PILOT'S HANDBOOK FOR NAVY MODEL F6F AIRPLANE

HELLCATS

OVER THE PACIFIC

WWII FLIGHT SIMULATOR

Pilot's Handbook

Graphic Simulations Corporation wishes to thank the following people for their contributions to this manual:

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Introduction

For specific instructions on operating **Hellcats Over the Pacific** on your computer, reference the **User's Manual** included with this package.

The **User's Manual** provides instructions on installation of **Hellcats**, commands, scoring, mission objectives and techniques.

The **Pilot's Handbook for Navy Model F6F Airplane**, contains additional reading material provided to give the user a glimpse into the real world of operating a WWII vintage fighter airplane. This book contains F6F flight operations material provided by the Grumman Corporation, the F6F manufacturer, as well as flight training material provided by the Federal Aviation Administration.

I. Pilot's Handbook for Navy Model F6F Airplane

This section contains the handbook given to Navy fighter pilots operating an F6F Hellcat. Each of the controls and instruments on the various F6F models are explained in detail.

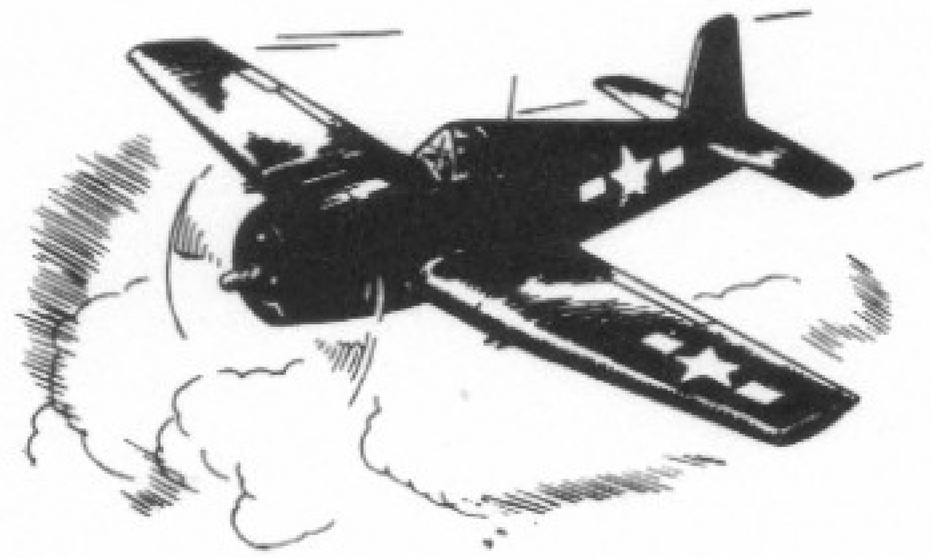
Pilot's Handbook

for

NAVY MODEL

F6F-3 • F6F-3N
F6F-5 • F6F-5N

Airplanes



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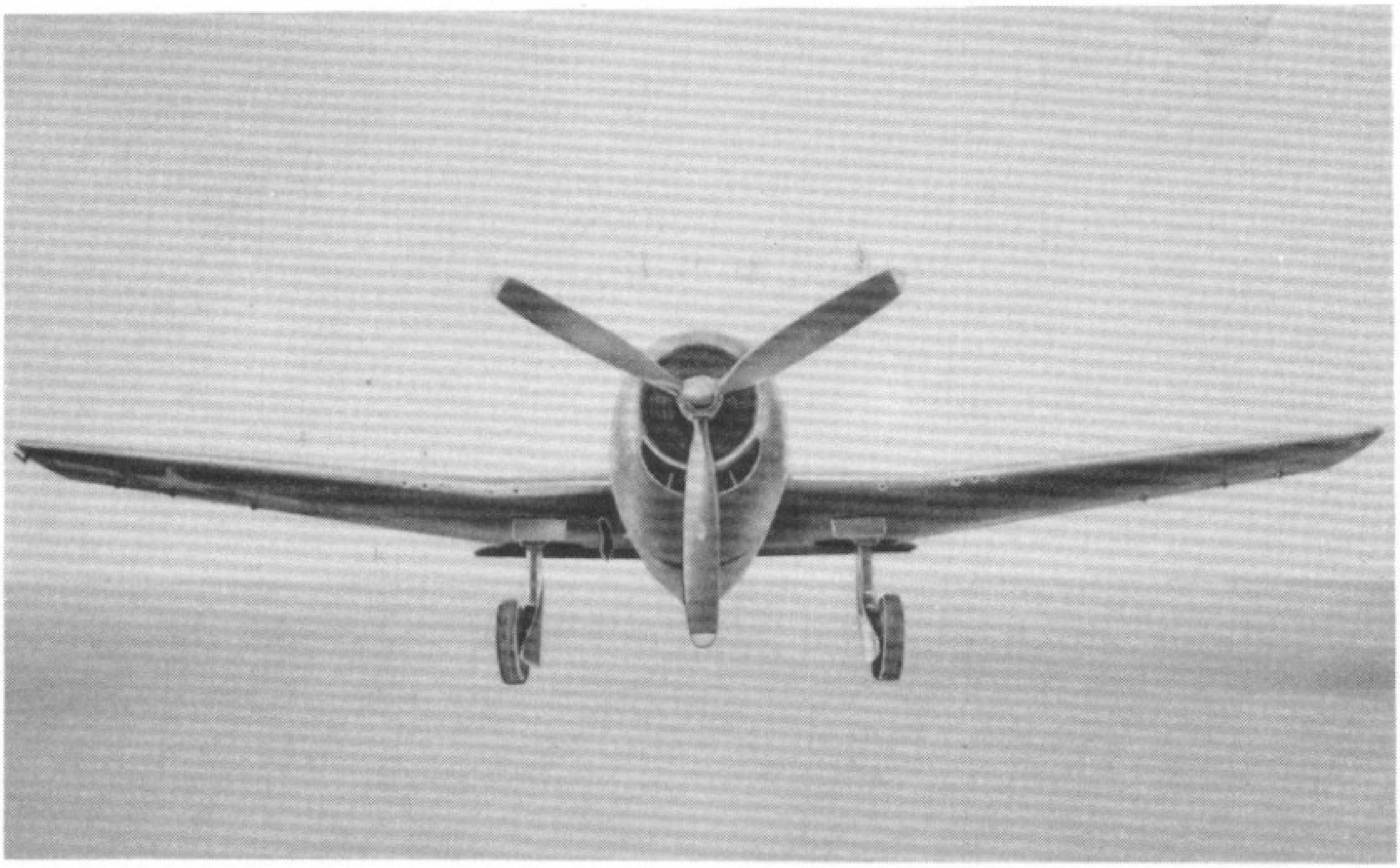
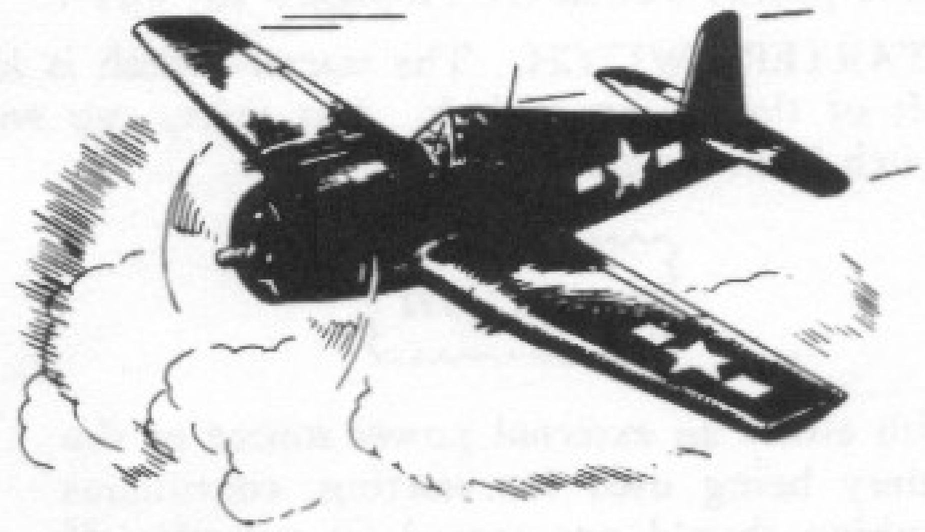


Figure 1-1. Airplane (F6F-5)—Front View

Figure 1-2. Airplane (F6F-5)—Three Quarter Left Rear View



SECTION I
DESCRIPTION

1-1. AIRPLANE.

1-2. The F6F-3, -3N, -5 and -5N airplane is a Class VF single engine, single place, folding low wing fighter. It is designed for either unassisted or catapult take-off from aircraft carriers. The landing gear, wing flaps, cowl flaps, intercooler and oil cooler shutters, wing hinge locking pins, and gun chargers are operated hydraulically. The arresting hook is operated electrically. The wings are folded and spread manually.

