical system. The external power switch and receptacle door are designed to prevent the door from being closed when the switch is in the "EXTERNAL" position, in order to avoid inadvertant operation of the aircraft with the electrical system de-energized.

EMERGENCY GENERATOR

The emergency generator (10, figure 1-2) is an air stream driven unit which delivers 115/200 volt, 3 phase, $400 \pm ^{20}_{40}$ cycle current. The unit is rated at 1.7 KVA at an airspeed of 165 knots and above. The generator is normally stowed in a compartment just forward of the left main landing gear door. When released from the stowed position, the generator extends into the airstream and is subsequently motored by a variable pitch propeller which maintains generator speed at the value required to provide constant frequency 400 cycle electrical power. Emergency generator power is connected directly to the a-c primary bus which further distributes current to the d-c primary bus through the transformer-rectifier.

EMERGENCY GENERATOR RELEASE HANDLE. The EMER PWR release handle (3, figure 1-5), on the right-hand console, is utilized to extend the emergency generator and is the only electrical system control provided in the cockpit. Pulling on the handle mechanically actuates a hydraulic valve and generator release latch. The emergency generator is then extended by utility hydraulic power and mechanical means. If utility hydraulic power is not available, the generator will extend by means of gravity and a powerful mechanical downspring. When the EMER PWR release handle is operated, it also disconnects the a-c monitored bus from the electrical system and renders the main generator inoperative by interrupting the exciter field circuit through the voltage regulator. Once the emergency generator is extended, it may be retracted if hydraulic power is available, by returning the EMER PWR release handle to the "normal" position.

A-C POWER DISTRIBUTION

Electrical power from the main generator is regulated by a voltage regulator which maintains a constant voltage output by varying the current in the generator exciter field as required. From the generator, electrical power flows through the "INTERNAL" position of the external power switch to the a-c monitor bus, from the a-c monitor bus through the "normal" position of the emergency generator switch to the a-c primary bus. The primary bus supplies power to a transformer which reduces the voltage to 26 volts to power the 26-volt a-c bus. Each of the busses in the a-c system distributes power through protective fuses to the individual electrically operated components connected to them. Refer to figure 1-9.

A-C MONITOR BUS. The a-c monitor bus is energized at all times when either external power is connected to the aircraft and the external power switch is in the "EXTERNAL" position, or when the main a-c generator is operating and the external power switch is in the "INTERNAL" position. The a-c monitor bus is disconnected when the emergency generator release handle has been actuated, and all electrical units tied to this bus will be inoperative during electrical emergencies.

A-C PRIMARY BUS. The a-c primary bus is energized at all times, either through the "normal" position of the emergency generator release handle, by the a-c monitor bus, or, through the "emergency" position of the emergency generator release handle, by the emergency generator. All equipment tied to this bus will be energized during emergency operation.

D-C POWER DISTRIBUTION

The a-c primary bus supplies 115/200 volt, 3 phase, 400 cycle, a-c power to a transformer-rectifier which converts the a-c power to 28 volts d-c and directly energizes the d-c primary bus. The d-c primary bus energizes the d-c monitor bus through the energized position of the d-c bus control relay when the emergency generator release handle is in the "normal" position. Each of the busses in the d-c distribution system distributes power through protective

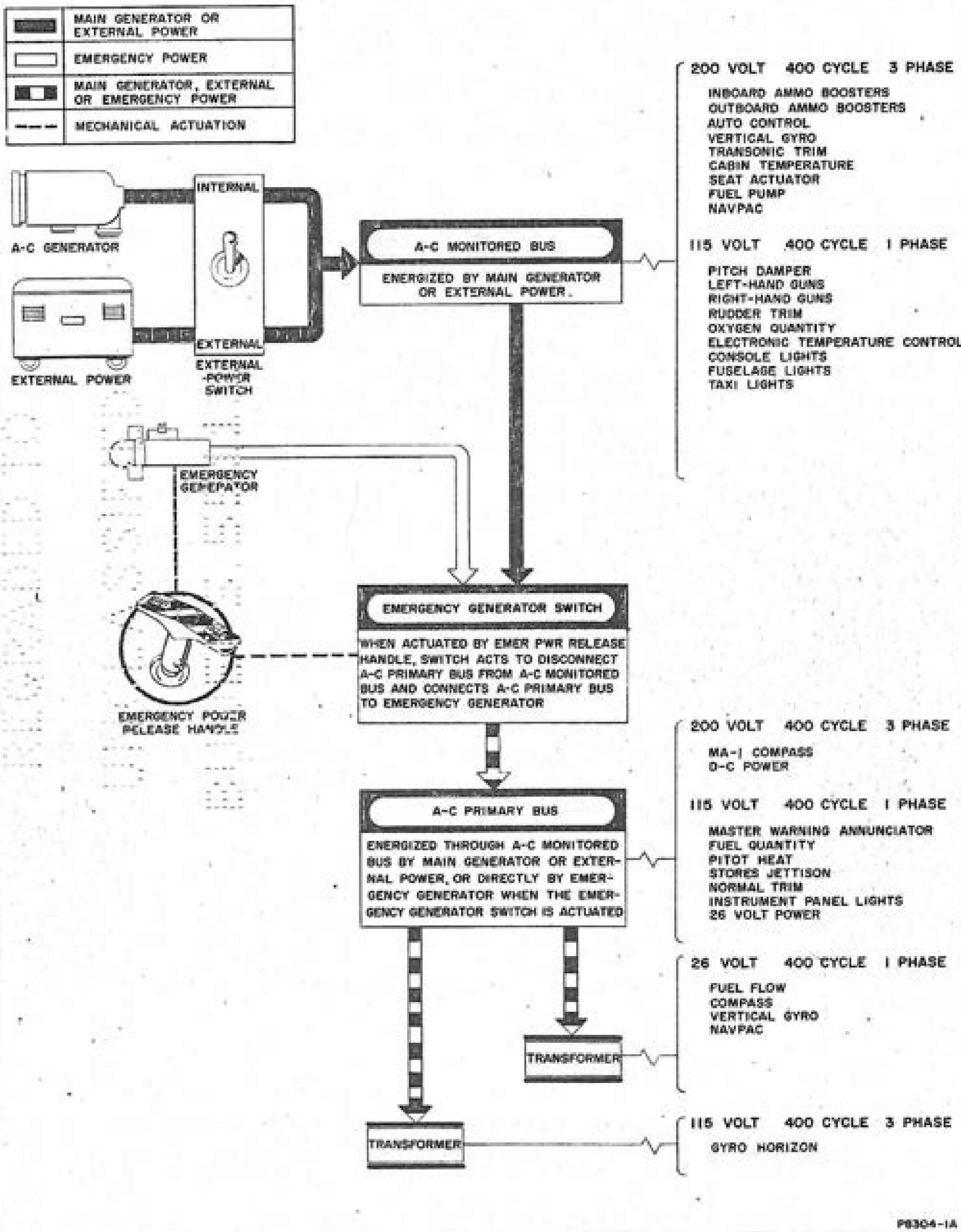

Figure 1-9. A-C Power Supply

fuses to the individual electrically operated components connected to them. Refer to figure 1-10. A D. C. POWER failure warning light (5, figure 1-5), provided on the right-hand console, will illuminate to indicate d-c power failure.

D-C PRIMARY BUS. The d-c primary bus is energized through the transformer-rectifier, a-c primary bus, and a-c monitor bus by external power when the external power switch is in the "EXTERNAL" position; or by main generator power when the external power switch is in the "INTERNAL" position and the engine is operating, or by emergency generator power when the emergency generator release handle has been actuated. All equipment tied to this bus will be operative during emergency operation.

D-C MONITOR BUS. The d-c monitor bus is energized through the energized position of the d-c bus control relay, when the emergency generator release handle is in the "normal" position, by the d-c primary bus. All electrical equipment tied to this bus will be inoperative during emergency generator operation.

D-C BUS CONTROL RELAY. The d-c bus control relay connects the d-c monitor bus to the d-c primary bus when the relay is energized by the "normal" position of the emergency generator release handle. The purpose of this relay, when de-energized, is to disconnect the electrical loads on the d-c monitor bus during emergency generator operation of the electrical system.

FUSE PANELS

All electrical circuits are protected by fuses in lieu of circuit breakers, in order to effect a weight saving and provide better wire protection in view of the fact that no control of the electrical circuits is provided in the cockpit. The fuses are located on two fuse panels, one on each side of the equipment compartment located between the cockpit and the fuselage fuel cells compartment. The fuse panels are inaccessible during flight and should be checked for burned out or improperly inserted fuses prior to flight.

HYDRAULIC SYSTEM

The aircraft is equipped with two independent hydraulic systems; a 3000 psi utility system, and a 3000 psi elevon system. Each hydraulic system is powered by a variable displacement engine driven pump. Two airless, pressurized hydraulic reservoirs of 265 cubic inch capacity each, supply the pumps with hydraulic

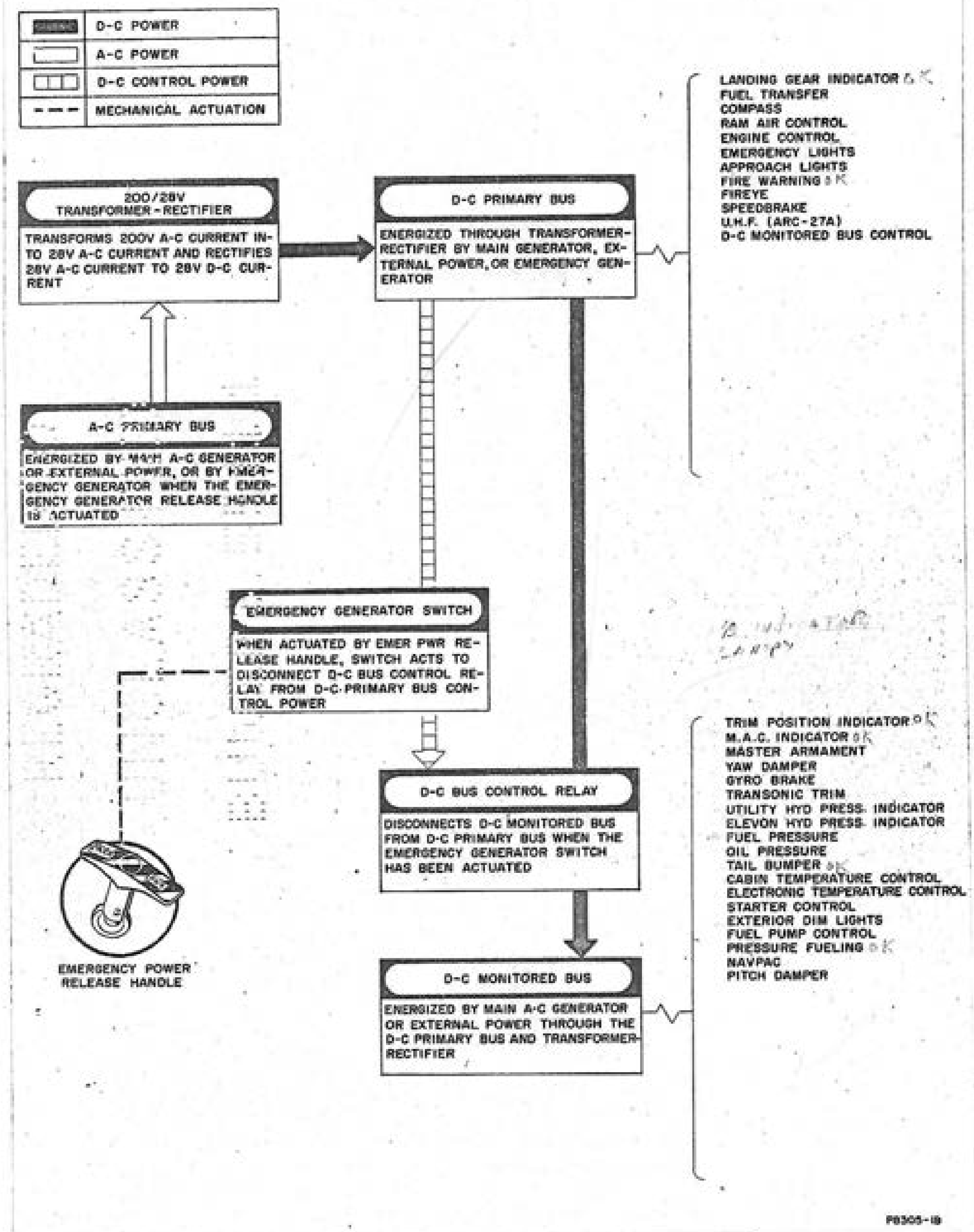

Figure 1-10. D-C Power Supply

fluid. The reservoirs are self pressurized at 60 psi when the engine driven pumps are operating. A hand pump, located in the left-hand air scoop, is incorporated in the hydraulic system for ground handling and filling the reservoirs. An emergency wind-driven drop-out hydraulic pump (5, figure 1-2) is provided to pressurize the elevon hydraulic system in the event of failure of both the engine driven hydraulic pumps. The hydraulic system is shown schematically in figure 1-11. For hydraulic fluid specification refer to Table I at the end of this section. For emergency operation of the hydraulic system refer to Section III.

UTILITY HYDRAULIC SYSTEM

The utility hydraulic system supplies pressure for extending and retracting the landing gear, retracting the tail bumper, retracting the arresting hook, closing the rocket doors, operating the power brakes, extending and retracting the emergency generator, extending and retracting the emergency hydraulic pump, operating the rudder servo, elevator servo, aileron servo, one-half of the elevon tandem actuating cylinders, and one-half of the inboard elevon tandem actuating cylinders. A UTILITY PRESSURE indicator (5, figure 1-5) on the right-hand console illuminates when the utility hydraulic system is not functioning normally. The light illuminates when utility system hydraulic pressure is below 850 ± 50 psi. In addition to the indicator light, a pressure indicator (32, figure 1-4) on the instrument panel reflects utility hydraulic system pressure in pounds per square inch. Utility system pressure may be read opposite the needle labeled "U". Indicator graduations must be multiplied by 1000 to obtain correct values.

ELEVON HYDRAULIC SYSTEM

The elevon hydraulic system supplies pressure for opening and closing the speedbrakes, and for operating one-half of the elevon tandem actuating cylinders and one-half of the inboard elevon tandem actuating cylinders. A ELEVON PRESSURE indicator (5, figure 1-5) on the right-hand console illuminates when the elevon hydraulic system is not functioning normally. The light illuminates when elevon system hydraulic pressure is below 850 ± 50 psi. In addition to the indicator light, a pressure indicator (32, figure 1-4) on the instrument panel reflects elevon hydraulic system pressure in pounds per square inch. Elevon system pressure may be read opposite the needle labeled "E". Indicator graduations must be multiplied by 1000 to obtain correct values.

EMERGENCY DROP-OUT HYDRAULIC PUMP. An emergency "drop-out" hydraulic pump (5, figure 1-2) is incorporated into the elevon hydraulic system. In the event of failure of both engine driven hydraulic pumps, this wind-driven pump will supply

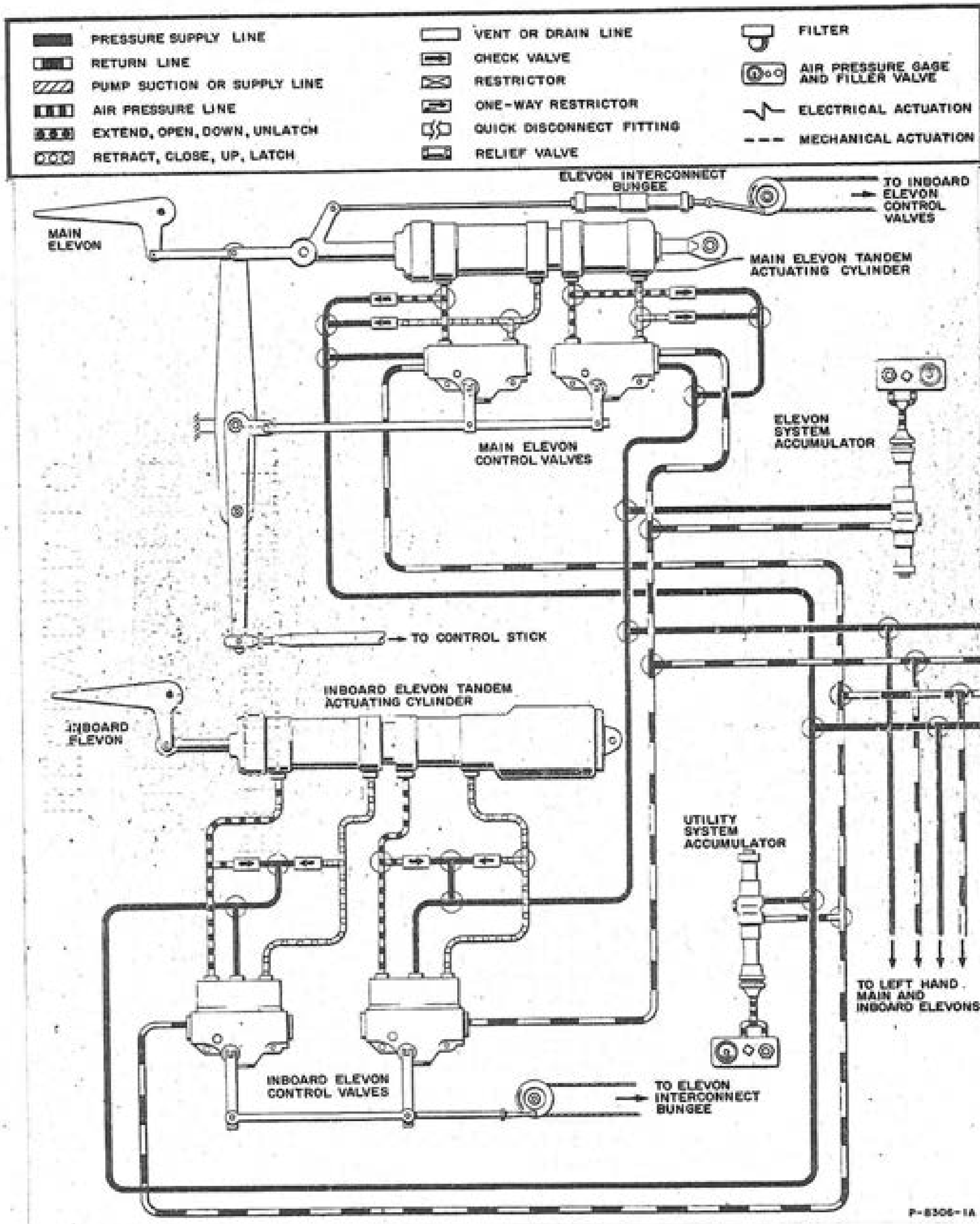

Figure 1-11. Hydraulic System (sheet 1)

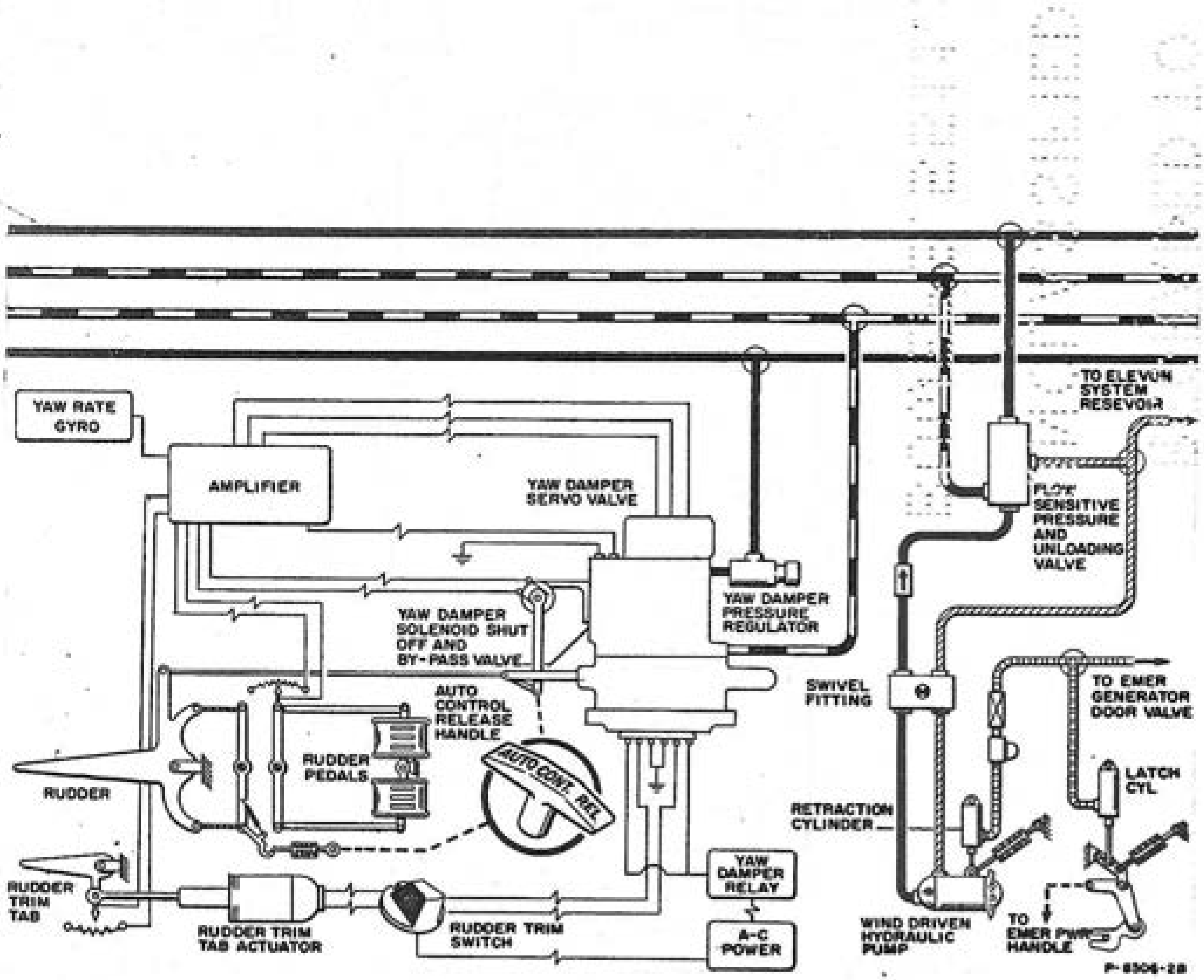

Figure 1-11. Hydraulic System (sheet 2)

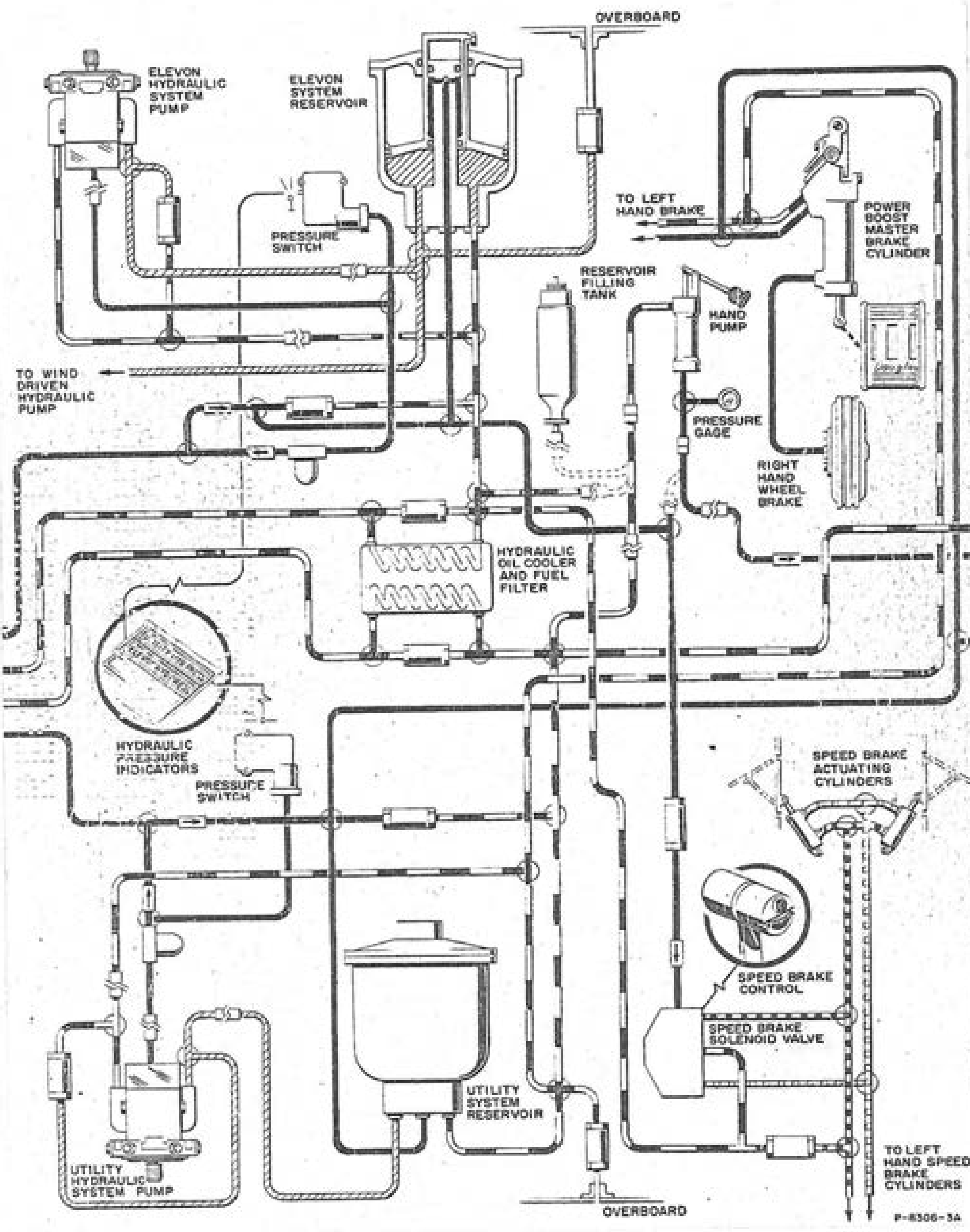

Figure 1-11. Hydraulic System (sheet 3)

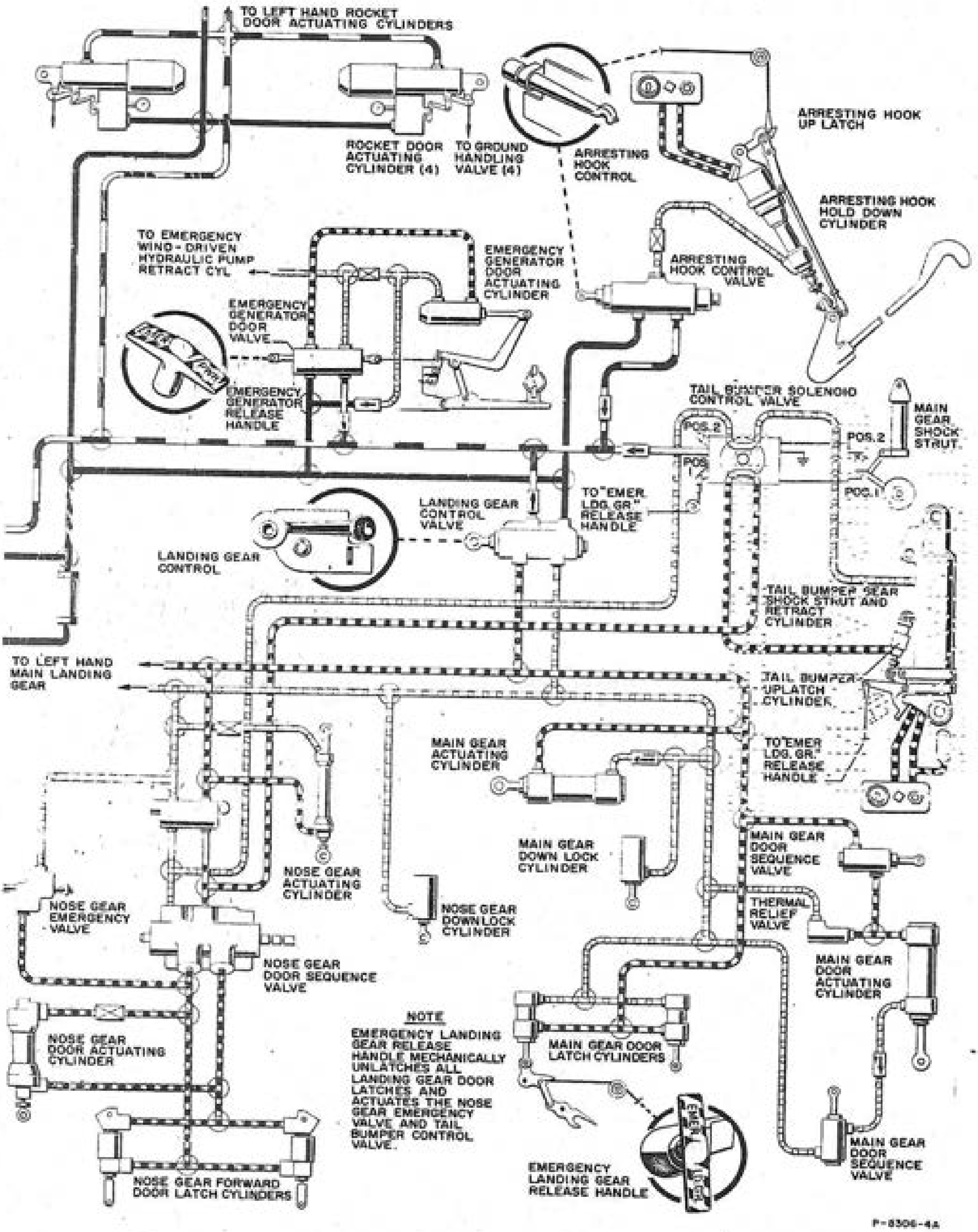

Figure 1-11. Hydraulic System (sheet 4)

hydraulic pressure to the elevons and speedbrakes. The pump is extended into the airstream by pulling the EMER PWR release handle (3, figure 1-5) on the right-hand console. The EMER PWR release handle extends both the "drop-out" hydraulic pump and the emergency generator simultaneously. Both may be retracted by returning the EMER PWR handle to the "normal" position.

FLIGHT CONTROL SYSTEM

The flight control surface arrangement is unconventional. The control surfaces consist of main elevons, inboard elevons, and a power rudder (1) incorporating automatic yaw damping provisions. The main elevons (16, 26, figure 1-2) and the inboard elevons (20, 24, figure 1-2) perform both the longitudinal and lateral control functions normally accomplished through the use of elevators and ailerons. Directional control is accomplished by a power rudder (18, figure 1-2) operated by an electronic-hydraulic servo mechanism. A conventional control stick and rudder pedals operate the control surfaces.

ELEVON CONTROL

The main elevon surfaces are divided into two sections on each side of the wing to permit folding the wings. These two sections of each main elevon operate as one continuous control surface at all times. The port and starboard inboard elevons operate simultaneously and in conjunction with each respective main elevon. The port and starboard main and inboard elevons act together symmetrically for longitudinal control (elevator function), and differentially for lateral control (aileron function). Both the main and inboard elevons are operated by tandem elevon actuating cylinders powered by utility and elevon hydraulic system pressure. Each tandem actuating installation is basically comprised of two integral power cylinders and pistons joined by a common piston rod. Each half of the power cylinders has a separate hydraulic power source controlled by hydraulic valves which are actuated mechanically by movement of the pilot's control stick. Since the elevon control system is a power system, the tandem actuating cylinders are irreversible and their output is completely independent of the manual effort required to displace the control stick. Consequently, there is no feed-back to the pilot of air loads imposed on the elevons. Artificial "feel" of control surface loads is provided through means of spring bungees and a fluid damped elevator bobweight which oppose movement of the control stick in quantities proportionate to the displacement of the control stick and applied airplane load factors. While this method of providing control system "feel" has the disadvantage of having nearly the same resistance to movement of the stick at all altitudes and airspeeds, overstressing of the airframe through overcontrolling is minimized by an automatic mechanical advantage changer system (MACS), discussed later in this section, and a lateral (aileron) sensitivity mechanism. The lateral sensitivity mechanism varies the rate of displacement of the

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elevon surfaces, when functioning as ailerons, for any given constant rate of control stick displacement. During the first seven degrees of elevon displacement this device proportions stick-to-elevon travel at 3:1, and from seven degrees of elevon travel to full control surface displacement, at 3:2. During maneuvers requiring more than twenty percent lateral control deflection, the pilot may feel an increase in control force gradient.

