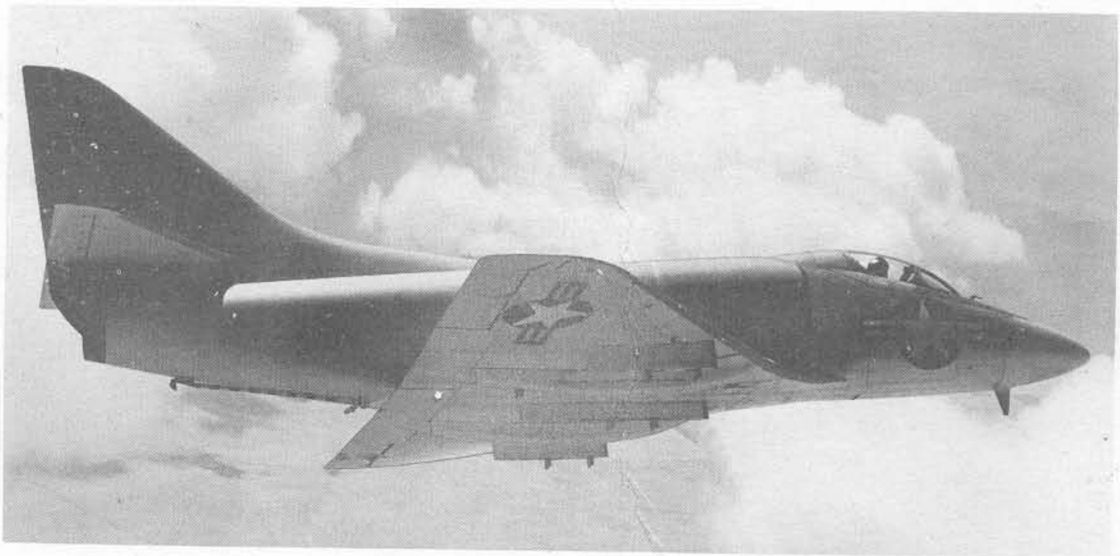
AN 01-40AVA-1

Flight Handbook

NAVY MODEL

A4D-1, A4D-2

AIRCRAFT



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1 October 1956

LIST OF REVISED PAGES ISSUED

INSERT LATEST REVISED PAGES. DESTROY SUPERSEDED PAGES.

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

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

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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.

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 aircraft he is assigned to fly. Thus, as complete a picture as is practicable of the basic structure and installations of the aircraft along with the fundamental operating procedures is included. It is not the function of this handbook to teach the pilot how to fly the airplane, as it is assumed he is competent in this respect. However, the handbook contains information regarding behavior peculiar to the aircraft in various conditions of flight and ground operation.

The book is divided into nine sections. Sections I, II and III are closely interrelated and contain 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 which is not 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 airplane is approached 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 which must be observed during normal operation are discussed in Section V. Section VI attempts to evaluate any unusual flight characteristics, both favorable and unfavorable, that the aircraft may possess. A portion of Section IV and all of Sections V and VI are currently classified Confidential and are published in a supplemental handbook, AN 01-40AVA-1A. In Section VII, operation of the various aircraft systems is discussed. Section VIII is not applicable to single-place aircraft. Section IX contains procedures and information pertaining to all-weather operation.

A Confidential Appendix, which is published under a separate cover, contains all flight operating data charts for the airplane and other Confidential material which is supplementary to this Flight Handbook. Refer to AN 01-40AVA-1A, Supplement to AN 01-40AVA-1 Temporary Flight Handbook for Navy Model A4D-1 Aircraft.

It should be noted that the information in this handbook will be kept current by frequent revisions. Since, however, a slight delay in the dissemination of revision material is to be expected, *it is imperative that pilots stay abreast of pertinent technical directives which frequently cover critical flight restrictions or new techniques in operation of the aircraft.*

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 control positions as they are identified in the airplane. For example, 'The SEAT switch is spring-loaded to the center (off) position, and a momentary movement to "UP" or "DOWN" will adjust the seat accordingly.'

Equipment and procedures not applicable to all models of the aircraft are called out by footnotes. A key to the footnote system is herein provided. The airplanes are listed in groups of major differences (designated by capital letters: A, B, C, etc.), which are in turn broken down into sub-groups (designated by small letters following the capital of the group: Aa, Ab, Ba, etc.). These sub-groups may again be divided and assigned an additional small letter, such as Aaa, Aab, etc. This system should aid the reader to quickly locate all the remarks applicable to his airplane, and to reject the inapplicable ones. For example, only footnotes containing either "Group A" or "Group Ad" are applicable to Airplane BuNo 137820.

Group		*Airplanes*	*Tail Designation*
A	a	BuNo 137813 through 137816	A4D-1
	b	BuNo 137817	A4D-1
	c	BuNo 137818	A4D-1
	d	BuNo 137819, 137820	A4D-1
	e	BuNo 137821	A4D-1
B	a	BuNo 137822	A4D-1a
	b	BuNo 137823	A4D-1a
	c	BuNo 137824 through 137830	A4D-1a
	d	BuNo 137831	A4D-1a
C	a	BuNo 139919 through 139938	A4D-1b
	b	BuNo 139939 through 139952	A4D-1c
	c	BuNo 139953	A4D-1c
	d	BuNo 139954 through 139970	A4D-1c

Group		*Airplanes*	*Tail Designation*
D	a	BuNo 142142 through 142161	A4D-1d
	b	BuNo 142162 through 142186	A4D-1e
	c	BuNo 142187 through 142211	A4D-1f
	d	BuNo 142212 through 142235	A4D-1g
E	a	BuNo 142082 through 142086	A4D-2a
	b	BuNo 142087 through 142096	A4D-2b
	c	BuNo 142097 through 142116	A4D-2c
	d	BuNo 142117 through 142141	A4D-2d

A difference in aircraft footnoted by any letter or combination of letters is applicable to all the sub-groups under that letter or letters, but not to the next major group. (Example: a footnote containing "Group A" applies to all airplanes from BuNo 137813 through 137821, while a footnote containing "Group Ae" pertains only to BuNo 137821).

An alphabetical index is included at the end of the book to facilitate reference to the text.

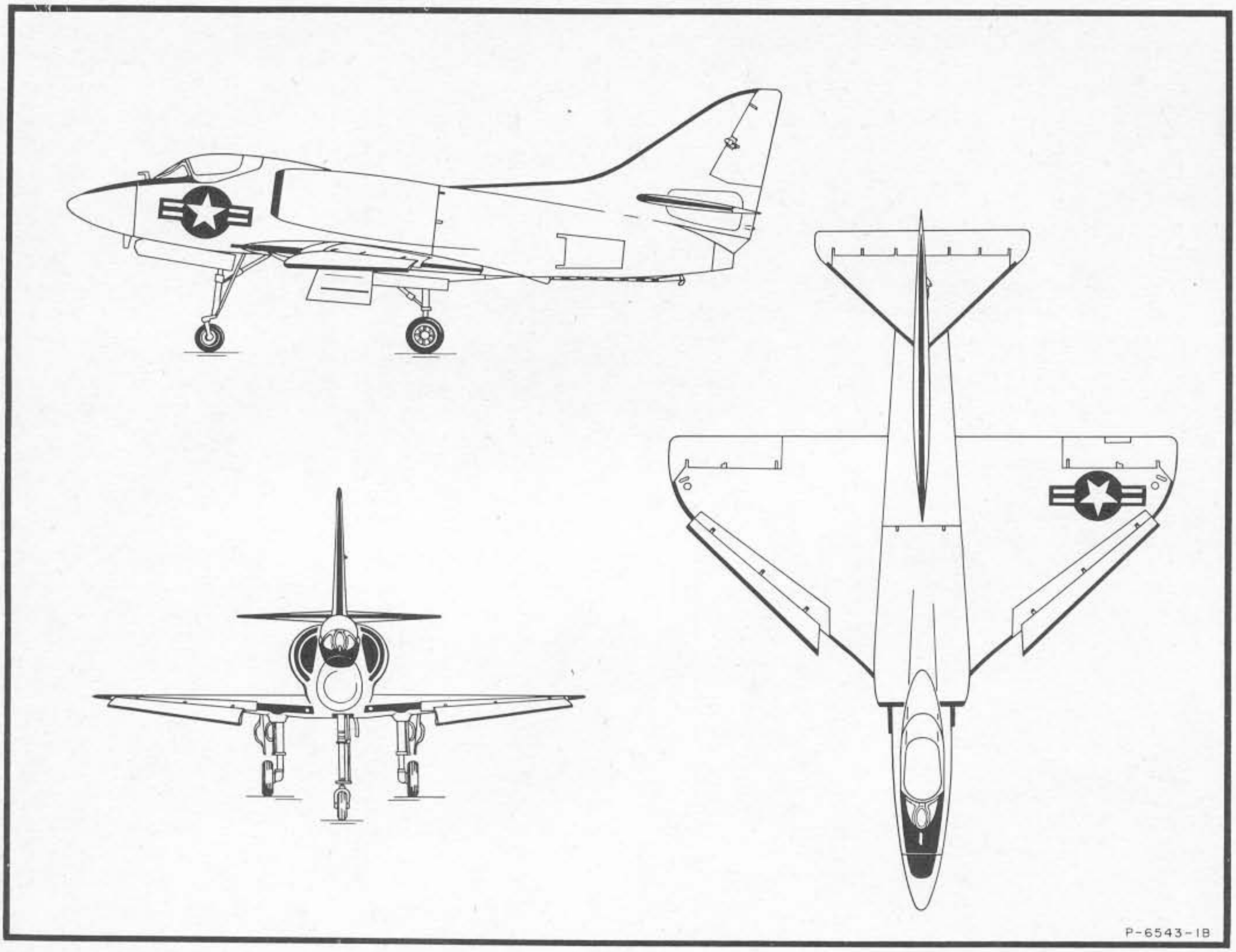


Figure 1-1. Model A4D-1, A4D-2 Airplane

SECTION 1
DESCRIPTION

THE AIRPLANE

The Navy Model A4D-1, -2 Skyhawk is a jet-propelled, single-place monoplane with a modified delta-planform wing, manufactured by the Douglas Aircraft Company, El Segundo Division. Designed as a high performance, light weight day-attack airplane, the Model A4D-1, -2 mounts two 20-mm guns internally[1], carries a variety of external stores, and is capable of operating either from a carrier or from a shore base.

DIMENSIONS. The principal three-point dimensions of the airplane are as follows:

Span	27 feet 6 inches
Length	39 feet 4 5/16 inches
Height	14 feet 11 9/16 inches

ENGINE

The airplane is powered by a Wright J65-W-4B[2] or J65-W-16A[3] turbo-jet engine with a multi-stage, axial-flow compressor driven by a two stage turbine assembly. Engine accessories are driven through an accessory gear box by the compressor. Components driven by the gear box are the engine-driven fuel pump, hydraulic pump, fuel control governor, tachometer generator, and the ignition generator. Cooling of the engine is accomplished by the routing of compressor air through various parts of the engine. The main bearings of the engine are cooled by air bled from the fifth compressor stage and are lubricated by oil from the oil system. The Wright J65-W-4B[2] and J65-W-16A[3] engines are rated at standard sea level static conditions as follows:

RATING	*THRUST*	*RPM*
Military	7700	"MIL" (100.0%)—8300
Normal (continuous)	6780	"NORM" (96.7%)—8030

ENGINE FUEL CONTROL SYSTEM

The engine fuel control system consists of a fuel pump, fuel control unit, six flow dividers, a fuel primer solenoid, and two fuel primers.

FUEL PUMP. The engine-driven fuel pump is composed of a centrifugal booster element and two parallel gear-type elements operating in series. Should either gear element fail, the operative element is capable of supplying the engine with enough fuel for full power requirements under static sea level conditions. An ENGINE FUEL PUMP FAILURE warning light in the right-hand wheel well provides the ground crew with an indication of the functioning status of the fuel pump.

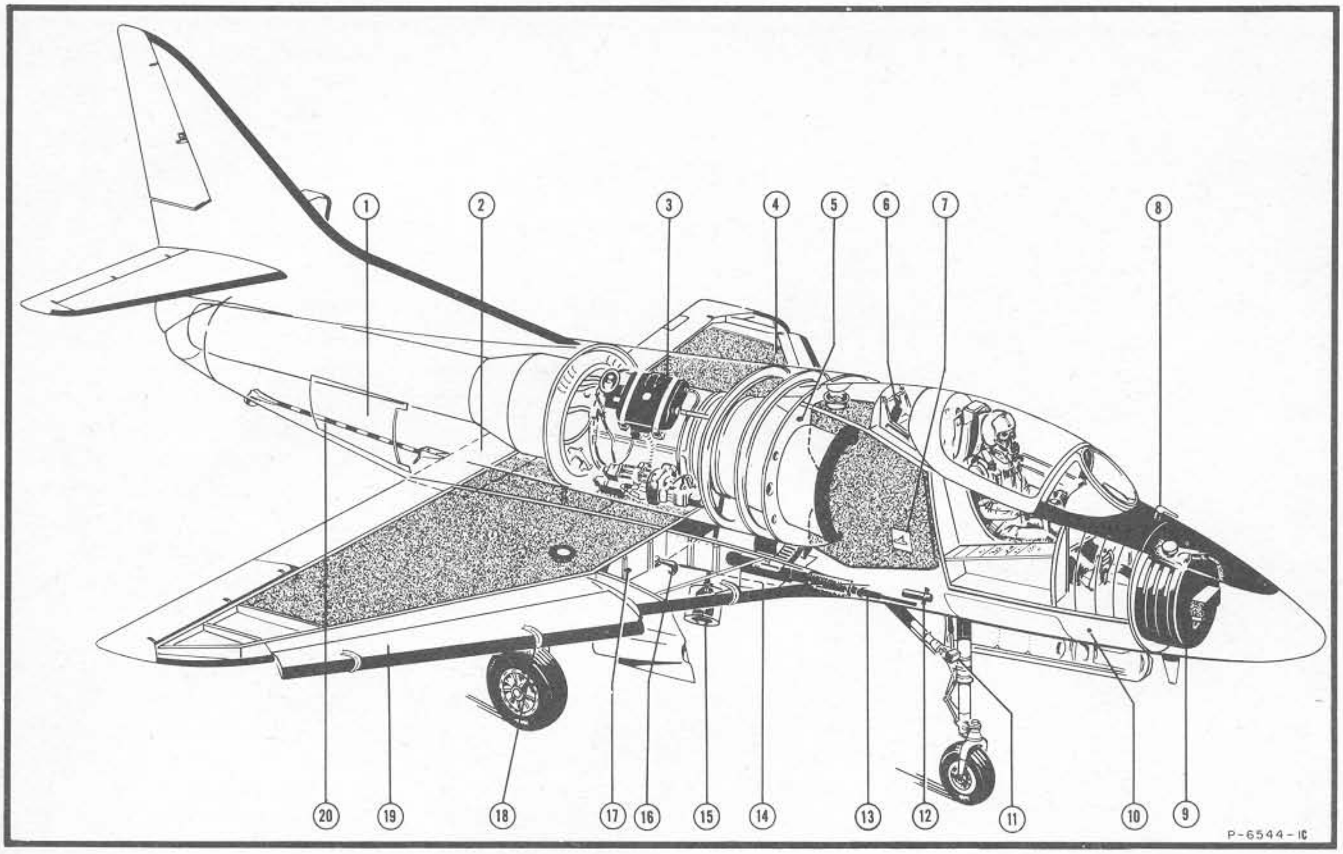
[1] Group Ab, Bb, C, D, and E Airplanes.
[2] Group A, B, C, and D Airplanes.
[3] Group E Airplanes.

FUEL CONTROL UNIT. The engine fuel control unit contains two control systems: a primary system for normal operation, and a manual system for use if the primary system should fail. When operating on the primary fuel control system, the engine fuel control unit provides fuel governing for all engine speeds, including idle. "Built-in" pre-determined operating schedules provide optimum rates of acceleration and deceleration for all throttle movements under any ambient condition. The acceleration schedule insures a safe surge margin during rapid advancement of the throttle, and the deceleration schedule prevents flame-out during rapid retarding of the throttle. When compressor discharge pressure exceeds a pre-determined value, a limiter causes a reduction in fuel flow with a corresponding reduction in rpm, which prevents exceeding compressor discharge pressure limits. This reduction in rpm normally occurs at low altitudes, high airspeeds, and at temperatures below standard, and should not be misconstrued as malfunctioning of the fuel control unit. Basically, the fuel control unit consists of two main valves in series: a governor valve, and a contoured regulator valve. The governor valve is adjusted by throttle position, which spring-loads the valve against a balancing force from a fly-ball governor. The fly-ball governor senses engine rpm and acts to open or close the governor valve as necessary to maintain the throttle-selected rpm. The correct fuel pressure drop across the governor valve is maintained by the regulator valve, which is positioned by, and compensates for, ambient conditions and engine speed.

When operating on the manual fuel control system, all fuel metering is accomplished manually by the throttle through direct linkage with an emergency throttling valve in the fuel control unit, but with no compensation for ambient conditions or engine speed. Two emergency transfer valves, actuated electrically by the FUEL CONTROL switch in the cockpit, isolate the automatic metering features of the fuel control unit by re-routing the flow of fuel to the engine and also deactivate the compressor pressure limiter.

CAUTION

Operation on the manual fuel control system requires that the throttle be moved slowly and smoothly to prevent overspeeding and excessive exhaust temperatures.



1. Speedbrake
2. Aft engine compartment access door
3. Oil tank
4. Integral wing tank
5. Fuel nozzle grounding receptacle
6. Cockpit canopy air bungee cylinder
7. External CANOPY JETTISON handle[1]
8. Pitot tube
9. AN/ASQ-17 Integrated Electronic Central
10. Static vent
11. Approach light
12. Emergency generator
13. MK 12 MOD 0 20-mm gun
14. Forward engine compartment and accessory section access door
15. External pneumatic starter
16. Catapult hook
17. Fuel nozzle grounding receptacle
18. Barricade strap detent
19. Wing slat
20. Arresting hook

Figure 1-2. General Arrangement

FUEL PRIMERS. When the throttle is moved from "OFF" to "IDLE" during starting, priming fuel is supplied automatically from the fuel control unit through the primer solenoid valve to the fuel priming tubes. The primer tube nozzles are installed in conjunction with the igniter plugs to supply fuel for ignition. When a centrifugal switch within the ignition generator closes at 1900 $\pm$ 50 rpm, the fuel primer solenoid valve is closed, thus stopping the flow of priming fuel.

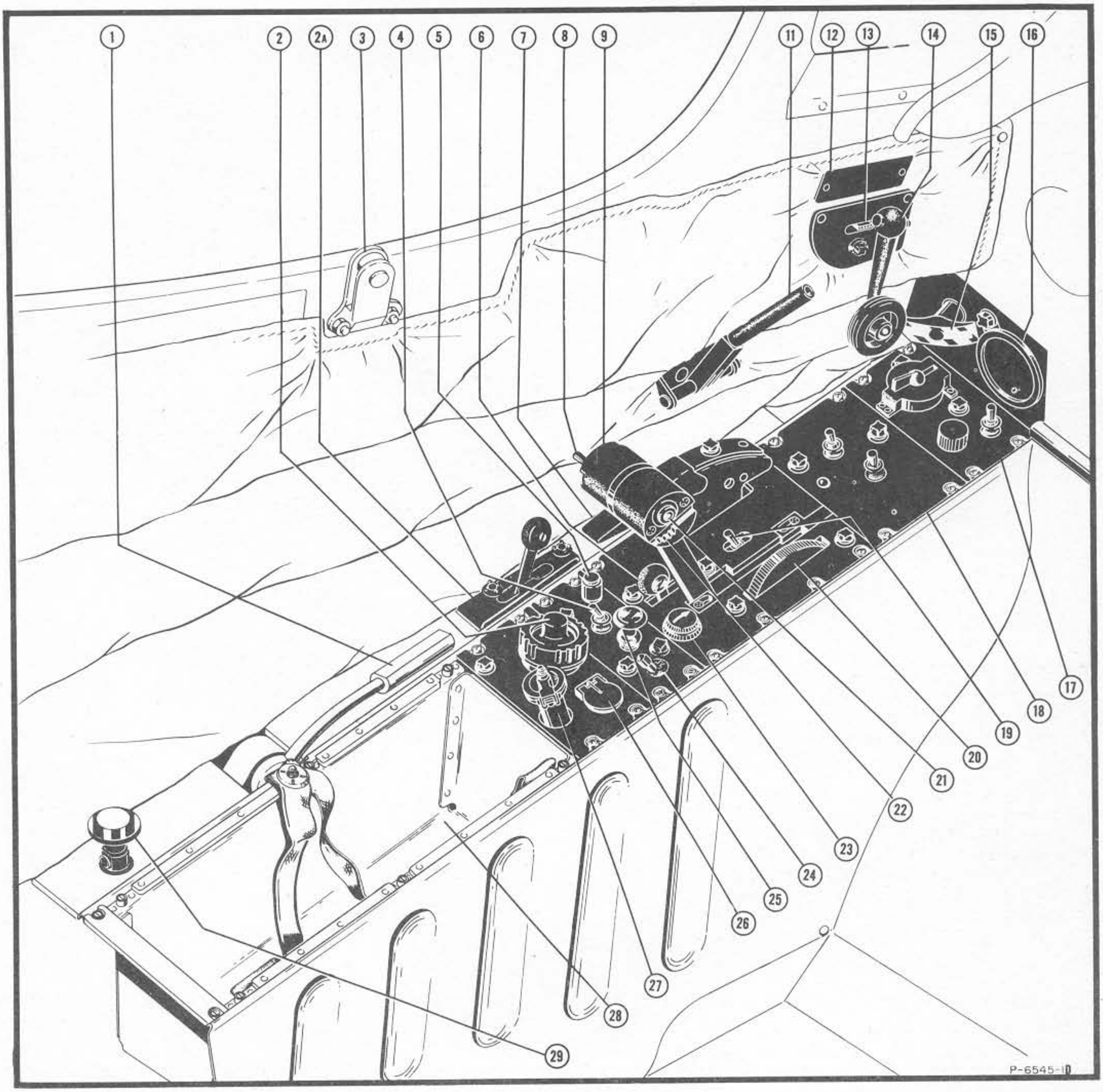
FLOW DIVIDERS. The flow dividers are fuel manifolds which are connected by lines to the fuel burner. Each of the six flow dividers contains an inlet port with filter, a pressure regulating piston, and an outlet port with pressure tap.

FUEL CONTROL SWITCH. A two-position FUEL CONTROL switch (24, figure 1-3) on the ENGINE control panel is used to select the mode of operation of the engine fuel control unit. With the switch at "PRIMARY", the automatic metering devices in the fuel control unit regulate the flow of fuel to the engine to maintain a given power setting as established by the position of the throttle. Pressure, temperature, and engine speed sensing devices are continuously acting to maintain the selected power condition, regardless of existing ambient conditions. When the FUEL CONTROL switch is placed at "MANUAL", fuel flow is routed around the automatic metering mechanisms, the MANUAL FUEL warning light (5, figure 1-3) is illuminated, and all fuel metering is then accomplished directly by manipulation of the throttle.

ENGINE CONTROLS

THROTTLE. The throttle (9, figure 1-3) on the left console, is mechanically linked to the engine fuel control unit and provides a means of selecting engine power

[1]Group D, Ec, and Ed Airplanes.

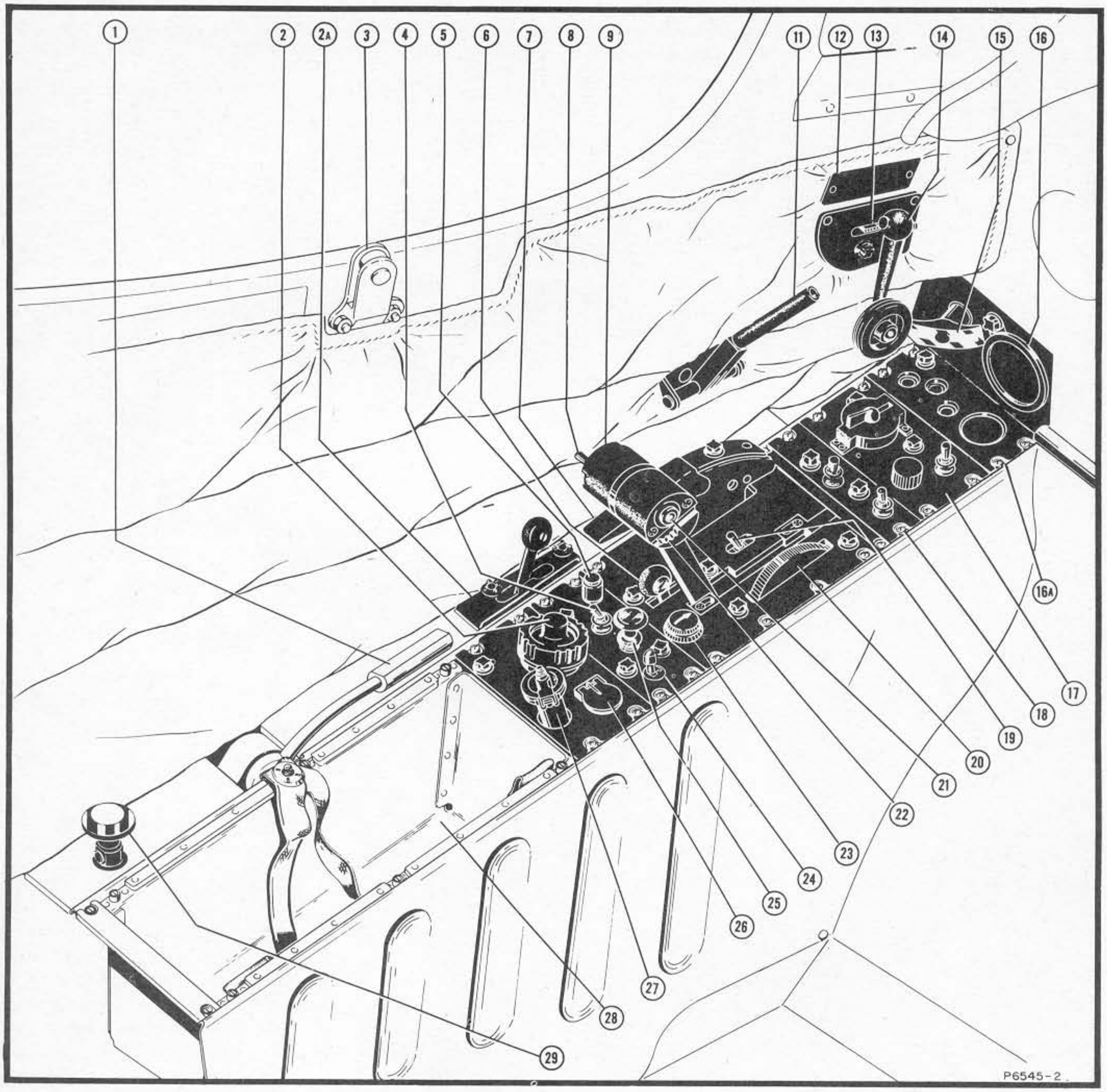


1. Canopy control
2. Anti-blackout control valve
2A. Manual fuel valve control
3. Canopy latch pin
4. External drop tanks pressurizing switch
5. Manual fuel control system warning light
6. Horizontal stabilizer manual override control
7. Wing flaps control
8. Exterior lights master switch
9. Throttle
10. Deleted
11. Catapult handgrip
12. Airspeed correction card
13. Landing gear retraction release switch
14. Landing gear control
15. Emergency landing gear release handle
16. Oxygen system pressure(1) or quantity(2) gage
17. AN/ARN-14E omni-range radio control panel
18. AN/ARN-12 marker beacon receiver control panel
19. Air starting ignition switch
20. Throttle friction wheel
21. Radio microphone switch
22. Speedbrakes switch
23. Rudder trim switch
24. Fuel control system selector switch
25. Starter switch
26. Oxygen and radio receptacle
27. Anti-blackout suit hose connection
28. Map case
29. Emergency speedbrake control

(1) Group A and B Airplanes.
(2) Group C, D and E Airplanes.

Group A, B, C and D Airplanes

Figure 1-3. Cockpit — Left Console (Sheet 1)



1. Canopy control
2. Anti-blackout control valve
2A. Manual feed valve control
3. Canopy latch pin
4. External drop tanks pressurizing switch
5. Manual feed control system warning light
6. Horizontal stabilizer manual override control
7. Wing flaps control
8. Exterior lights master switch
9. Throttle
10. Deleted
11. Catapult handgrip
12. Airspeed correction card
13. Landing gear retraction release switch
14. Landing gear control
15. Emergency landing gear release handle
16. Oxygen system quantity gage
16A. Wheels and flaps position indicators
17. AN/ARN-14E omni-range radio control panel
18. AN/ARN-12 marker beacon receiver control panel
19. Air starting ignition switch
20. Throttle friction wheel
21. Radio microphone switch
22. Speedbrakes switch
23. Rudder trim switch
24. Fuel control system selector switch
25. Starter switch
26. Oxygen and radio receptacle
27. Anti-blackout suit hose connection
28. Map case
29. Emergency speedbrake control

Group E Airplanes

Figure 1-3. Cockpit — Left Console (Sheet 2)

conditions for which the fuel control unit meters fuel to the engine. When operating on the manual fuel system, the throttle, through linkage with an emergency throttling valve, manually meters fuel to the engine with no compensation for engine speed or ambient conditions. Marked positions of the throttle are "OFF", "IDLE", "NORMAL" and "MILITARY". The "OFF" position closes a fuel cut-off valve in the fuel control unit, stopping all fuel flow to the engine. The "IDLE" position incorporates a detent to prevent inadvertent movement of the throttle to the "OFF" position. At "NORMAL", the engine should develop the maximum rpm allowed for continuous operation, and at "MILITARY" should develop maximum power. On the inboard side of the throttle grip are located switches for the operation of the radio microphone and speedbrakes, and on the outboard side is the exterior lights master switch. On the left console, inboard of the throttle, is the THROTTLE FRICTION wheel (20, figure 1-3) which is rotated forward to increase friction on the throttle. To prevent retardation of the throttle during catapulting, a handgrip (11, figure 1-3), which extends from its spring-loaded position against the cockpit rail is grasped in conjunction with the throttle.

ENGINE CONTROL PANEL. The ENGINE control panel, just aft of the throttle on the left console, contains all other controls for the operation of the engine. On the panel are the MANUAL FUEL warning light, the DROP TANKS switch, the START-ABORT switch, and the FUEL CONTROL switch.

ENGINE PERFORMANCE INDICATOR. The engine performance indicator (figure 1-4), on the instrument panel, is part of an integrated performance indicating system which provides engine speed, exhaust temperature, fuel boost pressure, and engine oil pressure readings on one instrument. Engine speed is presented in terms of engine operating conditions as well as in percentage of maximum rpm. These conditions are "START", "IDLE", "NORM", "MIL" and "ACC". Between "IDLE" and "NORM" the instrument is marked in percentage of maximum rpm from 70 to 95; the "MIL" mark is labeled 100.

Exhaust temperature is presented in terms of engine operating conditions as well as exhaust gas temperature. These conditions are "NORM", "MIL", and "ACC". The "MIL" exhaust temperature mark is based on an ambient temperature of 100°F, but as the ambient temperature decreases the "MIL" limitation also decreases. This is shown on the instrument by marked increments of 10°C between "MIL" and "NORM". Each mark represents a new "MIL" limitation as the ambient temperature changes by 40°F. Between the lowest reading of the instrument and "ACC", the exhaust gas temperature is graduated from 200° to 800° centigrade.

Fuel boost pressure and engine oil pressure are shown in small windows at the top and bottom, respectively, of the indicator. Fuel boost pressure is indicated by the words "OUT" and "NORM"; oil pressure is shown by the words "OUT", "NORM" and "HIGH".

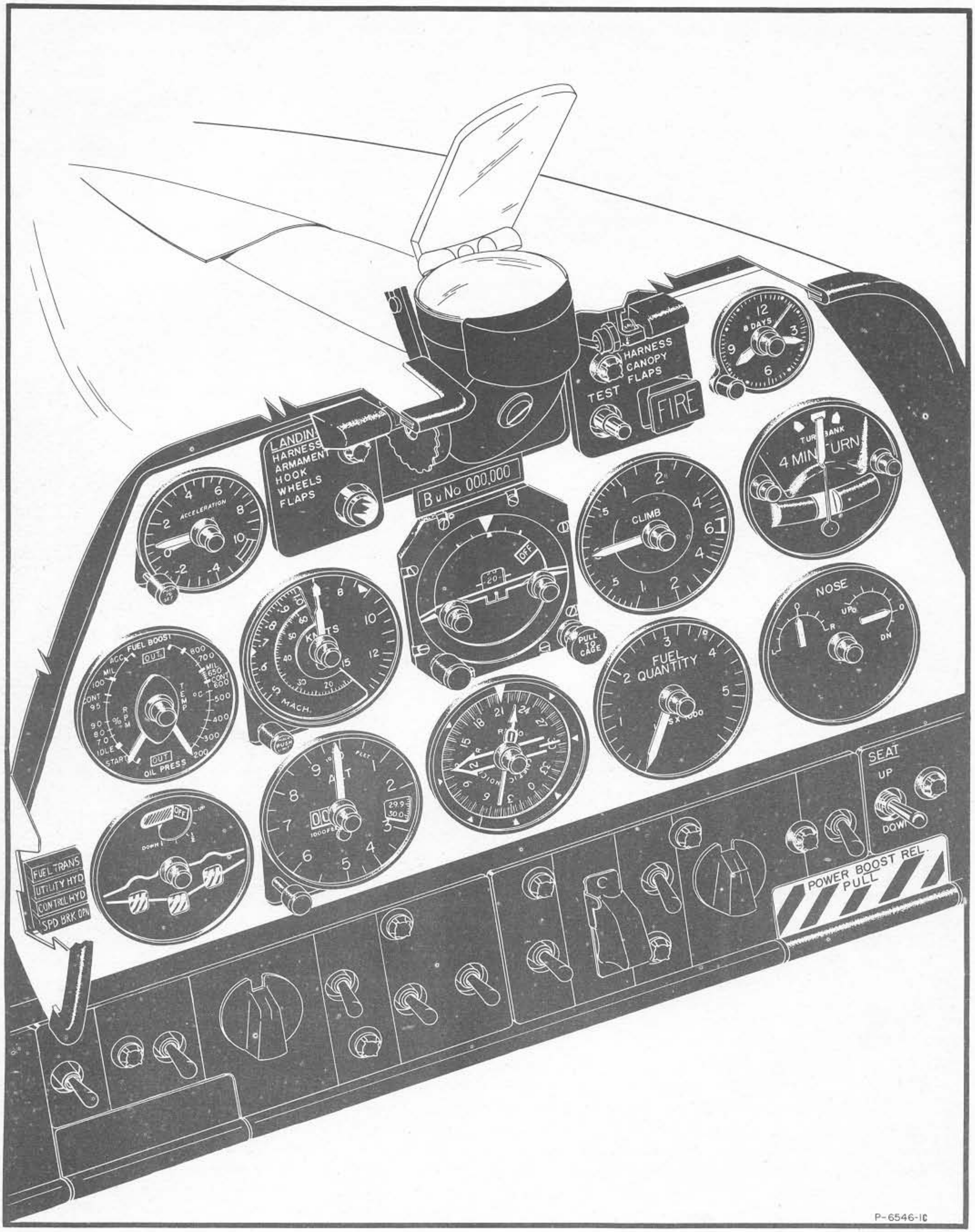
Instrument indications and their interpretations are listed below:

INDICATION		**INTERPRETATION**
	RPM	
"START"	830–1029	10.0%–12.4%
"IDLE"	3525–3985	42.0%–48.0%
"NORM"	8010	96.5%
"MIL"	8360	100.7%
"ACC"	8800	106.0%
	TEMP	
"NORM"		595°C
	(Ambient Temperature)	
"MIL"	100°F	650°C
	60°F	640°C
	20°F	630°C
	−20°F	620°C
	−60°F	610°C
"ACC"		800°C
Maximum indication		900°C
	FUEL BOOST	
"OUT"		Below 4 psig
"NORM"		4 psig and above
	OIL PRESS	
"OUT"		Below 20 psig
"NORM"		20 to 40 psig
"HIGH"		Above 40 psig

STARTER

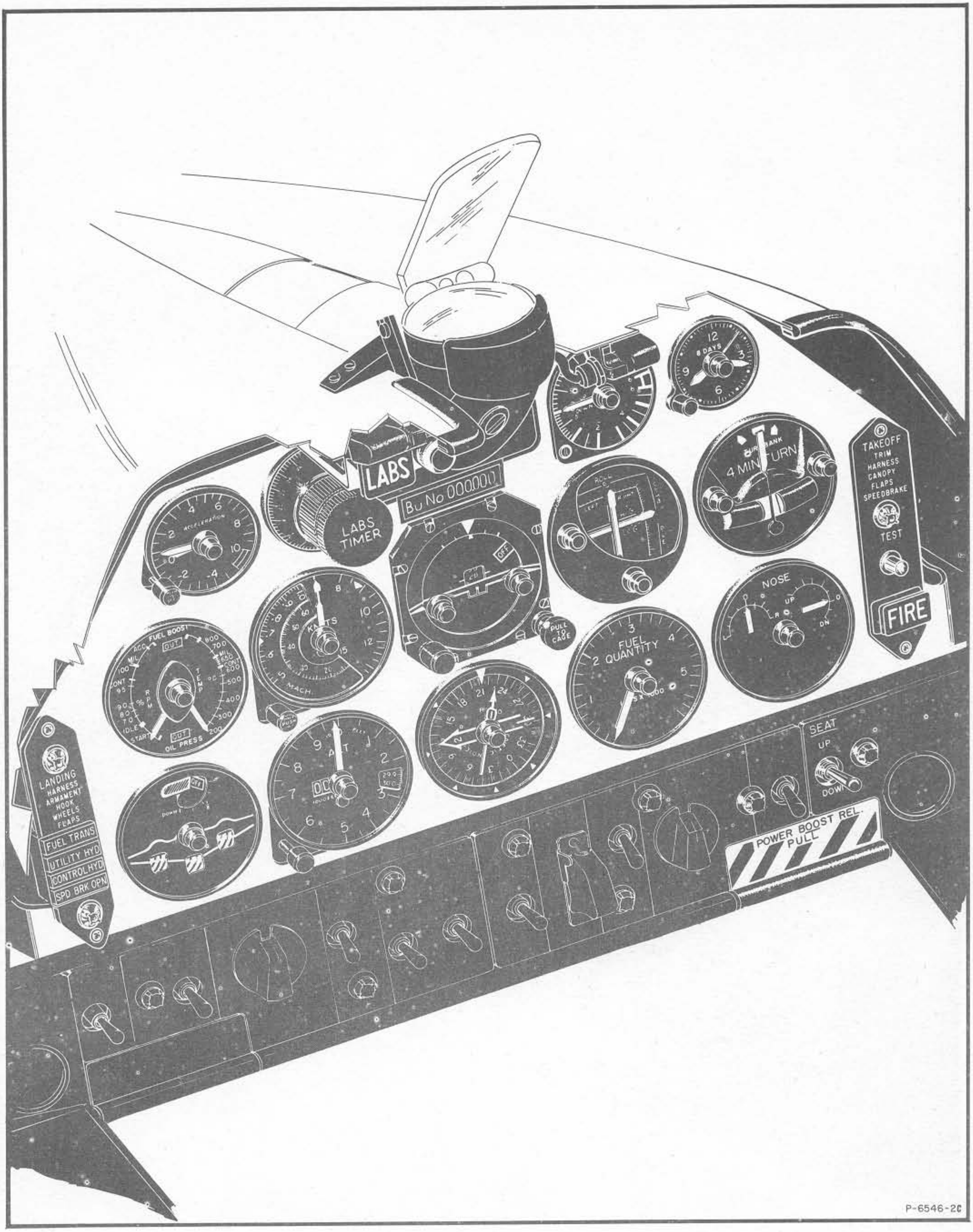
The engine is started on the ground by use of an external pneumatic starter driven by compressed air from a mobile gas turbine compressor. The starter shaft is inserted into a receptacle located inboard of the right main wheel well, where it engages the hydraulic ball pump generator drive, completing both mechanical and electrical connections and allowing actuation of the starter from the cockpit. Provision is made for carrying the starter internally in the aft fuselage under the engine for use when landing away from base. The gas turbine compressor can be carried externally on any store rack.

START-ABORT SWITCH. Actuation of the starter is controlled by the START-ABORT switch (25, figure 1-3) on the ENGINE control panel. When the switch is depressed, the starter air supply solenoid valve opens allowing compressed air from the gas turbine compressor to rotate the starter shaft. When the engine speed reaches a predetermined rpm, a starter overspeed switch opens, allowing the START-ABORT switch to pop out, thus stopping the air supply to the starter. Manually pulling out the START-ABORT switch will also stop the starter air supply.