1-3. Two main fuel cells located left and right of the centerline in the wing center section have a capacity of 87.5 gallons each. A 75 gallon reserve tank is located in the fuselage under the cockpit floor. A 150 gallon droppable tank can be installed on the fuselage bomb rack and a 100 or 150 gallon droppable tank on each wing bomb rack.

1-4. The armament consists of six .50 cal. machine guns, three in each wing outer panel. Later model airplanes are designed to accommodate mixed batteries of guns, four .50 cal. machine guns and two 20 mm cannon. Two 1000 lb. bombs can be carried under the wing center section or a bomb or torpedo under the fuselage. When the airplane is operating with the torpedo or bomb under the fuselage, 100 gallon droppable fuel tanks can be carried under the wing center section. Three sets of rocket launchers are installed on the underside of each wing outer panel outboard of the guns.

Normal Weight	12480 lbs.
Span (Wings Spread)	42'10"
Span (Wings Folded)	16'2"
Fuselage (Height Overall)	14'5"
Fuselage (Length)	33'10"

1-5. POWER PLANT.

1-6. GENERAL. A Pratt and Whitney R-2800-10 or 10W two stage, two speed, supercharged engine drives a Hamilton Standard Hydromatic three-bladed propeller. The R-2800-10W engine is equipped with a water injection system.

1-7. POWER PLANT CONTROLS.

1-8. CONTROL QUADRANT. The engine control quadrant is located on the left hand cockpit shelf and includes the throttle lever, supercharger lever, mixture lever, and propeller governor lever with vernier handwheel.

1-9. THROTTLE LEVER. Handle with the microphone switch button on the top.

a. Move AFT to "CLOSED".

b. Move FORWARD to "OPEN".

1-10. SUPERCHARGER LEVER. Handle marked (SC) in the center of the control quadrant.

a. Move FORWARD for "NEUTRAL".

b. Move to CENTER for "LOW".

c. Move AFT for "HIGH".

1-11. MIXTURE LEVER. Handle marked (M) on inboard side of control quadrant.

a. Move FULL AFT to "IDLE CUT-OFF".

b. Move FORWARD to "AUTO LEAN" and "AUTO RICH".

c. "FULL RICH" position has been rendered inoperative.

1-12. PROPELLER GOVERNOR LEVER AND VERNIER HANDWHEEL. The propeller pitch is controlled hydraulically by a governor unit located on the nose section of the engine. The propeller governor control lever, marked (P), is located on the aft end of