In the event of failure of either the elevon hydraulic system or the utility hydraulic system, complete control is maintained in the transonic and subsonic speed ranges. Control in the supersonic speed range may be limited when one hydraulic system is inoperative. If both hydraulic systems fail, the emergency "drop-out" hydraulic pump may be utilized to pressurize the elevon hydraulic system and maintain elevon control. In addition to the hydraulic power control system, each main elevon incorporates a fluid snubber to prevent elevon "buzz" at high speeds.

MECHANICAL ADVANTAGE CHANGER SYSTEM

The function of the mechanical advantage changer system (M.A.C.S.) is to change the ratio of relative stick-to-elevon travel (independently and in addition to the lateral sensitivity mechanism ratio) from one-to-one at slow speeds or high altitudes, to as much as five-to-one at high speeds and low altitudes (see figure 1-12). The M.A.C.S. automatically maintains a relatively constant stick-travel-to-control-response throughout the entire speed and altitude range, reduces control surface sensitivity at high speeds, and protects the structural integrity of the wing under torsion by limiting elevon deflection at certain altitude and airspeed conditions. The M.A.C.S. is automatically operated during normal conditions by an electric servo motor and cable system which positions a variable moment bell crank in the elevon control linkage. Actuation of the M.A.C.S. servo motor is programmed by an electronic amplifier receiving signals from Mach number and altitude sensing units. Both the servo motor and amplifier are powered by the a-c monitor bus.

MANUAL M.A.C.S. CONTROL. The M.A.C.S. operates automatically when the EMER ELEVON MECH ADVANTAGE crank (1, figure 1-3) is placed in the stowed position. This crank is provided for emergency use in the event of electrical failure, or as an alternate control if the pilot desires to select a particular mechanical advantage ratio manually. To operate the control, the EMER ELEVON MECH ADVANTAGE crank must be swung from its stowed position to the manual operating position. This action automatically disconnects the electrically operated automatic components of the M.A.C.S. To change the mechanical advantage ratio manually, the crank is then rotated towards "INCREASE" or "DECREASE" as required.

WARNING

The EMER ELEVON MECH ADVANTAGE crank must be in the stowed position to place the automatic M.A.C.S. into operation.

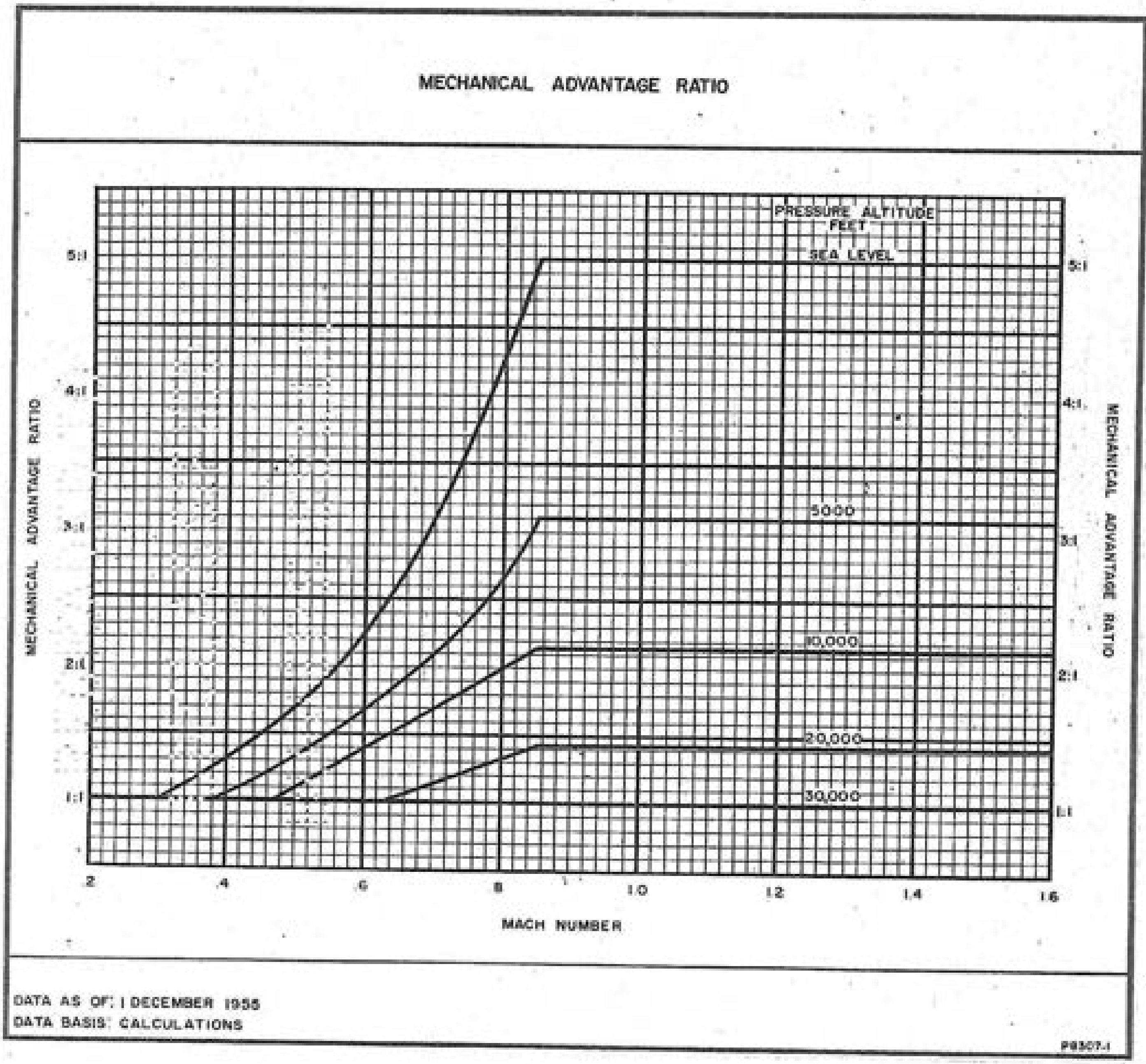

Figure 1-12. Mechanical Advantage Changer Schedule

M.A.C.S. INDICATOR. An ELEVON MECHANICAL ADVANTAGE indicator (20, figure 1-4) located on the instrument panel indicates the mechanical advantage ratio when the d-c monitored electrical bus is energized. The indicator is calibrated from "1" to "5". In addition to the electrical ELEVON MECHANICAL ADVANTAGE indicator on the instrument panel, an emergency manual mechanical advantage indicator (2, figure 1-3) is located on the EMER ELEVON MECH ADVANTAGE changer panel for use during emergency operation of the M.A.C.S. When the EMER ELEVON MECH ADVANTAGE crank is in use, the pilot may obtain the desired mechanical advantage ratio by turning the crank handle until the desired ratio appears in the indicator window. The indicator is calibrated from "T.O & LAND" to "5".

RUDDER CONTROL (1)

An electrically controlled, hydraulically powered rudder, incorporating automatic yaw damping and trim control, is installed on the airplane. The rudder is operated by conventional rudder pedals in the cockpit through an arrangement of force links, electrical elements, an amplifier, rudder force feel springs, a servo valve, and a hydraulic actuating cylinder. Application of pilot effort on the rudder pedals acts to send an electrical signal, proportionate to pedal force, through the amplifier and associated wiring to the rudder servo valve to effect rudder movement by means of the hydraulic actuating cylinder. Force feel springs in the rudder control system serve to keep the rudder pedals centered and to provide resistance to simulate control surface air loads. Rudder trim is effected either by an electrically driven trim tab or by rebalancing the steering coupler in the amplifier to reposition the null or "neutral" point of the servo valve. Refer to DIRECTIONAL TRIM CONTROL in this section. The rudder control system also incorporates a hydraulic snubber, to prevent rudder "buzz" at high speeds, and automatic yaw damping controlled by a yaw rate gyro in the auto pilot system. Refer to YAW DAMPER SYSTEM in this section. In the event of failure of the hydraulic servo valve or cylinder, the rudder may be operated mechanically by pulling out on the AUTO CONT release handle (40, figure 1-4). This action disconnects the hydraulic cylinder from the rudder, permitting mechanical control of the rudder through a system of cables and pulleys.

RUDDER CONTROL (2)

A power rudder, incorporating automatic electronic-hydraulic yaw damping and electrical trim control, is installed on the airplane. The rudder is operated by conventional rudder pedals in the cockpit through cables, a lost-motion mechanism, a rudder pedal position synchro, an amplifier, a servo valve,

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and a hydraulic power cylinder. A spring bungee, located in the dorsal just aft of the canopy, on the lost-motion mechanism, serves to keep the rudder pedals centered and to provide resistance to simulate control surface air loads. Rudder trim is effected both by an electrically driven trim tab and by hydraulic servo "neutral" point repositioning. Refer to DIRECTIONAL TRIM CONTROL in this section. The rudder control system also incorporates a fluid snubber, to prevent rudder "buzz" at high speeds, and automatic yaw damping controlled by a yaw rate gyro in the auto pilot system. Refer to YAW DAMPER SYSTEM in this section. In the event of failure of the hydraulic power servo valve or cylinder, the rudder may be operated mechanically by pulling out on the AUTO CONT release handle (40, figure 1-4) under the instrument panel. This action cages the lost-motion mechanism and disconnects the hydraulic cylinder actuating rod from the rudder, thereby providing direct mechanical linkage from the rudder pedals to the rudder through a system of cables and pulleys.

RUDDER PEDAL POSITION INDICATORS. (2) The rudder pedals are adjustable fore and aft by means of a foot operated lever at the side of each pedal. In order to preclude the possibility of uneven rudder pedal adjustment, two rudder pedal position indicators are provided. These indicators are mounted on the rudder pedal push-pull tubes, just above the cockpit floor and forward of each console. A window displaying the number of each individual rudder pedal position "click-stop" is incorporated, making it possible for the pilot to obtain even adjustment simply by matching the numbers in the indicators.

YAW DAMPER SYSTEM

Automatic yaw damping, through the hydraulic power operated control of the rudder, is provided to decrease directional oscillations during flight. When the yaw damper system is engaged by depressing the push-pull type YAW DAMP switch (18, figure 1-3) on the left-hand console during flight, the rudder is controlled by the servo valve receiving signals from a yaw-rate gyro and yaw damper amplifier in the automatic flight control system. Since the mechanical rudder and servo operated rudder are the same control surface, the rudder pedals will move back and forth against pilot effort at the will of the yaw damper system when engaged. (1) There is no "feed-back" of yaw-damper-system-induced rudder movement to the rudder pedals, due to the installation of the lost-motion mechanism in the rudder control system. (2) Engagement of the yaw damper system is accomplished by depressing the YAW DAMP switch after the YAW DAMPER AUTOCONTROL (1) or AUTOMATIC SYSTEMS (2) switch has been placed at "ON". If the yaw-rate gyro, amplifier, or servo valve fails, the yaw damper system may be isolated by pulling "up" on the YAW DAMP switch. The yaw damper control is automatically turned "off" by a "squat" switch through the airborne control relay during initial compression of the left landing gear strut when effecting a landing. (1) On three aircraft (2) the yaw damper system is turned "off" whenever the landing gear is extended.

Note

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PITCH DAMPER SYSTEM (2)

The aircraft is equipped with a pitch damper system to decrease longitudinal oscillations during flight. When the pitch damper system is engaged, by placing the YAW DAMPER AUTOCONTROL (1) or AUTOMATIC SYSTEMS (2) switch (7A, figure 1-3) at "ON" and depressing the PITCH DAMP button on the left-hand console, inboard elevon control is supplemented by pitch damping signals introduced through a pitch rate gyro, an amplifier, and the inboard elevon tandem actuating cylinders servo valves. Pitch damping signals do not take precedence over the slaving arrangement between the main and inboard elevons; consequently, the pitch damping action operates the inboard elevons (nose up or nose down as necessary) from whatever position they may be in as a result of main elevon command signals. The maximum deflection of the inboard elevons as a result of pitch damping signals is ± 4°. If inboard elevon position is such that the introduction of a pitch damping signal would exceed normal inboard elevon travel limits, the PITCH DAMP button (16A, figure 1-3) will pop-up to disengage the system. In order to reestablish automatic pitch damping, the system control switches must be positioned in sequence as follows; the YAW DAMPER AUTOCONTROL (1) or AUTOMATIC SYSTEMS (2) switch cycled to "EMER OFF", then "ON", YAW DAMP button depressed; and then the PITCH DAMP button depressed. When the YAW DAMPER AUTOCONTROL (1) or AUTOMATIC SYSTEMS (2) switch is "ON," slaving of the inboard elevons to the outboard elevons is accomplished electrically through main elevon position transmitting synchros and the inboard elevon servo valves. Pitch damping signals augment these slaving signals, thereby providing damping of longitudinal oscillations at any slaved inboard elevon position. When the YAW DAMPER AUTOCONTROL (1) or AUTOMATIC SYSTEMS (2) switch is placed at "EMER OFF", pitch damping signals are discontinued and the inboard elevons are slaved to the main elevons through a direct mechanical drive. In the event of failure of any pitch damping components resulting in erratic longitudinal control, the YAW DAMPER AUTOCONTROL (1) or AUTOMATIC SYSTEMS (2) switch may be placed at "EMER OFF" to isolate the pitch damping system and obtain mechanical elevon slaving.

CAUTION

Until further flight testing and final adjustment and scheduling of the pitch damper system, it is recommended that the YAW DAMPER AUTOCONTROL (1) or AUTOMATIC SYSTEMS (2) switch be left in the "EMER OFF" position when operating below 10,000 feet.

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CONTROL STICK

The control stick (see figure 1-4) is conventional in appearance and operation, although the combined lateral and longitudinal control with elevons requires a unique linkage in the control system. Fore and aft movement of the stick will provide symmetrical up and down elevon motion; lateral movement of the stick will provide differential up and down elevon motion; combined longitudinal and lateral control stick movement will "mix" elevon motion as required. Artificial "feel" springs in the control system serve to keep the control stick centered, and provide resistance to movement to simulate control surface loads. The stick grip is provided with a lateral-longitudinal trim switch (37, figure 1-4)and a rocketor guns firing trigger switch (39, figure 1-4).

TRIM CONTROL SYSTEM

The trim control system provides trim about all three axes of the aircraft. Directional trim is provided by an electrically actuated trim tab located at the trailing edge of the rudder, and by displacing the rudder surface hydraulically through the rudder servo valve. Longitudinal trim is accomplished by physically moving the longitudinal stick "feel" spring bungee in the control linkage. Movement of the longitudinal stick feel spring is controlled electrically by the trim switch on the control stick. Lateral trim is accomplished in the same manner as longitudinal trim, utilizing the lateral stick "feel" spring and lateral control function of the elevons. To compensate for large trim changes which occur when the aircraft is accelerating or decelerating through the transonic speed region, the aircraft is equipped with an automatic transonic trim conpensator which acts to relieve the severity of this tim change insofar as pilot action required is concerned.

DIRECTIONAL TRIM CONTROL. Directional trim is accomplished by an electrically actuated trim tab mounted on the trailing edge of the rudder and (1) or (2) rudder surface displacement through means of electrically repositioning the neutral or "null" point of the power rudder servo valve. Control of the trim tab and servo valve is through means of the RUDDER TRIM switch (4, figure 1-3) on the left-hand console. (1) On one aircraft (2) the RUDDER TRIM TAB switch (4, figure 1-3, sheet 3) actuates the rudder trim tab only. On this aircraft an additional RUDDER TRIM switch (6A, figure 1-3, sheet 3) is provided, which trims the rudder surface through the power rudder hydraulic servo valve. Positions of the switches are "L" and "R", for nose left and nose right rudder trim respectively. No emergency rudder trim is provided. A rudder trim tab position indicator (20, figure 1-4) on the instrument panel, shows rudder trim tab position at all times when the d-c monitor bus is energized. The indicator is identified as NOSE and is graduated if five degree

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units from "0" to "15", both left and right of center, toward the "L" and "R" positions.

LATERAL TRIM CONTROL. To trim the aircraft laterally, the control stick trim switch (37, figure 1-4) is moved to the "LWD" or "RWD" position. This action causes an electric motor powered by 115 volt and 26 volt a-c power to move the lateral stick "feel" spring bungee, relocating the center or "neutral" position of the control stick, thereby actuating the control system linkage and valves to re-position the elevons in their aileron function. The lateral trim actuator is powered by the a-c primary bus. No lateral trim position indicator is provided in the cockpit, consequently the only means of determining lateral trim position is by visual inspection of control stick or elevon position.

LONGITUDINAL TRIM CONTROL. To trim the aircraft longitudinally, the stick trim switch (37, figure 1-4) is moved to the "NOSE UP" or "NOSE DOWN" position. This action causes an electric motor powered by 115 volt and 26 volt a-c power to move the longitudinal stick "feel" spring bungee, relocating the center or "neutral" position of the control stick, thereby actuating the control system linkage and valves to reposition the elevons in their elevator function. The longitudinal trim actuator is powered by the a-c primary bus. A longitudinal trim position indicator (20, figure 1-4) is located on the instrument panel. The indicator is identified as NOSE, and is graduated from "0" through "10" in two degree increments from a horizontal position (corresponding to three o'clock), counterclockwise through a range of 135 degrees; and from "0" through "2" from a horizontal position (corresponding to three o'clock), clockwise through a range of approximately 25 degrees. The longitudianl trim position indicator is powered by the d-c monitor bus.

TRANSONIC TRIM CONTROL COMPENSATOR. The aircraft is equipped with a transonic trim compensator (TTC) to alleviate the trim change whcih occurs as the aircraft is accelerated or decelerated through the transonic speed region. Refer to HIGH SPEED FLIGHT, Section VI. The TTC is comprised of a Mach number sensing component, function generator, electric drive motor, crank assembly, synchro, amplifier, and associated wiring. As the aircraft accelerates into the trim change region, the Mach number transducer supplies signals to the function generator in the amplifier. which in turn programs the actuation of an electric drive motor and crank assembly to physically reposition the elevator load feel force trim bungee. This action moves the control stick aft, actuating the elevon control system linkages, elevon control valves, and dual tandem actuating cylinders to move the elevons up to compensate for the nose down or "tuckunder" trim change. When decelerating through the trim change region, the TTC components act to reverse this stick and elevon movement to counteract the nose up trim change or "pitch up" which occurs at this time.

NOTE

> Pilots should anticipate this fore or aft stick movement so as not to be startled when it occurs. This stick movement will at the same time tend to maintain a constant stick force. Pilots should also be prepared to make any necessary fine adjustments to stick position to completely cancel out the trim change.

The TTC system is placed into operation when the TRANSONIC TRIM switch (3A, figure 1-3) and YAW DAMPER AUTOCONTROL (1) or AUTOMATIC SYSTEMS (2) switch (7A, figure 1-3) on the left-hand console are placed at "ON." If it is desired to cancel out TTC signals, or in the event of malfunction, normal control stick position may be obtained by placing the TRANSONIC TRIM switch at "RETURN" TO NEUTRAL" for a minimum of 6 seconds, and subsequently to "OFF."

NOTE

> On one aircraft (1) the signals to the drive motor are furnished by relays rather than by a magnetic amplifier. This results in actuation of the system, and consequently stick movement, in steps or a series of "bumps."

CONTROL LOCK

The elevons are automatically centered and locked when the elevon lock handles at each wing fold joint are pulled down, or when the wing pin pulling handle is pulled down to unlock the wings, and at all times when the wings are folded. The rudder has no gust lock provision, but is prevented from swinging uncontrolled against its stops by the damping action of the rudder snubber, making it unnecessary to incorporate a control lock.

<u>WING SLATS</u>

Slats are installed on the leading edge of the wings to improve lateral and directional stability and control during take-off,

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approach and landing. The slats are fully automatic, opening and closing in response to aerodynamic forces. No fixed airspeeds can be published on the points at which the slats will begin to open or close, as the variables of airspeed, gross weight, and applied load factor will all affect the operation of the slats to a varying degree. In general, the slats may be expected to start opening at an airspeed below 325 knots, and to be fully opened at an airspeed below 205 knots. At Mach numbers in excess of 0.55 the slats will not open.

SPEED BRAKES

Four speed brakes are installed on the aircraft, one on the upper and lower surface of each wing. The speed brakes are actuated by elevon hydraulic system pressure and controlled by a two position SPEEDBRAKE switch (8, figure 1-3) mounted on the throttle grip. To open the speedbrakes, the switch is moved aft to the "OPEN" position; to close them, the switch is moved forward to the "CLOSED" position. As the SPEEDBRAKE switch has only two positions, the speedbrakes cannot be stopped at any intermediate point between fully opened or fully closed. The speedbrakes may be opened at any speed. A "blow-back" feature is incorporated which allows the speedbrakes to begin to close when the air load against them causes the hydraulic pressure in the actuating cylinders to exceed the pressure at which the "blow-back" relief valve opens, thus preventing damage to the speedbrake system. The speedbrakes will begin to blow-back at an indicated airspeed of approximately 380 knots. The speedbrakes require approximately two seconds to open or close.

WING FOLDING

The wings are folded and spread manually and locked mechanically. The wing folding controls are located in each wing at the wing fold joint. Access to the wing folding controls is through a hinged door on the bottom surface of each wing, midway between the leading and trailing edges at the wing fold joint. Opening the access door exposes the elevon lock and wing pin pulling handles. To fold the wing, the wing pin pulling landle (long handle) is pulled down and aft to the vertical position. This action automatically operates the elevon lock handle (short handle) to lock the elevons, and pulls the wing pins. The wing is then pushed manually to the folded position and locked in place by raising the wing pin pulling handle back to the horizontal position. When the wing is folded, a mechanical interlock prevents the elevon lock handle from being moved to the "unlocked" position, thereby making it impossible to unlock the elevons inadvertently. The wing fold control access door cannot be closed when the wing is folded. No interconnect mechanism is provided between the port and starboard wings, making it necessary to fold and spread each wing individually.

CAUTION

On three aircraft (1) the flight control system is rigged in such a manner that the elevons are positioned at two degrees airplane nose up when the control stick is centered. In order to prevent structural damage to the wing and/or control system when the wings are folded, do not apply any hydraulic pressure to the ship's system through any means. It is recommended that the wings be left spread at all times except as absolutely neccessary for servicing.

WING PIN LOCK INDICATORS

Two red "warning flag" indicators are located inboard of the wing fold joints, one on the upper surface of each wing. The indicators extend above the wing surface whenever the wing fold control access doors are open, regardless of whether the wings are spread and locked. The indicators are flush with the wing surface only when the access door is closed, indicating that the wing is locked since the door cannot be closed unless the wing pin pulling handle is in the "locked" position and the elevon lock handle is in the "unlocked" position.

LANDING GEAR SYSTEM

The aircraft is equipped with a tricycle landing gear and tail bumper gear. The main gear retracts forward, the wheels rotating ninety degrees and the shock strut telescoping to allow the gear to fit flush into the wings. When retracted, the main gear is held in the retracted position by hydraulic pressure in the retraction cylinders and by the wheel well doors. The nose gear retracts forward into the fuselage and is retained in the retracted position by hydraulic pressure and the nose wheel well door. The tail bumper retracts upward into the tail cone assembly. A tail bumper sequencing switch automatically retracts the tail bumper when the weight of the aircraft is on the main landing gear struts. The landing gear and wheel well doors are actuated by utility hydraulic system pressure. The tail bumper gear is extended pneumatically and retracted hydraulically. All landing gear wheel well doors remain extended when the gear is down except the forward nose gear door which is retracted to prevent severing the barrier engagement strap in the event of a barrier crash. Emergency extension of the landing gear is accomplished by mechanically releasing the wheel well doors up latches, by-passing the landing gear sequence valves

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retraction lines, and the effects of gravitational and aerodynamic forces.

LANDING GEAR CONTROL

The landing gear is extended or retracted by moving the LANDING GEAR control (13, figure 1-3) to "UP" or "DOWN". When the LANDING GEAR control is placed in the "UP" position, a red light in the wheel shaped control handle will illuminate until all four units of the landing gear are up and locked. When the LANDING GEAR control is placed in the "DOWN" position, the red light will illuminate until the tail bumper is extended and the main and nose gear are down and locked. A solenoid-operated safety latch (12, figure 1-3) is provided to prevent inadvertent retraction of the landing gear when the aircraft is on the ground. The solenoid is actuated, through the airborne control relay, by the "squat" switch on the left main landing gear strut when the weight of the aircraft compresses the strut. In normal operation, the landing gear retraction release solenoid is energized when the aircraft becomes airborne, thus unlatching the control handle. If the retraction release solenoid should malfunction, or if it becomes necessary during an emergency to retract the landing gear while on the ground, the safety latch may be released manually to unlatch the LANDING GEAR control.

LANDING GEAR POSITION INDICATOR. A wheel position indicator (17, figure 1-4) is provided on the instrument panel. When all units of the landing gear are locked up, the word "UP" appears in the indicator. When all landing gear units are locked down, a miniature wheel appears in the indicator. If any unit of the landing gear is not locked in either position, a cross-hatched warning signal will be visible. In the event that the cross-hatched warning signal is visible, it may be readily determined which landing gear unit is not locked up or down through use of the landing gear indicator selector switch adjacent to the indicator. The selector switch is labeled "N" (nose), "L" (left), "ALL", "R" (right), and "T" (tail). The indicator is powered by the d-c primary bus.

Note

> If the d-c electrical power supply fails the cross-hatched warning signal will be visible in the indicator regardless of landing gear or selector switch position.

EMERGENCY LANDING GEAR EXTENSION CONTROL

In the event of utility hydraulic system failure the landing gear may be extended by placing the normal LANDING GEAR control handle at "DOWN," and pulling the EMER LDG GR handle (14, figure 1-3). Operating this handle mechanically releases the landing gear wheel well door latches and opens emergency by-pass valves in the landing gear hydraulic system, allowing the gear to drop to the extended position. The tail bumper gear is released mechanically and extended pneumatically when the emergency landing gear release handle is operated.

WARNING

The normal LANDING GEAR control must be in the "DOWN" position in addition to pulling out the EMER LDG GR handle to effect emergency landing gear extension.

ARRESTING GEAR

An externally mounted arresting hook (22, figure 1-2) is installed in a fairing on the lower aft engine door structure. Retraction and extension of the hook is accomplished by a pneumatic-hydraulic hold-down cylinder in the aft engine access door. The hold-down unit is essentially a reservoir which is divided into two chambers by a relief valve and orifice arrangement. The upper chamber is partially filled with hydraulic fluid and then charged with compressed air to 550 psi with the hook retracted. The lower chamber contains the actuating piston which is attached to the arresting hook. Utility hydraulic system pressure is applied to the lower side to effect retraction of the arresting hook, which is then held in the retracted position by a mechanical latch. Compressed air pressure in the upper chamber, and the relief of hydraulic retracting pressure in the lower chamber cause extension of the hook when the mechanical latch is released. With the arresting hook extended, the relief valve and orifice provide snubbing action to keep the hook on the deck during arrested landings.

ARRESTING HOOK CONTROL

The ARREST HOOK control (1, figure 1-5) on the right-hand cockpit rail controls the operation of the arresting hook. When the control is moved to "DOWN", the arresting hook is mechanically unlatched and hydraulic retraction pressure is relieved, allowing pneumatic pressure in the hold-down unit to extend the hook. A red light in the hook-shaped handle of the control will illuminate when the control is moved to "DOWN", and will go out when the hook reaches the fully extended position. The "UP" position of the ARREST HOOK control mechanically positions the arresting hook control valve so that utility hydraulic pressure enters the lower chamber of the hold down unit, overriding the air pressure and causing the hook to retract and latch. If the cable system to the latching mechanism should fail, the latch will be automatically released; however, the

ARREST HOOK control must be moved to the "DOWN" position to relieve retraction hydraulic pressure before the hook will extend.

BRAKE SYSTEM

A hydraulic power boost brake system is provided in the aircraft. Each main wheel brake has an independent hydraulic boost brake cylinder operated by toe pressure on the rudder pedals. The power brakes are operated by utility hydraulic system pressure. In the event of utility hydraulic system failure, sufficient pressure for braking can be obtained by exerting approximately twice the normal toe pressure on the rudder pedals.

INSTRUMENTS

The aircraft is equipped with instruments and indicators necessary for operation of the power plant, and monitoring aircraft systems, emergency and auxiliary equipment. Pitot pressure is supplied by an electrically heated pitot tube (2, figure 1-2) located on the top of the fuselage just forward of the windshield. Static pressure is supplied by two static pressure ports (37, figure 1-2), one on each side of the fuselage above and forward of the nose gear wheel well door.

FLIGHT INSTRUMENTS

The flight instruments consist of an airspeed indicator, Mach meter, and pressure altimeter, operated by pitot and static pressure; a vertical gyro attitude indicator operated by electrical power; and a turn-and-bank indicator operated by engine compressor bleed air.

AIRSPEED INDICATOR. An airspeed indicator (6, figure 1-4) on the instrument panel is provided to indicate airplane velocity in knots. The outer perimeter of the instrument is graduated at one knot intervals and labeled at ever 10 knots. One revolution of the needle is equivalent to 100 knots. A window at the bottom of the instrument permits the pilot to view a rotary dial which is graduated in 50 knot increments and labeled every 100 knots. For example; if the aircraft were flying at 650 knots IAS, the indicator would display a "6" in the window and the needle would be positioned at "50" on the outer dial.

MACH NUMBER INDICATOR. A Mach number indicator (13, figure 1-4) is provided on the instrument panel. This instrument is graduated in increments of 0.02 Mach and is labeled at every 0.10 Mach from ".5 "through "1.8".

VERTICAL GYRO ATTITUDE INDICATOR. This aircraft is equipped with a Model K-4 remote indicating, vertical gyro control with a pictorial sphere presentation gyro INDICATOR (7, figure 1-4) located on the instrument panel. This instrument provides the pilot with a constant visual attitude indication of the pitch and roll of the aircraft. Aircraft attitude reference signals are supplied to the INDICATOR by electrical connection with the K-4 controller, which is mounted remotely in the aircraft. Pitch and roll attitudes are indicated by motions of a universally mounted sphere, which is displayed as the background for a miniature reference airplane attached to the instrument case, giving improved readibility and interpretation even during extreme maneuvers. The horizon is represented on the sphere as a white line dividing the top and bottom halves of the sphere. The upper half, symbolizing sky, is indicated by a light grey area above the horizon line, labeled "CLIMB"; the lower half, symbolizing earth, is indicated by a dull black area below the horizon line, labeled "DIVE". The sphere is graduated every five degrees of climb and dive. Thus, the indicator is very determinative of climb or dive angle in maneuvers as well as in instrument climb-outs, G. C. A. and high performance penetrations in instrument weather. The sphere is free to move a full 360 degrees in roll, without obstruction, at roll rates up to 275 degrees per second. Multiple rolls may be made without accumulative error. Bank angles can be read on a semi-circular bank scale located on the upper half of the instrument case. The miniature reference airplane is always in the logical physical relationship to a simulated earth, horizon or sky areas of the background sphere, and it is sufficiently realistic to enable the pilot to fly the airplane simply by flying the instrument. The indicator is capable of accurate performance in all pitch attitudes including loops and coordinated immelmans. However, the instrument will "tumble" if the aircraft is rolled in either a vertical dive or vertical climb.

The remote gyro control unit is driven by a 115-volt, 400-cycle, 3 phase, a-c electric motor which receives current from the a-c primary bus through a transformer. Control power is furnished by the 28-volt d-c primary bus. The gyro horizon indicator receives all its power from the K-4 gyro control. It is illuminated indirectly by the instrument lighting system.

The instrument features automatic caging, and is ready for use shortly after electrical power is induced into the system. Within two minutes the gyro is erected, the amplifier channels are warmed up and the "OFF" flag retracted from the indicator face. Failure at any time of the a-c or d-c power supply or the erection relay will cause the "OFF" flag to reappear.

An adjusting knob sets the potentiometer controlling the sphere setting to line up the horizon line on the ball with the reference airplane to accommodate different sized pilots and different trim and loading conditions of the aircraft.

WARNING

> The indicator is not reliable for flight indications if the power warning flag is visible.
>
> Rolling the aircraft in the vertical position will tumble the instrument.

A REMOTE ATTITUDE CAGING SWITCH located on the instrument panel (34, figure 1-4) is provided to cage the VERTICAL GYRO ATTITUDE INDICATOR.