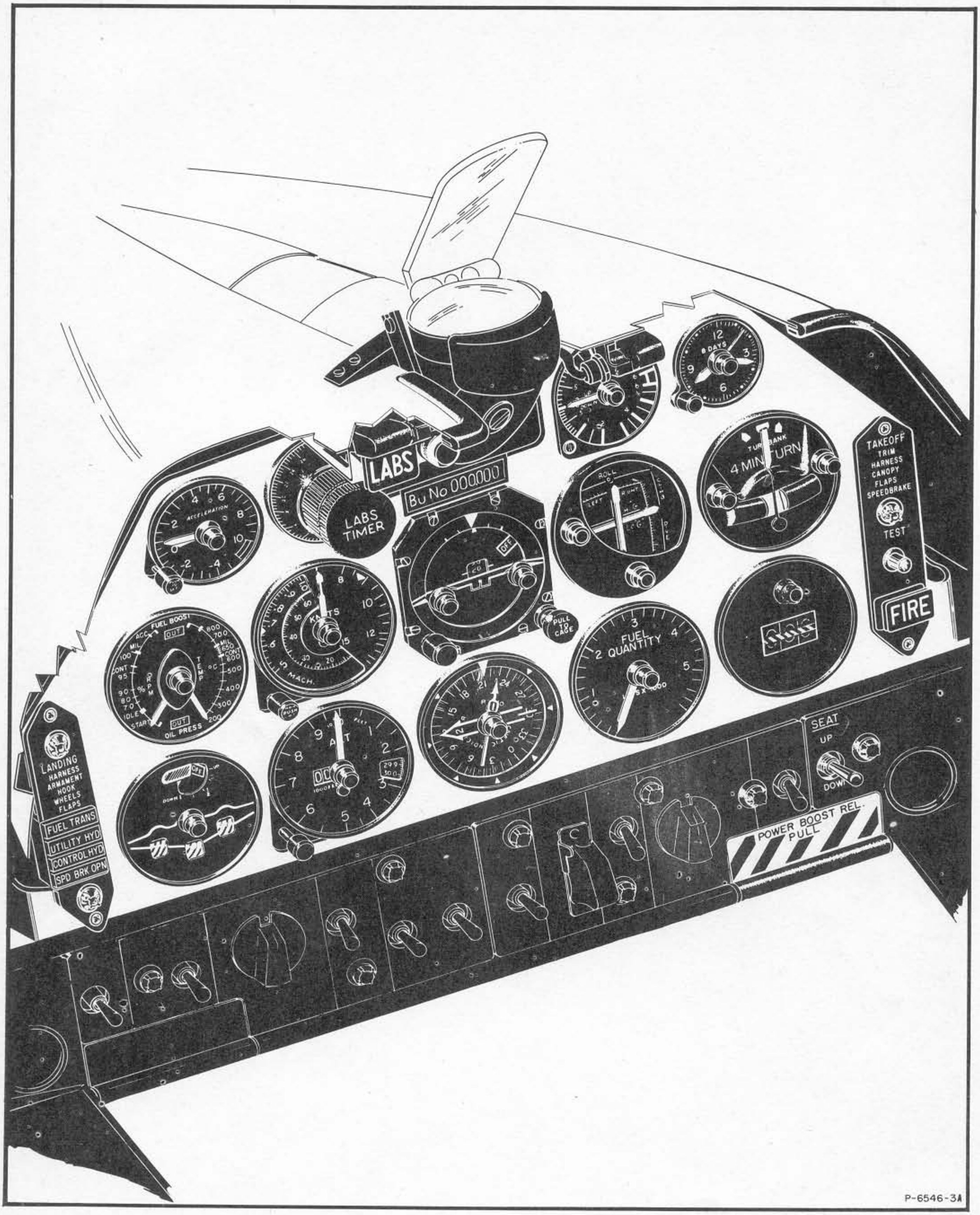


Group A and Ba Airplanes

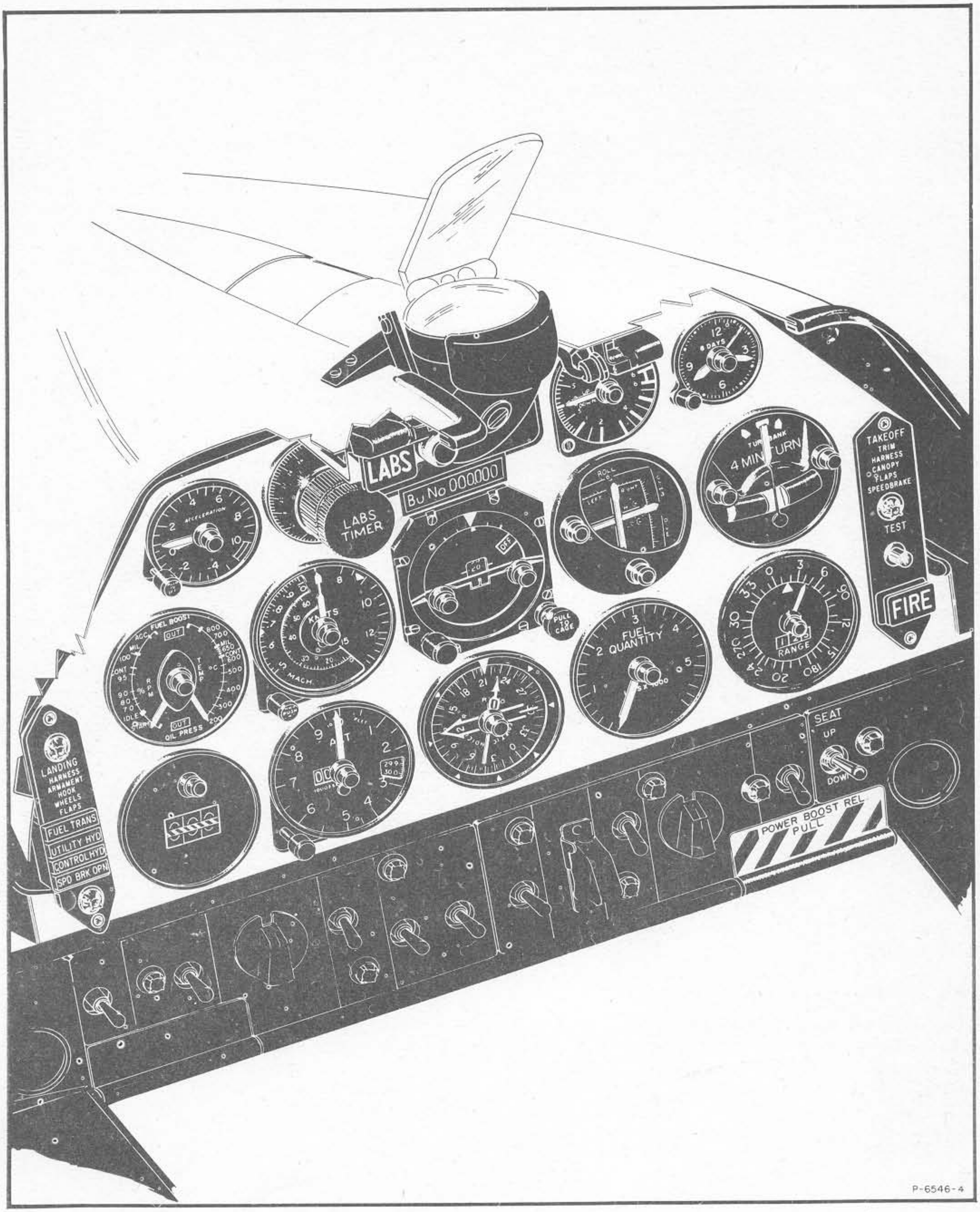
Figure 1-4. Cockpit — Instrument Panel (Sheet 1)



Group Bb, Bc, Bd, Ca, Cb and Cc Airplanes
Figure 1-4. Cockpit — Instrument Panel (Sheet 2)



Group Cd and D Airplanes
Figure 1-4. Cockpit — Instrument Panel (Sheet 3)



Group E Airplanes

Figure 1-4. Cockpit — Instrument Panel (Sheet 4)

IGNITION

Components of the ignition system are: a throttle lever switch, actuated by movement of the throttle from "OFF" to "IDLE"; an ignition generator; a dual ignition unit; a rectifier and filter assembly; and two igniter plugs. As the engine turns during starting, the ignition generator charges the dual ignition unit. When the throttle is advanced to "IDLE", the throttle ignition switch is opened, causing the dual ignition unit to discharge through the igniter plugs, thus igniting the primer fuel which began to flow when the throttle was opened. At 1900 ± 50 rpm, a centrifugal switch in the ignition generator closes, stopping ignition and closing the fuel priming valve.

AIR START SWITCH. The AIR START switch (19, figure 1-3), outboard of the throttle, is used to provide ignition during inflight starting of the engine. When the switch is moved to "ON" and the throttle is in the "IDLE" position, ignition is supplied through the igniter plugs. The "OFF" position discontinues the ignition cycle. The "OFF" position of the switch may be selected manually, or automatically simply by advancing the throttle to the normal operating range. Refer to SECTION III for air starting procedure.

OIL SYSTEM

The aircraft oil system is part of the engine and is automatic in operation whenever the engine is running. An oil tank (3, figure 1-2), located on the upper right side of the engine compressor housing, contains approximately three gallons of usable oil. A rotating supply tube within the oil tank furnishes oil during normal and inverted flight. An engine-driven gear-type pump and two metering pumps supply oil to all main bearings and accessory drives. A scavenge pump in the accessory gear box returns the oil, which has passed through a strainer, to the oil tank for re-circulation through the system. Pressure from the engine-driven gear pump is displayed in the OIL PRESS window of the engine performance indicator by the words "OUT", "NORM", and "HIGH". Normal oil consumption is approximately three-tenths gallon per hour. See figure 1-11 for oil specification.

FUEL SYSTEM

The engine fuel supply is carried in two tanks containing a combined total of 810 U.S. gallons. These tanks may be serviced by means of two gravity fuel tank filler caps or a single point pressure fueling system.[1] Three drop tanks may be carried externally to increase the total fuel quantity to 1410 U.S. gallons. A fuel transfer system is provided which transfers fuel through a piping system by means of tank pressurization and a fuel transfer pump. All fuel is delivered to the fuselage tank containing an electrically driven fuel boost pump which delivers the fuel under pressure to the engine driven fuel pump. A manual fuel supply shut-off valve control[2] is provided in the cockpit. Refer to figure 1-11 for fuel grades and specifications of recommended and alternate fuels. Refer to figure 1-7 for a schematic presentation of the fuel system.

FUEL TANKS

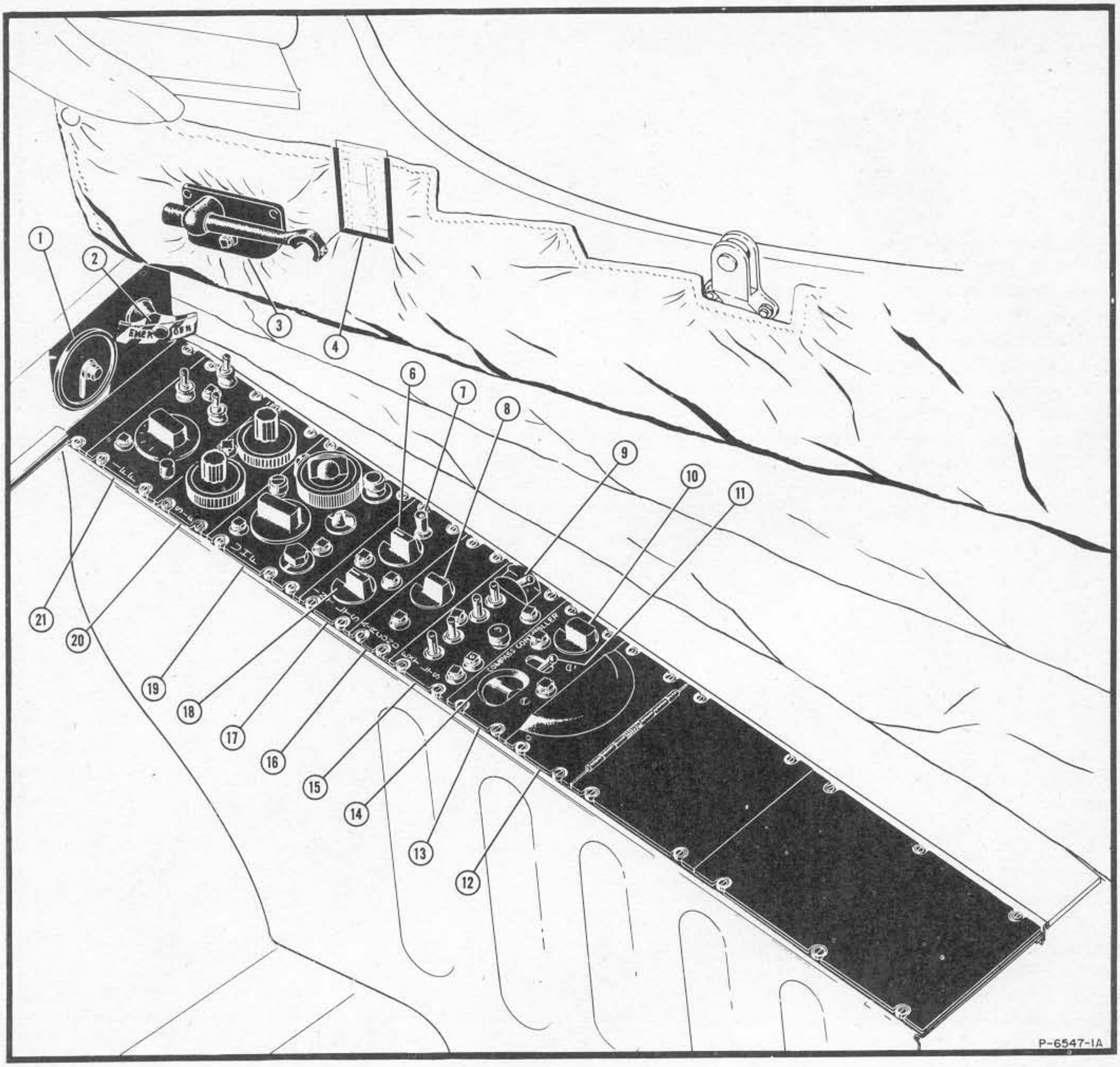
INTERNAL TANKS. The two fuel tanks provided in the airplane are comprised of an integral wing tank and a self-sealing type fuselage tank mounted between the cockpit and the engine bay. The fuselage tank contains the control valve, for regulation of transfer fuel flow, and the fuel boost pump which delivers fuel to the engine. All fuel aboard the airplane is transferred to this tank before delivery to the engine. The integral wing tank contains a fuel transfer pump which transfers fuel from the wing tank to the fuselage tank. Both fuel tanks are vented. The vent system exit is located aft of the right main landing gear strut and is designed to provide a small amount of ram air pressure in the fuel vent system to reduce the amount of collapse of the self-sealing type fuselage tank as a result of negative internal pressure when partially full. Both tanks incorporate provisions for gravity filling, pressure fueling and de-fueling,[1] and water and sediment drainage. For information concerning total and usable fuel capacities of each tank, refer to figure 1-6.

DROP TANKS. Provisions are made for carrying drop tanks singly or in combination. The external stores racks will accommodate either a 150 gallon or 300 gallon drop tank on the center-line rack and 150 gallon drop tanks on the wing racks. All drop tanks are vented, and contain provisions for gravity fueling, pressure fueling,[1] and pressurization to effect fuel transfer to the integral wing tank at the option of the pilot. The drop tanks may be jettisoned electrically in the same manner as other droppable external stores. Refer to RELEASING BOMBS, Section IV.

FUEL TRANSFER

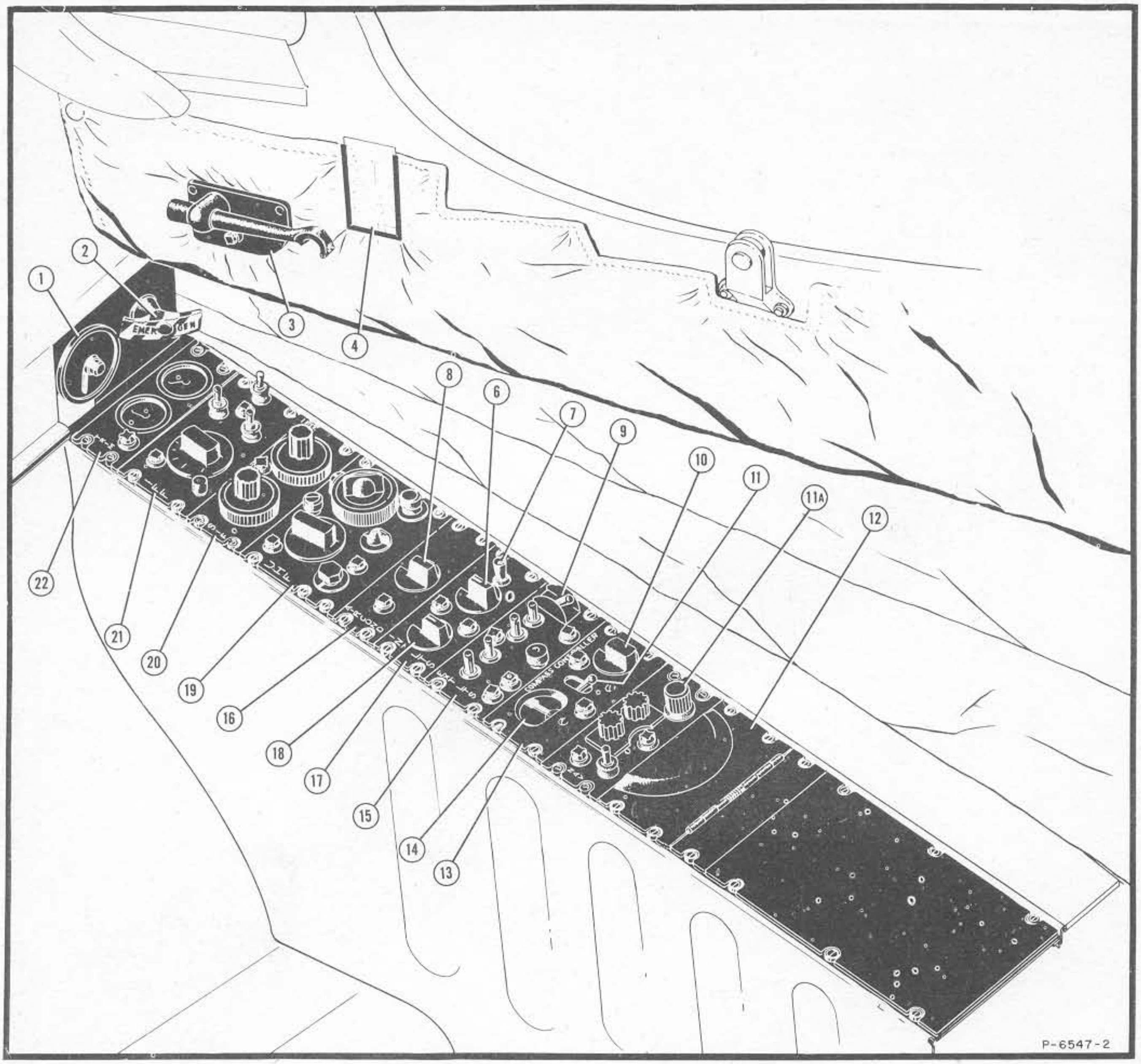
WING TANK TRANSFER. Placed internally in the wing tank is an air turbine driven transfer pump which transfers fuel from the wing tank to the fuselage tank. This pump utilizes engine compressor bleed air for power, and is operative whenever the engine is running. In lieu of the air turbine driven pump, some airplanes[3] are equipped with an electrically powered transfer pump which receives power from the aft monitored bus. In these airplanes, failure of the main generator will render the transfer pump inoperative. Since the wing tank transfer pump operates continuously, a float valve is placed in the fuselage tank to stop the flow of transfer fuel whenever the fuselage tank is full, in order to prevent transfer fuel being pumped overboard through the fuel vent system. A "FUEL TRANS" failure warning light[4] is provided on the left side of the instrument

[1]Group E Airplanes.
[2]Group C, D, and E Airplanes.
[3]Group Aa, Ad, Ba, and Bd Airplanes.
[4]Group Bd, C, and D Airplanes.



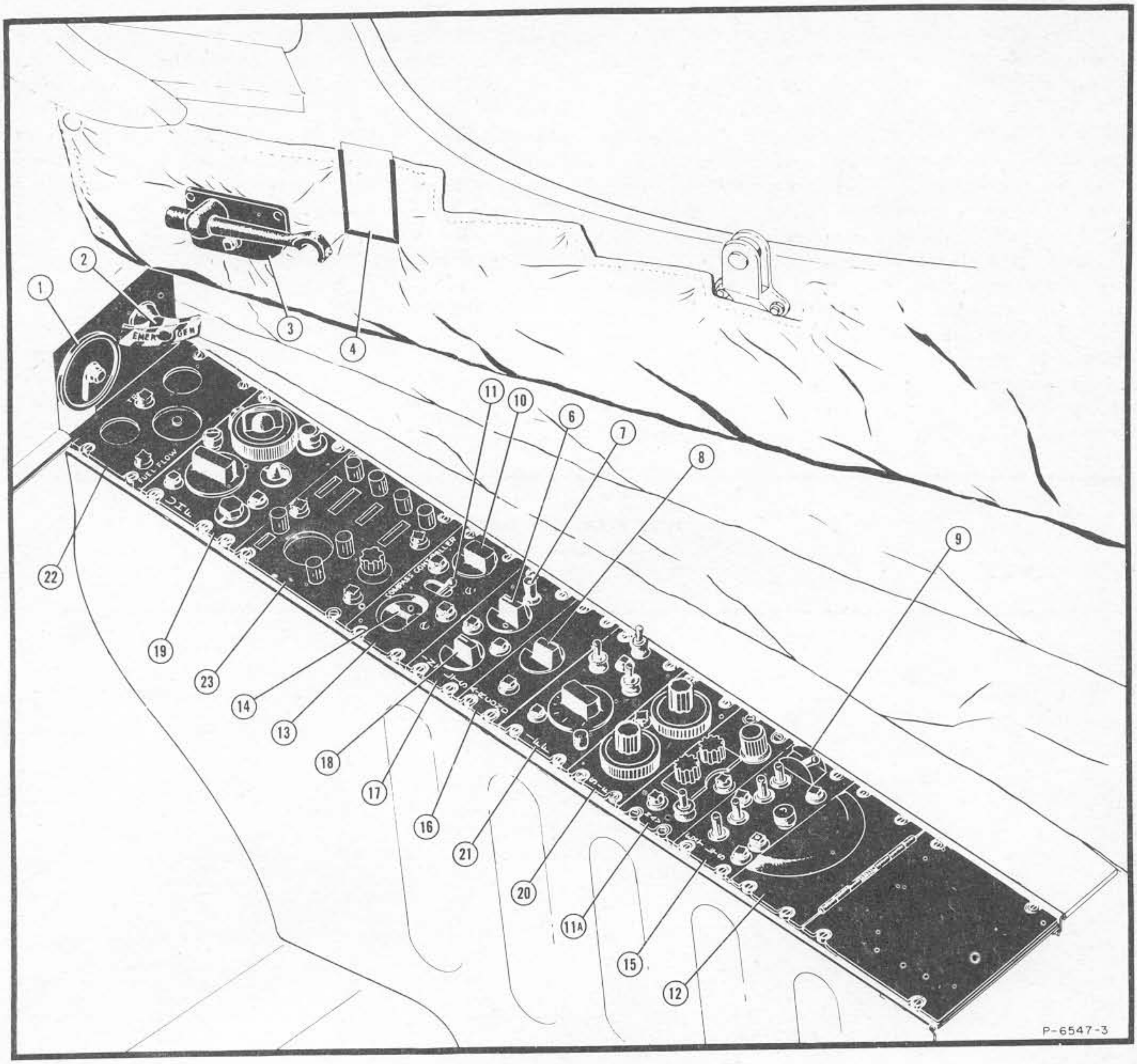
1. Cabin pressure altimeter
2. Emergency generator release handle
3. Arresting hook control
4. Standby compass correction card
5. Deleted
6. Console lights switch
7. Floodlight switch
8. Air conditioning control switch
9. Taxi light switch
10. SET HEADING FREE GYRO control
11. SLAVED GYRO-FREE GYRO selector switch
12. Spare lamps receptacle
13. Compass controller panel
14. Compass heading synchro. meter
15. Exterior lights control panel
16. Air conditioning control panel
17. Interior lights control panel
18. Instrument lights switch
19. AN/ARC-27A UHF radio control panel
20. AN/APA-89 SIF control panel
21. AN/APX-6B IFF radar control panel

Group A, B, Ca, Cb and Cc Airplanes
Figure 1-5. Cockpit — Right Console (Sheet 1)



1. Cabin pressure altimeter
2. Emergency generator release handle
3. Arresting hook control
4. Standby compass correction card
5. Deleted
6. Consoles lights switch
7. Floodlight switch
8. Air conditioning control switch
9. Taxi light switch
10. SET HEADING FREE GYRO control
11. SLAVED GYRO-FREE GYRO selector switch
11A. AN/ARN-21 NAV control panel
12. Spare lamps receptacle
13. Compass controller panel
14. Compass heading synchronization meter
15. Exterior lights control panel
16. Air conditioning control panel
17. Interior lights control panel
18. Instrument lights switch
19. AN/ARC-27A UHF radio control panel
20. AN/APA-89 SIF control panel
21. AN/APX-6B IFF radar control panel
22. Trim position indicators panel

Group Cd and D Airplanes
Figure 1-5. Cockpit — Right Console (Sheet 2)



1. Cabin pressure altimeter
2. Emergency generator release handle
3. Arresting hook control
4. Standby compass correction card
5. Deleted
6. Consoles lights switch
7. Floodlights switch
8. Air conditioning control switch
9. Taxi light switch
10. SET HEADING-FREE GYRO control
11. SLAVED GYRO-FREE GYRO selector switch

11A. AN/ARN-21 NAV control panel

12. Spare lamps receptacle
13. Compass controller panel
14. Compass heading synchronization meter
15. Exterior lights control panel
16. Air conditioning control panel
17. Interior lights control panel
18. Instrument lights switch
19. AN/ARC-27A UHF radio control panel
20. AN/APA-89 SIF control panel
21. AN/APX-6B IFF radar control panel
22. Trim position indicators and fuel flow meter
23. Automatic dead reckoning control panel

Group E Airplanes

Figure 1-5. Cockpit — Right Console (Sheet 3)

panel and will illuminate when wing tank fuel transfer pressure drops below 2 ($+\frac{1}{4}-\frac{1}{8}$) psi, indicating fuel transfer pump failure or wing tank fuel depletion.

DROP TANK TRANSFER. Fuel transfer from the drop tanks to the integral wing tank is effected by means of drop tank pressurization. Placing the DROP TANKS switch (4, figure 1-3) on the ENGINE control panel at "PRESS" opens a solenoid operated air shut-off valve which directs engine compressor bleed air to the drop tanks. Once pressurized, the flow of fuel from the drop tanks to the wing tank is controlled by float valves[2] or drop tank solenoid pilot valves[3] in the wing tank, which stop the transfer of fuel when the wing tank is full or allow it to continue when space is available. Placing the DROP TANKS switch at "OFF" energizes the drop tank air shut-off valve, thereby closing the valve and discontinuing transfer of fuel from the drop tanks. If electrical failure occurs, the drop tank air shut-off valve is automatically opened, providing immediate and automatic transfer of drop tanks fuel as wing tank space permits. To prevent drop tanks[1] 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 pressurizing air is introduced.

FUEL BOOST PUMP

An electrically driven fuel boost pump, powered by the

[1] Group E Airplanes.
[2] Group A, B, C, and D Airplanes.
[3] Group E Airplanes.

FUEL QUANTITY DATA
(GALLONS)

TANKS	USABLE FUEL	UNUSABLE FUEL — LEVEL FLIGHT	EXPANSION SPACE	TOTAL VOLUME
INTEGRAL WING	570	6	9	585
FUSELAGE	240	0	0	240
LH DROP (EXTERNAL AUXILIARY) AERO IA	*	*	*	150
CENTER DROP (EXTERNAL AUXILIARY) AERO IA (SMALL) OR AERO IA (LARGE)	* *	* *	* *	150 300
RH DROP (EXTERNAL AUXILIARY) AERO IA	*	*	*	150

REMARKS:

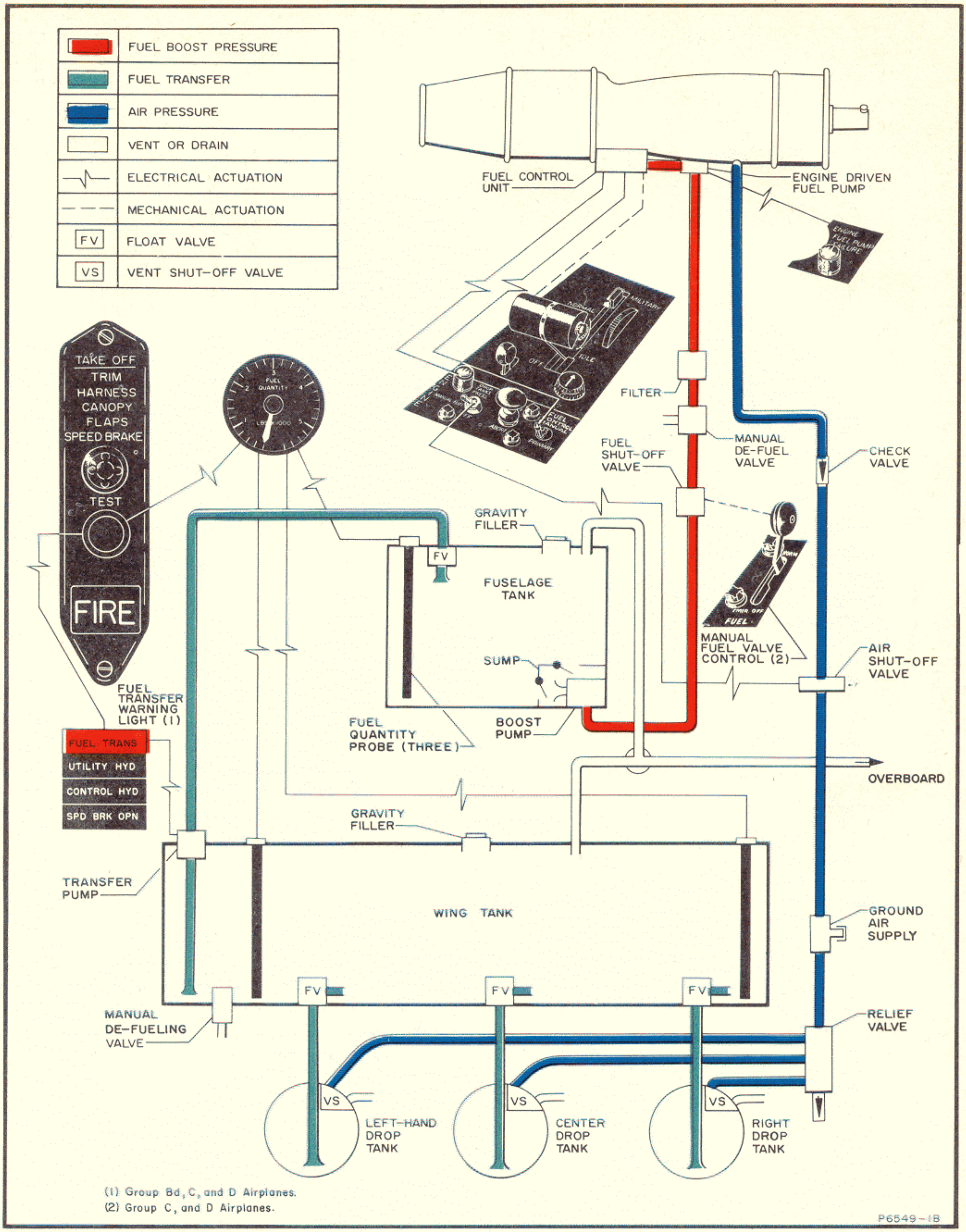
USABLE FUEL TOTALS

Fuselage and wing	810
Fuselage, wing, and small center drop	960
Fuselage, wing, and large center drop	1110
Fuselage, wing, l.h. drop, and r.h. drop	1110
Fuselage, wing, l.h. drop, r.h. drop, and small center drop	1260
Fuselage, wing, l.h. drop, r.h. drop, and large center drop	1410

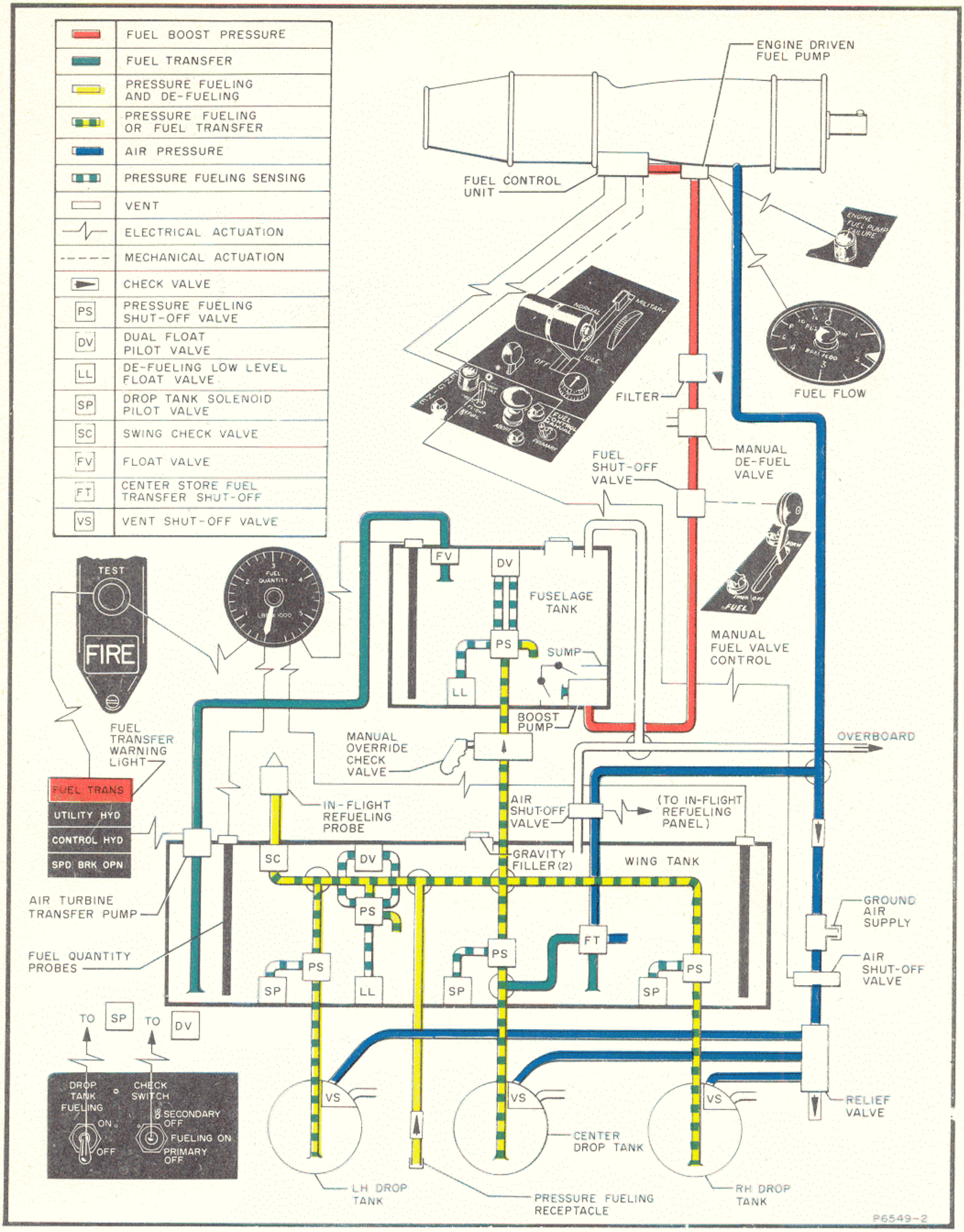
* Information to be submitted when available

P 6548-IA

Figure 1-6. Fuel Quantity Data



Group A, B, C and D Airplanes
Figure 1-7. Fuel System (Sheet 1)



Group E Airplanes

Figure 1-7. Fuel System (Sheet 2)

TO BE ADDED

Group E Airplanes
Figure 1-7A. In-Flight Refueling Store

aft monitored bus, is submerged in the fuselage tank sump. The fuselage tank sump incorporates flapper valves which act to keep the boost pump fuel inlet supplied with fuel at all airplane attitudes, including diving flight and negative "g" or inverted flight of short duration. Operation of the fuel boost pump is automatic as it is operative whenever the aircraft electrical system is energized by the main generator or external power. In the event of main generator failure the fuel boost pump will be inoperative, however, the fuselage fuel tank is so situated that gravity feed will provide the engine driven fuel pump a supply of fuel.

FUEL BOOST PRESSURE INDICATOR. Fuel boost pressure is shown in the FUEL BOOST window of the engine performance indicator (figure 5-1). The indicator is powered by the 28 volt d-c bus and has two positions, "NORM" and "OUT." The "NORM" position indicates that fuel boost pressure is above 4 psi, and the "OUT" position indicates that pressure is below this value.

MANUAL FUEL VALVE CONTROL. The fuel system incorporates a manually operated emergency FUEL valve control[(1)] (2A, figure 1-3) located outboard of the left hand console. This manual FUEL valve control has two positions, "NORMAL" and "EMERG OFF." The "EMERG OFF" position of the control stops all fuel flow from the airplane fuel system to the engine fuel control system. In earlier aircraft[(2)] no cockpit control is provided and the fuel valve can be operated only when the aircraft is on the ground.

FUEL QUANTITY INDICATING SYSTEM
The fuel quantity indicating system is comprised of dielectric-type fuel quantity probes, fuel quantity indicator, low level switch, fuel quantity test switch, and associated wiring. The wing tank contains two fuel quantity probes and the fuselage tank contains one fuel quantity probe. These probes are wired into the fuel quantity indicator in such a manner as to indicate the total quantity of fuel remaining in both tanks whenever the fuselage tank contains more than 170 gallons (1105 pounds). The fuselage tank contains a low level switch located approximately one-third the distance down the length of the fuel quantity probe at about the 170 gallon level. If the fuel supply in the fuselage tank falls below this level due to malfunction of the wing tank transfer system, or failure or mismanagement of the drop tank transfer system, the low level switch causes the reading of any remaining wing tank transfer fuel to be dropped out, indicating to the pilot that approximately 170 gallons of usable fuel remains. During normal fuel system operation the fuel gage will show a gradual decrease in fuel quantity indication, but if fuel transfer pump failure should occur, there would be a sudden drop in fuel quantity indication as soon as the fuel in the fuselage tank is depleted to the pre-determined 170 gallon level. The system does not show drop tank fuel quantity.

FUEL QUANTITY INDICATOR. The fuel quantity indicator (figure 1-4), located on the instrument panel, indicates the quantity of fuel available, in pounds. Readings of the indicator must be multiplied by 1000 to obtain the correct values. The fuel quantity indicator does not reflect the fuel available in the drop tanks when carried.

Note

A given volume of fuel will vary in weight, depending on its density, and although the indicating 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.

TEST SWITCH. The fuel quantity indicating circuit may be tested by pushing the multi-purpose TEST switch (figure 1-4) on the TAKEOFF check list panel. The fuel quantity indicating circuit is powered by the monitored primary bus. When the TEST switch is depressed, the fuel quantity indicator pointer will move in a counter-clockwise direction. When the switch is released, the pointer will return to the original indication if all units of the fuel quantity indicating circuit are functioning properly. In addition, when the test switch is depressed, the FIRE, FUEL TRANS,[(3)] UTILITY HYD, CONTROL HYD,[(4)] and SPD BRK OPN warning lights will illuminate to indicate that the respective circuits and light bulbs are operative.

PRESSURE FUELING AND DE-FUELING SYSTEM[(4)]

The pressure fueling system is designed to permit fueling at a rate of 200 gallons per minute through a single point pressure fueling receptacle, located at the trailing edge of the wing just inside the aft engine compartment access door. The system may be de-fueled through the same receptacle at a rate of approximately 100 gallons a minute. It is necessary to plug in external electrical power to fuel or de-fuel the airplane as described in the following paragraphs.

FUELING. When fueling the wing and fuselage tanks, fuel pressure opens the fueling shut-off valve in each tank, allowing fuel to enter the tank and also to flow through the sensing lines to the dual float pilot valve. When the tank becomes full, the floats of the pilot valve close the valve, causing pressure to increase behind the diaphragm of the shut-off valve and close it. This stops the flow of fuel into the tank. A three position momentary-contact CHECK SWITCH is provided on the pressure fueling switch panel to check the operation of the dual float shut-off valves. Each dual float shut-off valve consists of 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. Moving the CHECK SWITCH to either the "PRIMARY OFF" or "SECONDARY

(1) Group C, D, and E Airplanes.
(2) Group A and B Airplanes.
(3) Group Bd, C, D and E Airplanes.
(4) Group E Airplanes.

OFF" position causes solenoids to raise the respective float valve 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.

When drop tanks are installed, and it is desired to fuel the drop tanks by means of the pressure fueling system, it is necessary to place the DROP TANK FUELING switch on the pressure fueling switch panel at "ON." This action energizes the normally closed solenoid pilot valve, thus permitting fuel pressure to open the drop tanks shut-off valves and subsequently flow to the drop tanks. Under these conditions, fuel also flows through the sensing lines to the drop tanks solenoid pilot valves. As each drop tank becomes full, a float valve in the tank rises, breaking the electrical circuit to the energized solenoid pilot valve, causing the pilot valve to close and pressure to build up behind the diaphragm of the shut-off valve which then also closes, discontinuing the pressure fueling to that tank.