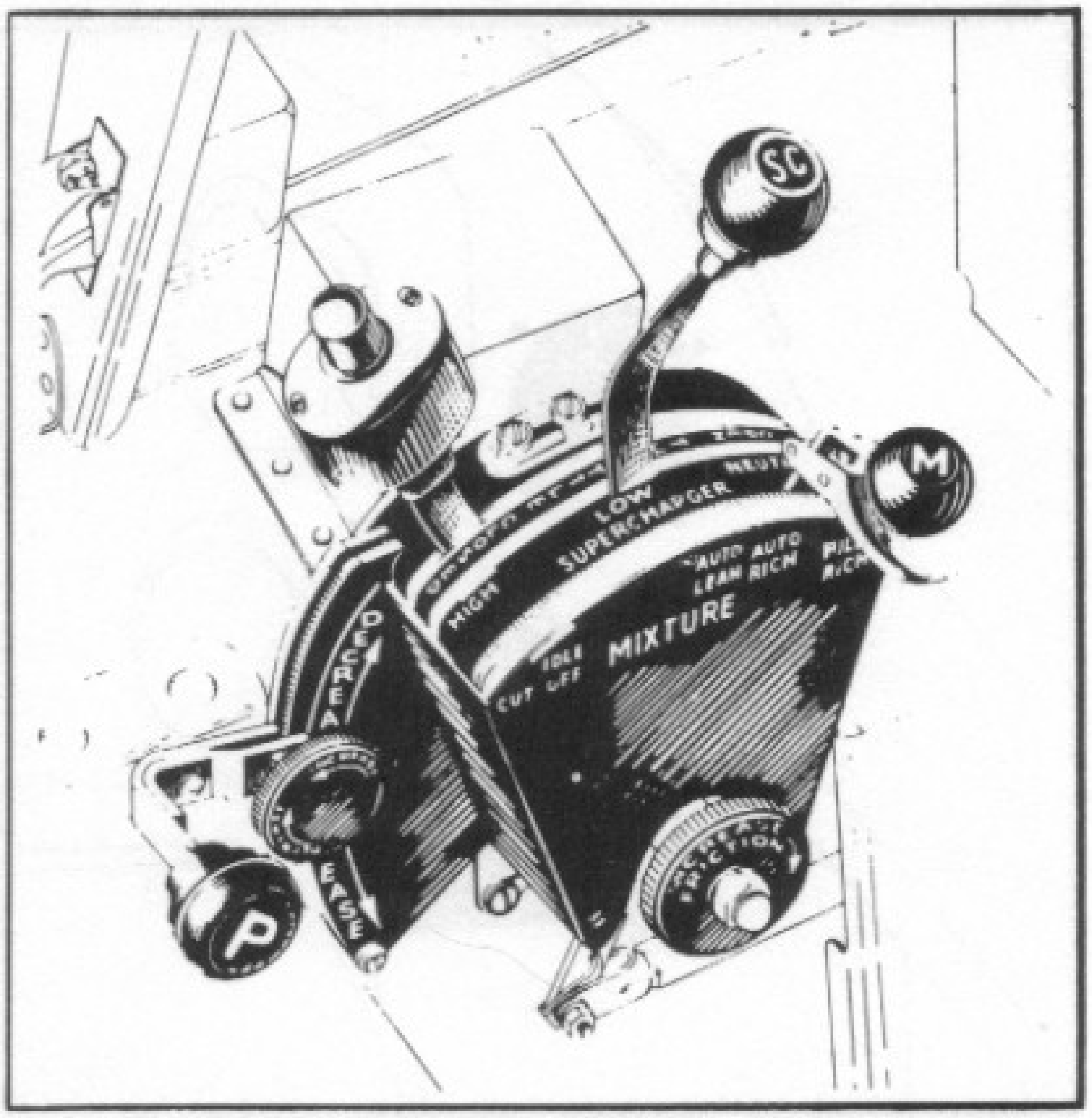


Figure 1-3. Engine Control Quadrant

the engine control quadrant. The vernier handwheel for fine pitch adjustment is located at the base of the governor lever.

a. Move lever UP to "DECREASE RPM" (Increase Pitch).

b. Move lever DOWN to "INCREASE RPM" (Decrease Pitch).

c. Rotate vernier handwheel CLOCKWISE to "DECREASE RPM".

d. Rotate vernier handwheel COUNTERCLOCKWISE to "INCREASE RPM".

1-13. FRICTION ADJUSTMENT KNOB. On inboard side at the base of the control quadrant (for throttle and propeller levers only).

a. Rotate CLOCKWISE to "INCREASE FRICTION".

b. Rotate COUNTERCLOCKWISE to "DECREASE FRICTION."

1-14. IGNITION SWITCH. The ignition switch is mounted to the left of the main instrument panel. The switch has four positions: "OFF", "R", "L", and "BOTH".

1-15. PRIMER AND STARTER CONTROLS. The primer and starter control switches are located at the forward end of the electrical distribution panel. Early model airplanes are equipped with cartridge starters and later models with direct cranking electric units.

1-16. PRIMER SWITCH. The primer switch is located at the forward end of the electrical distribution panel adjacent to the starter switch. The battery switch, also on this panel, must be "ON" to prime and start the engine.

a. Move primer switch INBOARD to "OFF".

b. Move primer switch OUTBOARD to "ON".

1-17. STARTER SWITCH. The starter switch is located aft of the primer switch. Lift safety cap and hold switch "ON" to start engine.

CAUTION

With either an external power source or the battery being used for starting, continuous cranking should not exceed 60 seconds. If the engine does not start, open the starter switch and allow the starter to cool for at least one minute. If the engine fails to start after a few attempts, check engine. Since the induction vibrator (booster coil) is designed for 60 seconds operation, continuous use of the starter with a cold engine may burn out the induction vibrator.

1-18. COWL FLAPS CONTROL. The spring loaded cowl flap hydraulic control lever is located on the left hand cockpit shelf.

a. Move lever FORWARD to "OPEN".

b. Move lever to CENTER to "NEUTRAL" (Lock).

c. Move lever AFT to "CLOSE".

1-19. The cowl flaps, can be operated by the hydraulic hand pump by turning the hand pump selector valve to "ENGINE FLAPS", when the engine-driven hydraulic pump is not operating. See Section IV.