ENGINE INSTRUMENTS

The engine instruments consist of a tachometer, tail pipe temperature indicator, oil and fuel pressure indicators, oil temperature indicator, pressure ratio indicator, fuel quantity indicators, fuel boost pump and engine fuel pump failure warning indicator. For description of the engine instruments, refer to applicable paragraphs under ENGINE, and FUEL SYSTEM, Section I.

NAVIGATION INSTRUMENTS

The navigation instruments consist of an ID-250A/ARN course indicator and magnetic STANDBY COMPASS (figure 1-4). For a description of the course indicator and STANDBY COMPASS, refer to NAVIGATION EQUIPMENT, Section IV.

OTHER INSTRUMENTS

Other instruments available to the pilot consist of a CABIN DIFF. PRESS. indicator (24, figure 1-4), mechanical advantage changer and rudder and elevator trim position indicator, SPEED BRAKES position indicator (10, figure 1-4), landing gear position indicator, dual hydraulic pressure indicator (32, figure 1-4), LIQUID OXYGEN quantity indicator, accelerometers (8, 14, figure 1-4), force gage (22, figure 1-4) and FIRE warning indicator. For descriptions of the mechanical advantage and rudder and elevator trim position indicator, landing gear position, hydraulic pressure, and FIRE warning indicators, see applicable paragraphs of Section I. For description of the LIQUID OXYGEN quantity indicator, refer to LIQUID OXYGEN SYSTEM, Section IV.

EMERGENCY EQUIPMENT

FIRE DETECTION SYSTEM

The fire detection system indicates the occurrence of fire in the engine accessory, and turbine and afterburner compartments. If a fire occurs in either of these two compartments, a push-to-test type FIRE warning light (5, figure 1-4) on the instrument panel will illuminate. The fire detection system may be checked for continuity by pressing the FIRE warning indicator. When the indicator is depressed, the FIRE warning light should illuminate to indicate a properly functioning circuit and light bulb.

FIRE EXTINGUISHING SYSTEM (1)

The fire extinguishing system consists of two pressure vessels containing the fire extinguishing agent bromotrifluoromethane (CF_3B_r), valves, piping and discharge nozzles, and associated wiring and controls. In the event of a fire in the engine cavity, the bromotrifluoromethane fire extinguishing agent may be discharged into the engine accessory, and turbine and afterburner compartments by operating the guarded FIRE EXT switch (5A, figure 1-4). The fire extinguishing system may be checked by depressing the FWD and AFT FIRE BOTTLE TEST switches (5B, figure 1-4) in turn. When either switch is depressed, the light adjacent to the switches will illuminate, indicating a properly functioning circuit.

BAROMETRIC PARACHUTE OPENER

Parachutes used with the integrated flight suit will be equipped with a barometric-pressure-actuated parachute opener. The opener is designed primarily to deploy the parachute automatically at a predetermined altitude, should the pilot be incapacitated to the extent where he is not able to open the parachute manually. In addition, the opener provides a four-seconds timed delay before opening the parachute after reaching the preset altitude. This delay is incorporated primarily for those instances when ejection is made below, at or slightly above the altitude for which the opener is set. The delaying period allows the pilot to decelerate prior to the opening of the parachute, thus reducing or eliminating any opening shock damage to the pilot or parachute. The delay also prevents the parachute from fouling on the seat when ejection is made at altitudes below that for which the opener is set, where deployment would otherwise occur immediately upon separation from the seat. The barometric parachute opener interferes in no way with the normal opening method, and the parachute ripcord grip may be pulled at any time to open the parachute. An arming pin is inserted through the opener mechanism to prevent inadvertent deployment of the parachute during normal operation whenever the aircraft descends through the altitude for which the opener is set. The arming pin is anchored by a static lanyard to a retaining slot on the HARNESS RELEASE handle (17, figure 1-6). When the pilot separates from the seat after ejection, the arming pin is automatically pulled and the opener is then armed.

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WARNING

If the pilot's automatic harness releasing mechanism should fail to operate and the HARNESS RELEASE handle must be pulled to free the pilot from the seat, the opener will not be armed and the parachute MUST be opened manually.

CANOPY

The cockpit canopy is the hinged "clamshell" type which moves slightly aft and then swings up to open. When closed, the canopy is held in place by latches on the cockpit rails which engage latch pins on the canopy rails. An air bungee cylinder, pressurized to 2870 psi, provides the force necessary to counter-balance the canopy during normal use and to jettison the canopy during emergencies. The canopy is closed by grasping either side and pulling down against the bungee counter-balancing pressure, and operating either the internal or external CANOPY controls to slide the canopy forward and lock it. The canopy may be jettisoned, if necessary, by five different methods: pulling the ejection seat face curtain control; pulling the EMER CANOPY JETT handle; pulling the (flight test) auxiliary front handle PILOT EJECTION control; pulling the external EMERGENCY CANOPY JETTISON handle; and by operating the normal interior CANOPY control while in flight, which will result in the force of the relative wind shearing off the canopy.

WARNING

- When the canopy is jettisoned, a cable linking the canopy with the seat catapult firing mechanism removes the safety pin and arms the catapult for subsequent seat ejection.

- The canopy is not designed to be opened in flight. Release of the canopy latches while the aircraft is airborne or with headwind conditions in excess of 107 knots will result in loss of the canopy. Damage to the canopy will occur during ground operations with headwinds in excess of 88 knots.

CANOPY CONTROLS

NORMAL INTERIOR CONTROL. The CANOPY control handle (3, figure 1-3) on the left side of the cockpit is mechanically linked to the canopy mechanism. To close and lock the canopy, the canopy must first be pulled down manually against the air bungee counter-balancing pressure, and then the CANOPY control may be moved forward and down, which slides the canopy forward against the windshield and latches it. To open the canopy, the CANOPY control is pulled up and aft, disengaging the latches and allowing the air bungee to raise the canopy.

NORMAL EXTERIOR CONTROL. The exterior CANOPY control handle (3, figure 1-2) is located on the left hand side of the fuselage below the edge of the cockpit. To open the canopy from the outside, the CANOPY control lever is moved down and forward, clockwise, until the canopy releases and swings up. To close the canopy from the outside, it is necessary to hold the canopy closed manually, then the protruding CANOPY lever is moved aft and up, counter-clockwise, until it is flush with the fuselage.

EMERGENCY INTERIOR JETTISON CONTROL. The canopy may be jettisoned by pulling the EMER CANOPY JETT handle (41, figure 1-4). Pulling out this handle at any time will unlatch the canopy and blow it clear of the aircraft through the jettisoning feature of the air bungee cylinder.

Note

> The air bungee cylinder is capable of blowing the canopy clear of the aircraft during a ground emergency when the airplane is at a standstill, or under water.

EMERGENCY EXTERIOR JETTISON CONTROL. An EMER CANOPY JETTISON control (4, figure 1-2) is located on the left-hand side of the fuselage, below and aft of the normal exterior CANOPY control. A second, similar control, labeled EMERGENCY CANOPY JETTISON is located on the right-hand side of the fuselage. In the event of a crash or fire, the canopy may be blown clear of the aircraft from the outside by pulling either handle. The handle on the left-hand side of the fuselage is situated behind and above a spring loaded access door. To gain access to the control it is necessary first to depress a flush latch forward of the access door; second, to push the door inboard with the knuckles of the hand with the palm up; and third, to pull down on the handle which will spring into the hand when the door is fully open.

The handle on the right-hand side of the fuselage is situated below a spring loaded door and is flush with the exterior surface of the fuselage. To gain access to the control it is necessary only to push inboard on the top edge of the door; whereupon the handle will spring out and may be grasped for actuation.

WARNING

> When canopy is jettisoned by any method, the ejection seat safety pin is pulled by a lanyard anchored to the canopy structure and the seat catapult is armed.

EJECTION SEAT CONTROL. The canopy is also jettisoned when the face curtain handle (4, figure 1-6) is pulled down. The first portion of face curtain travel will jettison the canopy at the interlock position through an interconnecting linkage from the seat ejection system.

WARNING

> Canopy jettisoning by means of partial face curtain extension is not recommended except during the ejection sequence, since no positive stops are provided which would prevent inadvertent seat ejection if tension is maintained on the face curtain after the canopy has jettisoned.

AUXILIARY FRONT HANDLE EJECTION CONTROL. The canopy is also jettisoned when the auxiliary front handle ejection system is employed for ejection. The results are the same as ejection of the seat and pilot with equipment as by means of the face curtain. The canopy is jettisoned by the air bungee and the seat is ejected when the handle marked PILOT EJECTION (13, figure 1-6), located between the pilot's legs, is pulled. (see AUXILIARY FRONT HANDLE EJECTION SYSTEM).

EJECTION SEAT

The pilot's seat (figure 1-6) is the ejection type accommodating a back-type parachute and a PK-3 pararaft kit, and incorporates provisions for attachment to a flight suit with integral harness. A non-adjustable head rest is built into the seat structure and houses the face curtain. Although the ejection seat is not equipped with foot rests, the seat pan (16, figure 1-6) projects forward of the seat basin to provide thigh support during ejection, the front surface of the seat basin serves as a buffer for the calves of the legs, and the sides of the basin extend above the pilot's thighs to protect the legs from side forces during ejection. Ejection of the seat is accomplished by a catapult which utilizes an explosive cartridge, Mark III, Mod "O", to separate the seat from the aircraft. The seat is equipped with a three-quarter second delay harness release mechanism utilizing a blank .38 caliber cartridge, which automatically frees the seat belts and shoulder harness from the seat structure three-quarters of a second after ejection. The ejection seat also provides a means of automatically arming a barometrically-controlled parachute opener by means of a lanyard attached to the seat at the HARNESS RELEASE handle.

SEAT ATTACHMENTS

The pilot is held in the seat by attachments to the integrated flight suit. This flight suit incorporates within its structure the combined attributes of a seat belt, shoulder straps, and parachute harness, thus leaving the pilot with few of the normal encumbrances. The shoulder straps are sewn to the parachute risers and attach to the inertia reel connection just below the head rest. The loose ends of the parachute risers have quick-action fittings (7, 20, figure 1-6) which engage other fittings that extend from the front shoulders of the flight suit. Short seat belts which are sewn to the pararaft cover on each side, are attached to the seat structure at the aft corners of the seat basin. The loose ends of the seat belts have quick-action fittings (12, 19, figure 1-6) which engage mating fittings protruding from the hip region of the flight suit. The seat belts are adjustable in length. When these connections are made, the pilot is held securely in the seat. The delayed action of the harness release actuator provides protection for the pilot in that the feature retains the pilot in the seat during the early period of ejection, when high velocity windblast could cause premature opening of the parachute with resultant parachute damage and severe opening shock inflicted on the pilot. During the delaying period, the pilot and seat will decelerate to a speed where the stresses placed upon the pilot and parachute are reduced from the critical stage, which might otherwise be reached if the separation occurred immediately after ejection. The harness release actuator is essentially a cylinder containing a piston, a slow burning .38 caliber cartridge and a firing mechanism. The firing mechanism is spring loaded and is held in a safe position by a pin, to which a lanyard is attached. A cable runs from the harness release mechanism through a pulley on the actuator piston and is anchored to the seat structure. When the seat is ejected the firing mechanism safety pin is pulled by the lanyard anchored to the aircraft structure, allowing the spring loaded firing mechanism to detonate the cartridge. Three-quarters of a second later the fired cartridge exerts enough pressure to actuate the piston. As the piston extends it pulls the cable and releases the seat belts and shoulder harness, freeing the pilot from the seat.

PARACHUTES

The parachute is furnished with an automatic barometrically-controlled opening device (see BAROMETRIC PARACHUTE OPENER) which will prevent automatic deployment of the parachute until after a preset time delay below the preset altitude (approximately 14,000 feet). All survival equipment automatic devices are furnished as a supplemental means of actuation under adverse conditions such as injury or unconciousness, with manual means of actuation (ripcord) retained in all cases. The lanyard from the arming mechanism running to the HARNESS RELEASE handle (17, figure 1-6) can be removed by squeezing the trigger of the HARNESS RELEASE handle without pulling the handle. This action pulls back a slotted pin, allowing the lanyard to be removed from the HARNESS RELEASE handle. The quick-action fittings attaching the shoulder harness and parachute risers to the flight suit permit the pilot to easily and rapidly release the parachute risers after a safe landing, to prevent possible injury from being dragged when on land or possible drowning when in water.

SEAT CONTROLS

SEAT SWITCH. The seat is electrically adjustable in height by movement of the SEAT switch (2, figure 1-5) to either of its two positions, "UP" or "DOWN", and is stopped at any desired position by releasing the switch to the center, or "off" position.

SHOULDER HARNESS INERTIA REEL CONTROL. The shoulder harness take-up mechanism is controlled manually by the SHOULDER HARNESS control (11, figure 1-6) on the left-hand side of the seat basin. The control is moved forward to "LOCK" and aft to "UNLOCKED". During seat ejection or sudden deceleration in excess of 2.5 g along the thrust line of the aircraft, the inertia reel automatically locks the shoulder harness. This safety feature helps to prevent injuries if the shoulder harness is not locked prior to an arrested landing or crash. When once locked and a force has been applied to the shoulder harness, the inertia reel will not release when the harness is merely slacked off, regardless of the position of the manual control. This is due to the "stay-lock" feature of the inertia reel. Only when the control is moved into the "UNLOCKED" position with the harness slacked off will the mechanism release.

HARNESS RELEASE. A "D-ring" handle, labeled HARNESS RELEASE (17, figure 1-6), is mounted on the right side of the pilot's seat. The HARNESS RELEASE "D-ring" is equipped with a guard and squeeze trigger. In the event the pilot needs to be free of his harness, the "D-ring" handle may be released for actuation by grasping it with the right thumb on the inside of the guard and the other four fingers squeezing the trigger. When the handle is pulled up a distance of about five inches the shoulder harness and seat belt attachments are released from the seat, allowing the pilot to leave the cockpit with the parachute and pararaft kit still at-ached to the integrated flight suit harness.

WARNING

> Do not pull the HARNESS RELEASE handle "D-ring" while airborne, or until the airplane comes to a complete stop. Pulling the "D-ring" releases the shoulder harness and lap belt end fittings and re-engagement cannot be made in flight.

In event, the integrated harness is not automatically released following ejection, seat separation may be accomplished by pulling the HARNESS RELEASE. In this case it will be necessary to pull the ripcord manually, as the automatic barometric parachute opener will not be armed.

EJECTION CONTROL. The ejection seat face curtain handle (4, figure 1-6) serves not only as a control for jettisoning the canopy and ejecting the seat, but also as a protective cover for the pilot's face during ejection. The face curtain, which is housed in the head rest structure with the handle protruding, is mechanically connected to the air bungee cylinder and the seat catapult firing mechanism. When the face curtain handle is pulled downward, the first portion of travel jettisons the canopy and the last portion causes the seat to be ejected.

NOTE

> When pulling the face curtain during the ejection sequence, pilots should anticipate a slight pause or hesitation in face curtain travel at the interlock position. This is due to a safety feature which prevents full face curtain travel to the ejection position until the canopy shears away from the aircraft and pulls the ejection seat arming safety pin.

AUXILIARY FRONT HANDLE EJECTION CONTROL. If the full pressure suit headgear prevents pulling the face curtain for ejection, or if "G pressures" prevent the pilot from using the face curtain, the auxiliary front handle ejection system should be used. This consists of a "D-ring" handle and cable, together with a hidden cranking lever for mechanical advantage, linked to the pilot seat catapult system. The "D-ring" ejection handle, labeled PILOT EJECTION (13, figure 1-6), is located between the pilot's legs in the notch in the seat pan and pararaft equipment. The handle is retained vertically by a clip type bracket, allowing easy access with either hand. Pulling up on the "D-ring" gives the same results as pulling the face curtain release. The ejection sequence should occur between three and six inches of cable pull.

WARNING

- Do not pull the PILOT EJECTION D-ring without first placing the head firmly on the headrest. In any other position the sudden acceleration may cause serious injury.
- The "D-ring" handle is not safety wired and all hands should be careful not to kick or lift this handle inadvertently. The hazard is obvious.

PARACHUTE SERVICING. To remove the parachute and pararaft for periodic inspection and servicing it is necessary to pull up on the HARNESS RELEASE handle.

AUXILIARY EQUIPMENT

Refer to Section IV for a discussion of the following auxiliary systems and equipment:

Air Conditioning and Pressurization System
Defrosting System
Anti-icing System
Communication Equipment
Liquid Oxygen System
Navigation Equipment
Miscellaneous Equipment

TABLE I

GRADE AND SPECIFICATION

FUEL	(Recommended)	MIL-F-5624A(JP-4)
	(Alternate)	MIL-F-5624(JP-3)
	(Emergency)AVGAS	MIL-F-5572-1(JET MIX 3:1)
OIL		MIL-L-7808B(PWA521A)
HYDRAULIC FLUID		MIL-O-5606(RED)

SECTION II

NORMAL PROCEDURES

BEFORE ENTERING THE AIRCRAFT

FLIGHT RESTRICTIONS

Refer to Section V, Operating Limitations, for airplane and engine operating limitations and flight restrictions.

CRUISE CONTROL

Fuel, power settings and airspeeds required to accomplish an assigned mission are normally provided in Appendix 1, Operating Data. This data, however, is not available at this printing and future publication is not contemplated.

WEIGHT AND BALANCE

Check gross weight and center of gravity for take-off and anticipated loading for landing. Refer to Section V for weight limitations to be observed. Insure that the distribution of load falls within the center of gravity fore and aft limits as specified in Section V.

EXTERIOR INSPECTION

Consult the ground crew and applicable records to determine the engineering status of the aircraft and that it has been fully serviced with fuel, oil, liquid oxygen, compressed air, hydraulic fluid, and bromotrifluoromethane (1) as required.

(1) Airplanes BuNo. 139208-139209.

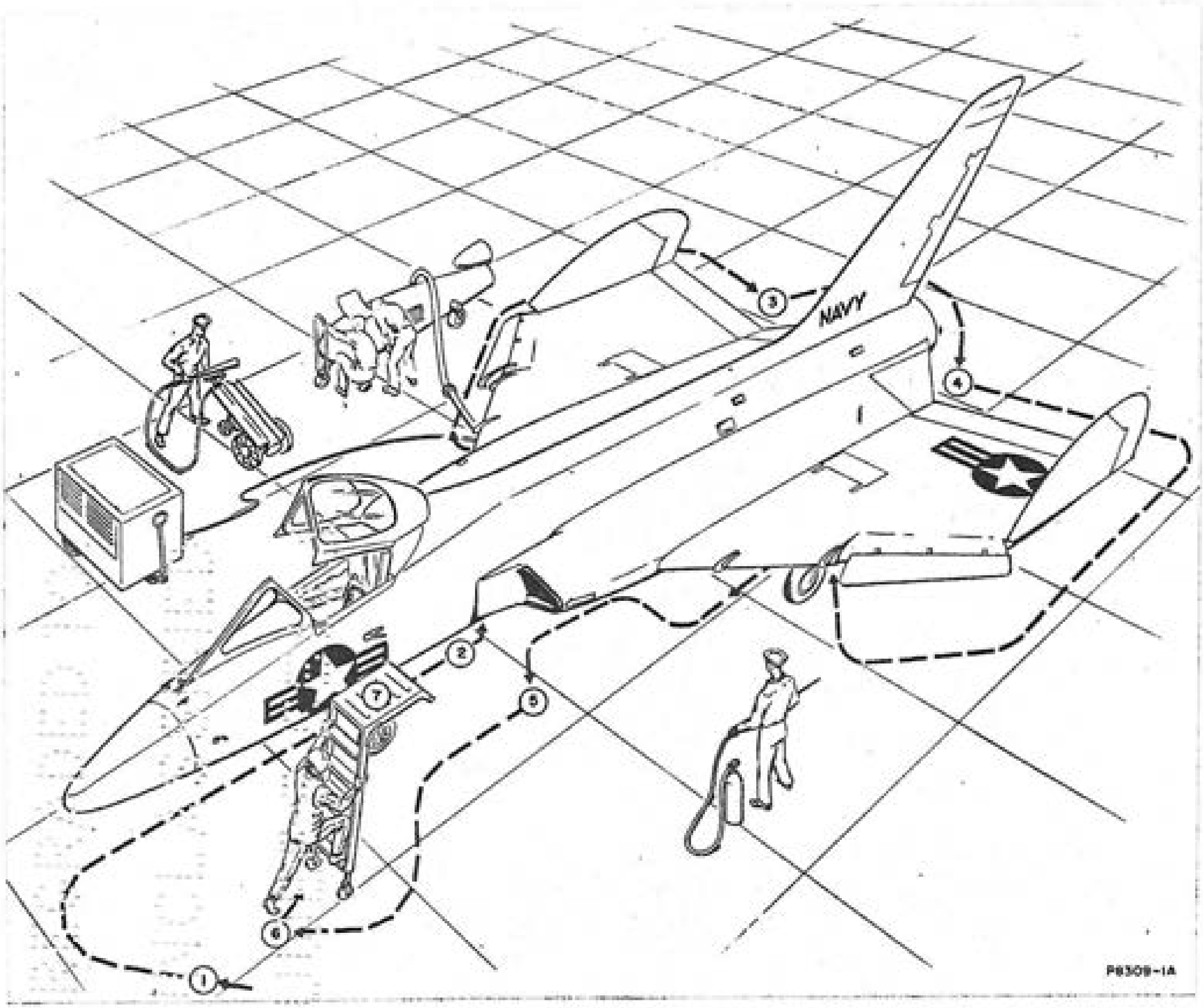

Figure 2-1. Exterior Inspection

Conduct an inspection of the exterior of the aircraft, proceeding as shown in figure 2-1, to check the following:

1-2. FUSELAGE

a. Static pressure vent (left side).................... clear

b. Radome........................ condition, security

c. Static pressure vent (right side).................. clear

d. Air conditioning intake and exhaust ducts (right side)..... clear

e. Nose wheel tire....................... properly inflated, no slippage

f. Nose wheel strut...................... proper extension, no leakage

g. Nose wheel downlock pin............... removed

h. Nose wheel well doors................. condition, security

i. Nose wheel compartment vent exit...... clear

j. Equipment compartment access hatch.... closed, security

k. EMERGENCY CANOPY JETTISON handle access door.................... condition, security

2-3. R-H INTAKE DUCT, WING AND LANDING GEAR

a. Fuselage fuel tanks compartment vent air intake (leading edge of boundary layer separator)............ clear

b. Engine air intake plug................ removed

c. Intake duct........................... free of foreign objects

d. Generator cooling air intake (inside intake duct).................. clear

e. Rocket doors cylinder pressure gages (doors closed).................. 750 ± 25 psi

f. Generator air turbine drive air exit.. clear

g. Generator cooling air exit............ clear

h. External power switch................. "INTERNAL"

i. External electric power............... plugged in

j. Forward wing tank water drain cock.... drained, secure

k. Aft wing tank water drain cock........ drained, secure

l. Transfer pump air turbine exhaust..... clear

m. Forward engine access door............ closed, secure

n.	External engine starter unit hose and electrical plug.......	connected
o.	Main wheel well door...........	condition, security
p.	Main wheel tire.................	properly inflated, no slippage
q.	Main wheel strut...............	proper extension, no leakage
r.	Main wheel downlock pin........	removed
s.	Utility hydraulic system elevon accumulator gage (wheel well)...................	1000 ± 100 psi
t.	Elevon hydraulic system accumulator gage (wheel well)..	1000 ± 100 psi
u.	Speed brake (lower)............	condition, security
v.	Access plates and covers.......	secure
w.	Bottom surface of wing free of stains indicating fuel or hydraulic fluid leaks	
x.	General condition (wrinkles, cracks, loose rivets)	
y.	Slat...........................	condition, security, operation
z.	Navigation lights..............	condition
aa.	Wing tip.......................	condition
ab.	Elevons........................	condition, security, bonding
ac.	Fuel system vent exit (forward of inboard elevon).............	clear
*ad.	Speed brake (upper)............	condition, security
*ae.	Forward and aft wing tank filler caps (1)...............	secure

* Items located on upper surface of wing.

(1) Airplanes BuNo. 139208-139209.

*af. Access plates and covers.............. secure

3-4. TAIL SECTION

a. Arresting hook pressure gage......... 550 ± 50 psi

b. Tail bumper pressure gage............ 235 ± 25 psi

c. Tail bumper and arresting hook....... up and secure

d. Spin chute (if installed)............ secure, rigged

e. Tail pipe plug....................... removed

f. Tail pipe............................ cracks, fuel deposits

g. Fin and rudder....................... general condition, bonding

h. Access plates and covers............. secure

4-5. L-H INTAKE DUCT, WING, LANDING GEAR, AND ARMAMENT BAY

*a. Access plates and covers............. secure

*b. Forward and aft wing tank filler caps (1)..................... secure

*c. Speed brake (upper).................. condition, security

d. Elevons.............................. condition, security bonding

e. Wing tip............................. condition

f. Navigation lights.................... condition

g. Slat.... .... condition, security operation

h. General condition (wrinkles, cracks, loose rivets)

i. Bottom surface of wing free of stains indicating fuel or hydraulic fluid leaks

* Items located on upper surface of wing.

(1) Airplanes BuNo. 139208-139209.

j. Access plates and covers........ secure

k. Speed brake (lower)............. condition, security

l. Utility hydraulic system elevon accumulator gage (wheel well)... 1000 ± 100 psi

m. Elevon hydraulic system accumulator gage (wheel well)... 1000 ± 100 psi

n. Main wheel well door............ condition, security

o. Main wheel tire................. properly inflated, no slippage

p. Main wheel strut................ proper extension, no leakage

q. Main wheel downlock pin......... removed

r. Pressure fueling switch panel:
   INTERNAL TANKS switch........ "MAIN"
   CHECK switch................. "FUELING ON"
   EXTERNAL TANKS switch........ "OFF"

s. Transfer pump air turbine exhaust......................... clear

t. Aft wing tank water drain cock.. drained, secure

u. Forward wing tank water drain cock........................... drained, secure

v. Emergency generator door........ closed, secure

w. Fuselage fuel tanks compartment vent air exit................... clear, no fuel drainage

x. Fire extinguisher discharge indicators (1).................. white

y. Drop-out hydraulic pump door.... closed, security

z. Engine air intake plug.......... removed

aa. Intake duct..................... free of foreign objects

(1) Airplanes BuNo. 139208-139209.

ab. Fuselage fuel tanks compartment vent air intake (leading edge of boundary layer separator)........ clear

ac. Forward auxiliary and aft fuselage sump tanks water drain cocks (in armament bay - rocket doors open)............ drained, secure

ad. Aft fuselage sump tank shut-off valve (armament bay)................ "down"

ae. Forward auxiliary and aft fuselage sump tanks fuel filler caps (top of fuselage) (1).... secure

5-6. FUSELAGE

a. EMER CANOPY JETTISON handle access door......................... condition, security

b. Air conditioning intake and exhaust ducts (left side)............ clear

c. Pitot pressure tube cover........... removed

d. Access ladder....................... secure

e. 100-pound CO_2 fire extinguisher..... 10 feet from engine

f. Area clear 100 feet to the rear, 10 feet forward, 5 feet to the sides of the engine.

7. COCKPIT AREA

a. Exterior CANOPY control.............. "OPEN"

b. Canopy.............................. general condition

c. Ejection seat arming safety pin...... in place

d. Canopy air bungee pressure gage...... 2870 psi

e. Canopy inflatable air pressure seal.. general condition

(1) Airplanes BuNo. 139208-139209.

ENTRANCE

Entrance into the aircraft is effected on the left side, using the ladder provided by the ground crew to gain access to the cockpit. After entering the aircraft, have the ground crew remove the ladder prior to starting the engine, due to its proximity to the engine air intake ducts.

ON ENTERING THE AIRCRAFT

INTERIOR CHECK

Upon entering the aircraft, check the general appearance of the cockpit and accomplish the following:

a. Make suit harness connections.

b. Make oxygen and radio, and anti-blackout hose connections.

c. Throttle........................ "off"

d. ENGINE MASTER switch............ "OFF"

e. ENG CONT switch................. "PRIMARY"

f. AUX FUEL TRANS switch........... "OFF"

g. PITOT & ENG ANTI-ICING switch... "OFF"

h. Direct the ground crew to energize the external electric power supply and to place the external switch at "EXTERNAL".

i. ATT. GYRO switch................ "UNCAGED"

j. EMER ELEVON MECH ADVANTAGE crank......................... stowed position

k. EMER ELEVON MECH ADVANTAGE indicator...................... "T. O. & LAND"

l. ANTI-BLACKOUT valve control..... "HI" or "LO" as desired

m. OXYGEN control.................. "ON"

n. SPIN & DRAG CHUTE CONTROLS panel
   READY-OFF switch.................... "OFF"
   BLOW-OFF-BLOW switch .... "OFF"
   OPEN handle......................... secure
   RELEASE handle...................... secure

o. TRANSONIC TRIM switch................. "OFF"

p. EMER STORES release handle............ secure

q. RUDDER TRIM switch.................... centered

r. SPEEDBRAKE switch..................... "CLOSED"

s. EXT LIGHTS switch..................... "OFF"

t. PITCH DAMP button (1)................. pulled out

u. YAW DAMP button....................... pulled out

v. YAW DAMPER AUTOCONTROL (3) or
   AUTOMATIC SYSTEMS (1) switch.......... "EMER OFF"

w. START switch.......................... pulled out

x. FACE MASK TEMP control panel.......... "OFF"

y. LANDING GEAR control.................. "DOWN"

z. EMER LDG GR handle.................... secure

aa. The landing gear position indicator should have a miniature wheel visible.

ab. FWD and AFT FIRE BOTTLE TEST
    switches (2)......................... press to test

ac. FIRE warning light................... push to test

ad. FIRE EXT (2)......................... guarded off

ae. Altimeter............................ field elevation

af. MASTER WARNING TEST switch........... press to test

(1) Airplanes BuNo. 139208, 142349-142350.
(2) Airplanes BuNo. 139208-139209.
(3) Airplane BuNo. 139209.

ag. Accelerometer PUSH TO SET knob.. depress

ah. SLATS OPEN lights............... illuminated

ai. GEAR DOOR GAPPING lights........ illuminated

aj. LIQUID OXYGEN quantity gage..... "F" (1) or "4" (2)

ak. FUEL QUANT TEST switch.......... press to test

al. AUTO CONT release handle........ secure

am. EMER CANOPY JETT handle......... secure

an. ARREST HOOK control............. "UP"

ao. Console FUEL QUANTITY PUSH TO TEST switch...................... push to test

ap. AUX TANK SELECTOR switch........ "BOTH"

aq. SEAT switch..................... "UP" or "DOWN" as desired

ar. AN/ARC-27A, OFF-T/R-T/R+G - ADF switch.......................... "T/R" or "T/R+G"

as. Air conditioning temperature control switch.................. "70"

at. AIR COND switch................. "NORMAL"

au. WSHLD DEFOG switch (1).......... "OFF"

av. CABIN AIR TEMP switch (3)....... "ON AUTO"

aw. EMER PWR release handle......... secure

ax. PULSE GENERATOR CONTROL switch.. "OFF"

ay. SELECTOR switch................. "OFF"

(1) Airplane BuNo. 139209.
(2) Airplanes BuNo. 139208, 142349-142350.
(3) Airplanes BuNo. 139208-139209.

az. COMPASS, FREE N. LAT-SLAVED-
FREE S. LAT switch.................... "SLAVED"

ba. SET TO LAT switch.................... set to latitude

bb. SPARE FUSES container................. supply

bc. SPARE LAMPS container................. supply

bd. FUEL VALVE control.................... "OPEN"

BEFORE STARTING THE ENGINE

Before starting the engine, check to see that the external GTC starter unit is operating and that no personnel or equipment are within the intake and exhaust blast danger areas. See figure 2-2 for danger areas. Make certain that fire fighting equipment is available and manned. The aircraft may be operated on alternate fuel Specification MIL-F-5624 JP-3. Refer to Section V for operating limitations to be observed when using alternate fuel.

STARTING THE ENGINE

NORMAL STARTING PROCEDURE

The following procedure should be followed when starting the engine while on the ground:

a. Throttle lever........................ "off"

b. ENGINE MASTER switch.................. "ON"

c. ENG CONT switch...................... "PRIMARY"

d. START switch.......................... depress

e. When the tachometer indicates 12-16%, move the throttle lever outboard to the "IGNITE" position to start the ignition cycle, and then forward to "idle".