DE-FUELING. To de-fuel the integral wing tank requires no procedure other than connecting the de-fueling hose to the pressure fueling receptacle. To de-fuel the fuselage tank it is necessary to operate the manual override check valve between the wing and fuselage tanks. When the de-fueling operation is begun, negative pressure in the pressure fueling shut-off valves will open the valves and allow the fuel to be removed. When either the fuselage or wing tank becomes empty, the de-fueling low level float valve opens, increasing the pressure behind the diaphragm of the shut-off valve, causing the valve to close. This prevents air from entering the de-fueling line and breaking the siphon when one tank empties ahead of the other.

To de-fuel the drop tanks through the pressure fueling system, it is necessary to connect a source of air pressure to the capped tee in the drop tanks pressurizing system. This will transfer the drop tanks fuel into the integral wing tank where it may be removed through the de-fueling receptacle.

PRESSURE FUELING SWITCH PANEL. The pressure fueling switch panel is located on the left side of the aft engine access compartment just inside the access door. This panel has two switches: the CHECK SWITCH, and the DROP TANK FUELING switch. The CHECK SWITCH is used to test the operation of the dual float pressure fueling pilot valves, and has three positions, "PRIMARY OFF," "FUELING ON," and "SECONDARY OFF." The DROP TANKS FUELING switch has two positions, "ON" and "OFF." The "ON" position of the DROP TANKS FUELING switch energizes the drop tanks solenoid pilot valves, permitting pressure fueling of the drop tanks.

IN-FLIGHT RE-FUELING SYSTEM

To be added in a subsequent revision.

ELECTRICAL POWER SUPPLY SYSTEM

Electrical power is normally supplied by an engine-driven main generator which furnishes 115/200 volts, 3-phase 400 cycle, constant frequency a-c power, and, through a transformer-rectifier, 28 volts d-c power. No d-c generator or battery is provided. An additional transformer modifies generator power to 26 volts a-c power for the operation of certain equipment. The aft monitored, forward monitored, primary, monitored primary[1], d-c, armament, and 26 volt a-c busses, serve to distribute power to the various electrical units. An air-stream-operated emergency generator provides 115/200 volts, 3-phase, 400 cycle a-c power to essential equipment in the event of main generator or engine failure, External power can be used to energize the system through an external power receptacle located in the left wing horn. Operation of the electrical system is completely automatic with the exception of the emergency generator, which must be activated by the pilot upon failure of the main generator. Refer to Section III for emergency operation of the electrical system, and see figure 1-8 for a schematic presentation of electrical power distribution.

MAIN GENERATOR

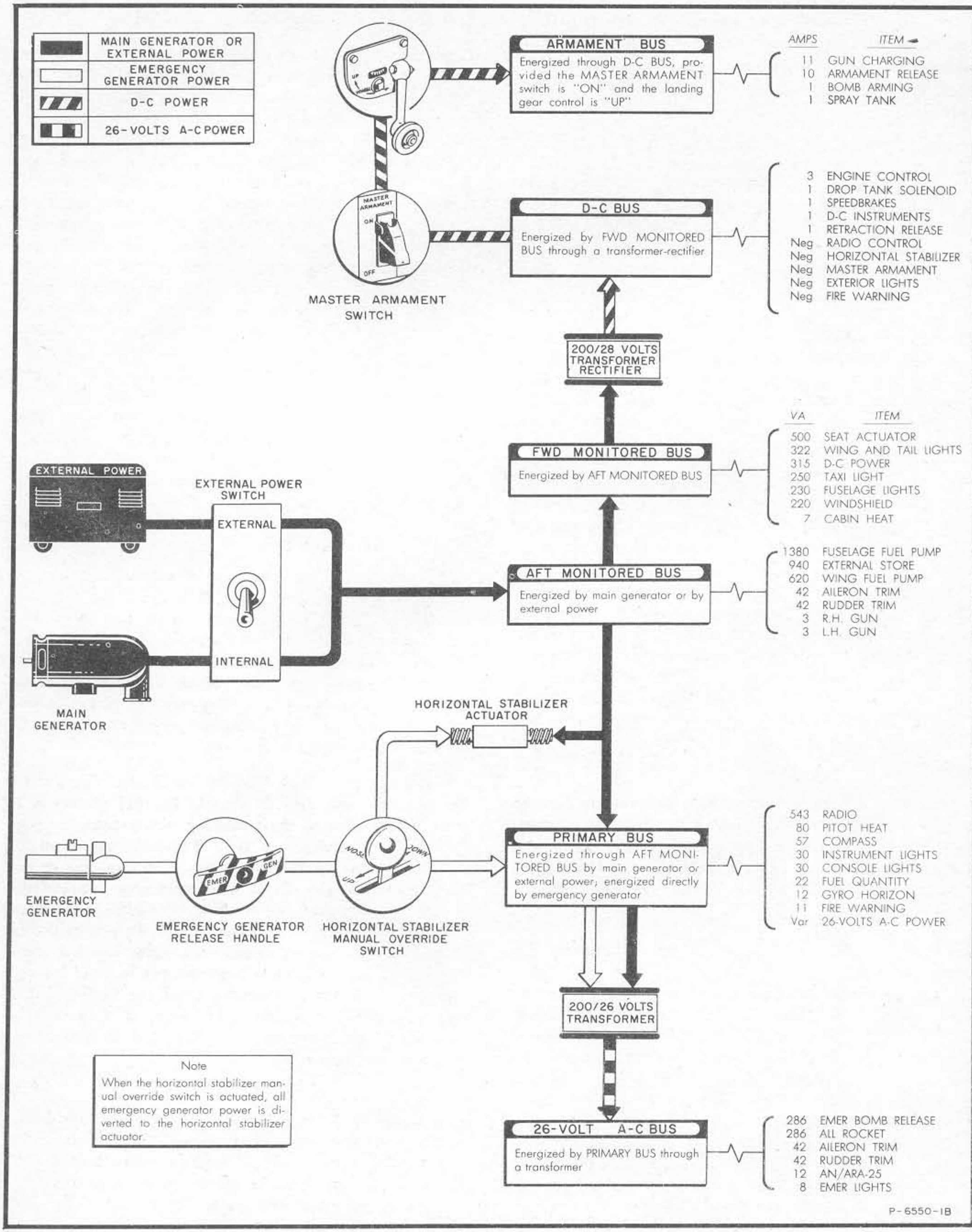
The main generator, rated at 9 kilovolt-amperes, is driven at a constant speed of 4000 rpm by the engine through a hydraulic ball-pump drive assembly. The drive assembly utilizes a separate hydraulic reservoir (figure 1-11) located in the forward engine compartment. The main generator, which delivers full rated capacity at engine idling speed, supplies power to the aft monitored bus, which further distributes power to other electrical busses in the system.

GENERATOR FIELD FLASH SWITCH. For use by the ground crew, a GEN FIELD FLASH SWITCH is installed adjacent to the hydraulic system pressure gage in the right hand wheel well. The momentary-contact switch is provided for the purpose of restoring or reversing the polarity of the main generator exciter field circuit by the process of "flashing the field". Power for the flashing circuit is supplied by the d-c power source in the external starter compressor store, which is connected to the circuit when the starter is inserted into its receptacle. Momentary actuation of the GEN FIELD FLASH SWITCH allows the current to flow briefly through the generator exciter field circuit, thus restoring it to a normal condition.

Note

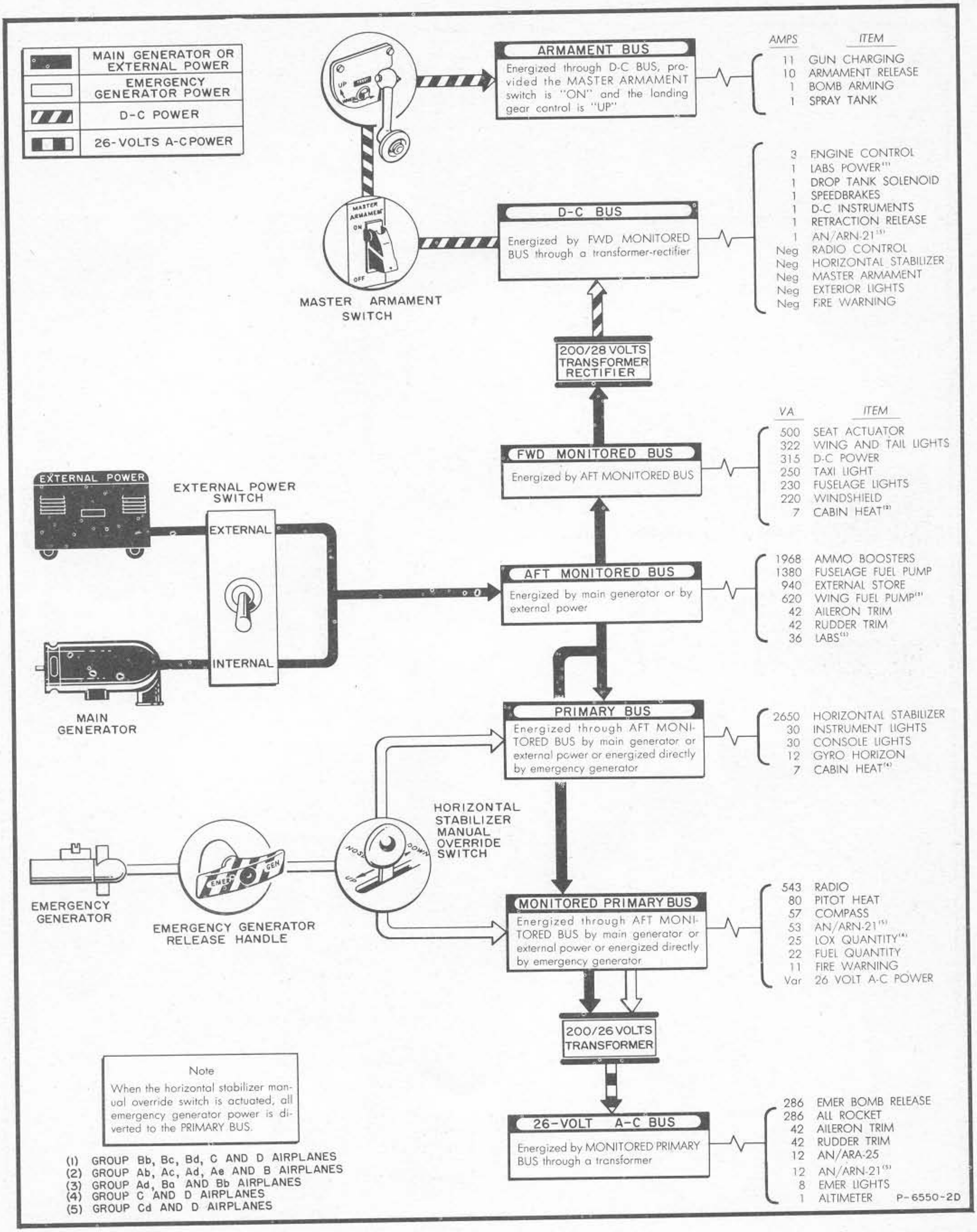
> A test unit provided by the ground crew should be plugged into the receptacle on the fuse panel in the nose wheel well during each start, to ascertain that the generator is operating within the prescribed limits.

EXTERNAL POWER SWITCH. The external power switch, located adjacent to the external power receptacle, must be placed at the "INTERNAL" position by the

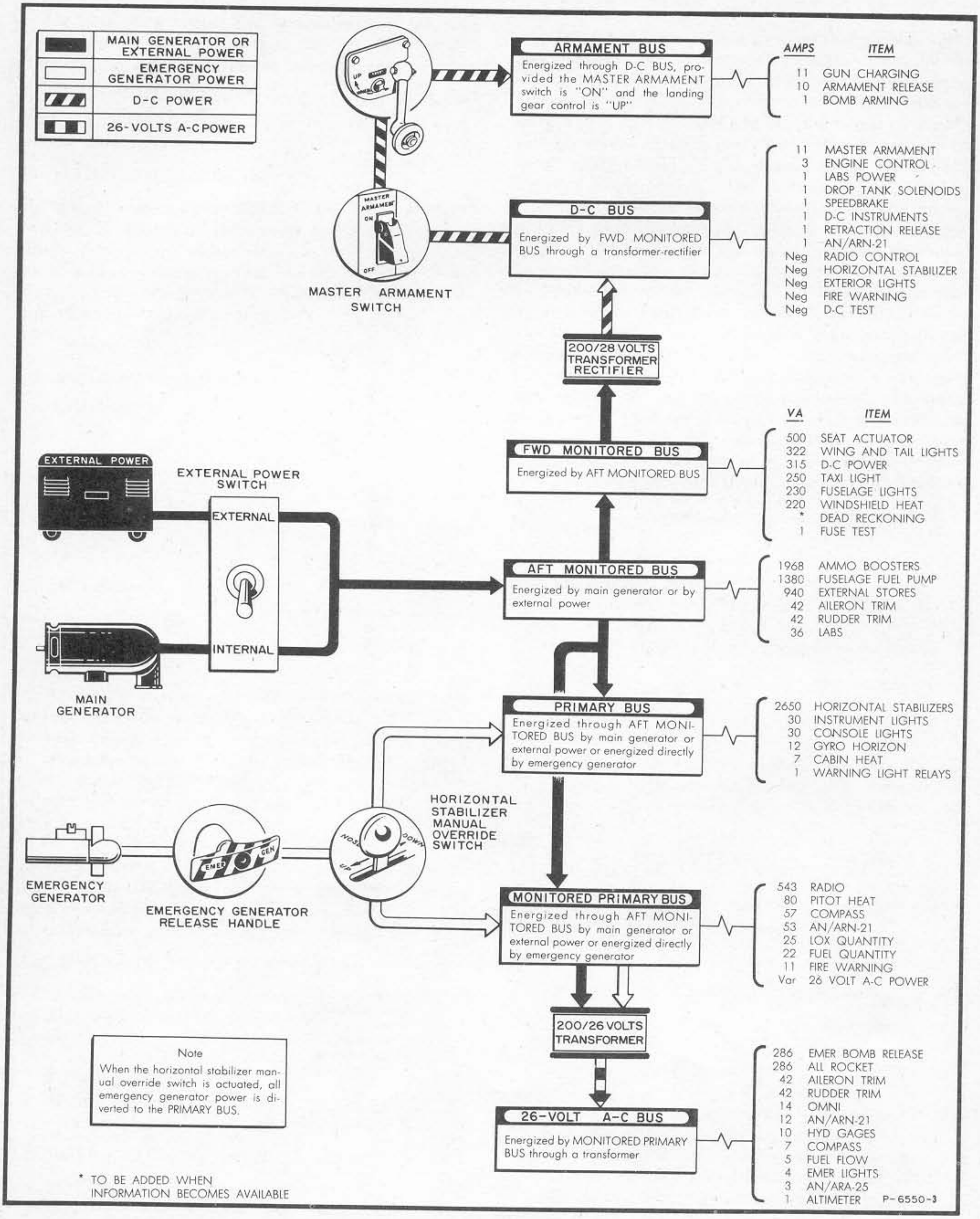


Group Aa Airplanes

Figure 1-8. Electrical System (Sheet 1)



Group Ab, Ac, Ad, B, C and D Airplanes
Figure 1-8. Electrical System (Sheet 2)



Group E Airplanes

Figure 1-8. Electrical System (Sheet 3)

ground crew before the main generator can energize the electrical system. When in the "EXTERNAL" position, the aft monitored bus is disconnected from the main generator and is connected to the external power receptacle, in order that power from an external source may be applied to the system. The external power receptacle door cannot be closed when the switch is in the "EXTERNAL" position.

EMERGENCY GENERATOR

The emergency generator, rated at 1.7 kilovolt-amperes[1] (1 KVA in early aircraft[2]), is normally carried internally in a compartment in the lower right side of the forward fuselage. When released into the airstream, a variable pitch propeller governs the speed of the generator at approximately 12,000 rpm to provide 400 cycle constant frequency power to the primary and monitored primary bus. In early aircraft[2] no monitored primary bus is provided. When the emergency generator is extended there is no d-c electrical power available for any circuits, except from a small transformer-rectifier which furnishes power to the S-2 Compass, and the landing gear warning and approach light relays.

EMERGENCY GENERATOR RELEASE HANDLE. The EMER GEN release handle (2, figure 1-5), on the extreme right corner of the armament panel, provides the only control of the electrical system that is available to the pilot. When the handle is pulled, the emergency generator drops into the airstream, disrupting the exciter field circuit of the main generator, rendering that generator inoperative, and connecting the primary, and monitored primary bus to the emergency generator. Once the emergency generator is extended, there is no way to retract it to the normal stowed position while in flight.

A-C POWER DISTRIBUTION

Power from the main generator flows to a voltage regulator which maintains a constant voltage output from the main generator by varying the current in the generator exciter field. From the voltage regulator, the power moves through the "INTERNAL" position of the external power switch to the aft monitored bus. The aft monitored bus distributes power to the electrical units connected to it, to the forward monitored bus, to the primary bus, and to the monitored primary bus. The monitored primary bus supplies power to a transformer which reduces the voltage to 26 volts to power the 26 volt a-c bus. In early aircraft[2] the transformer is powered by the primary bus. Each of the various busses in the electrical system distributes power, through protective fuses, to all electrical units connected directly to them.

EMERGENCY A-C POWER. Extending the emergency generator into the airstream breaks the main generator exciter field circuit, rendering the main generator inoperative, and transfers the primary bus and monitored primary bus from the aft monitored bus to the emergency generator. Thus, only those units powered from the primary bus, the monitored primary bus, and the 26-volts a-c bus will be operable when the emergency generator is operating. If, during emergency generator operation, it becomes necessary to adjust the horizontal stabilizer, the entire output of the emergency generator, except power to the cockpit lights,[1] gyro horizon,[1] cabin temperature control,[3] and warning light relays[4] on later aircraft, must be diverted to the horizontal stabilizer actuator. This can be accomplished only by using the horizontal stabilizer manual control on the left console. Upon release of the control, the output of the emergency generator is again directed to the bus system.

D-C POWER DISTRIBUTION

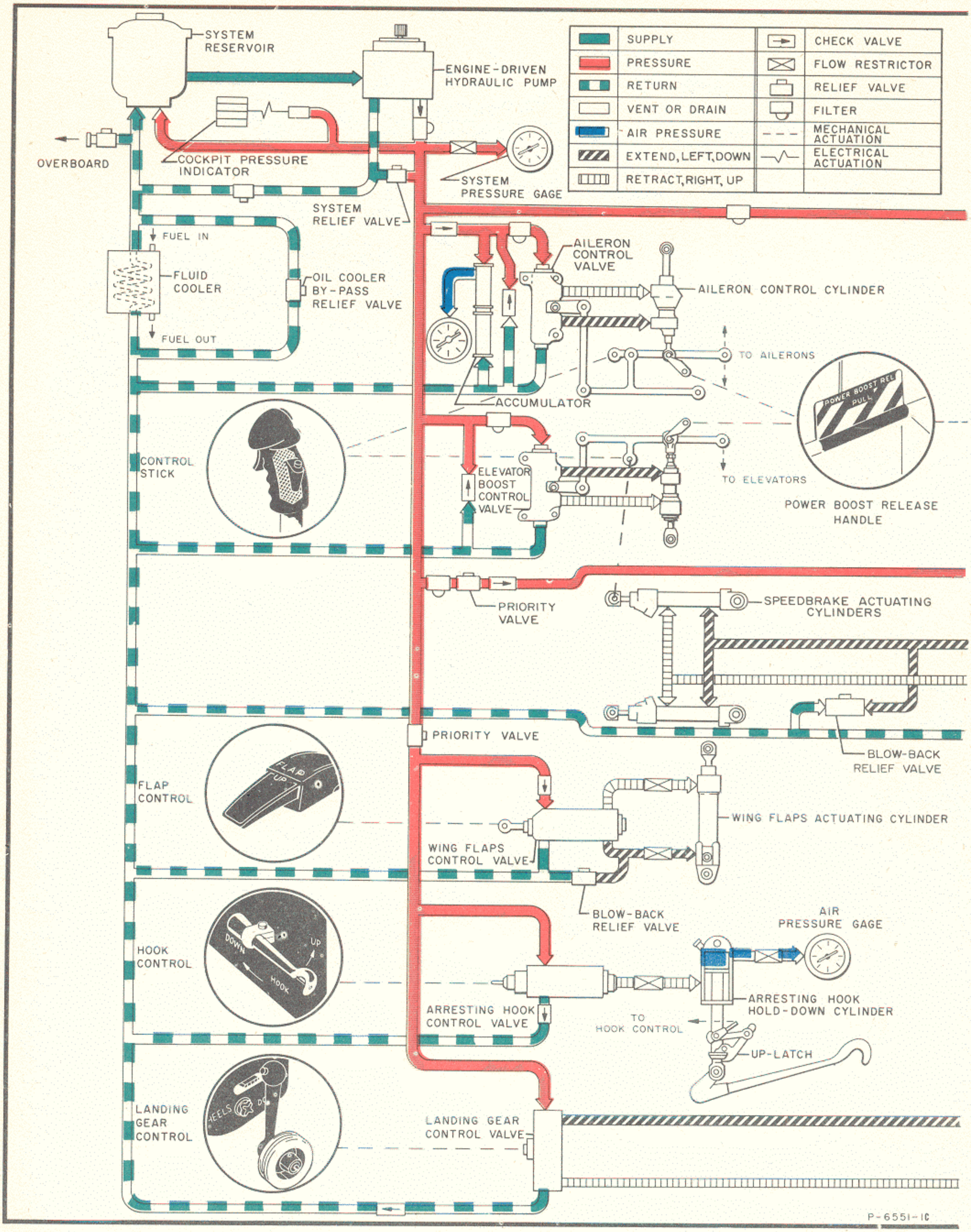
The forward monitored bus supplies 115/200 volts, 3-phase, 400 cycles a-c power to a transformer-rectifier which converts the a-c power to 28-volts d-c power to energize the d-c bus. The d-c bus further distributes power to connected electrical units and to the armament bus.

D-C BUS. The d-c bus is energized when the main generator is operating, or when external power is connected to the aircraft. The emergency generator will not energize the d-c bus.

ARMAMENT BUS. The armament bus receives power from the d-c bus, provided that the MASTER ARMAMENT switch is "ON" and the landing gear control is "UP". An armament safety switch, actuated by the "DOWN" position of the landing gear control, deenergizes the armament bus as a safety feature to prevent inadvertent firing of the guns or release of stores when the aircraft is on the ground or in the landing pattern with the wheels down. The armament bus may be energized on the ground by turning the MASTER ARMAMENT switch "ON" and depressing the ARMT. SAFE DISABLE SWITCH in the nose wheel well.

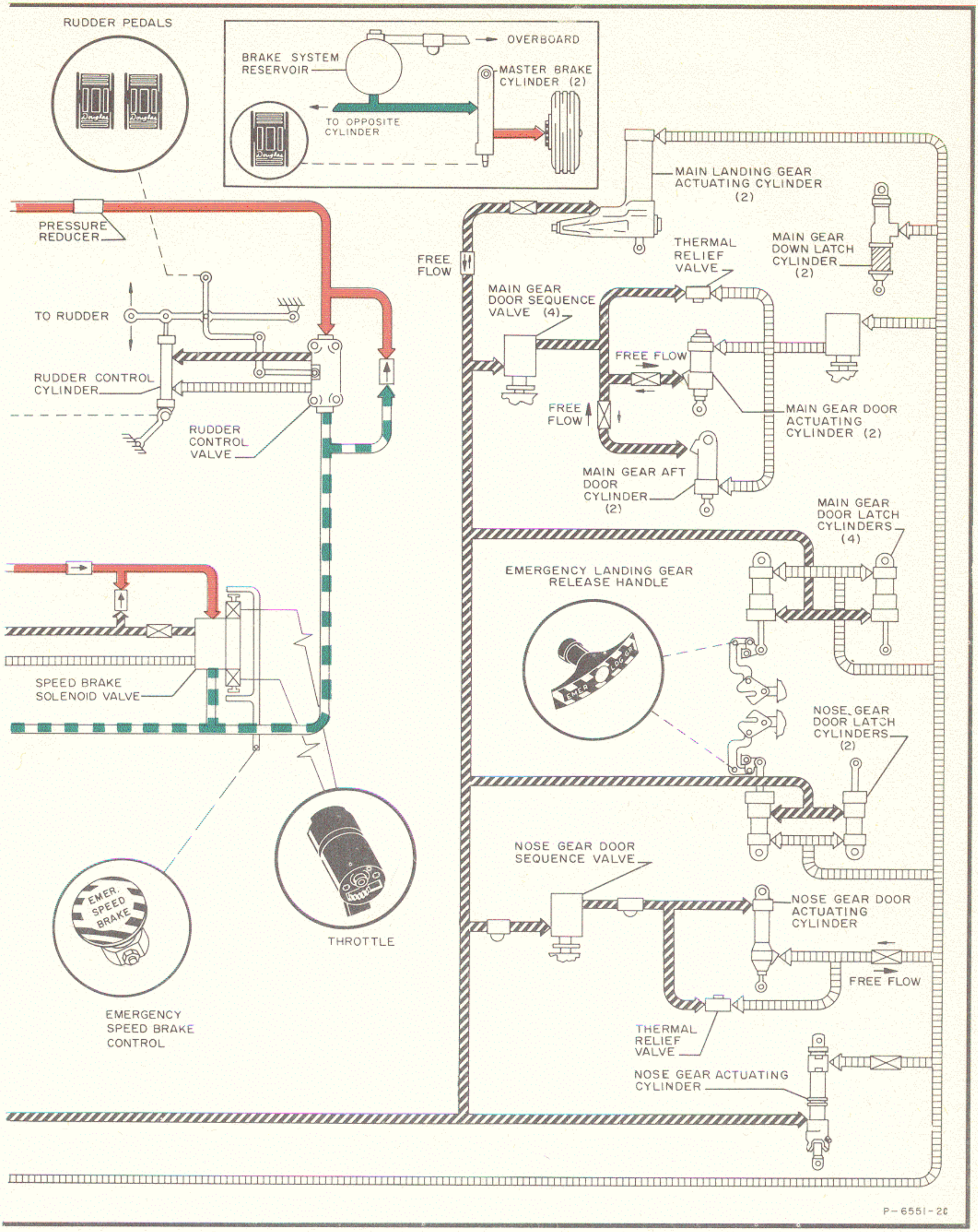
ARMAMENT SAFETY DISABLING SWITCH. The ARM SAFETY DISABLE SWITCH is located on the outboard side of the right-hand wheel well for ground testing of the armament circuit. In some airplanes[5] the ARMT. SAFE DISABLE SWITCH is located on the electrical TEST PANEL on the right side of the nose wheel well. When the MASTER ARMAMENT SWITCH is "ON" and the ARM SAFETY DISABLE SWITCH is momentarily depressed, an armament safety disabling relay is closed, allowing power to energize the armament bus. The relay will be held closed until the MASTER ARMAMENT switch is turned "OFF," or electrical power is disconnected from the aircraft. When this occurs, the armament safety feature is automatically reinstated.

[1] Group Ab, Ac, Ad, B, C and D Airplanes.
[2] Group Aa Airplanes.
[3] Group C and D Airplanes.
[4] Group E Airplanes.
[5] Group A, Ba, Bb, and Bc Airplanes.



Group A, B, C and D Airplanes

Figure 1-9. Hydraulic System (Sheet 1)



Group A, B, C and D Airplanes

Figure 1-9. Hydraulic System (Sheet 2)

FUSE PANELS

All electrical circuits are protected by fuses in lieu of circuit breakers in order to save weight and to provide better wire protection. The fuses are located on two fuse panels, one in the nose wheel well, the other in the forward engine and accessories compartment. Both panels are inaccessible during flight and therefore should be checked for burned-out or improperly inserted fuses prior to each flight. Fuse testing provisions and spare fuses are provided on each fuse panel.

HYDRAULIC POWER SUPPLY SYSTEM[2]

A self-pressurizing reservoir, located aft of the fuselage fuel tank, contains 1.25 gallons of hydraulic fluid with additional expansion space for 0.6 gallons. An engine-driven pump of variable displacement supplies hydraulic fluid at a pressure of 3000 psi for the aileron power control, rudder power control, and elevator power boost systems, the landing gear, wing flaps, speed brakes, and for retraction of the arresting hook. Two priority valves are provided to maintain a minimum pressure of 1500 psi in the aileron, rudder, and elevator power boost controls during operation of the secondary systems. A main system relief valve opens at 3500 psi to prevent damage to the lines and equipment from excessive pressure. A hydraulic pressure warning light is provided in the cockpit, and for use of the ground crew, a gage is installed in the right-hand wheel well. As there is no auxiliary pump, hydraulic pressure is not available for ground operation unless the engine is running. Whenever the engine is rotating, regardless of speed, normal hydraulic pressure will be supplied to the system; however, at engine windmilling speeds, the rate of fluid flow through the system will be reduced and the time required for the hydraulically operated units to fully respond to the actuation of their individual controls will be increased. See figure 1-9, sheets 1 and 2, for a schematic presentation of the hydraulic system, and refer to Section III for emergency procedures to be followed in the event of hydraulic system failure.

HYDRAULIC POWER SUPPLY SYSTEM[3]

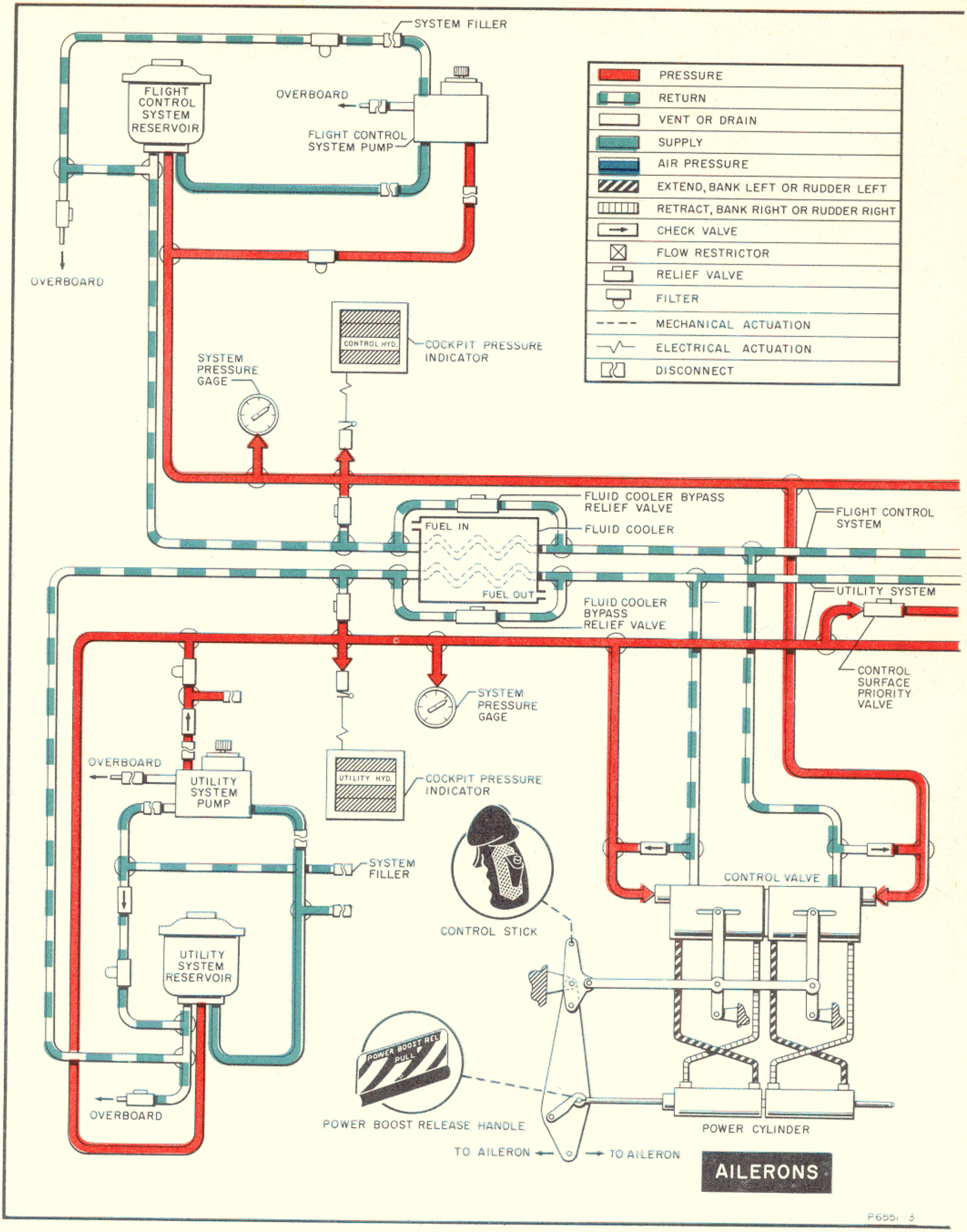
The tandem hydraulic system utilizes two self-pressurizing fluid reservoirs and two identical engine-driven variable displacement pumps. Both reservoirs are located in the upper right side of the fuselage over the center of the wing, with the flight control system reservoir just aft of the utility system reservoir. Capacity of the utility system reservoir is 1.21 gallons of hydraulic fluid, while the flight control system reservoir contains 0.30 gallons. Each system operates normally under a pressure of 3000 psi, and relief valves in each system open at 3650 psi to prevent damage to the lines and equipment that might be caused by excessive pressure. Tandem power cylinders are used in the aileron and elevator power controls, one side being operated by flight control system pressure, and the other side by utility system pressure. This arrangement allows the ailerons and elevators to be power-operated by either system in the event of failure of the other. During normal operation, the flight control system operates the rudder power control, as well as the aileron and elevator power controls. Utility hydraulic system pressure also operates the landing gear, wing flaps, speed brakes, and arresting hook. A hydraulic pressure warning light is provided in the cockpit for each of the two systems, and pressure in both systems may be checked by gages installed in the right-hand wheel well. There is no auxiliary pump, so no hydraulic pressure is available for ground operation unless the engine is running. Whenever the engine is rotating, regardless of speed, normal pressure will be supplied to both the flight control and the utility systems; however, at engine windmilling speeds fluid flow is somewhat reduced, and the time required for hydraulically operated units to fully respond will be increased. See figure 1-9, sheets 3, 4 and 5 for a schematic diagram of the tandem hydraulic system, and refer to Section III for emergency procedures to be followed in the event of failure of one or both of the hydraulic systems.

FLIGHT CONTROL SYSTEM

Flight control surfaces are conventional. To reduce control stick forces and increase maneuverability, separate hydraulic power controls are provided for operation of the ailerons and the rudder, while the elevator is provided with either a hydraulic power boost[2] or a hydraulic power control.[3] Conventional rudder pedals and an extendable control stick are installed in the cockpit. The rudder pedals are adjustable fore and aft by means of a lever on the inboard side of each pedal. The control stick may be extended to reduce the manual effort required to deflect the control surfaces during flight if the power control and boost systems should fail. Stick extension is accomplished by depressing a lever jut below the handgrip. The control stick may then be pulled up until the lever snaps into a detent in the control stick assembly, thus maintaining the stick in the extended position.

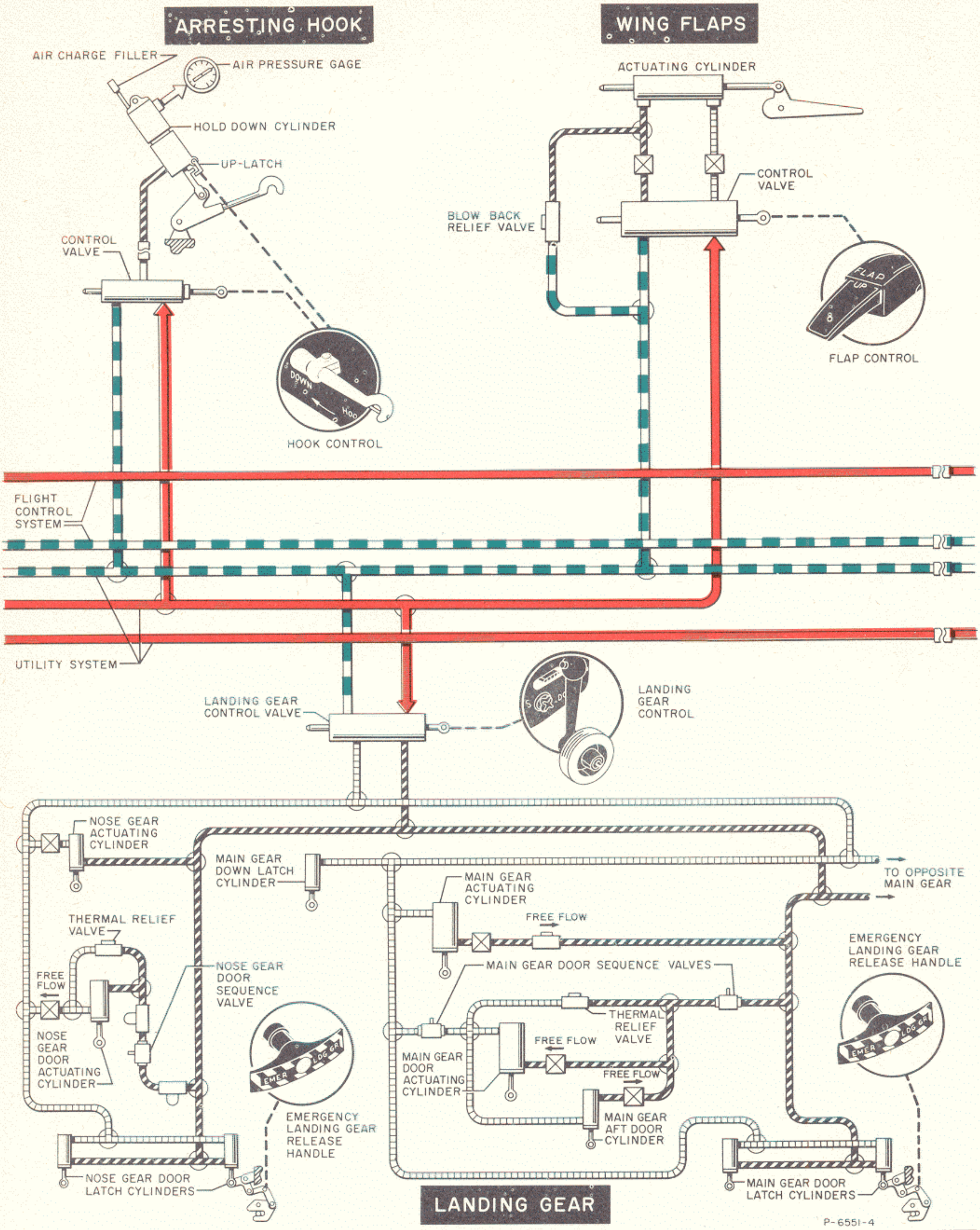
To eliminate aerodynamic "buzz" of the flight control surfaces at high airspeeds, independent hydraulic snubber systems are connected to the ailerons, rudder, and elevators. (Airplanes equipped with the tandem hydraulic system do not incorporate a snubber system for the elevator.) Each system consists of an accumulator, one-way restrictor, and a snubber cylinder assembly. The cylinder piston rod-end is attached to the control surface, and the cylinder barrel is anchored to the associated aircraft structure. An orifice in the piston controls the damping action by restricting the flow of fluid from one side of the piston to the other when the control surface is displaced. The degree of damping effect provided is proportional to the speed at which the con-

(1) Group A, Ba, Bb and Bc Airplanes.
(2) Group A, B, C and D Airplanes.
(3) Group E Airplanes.