The ignition circuit will remain energized for 30 seconds and light-off should occur within this period of time. The engine should accelerate smoothly to 55-65% rpm during which time the turbine outlet temperature must not exceed 630°C. After reaching idle rpm, the turbine outlet temperature should stabilize at or below 340°C. Normal oil pressure is 30-50 psi at idle rpm. After the engine is operating satisfactorily, direct the ground crew to disconnect the external GTC starter, place the EXTERNAL POWER switch at "INTERNAL", and remove external electrical power unit.

Note

During starting, when the engine reaches a speed of approximately 50% rpm, the START switch should pop up. If it does not, the switch must be pulled up to disengage the starter.

WARNING

If the throttle should at any time be inadvertently retarded to the "off" position, an immediate flame-out will occur. In this case the throttle lever must be left at "off" until a new start is made, since introducing unburned fuel into the engine will create a serious fire hazard.

For instructions to be followed in case of engine fire during starting, refer to Section III.

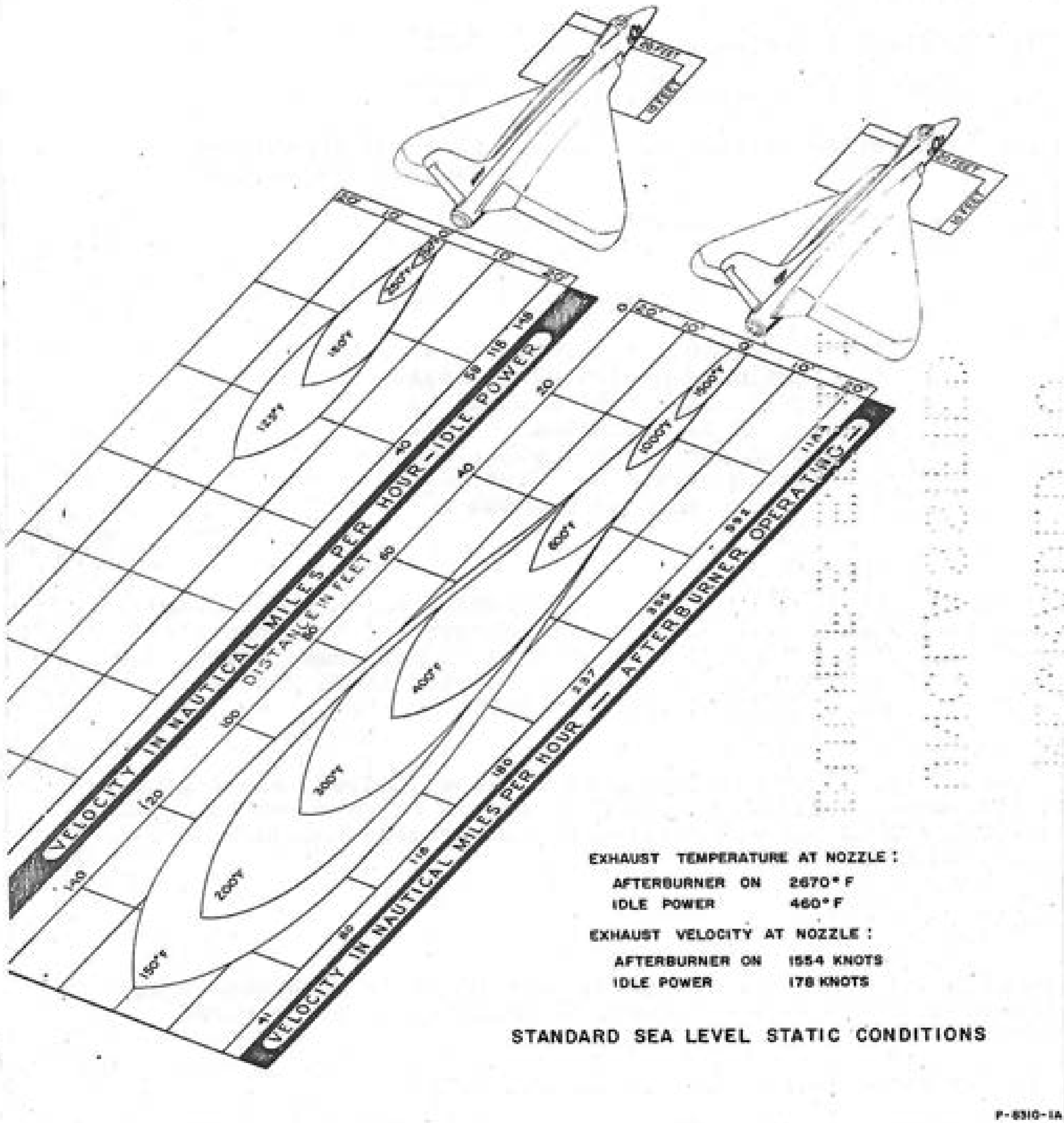

Figure 2-2. Danger Areas

UNSATISFACTORY STARTS

HOT START. A hot start is one during which the turbine outlet temperature exceeds the maximum starting acceleration temperature limit of 630°C. When a hot start is experienced, use the following procedure:

a. Throttle lever................. "off"

b. ENGINE MASTER switch........... "OFF"

c. START switch................... pull up, if starter has not disengaged

d. Investigate the cause of the hot start.

CAUTION

Before attempting a restart, a fuel drainage period of at least 30 seconds should be allowed to remove any accumulation of fuel in the engine or tail-pipe, and the engine should be "cleared" as outlined in this section of the handbook.

ABORTED START. An aborted start is one during which the engine does not start within 20 seconds after movement of the throttle to "idle", as evidenced by no rise in exhaust gas temperature and no increase in engine rpm. When an aborted start occurs, follow the procedure outlined for hot starts.

FALSE START. If, after moving the throttle to "idle", the engine starts but does not accelerate to idle rpm, a false start has occurred and the procedure stipulated for hot starts should be followed.

CLEARING THE ENGINE

When an abnormal start has occurred, or when it is suspected that fuel has accumulated in the engine, the engine may be "cleared" as follows:

a. Throttle lever................. "off"

b. ENGINE MASTER switch........... "ON"

c. START switch.......................... depress

d. Allow the starter to motor the engine for 10 to 20 seconds.

e. START switch.......................... pull up

ENGINE GROUND OPERATION

The engine requires little or no warm-up, and because of high fuel consumption while on the ground (approximately 2.5 gallons per minute at idle rpm), ground operation should be limited to as short a period as possible. The brakes are adequate to hold the airplane at military power. The aircraft must be tied down to make after-burner power checks.

GROUND TESTS

With the engine operating at idle rpm, external power disconnected, external power switch at "INTERNAL", and the wings spread and locked, perform or check the following to determine proper operation of the various aircraft systems:

a. Remote attitude indicator power warning flag.................... not visible

b. Fuel pressure indicator............... approximately 35 psi

c. Master Warning Annunciator............ not illuminated

d. Oil pressure indicator................ 30-50 psi

e. UTILITY and ELEVON PRESSURE indicators............................ 3000 psi

f. SPEEDBRAKE switch..................... "OPEN", check speedbrake position, then move to "CLOSED"

g. Operate the elevons to check for smooth, rapid response to control stick movements.

Note

The wings must be spread and locked and the elevon lock handles horizontal before the elevons can be operated, as the wing folding controls automatically lock the control system.

h. Operate the trim controls to check for proper trim position indicator reaction and control stick centering or "neutral" position displacement.

TAXIING INSTRUCTIONS

When ready to taxi, advance the throttle to an intermediate power setting above idle rpm, and when the aircraft is moving at the desired speed, retard the throttle to "idle". Idle rpm will be sufficient to maintain normal taxi speed under most conditions, and will probably require occasional braking action to prevent acceleration. Differential braking action must be used to steer the aircraft since the rudder is not effective at taxi speeds. Approximately 2.5 gallons (16.2 pounds) of fuel are consumed per minute while taxiing.

BEFORE TAKE-OFF

PRE-FLIGHT ENGINE CHECK

IDLE CHECK

After starting, allow the engine to run at idle rpm until the instrument readings have stabilized, then make the following check:

a. Throttle lever................. "idle"

b. ENG CONT switch................ "PRIMARY"

c. Tachometer..................... 55 to 65% rpm

d. Turbine outlet temperature.....340°C maximum

e. Oil pressure indicator......... 30-50 psi

f. Fuel pressure indicator........ approximately 35 psi

g. FUEL PUMP failure warning light......................... not illuminated

ENGINE POWER CHECK

a. Throttle lever................. "idle"

b. ENG CONT switch................ "PRIMARY"

c. Throttle lever........................ advance to military power, observing tachometer limitation of 102.2% rpm and 670°C turbine outlet temperature acceleration limit

d. Tachometer............................. below redline

e. Turbine outlet temperature............ 620°C maximum

f. Oil pressure indicator................ 45 ± 5 psi

g. Fuel pressure indicator............... approximately 35 psi

h. FUEL PUMP failure warning light....... not illuminated

AFTERBURNER CHECK. With the engine operating at military rpm, or at a minimum of 3% under military rpm, check afterburner operation as follows:

a. Throttle lever........................ move outboard to engage the AFTERBURNER detent and start the afterburner

b. Tachometer............................ no appreciable change

c. Turbine outlet temperature............ 630°C maximum

d. Throttle lever........................ retard to afterburner stop. (afterburner should remain lit) then, inboard to disengage AFTERBURNER detent, then to "idle"

CAUTION

If the afterburner nozzle fails to open after lighting-off the afterburner, as evidenced by a rapid rise in turbine outlet temperature and a decrease of approximately 4% rpm, immediately shut off the afterburner.

Note

After operating the afterburner on the ground, the engine should be allowed to run at idle rpm for as long as two minutes to allow fuel drainage.

EMERGENCY FUEL CONTROL CHECK

a. Throttle lever.................. "idle"

b. ENG CONT switch................. "MANUAL"

c. Throttle lever.................. slowly advance to "military" rpm

d. Tachometer...................... below redline

e. Turbine outlet temperatures..... 670°C during acceleration, stable at or below 620°C

CAUTION

Extra care must be taken not to over-temperature the engine when using the emergency fuel control system.

f. Throttle lever.................. "idle", 55 to 65% rpm

g. ENG CONT switch................. "PRIMARY"

PRE-FLIGHT AIRCRAFT CHECK

a. Canopy.......................... closed and latched

b. CANOPY UNLOCKED light........... out

c. Wings........................... spread and locked

d. SPEEDBRAKE switch............... "CLOSED"

e. AUX FUEL TRANS switch........... "OFF"

f. Air conditioning temperature control switch........................ temperature as desired

g. WSHLD DEFOG switch (1)................ "ON"

h. Trim

Rudder............................... "0"
Elevator............................. 6-8 degrees NOSE UP
Lateral.............................. "neutral"

i. SHOULDER HARNESS control.............. "LOCKED"

j. Check all controls for freedom of movement and response.

k. THROTTLE FRICTION & LOCK.............. as desired

TAKE-OFF

AIRFIELD

After lining up on the runway, advance throttle to the "TAKE-OFF" position observing engine instruments for satisfactory indications while holding the airplane in the take-off position.

Note

> If an afterburner assisted take-off is desired, the throttle lever is moved outboard to "AFTERBURNER" only after the instruments have stabilized at "military" rpm and the brakes have been released.

If satisfied with engine operation, release the brakes and begin the take-off run. The aircraft is inherently stable and has no unusual take-off characteristics. Maintain direction by use of differenctial braking; only a slight pressure or "tapping" on the brake pedals should be necessary for steering until the rudder becomes effective for directional control at approximately 50 knots. Do not lift the nose wheel from the runway prior to the actual take-off. Due to the configuration of the aircraft, lifting the nose wheel from the runway during the take-off run will greatly increase drag and lengthen the take-off run unnecessarily. Typical take-off speed is 150 knots IAS at 26,000 pounds gross weight. At take-off speed the aircraft should be lifted gently from the ground. Since the aircraft is not equipped with flaps or other controllable high-lift devices, no differentiation in take-off technique can be published for minimum run or obstacle clearance type take-offs. Refer to Section III for procedure in the event of an emergency during take-off.

(1)Airplane BuNo. 139209.

AFTER TAKE-OFF

a. Retract the landing gear as soon as the point is reached beyond which a safe landing cannot be made on the runway or any level area available immediately beyond the runway.

Note

Landing gear retraction time is a maximum of 6 seconds. Landing gear should be fully retracted before airspeed is allowed to exceed 250 knots. (This is a hydraulic system limitation).

b. Re-trim the aircraft as necessary.

c. Re-position the throttle lever as required by operating conditions.

Note

The afterburner may be shut-off after a safe airspeed is attained, after altitude is reached, or at the dictates of the mission.

CLIMB

The aircraft should be accelerated to climbing speed as soon as possible after take-off to obtain maximum performance. The airplane may be trimmed to obtain the desired climb and has no unusual characteristics. With no external stores, using military thrust at 26,000 pounds gross weight, the climb should be initiated at 380 knots IAS at sea level. Using afterburner with military power at 26,000 pounds gross weight, the climb should be scheduled to maintain the highest airspeeds consistent with maximum speeds set forth in Section V. A climb Mach number of 0.90 is considered optimum above 10,000 feet. For most economical operation, military thrust should always be used during a climb, observing at all times the engine rpm, turbine outlet temperature and time limitations set forth in Section V. After passing through 10,000 feet the automatic systems are placed into operation as follows:

a. YAW DAMPER AUTOCONTROL (2) or AUTOMATIC SYSTEMS (1) switch.."ON"

b. YAW DAMP button.................. depress

(1) Airplanes BuNo. 139208, 142349-142350.
(2) Airplane BuNo. 139209.

c. PITCH DAMP button [1]....... depress

d. TRANSONIC TRIM switch....... "ON"

AFTER CLIMB

If auxiliary fuel is carried in the forward fuselage and aft wing tanks, auxiliary fuel transfer must not be begun until after the climb is completed. To start the transfer operation, and at the same time check for proper functioning of the auxiliary fuel transfer system, perform the following:

a. Note FUEL QUANTITY indication of the FWD FUSE TANK and AFT WING TANK gages. The tanks should contain 265 gallons (1722 pounds) and 335 gallons (2177 pounds) respectively, if serviced to capacity.

b. AUX FUEL TRANS switch......."INTERNAL"

c. AUX TANK SELECTOR switch..... "FWD ONLY"

d. Ascertain that forward fuselage auxiliary tank fuel is transferring to the aft fuselage sump tank by observing FWD FUSE TANK FUEL QUANTITY indicator for a reduction in fuel quantity. Aft fuselage sump tank FUEL QUANTITY should increase from 168 gallons (1092 pounds) to some value up to as much as 180 gallons (1170 pounds) depending on engine power setting.

e. When forward fuselage tank fuel quantity is depleted to 110 gallons (715 pounds), place AUX TANK SELECTOR switch at.........."AFT ONLY"

f. Ascertain that aft wing auxiliary tank fuel is transferring to the aft fuselage sump tank by observing AFT WING TANK FUEL QUANTITY indicator for a reduction in fuel quantity. Aft fuselage sump tank fuel quantity should not show a reduction as a result of the switch-over.

g. When aft wing tank fuel quantity is depleted, place the AUX TANK SELECTOR switch at.."FWD ONLY"

h. When forward fuselage tank fuel quantity is depleted, place AUX FUEL TRANS switch at... "OFF"

If auxiliary fuel fails to transfer during any of the above procedure, the AUX FUEL TRANS switch should be placed at "OFF" immediately and the balance of the flight planned on a basis of aft fuselage sump and forward wing tank fuel availability only. Observe airplane sink speed limitation when landing. Refer to Section V.

(1) Airplanes BuNo. 139208, 142349-142350.

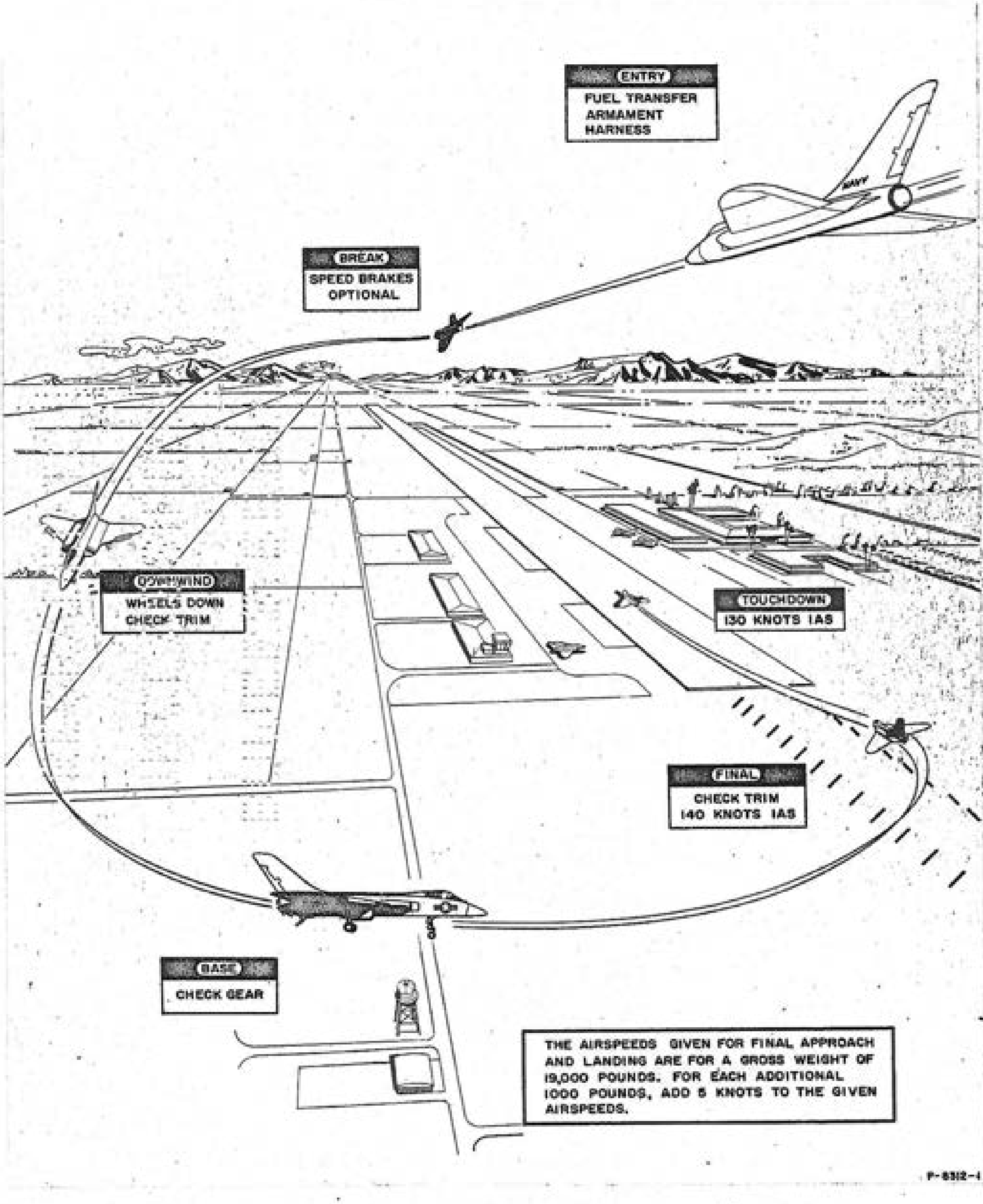

Figure 2-3. Landing Pattern Diagram

FLIGHT CHARACTERISTICS

For information regarding flight characteristics refer to Section VI.

DESCENT

Any type of descent may be made to meet the requirements dictated by local weather conditions, airfield position, fuel remaining, or established penetration procedures. Turn the TRANSONIC TRIM switch "OFF" at Mach numbers below 0.70. Turn the YAW DAMPER AUTOCONTROL (2) or AUTOMATIC SYSTEMS (1) switch to "EMER OFF" at altitudes below 10,000 feet. To make a maximum range descent, throttle back to idle rpm and maintain a trimmed airspeed of approximately 250 knots IAS. To obtain a maximum rate of descent, open the speedbrakes and dive, regulating power and dive angle as required to prevent exceeding the airspeed limitations stipulated in Section V. Refer to Section VI for diving characteristics of the aircraft.

PRE-TRAFFIC PATTERN CHECK LIST

Prior to entering the traffic pattern, the following checks should be made:

a. SHOULDER HARNESS control........"LOCKED"

b. YAW DAMPER AUTOCONTROL (2) or AUTOMATIC SYSTEMS (1) switch......"EMER OFF"

c. TRANSONIC TRIM switch..........."OFF"

d. YAW DAMP button.................out

e. PITCH DAMPER button (1).........out

f. SPEEDBRAKE switch...............as desired

g. Trim control....................as required

h. AUX FUEL TRANS switch..........."OFF"

TRAFFIC PATTERN CHECK LIST

Refer to Section V for landing gross weight and C. G. limitations.

a. ARREST HOOK control............."UP"

b. LANDING GEAR control............"DOWN"

c. WHEELS position indicator.......miniature wheel visible

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d. SPEEDBRAKE SWITCH............... as desired, but speedbrakes open not recommended

e. Trim control.................... as required

LANDING

It is advisable to maintain as high an rpm as is compatible with approach conditions in order to reduce the time delay for engine acceleration, should a wave-off become necessary. Refer to figure 2-3 for recommended approach and landing speeds. Refer to Section III for information concerning landing emergencies.

AIRFIELD

The landing characteristics of the airplane are normal. Never permit the airplane to stall in. At a gross weight of 18,000 pounds make the final approach at 135 knots IAS and touch down at approximately 125 knots IAS. After touch down, the aircraft can be steered with the rudder down to a speed of approximately 50 knots IAS, after which directional control must be maintained by differential braking action.

WARNING

Do not allow the airspeed to fall below the recommended minimums into the stall warning region. The drag at the high angles of attack resulting from operation at airspeeds in the stall warning region becomes of such proportions as to require full military power, with afterburning, to maintain altitude. See Section VI, LOW SPEED FLIGHT.

HEAVY WEIGHT

Approach and landing with the aircraft heavily loaded must be made at proportionately higher airspeeds. The aircraft is currently restricted to 18,906 pounds gross weight for field landings, and consumption of surplus fuel or jettisoning of external stores is necessary if the aircraft exceeds this weight.

CROSS WIND

The aircraft possesses no unusual characteristics during cross wind landings. Normal tricycle landing gear cross wind landing techniques may be used.

MINIMUM RUN

A landing utilizing the minimum amount of runway can be accomplished by simulating a carrier approach to the airfield. Make the approach as slow as possible, observing minimum recommended approach speeds. Attempt to cross the landing end of the duty runway at an altitude of 5 to 10 feet. Immediatley retard the throttle to "idle" rpm and land. After touchdown, keep the nosewheel firmly on the runway and apply the wheel brakes, being careful not to skid the tires.

WAVE-OFF

Wave-off characteristics are considered good if the approach speed is at or above the recommended minimums. Control is excellent and the rate of descent can be stopped almost immediately after applying power. In the event of receiving or taking a wave-off, proceed as follows:

a. Throttle lever........................"military" power

b. LANDING GEAR control................... "UP"

c. SPEEDBRAKE, if used.................... "CLOSED"

d. Trim control.......................... as required

WARNING

If use of the afterburner is necessary to recover from a critical condition such as airspeed below recommended minimums during a wave-off, the pilot must be cognizant of the fact that fuel consumption will be exceedingly high (600 pounds per minute). Do not operate the afterburner longer than is absolutely necessary to regain safe flying speed. Approximately two seconds time lapse should be expected between the time of afterburner application and thrust increase.

AFTER LANDING

After completing the landing run, check or perform the following:

a. YAW DAMP switch................ out

b. SPEEDBRAKES, if open............ "CLOSED"

c. Elevator trim................... "0" units NOSE up

d. Lateral trim.................... "neutral"

e. AIR COND switch................. "RAM"

f. WSHLD DEFOG switch (1) .... "OFF"

g. Canopy.......................... open if desired and relative headwind is below 88 knots

POST-FLIGHT ENGINE CHECK

Prior to stopping the engine, all instruments should be checked for proper indications. Any discrepancies noted must be referred to the proper personnel, and entries of such descrepancies must be inserted on the proper forms.

STOPPING THE ENGINE

Follwing high power operation, the engine should be allowed to idle for 5 minutes for cooling purposes prior to shut-down then:

a. Throttle lever.................. advance to 75-80% rpm for 20 seconds to clear engine oil sump, then to "off"

b. PITOT & ENG ANTI-ICING switch... "OFF"

c. ENGINE MASTER switch............ "OFF"

d. Check that the engine decelerates freely.

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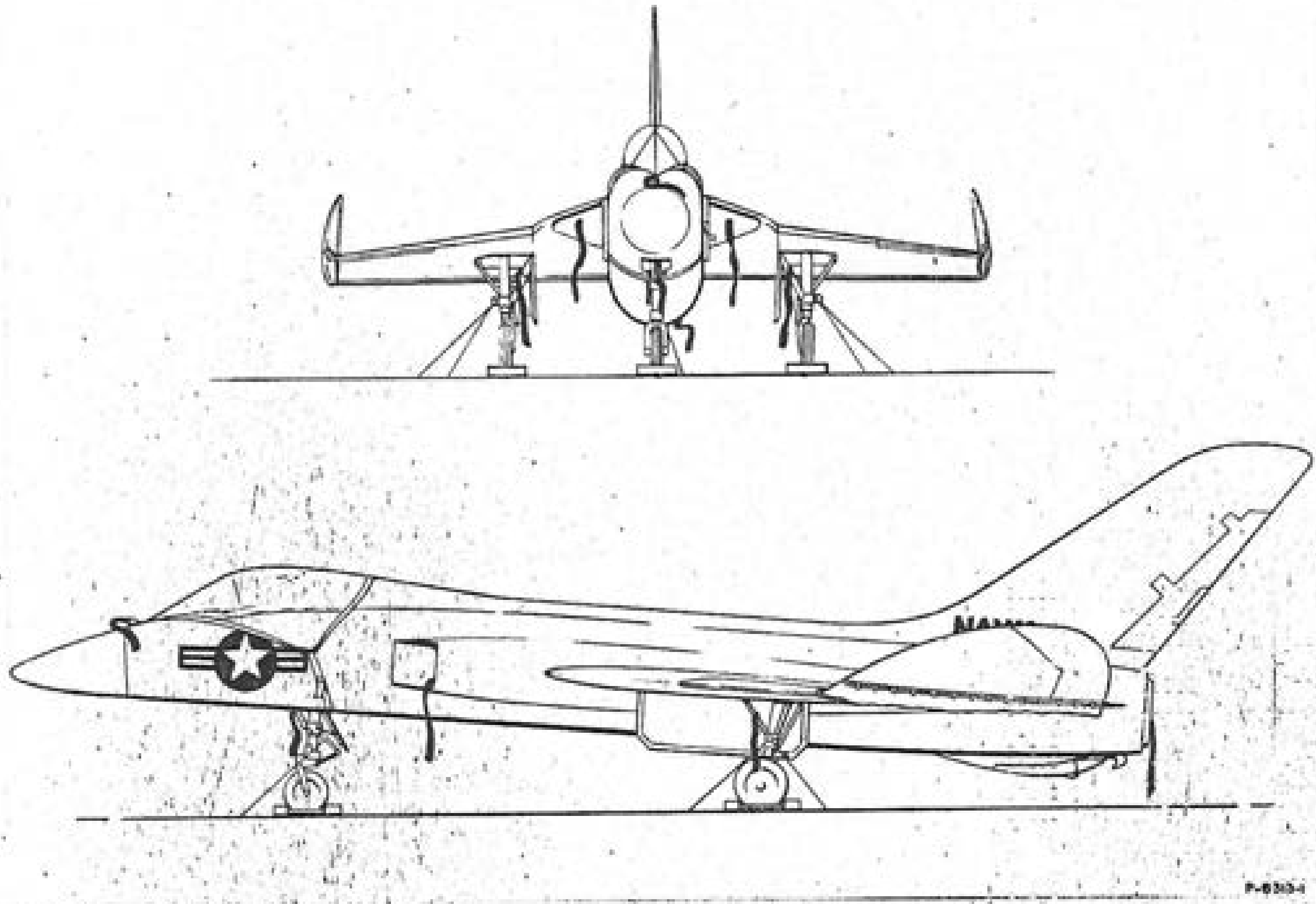

Figure 2-4. Mooring Diagram

BEFORE LEAVING THE AIRCRAFT

a. OXYGEN control........................ "OFF"

b. AN/ARC-27A function switch........... "OFF"

c. FUEL VALVE CONTROL................... "CLOSE"

MOORING

a. Wheels chocked

b. Intake and exhaust plugs inserted

c. Cockpit cover installed

d. Pitot tube cover in place

e. Landing gear downlock pins inserted

f. Aircraft tied down (see figure 2-4)

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SECTION III

EMERGENCY PROCEDURES

ENGINE FAILURE

Indication of impending engine failure will usually be in the form of unstable engine operation, and may be recognized by any, or a combination of the following symptoms:

a. Erratic increase of turbine outlet temperature.

b. Rapid reduction or fluctuation of engine rpm.

c. No increase in rpm when throttle is advanced.

d. Loss of thrust.

e. Compressor pulsation.

If unstable engine operation occurs, it is possible that the engine may run at some reduced thrust setting as a result of retarding the throttle to a point where the engine is "caught" before a flame-out occurs. If unstable operation occurs during engine acceleration or deceleration, retard the throttle to "idle" until engine operation becomes stable, then reset the throttle slowly to the desired position. If unstable operation occurs during steady state operation, immediately reduce power, reduce altitude, and increase airspeed by changing the attitude of the airplane. If necessary, due to continued excessive instability, shut down the engine.

PROCEDURE ON ENCOUNTERING ENGINE FAILURE

FLAME-OUT. In the event of a flame-out, proceed as follows:

a. Throttle lever........................ "off"

~~b. ENGINE MASTER switch.................. "OFF"~~ NEVER.
MAY NOT REOPEN

c. EMER PWR handle........................ pull

d. Check for evidence of fire by observing the following:

- FIRE warning light illuminated
- Smoke or fumes in the cockpit
- Emission of smoke from tailpipe, accessories compartment ventilating ducts, or burner, turbine, and afterburner compartment ventilating ducts.

e. If considered safe to do so, attempt to start the engine as prescribed under AIR STARTING PROCEDURE in this section.

ENGINE FUEL CONTROL FAILURE. If engine fuel control unit failure is suspected as the cause for unstable engine operation, proceed as follows:

a. Throttle lever.................."idle"

b. ENG CONT switch................ "MANUAL"

c. Advance throttle slowly to obtain desired thrust, observing maximum temperature and rpm limitations.

CAUTION

When safety of flight necessitates, the transfer to "MANUAL" fuel control may be made at any throttle lever setting but, whenever possible, should be made with the throttle lever at "idle" rpm.

ENGINE DRIVEN FUEL PUMP FAILURE. If a component of the engine combination fuel pump should fail, indications and effects to be anticipated are as follows:

Engine Stage. The FUEL PUMP failure warning light will illuminate, automatic transfer to the afterburner stage will be effected, and full fuel flow requirements will be met for the engine, with partial flow to the afterburner at low altitudes. At high altitudes, both engine and afterburner fuel requirements will be satisfied.

Afterburner Stage. No afterburning will be possible, but full fuel flow to the engine will be provided.

Centrifugal Boost Stage. Little or no effect on engine and afterburner operation will be experienced unless the electric fuel boost pump has also failed; then there will be some loss of power, especially on initial climb-out, but no flame-out should result.

AIR STARTING PROCEDURE. Successful air starts may be accomplished as high as 35,000 feet within an engine windmilling range of 12 to 30% rpm. Listed below are the approximate minimum and maximum indicated airspeeds, at various altitudes, which will furnish windmilling rpm within air starting requirements.

Pressure Altitude	IAS Range, Knots
Sea Level	150-350
5,000	150-335
10,000	150-310
15,000	150-284
20,000	150-258
25,000	150-232
30,000	150-206
35,000	150-180

It is recommended that the higher airspeed values be used to give more consistent starts, and that no air starts be attempted above 35,000 feet. The procedure for a normal air start is as follows:

a. Throttle lever........................ "off"

b. ENGINE MASTER switch.................. "ON"

c. ENG CONT switch....................... "PRIMARY" ("MANUAL" if fuel control malfunction is suspected)

d. EMER PWR handle....................... pull

e. Obtain correct windmilling rpm flight speed.

f. Throttle lever........................ "IGNITE", then forward to "idle"

g. Engine should start within 20 seconds and accelerate to idle rpm.

Note

Idle rpm increases with altitude, therefore an idle speed of as high as 80% rpm will not be unusual at high altitudes.

h. Move the throttle to the desired setting after the engine speed has stabilized.

i. Retract the emergency generator and hydraulic pump.

CAUTION

If light-off does not occur within 30 seconds after the throttle has been removed to "idle," or if an unsatisfactory start is indicated in any other way, retard the throttle to the "off" position and allow the engine to windmill for 30 seconds to allow fuel drainage before attemping another start.

If the afterburner should flame-out, pull the throttle inboard, disengaging the afterburner. Wait a minimum of 5 seconds and then move the throttle outboard to re-light the afterburner.

ENGINE FAILURE UNDER SPECIFIC CONDITIONS

DURING TAKE-OFF. If the engine fails during take-off, but before becoming airborne:

a. Throttle lever.................. "off"

b. ENGINE MASTER switch............."OFF"

c. Apply wheel brakes as necessary to stop the aircraft.

d. If there is insufficient runway to achieve a stop and the pilot elects to retract the gear, release the landing gear retraction release latch and move the LANDING GEAR control to "UP".

If the engine fails after becoming airborne, LAND STRAIGHT AHEAD. Perform as many of the following operations as possible:

a. Throttle lever........................ "off"

b. Jettison external stores.

c. LANDING GEAR control.................. as desired

d. ENGINE MASTER switch.................. "OFF"

e. EMER PWR handle....................... pull

DURING FLIGHT. If the engine fails during flight, attempt an air start as prescribed under AIR STARTING PROCEDURE if considered safe to do so. If an air start is not possible, turn off all unnecessary switches, establish the recommended gliding speed as shown under MAXIMUM GLIDE, and perform the operations set forth under LANDING WITH NO THRUST. Should the engine fail during night operations over unfamiliar terrain and the engine cannot be started, eject.

MAXIMUM GLIDE. Maximum range gliding speeds for various gross weights with the engine inoperative are as follows:

Gross Weight - Pounds	IAS - Knots
18,000	230
21,000	250
24,000	265
27,000	280

LANDING WITH NO THRUST. The recommended airspeed for a landing approach without thrust is 180 knots IAS. This airspeed will insure adequate control under every condition. Prior to making a forced landing accomplish as many of the following as possible:

a. Throttle lever........................ "off"

b. ENGINE MASTER switch.................. "OFF"

c. EMER PWR handle....................... pull

d. Jettison external stores.

e. SHOULDER HARNESS control........ "LOCKED"

f. FUEL VALVE control.............. "CLOSE"

g. AUTO CONT release handle........ pull

h. EMER ELEVON MECH
ADVANTAGE crank................. crank to "2" position

i. LANDING GEAR.................... as desired

FIRE

ENGINE FIRE

ON THE GROUND. If an engine fire should occur while on the ground, immediately perform the following:

a. Throttle lever.................. "off"

b. FUEL VALVE control.............. "CLOSE"

c. Ascertain that the starter has a source of external compressed air.

d. START switch.................... depress

CAUTION

It must be definitely established that the fire is in the engine before the starter is used. If the fire is in the accessory section beneath the engine, the introduction of starter exhaust air may intensify combustion.

e. Allow the starter to motor the engine until the fire has disappeared. If fire persists for a considerable length of time, continue cranking the engine and signal the fire guard to apply CO_2 to the engine air intake duct as required.

CAUTION

CO_2 must not be applied to the hot turbine blades through the exhaust duct.

IN THE AIR. When indications of fire occur during flight, make certain that a fire actually exists, then proceed as follows:

a. Throttle lever........................ "off"

b. ENGINE MASTER switch.................. "OFF"

c. Extend the emergency generator.

d. If the fire is extinguished by this action, and "PRIMARY" fuel control unit malfunction is suspected, attempt an air start as outlined under AIR STARTING PROCEDURE.

e. Should the fire continue to burn, move the FUEL VALVE control to "CLOSE". If this procedure does not extinguish the fire, it is up to the pilot to elect a forced landing or ejection.

FUSELAGE FIRE

ON THE GROUND. If a fuselage fire occurs, it will most likely be in the engine accessory compartment or the electrical equipment compartment. If the fire is of electrical origin, refer to the paragraph entitled ELECTRICAL FIRE. If the fire is in the engine accessory compartment, observe the following procedure:

a. Throttle lever........................ "off"

b. ENGINE MASTER switch.................. "OFF"

c. FUEL VALVE control.................... "CLOSE"

d. Direct the ground crew to place the external power switch at "INTERNAL" if operating on external power.

e. Ascertain that the fire guard with a javelin type CO_2 nozzle introduces CO_2 into the engine compartment through the starter access door and/or the engine access door latch covers.

f. If the previous step is unsuccessful, the fire guard may penetrate the forward engine compartment with a javelin type CO_2 extinguisher.

g. If necessary, abandon the aircraft.

IN THE AIR. If a fuselage fire developes during flight in the engine compartment, causing the FIRE warning light to glow, proceed as follows:

a. Throttle lever.................."off"

b. ENGINE MASTER switch............. "OFF"

c. FUEL VALVE control............... "CLOSE"

d. EMER PWR release handle.......... pull

e. FIRE EXT switch (1).............. actuate, to flood engine compartment with extinguishing agent

f. If the fire is extinguished by the preceding steps, it is left up to the judgment of the pilot whether to elect forced landing or ejection. If the fire continues to burn, eject.

If the fuselage fire is electrical in nature, as indicated by trailing smoke from the nose wheel and equipment compartment vent exit, and there is no indication of fire by the FIRE warning light, proceed as outlined under ELECTRICAL FIRE.

WING FIRE

ON THE GROUND. If a wing fire occurs, it is probably due to an integral fuel tank leak, or electrical in nature. While on the ground, shut down the engine, turn off all electrical switches, abandon the aircraft, and clear the area while the fire fighting crew takes the necessary steps to fight the fire.

IN THE AIR. Because of the location of the integral wing fuel tanks, a fire in the wing could be caused by either fuel leakage or defective electrical components.

(1) Airplanes BuNo. 139208-139209.

a. Jettison combustible external stores

b. Follow the procedure stipulated under ELECTRICAL FIRE

c. If the fire continues to burn, eject, as the fire is probably fuel-fed and no further corrective measures can be taken.

ELECTRICAL FIRE

ON THE GROUND. When a fire occurs on the ground and is suspected to be of electrical origin, proceed as follows:

a. Throttle lever........................ "off"

b. ENGINE MASTER switch.................. "OFF"

c. Turn off all electrical equipment.

d. Disconnect external power.

e. When the fire is extinguished, investigate to determine the cause

IN THE AIR. If an electrical fire occurs while in flight, perform the following:

a. Turn off all electrical equipment

b. Observe that the fire is extinguished.

c. Turn the switches on one by one to find the defective circuit.

d. Leave the offending circuit off and use the remaining circuits only as they become essential to aircraft operation.

e. Land as quickly as practicable.

f. If the fire cannot be extinguished by the preceding action, the pilot must use his own discretion whether to attempt an emergency landing or to abandon the aircraft.

SMOKE ELIMINATION

To dissipate smoke or irritating fumes in the cockpit, turn the AIR COND switch to "RAM". This action stops the flow of pressurized air from the engine compressor section, dumps cabin pressure, and directs ambient ram air into the cockpit.

WARNING

Do not turn the AIR COND switch to "RAM" above 43,000 feet because of decompressioning effects.

LANDING EMERGENCIES (EXCEPT DITCHING)

In the event an emergency landing is necessary over land, the pilot should consider a number of variables to determine his best landing configuration. These include altitude and type of terrain, direction and velocity of the wind, and the characteristics of the airplane. The decision to lower the landing gear is left to the pilot, but should not be made until certain of the condition of the landing area from close scrutiny at low altitude. All external stores should be jettisoned.

An emergency landing should be made at or slightly above recommended approach and landing speeds. Landings at or just slightly above stalling speeds should never be attempted due to the accompanying high angle of attack and resultant high rate of sink and loss of control. If the landing emergency consists of one landing gear being down and another retracted, place the LANDING GEAR control at "UP" to allow the gear to collapse upon landing. For landing emergencies with no power, refer to the paragraph on LANDING WITH NO THRUST.

WARNING

- Do not jettison the canopy for emergency landings as this action will arm the ejection seat.

EMERGENCY ENTRANCE

Emergency entrance into the aircraft from the outside may be effected by operating the normal external CANOPY control to "OPEN," or by pulling the EMER CANOPY JETTISON exterior control on either side of the fuselage below the cockpit. If the aircraft has overturned and the canopy cannot be opened, break the transparent panels with an axe or other tools to gain entrance.

EJECTION

Escape from the aircraft is accomplished through use of the ejection seat. The following procedure is recommended if time permits:

a. AIR COND switch...................... "RAM"

b. Slow the aircraft as much as possible.

c. Leave feet on the rudder pedals.

d. SHOULDER HARNESS control.............. "LOCKED"

e. Sit erect with the spine straight and the head firmly against the headrest.

f. Grasp the face curtain handle with both hands and pull down as far as possible, or pull up on the auxiliary front handle ejection control, marked PILOT EJECTION.

g. If ejection occurs above the preset altitude, the parachute will not automatically deploy until after descent below the preset altitude. If ejection occurs below the preset altitude and the automatic barometric ripcord opener seems not to function, open the parachute with the manual ripcord when clear of the seat. In event the integrated harness is not automatically released following ejection, seat separation may be attained while falling, by pulling the HARNESS RELEASE handle. Actuation of the HARNESS RELEASE handle following ejection will disconnect the automatic parachute opener and will therfore require manual pulling of either the ripcord or the automatic parachute actuating lanyard.

Note

The following WARNINGS do not apply when the automatic parachute openers are installed.

WARNING

- If high speed ejection is made, wait until your speed is reduced before opening the parachute.

- If high altitude ejection is made, free fall to approximately 14,000 feet before opening parachute.

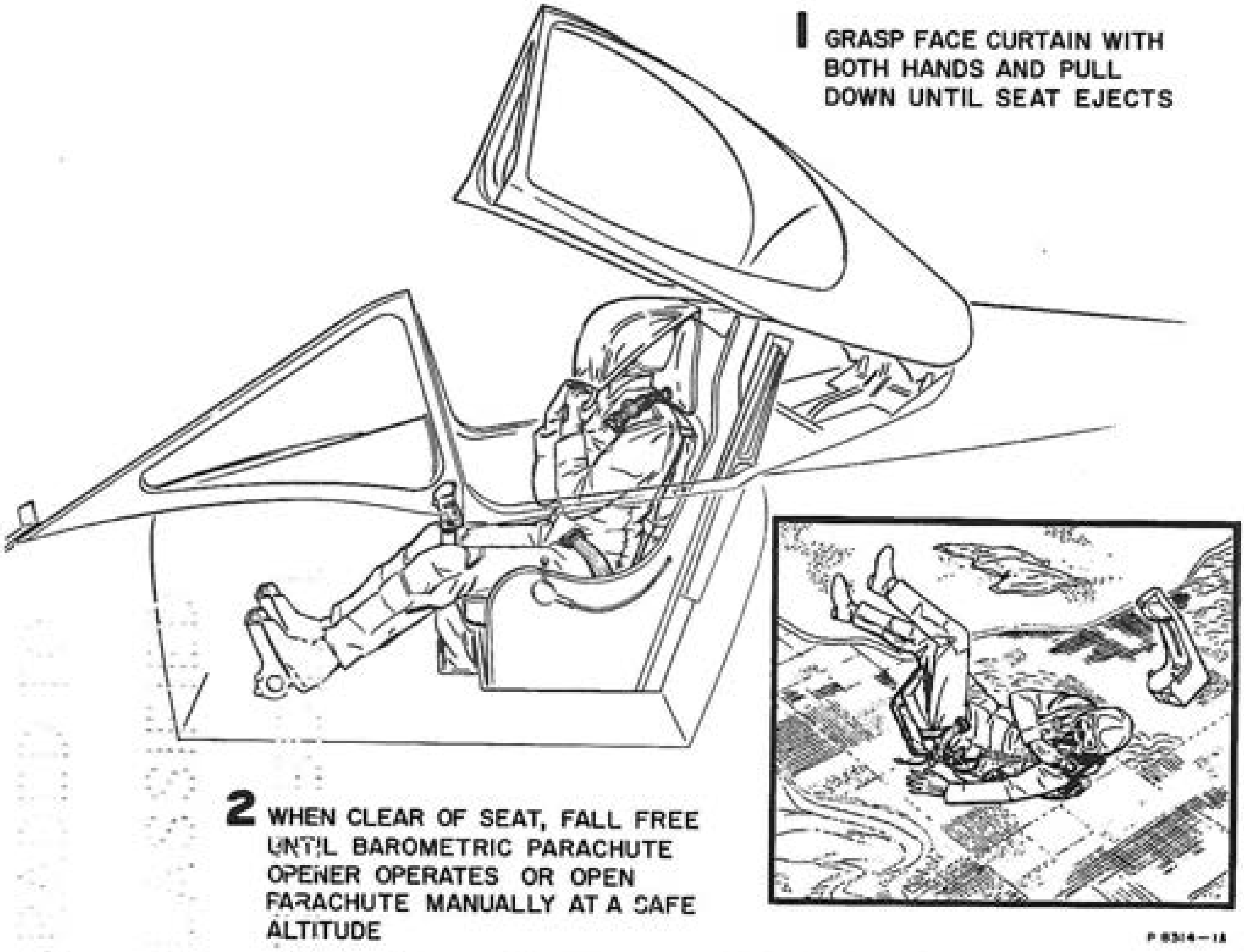

Figure 3-1. Canopy Jettison and Seat Ejection

When the emergency condition requiring ejection is such that ejection must be made without hesitation, simply grasp the face curtain handle and pull down. (See figure 3-1.)

AIRCRAFT SYSTEMS

FUEL SYSTEM

BOOST PUMP FAILURE. Failure of the fuel boost pump will be indicated by illumination of the FUEL BOOST PUMP failure warning light on the right-hand console. If the boost pump failure is electrical in origin, the engine combination fuel pump will draw enough fuel to sustain military power at high altitudes. If the boost pump failure is mechanical in nature, such as a locked rotor, the engine combination fuel pump will draw enough fuel for normal rated power. Afterburning will not be possible with the boost pump inoperative.

CAUTION

The approach for landing with the boost pump inoperative should be made with extra caution, as military power is not available at sea level in the event it becomes necessary to take a wave-off.

FUEL TRANSFER FAILURE. If one of the air driven forward wing tank transfer pumps fails, the remaining transfer pump will supply sifficient fuel flow to the aft fuselage sump tank to maintain military power fuel requirements. Some inter-transfer of fuel from the forward wing tank with the inoperative transfer pump to the forward wing tank with the operative transfer pump will occur by gravity flow through the pressure fueling piping system, (1) or crossfeed line. (2) If the engine is operated at a low power setting, nearly all the fuel in the wing tank containing the inoperative transfer pump will be available for consumption. With one transfer pump inoperative, the remaining transfer pump will not supply sufficient fuel flow during afterburning to maintain the aft fuselage sump tank fuel level. Indication of transfer pump failure during afterburner operation will occur when the fuel level in the aft fuselage sump tank drops below the level of the fuel quantity indicating system thermistor, which will result in a sudden drop from any existing fuel quantity indication on the TOTAL FUEL QUANTITY indicator to a value of approximately 923 pounds (142 gallons) in three aircraft, (3) or 650 pounds (100 gallons) in one aircraft. (4)

In the event that both air driven fuel transfer pumps fail, no indication of such failure will manifest itself until the aft fuselage sump tank fuel level drops below the fuel quantity indicating system thermistor, again resulting in a sudden drop from any existing fuel quantity indication on the TOTAL FUEL QUANTITY indicator to a value of approximately 923 pounds (142 gallons) in three aircraft, (3) or 650 pounds (100 gallons) in one aircraft. (4) Needless to say, it is recommended that the engine be operated at the lowest possible power setting and a landing be effected immediately.

CAUTION

Do not operate the aircraft over 30 minutes with the forward wing tanks empty and the fuel transfer pumps not submerged in fuel.

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(2) Airplanes BuNo. 139208-139209.
(3) Airplanes BuNo. 139208, 142349-142350.
(4) Airplanes BuNo. 139209.

~~AUXILIARY FUEL TRANSFER FAILURE.~~ (2) If the auxiliary fuel transfer pressurized air supply fails for any reason, transfer of auxiliary fuel to the aft fuselage sump tank will stop immediately. No immediate indication or signs of this failure will be apparent to the pilot on the TOTAL FUEL QUANTITY indicator, as the fuel in the forward wing tanks will begin to transfer automatically to the aft fuselage sump tank when the fuel level in the sump tank drops from the auxiliary fuel transfer float valve level to the forward wing tanks fuel transfer float valve level. With the AUX FUEL TRANS switch set at "INTERNAL", and the auxiliary fuel transfer system having failed, the fuel quantity indicator will read total fuel aboard until the forward wing tanks transfer fuel supply is exhausted and the fuel level in the aft fuselage sump tank drops to approximately 650 pounds. At this time, all fuel quantity readings will be dropped out automatically and the system will indicate aft fuselage sump tank fuel quantity only, signaling the pilot that the transfer of auxiliary fuel has ceased and sump tank fuel only is available for consumption. If this condition occurs, it will be necessary to land as soon as possible, using the lowest practicable power settings, as less than 650 pounds (100 gallons) of fuel remains. To check operation of the auxiliary fuel transfer system during flight, refer to Section II, AFTER CLIMB.

AUXILIARY FUEL TRANSFER FAILURE. (1) If the auxiliary fuel transfer pressurized air supply fails for any reason, transfer of auxiliary fuel to the aft fuselage sump tank will stop immediately. Indication of this failure will be almost immediate as the sump tank fuel level drops from the auxiliary fuel transfer float valve level to the forward wing tanks fuel transfer float valve level. Even with the AUX FUEL TRANS switch set at "INTERNAL" and the auxiliary fuel transfer system having failed, the reading of auxiliary fuel on the TOTAL FUEL QUANTITY indicator will be dropped out as a result of the action of the upper level thermistor and relay control unt. If this condition occurs it will be necessary for the pilot to plan the remainder of the flight on the basis of remaining available fuel in the forward wing and aft fuselage sump tanks. To check operation of the auxiliary fuel transfer system during flight, refer to Section II, AFTER CLIMB.

OIL SYSTEM

Upon indication of oil system failure by illumination of the OIL PRESSURE indicator, the engine should be operated at the lowest practicable thrust setting and a landing effected as soon as possible.

(1) Airplanes BuNo. 139208, 142349-142350.
(2) Airplane BuNo. 139209.

ELECTRICAL SYSTEM

MAIN GENERATOR. If the main generator should fail, all electrical equipment will be rendered inoperative immediately. The most readily recognizable indications of main generator failure will occur simultaneously as follows:

- AC GEN failure warning light.............. illuminated
- Landing gear position indicators.......... cross-hatched "unsafe" indication
- Gyro horizon indicator.................... "OFF"
- Complete radio failure

When main generator failure is indicated, extend the emergency generator.

CAUTION

- To prevent damage to the generator door, do not extend the emergency generator at airspeeds above 350 knots I.A.S.
- During emergency generator operation the fuel boost pump is inoperative. Afterburning will not be possible and high power settings at low altitudes should be avoided.
- The emergency generator will not deliver sufficient electrical power to operate the flight instruments and UHF radio at airspeeds below 165 knots I.A.S.

When operating on the emergency generator, perform the following:

a. AUTO CONT release handle.............. pull

b. EMER ELEVON MECH ADVANTAGE crank...... Move to operating position, then to "INCREASE" or "DECREASE" as desired.

c. Effect a landing as soon as practicable.

TRANSFORMER-RECTIFIER. If the transformer rectifier fails, no d-c electrical power will be available. Failure will be apparent by the malfunctioning of all d-c powered equipment with no discrepancies in the operation of a-c powered equipment. The most readily recognizable indications of transformer-rectifier failure will occur simultaneously as follows:

- Landing gear position indicators.... cross-hatched "unsafe" indication
- Trim position indicators............ inoperative
- M.A.C.S. indicator.................. inoperative
- SPEEDBRAKE switch................... inoperative
- Cabin temperature control........... inoperative

No remedial action can be taken to correct transformer-rectifier failure. It is recommended that a landing be made as soon as expediency permits.

WARNING

With the transformer-rectifier inoperative, the FIRE warning circuit is disabled, and re-lights in the event of flame-out are not possible.

HYDRAULIC SYSTEM

Failure of either the utility or elevon hydraulic systems will render all equipment connected to the affected system inoperative. Refer to Section I, HYDRAULIC SYSTEM, for a list of hydraulic equipment operated by each system. For action to be taken in the event that both hydraulic systems fail, refer to FLIGHT CONTROL SYSTEM, in this section.

FLIGHT CONTROL SYSTEM

HYDRAULIC POWER FAILURE. If either the elevon or utility hydraulic power systems fail during supersonic flight, terminate any accelerated maneuver and reduce speed. The remaining operative hydraulic power system will provide adequate control in the sonic speed range. If the utility hydraulic system fails, pull the AUTO CONT release handle to obtain manual control of the rudder. If both the elevon and utility hydraulic power systems fail, the following procedure should be followed:

a. Terminate any accelerated maneuver.

b. Reduce speed.

c. EMER PWR handle........................ pull

Note

> Do not extend the emergency drop-out hydraulic pump and electrical generator at speeds in excess of 350 knots I.A.S.

d. AUTO CONT release handle.............. pull

e. AUTOMATIC SYSTEMS (1) or YAW DAMPER AUTO CONTROL (2) switch........ "EMER OFF"

f. EMER ELEVON MECH ADVANTAGE crank....... operating position, crank to "INCREASE" or "DECREASE" as required

g. Trim control........................... as required

h. Terminate flight as soon as practicable. Do not operate the airplane as less than 125 knots I.A.S. when employing the wind-driven hydraulic pump for flight control hydraulic pressure.

EMERGENCY MECHANICAL ADVANTAGE CHANGER. In the event of a-c electrical power failure, or other failure of the M.A.C.S. amplifier or servo motor, the M.A.C.S. is operated as follows:

a. EMER ELEVON MECH ADVANTAGE crank...... move to operating position

b. EMER ELEVON MECH ADVANTAGE crank...... rotate toward "INCREASE" or "DECREASE" as required

c. EMER ELEVON MECH ADVANTAGE indicator window...................... observe indicator for desired ratio

(1) Airplanes BuNo. 139208, 142349-142350.
(2) Airplane BuNo. 139209.

YAW DAMPER RELEASE. If the yaw rate gyro, yaw damper amplifier, a-c electrical power, utility hydraulic system, or yaw damper servo fails, the rudder may or may not lock in the position prevalent at the time of failure. To de-activate the yaw damper system and/or disengage the yaw damper servo, perform the following:

a. YAW DAMP switch.................... pull up, to de-activate the system, or

b. AUTO CONT release handle.......... pull, to disengage the system, and

c. AUTOMATIC SYSTEMS (1) or YAW DAMPER AUTO CONTROL (2) switch........................... "EMER OFF"

PITCH DAMPER FAILURE. Failure of the pitch damper is evidenced by an abrupt pitch-up or pitch-down without accompanying stick movement. Pulling out the PITCH DAMP button will eliminate any further erratic pitch damper operation. If doubt exists as to whether the pitch damper or the transonic trim compensator is malfunctioning, turning the YAW DAMPER AUTOCONTROL (2) or AUTOMATIC SYSTEMS (1) switch to "EMER OFF" will simultaneously discontinue further operation of the pitch damper, yaw damper, and transonic trim compensator.

TRANSONIC TRIM COMPENSATOR FAILURE. Failure of the transonic trim compensator will be evidenced by a steadily accelerating pitch-up or pitch-down of the airplane, accompanied by movement of the control stick in the direction of airplane movement. Such a failure may readily be controlled by the pilot through application of a counter-force on the control stick. Turning the TRANSONIC TRIM switch "OFF" will prevent further stick movement. At the discretion of the pilot the TRANSONIC TRIM switch should be placed in the "RETURN TO NEUTRAL" position and subsequently back to "OFF."

Note

Holding the TRANSONIC TRIM switch in the "RETURN TO NEUTRAL" position will cause the control stick to be motored to its original longitudinally trimmed position.

(1) Airplanes BuNo. 139208, 142349-142350.

(2) Airplane BuNo. 139209.

NORMAL TRIM CONTROL FAILURE. If either the lateral or longitudinal trim actuator motor or circuit fails, there is no action that can be taken by the pilot to remedy the condition. It will be necessary for the pilot to hold pressure on the control stick to correct for any out-of-trim condition should this type of failure occur.

LANDING GEAR SYSTEM

EMERGENCY LANDING GEAR EXTENSION. In event of utility hydraulic power system failure, the landing gear can be extended by the following procedure:

a. LANDING GEAR control handle........... "DOWN"

b. EMER LDG GR release handle............ pull

LANDING GEAR SAFETY SOLENOID. If the solenoid operated safety latch to prevent inadvertent retraction of the landing gear on the ground fails to retract after take-off, manually lift the latch and move the LANDING GEAR control handle to "UP".

BRAKE SYSTEM

EMERGENCY BRAKE OPERATION. Failure of the utility hydraulic power system will cause loss of pressure in the power boost brake system. To operate the brakes, approximately twice the normal toe pressure must be exerted. Make allowance for a longer landing roll.

(1) Airplanes BuNo. 139208, 142349-142350.
(2) Airplane BuNo. 139209.

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SECTION IV

AUXILIARY EQUIPMENT

AIR CONDITIONING AND PRESSURIZING SYSTEM

Cockpit pressurization, heating, cooling, ventilation, and canopy defrosting is accomplished by an interconnected air conditioning and pressurizing system. Hot compressed air is bled from the engine compressor section and passed through a pre-cooler and refrigeration unit where it is expanded and cooled. The cold air produced is mixed proportionately with hot air from the engine compressor section (which has by-passed the refrigeration unit) by the action of a cockpit modulating temperature control valve. This mixed air provides heating or cooling as required. Desired cockpit pressurization is obtained through the action of a cockpit pressure regulating valve which meters the release of the compressed conditioned air from the cockpit. Refer to figure 4-1 for a schedule of cockpit pressurization. This air also serves to cool the radio and electronic equipment in the equipment compartment before being exhausted overboard through a duct aft of the nose wheel well. The conditioned, pressurizing air enters the cockpit through 3 ducts; one located near the pilot's feet, and two ducts leading to the canopy where the air circulates over the transparent plastic canopy panels to provide a degree of canopy de-frosting. If air passing over the canopy becomes hot enough to threaten safety of the canopy, a canopy emergency overheat switch actuates the canopy emergency overheat shut-off valve. The canopy defrosting air emergency overheat shut-off valve automatically returns to the open position when the conditioned air temperature is reduced to a safe value. An electronic temperature control box, operated by the pilot, automatically maintains a constant selected cockpit temperature.

COCKPIT TEMPERATURE CONTROL BOX

The cockpit temperature control box electronically regulates cockpit temperature through positioning of the cockpit modulating temperature control valve. The position of the air conditioning switch on the AIR COND panel (16, figure 1-5) establishes the criterion for desired cockpit temperature as a reference voltage in the cockpit temperature control box. The cockpit temperature control box then maintains cockpit temperature by positioning the cockpit modulating temperature control valve in response to signals from four temperature-sensing elements; the cockpit temperature control pick-up, cockpit temperature control anticipator pick-up, cockpit temperature control hi-limit pick-up, and cockpit temperature control hi-limit anticipator pick-up. The cockpit temperature control pick-up, located at the cockpit air outlet, is the basic sensing unit of the control system and provides regulating signals to the cockpit temperature control box. The cockpit temperature control anticipator pick-up controls the rate of change of cockpit temperature. If a relatively small change of temperature is called for, the anticipator prevents

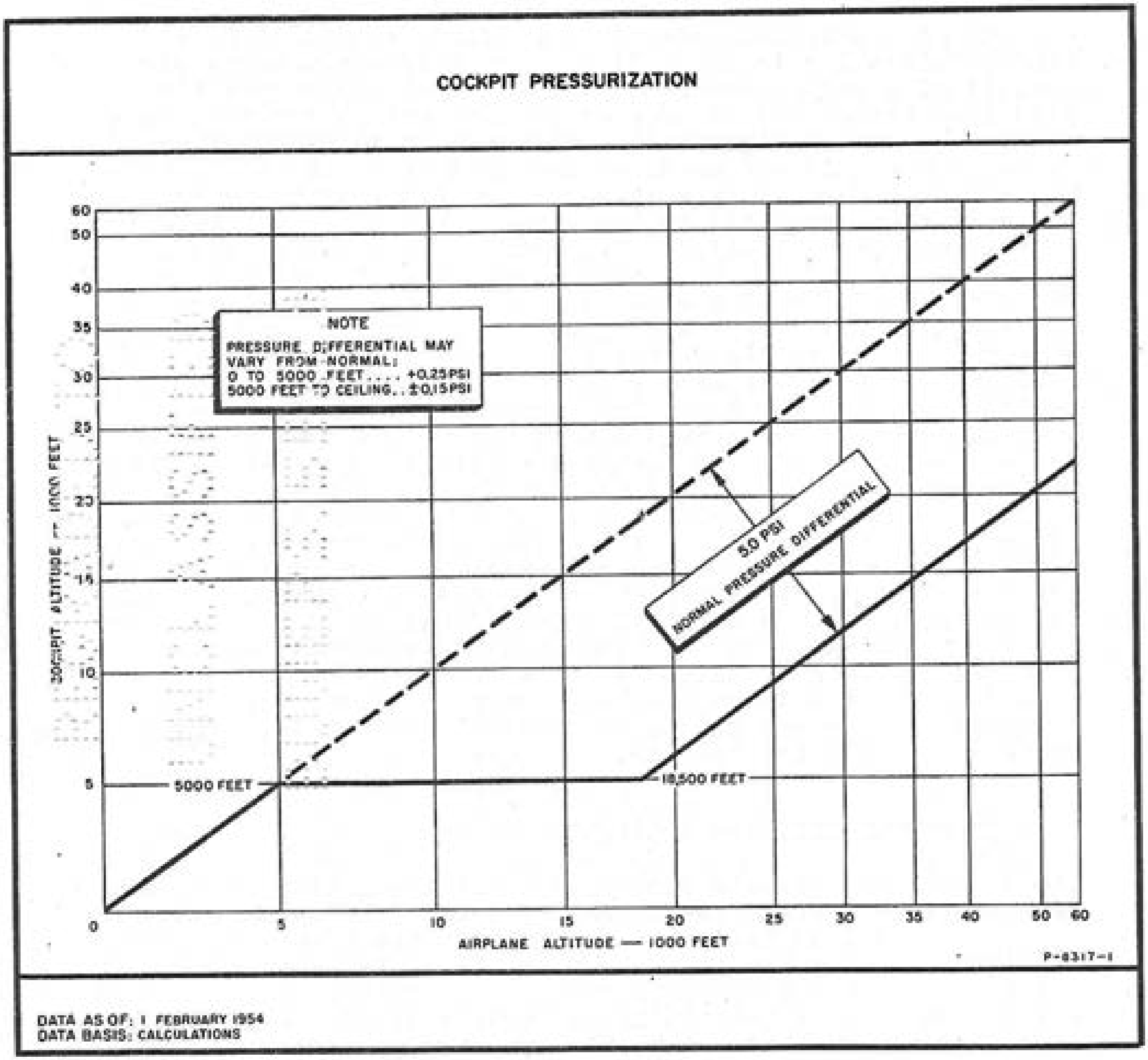

Figure 4-1. Cockpit versus Airplane Altitude Comparison Chart

an excessive amount of hot air from entering the cockpit by modifying the degree to which the cockpit modulating temperature control valve opens or closes, thus preventing cockpit temperature from fluctuating above and below the desired temperature before that temperature is established and maintained. In addition, the anticipator senses increases in temperature of the conditioned air, due to the variables of ambient conditions and engine power settings, before it arrives at the cockpit and acts to signal the control box to reduce the temperature in anticipation of such a signal from the cockpit temperature control pick-up. The cockpit temperature control hi-limit pick-up acts to limit the temperature of the conditioned air entering the cockpit to an absolute maximum of 190 ± 5° F, regardless of other signals, in order to prevent damage to heat sensitive materials in the cockpit. The cockpit temperature control hi-limit anticipator pick-up performs the same function for the cockpit temperature control hi-limit pick-up as the cockpit temperature control anticipator does for the cockpit temperature control.