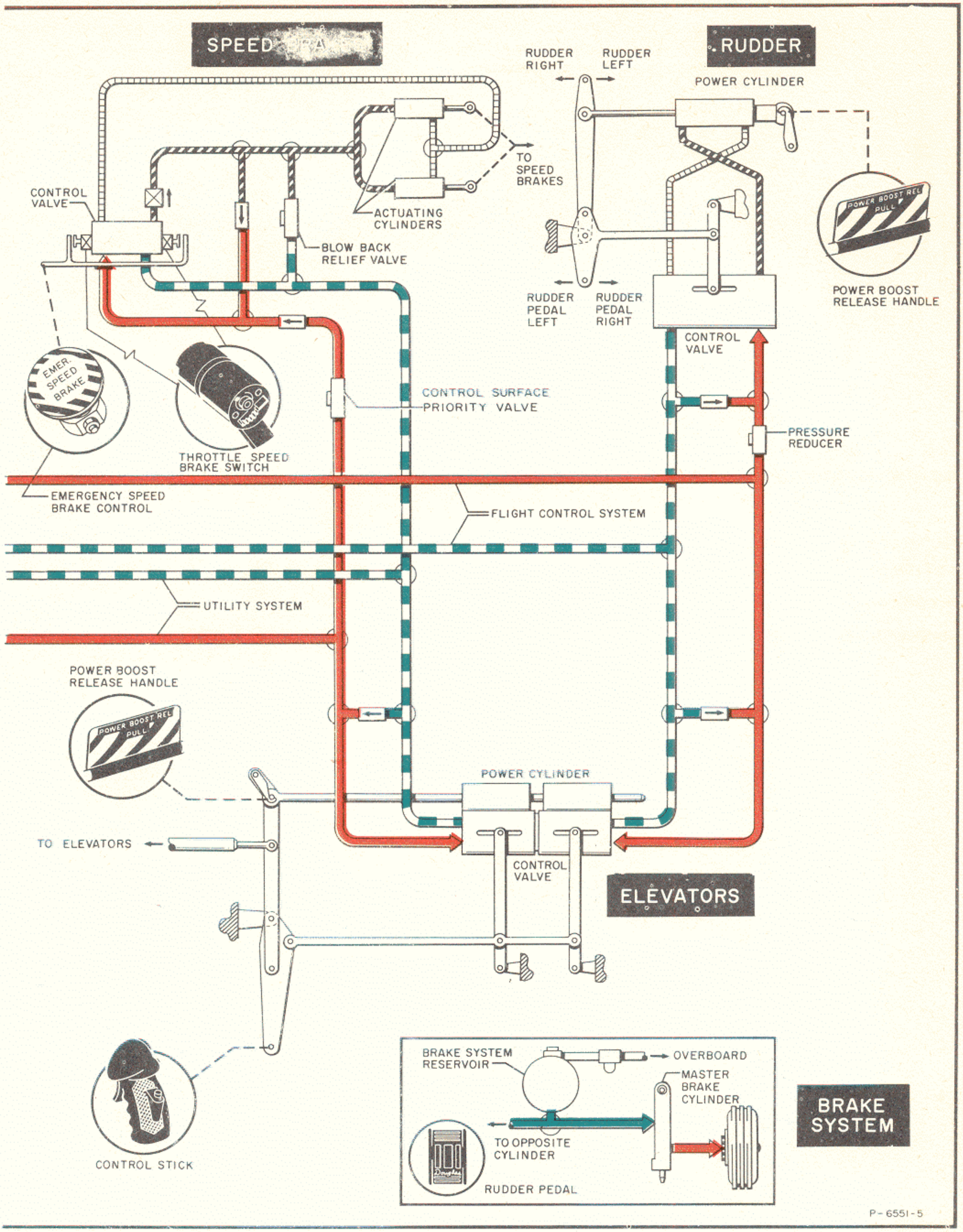
Group E Airplanes

Figure 1-9. Hydraulic System (Sheet 3)



Group E Airplanes

Figure 1-9. Hydraulic System (Sheet 4)



Group E Airplanes

Figure 1-9. Hydraulic System (Sheet 5)

trol surface is moved. Thus, when rapid displacement occurs, high damping action is provided; when the control surface is moved slowly, less damping action is supplied.

FLIGHT CONTROL POWER SYSTEMS (SINGLE HYDRAULIC SYSTEM)[1]

Hydraulic pressure for the aileron power control, rudder power control, and the elevator power boost is furnished by the main hydraulic system pump. These units consist basically of three separate control valves and power cylinders with associated mechanisms for operating the ailerons, rudder and elevators. The power cylinders receive pressure through the control valves which are actuated by movement of the control stick or rudder pedals. The elevator boost cylinder moves the connecting linkage to the elevators in the proper direction with a force corresponding to approximately 7 times the force applied to the control stick. The aileron and rudder power control cylinders are irreversible, and their output is independent of the manual effort required to displace the controls. When secondary systems are operated during flight, the two priority valves in the main hydraulic system prevent pressure to the flight controls from dropping below 1500 psi.

(TANDEM HYDRAULIC SYSTEM)[2]

The aileron and elevator power controls are furnished hydraulic pressure by the flight control and utility system pumps, while the rudder obtains hydraulic pressure from only the flight control system pump. These controls are made up of three separate sets of control valves and irreversible power cylinders for operating the ailerons, elevators, and rudder. Movement of the control stick or rudder pedals mechanically actuates the control valves, which in turn port fluid pressure to the power cylinders; mechanical linkages between the power cylinders and the control surfaces then displace the surface the desired direction and distance. Tandem control valves (for the aileron and elevators) are mounted side by side in one valve body. The power cylinders are also "double" in the same respect, except that the two pistons utilize one common piston rod. Flight control system pressure operates one side of each control valve and power cylinder, utility system pressure operates the other side, and since the two halves of each are connected mechanically it may be seen that the two sides operate simultaneously during normal operation. It is because of this tandem relationship that the aileron and elevator power controls will operate on either system alone, but the loss in maneuverability may be appreciable. Should the flight control system fail, the utility system will continue to operate the aileron and elevator power controls; likewise, if the utility system should fail the flight control system will operate the aileron and elevator controls.

AILERON CONTROL. Both ailerons are aerodynamically and statically balanced. Lateral movement of the control stick positions the aileron control valve so that hydraulic fluid at 3000 psi is ported to the aileron power cylinder. The aileron power cylinder operates push-pull tubes to the ailerons, causing the latter to be deflected in the desired direction. Because the aileron power control is irreversible, there is no feed-back to the pilot of air loads against the ailerons, therefore, artificial "feel" is induced by a spring bungee. The action of the spring bungee opposes the movement of the control stick.

AILERON TRIM SYSTEM. An electrically powered actuator moves the stick, power system linkages, and consequently the ailerons to the desired trim position by changing the neutral position of the aileron load feel and centering bungee. At the same time it positions a follow-up tab on the left aileron so that the airplane will remain approximately in trim whenever the power system is disconnected. In this event, the pilot continues to trim the ailerons in the same manner except that now the tab is positioning the surfaces. The trim actuator is controlled by movement of the trim switch (figure 4-6) on the stick grip to "LWD" (left wing down) or "RWD" (right wing down). As the trim actuator receives its power from the aft monitored bus, it will be inoperative when the emergency generator is in use. No indicator is provided to show the trim position of the ailerons and tab, but the control stick will be displaced from center to a new "neutral" position as the trim tab and ailerons are moved from their faired positions by the trim actuator.

ELEVATOR CONTROL. Fore and aft movement of the control stick moves a push rod attached to the elevator control valve, which ports hydraulic pressure to the elevator boost[1] (elevator power control)[2] cylinder. The cylinder then, through mechanical linkage, deflects the elevator surface as desired. The elevators are not equipped with trim tabs, as longitudinal trim is provided by a movable horizontal stabilizer. To provide additional nose-up trim, however, a bungee is installed in the elevator control system. This bungee is so geared to the horizontal stabilizer that when the stabilizer is moved to the nose-up position, a detent in the bungee assembly causes the elevators to be deflected upward approximately five degrees. The action also causes the control stick to be moved slightly aft. When the elevator bungee is in the detent, the force required to move the control stick fully back and stall the aircraft is somewhat decreased. Additionally, the elevators are inter-connected with the operation of the speedbrakes to assist the pilot in overcoming trim changes resulting from speedbrake operation. A system of cables and springs attached to the left speedbrake actuates the control cables between the stick and the elevator control valve. When the speedbrakes are opened, this system pulls the "nose-down" elevator cable, moving the stick forward and actuating the elevator to compensate for a "nose-up" trim change. When the

[1] Group A, B, C and D Airplanes.
[2] Group E Airplanes.

speedbrakes are closed, the stick moves aft to its original trimmed position, thus compensating for the "nose-down" trim change.

HORIZONTAL STABILIZER.[1] The entire surface of the horizontal stabilizer is moved by an electrically operated actuator to provide longitudinal trim. The actuator is controlled by forward and backward movement of the TRIM switch to "NOSE DOWN" or "NOSE UP." An arrangement of stabilizer limit switches and a stabilizer limit by-pass switch cause total stabilizer travel to be limited to 3 degrees nose up and 4 degrees nose down when the landing gear handle is in the "UP" position. When the landing gear handle is in the "DOWN" position, stabilizer travel is from 12 degrees nose up to 4 degrees nose down. This feature is installed in the aircraft to prevent overstressing of the tail surface through over-trimming at high speeds. The position of the horizontal stabilizer is shown on the trim position indicator.

HORIZONTAL STABILIZER.[2] The entire surface of the horizontal stabilizer is moved by an electrically operated actuator to provide longitudinal trim. The actuator is controlled by forward and backward movement of the TRIM switch to "NOSE DOWN" or "NOSE UP." Stabilizer travel is from 11 degrees nose up to 3 degrees nose down. The position of the horizontal stabilizer is shown on the trim position indicator.

MANUAL OVERRIDE SWITCH. A horizontal stabilizer manual override switch (6, figure 1-3) on the left console outboard of the throttle, will operate the horizontal stabilizer in the event the TRIM switch becomes inoperative. The positions of the manual override switch correspond to those of the TRIM switch, and as the switch is spring-loaded to the center or "off" position, it must be moved to the full extent of its travel in either direction to operate the horizontal stabilizer. When the emergency generator is operating, the manual override switch is the only means of actuating the horizontal stabilizer.

CAUTION

Use of the horizontal stabilizer manual override switch does not allow the limit switches on the stabilizer stops to cut out the actuator motor when the horizontal stabilizer reaches the full limit of its travel, consequently, continued operation of the manual override switch in one direction when the stabilizer is at the limit of travel in that direction will burn out the actuator motor, causing complete loss of stabilizer control.

RUDDER CONTROL. The aircraft is equipped with a power rudder operated at a reduced hydraulic pressure of 1150 psi. In tandem hydraulic system airplanes the rudder power control is operated by the flight control system, at the same reduced pressure, and is not connected to the utility system. Movement of the rudder pedals positions a hydraulic control valve in the vertical stabilizer through a cable and pulley system. The control valve in turn ports hydraulic pressure to the rudder actuating cylinder as required. Since there is no feed-back of air loads on the control surface of a hydraulic power system, a spring bungee is installed in the fin to center and restrain the control valve and rudder pedals, thereby simulating aerodynamic loads on the rudder surface to provide artificial "feel." Directional trimming is accomplished by displacing the entire rudder surface as a result of repositioning the center or "neutral" point of the spring bungee through the action of a 26 and 115 volt electric motor controlled by the RUDDER trim switch (23, figure 1-3) on the left hand console. Positions of the trim switch are "NOSE LEFT" and "NOSE RIGHT." The rudder trim actuator is powered by the aft monitored and 26 volt a-c busses. Rudder trim position is shown on the trim position indicator.

POWER BOOST RELEASE

A POWER BOOST REL handle (figure 1-4) on the lower right side of the armament panel may be used to release the power and boost systems from the flight controls in the event of hydraulic system failure.

TRIM POSITION INDICATOR[3]

The positions of the rudder trim and the horizontal stabilizer are shown on the trim position indicator (figure 1-4) on the lower right side of the instrument panel. The scale for rudder trim position is calibrated from "L" through "0" to "R" in units of approximately two degrees each. The total travel of the rudder trim is 7 degrees in either direction from "0." The scale for the position of the horizontal stabilizer is calibrated in increments of two degrees each from "DN" through "0" to "UP." Maximum travel of the stabilizer is four degrees down and 12 degrees up.

TRIM POSITION INDICATORS[4]

The positions of the rudder trim and the horizontal stabilizer are shown on the trim position indicators (22, figure 1-5, sheet 2) at the forward end of the right-hand console. The trim position indicators panel is labeled TRIM. The rudder trim position indicator is graduated in one-degree increments to the "L" (left) and "R" (right) of "O." Total travel of the rudder trim position indicator represents seven degrees of rudder travel left and right of center. All even degree marks are numbered from "0" through "6." The scale for horizontal stabilizer position is graduated in one-degree increments from "DN" (down) through "0," and in two degree increments from "0" through "UP."

[1] Group A, B, C and D Airplanes.
[2] Group E Airplanes.
[3] Group A, B, Ca, Cb, and Cc Airplanes.
[4] Group Cd, D, and E Airplanes.

All even numbered degree marks are identified numerically. Maximum indications of the stabilizer trim position indicator in early aircraft[1] are four degrees airplane-nose-down and twelve degrees airplane-nose-up. Maximum indications in later aircraft[2] are three degrees airplane-nose-down and eleven degrees airplane-nose-up.

WING FLAPS

Split flaps are installed on the trailing edges of the wings. Hydraulically actuated, the wing flaps are mechanically controlled by the FLAP control (7, figure 1-3) on the left console, outboard of the throttle. The wing flaps may be extended 50 degrees maximum by moving the FLAP control to "DOWN," or may be stopped at any intermediate position by placing the FLAP control at "STOP." When "UP" is selected, the flaps will fully retract. The position of the flaps is shown on the wheels and flaps position indicator (figure 1-4) on the pilot's instrument panel. A relief valve in the wing flap system allows the flaps to "blow-back" to prevent structural damage when the air load against them causes the hydraulic pressure within the actuating cylinder to exceed the pressure at which the relief valve opens (3500 psi). This automatic retraction will begin at approximately 165 knots IAS, and the flaps will not return automatically to the extended position if the FLAP control is in the "STOP" position.

WING SLATS

Aerodynamically controlled slats are installed on the leading edges of the wings to improve airflow characteristics over the wing surfaces at high angles of attack, primarily during approach and landing. The wing slats will open and close automatically as the aerodynamic loading on them dictates. Because so many variables—airspeed, gross weight, and applied load factor—affect the operation of the wing slats, no fixed airspeeds can be established as the points at which the slats begin to open or close. In general, however, they will begin to open at some airspeed below 200 knots, and will be fully opened at stalling speed.

SPEEDBRAKES

Two flush-mounted speedbrakes (1, figure 1-2) are installed, one on each side of the after fuselage, to provide deceleration during flight. Hydraulically operated, the speedbrakes are electrically controlled by the SPEEDBRAKE switch (22, figure 1-3) on the inboard side of the throttle grip. Movement of the switch to either "OPEN" or "CLOSE" actuates a solenoid valve which controls the flow of hydraulic pressure to the speedbrake actuating cylinders. As the SPEEDBRAKE switch has only two positions, the speedbrakes cannot be stopped at intermediate positions between fully opened and fully closed. A speedbrake position indicator is installed on the left-hand side of the instrument panel. The indicator illuminates whenever the speedbrakes are in any position other than fully closed. A "blow-back" feature allows the speedbrakes to begin closing when the airload against them causes the hydraulic pressure in the actuating cylinders to exceed the pressure at which the blow-back relief valve opens (3500 psi), thus preventing damage to the speedbrake system. At sea level, the speedbrakes will begin to close at an indicated airspeed of approximately 480 knots.

EMERGENCY SPEEDBRAKE CONTROL

The aircraft is equipped with an emergency speedbrake solenoid valve manual override control. The EMER SPEED BRAKE control (29, figure 1-3) at the aft end of the left-hand console may be used to open or close the speed brakes in the event of complete d-c electrical failure, or failure of one of the speed-brake control valve solenoids. The EMER SPEED BRAKE control is held in a "neutral" position by the action of a spring bungee and must be pulled up or pushed down momentarily to open or close the speedbrakes respectively. If speedbrake system failure is due to one solenoid being inoperative, it will be necessary to position the normal speedbrake control switch to the desired position before actuating the emergency manual override control.

In the event of hydraulic system and electrical failures with the speed brakes open, they may be closed by the manual control.

VORTEX GENERATORS

In order to combat buffet during high speed flight, vortex generators are installed on the aircraft. The vortex generators, which are small metal vanes set at various fixed angles relative to the normal airflow, extend along the span of the slats and up the sides of the fuselage.

LANDING GEAR SYSTEM

The tricycle landing gear is retracted and extended by utility hydraulic system pressure during normal operation. The main gear retracts up and forward, the wheels rotating to fit flush into the wheel wells in the wings. The struts do not retract into the wings, but are housed within the strut doors. When retracted, the main gear is held up by hydraulic pressure in the system and by latches on the wheel doors. The nose gear also retracts up and forward, the strut telescoping to allow the wheel to fit into the nose wheel well in the nose section. The nose wheel and strut are completely contained within the nose section when retracted, and are held up by the door latches and hydraulic pressure in the system. For emergency operation of the landing gear system, the door latches can be mechanically released by the pilot, allowing gravity and airstream pressure to lower and lock the gear, however the landing gear control must be "DOWN" before the wheels will extend.

LANDING GEAR CONTROL. The landing gear control (14, figure 1-3) on the left cockpit rail controls the normal operation of the landing gear system. Labeled WHEELS, the landing gear control has two positions, "UP" and "DOWN," and is mechanically linked to the

[1] Group A, B, C and D Airplanes.
[2] Group E Airplanes.

landing gear control valve. A warning light in the wheel-shaped handle of the control glows when the control is moved to either of its two positions. The light will remain illuminated until the wheels are locked in either the retracted or extended positions. The position of the wheels is shown on the wheels and flaps position indicator on the left side of the instrument panel. To prevent movement of the landing gear control to "UP" when the aircraft is on the ground, a safety solenoid is actuated by a switch on the left main gear strut when the weight of the aircraft compresses the strut, causing the landing gear control to be latched in the "DOWN" position. In normal operation, the safety solenoid is energized when the aircraft becomes airborne, unlatching the control handle. If the safety solenoid should malfunction, or if it should become necessary, during an emergency, to retract the landing gear while on the ground, a knurled knob (13, figure 1-3) on the landing gear control panel may be moved aft to unlatch the landing gear control.

EMERGENCY LANDING GEAR SYSTEM

In the event of utility hydraulic system failure, the landing gear may be lowered manually by means of the EMER LDG GR release handle (15, figure 1-3) on the extreme left end of the armament panel. When the landing gear control is moved to "DOWN" and the EMER LDG GR release handle is pulled, the wheel and strut doors are unlatched, allowing the landing gear to drop into the airstream. The gear extends and locks by a combination of ram air force and gravity. Once lowered by the emergency system, the landing gear cannot be retracted again unless utility hydraulic pressure is available. Refer to Section III for emergency operation of the landing gear system.

CAUTION

> After lowering the landing gear by the emergency system, the gear should not again be retracted until the ratchet in the emergency mechanism has been reset. Failure to comply with this instruction will result in a broken ratchet.

WHEELS AND FLAPS POSITION INDICATOR. The position of the landing gear and wing flaps is presented on the wheels and flaps position indicator (figure 1-4) on the lower left side of the instrument panel. When the wheels are down and locked, the image of a wheel appears in a small window provided for each wheel on the instrument. When the landing gear is up and locked, the word "UP" appears in each window. During the period when the landing gear is transient, or whenever the wheels are not locked in position, diagonally striped signals are shown in the windows. The position of the wing flaps is shown, with respect to the wing, in units from "UP" through "½" to "DOWN," each unit corresponding to one-quarter of the total amount of extension possible.

BRAKE SYSTEM

Single-disc, spot-type brakes are installed on the main wheels only. The brake system includes a separate hydraulic reservoir (figure 1-11) located in the nose section of the aircraft. Two master brake cylinders, operated by toe pressure on the upper part of the rudder pedals, provide the pressure necessary for operation of the brakes.

ARRESTING HOOK SYSTEM

An externally mounted arresting hook (20, figure 1-2) is installed on the lower aft fuselage. Retraction and extension of the hook is accomplished by a pneumatic-hydraulic hold-down cylinder in the aft engine compartment. 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 filled with hydraulic fluid to the full level and then charged with compressed air to 900 $\pm$ 50 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 of the piston to effect retraction of the arresting hook, which is then held in the retracted position by a mechanical latch. Compressed air pressure and the weight of the arresting hook cause extension when the 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 by restricting the flow of fluid between the lower and upper chambers of the hold-down unit whenever external forces tend to force the hook rapidly toward the retracted position.

ARRESTING HOOK CONTROL. A HOOK control (3, figure 1-5) on the right cockpit rail, controls the operation of the arresting hook. When the control is moved to "DOWN," the arresting hook is manually unlatched, allowing pressure from the hold-down unit and the force of gravity to extend the hook. A light in the handle of the HOOK control will glow when the control is moved to "DOWN," and will go out before the hook reaches the fully extended position. The "UP" position of the control manually positions the arresting hook control valve so that utility hydraulic fluid at 3000 psi flows into lower part of the arresting hook hold-down cylinder, overriding the air pressure and causing the hook to be retracted and latched against the lower surface of the fuselage. If the cable system to the latching mechanism should fail, the latch will automatically release the arresting hook; however, the HOOK control must be moved to the "DOWN" position before the hook will extend.

INSTRUMENTS

All flight and engine instruments are located on the instrument panel (figure 1-4). The altimeter, rate-of-climb, and airspeed indicators are connected to the pitot-static system; the turn and bank indicator is operated from compressor bleed air pressure, and the attitude gyro, fuel quantity, trim position and wheels and flaps position indicators are electrically operated. The eightday clock and the accelerometer are independent of other systems in operation. A light is mounted in the center of each instrument lens to provide equal illumination over the entire face of the instrument.

ALTIMETER. The pressure altimeter (figure 1-4) indicates the altitude of the aircraft above sea level to a height of 50,000 feet. The dial face is marked in increments of 100 feet, each complete revolution of the pointer indicating a change in altitude of 1000 feet. On the left of the center of the instrument is a window containing two rotating counters; the inner counter registers altitude in thousands of feet, while the outer registers ten thousands of feet. When the altimeter pointer makes twelve revolutions, for instance, the outer counter will indicate 1, and the inner counter will indicate 2, thus showing that the aircraft is at an altitude of 12,000 feet above sea level. At the extreme right side of the altimeter face is the barometric pressure window. The barometric pressure dial seen through the window is marked from 28.10 to 31.00 inches of mercury, and used to correct for variations in sea level barometric pressure by means of a knob on the lower left corner of the instrument case.

AIRSPEED INDICATOR. A combination airspeed indicator and Mach meter (figure 1-4) is mounted on the instrument panel. The airspeed portion of the dial is fixed in position, and is calibrated from 80 to 650 knots. The Mach meter scale is a rotating disc, marked from 0.50 to 2.20, turning beneath the airspeed dial. Only a portion of the disc can be seen through a cutout in the airspeed dial. Both airspeed and corresponding Mach number are indicated simultaneously by a single needle pointer. On the Mach number disc is a movable index which is used to set the limiting Mach number of the aircraft by depressing and turning a PUSH MACH LIMIT knob on the lower left corner of the instrument case. On the edge of the airspeed dial is an airspeed index pointer, which is adjustable through a range of from 80 to 145 knots merely by turning the PUSH MACH LIMIT knob. The airspeed index pointer is used as a reference point to indicate the minimum safe airspeed for a particular gross weight during approach and landing or slow speed flight. See figure 6-1 for stalling speeds at various gross weights and angles of bank.

ATTITUDE GYRO INDICATOR. An electrically operated attitude gyro indicator (figure 1-4) is mounted on the instrument panel. The instrument is non-tumbling and so provides true attitude indication throughout all flight maneuvers. If electrical power to the instrument fails, or if power phase rotation is incorrect, an OFF warning flag will drop into view on the face of the instrument to notify the pilot of the unreliability of the attitude gyro indicator. An erection device is provided to maintain the gyro in a stabilized neutral position when operating, but to compensate for any precession which might occur, a PULL TO CAGE knob is installed on the instrument case. When the knob is pulled, the attitude gyro mechanism is caged and erected; upon release of the knob, which is self-retracting, the mechanism is uncaged.

WARNING

The attitude gyro indicator should be uncaged only when the aircraft is in straight and level flight. Because the gyro is caged and erected to the vertical plane of the aircraft, and not to true vertical, uncaging of the gyro when the aircraft is not in straight and level flight will result in an erroneous attitude indication.

EMERGENCY EQUIPMENT

FIRE DETECTION SYSTEM

The fire detection system indicates the presence of fire in the area of the engine, tailpipe, and accessories section. If fire occurs in these locations, a push-to-test type FIRE warning light (figure 1-4) on the take-off check list panel will glow. The fire detection system may be checked by depressing the TEST switch adjacent to the light. When the switch is depressed, the FIRE warning light will glow, 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 an extent where he might not be able to manually open the parachute. 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 pin on the CANOPY-HARNESS REL handle (See figure 1-10). When the pilot separates from the seat after ejection, the arming pin is automatically pulled and the opener is then armed.

WARNING

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

COCKPIT ENCLOSURE

The cockpit enclosure consists of a fixed, three-piece windshield and a hinged "clamshell" canopy. The two windshield side panels are of moulded plastic, and the center panel is constructed of alternating layers of glass and of vinyl to provide a bullet resistant surface.

CANOPY

The cockpit canopy, which is hinged at the after end by shear bolts, moves back and up when opened. When closed, the canopy is held in place by latches on both sides of the forward edge of the canopy rail which engage latch pins on the cockpit rails. An air bungee cylinder (6, figure 1-2), mounted aft of the ejection seat, counterbalances the canopy during normal operation and provides snubbing to prevent the canopy from opening too rapidly after the latter is manually unlatched. The canopy is closed by grasping the ledge on either side and by pulling down, overriding the air bungee cylinder pressure. The canopy may then be locked by moving the CANOPY control handle forward.

CANOPY CONTROLS. The CANOPY control handle (1, figure 1-3), on the left console, is mechanically linked to the canopy mechanism. When the CANOPY control handle is moved forward, it slides the canopy forward, causing the canopy latches to engage the latch pins. When the handle is moved aft, the canopy slides aft, disengaging the latches and allowing air bungee pressure to open the canopy. An external CANOPY release handle, which can be reached from the ground, is set flush into the left side of the fuselage below the cockpit. Pulling the external CANOPY release handle out and forward unlatches the canopy, allowing it to open in the normal manner. To close and lock from the exterior, the canopy must be manually held down and the external handle moved aft and in until it is flush with the fuselage.

INTERIOR CANOPY JETTISONING

The canopy may be jettisoned by pulling the CANOPY-HARNESS REL handle (10, figure 1-10) on the right side of the ejection seat, or by moving the CANOPY control handle aft while in flight. When the CANOPY-HARNESS REL handle is pulled, the canopy slides slightly aft to unlatch, swings open, and shears at the hinges as the full pressure of the air bungee cylinder is released against the canopy structure. Opening the canopy by the CANOPY control while in flight should cause the canopy to shear at the hinges from the force of the relative wind. The canopy is also jettisoned when the face curtain handle (2, figure 1-10) is pulled down. (See FACE CURTAIN, in this section.)

EXTERIOR CANOPY JETTISONING(1)

An emergency canopy jettison control (7, figure 1-2) is provided on each side of the fuselage, just forward of the wing root, for jettisoning the canopy during rescue. The control is a red handle, marked PULL CANOPY JETTISON, and is installed in a recess behind a spring-loaded door. This door is plainly pointed out by a RESCUE arrow. When the door is pushed in, the PULL CANOPY JETTISON handle extends and may then be grasped and pulled to fire the air bungee cylinder.

WARNING

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

EJECTION SEAT

The pilot's seat (figure 1-10) is of the ejectable type, can accommodate a back-type parachute, a modified PK-2 pararaft kit and seat pan, and is designed for use with a flight suit which incorporates an integral harness. A non-adjustable head rest is part of the seat structure and houses the face curtain. Although the ejection seat is not equipped with conventional foot rests, 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 to separate the seat from the aircraft. Upon ejection, a lanyard anchored to the aircraft structure fires a harness release actuator which automatically frees the seat belts and shoulder harness from the seat structure three-quarters of a second after ejection, causing the pilot and the seat to separate.

SEAT ATTACHMENTS. The pilot is held in the seat by attachments to the integrated flight suit (View "A," figure 1-9A). This flight suit incorporates within its structure the combined attributes of a seat belt, shoulder straps and a parachute harness, thus leaving the pilot with few of the normal encumbrances. The shoulder harness 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 which engage other fittings that extend from the front shoulders of the flight suit. Short seat belts, which are sewn to the parachute harness on each side, and attached to the seat structure at the after corners of

(1)Group D, Ec and Ed Airplanes.

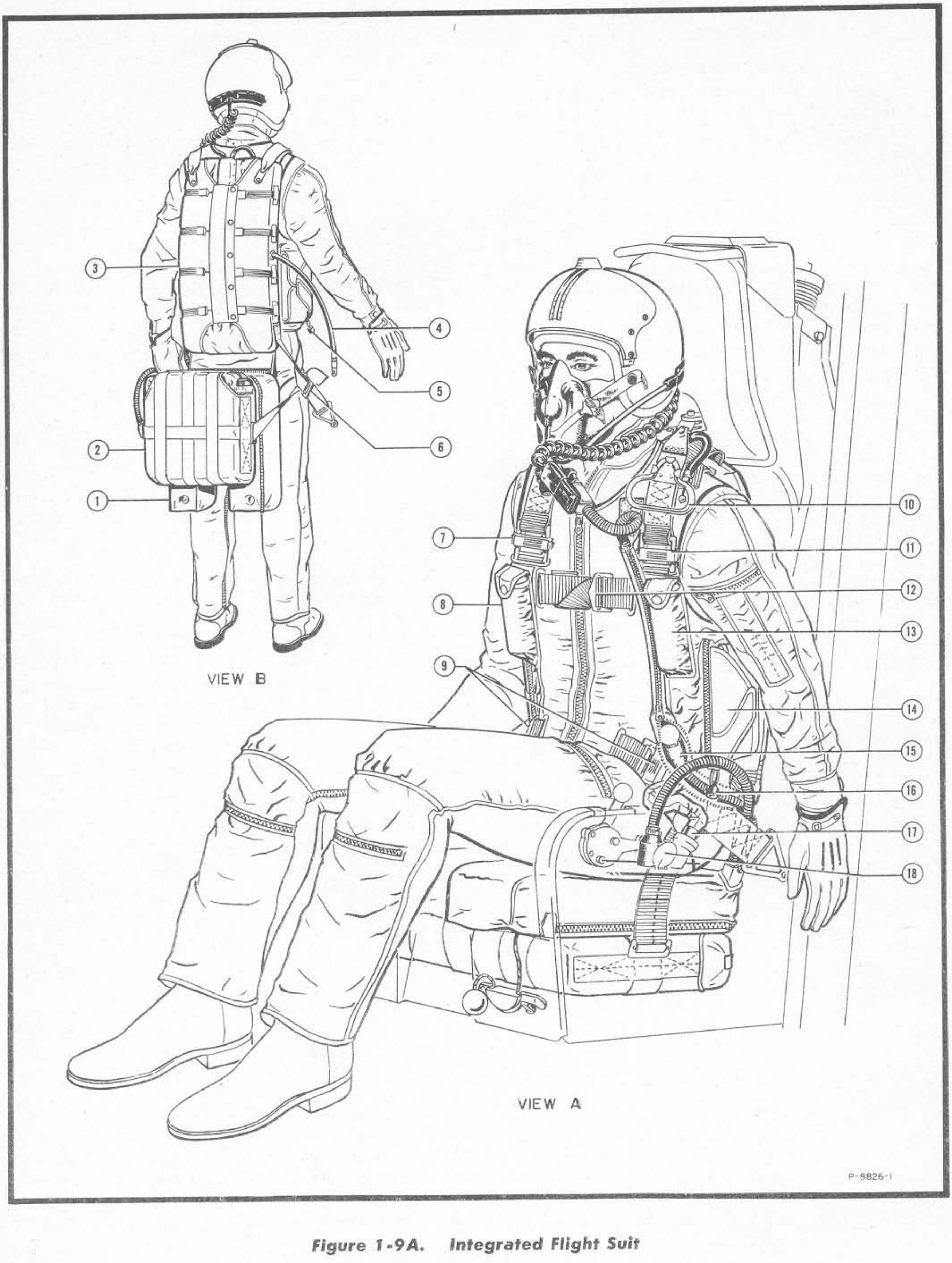


Figure 1-9A. Integrated Flight Suit

Key to Figure 1–9A Integrated Flight Suit

1. Seat cushion
2. PK-2 Pararaft
3. Parachute
4. Barometric parachute opener lanyard
5. Flotation gear actuating lanyard
6. Lap belt adjustment buckle
7. Shoulder harness and parachute attachment
8. Signal flare
9. Seat belt and parachute attachments
10. Parachute ripcord grip
11. Shoulder harness and parachute attachment
12. Suit harness adjustment buckle
13. Signal flare
14. Flotation gear
15. Suit oxygen and radio connection
16. Flotation gear actuating lanyard
17. Anti-g suit connection
18. Console oxygen and radio connection

the seat basin, are adjustable in length. The loose ends of the seat belts have quick-action fittings which engage fittings protruding from the hip region of the flight suit. 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 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 to which a pulley is attached, 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 the pulley on the actuator piston and is anchored to the seat structure. Thus, when the seat is ejected, the actuator safety pin is pulled by the lanyard anchored to the aircraft structure, and the firing mechanism detonates the cartridge which, three-quarters of a second later, exerts enough force to actuate the piston. The piston extends, pulling the cable, and the harness release mechanism opens, freeing the seat belts and shoulder harness.

SEAT CONTROLS

SEAT SWITCH. The seat is electrically adjusted in the vertical plane by movement of the SEAT switch (figure 1-4) to either of its two positions, "UP" or "DOWN," and is stopped at the desired position by releasing the switch to the center, or "off" position.

SHOULDER HARNESS CONTROL. The SHOULDER HARNESS control (5, figure 1-10), on the left side of the seat basin, controls the movement of the shoulder harness inertia reel. When the control is in the "LOCKED" position, the shoulder harness will not extend and the pilot's freedom of movement is restricted. The "UNLOCKED" position allows the shoulder harness to extend or retract as the pilot moves about; however, the reel will lock automatically if the aircraft is subjected to a deceleration in excess of 2.5g along the thrust line. This safety feature helps to prevent injuries if the shoulder harness is not locked prior to an arrested landing or a crash.

CANOPY-HARNESS RELEASE HANDLE. A "D-ring" handle (10, figure 1-10), labeled CANOPY-HARNESS REL, is mounted on the right side of the ejection seat basin. From the aft end of the handle protrudes a pin which extends down through the edge of the ejection seat to anchor the arming lanyard of the barometric parachute opener. A spring-loaded latch, which is grasped in conjunction with the CANOPY-HARNESS REL handle, serves to retain the handle in the proper position and must be squeezed before the latter can be pulled. When the handle is pulled up, the canopy is jettisoned by the air bungee cylinder, and the shoulder harness and seat belt attachments are severed from the seat, allowing the pilot to leave the cockpit with a parachute and pararaft kit still attached to the flight suit harness (View "B," figure 1-9A).

WARNING

The CANOPY-HARNESS REL handle "D-ring" should not be pulled 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.

FACE CURTAIN. The ejection seat face curtain (2, figure 1-10) serves not only as a control for jettisoning the canopy and ejecting the pilot's 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 is pulled downward, the first portion of travel jettisons the canopy and the last portion causes the seat to be ejected.

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 will prevent seat ejection if tension is maintained on the face curtain after the canopy has jettisoned.

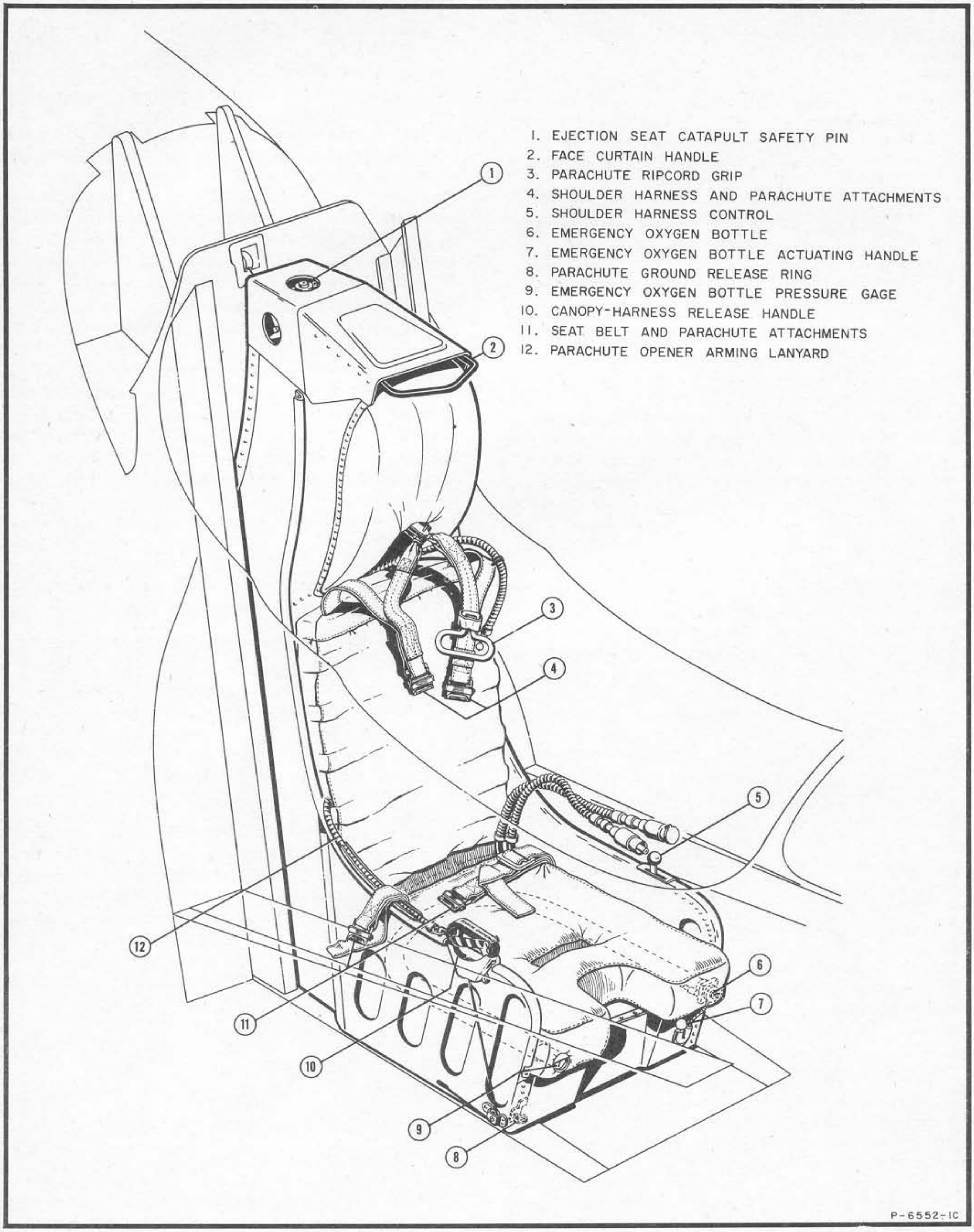


Figure 1-10. Ejection Seat

PARACHUTE RELEASE RING. In order to remove the parachute for ground inspection and maintenance, a parachute release ring (8, figure 1-10) is installed under the lower right side of the ejection seat. Pulling the ring actuates the pilot release mechanism without operating the canopy jettisoning feature. When the parachute release ring is pulled, the pilot release mechanism will remain in the "release" position. When the parachute is replaced, a screwdriver or similar object must be inserted in a hole, labeled PUSH TO RESET PILOT RELEASE, in the seat back, in order to reset the release mechanism.

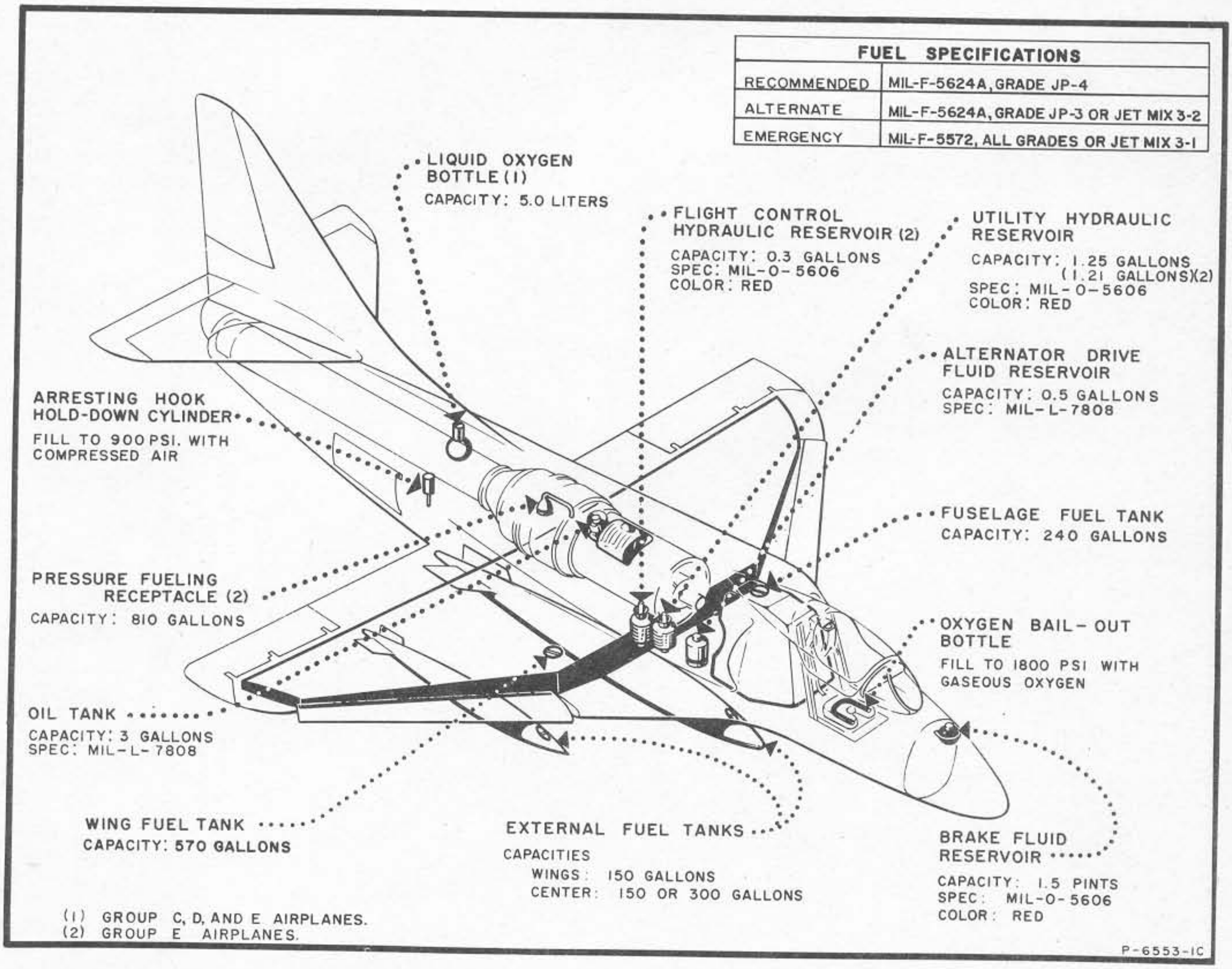
CAUTION

Disconnect the barometric parachute opener arming lanyard from the retaining pin on the CANOPY-HARNESS REL handle before removing the parachute from the seat. If this action is not taken, the arming pin will be pulled and the parachute will open immediately.

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 and Associated Electronic Equipment
Lighting Equipment
Oxygen System
Navigation Equipment
Armament Equipment
Miscellaneous Equipment



FUEL SPECIFICATIONS	
RECOMMENDED	MIL-F-5624A, GRADE JP-4
ALTERNATE	MIL-F-5624A, GRADE JP-3 OR JET MIX 3-2
EMERGENCY	MIL-F-5572, ALL GRADES OR JET MIX 3-1

Figure 1-11. Servicing Diagram

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SECTION II
NORMAL PROCEDURES

BEFORE ENTERING THE AIRCRAFT

FLIGHT RESTRICTIONS. Refer to Section V for restrictions to be observed during operation of the aircraft and engine in flight.

CRUISE CONTROL. To determine the required fuel, power settings and airspeeds necessary to accomplish the assigned mission, refer to Appendix I.

WEIGHT AND BALANCE. Determine the weight of ammunition, bombs and other stores which are loaded. Check gross weight for take-off and determine the anticipated gross weight for landing by computing the weight of stores to be dropped and fuel to be used during the mission, referring to Section V for weight limitations. Insure that the distribution of loads falls within the center of gravity fore and aft limits as specified in the Handbook of Weight and Balance Data, AN 01-1B-40. It is necessary that a Weight and Balance Form F be completed when the aircraft is loaded in a manner for which no previous form has been filed.

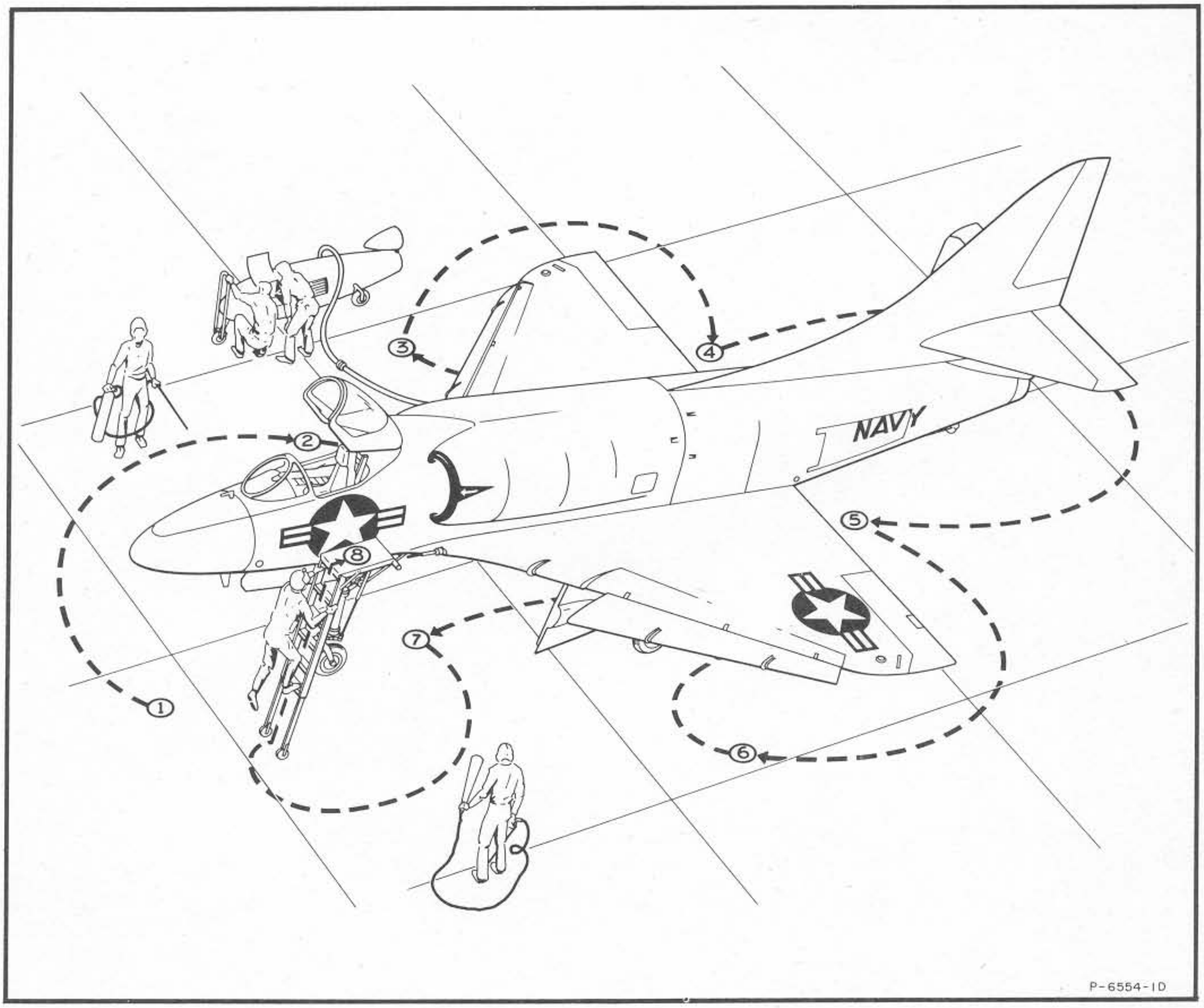

Figure 2-1. Exterior Inspection

CHECK LISTS

See figure 2-3 for a reproduction of the TAKEOFF and LANDING check lists appearing in the cockpit of the aircraft.

EXTERIOR INSPECTION

Consult the Naval Aircraft Flight Record (yellow sheet) to determine the status of the aircraft and that it has been fully serviced with fuel, oil, oxygen, compressed air, and hydraulic fluid.

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

1—2. FORWARD FUSELAGE

a. Air conditioning intake and exhaust ducts . . . . clear

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

c. Nose compartment panels . . . condition, security

d. Nose compartment cooling air inlet . . . . clear

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

f. Controls access panel (right side) . . . . secure

g. Nose wheel well door . . . . . . condition, security

h. ARMT. SAFE DISABLE switch[1] . . . . "SAFE"

i. Approach light . . . . condition, security

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

k. Nose wheel tire . . . . proper inflation, no slippage

l. Nose wheel downlock pin . . . removed

m. Emergency generator . . . . retracted, secure

n. CANOPY JETTISON handle[2] . . . . stowed; access door closed

o. Engine intake plug . . . . removed

p. Intake duct . . . . free of foreign objects

*q. Wing tank filler cap . . . . secure

2—3. R-H WHEEL WELL

a. Main wheel well doors . . . . . . condition, security

b. GUN PNEUMATIC PKG pressure gage . . . . 3200 ± 200 psi

c. ARM SAFETY DISABLE SWITCH[3] . . . . "SAFE"

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

e. Main wheel tire . . . . proper inflation, no slippage

f. Main gear downlock pin . . . . removed

g. Fuel system vent exit . . . . clear

3—4. R-H WING

a. General condition . . . . wrinkles, cracks, loose rivets; bottom surface free of stains indicating fuel or hydraulic fluid leaks

b. Wing rack store . . . . secure

c. Wing slat . . . . free movement

d. Starting power . . . . connected or available, if required

e. Navigation and formation lights . . . . condition

f. Aileron and wing flap . . . . . . condition, bonding

4—5. AFT FUSELAGE AND TAIL SECTION

a. Aft engine compartment access door . . . . closed

b. Wing tank water/sediment drain cock . . . . drained

c. Speed brake . . . . condition, security

d. Tailpipe cover . . . . removed

e. Tailpipe condition . . . . cracks, wrinkles, burns, fuel deposits

f. Tail light . . . . condition

g. Rudder, elevator, and horizontal stabilizer . . . . condition, bonding

h. Speed brake . . . . condition, security

i. Arresting hook . . . . retracted and locked, condition

j. Arresting hook hold-down cylinder pressure gage . . . . . . 900 ± 50 psi

5—6. L-H WING

a. Wing flap and aileron . . . . . . condition, bonding

b. Aileron tab . . . . condition

c. Navigation and formation lights . . . . condition

d. Wing general condition . . . . . wrinkles, cracks, loose rivets; bottom surface free of stains indicating fuel or hydraulic fluid leaks

e. Wing slat . . . . free movement

f. Wing rack store . . . . secure

*Item located on upper surface of right wing.
[1]Group A, Ba, Bb and Bc Airplanes.
[2]Group D, Ec and Ed Airplanes.
[3]Group Bd, C, D and E Airplanes.

6—7. L-H WHEEL WELL

a. Main wheel tire .... proper inflation, no slippage

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

c. Main gear downlock pin .... removed

d. Aileron accumulator pressure gage(1) .... 700 psi

e. Main gear doors .... condition, security

7—8. CENTER FUSELAGE UNDERSIDE

a. Fuselage rack store .... secured

b. GUN PNEUMATIC PKG pressure gage .... 3200 ± 200 psi

c. Forward engine and accessories section access door .... closed

d. Link and case ejection chutes .... clear

e. Fuselage tank water/sediment drain cock .... drained

8—9. COCKPIT AREA

a. Engine intake plug .... removed

b. Intake duct .... free of foreign objects

c. CANOPY JETTISON handle(2) .... stowed; access door closed

d. Canopy cover .... removed

e. Entrance ladder .... in place

f. Controls access panel (left side) .... secure

g. Canopy control .... pull out and forward to open canopy

h. Fuselage tank filler cap ..... secure

i. Pitot tube cover .... removed

j. Canopy air bungee cylinder air pressure gage .... 2500 psi

k. Canopy surface and seal .... condition

l. Ejection seat catapult safety pin and lanyard .... installed

ON ENTERING THE AIRCRAFT

INTERIOR INSPECTION. Check the general appearance of the cockpit, and ensure that all gear is properly stowed and secure. Check the oxygen bailout bottle pressure (1800 psi), make sure harness connections, oxygen, radio, and anti-black-out connections, and perform the following checks before starting the engine:

a. EMER SPEED BRAKE control . . "Normal"

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

c. DROP TANKS switch .... "OFF"

d. START-ABORT switch .... Pulled out

e. FUEL CONTROL switch .... "PRIMARY"

f. FUEL valve control .... "NORMAL"

g. Throttle .... "OFF"

h. SPEEDBRAKE switch .... "CLOSE"

i. Master exterior lights switch .... Off

j. Horizontal stabilizer manual override switch .... Centered

k. FLAP control .... "UP"

l. AIR START switch .... "OFF"

m. THROTTLE FRICTION wheel. As desired

n. Landing gear control .... "DOWN"

o. OXYGEN switch .... "ON"

p. Oxygen pressure gage(3) .... 1800 psi

Oxygen quantity gage(4) .... "4" (min)

q. All armament switches .... Off

r. Altimeter .... Field elevation

s. Airspeed indicator .... Set limiting Mach number and minimum airspeed

t. Accelerometer PUSH TO SET knob .... Depress

u. Clock .... Set to proper time

v. PITOT HEAT switch .... "OFF"

w. HOOK control .... "UP"

x. IFF MASTER switch .... "OFF"

y. UHF radio control panel .... Desired frequency

z. INT LTS control panel .... All switches "OFF"

aa. AIR COND control .... "OFF"

ab. EXT LTS control panel .... TAXI light "OFF"

ac. SPARE LAMPS container ...... Adequate supply

INTERIOR INSPECTION (NIGHT FLIGHTS). In addition to the normal interior inspection, check the operation of all interior and exterior lights when flight is to be conducted at night.

BEFORE STARTING THE ENGINE

Ascertain that the areas forward and aft of the aircraft

(1) Group A, B, C and D Airplanes.
(2) Group D, Ec and Ed Airplanes.
(3) Group A and B Airplanes.
(4) Group C, D and E Airplanes.

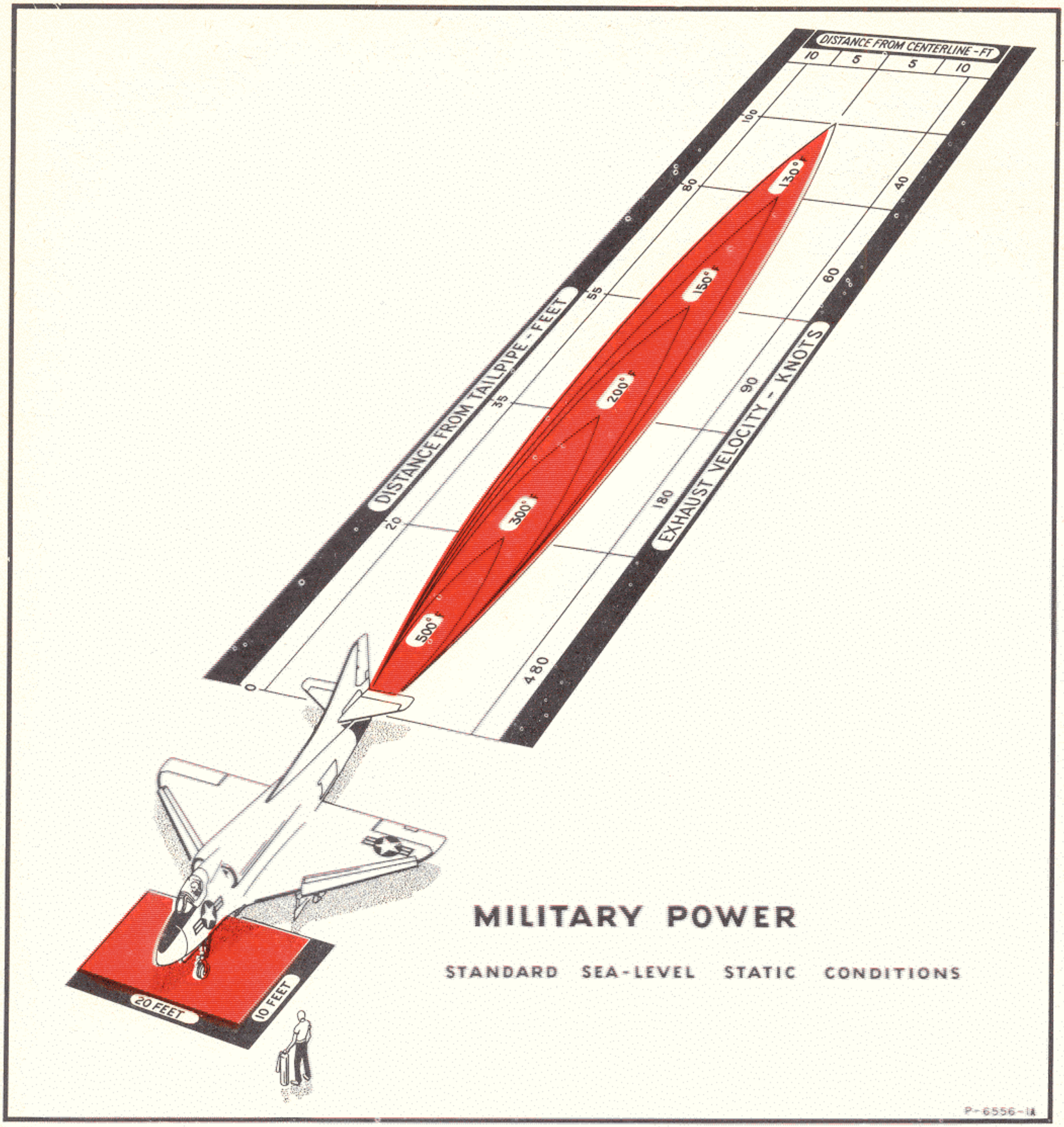

Figure 2-2. Danger Areas

are clear of personnel and loose objects. (See figure 2-2 for danger areas.) Make certain that fire fighting equipment is available and manned.

STARTING THE ENGINE

The engine should always be started on the ground in the following prescribed manner:

a. Throttle ...."OFF"

b. START-ABORT switch ....Depress

c. Throttle ...."IDLE" when the RPM pointer reaches "START"

Note

- The R.P.M. pointer may give erratic indications until starting speed is attained, then the pointer should stabilize at "START."
- The ground crew should check the output of

the generator on all three phases with the proper electrical test unit during each start, and "flash" the generator exciter field circuit if necessary.

When the engine starts, as indicated by a rise in exhaust temperature and rpm, it may be necessary to control the temperature by manipulation of the throttle to prevent exceeding the starting limitations stated in Section V. The starting limitations are particularly likely to be exceeded unless some throttle manipulation is done when the ambient temperature is over about 60°F. Throttle manipulation is effective only in the positions from "IDLE" to about halfway back toward "OFF." Below this position fuel flow to the engine may be cut off. When the engine speed is stabilized with the throttle at "IDLE," check the following:

RPM .... "IDLE"

TEMP .... One needle width above "MIL" (maximum)

Note

At "IDLE" rpm, the TEMP pointer will normally stabilize at a position below the maximum indicated above.

FUEL BOOST .... "NORM"

OIL PRESS .... "NORM"

Note

If the OIL PRESS window reads "OUT" at "IDLE" rpm advance the throttle to slightly above "IDLE." If "NORM" oil pressure is not obtained at this higher RPM, shut down the engine and determine the reason for the lack of oil pressure.

ABNORMAL STARTS

HOT START. Any start in which the exhaust temperature exceeds "ACC" is a "hot start." When a hot start is encountered, immediately stop the engine and have the ground crew insure that all excess fuel has drained from the combustion chamber before initiating another start. If, during five consecutive starts, the exhaust temperature exceeds "ACC," or if during any one start, the exhaust temperature reaches the maximum position on the indicator, do not attempt further starting, but have the engine thoroughly inspected to determine the reason for the hot starting.

SLOW START. Normally, engine starting speed should be attained within thirty seconds after depressing the START-ABORT switch, and light-off should occur within fifteen seconds after movement of the throttle to "IDLE." If these do not occur within the prescribed times, retard the throttle to "OFF" and pull up on the START-ABORT switch to deactivate the starter.

FALSE START. If the engine should light-off in the proper manner during starting but does not accelerate to idling speed, and the exhaust temperature does not rise, retard the throttle to "OFF" and have the ground crew investigate the difficulty.

CLEAR ENGINE PROCEDURE. To clear the engine of excess fuel after abnormal starts, the starter should be engaged for approximately 15 to 20 seconds without supplying ignition.

a. Throttle .... "OFF"

b. Depress the START-ABORT switch.

c. Allow the engine to turn for 15 to 20 seconds.

d. Pull up on the START-ABORT switch.

AFTER STARTING THE ENGINE

When electrical power becomes available after starting the engine, check the following items:

a. MANUAL FUEL warning light . . Push to test

b. LABS light[1] .... Push to test

c. Marker beacon indicating light . . Push to test

d. Attitude gyro indicator .... Uncaged and warning flag not in view

e. Depress TEST switch to check FIRE, FUEL TRANS,[2] UTILITY HYD, CONTROL HYD,[3] and SPD BRK OPN warning lights, and fuel quantity indicating circuits.

f. SEAT switch .... "UP" or "DOWN" until seat reaches desired position

g. COMPASS CONTROLLER panel .... Set course indicator

Note

In addition to the normal inspection after starting the engine, the operation of all interior and exterior lights should be checked when flight is to be conducted at night.

ENGINE GROUND OPERATION

No warm-up period is necessary, and engine operation should be held to a minimum while on the ground because fuel consumption is high, even at low power settings. Certain engine operating checks must be conducted on the ground, however, to insure satisfactory performance when airborne.

Note

To avoid possible rupture of second stage compressor blades during extended continuous ground operation, do not make steady state ground runs within the 60% to 82% rpm range.

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[1] Group Bb, Bc, Bd, C, D and E Airplanes.

[2] Group Bd, C, D and E Airplanes.

[3] Group E Airplanes.

GROUND TESTS

The following tests should be made to check satisfactory operation of the aircraft systems:

Note

Direct the ground crew to check the hydraulic system pressure gage or gages[2] and the ENGINE FUEL PUMP FAILURE warning light in the right hand wheel well to insure that the respective pumps are operating properly.

a. FLAP control . . Move to "DOWN," checking movement on the wheels and flaps position indicator. Move to "UP" and then to "STOP," noting movement. Move to "UP" to completely retract the flaps.

b. SPEEDBRAKE switch . . . . "OPEN"; check SPD BRK OPN indicator light, ascertain that control stick moves forward, have ground crew check speedbrake travel, then move to "CLOSE."

c. TRIM switch . . . Select "NOSE UP" and "NOSE DOWN," checking trim position indicator for correct movement. Move to "RWD" and "LWD," checking left aileron for trim tab movement. Fair the ailerons.

d. Rudder trim switch . . . . Move to "NOSE LEFT" and "NOSE RIGHT," checking correct movement on the trim position indicator.

e. Horizontal stabilizer manual override switch . . . . Select "UP" or "DOWN," noting proper movement on the trim position indicator.

TAXIING INSTRUCTIONS

When ready to taxi, advance the throttle to some value above "IDLE," and allow thrust to develop before releasing the brakes. Release the brakes, and when the desired taxiing speed is reached, retard the throttle to a point at which that speed is maintained. Under normal conditions, "IDLE" RPM should be adequate to maintain a safe taxiing speed. Do not taxi with the canopy open at speeds which, coupled with headwinds, would cause the relative wind to exceed 60 knots. Use the brakes to turn the aircraft, as the rudder will not be effective at normal taxiing speeds, but avoid sharp turns to prevent excessive wear on the tires. Hold taxi time to a minimum to conserve fuel as fuel consumption is approximately three gallons per minute at idle rpm when at sea level.

BEFORE TAKE-OFF

PRE-FLIGHT AIRCRAFT CHECK

a. Canopy . . . . Closed

b. AIR COND control . . . . Desired temperature

c. Trim

Aileron	Stick centered and aileron tab faired
Rudder	0
Horizontal stabilizer[1] .	4 units "UP" for half flaps or full flaps take-off.
Horizontal stabilizer[2] .	4 units "UP" for half flaps or full flaps take-off.

d. Wing flaps . . . . Half flaps or full flaps, as desired, for airfield take-off; full flaps for catapulting

e. SPEEDBRAKE switch . . "CLOSE"

f. EMER SELECT switch . . "ALL"

g. Move all flight controls through their entire range of movement, checking correct deflection with respect to control stick and rudder pedal positions.

h. Observe fuel quantity indicator for adequate supply of fuel.

i. SHOULDER HARNESS control . . . "LOCKED"

j. Check oxygen.

PRE-FLIGHT ENGINE CHECK

Select a run-up spot which is clear of loose objects around the engine air intakes and the tailcone.

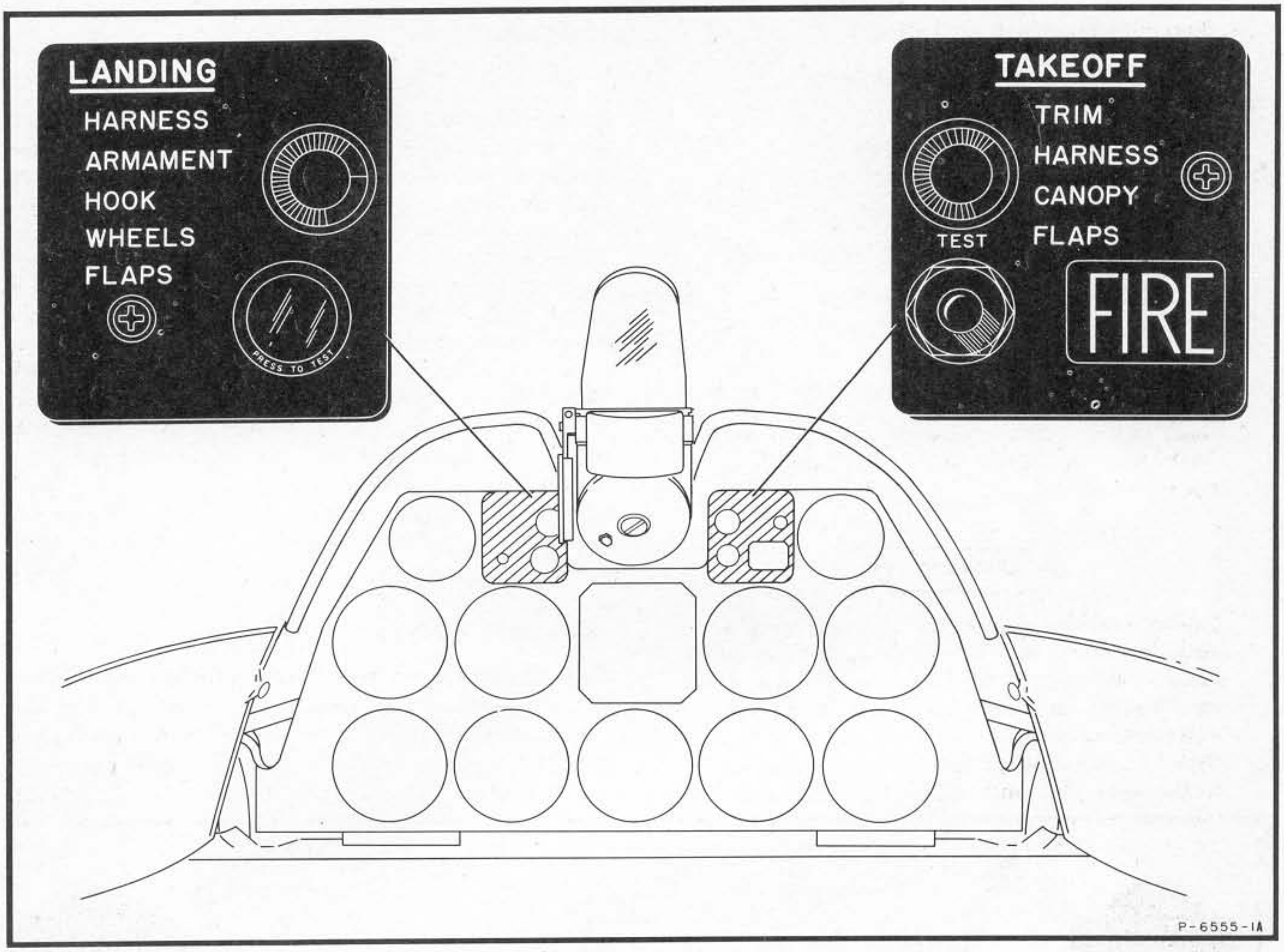
Note

Recommended engine checks at "MILITARY" power will generally have to be made during the take-off roll if the aircraft is not tied down. Static engine tests without tie-down will have to be made at the maximum obtainable power settings as determined by the aircraft beginning to slide with the brakes locked.

MANUAL FUEL CONTROL CHECK

Because overtemperaturing is so often encountered during accelerations when operating on the manual fuel control system, the ground check of the system will consist only of a test of the electric transfer solenoids in the fuel control unit to insure that they will operate properly.

(1) Group A, B, C and D Airplanes.
(2) Group E Airplanes.



Group A and Ba Airplanes
Figure 2-3. Check-off List (Sheet 1)

Note

A complete test of the manual fuel control system will be conducted only at the 30, 60, 90, and 120-hour inspections of the engine, and whenever a new engine fuel control unit is installed. Refer to Section VII for the recommended ground checking procedure at these times.

a. Increase the RPM to 90%.

b. Move the FUEL CONTROL switch to "MANUAL."

"MANUAL." MANUAL FUEL warning light should be illuminated.

Note

The engine rpm will increase or decrease immediately upon moving the FUEL CONTROL switch to "MANUAL," depending on the variation between existing ambient conditions and those comprising a standard day. If standard day conditions prevail, no apparent change in engine speed will occur.

c. Retard the throttle to "IDLE."

d. FUEL CONTROL switch .... "PRIMARY" when RPM decreases past 70%

PRIMARY FUEL CONTROL CHECK

a. FUEL CONTROL switch .... "PRIMARY"

b. RPM .... "IDLE"

c. Rapidly advance the throttle to "MILITARY."

d. Check time of acceleration to "MIL" RPM at 15 seconds or less.

Note

Following a ground start and before the engine is fully heated, the acceleration from "IDLE" to "MIL" RPM may take as long as 17 seconds. The heating effect of a flight should regain the ability to accelerate in 15 seconds. The engine is not expected to meet the 15-second time limit until it has been operated the equivalent of five minutes at 95% RPM.

e. Retard the throttle to 80% RPM.

f. Again advance the throttle rapidly to "MILITARY."

g. Check time of acceleration from 80% to "MIL" RPM at five seconds or less and note the following:

RPM .... "MIL"
TEMP .... "MIL" (maximum)
FUEL BOOST ..... "NORM"
OIL PRESS .... "NORM"

Note

Rapid manipulation of the throttle back and forth between "IDLE" and "MILITARY" is not recommended.

CAUTION

During accelerations, rpm and exhaust temperature limitations must be observed. Continuous satisfactory operation of the engine is based on strict observance of the prescribed limitations (refer to Section V). Any deviations from these limitations will result in serious reduction in the service life and reliability of the engine.

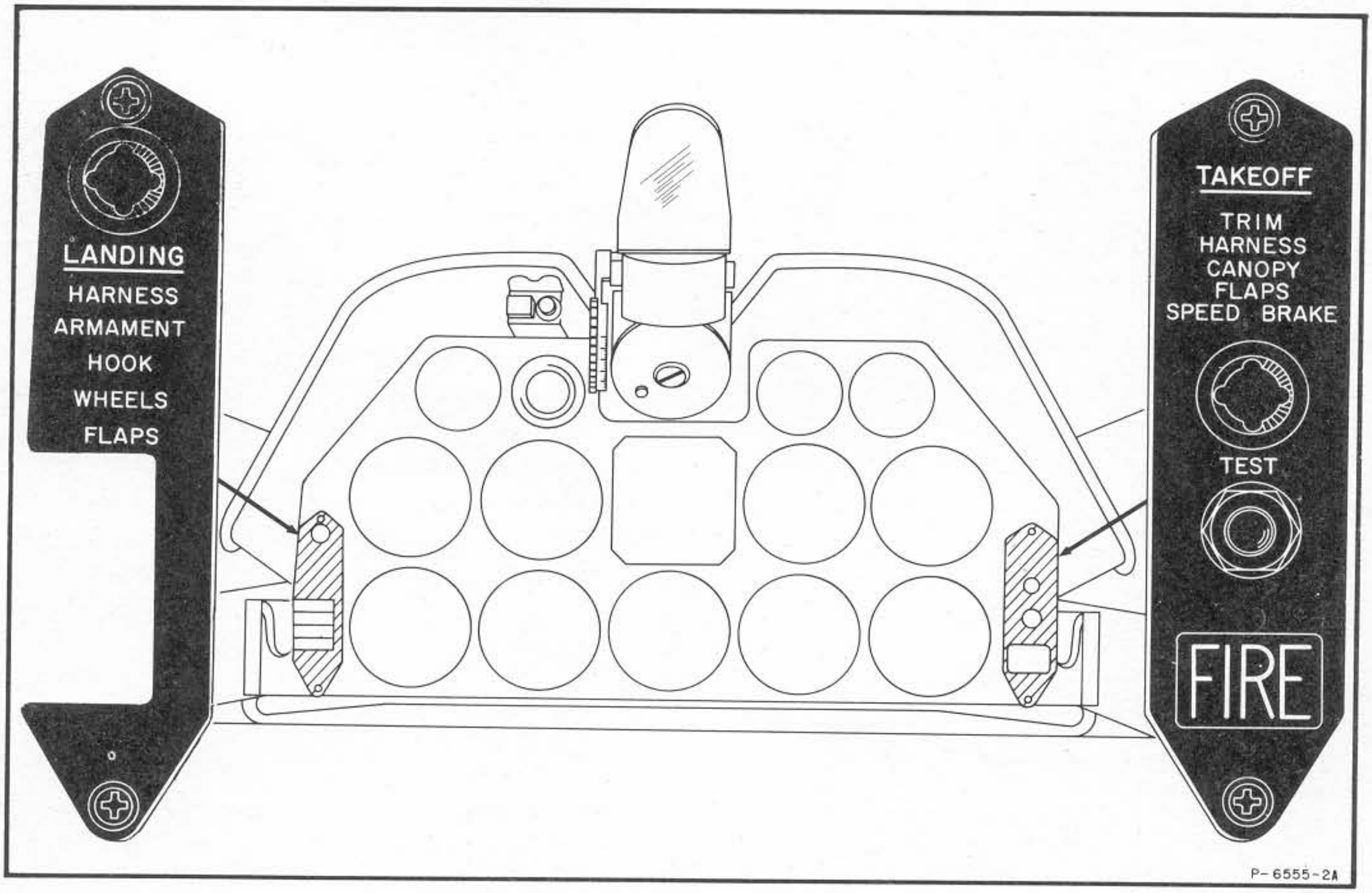
TAKE-OFF

AIRFIELD

Hold brakes and advance the throttle to the maximum thrust obtainable without sliding on the locked wheels. Check all instruments for satisfactory operation. When satisfied, advance the throttle to "MILITARY," release the brakes and begin the take-off run. Observe engine instruments for proper indications during the take-off run. Maintain direction by use of the brakes; only a slight pressure or "tapping" should be necessary for directional control. The rudder will become effective enough for steering at approximately 70 knots IAS. When the speed increases to approximately 115 knots IAS, the nose wheel should be lifted well off the runway to obtain higher lift. At normal take-off gross weight (clean configuration), the aircraft will become airborne at a minimum speed of 120 knots IAS. Refer to Section III for procedure to follow in the event of engine failure during take-off.

MINIMUM RUN

To accomplish a minimum run take-off, full nose up trim (6 units) and full flaps should be employed. After brake release, as the aircraft accelerates down the runway, a generous amount of aft stick should be used to effect a nose wheel lift-off at about 105 knots IAS. During air-



Group Bb, Bc, Bd, C, D and E Airplanes
Figure 2-3. Check-off Lists (Sheet 2)

plane lift-off, at approximately 110 knots IAS a nose up rotation of the aircraft will occur which will require an immediate reduction in aft stick pressure to control. As the airplane accelerates to climbing speed after take-off, almost constant re-trimming of the stabilizer will be necessary. This nose up rotation of the airplane at take-off is reduced in abruptness and severity by an increase in gross weight or by use of less airplane nose up trim. However, if less nose up trim is employed, the effect will be to increase the minimum nose wheel lift off speed about 4 knots per unit (2 degrees) of reduced nose up trim (less than full nose up trim), and to increase the take-off run approximately 300 feet per unit of reduced nose up trim.

CARRIER

When the aircraft is properly rigged on the catapult, and upon signal from the catapult officer:

a. Advance the throttle to "MILITARY."

b. THROTTLE FRICTION wheel . . . . . Full forward

c. Grasp the catapult handgrip in conjunction with the throttle.

d. Check all instruments for proper indications.

e. Signal the catapult officer when prepared for launching.

f. Immediately place head firmly back against the headrest, grasp the control stick and anchor the right elbow on the hip or stomach, keeping the feet against the lower portion of the rudder pedals.

CAUTION

Do not use brakes during catapulting.

g. Pilot technique for catapult launches consists of placing the control stick in the neutral position and holding it there throughout the launch. Very slight back stick pressure is required to maintain a level flight path off the bow near minimum end air speeds. This technique, plus the longitudinal trim tab settings listed below are mandatory for safe launches when end air speeds are within ten knots of the prescribed minimum.

Trim settings are as follows:

Gross Wt. Lb.	*Longitudinal Trim (Nose Up°)*	*Lateral Trim*	*Directional Trim*
13,850	10-12	Neutral	0
15,700	12 (Full nose up)	Neutral	0
17,600	12 (Full nose up)	Neutral	0
19,600	12 (Full nose up)	Neutral	0

AFTER TAKE-OFF

a. Leave the throttle at "MILITARY" until a safe altitude is reached.

b. Landing gear control . . . . "UP"

Note

When the nose gear has fully retracted, a vibration may be felt in the airplane. This is due to imbalance in the spinning nose wheel being transmitted through the nose gear strut when retracted against the stop.

c. FLAP control . . . . "UP"

Note

Almost constant trimming of the horizontal stabilizer will be necessary after take-off during the period of acceleration to best climbing speed.

CLIMB

Accelerate to the best climbing speed shown in Appendix I before starting climb. Use the airspeeds recommended in the chart to obtain the best rate of climb, however speeds may be varied 10 knots above or below those stipulated without appreciably affecting climb performance. Maintain "MIL" RPM throughout the climb for best results, observing at all times the engine rpm, exhaust temperature and time limitations as set forth in Section V.

FLIGHT CHARACTERISTICS

Refer to Section VI for information pertaining to the flight characteristics of the aircraft.

SYSTEMS OPERATION

Section VII contains information regarding operation of the various aircraft and engine systems.

DESCENT

Descents may be made very rapidly by using power and speedbrakes. Refer to Section V for limiting airspeeds, and to Section VI for diving characteristics of the aircraft. If a descent is desired in which maximum range is obtained, throttle back to "IDLE" RPM and maintain approximately 200 knots IAS during the let-down from altitude. See figure A-43.

PRE-TRAFFIC PATTERN CHECK LIST

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

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

b. Fuel quantity indicator . . . . Adequate supply for normal entry, approach and landing

c. GUNS switch . . . . "SAFE"

d. MASTER ARMAMENT switch and miscellaneous armament switches . . . . "OFF"

e. UHF radio . . . . Tower frequency

f. Master exterior lights switch . . . . On (night)

g. EXT LTS control panel . . . . All switches "BRIGHT" (night)

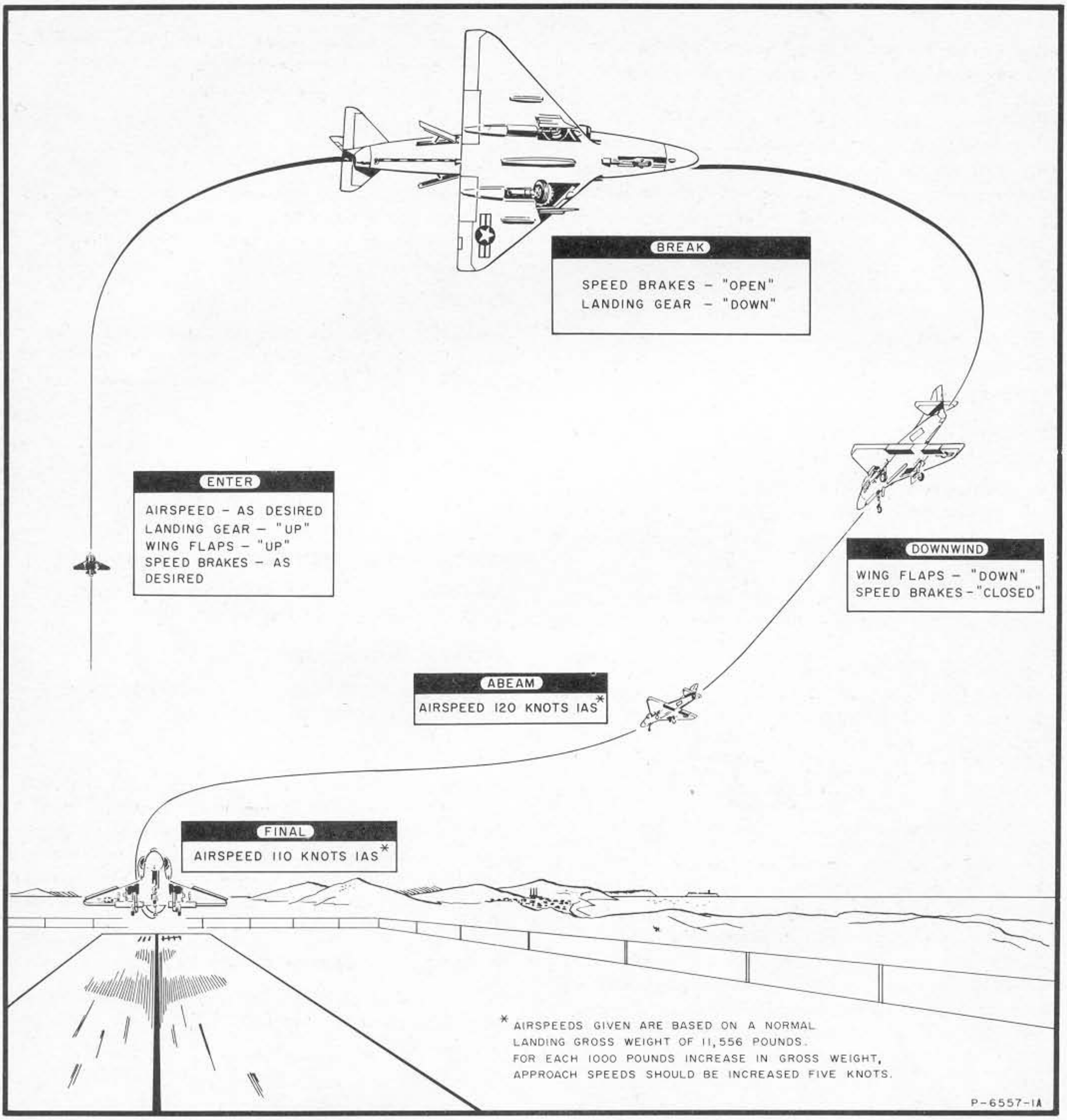


Figure 2-4. Landing Pattern Diagram

h. Trim . . . . As desired

i. SPEEDBRAKE switch . . . . As desired

TRAFFIC PATTERN CHECK LIST

a. Canopy . . . . Closed

b. HOOK control . . . . Field: "UP" Carrier: "DOWN"

c. Landing gear control . . . . "DOWN"

CAUTION

Slow movement of the landing gear handle, or failure to place the handle full down in the detent may cause damage to the landing gear doors. When lowering the landing gear use a brisk, positive, movement to full down and make sure the handle is in the detent.

d. FLAP control .... "DOWN"

e. SPEEDBRAKE switch .... "CLOSE"

Note

When normal wind conditions exist, full flaps should be used for landing on an airfield, but if high wind conditions are prevalent, use half flaps. Full flaps are always used for landing aboard a carrier.