AIR CONDITIONING TEMPERATURE CONTROL. The air conditioning temperature control switch on the AIR COND panel (16, figure 1-5) permits manual or automatic cockpit temperature control. This switch is calibrated from "40° -100°" F. Temperature selected in this range are automatically maintained by the cockpit temperature control box. The switch is provided with "MAN HOT" and "MAN COLD" positions to supply full hot or full cold air if desired. Placing the switch in either of the manual positions moves the cockpit modulating temperature control valve in a corresponding direction as long as the switch is in that position. When rotating the knob to the "MAN COLD" position, it must be held against spring pressure in that position for as long as it is desired to motor the modulating valve in the "cold" direction. When the knob is released it will spring into a neutral band between "MAN COLD" and "40°", and the valve will remain at the position in which it was stopped. The "MAN HOT" position of the knob motors the modulating valve towards the "hot" direction, but is not spring loaded; consequently the control will remain in "MAN HOT" when released. If full "MAN HOT" air is not desired it is necessary to return the control to the neutral position, between "MAN HOT" and "100°", manually.

AIR CONDITIONING SWITCH

The AIR COND switch (4, figure 1-5) on the right-hand console has two positions, "NORMAL" and "RAM". The "NORMAL" position of the switch allows the cockpit to be pressurized and places the cockpit temperature control system into operation. The "RAM" position of the switch closes the engine air shut-off valve and opens the cockpit pressure relief and dump valve, thus depressurizing the cockpit and admitting ram air.

COCKPIT AIR CONTAMINATION

Cockpit air contamination may occur periodically. This contamination can consist of smoke, visible oil vapor, or irritating fumes caused by the combustion of residual oil collected in the engine compressor section due to small leaks and seepage around the rotor bearings. After the engine is started, the residual oil is carried through the compressor and subsequently contaminates the air bled off to operate the air conditioning system.

The occurrence of cockpit air contamination is most likely when the airplane has been shut-down for several days or when the engine has been operated at idle rpm for extended periods. Under these circumstances, contamination will probably occure for a period of approximately 30 seconds to 2 minutes when the throttle lever is advanced to military rpm for take-off. When this condition arises, the cockpit may be ventilated by placing the AIR COND switch at "RAM" for several minutes.

COCKPIT FOG AND SNOW SUPPRESSION

The windshield de-fogging system and canopy de-frosting provisions normally require no control. Small quantities of fog, however, or even finely divided snow will on numberous occasions appear at the air conditioning outlets. While this is a normal condition resulting from the rapid cooling of air by the air conditioning unit, an excessively large volume of fog which obstructs vision can occur under extreme conditions of high humidity and high ambient air temperatures at very low altitude. This fog may be eliminated by turning the air conditioning control to increase the temperature of cockpit air. In some cases the ducting may have cooled to a point where fog will persist for a short time after the cockpit temperature has been increased. After the fog has been suppressed, a temperature setting should be selected that will provide the most comfortable temperature above the fogging point. In take-offs or landings during which fogging conditions exist, it is suggested that the AIR COND switch be turned to the "RAM" position until the climb is established or the landing is completed.

NORMAL OPERATION

BEFORE TAKE-OFF. Before take-off, place the air conditioning temperature control switch at "70°" or as desired

DURING FLIGHT. Rotate the air conditioning temperature control switch to a temperature which provides maximum confort.

AFTER FLIGHT. Place the AIR COND switch in the "RAM" position.

EMERGENCY OPERATION

If the air conditioning unit fails, the cockpit air temperature will become excessively high and operation of the "MAN COLD" position of the air conditioning control will be ineffective. In this event, place the AIR COND switch in the "RAM" position.

WARNING

> Placing the AIR COND switch in the "RAM" position depressurizes the cockpit. Aircraft altitude should not be more than 43,000 feet when the pilot is not wearing a pressure suit because of de-compressioning effects.

WINDSHIELD DE-FOGGING SYSTEM

The windshield de-fogging system is comprised of electrical heating elements, temperature control unit, and control switches. The system is completely automatic and is placed into operation whenever the LANDING GEAR control handle is in the "UP" position[1]. If the emergency generator is extended for any reason during flight, two switches, connected mechanically to the "EMER PWR" release handle, disable the right-hand windshield element in order to reduce the electrical load on the generator. One aircraft [2] incorporates a WSHLD DEFOG switch (16A, figure 1-5). The switch has two positions, "ON" and "OFF". Placing the switch at "OFF" disables the windshield de-fogging system.

ANTI-ICING SYSTEM

ENGINE

For a description of engine anti-icing provision refer to ENGINE, Section I.

PITOT TUBE

PITOT HEAT CONTROL. Heat for the pitot tube is provided by an electrical resistance element in the pitot tube head. The system may be turned "ON" or "OFF" through use of the PITOT & ENG ANTI-ICING switch on the ENGINE control panel (16, figure 1-3).

(1) Airplanes BuNo. 139208, 142349-142350.
(2) Airplane BuNo. 139209.

COMMUNICATION EQUIPMENT

MICROPHONE AND HEADSET JACK

The microphone and headset jack is incorporated into the oxygen supply tube and comprises a component of the personnel gear adapter (28, figure 1-3). The connection is located on the left-hand console adjacent to the seat.

MICROPHONE SWITCH

The mocrophone switch (9, figure 1-3) is located on the throttle grip. With the UHF receiver-transmitter in operation, the microphone switch is depressed to transmit.

UHF RADIO

AN/ARC-27A RECEIVER-TRANSMITTER

The AN/ARC-27A radio equipment provides two-way voice communication with other aircraft or surface facilities. The radio is designed to transmit or receive on any one of 1750 frequencies within a range of 225-400 megacycles. Transmission and reception are on the same frequency and through the same antenna. Remote tuning of the RT-178/ARC-27A transmitter-receiver is accomplished through use of the C-1015/ARC radio control panel.

CONTROL PANEL (C-1015/ARC-27A). The C-1015/ARC-27A radio control panel, identified as UHF and located on the right-hand console (17, figure 1-5) provides the pilot with 20 pre-set channels, 1750 manual channels, or the guard channel. The CHAN (channel) selector switch provides selection of #1 through #20 preset channels, the "G" (guard) channel, or the "M" (manual) position, the three concentric dials (frequency selectors) on the right side of the panel control the equipment frequency directly. The outer dial sets the first two digits of the frequency, the center dial sets the third digit, and the inner dial sets the digit to the right of the decimal point. The frequency of preset channels is normally set by maintenance personnel. However, if the pilot desires to select a frequency which has not been preset, the procedure is as follows:

1. Set the CHAN (channel) selector to the desired preset channel number.

2. Set the three concentric dials (frequency selectors) to the desired frequency.

3. Turn the preset button PUSH TO SET CHAN in the direction shown by the arrow next to the word "UNLOCK", until a stop is felt, and then push the button into the panel until another stop is felt.

A standard function switch provides for mode of operation as follows:

Setting	Function
"OFF"	Set inoperative
"T/R"	Transmitter and main receiver in operation; guard receiver in standby; ADF in standby
"T/R+G"	Transmitter and main receiver in operation; guard receiver in operation ADF in standby.
"ADF"	Transmitter in standby; guard receiver in standby; ADF in operation through main receiver.

The SENS control adjusts the gain of the reception. The VOL control adjusts the volume of the reception.

ADF RADIO (AN/ARA-25)

The AN/ARN-25 automatic direction finding equipment operates in conjunction with the AN/ARC-27A UHF radio communications system to provide a continuous directional indication of the source of signals in the 225-400 megacycle band. Approximate source indication in degrees of relative bearing is provided by the single-bar pointer (number one needle) of the ID-250/ARN course indicator (21, figure 1-4) for homing or direction finding purposes. Reception depends primarily on the power of the transmitting station; however, signals may be readily relied upon at distances upwards of 100 nautical miles.

OPERATION OF AN/ARN-25. The AN/ARN-25 is placed in operation by the function selector switch, labeled "OFF-T/R-T/R+G-ADF," on the UHF control panel.

a. Rotate the function switch on the UHF radio set control panel to the "ADF" position.

b. Select the desired frequency with the CHAN control.

c. Observe the direction of the signal source (relative bearing) as indicated on the azimuth scale under the single-bar pointer of the course indicator for direction finding; or, turn the aircraft until the narrow end of the single-bar pointer is under the lubber line of the course indicator to accomplish homing.

Note

Allow a three minute warm-up period if the function switch is moved to "ADF" from the "OFF" position.

CAUTION

Because ADF readings may be subject to errors of ± 20 degrees, this equipment should not be considered reliable for navigational purposes.

OXYGEN SYSTEM

The aircraft is equipped with a positive pressure, demand type liquid oxygen system. The liquid oxygen system is comprised of a five liter liquid oxygen tank, check valve, filler valve, relief valves, evaporator, shut-off valve, regulator, and associated piping. Liquid oxygen stored in an insulated tank flows through an evaporator where it is converted to gaseous oxygen and delivered to the oxygen regulator at a pressure of 70 ± 5 psi.

The oxygen regulator is mounted directly on the pilot's A-13A oxygen mask and delivers 100% oxygen under positive pressure at all altitudes. An oxygen shut-off valve, located on the left console, is the only control of the oxygen system. A liquid oxygen quantity gage is provided on the instrument panel.

OXYGEN SUPPLY. The main liquid oxygen (LOX) supply is stored in an insulated tank mounted in the left engine air intake structure. The tank filler valve is reached through an access door for servicing. The tank contains 5.0 liters of liquid oxygen when serviced to capacity. Evaporation loss is constant when the system is not is use, and this loss is used to pressurize the system. By venting any excess pressure overboard through relief valves, pressure is maintained at 70 ± 5 psi.

OXYGEN CONTROL AND EQUIPMENT. The OXYGEN control (25, figure 1-3) on the left-hand console has two positions, "ON" and "OFF". Placing the control at "ON" opens the oxygen shut-off valve and permits oxygen to be delivered to the oxygen regulator on the A-13A face mask, through the personnel gear adapter and oxygen supply tube.

CAUTION

- The type A-13A face mask used with this oxygen system should be properly fitted to the pilot's face for best results. Relatively small leaks around a mask are cumulative in effect and result in considerable oxygen loss over long periods of operation.
- Extra caution should be taken to check the personnel gear adapter for security before moving the OXYGEN control to "ON", as escaping oxygen creates a fire hazard in the proximity of oil or grease.

OXYGEN QUANTITY INDICATOR[(1)] An OXYGEN quantity gage (16, figure 1-4), located on the instrument panel, is marked "5" (full), "4", "3", "2", "1" and "0" (empty) to show the number of liters remaining. The quantity gage is electrically operated and has a small "OFF" window to indicate that the gage is not showing an accurate reading of the quantity when electrical power is lost. Also to be found on the indicator is a red "LOW LEVEL" warning light which comes on when the quantity falls below 1/2 liter.

OXYGEN QUANTITY INDICATOR.[(2)] One aircraft uses two 2 1/2 liter bottles, interconnected, to supply the five liters. Because of two pickup probes and only one indicator (16, figure 1-4), the indicator reads for the first bottle, and a "PUSH FOR AUX. BOTTLE" button (16A, figure 1-4) picks up the reading of LOX remaining in the second bottle.

OXYGEN DURATION. Figure 4-2 provides a tabulation of oxygen duration for the liquid supply at various combinations of cabin altitudes and amounts of liquid oxygen remaining in liters. It is noted that although 100% oxygen is used at all times, duration is greater at high altitudes. This is explained by the physical property of gases as affected by pressure. The volume of oxygen increases in direct proportion to the decrease in atmospheric pressure as altitude increases. Thus, while the volume of oxygen required by the pilot's lungs is approximately the same at any altitude, the 100% oxygen delivered in reduced cockpit pressure is lower in density and less of the supply is required to satisfy the demand.

[(1)] Airplanes BuNo. 139208, 142349-142350.
[(2)] Airplane BuNo. 139209.

LIQUID OXYGEN DURATION

CABIN PRESSURIZED — PILOT IN PRESSURE SUIT

HOURS REMAINING

AIRCRAFT ALTITUDE, ACTUAL, FEET	COCKPIT PRESSURE ALTITUDE, INDICATED, FEET	GAGE READING (LITERS)				
		"5"	"4"	"3"	"2"	"1"
60,000	22,500	5.3	4.3	3.2	2.1	1.1
55,000	21,500	5.2	4.2	3.1	2.1	1.0
50,000	20,500	5.0	4.0	3.0	2.0	1.0
45,000	18,750	4.7	3.7	2.8	2.8	0.9
40,000	16,750	4.3	3.4	2.6	2.6	0.9
35,000	14,500	4.0	3.2	2.4	2.4	0.8
30,000	12,000	3.6	2.9	2.2	2.2	0.7
25,000	9000	3.2	2.5	1.9	1.9	0.6
20,000	6000	2.9	2.3	1.7	1.7	0.6
15,000	5000	2.8	2.2	1.7	1.7	0.5
10,000	5000	2.8	2.2	1.7	1.7	0.5
5000	5000	2.3	1.8	1.4	1.4	0.5
SEA LEVEL	SEA LEVEL	1.9	1.5	1.2	1.2	0.4

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REMARKS:

(1) Data assumes the use of a properly fitted face mask, oxygen system pressurized at 70 psi and delivering 100% oxygen when in use. Many variables affect these calculations.

(2) Figures given for altitudes of 10,000 feet and below also allow for shorter oxygen duration at very high aircraft speeds; medium and low speeds at these altitudes will increase the oxygen duration slightly.

(3) Altitude readings must be read from the proper indicator.

(4) For flight condition of CABIN PRESSURIZED--PILOT IN UNPRESSURIZED (integrated) FLIGHT SUIT, oxygen duration will be 5% to 6% greater above 30,000 feet true altitude and 1% to 2% greater below 30,000 feet true altitude.

DATA AS OF: 15 January 1957.

DATA BASIS: Calculations.

Figure 4-2. Oxygen Duration Chart (Sheet 1)

LIQUID OXYGEN DURATION

CABIN UNPRESSURIZED — PILOT IN PRESSURE SUIT

HOURS REMAINING

AIRCRAFT ALTITUDE, ACTUAL, FEET	PRESSURE SUIT ALTITUDE, INDICATED, FEET	GAGE READING (LITERS)				
		"5"	"4"	"3"	"2"	"1"
60,000	35,000	10.0	8.0	6.0	4.0	2.0
55,000	35,000	10.0	8.0	6.0	4.0	2.0
50,000	35,000	10.0	8.0	6.0	4.0	2.0
45,000	35,000	10.0	8.0	6.0	4.0	2.0
40,000	35,000	10.0	8.0	6.0	4.0	2.0
35,000	35,000	8.8	7.0	5.3	3.5	1.7
30,000	30,000	7.1	5.7	4.4	2.8	1.4
25,000	25,000	5.8	4.6	3.5	2.3	1.1
20,000	20,000	4.8	3.8	2.9	1.9	0.9
15,000	15,000	4.0	3.1	2.3	1.6	0.8
10,000	10,000	3.3	2.6	2.0	1.3	0.6
5000	5000	2.7	2.2	1.6	1.1	0.5
SEA LEVEL	SEA LEVEL	2.3	1.8	1.3	0.9	0.4

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REMARKS:

(1) Data assumes the use of a properly fitted face mask, oxygen system pressurized at 70 psi and delivering 100% oxygen when in use. Many variables affect these calculations.

(2) Altitude readings must be read from the proper indicator.

(3) Flight condition of "CABIN UNPRESSURIZED--PILOT IN PRESSURE SUIT" (shown above) is not the ordinary condition. Normally, both cabin and pressure suit will be pressurized (see figure 4-3, sheet 1). Data shown above is for reference in the event of failure of cabin pressurization.

DATA AS OF: 15 January 1957.

DATA BASIS: Calculations.

Figure 4-2. Oxygen Duration Chart (Sheet 2)

EMERGENCY OXYGEN BOTTLE

An emergency gaseous oxygen supply is provided in a cylindrical "U" shaped emergency oxygen bottle installed in the pilot's seat cushion. The emergency oxygen bottle is charged to 1800 psi and has a capacity of 70 cubic inches. This oxygen supply may be checked by noting the reading of the pressure gage (15, figure 1-6) which is visible under the forward right-hand corner of the seat cushion. Oxygen is supplied from the emergency oxygen bottle for a period of approximately ten minutes, depending on the altitude when the green ball (14, figure 1-6) hanging from the left front side of the cushion is pulled. The ball, attached to a cable, releases a plunger on the pressure reducer of the emergency oxygen bottle and permits oxygen to flow at 70 ± 5 psi through a coupling assembly and pilot's supply tube to the oxygen regulator for delivery to the face mask. A check valve in the supply tube, which plugs into the connection on the left side of the pilot's seat, prevents loss of oxygen when the emergency oxygen bottle is actuated. During ejection, the emergency oxygen system is actuated automatically by a lanyard, anchored to the floor of the aircraft, when the ejection seat is catapulted.

NORMAL OPERATION

BEFORE FLIGHT. Before each flight requiring oxygen, the oxygen system and mask should be checked as follows:

a. Check general condition of fittings, hose, regulator and mask.

b. Inspect the exhalation valve for foreign matter (sand, grit, lint, etc.).

c. Inspect the inhalation valves to see that the valve body is properly seated in the mask.

d. Inspect the inhalation valves for foreign matter and proper mounting of plastic covers. The arrow scribed in the cover must point down.

e. Insert the end of the oxygen hose into the mouth, suck and seal the hose with the tongue. The inhalation valves should stay firmly seated for four to five seconds.

f. Place the mask on the face without connecting the oxygen hose to the aircraft's oxygen system and exhale. Exhalation should be possible without difficulty or resistance if the exhalation valve is unseating properly.

g. Under the same conditions, inhale. Inhalation should be difficult and will confirm that the exhalation valve is seating properly. Some air will enter the mask through the inhalation valves via the oxygen regulator and the resticتor in the supply tube.

h. With the mask in place, connect the mask oxygen supply tube to the supply tube on the seat cushion coupling assembly on the left side of the pilot's seat and turn the OXYGEN switch "ON". Inhalation should be very easy to accomplish if the regulator is operative and delivering oxygen at a slight positive pressure. Exhalation should also be possible without difficulty. If exhalation is difficult, there is inhalation valve leakage.

i. Check oxygen supply and security of both the hose couplings and radio connections.

DURING FLIGHT. The following should be checked frequently while on oxygen during flight:

a. Oxygen supply.

b. Oxygen mask for secure fit.

c. Security of oxygen disconnect couplings.

CAUTION

Loss of radio communication may indicate separation of the oxygen tube couplings. Check these connections for security before making any other check of communications equipment.

AFTER FLIGHT. Following each flight during which oxygen has been used, check:

a. OXYGEN switch . . . . "OFF"

b. Disconnect the mask-to-seat-cushion oxygen connection and ascertain that the dust cover on the supply hose snaps into place.

c. Report any oxygen system discrepancies and see that they are corrected.

EMERGENCY OXYGEN OPERATION

In event of oxygen system failure or depletion of supply, activate the emergency oxygen system by pulling the green ball ("green apple") on the left front of the seat, and descend to lower altitude.

NAVIGATION EQUIPMENT

MA-1 COMPASS SYSTEM

The MA-1 compass-controlled direction-gyro system provides a visual indication of the magnetic heading of the airplane. The compass system is comprised of a compass transmitter (flux valve), directional gyro, amplifier, controller, and a repeater indicator. In addition to its normal function as a gyro-stablizer magnetic compass, the system is used as a free directional gyro indicator. When operated as a free gyro, the MA-1 system utilizes a unique leveling provision for cancelling out drift error due to the aircraft's position in latitude and the effects of the rotation of the earth. Due to this feature, the total precession rate is less than four degrees per hour. The compass system requires both a-c and d-c electrical current and is powered by the a-c and d-c primary busses. Directional indications of the MA-1 compass system in either the compass-controlled or free-gyro modes are displayed on the ID-250/ARN course indicator (21, figure 1-4).

COMPASS CONTROLLER

The MA-1 compass system controller (12, figure 1-5) on the right-hand console is labeled COMPASS and consists of a heading-set-knob, mode of operation switch, synchronization indicator, latitude-compensation control, and power failure warning flag.

MODE-OF-OPERATION SWITCH. The mode-of-operation switch is used to select either slaved or free gyro operation. The switch is labeled "FREE N. LAT," "SLAVED," and "FREE S. LAT." To obtain compass-controlled directional-gyro indications the switch is placed at "SLAVED." To obtain free-gyro operation the switch is placed at "FREE N. LAT" or "FREE S. LAT" as required, depending on whether the airplane is flying north or south of the equator.

HEADING-SET KNOB. The heading-set knob, labeled PULL TO SET, is used to precess the directional-gyro electrically to obtain a specific heading indication on the compass card of the ID-250/ARN indicator when using the free gyro mode, or to precess the gyro to align it with the magnetic flux vlave as indicated by the synchronization indicator when using the compass-controlled gyro mode.

SYNCHRONIZATION INDICATOR. The synchronization indicator is located at the top center of the controller and incorporates a white pointer that is visible through the window in the controller. The indicator is used to synchronize the magnetic flux valve and directional gyro when operating with the mode-of operation switch placed at "SLAVED." To synchronize the system, the PULL TO SET knob must be rotated until the white line of the indicator is aligned with the white arrow on the face of the control panel. The pointer should approach the arrow from the right when the knob is turned counterclockwise or from the left when the knob is turned clockwise before the system and be considered to be in proper synchronization.

Note

> When the system is set for free gyro operation the synchronization indicator is inoperative and the pointer will remain at or near the white arrow.

After synchronization the pointer may wander back and forth but will not remain off to one side for any appreciable length of time. This condition is normal and indicates that the flux valve is oscillating due to aircraft acceleration, turn error, etc. Incorporated into the synchronization indicator is a power warning flag which is visible whenever a-c or d-c electrical power is not available, indicating that the compass is unreliable or inoperative.

LATITUDE-COMPENSATION CONTROL. The latitude-compensation control, labeled SET TO LAT, is graduated clockwise from "0" through "90". This control compensates for the apparent drift of the gyro at various latitudes, due to the rotation of the earth, by electrically applying torque to the gyro in a quantity sufficient to precess the gyro in the same amount but opposite to the direction of the drift. The control will compensate for all latitudes from 0° to 90° north or south.

Note

> When the system is set for compass-controlled operation, the latitude-compensation control is inoperative.

OPERATION OF THE MA-1 COMPASS

SLAVED GYRO MODE. To establish slaved gyro operation, perform the following:

a. Mode-of-operation switch . . . . "SLAVED"

b. PULL TO SET knob . . . . pull and rotate as necessary to align synchronization indicator.

FREE GYRO MODE. To establish free gyro operation, perform the following:

a. Mode-of-operation switch . . . . "FREE N. LAT" or "FREE S. LAT" as necessary.

b. SET TO LAT control . . . . rotate to existing latitude.

c. PULL TO SET knob . . . . pull and rotate as necessary to obtain desired heading on ID-250/ARN.

Note

During carrier based operations it is recommended that the free gyro mode be used for take-off because of shipboard magnetic distrubances.

STAND-BY COMPASS

A standard magnetic compass (see figure 1-4) is mounted on the forward edge of the canopy. The compass is visible to the pilot when the canopy is closed and is used for reference to set the ID-250/ARN course indicator when operating on free gyro course indications or in the event of failure of the MA-1 compass system.

ARMAMENT EQUIPMENT

(Data not available.)

MISCELLANEOUS EQUIPMENT

ANTI-BLACKOUT SYSTEM

The anti-blackout system operates on engine compressor bleed air. This air is directed through a pipe to the ANTI-BLACKOUT valve (26, figure 1-3) on the left-hand console. The valve meters the air to the pilot's anti-G suit when a force of approximately 1.75 G's is applied to the aircraft. A "HI" and "LO" control allows for adjustment of rate of inflation of the anti-G suit. In the "LO" range, the valve opens at 1.75 G and allows 1 psi of air pressure to pass to the suit for every increase of 1 G force thereafter. In the "HI" range, the valve also opens at 1.75 G but delivers 1.5 psi per G force thereafter. A button is provided on top of the anti-G valve for manually inflating the anti-G suit on the ground with the engine running, or in straight and level flight. Prior to each flight, with engine running and the anti-G suit connected, depress this button manually several times to check the operation of the anti-blackout system. If the valve has any tendency to stick or fails to return to the closed position, it should be replaced. On long flights, this feature makes it possible for the pilot to occasionally inflate the suit for body massage to lessen fatigue. The pilot's anti-G suit connection plugs into a receptacle (27, figure 1-3) adjacent to the control valve.

SPIN AND DRAG CHUTES

SPIN CHUTE. The spin chute installation, designed as an aid for pilot familiarization of spin recovery technique as well as an emergency device where normal spin procedure is ineffective, consists of the spin chute assembly, the spin chute control panel and operating and releasing control mechanisms. The spin chute assembly consists of a 24 foot ribbon type parachute, pilot chute and ejector spring, bridle line, tow line, deployment bag and "bathtub" pan. The pan assembly contains spring loaded latching linkage. The cockpit controls are found on the left hand console (24, figure 1-3) labeled SPIN AND DRAG CHUTE CONTROLS. The OPEN handle is for deployment of the spin chute.(1) In place of the OPEN handle, one aircraft(2) utilizes the ARREST-HOOK control (1, figure 1-5) to deploy the spin chute when the handle is placed in the "DOWN" position. The RELEASE handle is for manual jettisoning of the spin chute after use. An interlocking device prevents operating the RELEASE handle prior to opening the spin chute. There are two guarded SPIN CHUTE DETONATOR SWITCHES, the left one labeled READY-OFF arms the explosive actuated release mechanism for emergency use in event of manual RELEASE handle fails to jettison the chute. The right switch, labeled BLOW-OFF-BLOW, fires the explosive release when moved either fore or aft out of the "OFF" position. A shorting fuse prevents inadvertent firing by static electricity.

DRAG CHUTE. The drag chute is operated in the same manner when there is a need for high speed deceleration on short runways. It is a 10 foot ribbon type chute. It can only be used as an alternate to the spin chute: i.e., both are not carried at the same time.

NORMAL OPERATION OF SPIN CHUTE. Should an unintentional spin be entered and normal spin recovery procedures are not effective, the following procedure is suggested:

a. Reduce throttle to "IDLE". This is mandatory to prevent burning of the chute.

b. Neutralize controls.

c. Deploy the spin chute.

(1) Airplanes BuNo. 139208, 142349-142350.
(2) Airplane BuNo. 139209

d. After spin recovery, maintain a safe airspeed while the chute is attached to prevent another stall. With the chute attached, after the spin gyration has ceased at least 180 knots must be maintained by staying in about a 60-degree dive angle until the chute is jettisoned.

e. Pull up sharply on the "RELEASE" handle to manually jettison the chute.

f. Recover to straight and level flight.

EMERGENCY OPERATION OF THE SPIN CHUTE. In the event that the manual jettisoning method does not effect release of the chute, or if the spin chute is used in an accidental spin and quick release is desired:

a. Utilize the two guarded switches on the spin chute panel. Throw the left switch, labeled READY-OFF, forward, out of the "OFF" position, to "READY". This arms the electrically actuated explosive release.

b. Move the right guarded switch, labeled BLOW-OFF-BLOW, out of the "OFF" position, either fore or aft, to "BLOW", which fires the explosive actuated release to jettison the chute. If one position labeled "BLOW" is inoperative, throw the switch to the other "BLOW" position.

c. In the event this procedure is not effective, use the AFTERBURNER to burn the chute off. If this procedure fails, eject.

WARNING

- "IDLE" power setting must be used to preclude jet exhaust burning the chute or tow line.
- Deceleration is quite rapid with the spin chute open. Maintain approximately a 60-degree dive to maintain 180 knots with "IDLE" thrust with chute attached prior to jettisoning the spin chute.
- Do not attempt to land with the spin chute attached.
- Spin chutes should be packed by authorized parachute loft personnel. Prior to installation, all hands should be certain that the spin chute is not damaged, tampered with or placed in a reversed position.

- Snow or moisture in parachutes during cold weather operation or high altitude operation in warm weather may result in parachute failure, due to freezing of the pilot parachute or canopy.

SPIN RECOVERY INSTRUMENT

One aircraft[(1)] is equipped with a spin recovery instrument, located above the glareshield and labeled "SPIN RECOVERY DIRECTOR". It incorporates a vertical needle indicating the direction of the spin. The instrument also indicates fore-and-aft stick position required for spin recovery by means of a "flip-flop" indicator calling for forward, aft or neutral stick position. An "OFF" flag is provided to indicate failure of the instrument or power supply.

MASTER WARNING ANNUNCIATOR

The aircraft is equipped with a master warning annunciator system consisting of a red MASTER WARNING light, TEST switch, RESET switch, and 'panic panel'. The MASTER WARNING light (1, figure 1-4) on the instrument panel illuminates whenever a monitored aircraft system or component fail. Simultaneously, one of nine individual lights identifying the failed system or component will illuminate in the 'panic panel' (5, figure 1-5) on the right-hand console. The items displayed on the 'panic panel' are as follows:

a. D.C. POWER
b. A.C. GEN
c. FUEL PUMP (engine)
d. FUEL BOOST PUMP
e. EMERG FUEL (control)
f. OIL PRESSURE
g. ELEVON PRESSURE
h. UTILITY PRESSURE
i. FUEL QUANTITY

Detailed descriptions of conditions causing the foregoing lights to illuminate are set forth in applicable portions of Section I.

If the condition resulting in illumination of the MASTER WARNING light and applicable 'panic panel' light is corrected, both lights will be extinguished automatically. If the condition causing illumination of the warning lights is not, or cannot, be corrected, the pilot may push the "RESET" button (1A, figure 1-4) adjacent

(1)Airplane BuNo. 139209

to the MASTER WARNING light. This action will extinguish the MASTER WARNING light, leaving the individual 'panic panel' light illuminated, and re-establish the function of the MASTER WARNING light in the event of subsequent failure of an additional system or component. Pressing the "TEST" switch (2, figure 1-4) will cause the MASTER WARNING light and all 'panic panel' lights to illuminate simultaneously, thereby testing the continuity of the circuits and condition of the bulbs.

REAR VIEW MIRRORS

Two rear view mirrors (see figure 1-4) are provided, one on each side of the canopy bow.

SPARE LAMPS AND FUSES

Spare lamps for the instruments (8, figure 1-5) and consoles (7, figure 1-5) are provided on the SPARE LAMPS panel on the right-hand console, SPARE FUSES (6, figure 1-5) are also provided on the right-hand console.

Spare fuses for yaw damper, ignition and fuel flow are located on the left-hand console (29, figure 1-3).

WIRE RECORDER

A wire recorder is installed in the aircraft to record pilot voice transmissions through the oxygen mask microphone. The recorder is controlled by the EXT LIGHTS switch (7, figure 1-3) on the outboard side of the throttle lever. Aft movement of the switch provides momentary recording for as long as the switch is held. Forward movement of the switch provides continuous wire recording.

SECTION V

OPERATING LIMITATIONS

INTRODUCTION

This section lists limitations that must be observed during normal operation of the airplane. Cognizance shall be taken of the instrument markings illustrated in figure 5-1, since limitations portrayed by the illustration are not necessarily repeated in the test. The operating limitations set forth herein may be raised from time to time as flight and static testing dictate. Current Bureau of Aeronautics correspondence and/or pertinent Contractor Flight Test Data should be consulted prior to flight.

ENGINE LIMITATIONS

The engine and its supporting structure will mechanically withstand any flight maneuver or landing force within the design load factor limitations of the aircraft structure.

ENGINE OPERATING LIMITATIONS

The engine operating limitations are based on combinations of engine speeds and exhaust gas temperatures. The maximum allowable engine speed under any condition is "red-lined" at 102.2% (10,200) rpm. Should this limiting value be exceeded, reduce thrust and land immediately. Adhere strictly to maximum allowable exhaust gas temperatures to prolong the life of the engine. Table II lists maximum exhaust gas temperatures and time limits under various conditions.

The engine operating limitations shown in Table II are based on the use of MIL-F-5624A, grade JP-4 fuel. The use of alternate fuel MIL-F-5624, grade JP-3, is acceptable as an emergency procedure only. When using alternate fuel, it is necessary to exercise extra caution not to overtemperature the engine, and after such use, a "hot section" inspection must be conducted in accordance with existing practices.