LANDING

AIRFIELD

At the basic design landing gross weight with power off and wheels and flaps down, the stalling speed is approximately 100 knots IAS. Recommended approach speed is 120 knots IAS at the abeam position (see figure 2-4). Begin the turn into the base leg at a point slightly downwind of the landing end of the runway in order to have adequate straightaway for corrections of the final approach. Gradually reduce airspeed to approximately 110 to 115 knots IAS on the final leg. Long, low, flat approaches are unnecessary as the aircraft decelerates rapidly upon taking the "cut," particularly when a combination of a low approach speed and high power setting is used. Full nose-up elevator is required to hold the nose wheel off at touchdown, which should be made at approximately 105 knots IAS. After touchdown, the nose wheel can be held off only to a speed of 5 to 10 knots below that of landing.

CAUTION

For each 1000 pounds increase in the gross weight of the aircraft, the approach speeds should be increased by five knots.

MINIMUM RUN. A landing requiring the minimum amount of runway distance can be accomplished by simulating a carrier approach to the airfield. Make the approach as slow as possible, but maintain a safe margin above stalling speed. Attempt to cross the landing end of the runway at an altitude of 5 to 10 feet. Immediately retard the throttle and land. At touchdown, retract the wing flaps to reduce lift, thus producing more braking action from the wheels. When the nose wheel drops to the runway, apply wheel brakes and open the speedbrakes to increase the drag.

CROSS-WIND. Use standard cross-wind approach methods to effect a landing. In light crosswinds the aircraft tracks well after touchdown and can be easily controlled directionally by use of the wheel brakes. In crosswinds above 15 knots, however, the upwind wing rises and the nose of the aircraft tracks toward the downwind side instead of "weather-cocking" into the wind as is normally to be expected. To correct for this characteristic the following procedure is recommended immediately after touchdown:

a. Raise the flaps to reduce lift of the wing.

b. Apply full aileron into the wind. (Wind from the right—right aileron.)

c. Simultaneously with aileron, apply full down elevator to further reduce lift of the wing by decreasing the angle of attack.

d. Apply maximum braking effort obtainable without skidding the tires, to stop the aircraft as rapidly as possible.

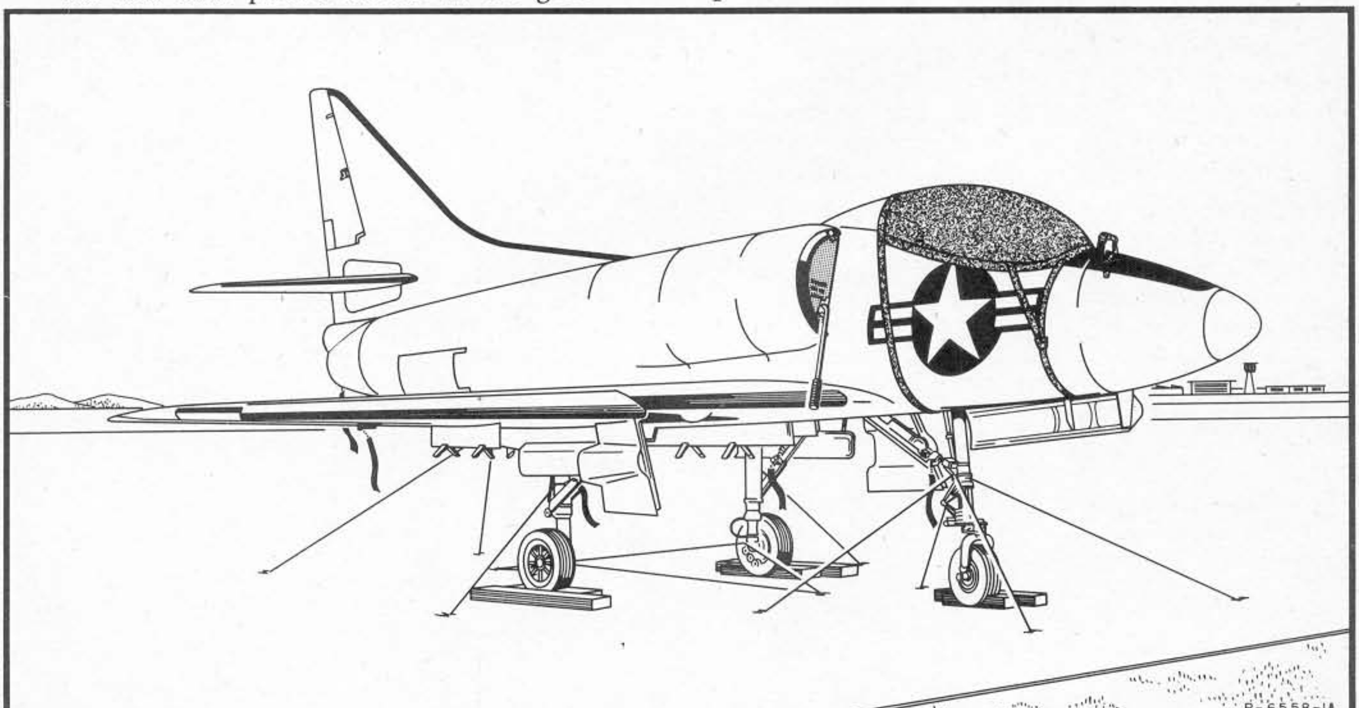


Figure 2-5. Mooring

If either wing rises it is easy to lock the wheel on the side of the raised wing as a result of braking action and the decreased weight on the landing gear. Extra caution should be exercised when applying the brake on the upwind side of the aircraft when the wing rises. If the upwind wing comes down due to aileron deflection into the wind, reduce the aileron throw to maintain the airplane level laterally.

CARRIER

The downwind leg of the carrier landing pattern should be sufficiently abeam of the carrier to permit a constant bank angle of 17 to 18 degrees throughout the base leg approach. When landing gross weight is under 11,000 pounds it is important to be at final approach speed and on altitude well out on the base leg because power required is in an RPM range where engine acceleration lags behind throttle movement making precise power adjustments difficult. Speed brakes are ineffective in the light gross weight range and longitudinal trim rate is slow. The recommended final approach speed is 115-118 knots IAS at a gross weight of 11,000 pounds. Add or subtract to this figure, 5 knots for each 1,000 pound increase or decrease in gross weight. In the final approach, mild lateral oscillations will be encountered from turbulence created by the ship. This phenomenon has no effect on the airplane flight path and should be ignored since corrective aileron deflections invariably are out of phase and serve to aggravate rather than alleviate this condition.

CAUTION

- Carrier landings on angled deck ships should be made with particular attention to achieving a good line up and avoiding landings with right to left drift which, when associated with the increased runout of angled deck arresting gears, can result in the aircraft coming to rest in the port catwalk even though a pendant is engaged.
- A burble effect, present under all wind conditions, produces a definite tendency for the right wing to drop as the airplane approaches and passes the round-down at the forward end of the landing area on angled deck ships.

WAVE-OFF

If it becomes necessary to abort the landing,

a. Advance the throttle rapidly but smoothly to .... "MILITARY"

b. Landing gear control .... "UP"

c. FLAP control .... "UP" at a safe altitude

Note

Under standard sea level conditions, the engine will accelerate from idle to military rpm in from 8 to 10 seconds.

AFTER LANDING

The throttle should be kept at "IDLE" for minimum thrust, and the speedbrakes may be opened for additional drag after landing. Use the wheel brakes as necessary to slow the aircraft, applying pressure lightly and evenly to prevent excessive wear and the possibility of blowing the tires. When clear of the runway:

a. FLAP control .... "UP"

b. SPEEDBRAKE switch .... "CLOSE" if used

c. AIR COND switch............ "OFF"

d. Canopy .... Open if desired and relative headwind is less than 60 knots

STOPPING THE ENGINE

If the engine is to be stopped from above "NORM" RPM, pause at "IDLE" for at least one minute to allow the exhaust temperature to stabilize, then move the throttle to "OFF." From below "NORM," the throttle may be moved immediately to "OFF."

BEFORE LEAVING THE AIRCRAFT

a. All electrical equipment .... Off

b. OXYGEN switch .... "OFF"

MOORING

Ascertain that the following action has been taken:

a. Wheels chocked.

b. Rudder and aileron gust locks installed.

c. Engine intake and exhaust duct covers inserted.

d. Cockpit cover installed.

e. Pitot cover in place.

f. Aircraft secured to the deck with lines (see figure 2-5).

SECTION III
EMERGENCY PROCEDURES

ENGINE FAILURE

Evidence that the engine is malfunctioning or that engine failure is imminent may be recognized by the following indications:

- Loss of thrust
- Fluctuating rpm and exhaust temperature
- Drop in engine oil pressure and/or fuel boost pressure
- Vibration

When any of these symptoms appear, reduce power if conditions permit, and land as soon as possible.

PROCEDURE ON ENCOUNTERING ENGINE FAILURE

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

a. Throttle .... "OFF"

b. EMER GEN release handle .... Pull

Note

- It is necessary to activate the emergency generator when the engine is not operating to provide electrical power for the FIRE warning light.
- Once the emergency generator has been extended it cannot be retracted while in flight.

c. Check for evidence of fire by noting the following points:

- FIRE warning light illuminated
- Presence of smoke or fumes in cockpit
- Emission of smoke from tailpipe or other areas

WARNING

If fire is present or existed prior to shut-down, do not attempt to restart the engine (refer to FIRE).

c. If considered safe, start the engine in the manner prescribed under AIR START.

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

a. Throttle .... "IDLE"

b. FUEL CONTROL switch....... "MANUAL"

CAUTION

When safety of flight necessitates, the transfer to "MANUAL" fuel control may be made at any throttle setting but, whenever possible, should be made with the throttle at "IDLE" rpm. The "MANUAL" control may be selected at any altitude, but should not be used just for practice.

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

CAUTION

- At altitude on "MANUAL" fuel control it will be possible to obtain 100% rpm with less than full throttle travel. Small movements of the throttle may result in large changes in rpm and EGT.
- Do not return the FUEL CONTROL switch to "PRIMARY" while in flight.

If engine fuel control unit failure results in a *flame-out,* and the engine has decelerated to windmilling rpm it may not be possible to obtain a re-light using the "MANUAL" fuel control system. No d-c electrical power is available to energize the "MANUAL" fuel control solenoids when the main generator has failed, and may not be available when the generator is being motored by engine windmilling rpm. D-c electrical power will also be lost if the transformer-rectifier has failed, or when the emergency generator is extended. Conversely, if the engine is being operated on the "MANUAL" fuel control system, due to "PRIMARY" system failure, and d-c electrical power is interrupted for any of the foregoing reasons, the de-energized "MANUAL" fuel control solenoids will automatically transfer the fuel control system back to "PRIMARY." No corrective action can be taken.

AIR START

Air starts may be accomplished within an engine windmilling range of 10 to 24% rpm. It may be difficult, however, to obtain a successful air start above 25,000 feet. Listed below are the approximate maximum and minimum indicated airspeeds, at various altitudes, which will furnish windmilling rpm within air starting requirements.

TABLE I

Pressure Altitude	*IAS Range*
Sea Level	215 Knots and above
5,000	195 Knots and above
10,000	175 Knots and above
15,000	160-220 Knots
20,000	160-220 Knots
25,000	160-220 Knots
30,000	160-220 Knots
35,000	160-210 Knots

It is recommended that the higher airspeed values be used for more consistent starts. The procedure for a normal air start is as follows:

a. Throttle .... "OFF"

b. Descend to 20,000 feet or below if practical.

c. Airspeed .... As required for proper engine rpm (See TABLE I)

d. FUEL CONTROL switch .... "PRIMARY"

e. Nose up slightly to drain fuel. (Do not allow air speed to fall below 160 knots.)

f. AIR START switch .... "ON"

g. Throttle .... "IDLE"

h. If relight has not occurred within 30 seconds, abort the start and repeat the air start procedure at a lower altitude.

i. AIR START switch .... "OFF" when relight is completed

ENGINE FAILURE UNDER SPECIFIC CONDITIONS

DURING TAKE-OFF

BEFORE AIRBORNE. Should the engine fail during take-off, but prior to becoming airborne:

a. Throttle .... "OFF"

b. FLAP control .... "UP"

c. Apply brakes as necessary to stop.

d. SPEEDBRAKE switch .... "OPEN" to aid in stopping

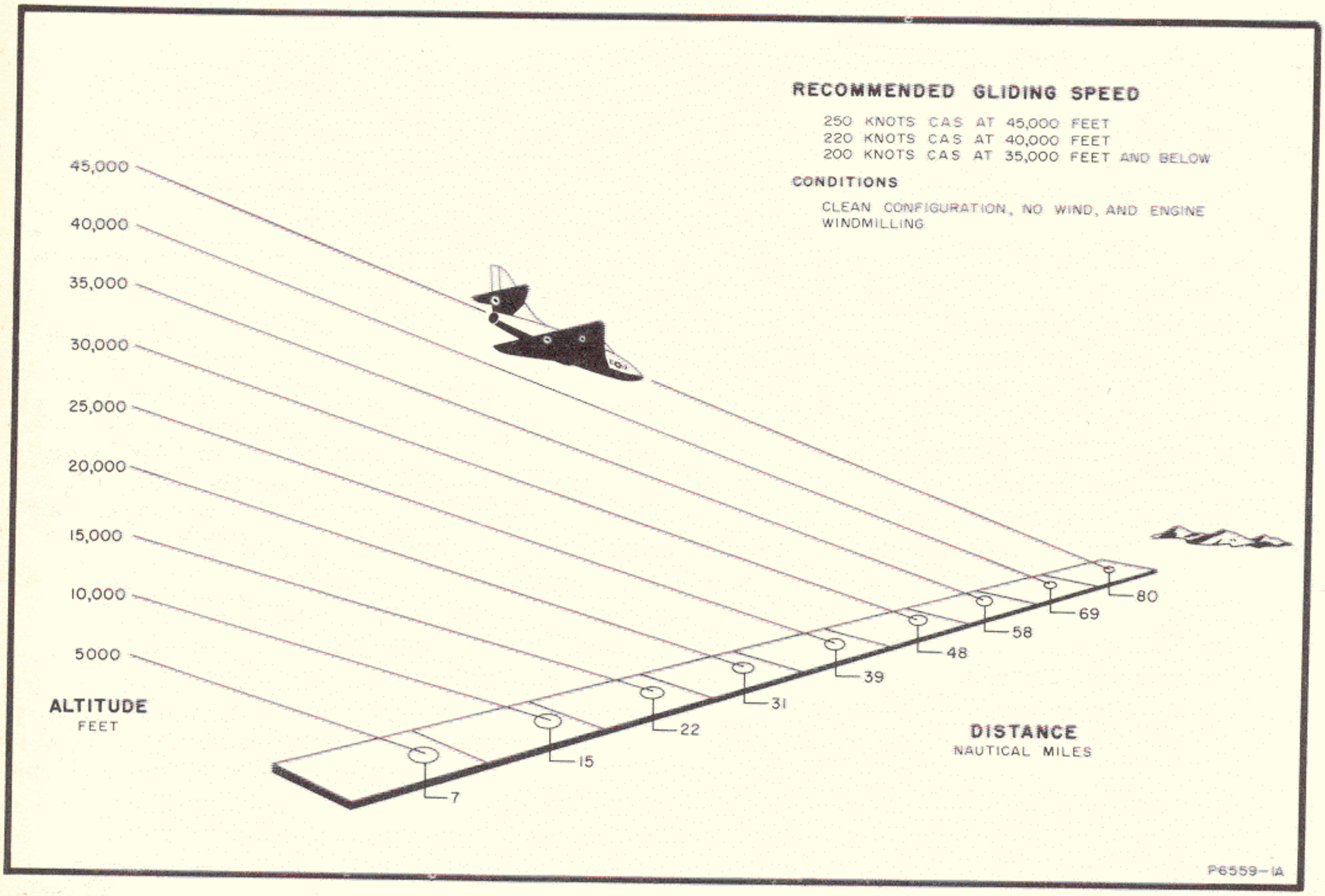


Figure 3-1. Maximum Glide

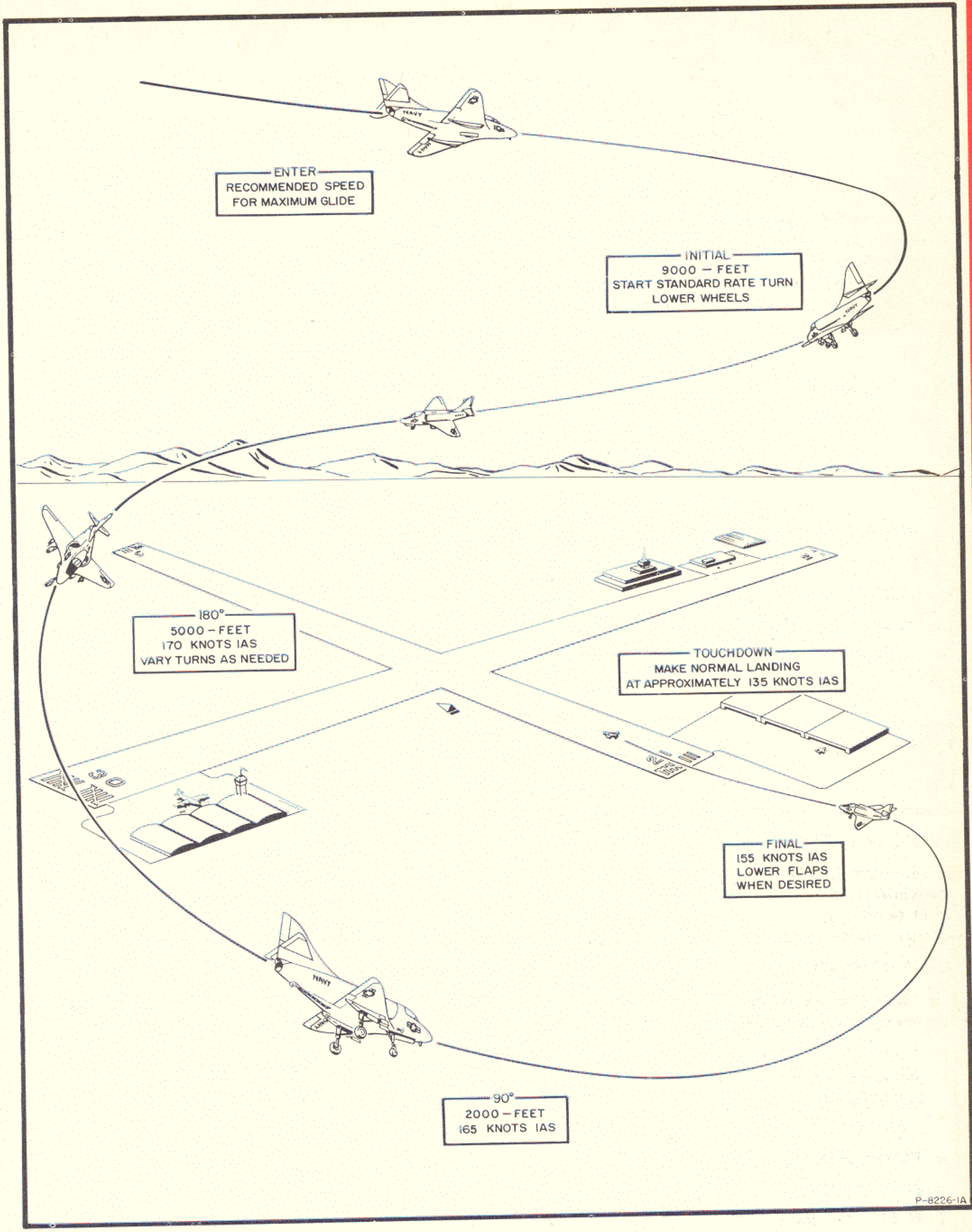


Figure 3-1A. Flame-out Landing Pattern Diagram

e. If the necessary stopping distance (refer to Appendix I) is not available and hazards exist at the end of the runway, the landing gear may be retracted by actuating the retraction release solenoid and moving the landing gear control to "UP."

AFTER AIRBORNE. If the engine fails after becoming airborne during take-off:

a. MAINTAIN FLYING SPEED.

b. Immediately place the FUEL CONTROL switch at "MANUAL" and retard the throttle enough to keep the tailpipe temperature within limits in the event the engine should continue to operate. Advance throttle slowly as the temperature permits. If the engine does not respond, complete the following steps.

c. Throttle .... "OFF"

d. Jettison all external stores

e. Landing gear control .... "UP" if insufficient runway remains for a "wheels down" landing

f. FLAP control .... "DOWN"

g. Land straight ahead.

DURING FLIGHT

a. Throttle .... "OFF"

b. Attempt to start the engine as outlined under AIR START.

c. If the air starting attempts are unsuccessful, perform the procedure set forth in LANDING WITH NO POWER and maintain the recommended gliding speed for maximum range (refer to MAXIMUM GLIDE). Should engine failure occur at night over unfamiliar territory, bail out if the engine cannot be started.

MAXIMUM GLIDE

The recommended speed for maximum gliding range is approximately 200 knots IAS for gross weights up to 16,000 pounds in a clean configuration. See figure 3-1 for approximate gliding ranges from various altitudes.

LANDING WITH NO POWER

If a landing is necessary because of engine failure, the following procedure should be used:

a. Throttle .... "OFF"

b. Jettison all external stores.

c. SHOULDER HARNESS control .... "LOCKED"

d. EMER GEN release handle .... Pull

e. POWER BOOST REL handle . . Pull

f. FUEL valve control[1] .... "EMERG OFF"

g. Extend the control stick.

Regulate the aircraft gliding speed at that recommended for maximum range. Attempt to arrive over the intended landing area headed into the wind at an altitude of approximately 9000 feet above the terrain. Immediately start a standard rate (two-minute) turn to the left and lower the landing gear if the area is sufficiently firm. After extending the wheels, adjust the gliding speed to 170 knots IAS. The abeam position should be reached at an approximate altitude of 5000 feet, and as the turn onto the base leg is begun, decrease the airspeed to 165 knots IAS. The turn to the base leg should be varied according to the direction and velocity of the wind, but attempt to reach the 90 degree position with approximately 2000 feet altitude. Gradually slow to 155 knots IAS on the final approach and maintain this airspeed throughout the final to insure adequate speed for breaking the glide without incurring an excessive sinking rate. If the engine is windmilling, the wing flaps may be used when desired, however flap extension time will be considerably increased. Following the flare-out, making a touchdown at approximately 135 knots IAS and retract the wing flaps, if used, to increase braking action. See figure 3-1A for a typical flame-out landing pattern.

Note

- All airspeeds given were established using the basic flight design gross weight of the aircraft (12,504 lbs.). Add approximately five knots for each 1000 pounds increase above this weight.
- The wheels and flaps position indicator will show an "unsafe" condition when operating on the emergency generator, even though the wheels are down and locked; however, the warning light in the landing gear control handle will glow until the gear is locked down.

FIRE

ENGINE FIRE

ON THE GROUND. If fire develops in the engine while on the ground, perform the following:

a. Throttle .... "OFF"

b. Make certain that the starter has a source of power.

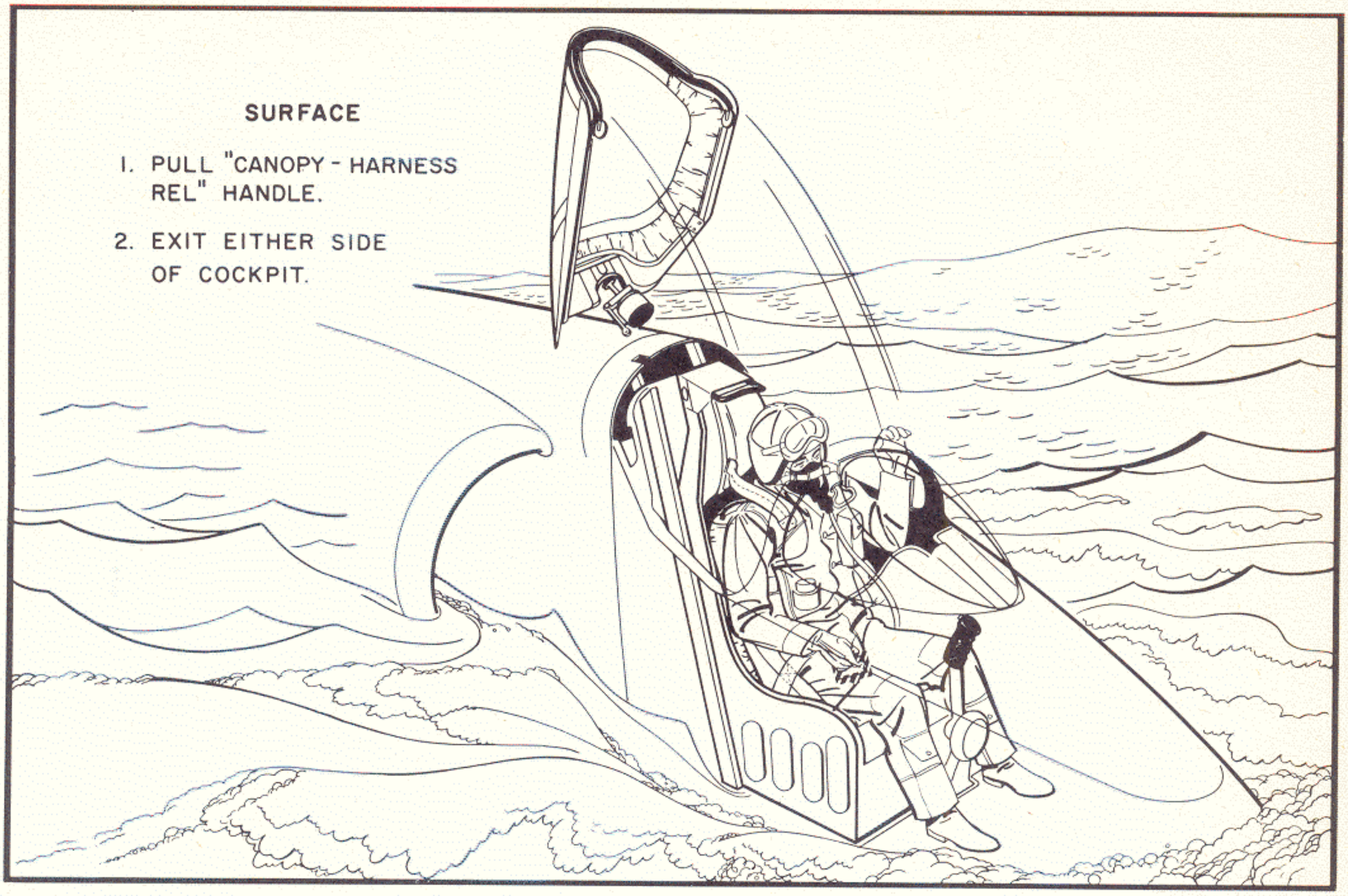
c. START-ABORT switch .... Depress

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, cranking may intensify combustion by circulating engine compartment cooling air.

d. Allow the starter to motor the engine until the fire has disappeared. If fire persists for a considerable

[1]Group C, D and E Airplanes.



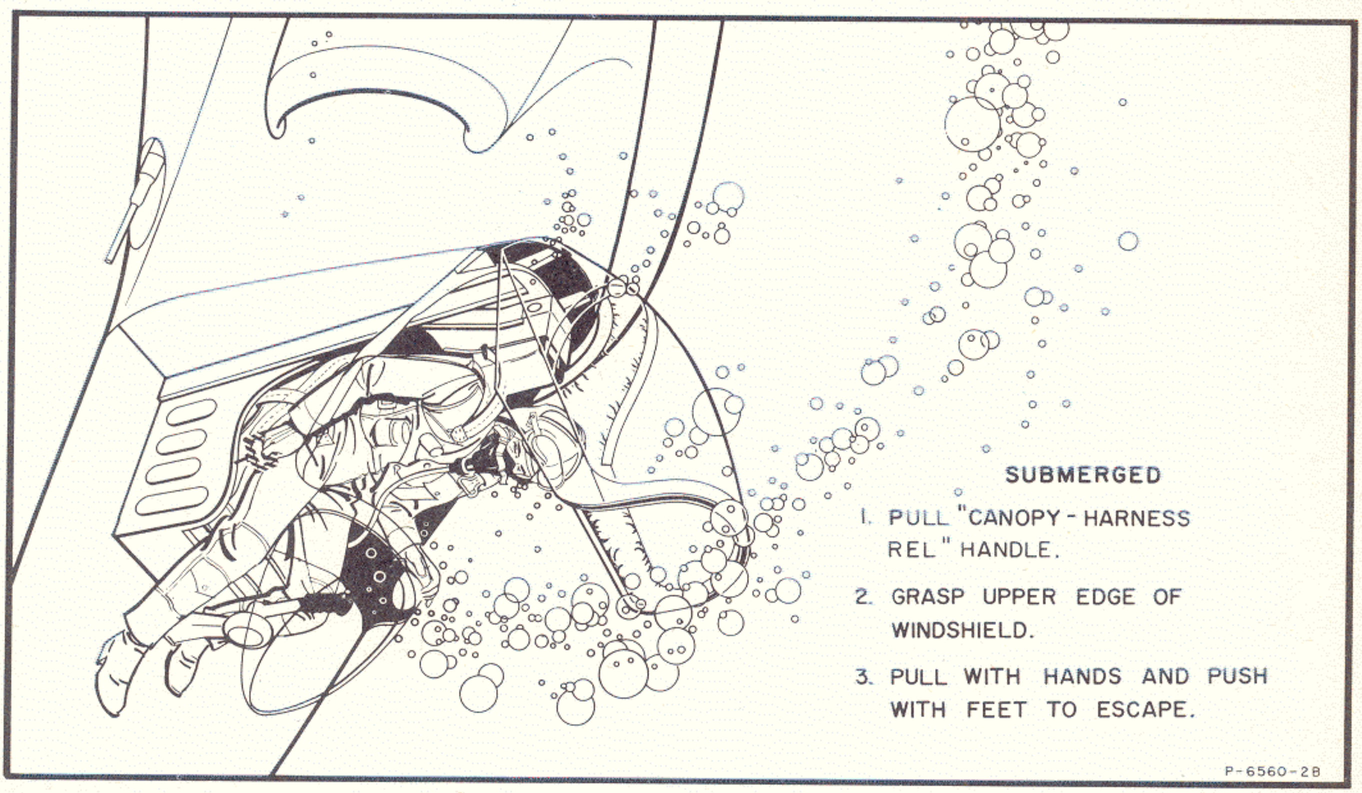


Figure 3-2. Ditching Procedure

length of time, continue cranking the engine and signal the fire guard to apply CO_2 to the engine air intake duct as required.

e. FUEL valve control .... "EMERG OFF," if fire persists

IN THE AIR. Should an engine fire occur during flight, proceed as follows:

a. Throttle .... "OFF"

b. EMER GEN release handle .... Pull

c. If the fire is extinguished by movement of the throttle to "OFF", as evidenced by the FIRE warning light going out, or by the disappearance of smoke, fumes, or actual fire, do not attempt to restart the engine. The decision to attempt an emergency landing with no power or to abandon the aircraft remains with the pilot.

d. Should the fire continue to burn after moving the throttle to "OFF," move the FUEL valve control to "EMERG OFF." If this procedure does not extinguish the fire, it is up to the pilot to elect forced landing or ejection.

FUSELAGE FIRE

There is little that can be done to combat a fire in the fuselage during flight, as no fire fighting equipment is available. There is always the possibility that the fire could be electrical in origin, however, and the procedure stated in ELECTRICAL FIRE should be followed.

WING FIRE

Because of the location of the integral wing fuel tank, a fire in the wing could be caused by either fuel leakage or defective electrical wiring.

a. Jettison all combustible external stores.

b. Follow the procedure as stated under ELECTRICAL FIRE.

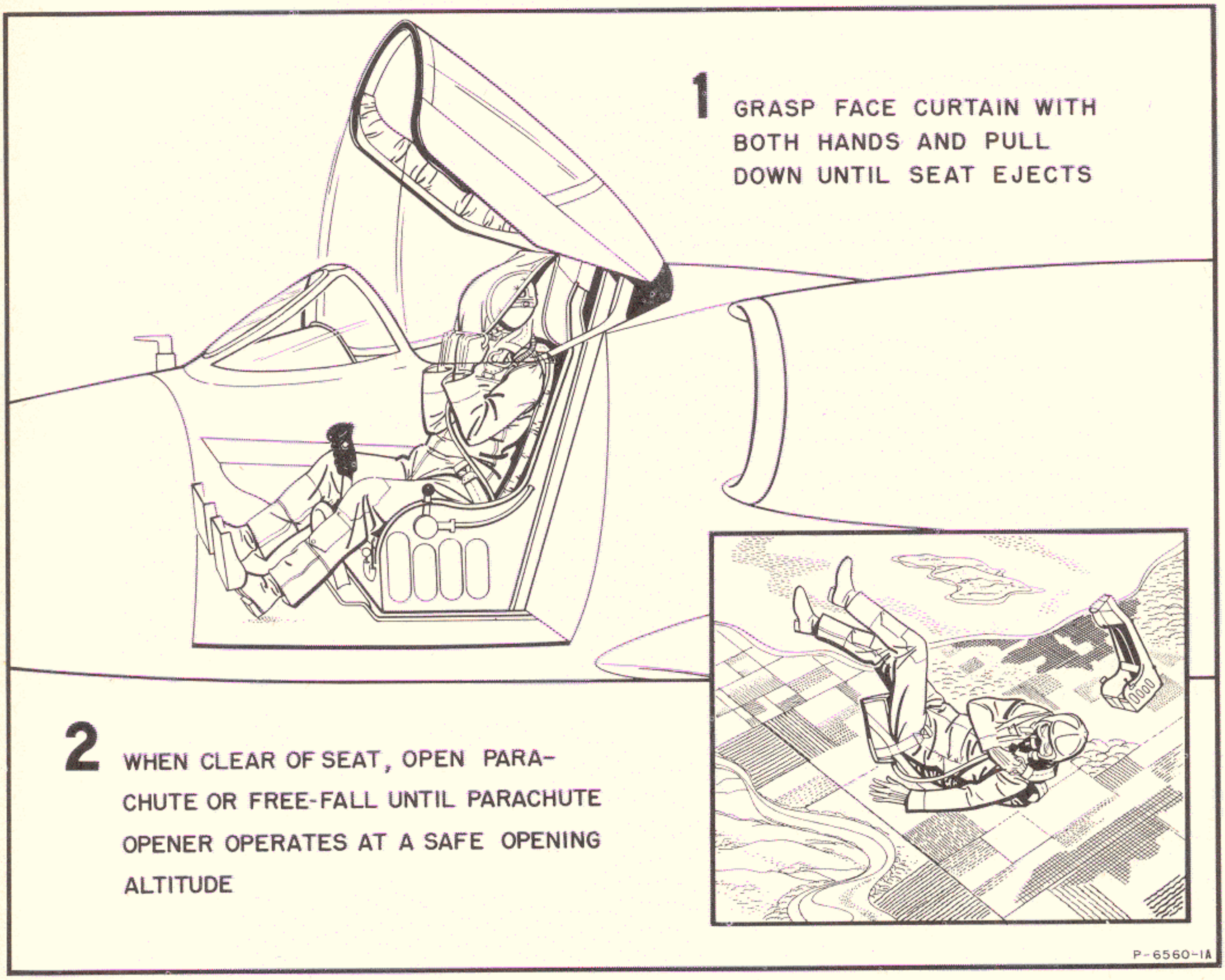


Figure 3-3. Ejection Procedure

c. If the fire continues to burn, eject, as the fire is probably fuel-fed and there is no way of stopping the fuel leakage while in flight.

ELECTRICAL FIRE

When a fire is suspected of being electrical in origin, proceed as follows:

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 turn the necessary equipment on.

e. If the fire cannot be extinguished in this manner, the pilot must use his own discretion in determining whether to abandon the aircraft or attempt an emergency landing.

SMOKE ELIMINATION

If smoke or irritating fumes are present in the cockpit, turn the air conditioning switch to "OFF."

LANDING EMERGENCIES (EXCEPT DITCHING)

When an emergency landing is necessary and power is available, more deliberation can usually be given in evaluating the many variables affecting a safe landing than when the engine is inoperative. Careful consideration should be taken of the direction and velocity of the wind and the type of terrain in which the landing is to be made. On areas other than prepared runways and surfaces of known adequate hardness, the landing should always be made with the wheels retracted. If the nature of the emergency is such that all wheels will not fully extend, then the landing gear should be left in the retracted position. Prior to any "wheels up" landing, all external stores should be jettisoned. Use the recommended approach speeds throughout the landing pattern, and attempt to touch down at, or slightly above, the recommended landing speed.

CAUTION

Airspeeds throughout the approach and landing should be increased five knots for each 1000 pounds increase in the weight of the aircraft over the normal landing gross weight.

EMERGENCY ENTRANCE

When it is necessary to gain entrance to the cockpit in an emergency, it might also be necessary to affect the quick and safe removal of the pilot from his seat and parachute. This entry and rescue operation requires that certain procedures be followed and that certain precautions be taken. Since the procedures are not the same for all aircraft, they will be discussed separately.

EARLY AIRCRAFT[1]

The external CANOPY RELEASE handle must be actuated to gain entrance to the cockpit. If the aircraft is inverted or for any other reason the canopy will not open, use an axe or other tools to smash the canopy and accomplish entry. To free the pilot from his seat and parachute, unfasten the harness attach fittings.

WARNING

If the canopy is open, do not pull the CANOPY-HARNESS REL handle to free the pilot from the seat. When the compressed air cylinder is expended, the canopy will usually slam shut, causing injury to anyone leaning across the cockpit rail.

LATER AIRCRAFT[2]

An external control for jettisoning the canopy is provided on each side of the fuselage. Push in on the RESCUE access door and pull the red PULL CANOPY JETTISON handle that extends: this will fire the canopy if it is closed and locked.

WARNING

When the canopy is jettisoned, the seat catapult safety pin is extracted, and care must be exercised to avoid firing of the seat catapult charge.

To release the pilot from the seat and parachute, unfasten the harness attach fittings or pull the CANOPY-HARNESS REL handle.

Note

When the canopy is open or unlocked, a disconnect mechanism prevents firing of the 2500 psi compressed air cylinder, thus eliminating danger to rescue personnel from canopy motion when the CANOPY-HARNESS REL handle is pulled.

DITCHING

In preparation for ditching, the following checks should be performed:

a. Jettison all external stores.

b. SHOULDER HARNESS control .... "LOCKED"

c. Lower the seat.

d. Landing gear control .... "UP"

[1] Group A, B, C, Ea and Eb Airplanes.
[2] Group D, Ec and Ed Airplanes.

e. HOOK control ...."DOWN"

f. FLAP control ...."DOWN"

If power is available prior to ditching, use up as much fuel as possible. This will not only reduce the aircraft weight, but will create added buoyancy by nearly emptying the fuel tanks. Ditch while power is still available, however, in order to be able to choose the spot where best sea conditions exist. Ditch along the top of, and parallel to a row of swells. By lowering the arresting hook, some indication will be given of the altitude of the aircraft when the hook strikes the water. At the time of contact attempt to have the lowest possible forward speed consistent with good control. Do not drop the aircraft in, as the possibility of bouncing and subsequent "nosing in" always exists. Attempt to hold the nose up after the first contact with the water, and when the aircraft comes to a complete stop, pull the CANOPY-HARNESS REL handle to jettison the canopy and escape.

WARNING

- The CANOPY-HARNESS REL handle "D-ring" should not be pulled 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.
- When the canopy is jettisoned, the seat catapult safety pin is extracted; care must be exercised to avoid firing of the catapult charge.

The canopy should be locked when the CANOPY-HARNESS REL handle is pulled. If it is not locked on early airplanes[1] the canopy will not jettison, and might even become jammed in a partly-open position when the compressed air cylinder fires. In later airplanes[2] the canopy will not jettison at all unless it is locked. Should the aircraft "nose in" or overturn, the canopy will not jettison, but will open upon actuation of the CANOPY-HARNESS REL handle. In this case, since the canopy is not actually jettisoned, the catapult safety pin will not be pulled.

EJECTION

Escape from the aircraft can be accomplished from all attitudes and airspeeds through use of the ejection seat, however, the following procedure is recommended if time permits:

a. Turn air conditioning system off.

b. Slow the aircraft as much as possible.

c. Leave feet on the rudder pedals.

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

e. Lock the shoulder harness.

f. Grasp the face curtain handle with both hands and pull down to the fullest extent.

g. When free of seat, open the parachute, fall to a safe altitude before opening the parachute, or fall until the parachute opener functions.

h. After entering the water or landing on the ground, open the disconnects at the shoulder to release the parachute risers from the harness to prevent being dragged when a strong wind is blowing.

WARNING

Do not disconnect the parachute riser releases until a safe descent and landing have been made. A cross-connector between the parachute risers will prevent spilling of the canopy if one release is inadvertently opened.

When conditions are such that ejection must be made without the slightest hesitation, merely grasp the face curtain handle and pull down. (See figure 3-3.)

AIRCRAFT SYSTEMS

FUEL SYSTEM

FUEL BOOST PUMP. Failure of the fuel boost pump will result in a reading of "OUT" in the FUEL BOOST window of the engine performance indicator. Gravity flow of fuel to the engine-driven fuel pump will assure the engine an adequate supply of fuel for all power conditions below 6000 feet altitude, but negative or zero "g" flight should be avoided to prevent a break in fuel flow and subsequent flame out.

TRANSFER PUMP. If the wing tank transfer pump fails, the only fuel available will be that in the fuselage fuel tank. In later aircraft,[3] fuel transfer pump failure will be indicated by illumination of the FUEL TRANS warning light on the instrument panel. In earlier aircraft[4] however, the first indication of transfer pump failure will be an abrupt drop in fuel quantity indication to a reading of approximately 1100 pounds (the fuel level at which the wing tank capacitance units are removed from the indicating circuit). When this occurs, land as soon as possible to prevent fuel exhaustion.

ENGINE-DRIVEN FUEL PUMP. Either gear stage of the engine-driven fuel pump is capable of supplying the engine with enough fuel for all operations. Should one gear stage fail, a bypass routes the fuel flow around the defective gear stage, allowing the other stage to function without interference. There will be no indication to the pilot of such failure.

(1) Group A, B, C, Ea and Eb Airplanes.
(2) Group D, Ec and Ed Airplanes.
(3) Group Bd, C, D and E Airplanes.
(4) Group A, Ba, Bb and Bc Airplanes.

FUEL CONTROL UNIT. A malfunction of the fuel control unit may result in fluctuating rpm and exhaust temperature, or a complete loss of power. When any of these occur, proceed as follows:

a. Throttle .... "IDLE"

b. FUEL CONTROL switch ...... "MANUAL" when the RPM begins to decrease

c. Check MANUAL FUEL warning light illuminated.

d. Advance the throttle slowly and smoothly to the desired power setting.

CAUTION

When operating on the manual fuel control system, it is mandatory that the throttle be moved slowly and smoothly to prevent over-speeding of the engine and excessive exhaust temperatures.

OIL SYSTEM

Failure of the oil system will be indicated by a reading of "OUT" in the OIL PRESS window of the engine performance indicator. Continued engine operation should be governed by the following:

a. Absence of oil pressure for a maximum period of one minute is permissible and no damage should result from continued operation, providing "NORM" oil pressure is indicated after such period.

b. Lack of oil pressure for a period over one minute, but less than 2½ minutes, allows the flight to be continued at the pilot's discretion; however, the engine should be inspected in accordance with existing instructions.

c. If oil pressure is not indicated for a period exceeding 2½ minutes, the flight should be terminated as soon as possible, and the engine should be removed for overhaul.

Note

Electrical power failure will also cause the OIL PRESS window to read "OUT"; this will occur, however, in conjunction with other symptoms of electrical failure. Refer to ELECTRICAL SYSTEM in this section.

ELECTRICAL SYSTEM

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

- FUEL BOOST .... "OUT"
- OIL PRESS .... "OUT"
- Wheels and flaps position indicator .... Wheels: Unsafe Flaps: OFF
- Attitude gyro indicator .... OFF warning flag visible
- Radio failure.
- All lights out.

When the main generator fails, pull the EMER GEN release handle.

CAUTION

Do not release the emergency generator at airspeeds above 500 knots IAS or .91 indicated Mach number to prevent damage to the generator.

TRANSFORMER-RECTIFIER. If the transformer-rectifier fails, no d-c power will be available except for the S-2 compass and the landing gear handle warning and approach light relays. Failure will be apparent by the malfunctioning of all d-c powered equipment, but with no discrepancies in the operation of a-c powered units. Should the transformer-rectifier fail, the following will occur:

- FUEL BOOST .... "OFF"
- OIL PRESS .... "OFF"
- Wheels and flaps position indicator .... Wheels: Unsafe Flaps: OFF
- Armament equipment .... Inoperative
- Trim position indicator .... Inoperative

No action can be taken to remedy the situation.

HYDRAULIC SYSTEM

Failure of any hydraulic system will render all equipment serviced by that system inoperative. Refer to Section I, HYDRAULIC POWER SUPPLY SYSTEM, for a list of equipment operated by hydraulic system pressure. Complete hydraulic system failure, in airplanes equipped with the single hydraulic system,(1) is indicated to the pilot by illumination of the UTIL HYD warning light and by stiffening of the controls due to loss of the flight control power systems. On airplanes utilizing the tandem hydraulic system,(2) illumination of either the UTIL HYD warning light or the CONT HYD warning light will indicate loss of pressure to one or the other system. No stiffening of the control stick will be encountered with the tandem system except with complete failure of both the flight control and the utility hydraulic systems, with both warning lights illuminated.

Note

In tandem hydraulic system airplanes the rudder will stiffen in the event of flight control system failure, because it is not served by the utility system.

(1) Group A, B, C and D Airplanes.
(2) Group E Airplanes.

No means is available to the pilot to correct hydraulic system failure. For action to be taken in the event of failure, refer to FLIGHT CONTROL SYSTEM in this section.

FLIGHT CONTROL SYSTEM

In the event of flight control system failure, due to loss of pressure in the single hydraulic system or complete failure of both systems in the tandem installation, the following procedure should be followed:

a. Terminate any accelerated maneuver.

b. Trim the aircraft laterally.

Note

An aileron accumulator, used only on single hydraulic system airplanes[1], can furnish adequate hydraulic pressure for several actuations of the ailerons after hydraulic failure. As aileron accumulator pressure diminishes, stick forces will progressively grow "stiffer," and trimming should be accomplished before this time.

c. POWER BOOST REL handle . . . . Pull

Note

- The POWER BOOST REL handle should not be pulled on airplanes equipped with the tandem hydraulic system unless both the flight control and utility systems have failed.
- The POWER BOOST REL handle should be pulled fully out with a rapid and positive motion. This action will assure a "clean" disconnect and will aid in preventing a tendency for the airplane to roll as the switch to manual control is made.
- The POWER BOOST REL mechanism cannot be reset in flight once it has been disengaged.

d. Extend the control stick.

(1)Group A, B, C and D Airplanes.
(2)Group E Airplanes.

e. Reduce airspeed.

Note

Control stick forces are very high on manual control except at low speeds.

f. Terminate flight as soon as practicable.

SPEEDBRAKES

In the event of speedbrakes control valve solenoid or d-c electrical failure, operate the speedbrakes as follows:

a. SPEEDBRAKE switch . . . . "OPEN" or "CLOSE" as required

b. EMER SPEED BRAKE control . . "Pull" to open, or "push" to close, as required

In the event of hydraulic system and electrical failures with the speed brakes open they may be closed by the manual control.

LANDING GEAR SYSTEM

Utility hydraulic system failure will necessitate lowering the landing gear by the manual release system.

a. Airspeed . . . . 130 knots, but not to exceed 225 knots

b. Landing gear control . . . . "DOWN"

c. EMER LDG GR release handle . . Pull

d. If the landing gear does not fully extend or lock down, increase the airspeed.

CAUTION

After lowering the landing gear by the emergency release system, the gear should not be retracted until the ratchet in the emergency release system has been reset. Failure to comply will result in failure of the ratchet.

SECTION IV

AUXILIARY EQUIPMENT

AIR CONDITIONING AND PRESSURIZATION SYSTEM

A combination air conditioning and pressurization system heats, cools, ventilates, and pressurizes the cockpit. The system comprises an air cycle system refrigeration unit, cockpit pressure regulator, pressure relief valve, and temperature control components. (See figure 4-1 for a schematic of the air conditioning and pressurization system.)

AIR CONDITIONING

Hot high pressure air is bled from the engine compressor section, and is ducted either through or around the refrigeration unit as governed by a cockpit temperature controller. Air passing through the refrigeration unit is directed through a heat exchanger and turbine where it is expanded and cooled. The cooled air from the refrigeration unit is further mixed with hot air which has by-passed the unit, and is delivered to the cockpit. The degree of mixing of the conditioned air is controlled automatically by an air temperature control valve, which maintains the air at the temperature selected from the cockpit. When the AIR COND control switch is turned to "OFF," or when electrical failure occurs, a solenoid in the air control valve de-energizes and opens the valve to allow only cold air to be directed into the cockpit.

PRESSURIZATION

When the air conditioning system is in normal operation, the air provided for heating, cooling, and ventilating is also used to pressurize the cockpit. Pressure is automatically maintained at a pre-determined schedule by a cockpit pressure regulator. The pressurizing schedule (figure 4-2) provides for cockpit pressure to equal atmospheric pressure from sea level to 8000 feet altitude. The cockpit pressure at 8000 feet is then maintained to an altitude of 17,500 feet. From this point on up, the cockpit pressure is maintained at 3.3 psi above the existing atmospheric pressure. Cockpit pressure is shown in terms of altitude by the cabin altimeter (1, figure 1-5) located on the right side of the armament panel.

In order to prevent excessive positive or negative pressure differentials because of possible malfunctioning of the pressure regulator, a pressure relief valve will open at a positive pressure differential of 3.6 psi and at a negative differential of -0.10 psi. The pressure relief valve also incorporates a pneumatically-actuated emergency valve which dumps cockpit pressure when the air conditioning system is turned off.

AIR CONDITIONING CONTROL PANEL. An AIR COND control panel (16, figure 1-5) on the right console contains a single rotary switch for the operation of the air conditioning and pressurization system. Marked positions of the switch are "OFF," "40°," "70°F," and "100°." The "OFF" position turns off the air conditioning system, pneumatically opens the pressure relief valve to dump cockpit pressure, and opens a valve in the ram air line, allowing outside air to ventilate the cockpit. To enter the "OFF" position, it is necessary to apply an increased force on the rotary switch. This feature prevents inadvertent selection of "OFF" when regulating the cockpit temperature. Adjustment of the switch to any position between "40°" and "100°" causes the cockpit temperature to be maintained at the level selected.

DEFROSTING SYSTEM

Defrosting of the cockpit windshield side panels is accomplished by utilizing hot air from the air conditioning system and is automatic in operation whenever the switch on the AIR COND control panel is rotated from "OFF." A thermostatically controlled valve is installed in the windshield air duct to prevent overheating of the windshield. If the temperature of the windshield air exceeds a pre-determined limit, the thermostat control diverts air from the anti-blackout system to the overheat valve. The valve then closes, shutting off the air to the windshield until the temperature is reduced.

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 numerous 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.

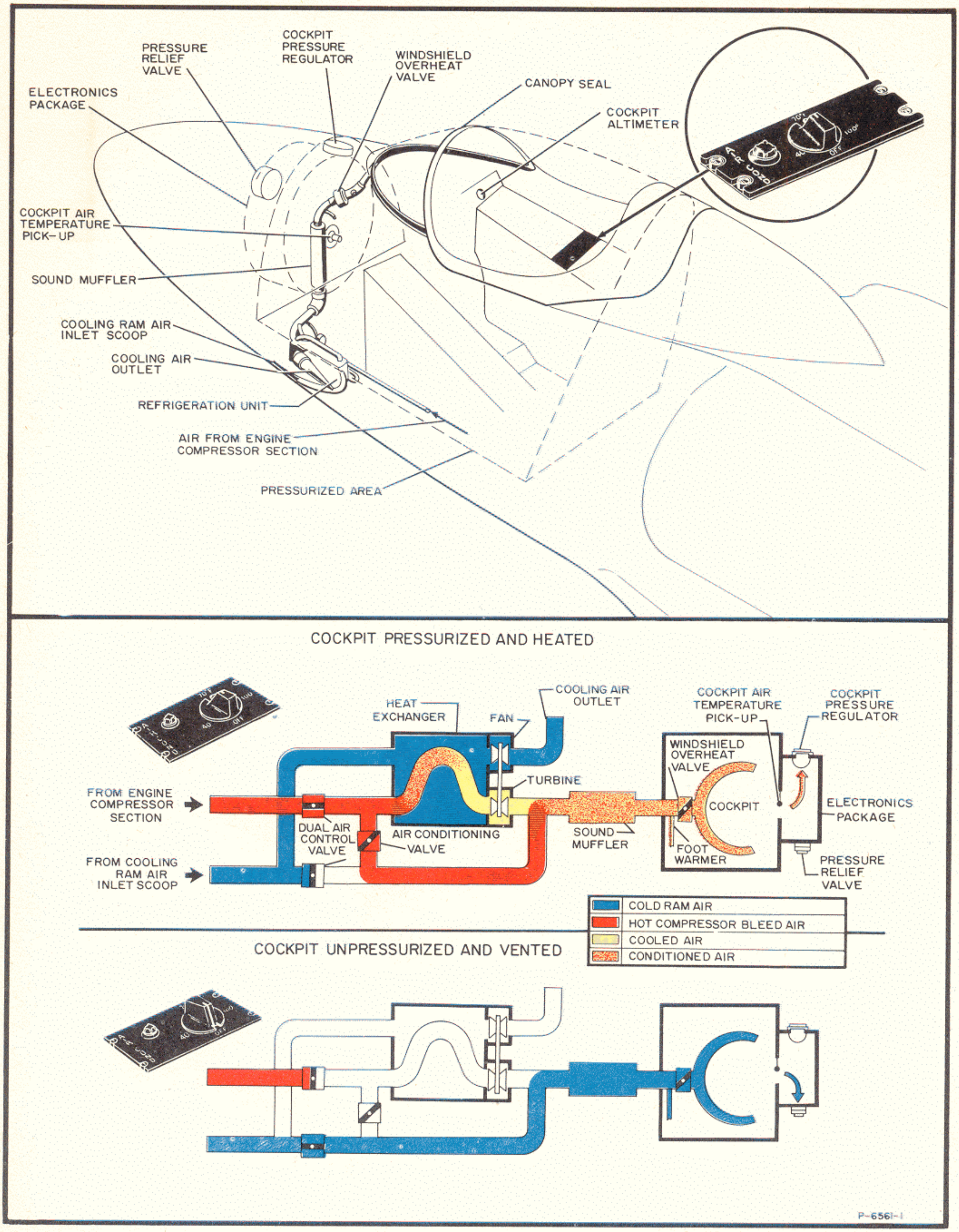
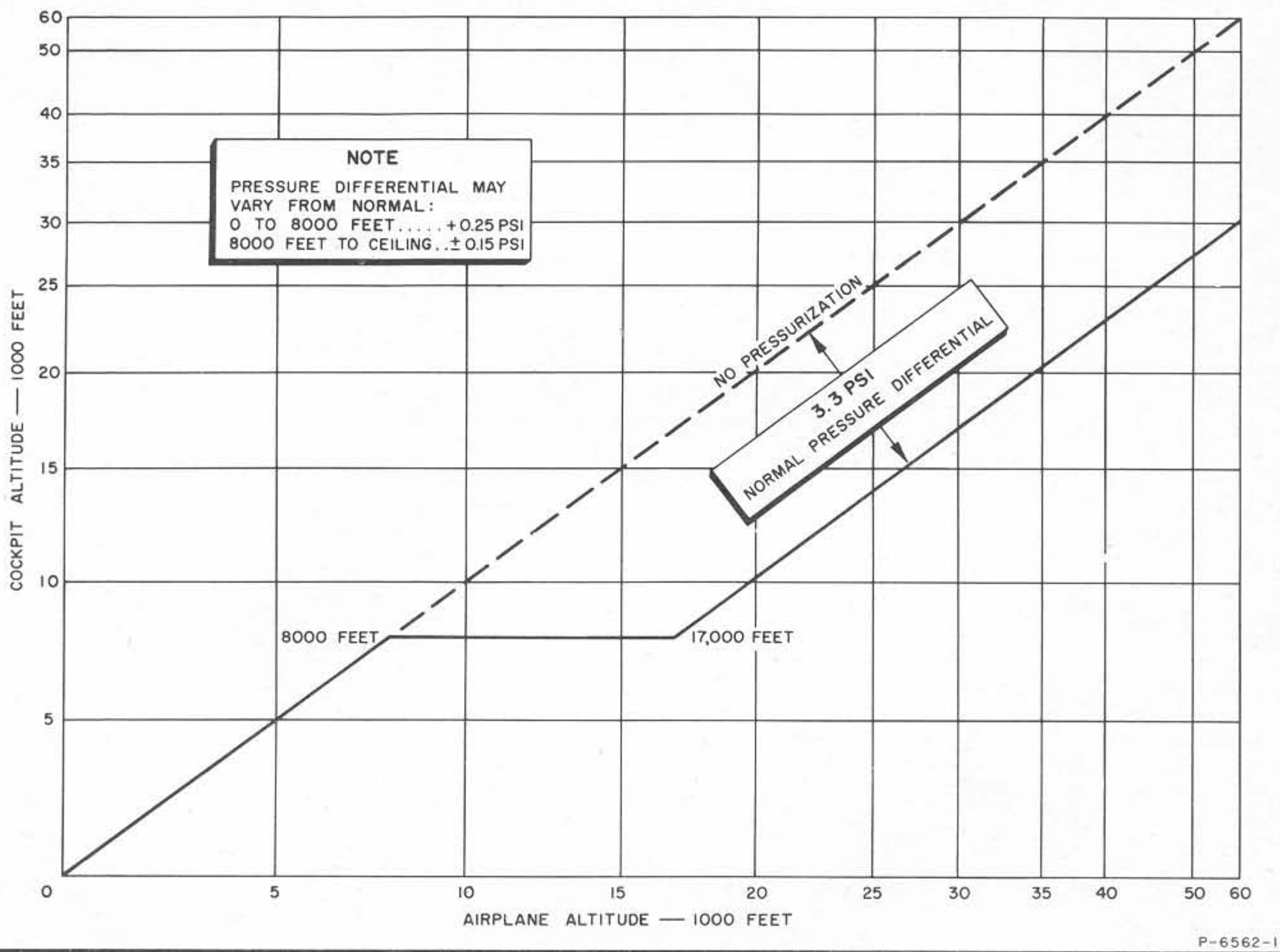


Figure 4-1. Air Conditioning and Pressurization System

COCKPIT PRESSURIZATION



DATA AS OF: 1 February 1955
DATA BASIS: Calculations

Figure 4-2. Cockpit Pressurization Chart

This fog may be eliminated by adjusting the AIR COND control knob for a higher cockpit temperature. 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.

NORMAL OPERATION

BEFORE TAKE-OFF. Before take-off, place the AIR COND switch at "70°" or as desired. (A setting for approximately 85° will insure against fogging during take-off and the initial part of climb.)

DURING FLIGHT. Rotate the AIR COND control knob in either direction to a temperature which provides the most comfort.

AFTER FLIGHT. Place the AIR COND control knob in the "OFF" position.

EMERGENCY OPERATION

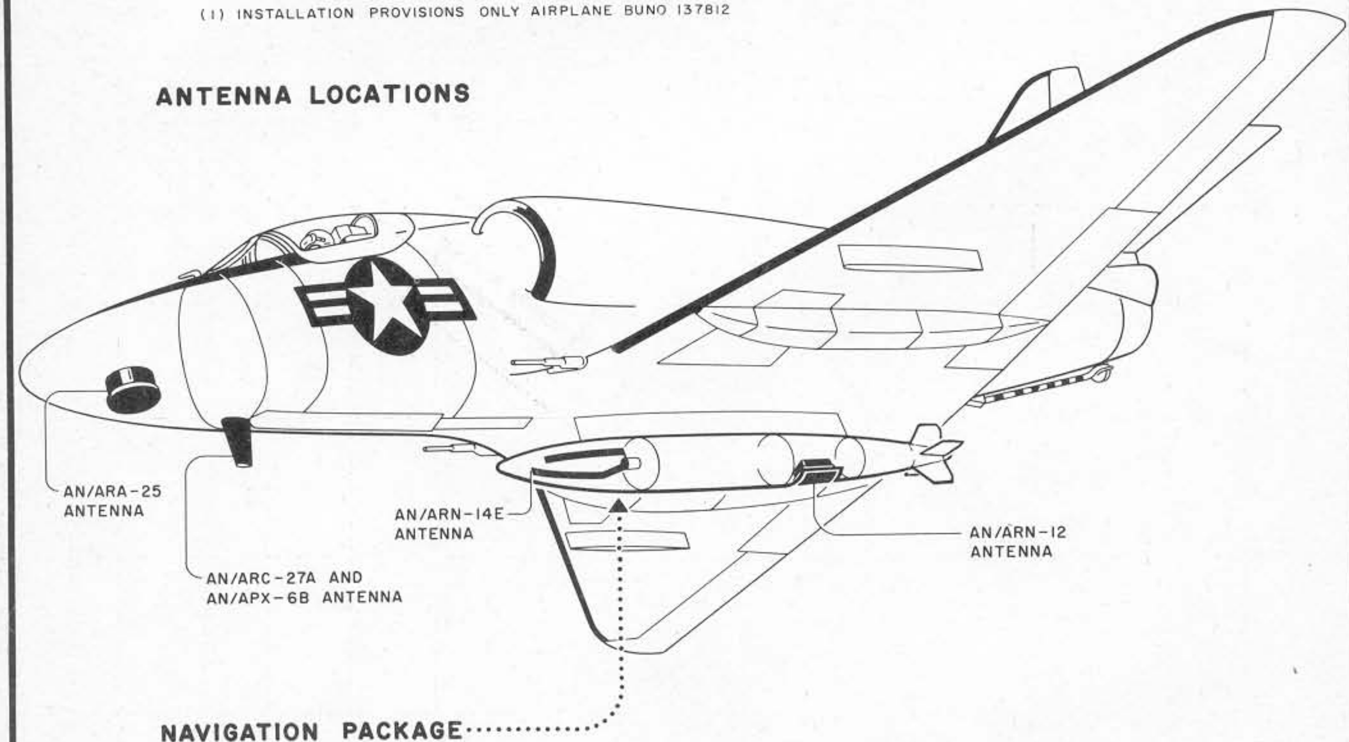
If the air conditioning unit fails, or loses electrical power, the cockpit air temperature may go to full cold or full hot. If the unit itself has failed, turning the AIR COND control knob to "OFF" will provide a more comfortable temperature, although pressurization will be lost; if failure is due to loss of electrical power, temp-

AIRCRAFT INSTALLATIONS

TYPE	DESIGNATION	FUNCTION	RANGE	LOCATION OF CONTROLS
COMMUNICATION				
UHF RADIO	AN/ARC-27A	SHORT RANGE - TWO WAY VOICE COMMUNICATION	LINE-OF-SIGHT	RIGHT CONSOLE
NAVIGATION				
UHF - ADF	AN/ARA-25	DIRECTIONAL HOMING	LINE-OF-SIGHT	RIGHT CONSOLE
IDENTIFICATION				
IFF RADAR	AN/APX-6B	IDENTIFIES AS FRIENDLY	LINE-OF-SIGHT	RIGHT CONSOLE
SIF RADAR (1)	AN/APA-89	IDENTIFICATION CODER	LINE-OF-SIGHT	RIGHT CONSOLE

(1) INSTALLATION PROVISIONS ONLY AIRPLANE BUNO 137812

ANTENNA LOCATIONS



TYPE	DESIGNATION	FUNCTION	RANGE	LOCATION OF CONTROLS
NAVIGATION				
OMNI RANGE RECEIVER	AN/ARN-14E	DIRECTIONAL HOMING	LINE-OF-SIGHT	LEFT CONSOLE
MARKER BEACON RECEIVER	AN/ARN-12	INDICATES MARKER BEACON POSITION	WITHIN MARKER BEACON SIGNAL PATTERN	LEFT CONSOLE

P-6563-1B

Group A, B and Ca Airplanes

Figure 4-3. Table of Electronic Equipment (Sheet 1)

erature control will be restored when the emergency generator is put in operation. This is true only on later airplanes[1] in which the air conditioning unit is powered by the primary bus.

DE-FOGGING SYSTEM

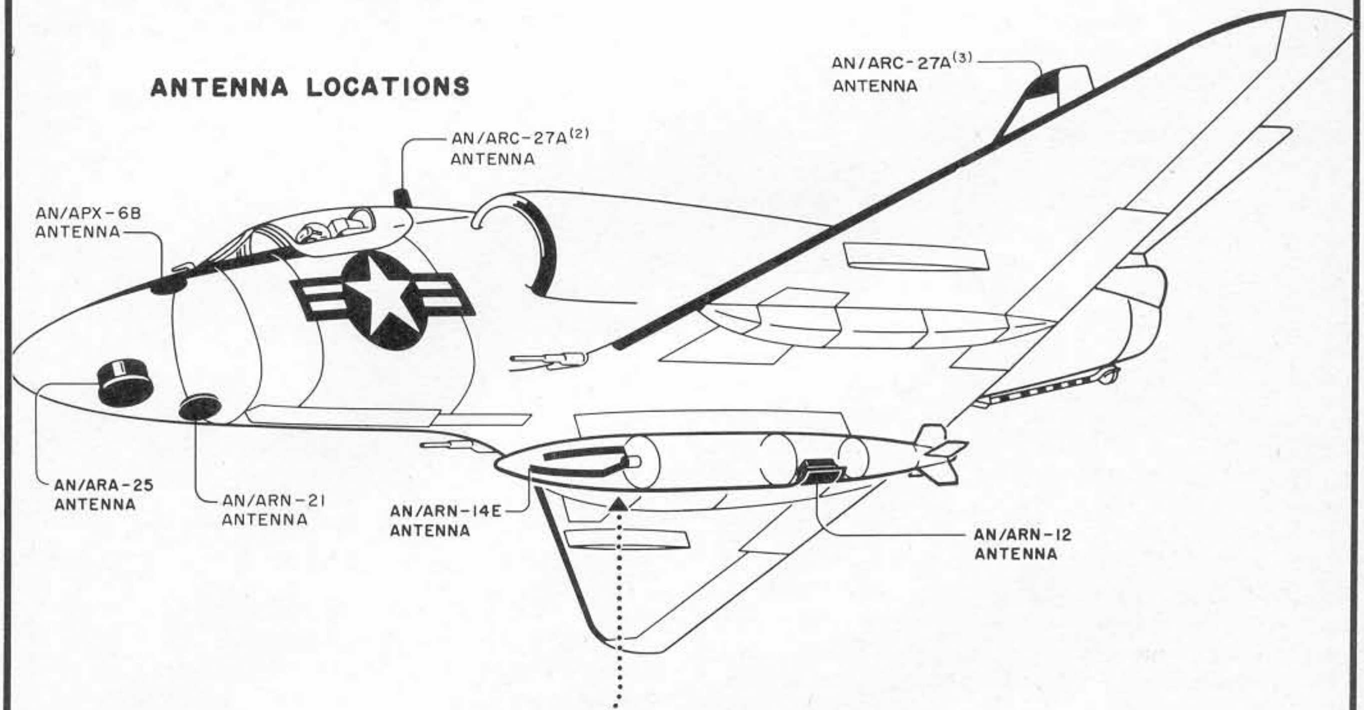
Fogging or frosting of the bullet resistant glass center panel of the windshield is prevented by electrical means. This system is energized whenever the main generator is operating or when external power is supplied to the a-c system. A transparent layer within the glass panel provides resistance for electrical heating, and a sensing element contained in the panel causes a heating con-

(1)Group C, D and E Airplanes.

AIRCRAFT INSTALLATIONS

TYPE	DESIGNATION	FUNCTION	RANGE	LOCATION OF CONTROLS
COMMUNICATION				
UHF RADIO	AN/ARC-27A	SHORT RANGE - TWO WAY VOICE COMMUNICATION	LINE-OF-SIGHT	RIGHT CONSOLE
NAVIGATION				
UHF - ADF	AN/ARA-25	DIRECTIONAL HOMING	LINE-OF-SIGHT	RIGHT CONSOLE
OMNI BEARING-DISTANCE RADIO (1)	AN/ARN-21	PROVIDES BEARING AND DISTANCE TO A SELECTED STATION	LINE-OF-SIGHT TO 200 MILES	RIGHT CONSOLE
IDENTIFICATION				
IFF RADAR	AN/APX-6B	IDENTIFIES AS FRIENDLY	LINE-OF-SIGHT	RIGHT CONSOLE
SIF RADAR	AN/APA-89	IDENTIFICATION CODER	LINE-OF-SIGHT	RIGHT CONSOLE

ANTENNA LOCATIONS



NAVIGATION PACKAGE

TYPE	DESIGNATION	FUNCTION	RANGE	LOCATION OF CONTROLS
NAVIGATION				
OMNI RANGE RECEIVER	AN/ARN-14E	DIRECTIONAL HOMING	LINE-OF-SIGHT	LEFT CONSOLE
MARKER BEACON RECEIVER	AN/ARN-12	INDICATES MARKER BEACON POSITION	WITHIN MARKER BEACON SIGNAL PATTERN	LEFT CONSOLE

(1) GROUP Cb AND Cc AIRPLANES HAVE SPACE, WIRING, AND WEIGHT AND BALANCE PROVISIONS ONLY.

(2) GROUP Cb,Cc,Cd, AND D AIRPLANES

(3) GROUP E AIRPLANES

P6563-2A

Group Cb, Cc, Cd, D and E Airplanes

Figure 4-3. Table of Electronic Equipment (Sheet 2)

troller to automatically regulate the temperature of the surface to maintain it between two predetermined limits. As the electricity for the heating element is supplied from the forward monitored bus, actuation of the emergency generator will de-energize the electrical de-fogging system; however, the hot air provided for defrosting of the side panels of the windshield should be sufficient to prevent defogging of the center panel also.

ANTI-ICING SYSTEM

Anti-icing is provided for the pitot system only. An electrical heating element in the pitot head is controlled by the PITOT HEAT switch (5, figure 4-5) on the armament panel. Two positions, "ON" and "OFF," are provided for the operation of the pitot heat. The "ON" position should normally be used only in flight, as prolonged ground operation without cooling air flow can damage the heating element.

COMMUNICATIONS AND ASSOCIATED ELECTRONIC EQUIPMENT

All communications and associated electronic equipment are listed in the Table of Electronic Equipment, figure 4-3. Major units of the AN/ARC-27A UHF radio, the AN/ARA-25 direction finder, and the AN/APX-6B IFF equipment are mounted in an integrated electronics central, which is designated AN/ASQ-17 and located in the nose section of the airplane, along with the AN/APA-89 coder group. Also installed in the nose section of later aircraft[1] is the AN/ARN-21 omni bearing-distance radio. Some earlier airplanes [2] have installation provisions for the AN/ARN-21. Auxiliary navigation equipment, the AN/ARN-12 marker beacon receiver and the AN/ARN-14E VHF radio, can be carried as an external store in a navigation package (NAVPAC). No master radio switch is provided to energize the radio circuits, and each piece of equipment must be turned on at its associated control panel, with the exception of the AN/ARN-12 marker beacon receiver and the AN/ARC-27 UHF radio. The latter are energized whenever the main generator is operating or external power is connected to the aircraft.

UHF RADIO

The AN/ARC-27A UHF radio equipment provides two-way voice communications with other aircraft and with surface stations. The equipment can transmit or receive through the same antenna on any one of 1750 channels within a frequency range of 225.0 through 399.9 megacycles.

UHF RADIO CONTROL PANEL. The C-1015/ARC-27A control panel (19, figure 1-5) is located on the right console. The panel contains a VOL control, SENS control, function selector switch, CHAN selector switch, and a manual tuning control. The VOL control regulates the volume of incoming signals, and the SENS control provides a vernier adjustment of the sensitivity. The CHAN selector switch has twenty channel positions, numbered 1 through 20, a "G" (guard) position, and an "M" (manual) position. In early aircraft[3] the "G" position is inoperative as no guard receiver is installed.

The manual tuning control consists of three concentric dials. The outer dial is used to select the first two digits of the desired frequency, the center dial selects the third digit, and the inner dial selects the digit to the right of the decimal point. The CHAN selector switch must be placed at "M" before channels can be selected manually. The function selector switch is used to place the set in operation, and has four positions marked as follows:

Position	*Function*
"OFF"	Inoperative (the transmitter-receiver is energized when the electrical system is energized).
"T/R"	Transmitter on standby and receiver in operation.
"T/R + G REC"	Same as "T/R" plus reception on the guard frequency except in early aircraft[3] which have no guard receiver installed.
"ADF"	Transmitter on standby and AN/ARA-25 direction finding equipment in operation through receiver.

OPERATION OF THE UHF RADIO

a. Function selector switch ...."T/R"

b. CHAN selector switch ....Desired channel

c. Depress throttle MIC switch to transmit.

d. For operation on "ADF" refer to AUTOMATIC DIRECTION FINDING EQUIPMENT.

AUTOMATIC DIRECTION FINDING EQUIPMENT

The AN/ARA-25 automatic direction finding equipment operates in conjunction with the AN/ARC-27A UHF radio to provide a continuous directional indication of the source of signals received in the 225 to 400 megacycles band. Source indication in degrees of magnetic bearing for homing or navigational purposes is provided by the number 1 needle of the ID-250/ARN course indicator.

OPERATION OF THE AN/ARA-25. The AN/ARA-25 ADF equipment is energized when the aircraft electrical system is energized, and is placed in operation by moving the function selector switch on the UHF radio control panel to the "ADF" position.

RADAR IDENTIFICATION EQUIPMENT

The aircraft radar identification equipment consists of the AN/APX-6B IFF transponder and AN/APA-89 equipment. The AN/APX-6B IFF transponder provides the aircraft with automatic identification when properly challenged by surface or airborne radar equipment. The identification system also permits surface tracking of the aircraft in which it is installed. Function-

[1] Group Cd, D and E Airplanes.
[2] Group Cb and Cc Airplanes.
[3] Group A and B Airplanes.

ally, the AN/APX-6B equipment receives challenges and transmits coded replies for display upon the challenger's radar indicator.

IFF CONTROL PANEL. A C-1159/APX-6B control panel (21, figure 1-5) is installed on the right console and is identified as IFF. The panel contains a rotary selector switch labeled MASTER with five designated positions: "OFF," "STDBY," "LOW," "NORM," and "EMERGENCY." Two MODE switches are located adjacent to the MASTER switch; a third toggle switch provides selection of "I/P" and "MIC." The equipment is placed in operation when the MASTER switch is rotated out of the "OFF" position, provided the main generator is operating or external power is available.

SIF CONTROL PANEL. A C-1272/APA-89 control panel (20, figure 1-5), labeled SIF, is located on the right console adjacent to the IFF control panel.

OMNI BEARING-DISTANCE EQUIPMENT[(1)]

The AN/ARN-21 airborne equipment operates in conjunction with surface navigation beacons to provide continuous directional and distance information to the pilot. Visual indication of bearing to a selected station is provided by the number 2 pointer of the ID-250/ARN course indicator (figure 1–4, sheet 3), and distance information is indicated on the ID-310/ARN range indicator (figure 1–4, sheet 3). Beacon identification tone signals are received through the regular headset.

ARN/21 CONTROL PANEL. The C-866/ARN-21 control panel (11A, figure 1–5, sheet 2) is identified as NAV and is located on the right-hand console. Operating controls include the power switch with "OFF-REC-T/R" positions, two CHAN (channel selector) knobs, and a VOL (volume) control.

ARN-21 TRANSFER RELAY.

Because the ID-250/ARN course indicator is common to both the AN/ARN-14 radio and the AN/ARN-21 radio, a switching means is provided for the selection of proper receiver-indicator combinations to meet specific navigational requirements. The ARN-21 transfer relay will automatically connect the ID-250/ARN course indicator to either the AN/ARN-14 or the AN/ARN-21 radio when their respective console control panel is turned on. The AN/ARN-21 transfer relay is energized by either the "REC" or "T/R" position of the power switch on the C-866/ARN-21 control panel, disconnecting the AN/ARN-14 radio and connecting the AN/ARN-21 radio to the ID-250/ARN course indicator. Thus if both radio sets are inadvertently turned on at the same time, only the AN/ARN-21 equipment will be operative.

OPERATION OF AN/ARN-21. To operate the AN/ARN-21 radio, proceed as follows:

a. Power switch . . . . "REC"

Note

A warm-up period of 90 seconds is required after the power switch is moved from "OFF".

b. CHAN selector switch . . . . Set channel number in indicator dial

c. Identify beacon by tone signals in headset and read bearing to station at the number 2 pointer of the ID-250 indicator.

d. For distance information, the power switch must be turned to "T/R". Read distance to the beacon in nautical miles on the ID-310 range indicator.

MARKER BEACON EQUIPMENT

The AN/ARN-12 marker beacon equipment, which is placed in operation when the aircraft electrical system is energized, is located in the external navigation package. Functionally, the system receives a 75 megacycle signal, modulated at 400, 1300, or 3000 cycles, from marker beacon transmitters. From these signals, the relative position of the aircraft can be checked in respect to specific marker beacon stations. An aural indication of marker beacon interception is provided by the aircraft audio system. In addition, a marker beacon indicating light, located on the landing check list panel, or, in later aircraft[(2)], adjacent to the LABS light beneath the glareshield, glows an amber color to give visual indication of marker beacon interception.

Note

The marker beacon indicating light is also used in conjunction with certain special weapon installations when the NAVPAC is not carried. In this event a red lens cover should be placed over the light by the ground crew.

MARKER BEACON CONTROL PANEL. A MARKER BEACON control panel (18, figure 1-3), on the left console forward of the throttle, contains two toggle switches: an AUDIO switch with "ON" and "OFF" positions; and a sensitivity switch with positions marked "BELOW 20,000 FT" and "ABOVE 20,000 FT". Since no volume control is provided, the AUDIO switch is used to continue or discontinue the reception of aural signals when their volume level interferes with other radio receptions. Regardless of the position of the AUDIO switch, visual indication of marker beacon interception will be provided by the indicator light. The sensitivity switch should be placed in the position corresponding to the general altitude level at which the aircraft is flying to achieve the best reception.

OMNI-RANGE RADIO

The AN/ARN-14E omni-range radio includes an R-540/ARN-14C VHF receiver, an ID-251/ARN indicator con-

[(1)]Group Cd, D and E Airplanes.
[(2)]Group Bb, Bc, Bd, C, D and E Airplanes.

trol, and an antenna; all are contained in the NAVPAC external stores. The omni-range radio provides for the reception of VOR, VAR and localizer signals in a frequency spectrum of 108.0 to 136.0 megacycles. Use of the equipment in this aircraft is described as follows:

a. Indication of the magnetic bearing of the VHF omni-range station from the aircraft by the number 2 needle of the ID-250/ARN course indicator.

b. Reception of voice facilities on communications channels or superimposed on navigation signals in the localizer or omni-range channels.

OMNI-RANGE RADIO CONTROL PANEL. The omni-range receiver is remotely tuned by means of a C-760A/ARN control panel (17, figure 1-3) labeled VHF NAV. The panel contains a POWER switch with "ON TONE" and "OFF" positions, a VOLUME control, and a FREQUENCY selector which consists of two concentric dials. A window is provided to show the selected frequency.

OPERATION OF THE OMNI-RANGE RADIO

a. POWER switch ...."ON TONE"
b. FREQUENCY selector ....Desired frequency

c. Read the magnetic bearing to the station on the number 2 needle of the ID-250/ARN course indicator.

Note

The "OFF-REC-T/R" switch on the C-866/ARN-21 control panel must be positioned at "OFF" in order for the AN/ARN-14E radio to be connected to the ID-250/ARN course indicator.

NAVIGATION EQUIPMENT

S-2 COMPASS SYSTEM

The S-2 compass system consists of a directional gyro control (gyrosyn compass) which can be electrically slaved by a switch in the cockpit to a magnetic flux valve located in the tail section. An amplifier and junction box are installed on the left side of the forward engine compartment. The compass card of the ID-250/ARN course indicator is slaved to directional gyro control in the S-2 compass system to provide either magnetic or free directional indications of the aircraft heading. The compass system receives power from the primary bus and operates whenever the aircraft electrical system is energized. Approximately three minutes are required for the leveling and torque units to slave the gyro to the flux valve after the system is energized.

COMPASS CONTROLLER. Controls for the operation of the S-2 compass system are located on the COMPASS CONTROLLER panel (13, figure 1-5) on the right console. On the panel is a SLAVED GYRO-FREE GYRO toggle switch which permits selection of either function of the S-2 compass; a SET HEADING FREE GYRO control which, through its two positions ("DEC" and "INC") is used to rotate the ID-250/ARN compass card to any desired heading, or to synchronize the compass card reading with that of the magnetic flux valve; and a SYNC SIGNAL indicator, whose needle pointer indicates whether the compass card is synchronized with the flux valve. Deflection of the needle to the left or right of a center mark indicates that the compass card reading should be numerically increased or decreased, respectively. When the needle points to the center mark, the compass card is synchronized with the flux valve.