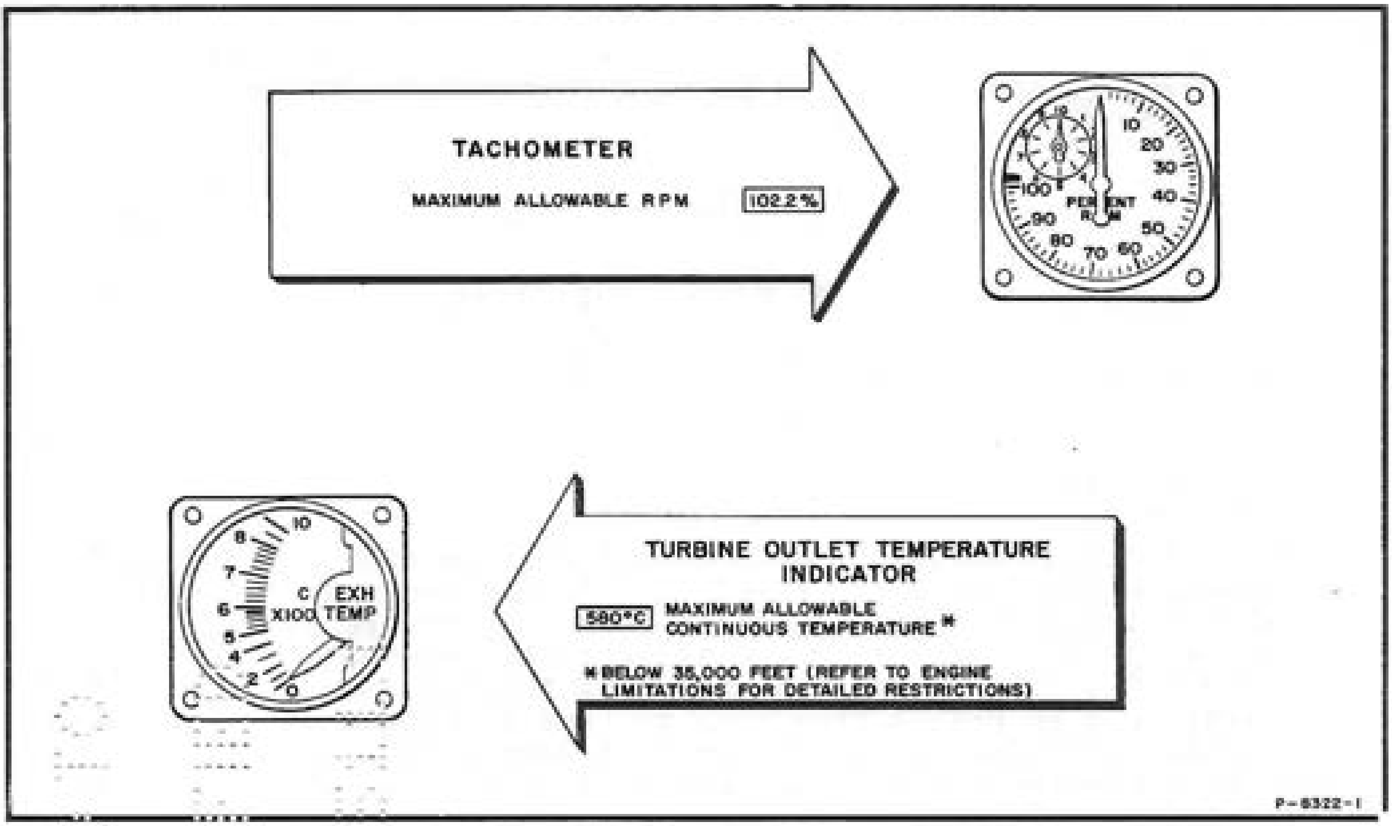


Figure 5-1. Instrument Markings

TABLE II

OPERATING CONDITIONS	MAX. EXHAUST GAS TEMP. BELOW 35,000	MAX. EXHAUST GAS TEMP. ABOVE 35,000	% OF MAX RPM	TIME LIMIT (MIN.)
"TAKE-OFF" (WITH AFTERBURNER)	630	660	#	5 (TAKE-OFF) 15 (FLIGHT)
MILITARY	620	650	*	30
NORMAL RATED	580	610	*	CONTINUOUS
CRUISE - 90% N.R.	540	570	*	"
CRUISE - 80% N.R.	500	530	*	"
CRUISE - 70% N.R.	470	500	*	"
IDLE	340	---	55-65	"
STARTING	630	630	---	---
ACCELERATION	670	670	*	2

Individual engine trim speed.
* These values vary with inlet temperature.

ENGINE GROUND OPERATION. For sustained engine operation on the ground the forward engine access door must be open. Under this condition, engine operation is limited to cycles of 5 minutes at military power plus 30 seconds of afterburning followed by a minimum of 5 minutes at idle power. Ground operation with the forward engine access door closed is limited to 3 minutes of military power plus 30 seconds of afterburning. Prior to repeat runs it is necessary to shut down the engine for a 30 minute cooling period or to open the forward engine access door. Idle operation is unlimited with the forward engine access door open or closed.

AIRSPEED LIMITATIONS

The maximum permissible indicated airspeeds during various operating conditions are as follows:

a. In smooth or moderately turbulent air, landing gear retracted, speed brakes open or closed . . . . as shown in figure 5-2

b. With landing gear extended . . . . 260 knots

c. With canopy open (taxi) . . . . . . 88 knots IAS

d. Emergency generator and hydraulic pump,

   Extension . . . . 350 knots IAS
   Extended . . . . 470 knots IAS

e. With forward rocket door open, aft rocket door closed:

   Sea level. . . . . Mach 0.85
   Above 20,000 feet . . . . Mach 1.15

f. Minimum airspeed with control system powered by wind-driven hydraulic pump:

   Recommended for landing. . . . . 140 knots IAS
   Permissible in flight. . . . . . 125 knots IAS

MANEUVERS

The following maneuvers are permitted:

a. Wing-over

b. Chandelle

c. Immelman

d. Loop

e. Inverted flight not to exceed 10 seconds.

f. Aileron rolls except as follows:

   1. A roll angle of 360 degrees and one-half lateral control displacement shall not be exceeded below 10,000 feet.

   2. A roll angle of 120 degrees and one-half lateral control displacement shall not be exceeded at altitudes above 10,000 feet.

   3. A roll angle of 120 degrees and one-fifth lateral control displacement shall not be exceeded above an indicated Mach number of 0.90.

g. Full lateral control displacement is permitted during take-offs and landing approaches.

Maneuvers not complying with the above restrictions may be performed by flight test personnel as authorized by the Bureau of Aeronautics, except that "zero" or "negative g" flight should be limited to 10 seconds at a fuel flow not to exceed 8400 pounds per hour under all circumstances, unless special equipment and instrumentation have been incorporated for the performance and measurement of certain tests. Rolling pull-outs and push-downs are prohibited.

ACCELERATION LIMITATIONS

Except that accelerations at which light to moderate buffeting is encounted shall not be exceeded, the maximum permissible acceleration for flight at gross weights of 22,500 pounds or less are as shown on figure 5-2 (sheet 1). As gross weights are increased above 22,500 pounds, the permissible acceleration decreases. To determine the maximum permissible acceleration at gross weights in excess of 22,500 pounds, multiply the acceleration shown in figure 5-2 (sheet 1), by the ratio of 22,500 pounds to the new gross weight.

RUDDER

The maximum permissible rudder deflection varies linearly from full at 130 knots IAS to that obtained with 150 pounds rudder pedal force at 450 knots IAS or at 0.81 IMN, whichever is less. Above these airspeeds intentional sideslips are not permitted.

WEIGHT LIMITATIONS

Current recommended maximum gross weights are as follows:

Field take-off	27,700 pounds
Field landing	26,250 pounds

CENTER OF GRAVITY LIMITATIONS

The forward center of gravity shall not be less than 21% Mean Aerodynamic Chord (MAC). The aft center of gravity shall not be more than 24.3% MAC with the gear up in level attitude, nor more than 27% MAC in the landing configuration. See AFTER CLIMB, Section II, for instructions concerning auxiliary transfer fuel center of gravity management.

SINKING VELOCITY LIMITATION

The aircraft sinking velocity during landing shall not exceed 13 feet per second. 780 fpm

STORES

The carrying of stores is not permitted except that Sparrow II missiles may be carried under the same restrictions on flight that apply without such stores, except as follows:

a. Spanwise load distribution shall be approximately symmetrical.

b. The maximum permissible indicated airspeed is 500 knots or 1.25 IMN, whichever is less, except that 1.40 IMN may be flown in 1.0g flight between the altitudes of 30,000 to 35,000 feet.

c. Except that accelerations at which buffeting is encountered shall not be exceeded, the maximum permissible acceleration for any gross weight is 3.0g.

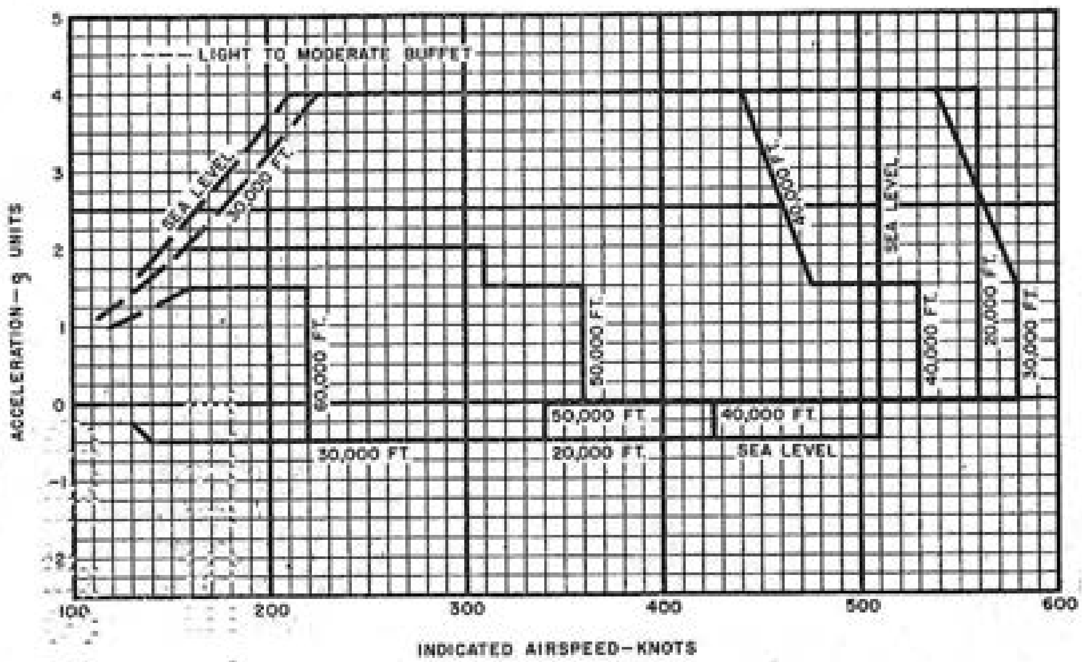
STABILITY AUGMENTATION LIMITATION

The minimum permissible altitude with stability augmentation (AUTOMATIC SYSTEMS) engaged is 10,000 feet.

OPERATING FLIGHT STRENGTH DIAGRAM

CLEAN CONFIGURATION

GROSS WEIGHT — 22,500 POUNDS



REMARKS:
(1) INDICATED AIRSPEEDS ARE APPROXIMATE
(2) OPERATING FLIGHT STRENGTH DIAGRAM BASED ON LIMITATIONS IMPOSED BY BAR EL SEGUNDO SPEEDLETTER, SERIAL 02C??, 20 MARCH 1957

DATA AS OF: MARCH 1957

DATA BASIS: CONTRACTOR FLIGHT TEST DEMONSTRATION

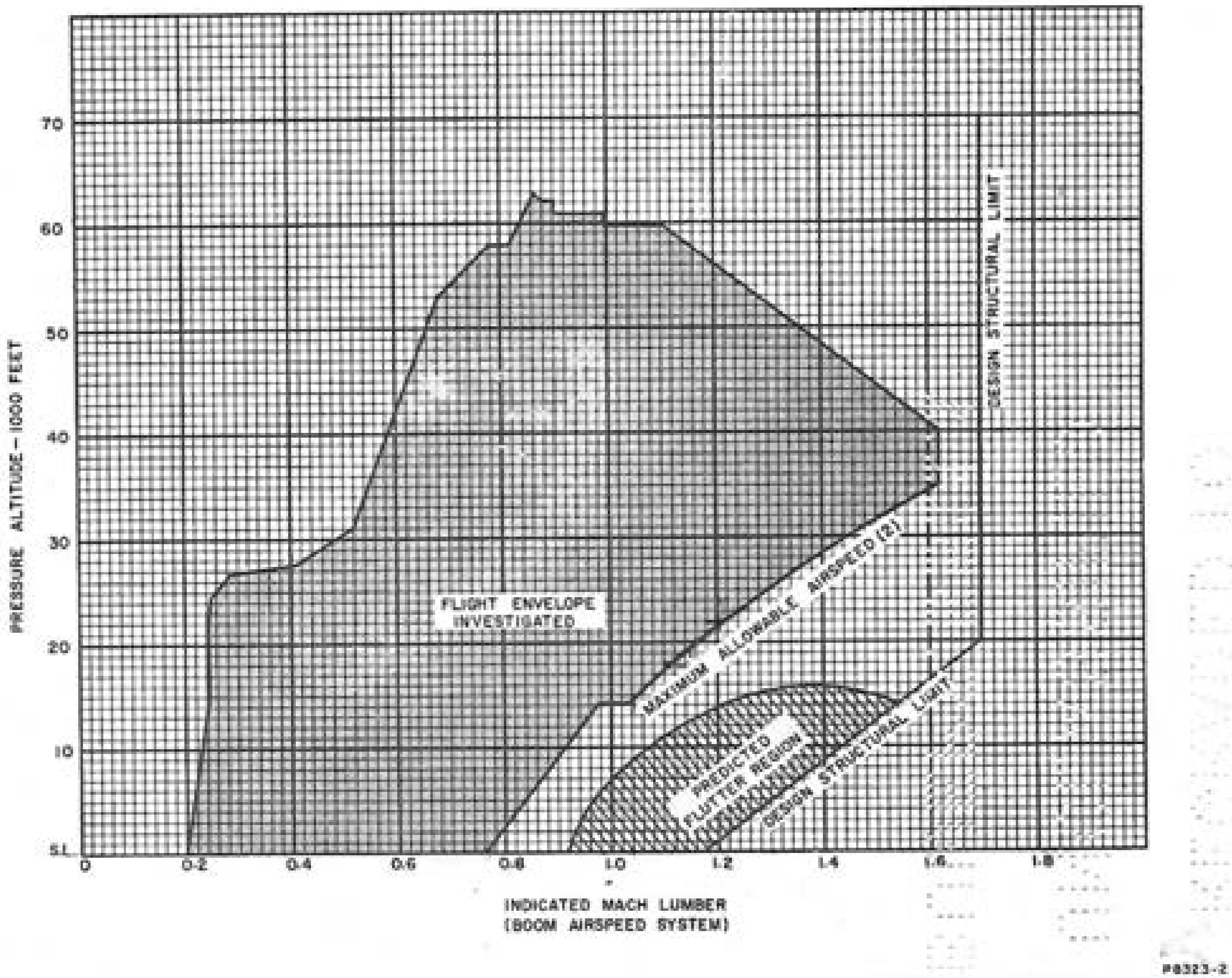
Figure 5-2. Operating Flight Strength Diagram (Sheet 1)

ARRESTED LANDING

Arrested landing operations are not permitted.

OPERATING FLIGHT STRENGTH DIAGRAM

CLEAN CONFIGURATION
GROSS WEIGHT – 22,500 POUNDS
ONE "G" ACCELERATION



REMARKS:

(1) INDICATED MACH NUMBERS ARE APPROXIMATE.

(2) THE "MAXIMUM ALLOWABLE AIRSPEED" LINE SHOWN ABOVE REFLECTS THE AIRSPEED LIMITATIONS IMPOSED BY BAR EL SEGUNDO SPEEDLETTER, SERIAL 03082, 20 MARCH 1957.

(3) THE "PREDICTED FLUTTER REGION" SHOWN ABOVE IS BASED ON STATIC TESTS AND CALCULATIONS.

DATA AS OF: MARCH 1957

DATA BASIS: CONTRACTOR FLIGHT TEST DEMONSTRATION

Figure 5-2. Operating Flight Strength Diagram (Sheet 2)

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SECTION VI

FLIGHT CHARACTERISTICS

GENERAL FLIGHT CHARACTERISTICS

Despite its unconventional configuration the aircraft possesses normal flight characteristics during most flight conditions and attitudes. Due to extensive differences in airflow conditions between subsonic and supersonic speeds, certain flight characteristics undergo changes as sonic speeds are approached; however the aircraft is designed to possess positive stability at both subsonic and supersonic speeds. While control effectiveness will be considerably reduced at sonic and supersonic speeds, the airplane may be expected to display excellent maneuverability throughout the entire speed range. Directional stability will deteriorate at higher Mach number, but is considered to be relatively good throughout the flight regime.

STALLS

Characteristics of the complete stall are abnormal, in that a spin may be entered inadvertently. The approach to the stall is characterized by adequate aerodynamic stall warning in the form of airframe buffeting. This buffet starts about 40 knots above the complete stall and builds up in intensity from very light at onset to moderate at the stall. While in the stall warning region, a yawing tendency and/or wing heaviness will develop at a point approximately 15 knots below the stall warning onset. The speeds at which this phenomenon will occur are shown in Table III and constitute the minimum recommended airspeeds. If airspeed is decreased below recommended minimums a yaw may develop which cannot be adequately controlled with the rudder, resulting in a spin. With approach power, a slight nose-down pitch becomes apparent prior to the divergence in yaw. With idle power, no nose-down pitch is apparent. Increased thrust effects the stall characteristics slightly, increasing the buffet intensity and reducing the descent rates and stalling airspeeds (see figure 6-1). Practicing complete stalls is not recommended. Low airspeed airplane characteristics investigation to the minimum recommended airspeeds should be performed at a safe altitude, as the rate of descent in the approach to the stall will be approximately 4000 fpm, power-off. Accelerated turns produce a tendency for the airplane to roll out of the turn when in the stall warning region. No further increase in angle of attack is recommended after encountering a tendency for the airplane to roll during accelerated flight. Since there is more than adequate stall warning, and since recommended minimum take-off and landing approach speeds are above the stall warning speeds, there is little likelihood of stalling during normal take-off and landing operations.

TABLE III

RECOMMENDED MINIMUM AIRSPEEDS - IAS (KNOTS)

GROSS WEIGHT (WINGS LEVEL)	MAXIMUM THRUST	APPROACH THRUST	POWER OFF
17,000	92	107	112
24,000	115	126	131
31,000	138	145	152

SPINS

Unauthorized intentional spinning of the airplane is prohibited until spin tests are completed. All possible precautions must be taken to avoid entering a spin inadvertently. Although it is possible to terminate and recover from spins by use of correct procedures, certain characteristics of the airplane in the spin make it highly desirable that spinning be avoided. The following spin characteristics may be expected:

STALLING SPEEDS

GEAR UP OR DOWN

ANGLE OF BANK DEGREES	APPROXIMATE I.A.S. (KNOTS)								
	MAXIMUM THRUST			APPROACH THRUST			POWER OFF		
	GROSS WEIGHT - POUNDS								
	31,000	24,000	17,000	31,000	24,000	17,000	31,000	24,000	17,000
0	125	106	83	132	116	98	139	122	103
15	127	108	85	134	118	100	141	124	105
30	135	114	90	142	125	106	149	131	111
45	149	126	99	157	138	117	165	145	122

P-8324-1

DATA AS OF: 15 NOVEMBER 1955
DATA BASIS: CALCULATIONS

Figure 6-1. Stalling Speeds

a. The airplane requires a unique recovery procedure for best results. Instinctive (normal) reactions of the pilot may not achieve rapid recovery.

b. Spins are oscillatory, particularly in pitch. In some instances the airplane may show a tendency to reverse spin direction even though pro-spin rudder is maintained. The erratic nature of the spin tends to cause pilot confusion and improper recovery procedures.

c. The aircraft will show some tendency to recover from normal spins by rolling into inverted spins. When this happens, extreme pilot disorientation results. There also exists a tendency for reversal in spin direction if recovery position of the controls is maintained too long.

RECOVERY PROCEDURE

The following recovery procedures are recommended in case a spin is inadvertently entered.

Note

> To distinguish between the two separate actions of the elevons, the words "aileron" and "elevator" are used in the following text to reference the lateral and longitudinal functions of the elevon controls, respectively.

NORMAL SPIN. The rudder should immediately be applied full against the spin. The ailerons should then be moved gradually to full throw (if necessary) with the spin (right stick deflection in a right spin) while the stick is held back. Deflecting the ailerons with the spin has proved more effective than opposite rudder in terminating the spin in similiar aircraft, but abrupt aileron motion introduces some tendency to roll inverted. Holding the stick aft during initial recovery will reduce this tendency. As soon as recovery has started, the stick should be moved forward to neutral and the ailerons and rudder neutralized. The stick should not be moved forward of neutral if recovery is made in a vertical attitude. This is to prevent possible entry into an inverted spin.

Because the ailerons are the prime recovery control, extreme care must be taken to insure that they are deflected in the correct direction, i.e., in the same direction as the aircraft is spinning. It is also highly recommended that the turn-and-bank indicator be watched during the spin as a reference to determine the correct rudder deflection direction for recovery. The turn rate will be high enough to peg the indicator during the spin and, upon recovery, the needle will reverse direction rapidly. To prevent entering a spin in the opposite direction, the controls should be neutralized as soon as the needle leaves the peg.

Considerable altitude loss will occur during a spin. A one-turn spin will require about 8000 feet for recovery to level flight and a five-turn spin will require about 15,000 feet. If a reversal of direction occurs during the spin or recovery from the spin, an additional 4000 to 5000 feet may be lost.

INVERTED SPIN. In inverted spins, recovery should be effected by applying rudder against the spin with the ailerons and elevator held in neutral. Extreme caution must be exercised because the pilot may become so disoriented due to the violent pitching motion and high rate of rotation that he will have difficulty in determining the direction of spin and whether the spin is inverted or upright. For this reason, reference to the cockpit turn-and-bank indicator should be made to determine whether to apply left or right rudder. Also, care must be taken during the recovery to release the anti-spin rudder position at the proper time so as to avoid entering a spin in the opposite direction. This will occur if the controls are held in the anti-spin position too long.

FLIGHT CONTROLS

ELEVONS

The elevons provide both longitudinal and lateral control. The inboard elevons are an integral part of the primary control and are operated through a "slaving" arrangement to the main elevons. The elevons are powered by independent dual hydraulic systems, each of which is capable of providing full maneuverability except at supersonic speeds. In the event that both hydraulic power systems become inoperative, the emergency hydraulic pump will furnish sufficient hydraulic pressure and flow for normal maneuvering.

MECHANICAL ADVANTAGE CHANGER

The mechanical advantage changer automatically changes the stick-travel to elevon-deflection ratio as the Mach number and altitude change. It maintains a relatively constant stick-force to control-response throughout the flight range and limits the amount of elevon deflection at certain speeds and altitudes to prevent over-stressing the airframe during accelerated maneuvers. Refer to figure 1-12 for information concerning the programming of this control. If the automatic mechanical advantage changer fails, the manual crank may be used to position the control at any desired setting.

RUDDER(1)

The rudder is electrically and hydraulically operated by the rudder pedals. Rudder pedal forces are light and directional control can

(1)Airplanes BuNo. 139208, 142349-142350.

be maintained at all speeds. Yaw damping is provided through servo controlled actuation of the single piece rudder. Rudder deflections originating from the yaw damper system will not be sensed by the pilot through movement of the rudder pedals. Failure of the yaw damper system will result in a reduction of lateral-directional stability, but the airplane is safe and easily controllable with this system inoperative. Failure of the power rudder electrical or hydraulic components will require centering the rudder pedals and actuating the AUTO CONT release handle to obtain manual rudder control. Rudder pedal forces can be expected to be moderately high when on manual control.

RUDDER(1)

In one airplane, the rudder control system is such that rudder deflections originating from the yaw damper system will be sensed by the pilot through movement of the rudder pedals.

SLATS

Slats are provided to improve directional and lateral stability during low speed flight and the approach to the stall. The slats operate automatically and have no noticeable effect on flight characteristics at the time of opening. The slats are expected to be open at 200 knots. During turns or accelerated maneuvers, the slats will open at slightly higher speeds.

SPEED BRAKES

The hydraulically operated speed brakes may be opened at any airspeed to provide deceleration with no appreciable trim change or buffet. The speed brakes will begin to blow back at airspeeds in excess of 400 knots. Use of the speed brakes during a landing approach is only mildly effective in providing deceleration. In this condition, furthermore, either opening or closing the speed brakes may effect the longitudinal trim and alter the landing attitude of the aircraft.

CAUTION

> It is recommended that the pilot avoid use of the speed brakes during low airspeed landing approach. If speed brakes have been opened previously, do not close them until the landing is completed.

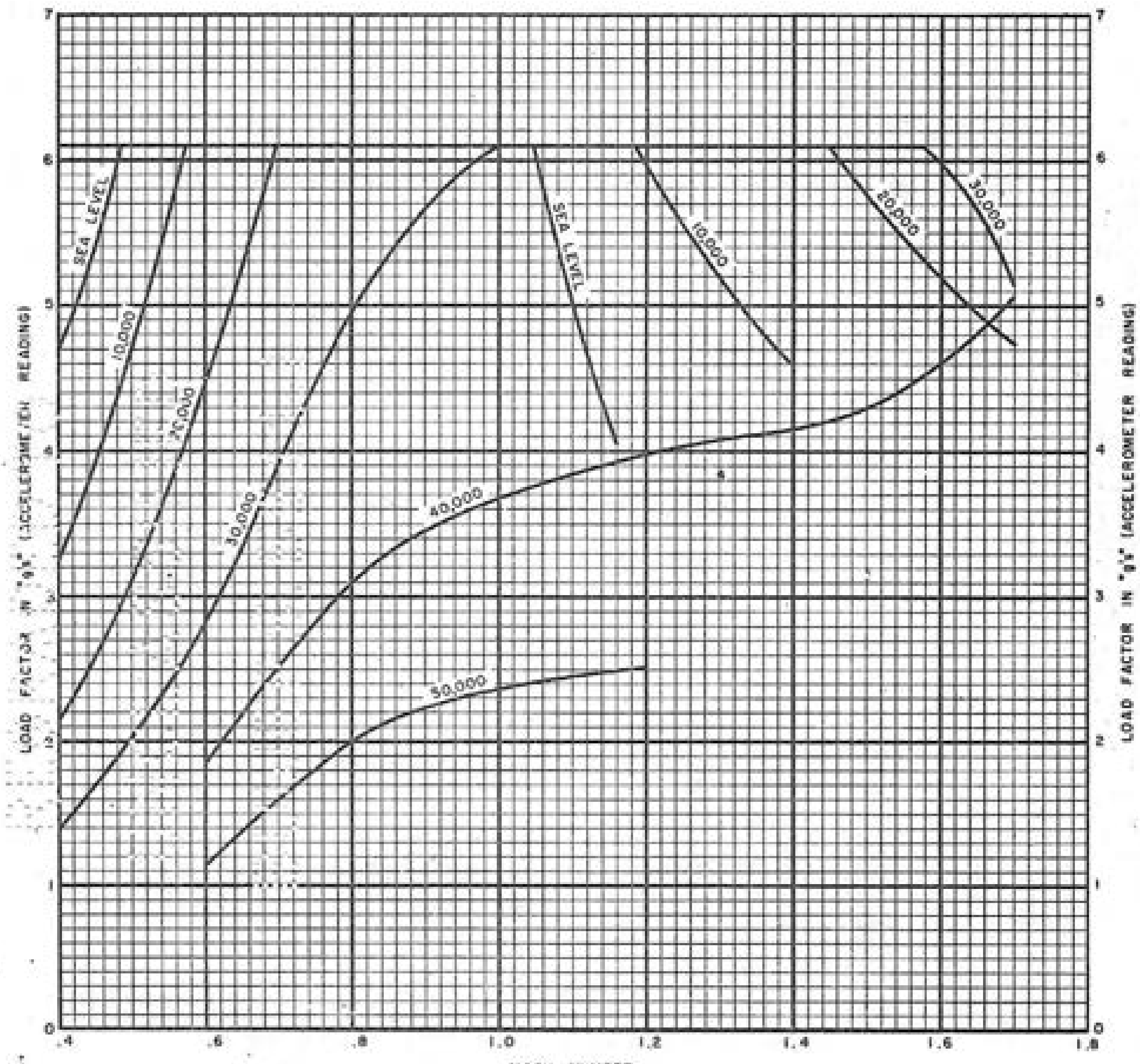
TRIM CONTROL

LONGITUDINAL TRIM. Longitudinal trim is accomplished by repositioning the neutral point or "no load" position of the control stick, which trims the elevator function of the elevons through

(1)Airplane BuNo. 139209.

MAXIMUM AVAILABLE LOAD FACTOR

BOTH ELEVON POWER SYSTEMS OPERATIVE



REMARKS:

(1) WEIGHT = 22,500 POUNDS

(2) C.G. AT 22.6% MAC

DATA AS OF: 1 DECEMBER 1955
DATA BASIS: CALCULATIONS

Figure 6-2. Maximum Available Load Factors (sheet 1)

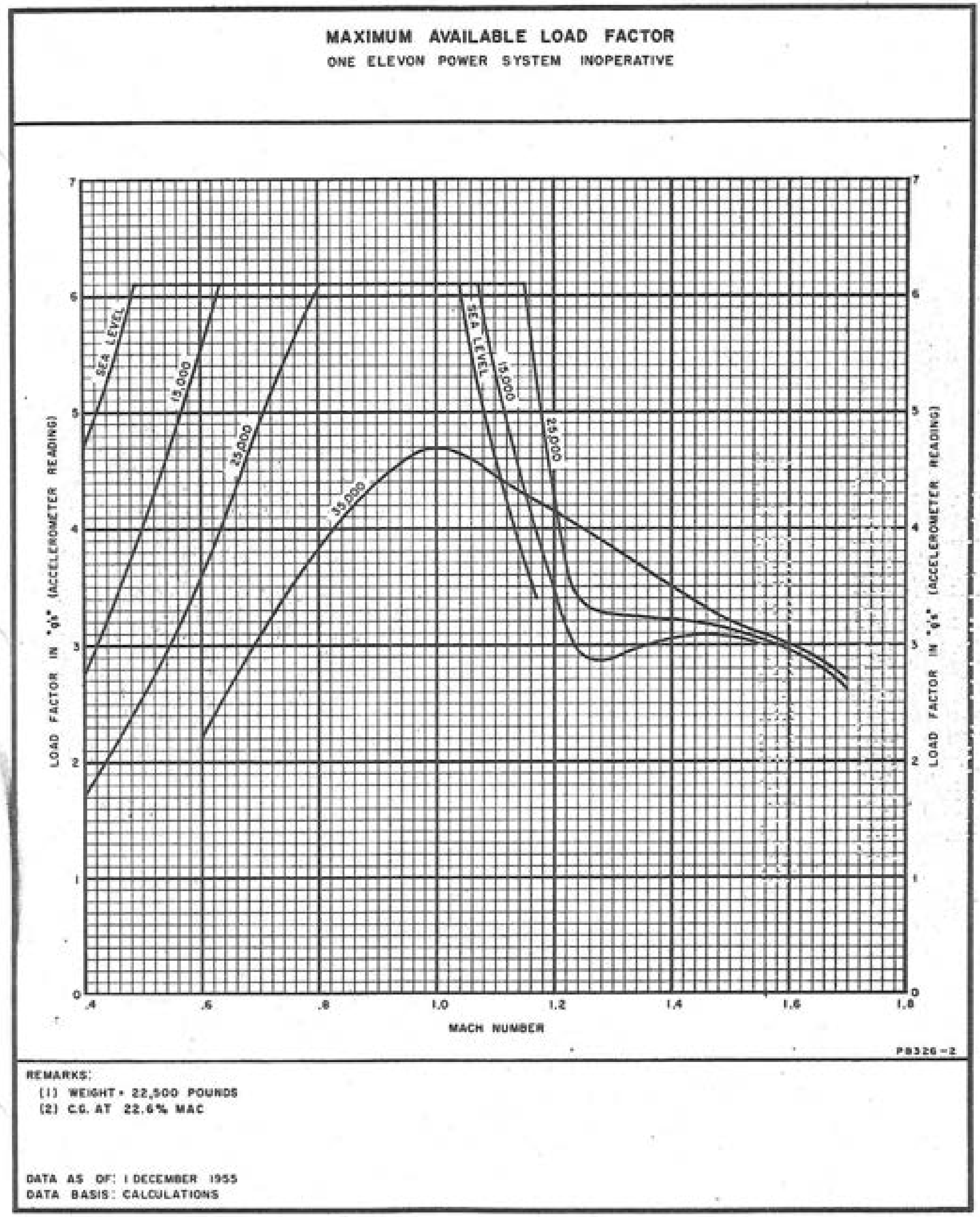


Figure 6-2. Maximum Available Load Factors (sheet 2)

the elevon control system. Longitudinal trim is not affected by hydraulic power system failure.