OPERATION OF THE S-2 COMPASS SYSTEM. When the compass system is warmed up, perform the following:

a. SLAVED GYRO-FREE GYRO switch ...."FREE GYRO"

b. Note direction of deflection of the SYNC SIGNAL pointer.

c. Move SET HEADING FREE GYRO control to "DEC" or "INC" as necessary until the pointer centers.

d. SLAVED GYRO-FREE GYRO switch ...."SLAVED GYRO"

AUTOMATIC DEAD-RECKONING SET[1]

The automatic dead-reckoning set continuously computes and indicates throughout a flight the great circle bearing and range to a target or home base. It also provides a continuous indication of present latitude and longitude and of the ground track being flown. This system consists of a true airspeed (TAS) transducer just below the cockpit floor, a data setting box on the right hand console (23, figure 1-5, sheet 3), a servo amplifier in the nose section, and an indicator on the pilot's instrument panel (figure 1-4, sheet 4). The set is energized by the forward monitored bus, and may be operated only when the main generator is running or when external power is applied to the aircraft. The TAS transducer, through rotation of a synchro control transformer rotor (which is varied by the outside air temperature and pitot/static pressure values), computes TAS information and transmits it to the data setting box. The data setting box accepts manual inputs of base position, target position, wind and variation, as well as the automatic input from the TAS transducer and from the aircraft compass system. These inputs are there resolved to components of great circle bearing and distance, and are supplied to the indicator, along with a function of ground track in the form of a synchro signal. The data setting box also provides a continuous reading of present latitude and longitude, which may be read in windows on the data setting box itself. Relative great circle bearing and distance to target or base are presented on the indicator by means of a pointer and counter, respectively, and ground track is presented by a moving card.

[1]Group E Airplanes.

DEAD RECKONING SET CONTROLS

FUNCTION SWITCH. The function selector switch is used to place the set in operation, and has five positions marked as follows:

Position	*Function*
"OFF"	The set is completely de-energized.
"TARGET"	Computations are continuously performed so that distance to the target and target bearing appear on the indicator.
"BASE"	Distance to base and base bearing appear on the indicator in place of target data.
"STANDBY"	System is energized, but is inoperative.
"RESET"	Computed data is erased. Used for establishing a new base reference.

LATITUDE AND LONGITUDE INPUT. Four knobs are provided to set position latitude, position longitude, target latitude, and target longitude. Each of these settaings is read on an adjacent counter on the data setting box.

WIND INPUT. Two knobs are used to set wind direction and velocity into the data setting box. These values are read on a dial and a counter, respectively.

Note

The computer accepts wind velocities from 0-299 knots only.

VARIATION INPUT. Magnetic variation up to 180° East or West may be introduced into the data setting box by means of a single knob adjacent to the variation dial.

OPERATION OF THE DEAD-RECKONING SET

Before take-off (or while flying over a known location), when it is desired to put the set in operation, turn the function selector switch to "STANDBY." Set the VAR knob to the applicable East or West variation, and the WIND VEL or WIND DIR according to the latest available wind information. Adjust the POSITION LAT and POSITION LONG knobs to the present location and the TARGET LAT and TARGET LONG knobs to the location of the target or destination. Turn the function switch to "RESET" momentarily; then back to "STANDBY." This stores the location set in the POSITION windows for later use as a base reference. After take-off, and above 150 knots, "TARGET" may be selected on the function switch, and the bearing and distance to the target will be continuously shown on the instrument panel indicator.

Note

When flying at true airspeeds of less than 150 knots, with the function switch on either "TARGET" or "BASE," computations of the set will be in error. Since the airspeed input is limited to a minimum of 150 knots, the computer will assume a TAS of 150 knots even at lesser speeds.

At any time during the flight the TARGET position settings may be changed without effect on previous computations. Aircraft location is continuously shown during the flight by the LAT and LONG POSITION windows. For the return to base, the function switch should be moved to the "BASE" position, which will cause the instrument panel indicator to show bearing and distance to the base reference which has been stored in the computer.

Note

- The base reference will be erased whenever the function switch is moved to "RESET," and the location seen at that time in the POSITION windows will be accepted as a new base reference.
- The only way to erase the base reference is to place the function switch in "RESET"; thus the same base reference could be used for two or more successive flights, as long as the "RESET" position has not been entered.

STANDBY COMPASS

A direct-reading magnetic compass is mounted on the forward edge of the canopy. The reading obtained from the standby compass must be compensated for deviation by applying the corrections given on the compass correction card (4, figure 1-5) located on the right side of the cockpit adjacent to the arresting hook control.

LIGHTING EQUIPMENT

INTERIOR LIGHTS

The interior lighting system includes all instrument and console lights, and two cockpit floodlights. The instrument faces are concave and each has an individual light mounted in the center to provide even illumination over the entire instrument face. The two floodlights are mounted on each side of the gunsight beneath the glareshield to provide auxiliary or emergency red lighting of the instrument panel. The console lights are standard. The instrument and console lights receive power from the primary bus and the floodlights are powered from the 26-volt a-c bus.

INTERIOR LIGHTS CONTROL PANEL. An INT LTS control panel (17, figure 1-5), mounted on the right console, contains switches for the operation of all interior lights. Two rotary switches, marked INST and CONSOLES, are rotated clockwise from "OFF" to turn on the instrument lights and console lights, respectively. Further rotation in a clockwise direction toward the "BRIGHT" position increases the intensity of the illumination. A toggle-type FLOOD switch with three positions, "BRIGHT," "OFF," and "DIM," controls the operation of the floodlights after the CONSOLES switch is rotated from the "OFF" position.

EXTERIOR LIGHTS

The exterior lights system includes position, formation, and fuselage-wing lights, an approach light, and provisions for a removable taxi light.[1] Standard semi-flush wing tip position lights and a single white tail light are installed on the aircraft. Formation lights are rectangular windows on the top and bottom of each wing tip and have colors corresponding to the position lights. A semi-flush white fuselage-wing light is located under the leading edge of each wing and one is mounted on top of the fuselage. The approach light is mounted on the nose wheel strut; the taxi light can be installed on the right hand main landing gear door. The exterior lights receive power from the forward monitored bus.

EXTERIOR LIGHTS CONTROLS

MASTER EXTERIOR LIGHTS SWITCH. A master exterior lights switch (8, figure 1-3), on the outboard side of the throttle grip, controls power to the exterior lights. The switch has a forward "on" position, a center "off" position, and an aft, or "momentary" position. The master exterior lights switch is spring-loaded from the aft to the center position, thus providing a means of signaling with the exterior lights.

EXTERIOR LIGHTS CONTROL PANEL. An EXT LTS control panel (15, figure 1-5) on the right console contains switches for functional control of the exterior lights. Two FUSELAGE switches control operation of the fuselage-wing lights; one switch has "FLASH" and "STEADY" positions, and the other has "BRIGHT," "MAN," and "DIM." The "MAN" position is used in conjunction with an adjacent MAN KEY switch and associated IND (indicator) light for sending code with the fuselage-wing lights. The WING & TAIL switch and the FORM (formation lights) switch have "BRIGHT," "OFF," and "DIM" positions. A guarded TAXI switch with "ON" and "OFF" positions controls the operation of the taxi light, when the latter is installed.

OPERATION OF THE EXTERIOR LIGHTS

a. Master exterior lights switch . . . "On"

b. FUSELAGE switches . . . . "FLASH" and "BRIGHT"

c. WING & TAIL switch . . . . "BRIGHT"

d. FORM switch . . . . "BRIGHT"

e. To manually send code signals, leave the FUSELAGE switches on "STEADY" and "MAN," and depress the MAN KEY switch to signal, using the IND light to monitor the signals.

APPROACH LIGHT OPERATION. The approach light circuit is energized by the a-c primary bus and is controlled by a tail hook micro-switch, a landing gear micro-switch, and by a manually operated tail hook by-pass switch. The approach light functions when the master exterior lights switch is turned on and provides the landing signal officer with information as to flight attitude on approach, as well as indicating the positions of the landing gear and arresting hook. The following chart describes approach light operation:

WHEELS	*HOOK*	*APPROACH LIGHT*
Locked down	Down	On
Locked down	Not down	Off
Not locked down	Any position	Off
Locked down	Not down but tail hook by-pass switch actuated	On

TAIL HOOK BY-PASS SWITCH. A momentary contact switch, located on a test panel on the right-hand side of the nose wheel well, is provided for by-passing the tail hook micro-switch so the approach light can be used for night field carrier landing practice. This switch, protected by a red guard, is identified as HOOK BY-PASS APPROACH LIGHT (or APPROACH LIGHT TAIL HOOK BY-PASS[2]). When the switch is pressed momentarily to the up position with the electrical system energized, the approach light will then be illuminated with the landing gear down and locked even though the arresting hook remains in the retracted position. Normal operation of the approach light is regained when the arresting hook control is moved to the "DOWN" position, or when electrical power to the approach light circuit is interrupted.

OXYGEN SYSTEM

A positive pressure, demand-type oxygen system provides 100% oxygen at all times when in use. The oxygen supply is contained in either one or two 514-cubic inch cylinders, depending on requirements, if a gaseous system is installed. If the liquid system is used in the aircraft, oxygen is contained in an insulated tank. Oxygen equipment and controls within the cockpit are the same for either system, except that a pressure gage is used to measure oxygen content in the gaseous system, whereas the liquid oxygen content is indicated by a quantity gage.

GASEOUS SUPPLY SYSTEM.[3] In aircraft which have a gaseous oxygen supply, oxygen is contained in either one or two 514-cubic inch cylinders which are serviced through a filler valve adjacent to the cylinder installation. Cylinders are serviced to a maximum of 1800 psi which is registered on a gage (16, figure 1-3) located on the left side of the armament panel in the cockpit. A reducer valve is used in the system to provide low pressure delivery (70 psi) to the receptacle on the left console.

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(1) Group Cc, Cd, D and E Airplanes.
(2) Group Aa Airplanes.
(3) Group A and B Airplanes.

LIQUID SUPPLY SYSTEM[1]. Aircraft provided with a liquid oxygen supply have an insulated tank mounted in the aft fuselage section. The tank filler valve is reached through an access door on the left side of the fuselage for servicing. The tank contains 5.0 liters of liquid oxygen when serviced to capacity. Evaporation loss is constant when the system is not in 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. A LIQUID OXYGEN quantity gage (16, figure 1-3) is located on the left side of the armament panel, and 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 ½ liter. Pressure is maintained in the system by the constant evaporation and is limited to 70 ± 5 psi by venting any excess pressure overboard through relief valves.

CONTROLS AND EQUIPMENT. The oxygen supply gage (16, figure 1-3) is located on the left side of the armament panel. An OXYGEN switch (14, figure 4-5), installed adjacent to the oxygen supply gage, places the system in operation when moved from "OFF" to "ON."

When the OXYGEN switch is turned on, oxygen is delivered from the supply system at a pressure of 70 psi to the OXYGEN receptacle (26, figure 1-3) on the left console. The pilot's supply tube is plugged into the receptacle to allow the oxygen to flow to the system regulator, installed just below the pilot's face mask, which meters oxygen to the pilot under positive pressure at all altitudes.

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.
- Always check the oxygen supply tubes for tight connections before moving the OXYGEN switch to "ON." Escaping oxygen creates a fire hazard in the presence of oil or grease.

BAIL-OUT BOTTLE. Emergency oxygen is contained in a cylindrical "U" shaped bail-out bottle (6, figure 1-10) installed in the seat cushion. A pressure gage (9, figure 1-10), which extends through the lower forward right hand corner of the seat cushion, should register 1800 psi when the bottle is filled. The duration of the emergency oxygen supply is approximately 10 minutes, dependent upon the altitude since the oxygen is delivered by the mask regulator only upon demand. Oxygen is supplied from the bail-out bottle when a green ball (1, figure 1-10) hanging from the left front side of the seat is pulled. The ball, attached to a cable, releases a plunger on the pressure reducer of the bail-out bottle allowing oxygen to flow at a reduced pressure of 70 psi. The emergency oxygen then flows through the pilot's supply tube to the oxygen regulator for delivery to the face mask. A check valve in the segment of supply tube which plugs into the OXYGEN receptacle on the left console prevents loss of oxygen when the bail-out bottle is actuated. During ejection of the seat in case of emergency bailout, the ball release will be actuated automatically when the pilot is separated from the ejection seat.

[1]Group C, D and E Airplanes.

OXYGEN DURATION

Figure 4-4 (sheet 1) provides a tabulation of oxygen duration for the gaseous supply at various combinations of altitudes and oxygen cylinder pressures. Figure 4-4 (sheet 2) is a tabulation of hours remaining for various altitude-tank supply combinations for the liquid oxygen supply system. It will be 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.

CAUTION

When operating an aircraft in which the gaseous supply system is installed, do not exhaust the oxygen supply below 300 psi except in an emergency, as this necessitates removal and purging of the supply cylinder to prevent contamination of the system.

NORMAL OPERATION

BEFORE FLIGHT. Before each flight requiring oxygen, the oxygen system and mask may 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 the 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 ship's 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 restrictor in the supply tube.

h. With the mask in place, connect the oxygen supply tube to the connector on the seat cushion 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 seat cushion and console oxygen 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

GASEOUS OXYGEN DURATION

HOURS REMAINING

CABIN PRESSURE ALTITUDE —FEET—	GAGE PRESSURE — P.S.I. 1800	1600	1400	1200	1000	800	600	400	300 AND BELOW
30,000	2.9	2.5	2.1	1.7	1.3	1.0	0.6	0.2	EMERGENCY DESCEND TO ALTITUDE NOT REQUIRING OXYGEN
25,000	2.2	1.9	1.6	1.3	1.0	0.7	0.4	0.2	
20,000	1.7	1.6	1.3	1.0	0.8	0.6	0.3	0.1	
15,000	1.4	1.2	1.0	0.8	0.6	0.4	0.3	0.1	
10,000	1.1	0.9	0.8	0.7	0.5	0.4	0.2	—	
5000	1.0	0.8	0.7	0.5	0.4	0.3	0.2	—	
SEA LEVEL	0.8	0.7	0.6	0.4	0.3	0.2	0.1	—	

P6564 –1A

REMARKS:

(1) Data is computed for oxygen supply contained in one 514-cubic inch cylinder. Oxygen duration may be obtained by multiplying the chart figures by the number of cylinders installed.

(2) The altitude indicated by the cabin altimeter must be used for all oxygen duration calculations.

(3) Data is based on the use of a properly-fitted face mask and the oxygen system delivering 100% oxygen whenever the equipment is in use.

DATA AS OF: 1 February 1956
DATA BASIS: Calculations

Group A and B Airplanes

Figure 4-4. Oxygen Duration (Sheet 1)

these connections for security before making any other check of communications equipment.

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

a. OXYGEN switch . . . . "OFF"

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

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

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(1)Group Ab, Bb, Bc, Bd, C, D and E Airplanes.

ARMAMENT EQUIPMENT

All bombs, rockets, and other droppable stores are carried externally on three racks. Weapons which can be carried are bombs, rockets, chemical tanks, mines, torpedoes and special stores. A four-hook ejector-type bomb rack is installed on the centerline (fuselage) station and two-hook ejector racks are provided on the wing stations. The centerline rack, which is limited to a load of 3575 pounds, can be used to carry stores which require either 30-inch or 14-inch suspension. Wing racks are provided with 14-inch suspension only and have a load limit of 1200 pounds each. In addition to external armament, two 20-millimeter guns can be installed as alternate equipment(1). Use of the guns, however, necessitates a reduction of approximately 450 pounds in the maximum allowable external load.

LIQUID OXYGEN DURATION

HOURS REMAINING

CABIN PRESSURE ALTITUDE —FEET—	GAGE READING (LITERS)				
	"5"	"4"	"3"	"2"	"1"
45,000 ▶	6.5	5.2	4.0	2.6	1.3
40,000 ▶	5.9	4.7	3.6	2.4	1.2
35,000 ▶	5.3	4.3	3.2	2.1	1.1
30,000 ▶	4.7	3.8	2.8	1.9	0.9
25,000 ▶	4.1	3.3	2.5	1.4	0.8
20,000 ▶	3.6	2.9	2.1	1.4	0.7
15,000 ▶	3.2	2.6	2.0	1.3	0.6
10,000 ▶	2.9	2.3	1.7	1.2	0.6
5000 ▶	2.4	1.9	1.4	1.0	0.5
SEA LEVEL ▶	2.0	1.6	1.2	0.9	0.4

P6564—3

REMARKS:
(1) The altitude indicated by the cabin altimeter must be used for all oxygen duration calculations.
(2) Data assumes the use of a properly-fitted face mask, oxygen system pressurized at 70 psi and delivering 100% oxygen when in use.
(3) All figures given are conservative. Figures given for altitudes of 15,000 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.

DATA AS OF: 1 June 1956
DATA BASIS: Calculations

Group C, D and E Airplanes
Figure 4-4. Oxygen Duration (Sheet 2)

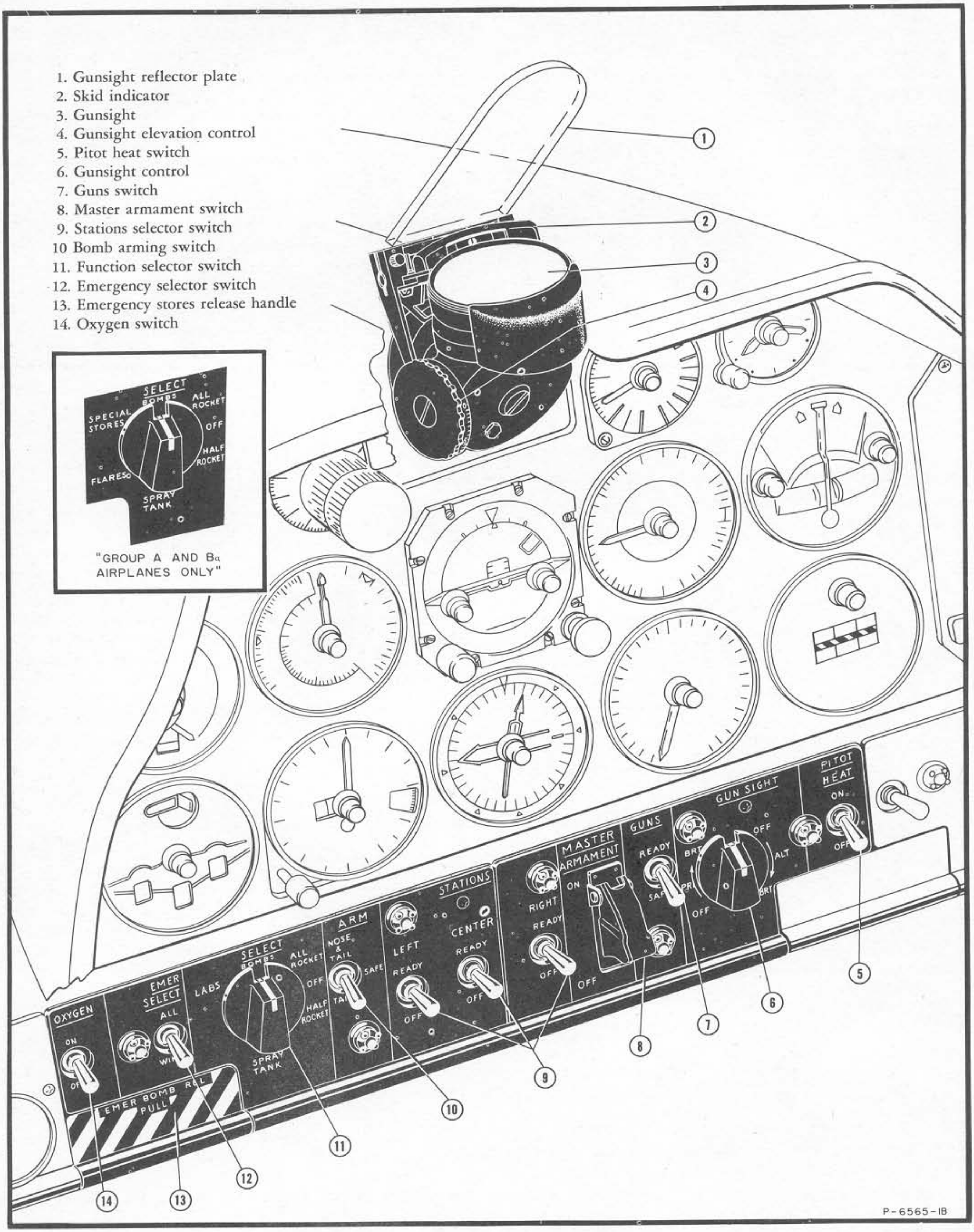

Figure 4-5. Gunsight and Armament Panel

Control of all armament is effected through an armament panel and, when various special stores are loaded, through additional panels installed on the left-hand console. Electrical release of stores is controlled directly by the pilot.

MASTER ARMAMENT SWITCH. All armament equipment is controlled by the MASTER ARMAMENT switch (8, figure 4-5) except during emergency jettisoning of stores. The armament circuits can be energized only when this switch is in the "ON" position. An armament safety circuit prevents operation of the armament system when the landing gear is extended due to the action of a micro-switch which causes all armament circuits to be de-energized.

STATIONS SELECTOR SWITCHES. Three STATIONS switches (9, figure 4-5) are provided for LEFT, CENTER and RIGHT armament release. Each switch has "READY" and "OFF" positions, providing single or multiple release of stores with the armament firing switches.

FUNCTION SELECTOR SWITCH. A rotary SELECT switch (11, figure 4-5) is provided for selecting the type of armament to be released. The switch may be set on any one of six[1] or seven separate detents which are labeled "SPECIAL STORES" or "LABS"[1], "BOMBS", "ALL ROCKET", "OFF", HALF ROCKET", "SPRAY TANK", and "FLARES" or blank[1]. When the "HALF ROCKET" is selected, only a portion of rockets is released from each rocket package.

Note

> The "HALF ROCKET" position of the function SELECT switch is not utilized with the rocket packages presently in use.

ARMAMENT SAFETY CIRCUIT DISABLING SWITCH. When the airplane is on the ground, an alternate circuit may be energized for the purpose of checking the armament system. This circuit is engaged by momentarily closing the ARM SAFETY DISABLE SWITCH, located on the outboard side of the right-hand wheel well. (On some airplanes[2] this switch is installed on a test panel on the right-hand side of the nose wheel well, and is labeled ARMT. SAFE DISABLE SWITCH.) Raising the landing gear or moving the MASTER ARMAMENT switch to "OFF" will restore the armament safety circuit to normal operation.

GUNSIGHT

An illuminated gunsight (3, figure 4-5) is provided for gunnery and for rocket and bomb aiming. The gunsight, mounted on the top center of the glareshield, contains one ladder type reticle and a glass reflector upon which the recticle image is focused.

GUNSIGHT ELEVATION CONTROL. A sight elevation knob (4, figure 4-5) on the left-hand side of the gunsight controls the glass reflector angle for raising and lowering the recticle image.

GUNSIGHT SWITCH. A single rotary knob (6, figure 4-5) on the armament panel, identified as GUNSIGHT serves as the gunsight light control. By rotating the knob, either of two filaments may be selected for lighting. Light intensity can be adjusted between the "OFF" and "BRIGHT" positions for either the primary (PRI) filament or for the alternate (ALT) filament.

GUNNERY EQUIPMENT

Two forward firing Mark 12 Mod 0 20-millimeter guns can be installed in the airplane as alternate armament. Ammunition boxes, located in the main access area between the guns, can be loaded with one hundred rounds of ammunition for each gun.

The Mark 12 Mod 0 gun is a combination blowback and gas-operated weapon which fires ammunition electrically. The weapon, which is air-cooled, has a rate of fire of 1000 rounds per minute in short bursts. A pneumatic charger mechanism operates the breechblock for first round loading or for round clearing. Although firing is simultaneous from both guns, individual compressed air tanks, filled to 3200 $\pm$ 200 psi, permit each gun to be remotely charged as a separate unit. The gun electro-pneumatic packages are located in the right hand wheel well, and in the forward engine compartment and accessory section.

GUN CHARGING SWITCH. The gun charging switch (7, figure 4-5) on the armament panel is identified as GUNS and has two positions, "SAFE" and "READY." With the switch on "READY," the charged gun is ready to fire. To re-charge or "cycle" the guns, the switch is moved to "SAFE" position for at least two seconds and then returned to "READY." In this operation, the breechblock is retracted by pneumatic pressure in the charger cylinder and held aft as long as the switch remains on "SAFE." When the switch is moved to "READY" the breechblock and a round of ammunition are brought forward to charge the gun.

WARNING

> If stoppage occurs when firing the guns, do not cycle the gun charger. Any attempt to recharge the weapon can cause the explosion of jammed rounds and possible destruction of the aircraft.

The armament safety circuit functions when the landing gear is lowered, de-energizing the gun-charging circuits and causing the breechblock to be retracted to the safe position.

[1] Group Bb, Bc, Bd, C, D and E Airplanes.
[2] Group A, Ba, Bb and Bc Airplanes.

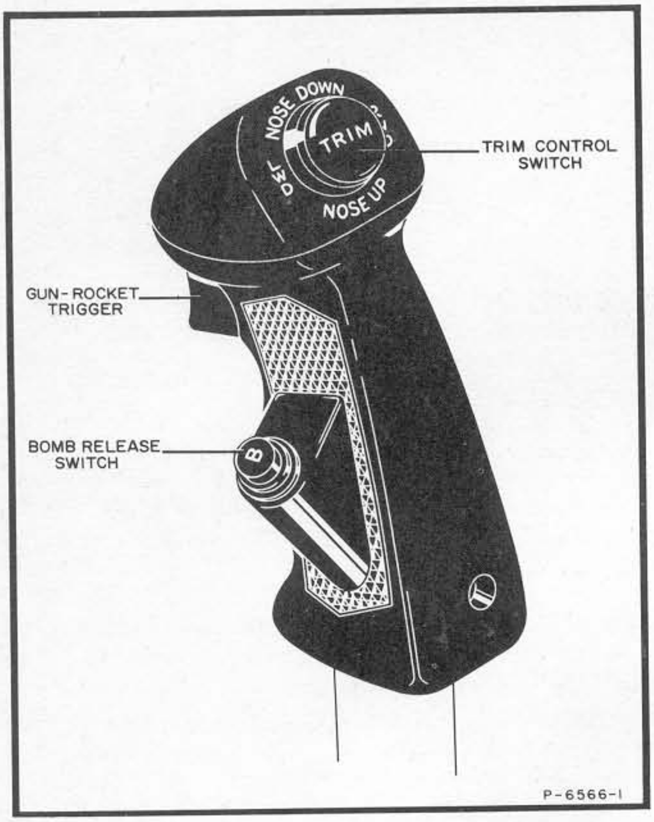

Figure 4-6. Control Stick Switches

GUN-ROCKET FIRING SWITCH. The gun-rocket trigger switch on the control stick grip (figure 4-6) is used to fire both guns and rockets. The guns will fire when the switch is depressed, providing the MASTER ARMAMENT switch is turned "ON" and the GUNS charging switch is at "READY."

Note

To prevent rockets from firing when operating the guns, the function SELECT switch should be set on the "OFF" position or the STATIONS selector switches turned "OFF."

FIRING GUNS

a. MASTER ARMAMENT switch . "ON"

b. Function SELECT switch, or STATIONS selector switch .... "OFF"

c. GUNS switch .... "READY"

d. Gun-rocket trigger .... Squeeze

WARNING

If stoppage occurs during firing of the guns, do not cycle the gun charger. Such action may result in the explosion of jammed rounds.

ROCKET EQUIPMENT

Forward firing air-to-ground rockets of two sizes can be carried in rocket launchers suspended from the bomb racks. The rockets are carried in the package launchers in any combination of the following configurations:

EACH WING STATION	*CENTERLINE STATION*
7—2.75″ rockets	7—2.75″ rockets
19—2.75″ rockets	19—2.75″ rockets
4—5.00″ rockets	4—5.00″ rockets

FIRING ROCKETS

Release of rockets from packages is accomplished by ripple fire. To fire the rockets, observe the following procedure:

a. MASTER ARMAMENT switch . "ON"

b. Function SELECT switch ...... "ALL ROCKETS"

c. STATIONS selector switches (LEFT, CENTER and RIGHT stations) .... "READY" on each station as required

d. GUNS switch .... "SAFE"

e. Gun-rocket trigger .... Squeeze

Note

- The "HALF ROCKET" position of the function SELECT switch is not applicable to rocket package equipment presently in use.
- When it is desired to fire rockets only, make certain that the GUN charging switch is on "SAFE," to prevent the guns from firing simultaneously with the rockets.

JETTISONING ROCKETS

Rockets are carried only in packages hung directly on the bomb racks. The packages may be jettisoned in the same manner as other external stores, using either the normal bomb releasing methods or the emergency releasing controls. (Refer to RELEASING BOMBS.)

BOMBING EQUIPMENT

Release of bombs and other stores from the ejector-type racks is accomplished by electrical detonation of cartridges. When a cartridge is fired by pressing the bomb release switch, the initial force is an upward thrust which opens the hooks, followed by a downward ejector thrust of several inches which forces the bomb clear of the aircraft. The ejector foot is located aft of center on the bomb rack in order to counteract the twisting moment on the bomb caused by drag forces in high speed flight. Each rack contains two cartridges, one for normal release and the other for emergency release.

BOMB ARMING SWITCH. The bomb arming switch (10, figure 4-5), identified as ARM, is located adjacent to the function SELECT switch. The ARM switch can be set at "NOSE & TAIL" or "TAIL" depending on arming requirements, or can be placed in the "SAFE" position for releasing stores in an unarmed condition. As d-c electrical power is needed to arm bombs, the ARM switch will be ineffective during emergency generator operation.

BOMB RELEASE SWITCH. The "pickle" switch (figure 4-6), used for releasing bombs and other external stores, is located on the left side of the control stick grip and is identified by the letter "B." This switch is also used with an alternate installation, the chemical spray tank.

EMERGENCY SELECTOR SWITCH. The EMER SELECT switch (12, figure 4-5), located directly above the EMER BOMB REL handle, provides selection of "WING" or "ALL" stations when jettisoning stores with the EMER BOMB REL handle.

EMERGENCY BOMB RELEASE HANDLE. An EMER BOMB REL handle (13, figure 4-5) on the lower left-hand side of the armament panel is used to jettison external stores. Pulling the handle closes a switch in the emergency release circuit, by-passing the normal release controls. Power to the emergency circuit is supplied by the primary a-c bus which provides either normal or emergency generator power. The emergency bomb release circuit is not affected by any other operation of the aircraft, and stores may be released irrespective of the position of the landing gear control or MASTER ARMAMENT switch. (See EMERGENCY SELECTOR SWITCH.)

RELEASING BOMBS

SINGLE RELEASE

a. MASTER ARMAMENT switch . . "ON"

b. Function SELECT switch . . . . . . "BOMBS"

c. STATIONS selector switches (LEFT, CENTER or RIGHT) . . "READY" as required for single release

d. Bomb release switch . . . . Depress

SALVO RELEASE

a. MASTER ARMAMENT switch . . "ON"

b. Function SELECT switch . . . . . . "BOMBS"

c. STATIONS selector switches (LEFT, CENTER and RIGHT) . . "READY"

d. Bomb release switch . . . . Depress

JETTISON

a. EMER SELECT switch . . . . "WING" or "ALL" (as required)

b. EMER BOMB REL handle . . . . . Pull

CAUTION

When the EMER BOMB REL handle is used to release wing stores only (EMER SELECT switch set on "WING"), make sure the center STATIONS selector switch is in "OFF" position to prevent an electrical feedback through the normal bomb release circuit and inadvertent release of the center store.

FLARE DISPENSER

Information will be supplied when available.

TORPEDOES

The centerline station rack is designed to accommodate a Mk 13 parachute-stabilized torpedo. Other information will be supplied when available.

MINES

Information will be supplied when available.

SPRAY TANK

A chemical spray tank, designated Aero 14A or Aero 14B, can be installed on the fuselage station only.

MISCELLANEOUS EQUIPMENT

ANTI-BLACKOUT SYSTEM

The anti-blackout system utilizes high pressure air bled from the air conditioning refrigeration unit air supply. Air is filtered and directed through a line to the valve located on the ANTI-BLACKOUT panel on the left-hand console. The valve control knob (2, figure 1-3) can be turned to "HI" or "LO" to regulate the amount of air pressure increase in the suit and should be set according to the ability of the pilot to withstand "g" forces. As level flight is attained, and "g" forces are reduced, the valve will automatically reduce the pressure in the suit. A push button on the top of the valve control knob may be manually operated to test the system on the ground or in level flight. *Prior to each flight*, with the engine running and anti-blackout suit connected, depress this button 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 inflate the suit occasionally for body massage to lessen fatigue. The pilot's anti-blackout suit connection plugs into a receptacle (27, figure 1-3) adjacent to the control knob.

MAP CASE

A map case (28, figure 1-3) is installed at the aft end of the left console.

SPARE LAMPS RECEPTACLE

Replacement bulbs for the instrument lamps and for fire warning lamps are contained in the SPARE LAMPS receptacle (12, figure 1-5) on the right-hand console.

RELIEF TUBE

The relief tube is contained in a clip beneath the center of the armament panel. A valve on the relief tube cone is normally closed to prevent loss of cabin pressure. Use of the tube will require compression of the thumb and finger grip at the base of the cone.

BARRICADE STRAP DETENTS

Three barricade strap detents are installed on each leading edge to insure proper barricade engagement. Two of the detents are spaced evenly on the wing slat; the third is on the leading edge of the wing, inboard of the slat (18, figure 1-2).

REAR VIEW MIRRORS

A rear view mirror is installed on each side of the canopy bow to provide rearward vision during flight and taxiing.

SECTION V
OPERATING LIMITATIONS

See AN 01-40AVA-1A, Confidential Supplement
to Flight Handbook

SECTION VI
FLIGHT CHARACTERISTICS

See AN 01-40AVA-1A, Confidential Supplement
to Flight Handbook

SECTION VII
SYSTEMS OPERATION

INTRODUCTION

A pilot's proficiency is not measured solely by the finesse with which he performs an inverted corkscrew clover-leaf or like maneuver designed to impress the uninitiated, but by the manner in which he gains the utmost from his aircraft through intelligent employment of the many systems and components which have been placed in the aircraft for just that purpose. This can be attained only by diligent study of the airplane as an entity . . . learning the theory and operation of the systems, determining the relationship of one system to another, knowing the maximum capabilities of each, and then correlating his ability to fly with his understanding of the aircraft. Then, and only then, can a pilot be considered proficient. With this goal in mind, the following section is included.

ENGINE FUEL CONTROL SYSTEM

MANUAL FUEL CONTROL SYSTEM OPERATION

GROUND CHECK. The manual fuel control system lacks the automatic compensating devices found in the primary system; thus, when operating on the former, it is necessary to exercise great caution in manipulation of the throttle. Slow, steady power changes must be made to prevent excessive overspeeding and extreme exhaust gas temperatures. High temperatures have a tendency to break down the structure of the metals in the engine, causing distortion and subsequent failure. Because of this, the failure of operating personnel to properly govern these critical aspects of engine operation has led to restricted ground use of the manual fuel control system. A complete ground check of the system will no longer be made by the pilot prior to flight, nor will the ground crew fully check it before each day's operations. Instead, at each 30-, 60-, 90-, and 120-hour inspection, and whenever a new fuel control unit is installed, the responsible personnel will conduct a test of the manual fuel control system as follows:

a. FUEL CONTROL switch......."PRIMARY"

b. RPM 90%

c. FUEL CONTROL switch......."MANUAL"

MANUAL FUEL warning light should be illuminated.

Note

The engine rpm will increase or decrease immediately upon moving the FUEL CONTROL switch to "MANUAL", depending on the variation between existing ambient conditions and those comprising a standard day. If standard day conditions prevail, no apparent change in engine speed will occur.

d. *Slowly* advance the throttle to "MILITARY".

RPM .... "MIL" ± 2 needlewidths
TEMP .... "MIL" (maximum)
FUEL BOOST .... "NORM"
OIL PRESS .... "NORM"

CAUTION

Do not hold RPM at two needlewidths above MIL".

Note

If the manual fuel control system is functioning properly, the RPM will be 1 to 1½ needlewidths below "MIL" on a standard day. At 80°F, however, the RPM should read "MIL".

e. *Slowly* retard the throttle to "IDLE".

RPM .... "IDLE"
TEMP .... "MIL" (maximum)
FUEL BOOST .... "NORM"
OIL PRESS .... "NORM"

f. *Slowly* increase the RPM to 90%.

g. Retard the throttle to "IDLE".

h. Move the FUEL CONTROL switch to "PRIMARY" when the RPM decreases past 70%.

IN-FLIGHT SWITCH-OVER. The engine fuel control operation may be changed from the primary to the manual system during flight at all altitudes by retarding the throttle to "IDLE" and moving the FUEL CONTROL switch to the "MANUAL" position. The switch-over to the manual system may be accompanied by a surge in engine speed and tailpipe temperature, the severity of which depends on the difference between ambient atmospheric conditions and those of a standard day. As the primary system incorporates devices which compensate for these differences and the manual system does not, when the switch-over is made, the sudden lack of compensation results in a change in fuel flow, thus causing the temporary aberration in operating characteristics.

After a switch-over, and when tailpipe temperature permits an acceleration, the throttle may be advanced slowly and smoothly to the desired power setting, provided that the engine operating limitations given in Section V are not exceeded. It must be remembered that when operating on the manual fuel control system, all fuel metering to the engine is accomplished by direct movement of the throttle; therefore, all subsequent power changes must be made with the same care, not only to prevent overspeeding and extreme temperatures, but also to avoid a flame-out from the possible inability of the engine to parallel, in speed, the rapidly changing fuel flow during quick accelerations and decelerations.

During take-off, if it becomes expedient to switch to the manual fuel control system, it is inadvisable to retard the throttle to "IDLE" prior to the switch-over. Because of the rapidity with which the engine decelerates upon failure of the primary fuel control system—the deceleration from 100% to 80% RPM requires approximately ½ second—the change must be made very quickly to prevent a complete loss of power. The throttle should be retarded only enough to keep the tailpipe temperature within limits after the switch-over, then advanced slowly to the necessary power setting.

HYDRAULIC POWER SUPPLY SYSTEMS

Since both of the engine-driven hydraulic pumps in the tandem hydraulic system and the one pump in the single hydraulic system are all of the constant pressure, variable displacement type, the flow of fluid through each system will vary in rate (gallons per minute) with the operating speed of the associated pump. Each pump is geared to the engine in such a manner that for each revolution of the engine, the hydraulic pump will complete approximately one-half revolution. At military power, fluid flows through the system at the approximate rate of six gallons per minute, while at sea level idle approach power the flow is reduced to slightly over two gallons per minute. As rate of fluid flow determines the speed at which the various hydraulically operated units respond to actuation of their individual controls, this variation in rate of flow with power changes during normal operation might ordinarily produce objectionable characteristics in operation of the hydraulic power supply system. Therefore, flow restrictors have been installed in the subsystems to regulate the maximum rate of flow. The flow restrictors prevent the landing gear, wing flaps, speed brakes, and arresting hook from operating too fast when fluid flow is at its peak, yet do not affect the time of operation when flow is reduced, as at low engine speeds.

Each hydraulic actuating cylinder within the systems requires a certain quantity of fluid to be delivered through its inlet port to completely extend the cylinder piston. The rate of flow determines how quickly the piston extends, and the pressure at which the fluid is delivered dictates the maximum load against which the piston will extend. Since the pump delivers fluid at a constant pressure, regardless of rotating speed, it is easily seen that as long as the engine is turning, even windmilling under flame-out conditions, the hydraulically operated units will operate satisfactorily against the usual loads; however, as rate of fluid flow diminishes with decreasing engine speed, the full extension of the hydraulically actuated pistons will require a greater period of time. The effects of these characteristics in the single and tandem hydraulic systems will be discussed separately.

SINGLE HYDRAULIC SYSTEM[1]

When two or more controls are actuated simultaneously during normal or flame-out conditions, one system naturally responds more quickly than the other, depending on which system offers the least resistance. When this system is satisfied, or reaches the point where its resistance to pressure is greater than the other actuated system, flow is diverted to the latter. This is true of all sub-systems except the flight control power systems. The two priority valves in the main hydraulic system serve to maintain approximately 1500 psi within the aileron power control and elevator boost when other components are actuated, thus preventing complete loss of flight control power during the period that other units are being operated.

TANDEM HYDRAULIC SYSTEM[2]

In the tandem hydraulic system the flight controls are powered by two completely separate hydraulic systems, however, when units other than the flight controls are being operated by the utility hydraulic system, the effectiveness of the flight controls will be reduced, especially at high air speeds. Within the utility system, when two controls are operated at the same time at very low engine speeds, one will respond more quickly than the other, depending on which offers the least resistance. When operating on the utility system alone, actuations of various units normally operated by the utility pressure will cause a temporary decrease in the effectiveness of the flight controls. Total effectiveness at high speed will be approximately one quarter of normal, and at reduced engine speed effectiveness will be still less.

[1] Group A, B, C and D Airplanes.
[2] Group E Airplanes.

SECTION VIII
CREW DUTIES

NOT APPLICABLE

ALL-WEATHER OPERATION
SECTION IX

TO BE ADDED

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