LATERAL TRIM. Lateral trim is accomplished by repositioning the neutral point or "no load" position of the control stick, which trims the aileron function of the elevons through the elevon control system. Lateral trim is not affected by hydraulic power system failure.

RUDDER TRIM[1]

Rudder trim is accomplished by both an electrically driven rudder trim tab and by rebalancing the "null" in the steering coupler of the yaw damper amplifier, thus displacing the entire rudder surface hydraulically. During manual rudder control, only the trailing edge trim tab is operative.

RUDDER TRIM[2]

Rudder trim is provided by a conventional electrically driven trim tab.

LEVEL FLIGHT CHARACTERISTICS

SLOW FLIGHT

The aircraft possesses positive stability and good handling characteristics in the low speed range. However, landing approach speeds much below the recommended values will result in high rates of descent and poor wave-off characteristics. Refer to LANDING, Section II, for recommended approach speeds under various conditions.

WARNING

At high angles of attack just prior to the stall, military power may be inadequate to maintain level flight. If sufficient altitude to lower the nose and gain airspeed is not available, military power with afterburning must be employed immediately.

CRUISING

Cruise characteristics are normal except for a high degree of lateral sensitivity at high altitude. The rate of roll at high altitude using full lateral control deflection may approach 550° per second. Refer to Section V for roll rate limiations.

[1]Airplanes BuNo. 139208, 142349-142350.
[2]Airplane BuNo. 139209.

HIGH SPEED FLIGHT

Sonic or supersonic speeds are readily obtainable at all operational altitudes. Considering the airplane with the trim change compensator inoperative, the airplane will encounter a "tuck-under" or trim change at about 0.94 Mach number. The pull forces required to overcome this trim change are light, however they do increase with altitude. A definite reduction in control effectiveness and an increase in maneuvering forces becomes apparent at supersonic speeds. The "tuck-under" characteristic although most pronounced from 0.95 to 1.05 Mach number, continues well into the supersonic speed range. As the airplane is decelerated through the transonic speed range the trim change will be nose up. This pitch up is controllable but is more serious for stick fixed decelerations at high load factors. The trim change compensator is scheduled to eliminate the "tuck-under" characteristic.

CAUTION

- Rapid decelerations from supersonic speeds should be avoided until the pilot becomes familiar with the transonic trim changes.

- Do not turn the TRANSONIC TRIM compensator switch to "OFF" above Mach number 0.90 without first placing the switch at "RETURN TO NEUTRAL."

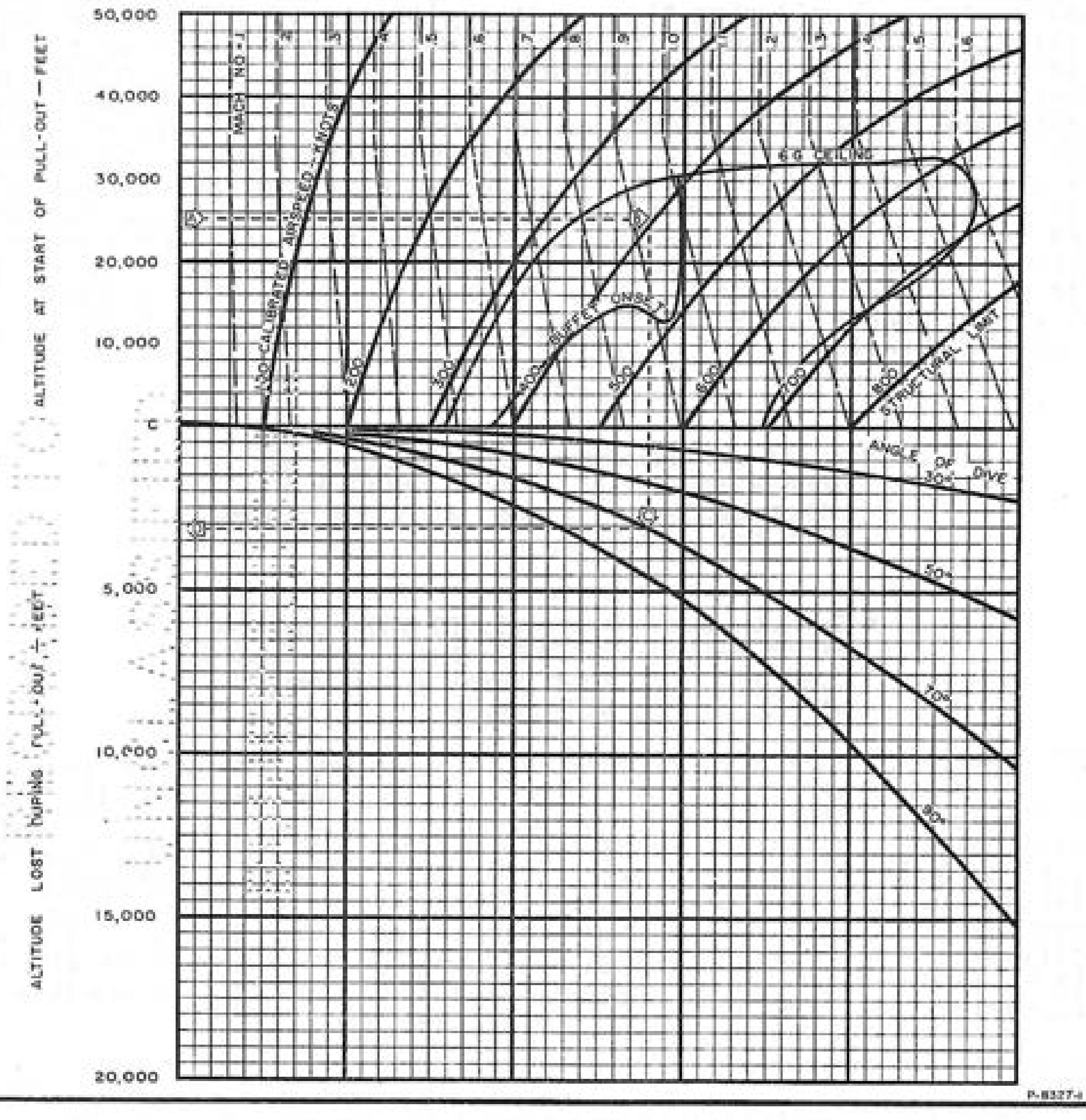
At supersonic speeds an increase in directional trim is necessary over that required at subsonic speeds. This increase in directional trim requirements results in insufficient directional trim being available at supersonic speeds at altitudes below 20,000 feet. Slight rudder pedal force may be necessary to maintain unyawed flight. At supersonic speeds below 30,000 feet, a loss of yaw damping efficiency results in weak damping of any directional oscillation. This may result in a persistent small amplitude directional oscillation when flying at supersonic speeds at low altitudes in conditions of atmospheric turbulence.

MANEUVERING FLIGHT

Maneuvering characteristics are normal except for the lateral control effectiveness existing at high altitude, where extremely high roll rates can be developed. Excellent maneuvering characteristics exist throughout the speed-altitude range. Refer to the Maximum Available Load Factor Chart, figure 6-2, sheet 1. In the event of a single hydraulic power system failure in the elevon control system, full maneuverability is retained except at supersonic speeds. Refer to the Maximum Available Load Factor Chart, figure 6-2, sheet 2.

DIVE RECOVERY CHART

CONSTANT 6G PULL-OUT



REMARKS:

(1) Constant Mach number lines are for Standard NACA Atmosphere
(2) Speed brakes open or closed
(3) Buffet onset line indicates the altitude above which buffeting will occur at a given airspeed and 6g acceleration. Pull-outs may be initiated above the buffet onset altitude without damage to the aircraft.

DATA AS OF: 1 December 1955
DATA BASIS: Estimates

EXAMPLE:

(1) Enter chart at altitude at start of pull-out (for example, 25,000 feet) (A)
(2) On scale along altitude line select the airspeed at which pull-out is started (400 knots, calibrated airspeed) (B)
(3) Sight vertically down to point on curve of dive angle (70°) directly below airspeed (C)
(4) Sight back horizontally to scale at left to read altitude lost during pull-out (D)

Figure 6-3. Dive Recovery Chart (sheet 1)

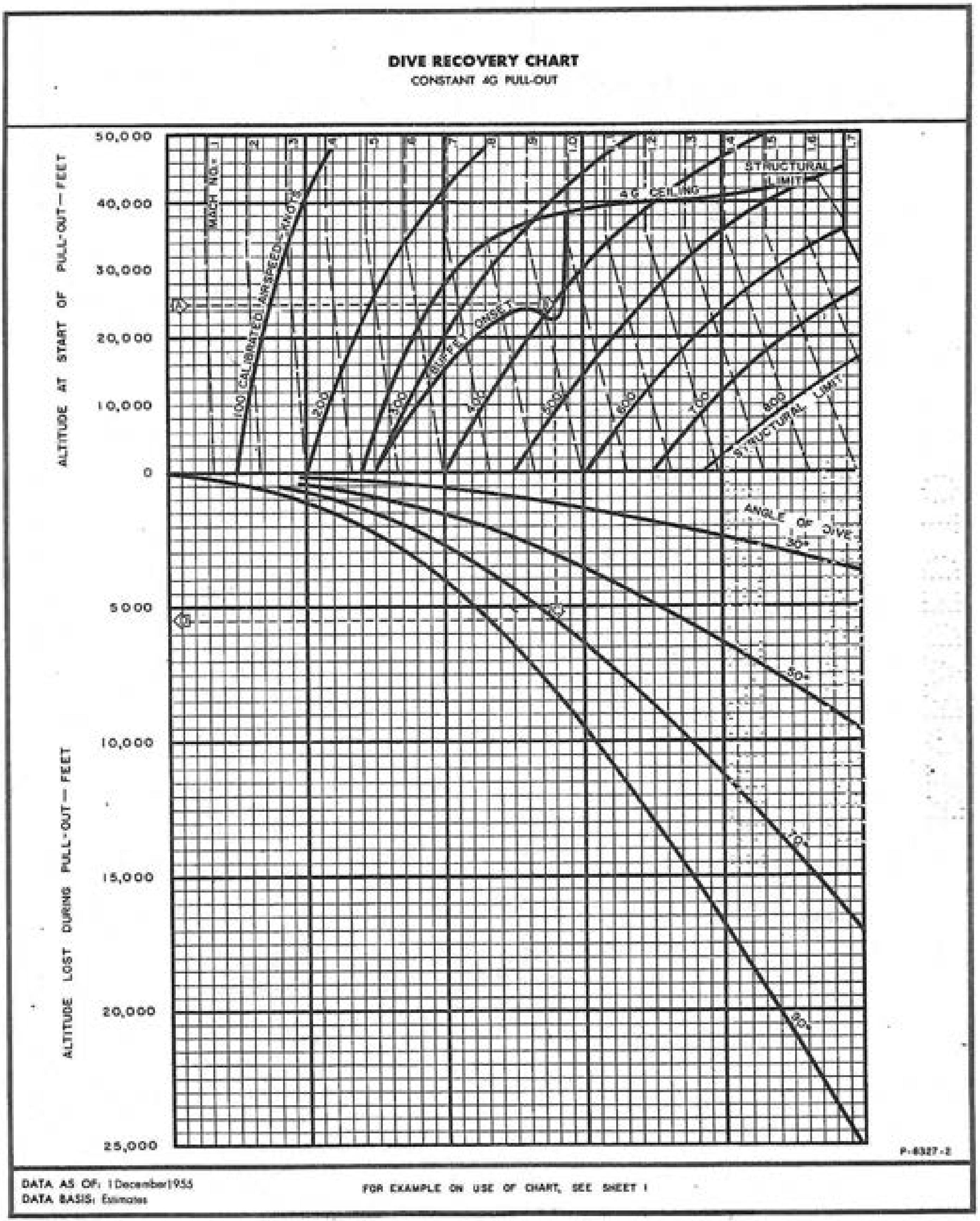


Figure 6-3. Dive Recovery Chart (sheet 2)

Because of increased drag in conjunction with increased load factor, a maximum "g" turn can not be accomplished without a considerable loss of speed and/or altitude.

WARNING

> High load factor level turns initiated at speeds above the "tuck-under" Mach number of .94 will result in rapid deceleration to subsonic speed, resulting in an abrupt "pitch-up."

Since control forces are not transmitted to the control stick the entire control "feel" is accomplished by springs which center and restrain the control stick. Full control stick deflection will always require approximately 35 pounds of force at the stick grip and full maneuverability can be obtained by the application of this force or less. Some variation in stick force required to produce a given "g" will occur as speed or altitude is varied.

ROLL CHARACTERISTICS

Subsonic roll characteristics are normal, with high rates of roll being available. In the supersonic flight regime, lateral control deflections in excess of 20 percent result in objectionable yawing (lateral) accelerations being imposed upon the pilot. Adherence to Mach number and lateral control deflection limitations as outlined in Section V will prevent exceeding airplane structural limitations.

DIVING

The airplane is capable of relatively high forward flight speeds with afterburning power. At such high speeds, rapid rates of descent result from relatively small angles of descent. Extreme care should be exercised in making any descents below 30,000 feet.

WARNING

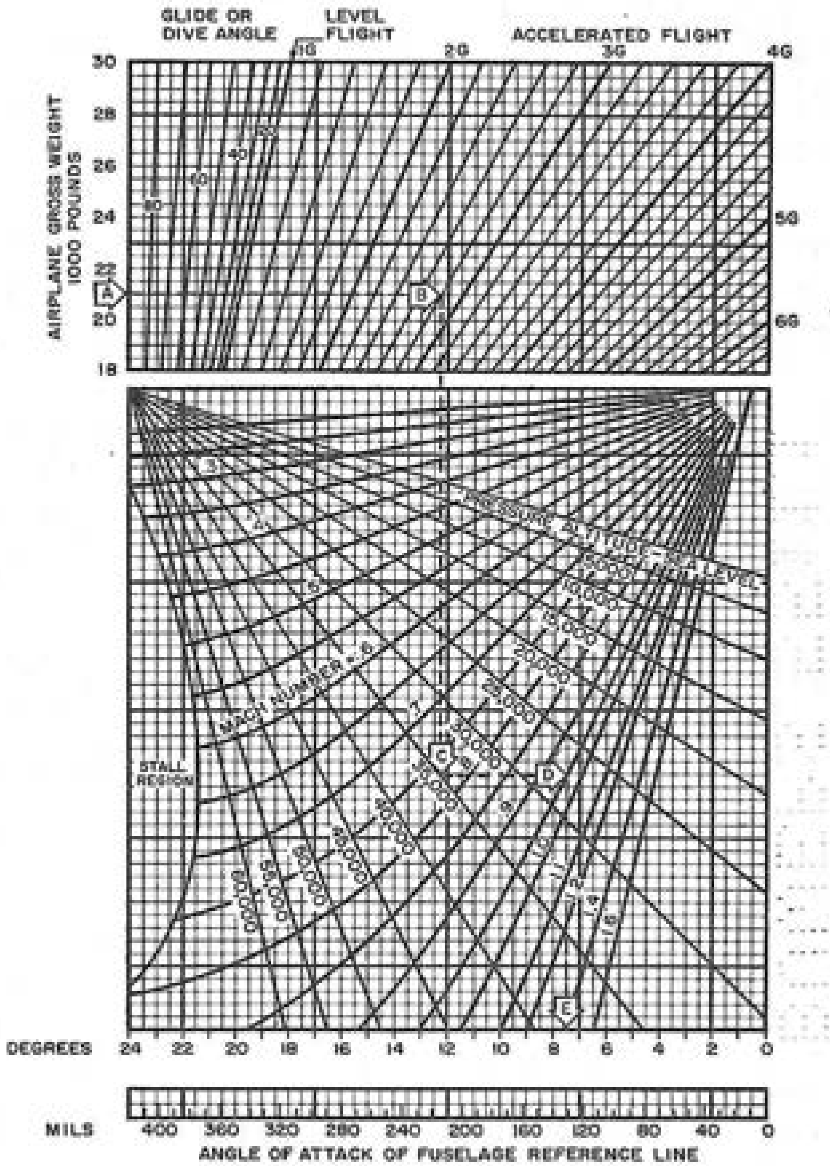
> The airplane is capable of exceeding its limiting airspeeds with alarming ease. Refer to figure 5-2 for airspeed limitations. Refer to the flyleaf at the front of this handbook for description of the airplane's flutter limitation.

FLIGHT WITH EXTERNAL STORES

Flight characteristics with external stores are expected to be normal, but current restrictions should be consulted prior to flight.

ANGLE OF ATTACK RELATIONSHIP

SPEED BRAKES OPEN OR CLOSED



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REMARKS:

Subtract 0.8 degree (14 Navy mils) to obtain angle of attack of Armament Datum Line.

EXAMPLE:

Enter chart at applicable gross weight (e.g., 21,000 pounds).

Select load factor for anticipated flight condition, i.e., level, accelerated, glide or dive (e.g., 2.8g)

Move vertically down to pressure altitude at start of pull-out (e.g., 35,000 ft.).

Move horizontally to intersection with line for selected Mach number (e.g., Mach = 0.95).

From Mach number line, move vertically down and read angle of attack of fuselage reference line in degrees or mils (7.5 degrees or 131 mils).

DATA AS OF: September 1955
DATA BASIS: Estimates

Figure 6-4. Angle of Attack Relationship

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F5D-1 QUESTIONNAIRE Rev.: 7-3-56

A. COMBINATION FUEL PUMP

1. The combination fuel pump has a centrifugal boost stage, supplying two separate gear type pump stages, one provides fuel for the ENGINE the other for the AFTER BURNER.

2. If either pump fails, what is the effect on engine power available? NO A/B AVAILABLE - MIL THRUST IS AVAILABLE

B. FUEL CONTROL SYSTEM

1. If the primary fuel control system malfunctions, how is the emergency system chosen? BY MOVING TOGGLE SWITCH FROM "PRIMARY" TO "MANUAL"

2. When the emergency fuel control system is selected, how is fuel metering controlled? MECHANICAL LINKAGE FROM THROTTLE POSITION TO THROTTLING VALVE IN THE FUEL CONTROL UNIT.

C. ENGINE

1. The engine master switch receives power from the primary DC electrical bus and energizes the following circuits.
   a) ENGINE FUEL CONTROL CIRCUIT
   b) A/C FUEL BOOST PUMP

   and arms the following circuits.

   a) FUEL PUMP FAIL LIGHT
   b) IGNITION SWITCH & IGN TIMER
   c) A/B SWITCH
   d) PITOT & ENG ANTI ICING SWITCH
   e)

unclassified
BuWeps Instruction 5511.1
1 Dec 59

2. How is engine ignition obtained? MOVING THROTTLE OUTBOARD WHICH ALLOWS THE IGNITION TIMER TO ENERGIZE THE IGNITORS (30 SEC)

3. The exhaust nozzle control is actuated by FUEL PRESS from the A/B STAGE OF ENGINE COMBINATION FUEL PUMP
   The exhaust nozzle actuating cylinders are motivated by HIGH PRESSURE COMPRESSOR BLEED AIR

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4. The afterburner fuel control meters fuel to the afterburner fuel nozzles as a function of *COMPRESSOR DISCHARGE PRESSURE*

D. FUEL SYSTEM

1. The fuel system consists of six tanks. Three of the tanks are intended to be filled on all flights while the remaining three are intended as use as auxiliary tanks. What are the names of the respective tanks and their approximate capacity in pounds (assuming 6.5 lbs/gal.)

Regular Tanks / Auxiliary Tanks

	Name	Capacity	Name	Capacity
1.	*AF FUS*	*1270*	*FWD FUS AUX*	*1800*
2.	*LT FWD WING*	*1870*	*RT AFT WING AUX*	*1105*
3.	*RT FWD WING*	*1870*	*LFT AFT WING AUX*	*1105*
	Total	*5010*	Total	*4010*

2. All fuel to the engine comes from the *AFT FUSELAGE* fuel tank.

3. Fuel transfer from the forward wing tanks to the aft fuselage sump tank is accomplished by *AIR DRIVEN FUEL TRANSFER PUMPS (ENGINE BLEED)*
This operation is automatic and controlled by the level of fuel in the *AFT FUSELAGE* fuel tank.

4. Fuel tansfer from the auxiliary tanks to the aft fuselage sump tank is accomplished by *ENGINE BLEED AIR TANK PRESSURIZATION*

The *AUX FUEL TRANS* switch must be in the INTERNAL position to transfer fuel from the auxiliary tanks.

5. The fuel quantity indicating system will reflect the total fuel quantity in all six fuel tanks with the AUX FUEL TRANS switch set at the *INTERNAL* position. With the AUX FUEL TRANS switch set at the *OFF* position, the system will indicate only the fuel quantity of the aft fuselage sump and forward wing tanks, regardless of fuel remaining in the forward fuselage and aft wing auxiliary tanks.

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6. If the fuel supply in the aft fuselage sump tank falls below the 100 GAL gallon level, the thermistor in the aft fuselage sump tank causes the reading of any remaining fuel in the other tanks to be dropped out, resulting in the quantity gage indicating the fuel remaining in the aft fuselage sump only.

E. ELECTRICAL SYSTEM

1. Electrical power is normally supplied by an air turbine driven main a-c generator which delivers 115/200 volt, 3 phase, 400 cycle constant frequency a-c power.

2. The aircraft ~~is~~/is not equipped with a battery or d-c generator.

3. D-c power is provided by conversion of a portion of the a-c power into 28 volt d-c current by means of a TRANSFORMER RECTIFIER.

4. Operation of the electrical system is completely automatic and no controls are furnished in the cockpit except the EMER GENERATOR RELEASE HNDL.

F. HYDRAULIC SYSTEM

1. The aircraft is equipped with two independent 3000 psi hydraulic systems; a utility and an elevon system. The utility system supplies pressure for;

a) LANDING GEAR (EXTENSION & RETRACTION)

b) TAIL BUMPER

c) ARRESTING HOOK

d) SPEED BRAKES

e) ROCKET DOORS

f) POWER BRAKES

g) EMER GENERATOR (EXT & RETRACTION)

h) SERVO RUDDER

i) ELEVATOR SERVO

j) AILERON SERVO

k) 1/2 TANDEM ELEVON ACTUATING CYL

l) 1/2 INBOARD TANDEM ELEVON ACTUATING ASSM.

2. The elevon hydraulic system supplies pressure for:

a) 1/2 TANDEM ELEVON ACTUATING CYL

b) 1/2 INBOARD TANDEM ELEVON ACTUATING ASSM.

3. In the event of utility hydraulic system failure, how can the main gear brakes be utilized? BY EXERTING TWICE NORMAL TOE PRESSURE ON RUDDER PEDALS

G. FLIGHT CONTROL SYSTEM

1. During normal operation the inboard elevon is slaved to the OUTBOARD ELEVON

2. During flight on manual control, the position of the inboard elevon is controlled by the TRIM SWITCH - LATERALLY & LONGITUDINALLY

3. During normal operation lateral and longitudinal trim is accomplished by RESPOSITING CONTROL STICK (CENTERING RELOCATION)

4. During manual operation lateral and longitudinal trim is accomplished by DC POWER - TRIMER SWITCH - INBOARD ELEVONS ARE DEFLECTED SYM FOR PITCH & ASYM FOR LATERAL TRIM

5. In the event of failure of either of the hydraulic systems, control in the HIGHER speed range will be limited.

6. In the event of an electrical failure, how is the MACS setting changed? MANUALLY POSITIONING THE MAC CRANK TO THE UNSTOWED POSITION. THE SETTING MAY THEN BE CHANGD BY OPERATING CRANK MANUALLY

7. What is the mechanical advantage ratio at 0.80 Mach number at sea level? 4.3 to 1 ; at 40,000 feet 1:1

8. Since the yaw damper servo is connected to the mechanical rudder. What happens to the rudder pedals with the yaw damper engaged? THE PEDALS WILL MOVE BACK & FORTH AGAINST PILOT EFFORT AT WILL OF THE YAW DAMPING SYSTEM

9. Pulling the emergency ELEVON REL handle results in the following:

a) DISCONNECTS THE TANDEM ELEVON ACTUATING CYL'S

b) RELEASES LATERAL STK "FEEL" SPRING

c) DISABLES NORMAL LAT & LONG TRIM ACTUATORS

d) ENGAGES EMER INBOARD ELEVON JACKSCREW ACTUATORS

e) COMPLETES PART OF AN ELECTRICAL CIRCUIT TO OPERATE TRIM

f) STICK TRIMER WILL THEN TRIM CONTROL SURFACES

10. To avoid an engagement transient the trim change compensator (TCC) should not be engaged above M .70 Mach number.

11. In order to cancel a possible residual signal after using the TCC, the TCC switch must be moved to the EMER OFF position.

H. MISCELLANEOUS

1. Actuating the fire extinguishing system results in the fire extinguishing agent to be discharged into the ENG ACCESSORY and TURBINE and AFTER BURNER compartments.

2. The canopy should not be opened in flight and on the ground at relative winds in excess of 88 knots.

3. The Master Warning Light is energized by a warning circuit on the "Panic Panel" being energized. The specific condition energizing a particular warning circuit is as follows:

ITEM	SPECIFIC CONDITION RESULTING IN ENERGIZING THE WARNING LIGHT
Fuel Boost Pump	LOW FUEL PRESS 5 ± 1 PSI
Fuel Pump	FAILURE OF ENG STAGE
Elevon Pressure	LOW HYDRAULIC PRESS 850 ± 50 PSI
Utility Pressure	" " " 850 ± 50 PSI
Fuel Quantity	LOW FUEL QUALITY 650 lbs AFT FUS
A-C Generator	A-C GEN HAS FAILED
Oil Pressure	OIL PRESS BETWEEN 25 / 35 PSI
Emergency Fuel Control	EMER POSITION
D-C Power	NO POWER AVAILABLE

I. OPERATIONAL PROCEDURES

1. Prior to engine start, the following major items should be checked for correct condition:

ITEM	CORRECT CONDITION
a) Landing gear handle	DOWN
b) Eng Cont switch	PRIMARY
c) Throttle	OFF
d) ENGINE MASTER switch	OFF
e) Fuel valve	OPEN

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2. Prior to <u>take-off</u>, the following major items should be checked for correct condition;

<u>ITEM</u>	<u>CORRECT CONDITION</u>
a) Mechanical advantage ratio	T.O. #1
b) Canopy	Closed & Latched
c) ENG CONT switch	Primary
d) Speed brakes	CLOSED
e) Trim Change Compensator	OFF
f) AUX FUEL TRANS switch	OFF
g) Engine bleed valve	AUTO
h) Trim	
Longitudinal	6° UP
Lateral	0
Directional	0
i) Warning Lights	OFF

3. In case of a utility system hydraulic failure, list the two steps necessary to extend the landing gear.

a) LND GEAR HNDL DWN

b) PULL EMER LNDG GEAR RELEASE

4. In case of a landing gear "barber pole" indication, how may it be ascertained which component of the gear is at fault?

By visually observing the 4 wheel position indicators on the instrument panel

5. What is the recommended seat ejection procedure?

AIR CONDITIONING "OFF" — SLOW A/C DOWN — FEET ON PEDALS — SHOULDER HARNESS LOCKED — SIT ERECT PULL FACE CURTAIN — OPEN CHUTE WHEN CLEAR OF SEAT

6. Air start procedure:

ITEM	DESIRED CONDITION
a) Throttle (after flame-out)	OFF
b) Fuel control	PRIMARY or MANUAL (FUEL CONT MALFUNCTION)
c) Engine Master switch	ON
d) EMER. GEN. release handle	PULL
e) Recommended airspeed	160 - 250
f) Recommended engine speed	12% to 30%
g) Throttle	IGNITE then IDLE
h) After successful start, EMER. GEN release handle	PUSH IN TO "NORMAL"

7. Complete control system failure:

ITEM	DESIRED CONDITION
a) Control stick	EXTEND
b) Mechanical advantage ratio	2:1
c) Recommended airspeed	230 - 280
d) Elevon release handle	PULL

8. What is the indication of main generator failure? ALL ELECTRICAL EQUIPMENT WILL BE RENDERED INOPERATIVE IMMEDIATELY

9. What is the corrective procedure for main generator failure? PULL EMER GEN RELEASE PULL AUTO CONTROL RELEASE HANDLE - POSITION MAC AS DESIRED MANUALLY LAND AS SOON AS POSSIBLE

10. What is the corrective procedure for a complete D-C power failure?
NONE - LAND SOON AS POSSIBLE

11. List the inoperative items after a complete D-C power failure:

a) FUEL PRESS INDICATOR OUT

b) OIL PRESS " OUT

c) ELEV HYD PRESS OUT

d) UTILITY HYD PRESS OUT

e) LND GEAR INDICATORS OUT

f) TRIM POSITION INDICATORS INOP

g) MACS INDICATORS INOP

h) SPEED BRAKE SWITCH INOP

i) CABIN TEMP CONTROL INOP

j) ARMAMENT EQUIPT INOP

12. What is the proper throttle position should it become necessary to transfer the fuel control system from normal to emergency?

____

J. LIMITATIONS

1. Engine:

a)

OPERATING CONDITION	MAX. EXHAUST GAS TEMPERATURE BELOW 35,000 feet	ABOVE 35,000 feet	Percent of Maximum rpm	Time Limit (minimum)
Afterburner	630	660	102.2	15
Military	620	650	*	30
Acceleration	670	670	*	2

*These values vary with inlet temperature.

b) Normal oil system pressure________________psi

c) Normal fuel boost pressure________________psi

2. Airplane

a) Maximum indicated airspeed______________knots

b) Maximum indicated Mach number____________

c) Maximum load factor at__________#fuel remaining

	Subsonic	Supersonic
Positive	______	______

d) Maximum indicated speed for landing gear extension__________knots

e) Maximum indicated airspeed for speed brake extension_________knots

f) Maximum speed for full rudder pedal deflection_______________knots

g) Maximum relative wind for canopy open__________88__________knots

h) Maximum allowable sinking speed for landing_________________fps

F5D QUESTIONNAIRE

A. 1. MAIN ENG. FUEL CONTROL
AFTERBURNER FUEL CONTROL

~~[illegible] possible [illegible]~~
2. ONLY MILITARY PWR. WILL BE AVAILABLE, NO A/B.

B. 1. By selecting MANUAL ON ENG. CONT. SW. which positions solenoid valve to ROUTE fuel through emerg. fuel control. Metering is DONE MANUALLY by DIRECT THROTTLE LINKAGE.

2. By MECHANICAL LINKAGE BETWEEN THROTTLE & EMERG. FUEL CONT.

C. 1. d. c.
a.) eng. fuel cont. CIRCUIT
b.) a-c elect fuel boost if a-c elect pwr. is AVAIL.
a.) Fuel pump fail. LIGHT
b.) PITOT & ENG. ANTI-ICING SW.
c.) A/B SW.
d.) IGNITION SW
e.) IGNITION TIMER

2. Two plugs in lwr. Two combustors

3. fuel press. A/B fuel CONT. HI - PRESS comp.

bleed air

4. Comp. OUTLET PRESS.

D. 1. REG. TANKS

REG. TANKS		AUX. TANKS	
REAR FUS. SUMP -	1202.5	FWD. FUS. TANK -	1722.5
L. FWD. WING -	1829.8	L. AFT. WING -	1092
R. FWD. WING -	1829.8	R. AFT. WING -	1092
TOTAL -	4862	TOTAL -	3906.5

2. AFT. FUS. Sump

3. AIR DRIVEN BOOST PUMP AND PRESS. TANKS. SUMP

4. AIR DRIVEN BOOST PUMPS & PRESS. TANKS.

AUX FUEL TRANSFER CONT.

5. INTERNAL

OFF OR DROPS