1-20. CARBURETOR AIR CONTROL. The two position carburetor air control (auxiliary stage) "T" handle is located on the left hand side of the instrument panel.

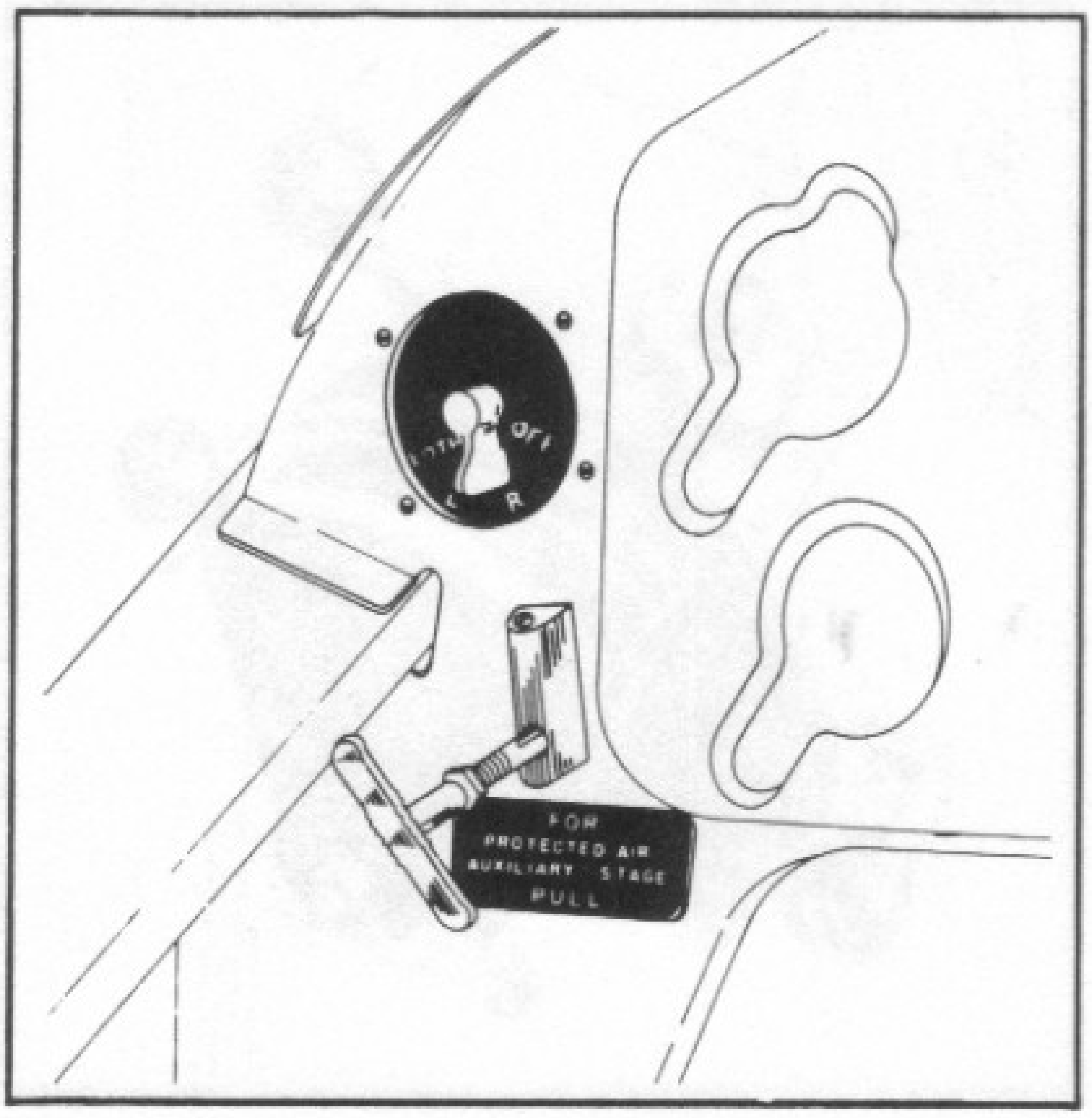


Figure 1-4. Ignition Switch and Carburetor Air Control

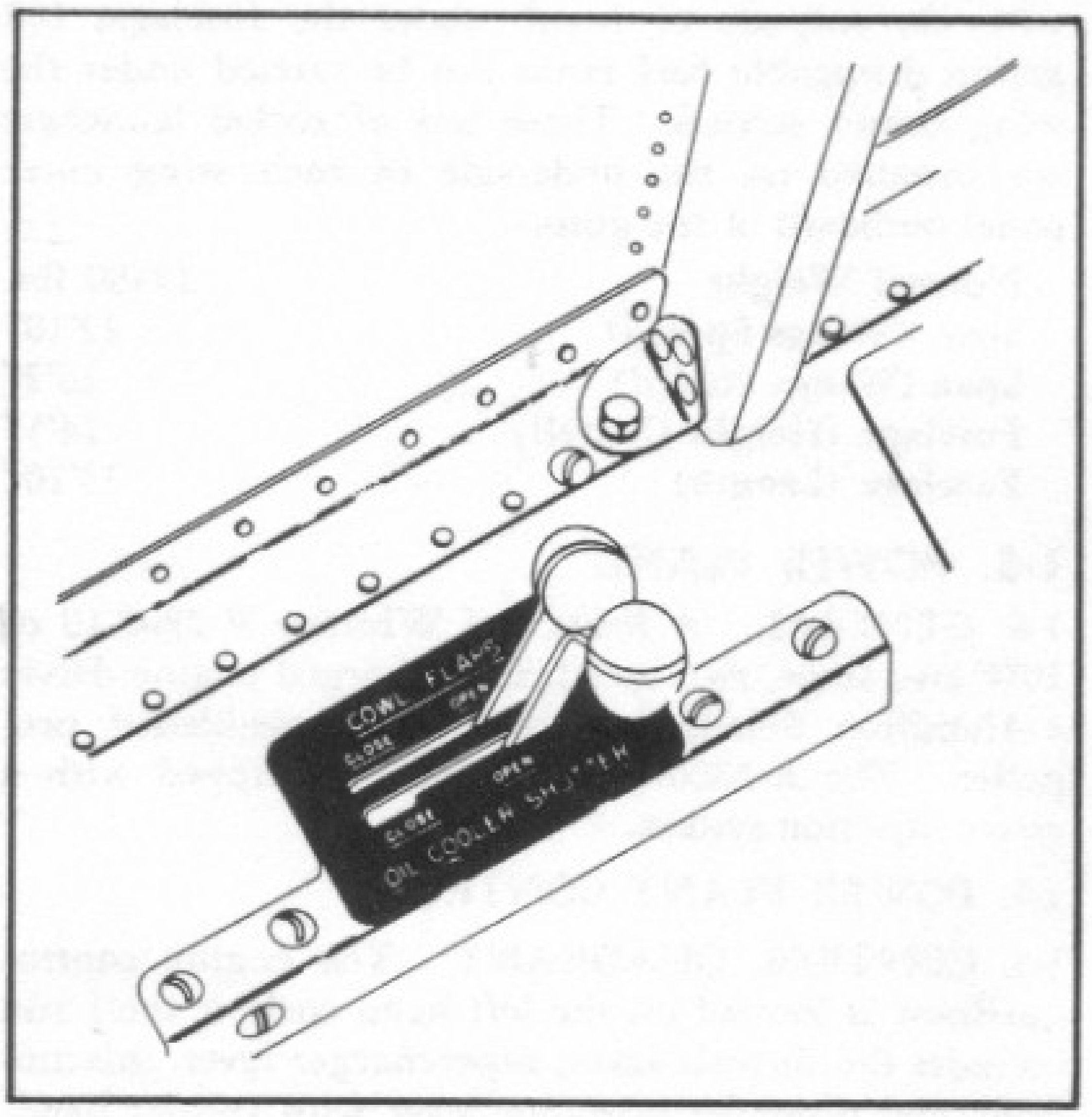


Figure 1-5. Cowl Flaps Control

a. Push "T" handle FULL FORWARD for "DIRECT".

b. Pull "T" handle FULL AFT for "PROTECTED AIR AUXILIARY STAGE".

1-21. This handle regulates only air coming from the auxiliary stage inlet duct located in the lower section of the nose spinning, and does not affect main stage air which is taken from the accessory compartment when operating in neutral blower. The primary function of the control is to prevent direct ramming air from entering the auxiliary stage when carburetor air filters are being used to filter main stage air while operating in neutral blower.

1-22. The combination of the Stromberg Injection Carburetor and the Pratt & Whitney blower-throat, fuel discharge nozzle and spinner, with the absence of distribution vanes in the blower throat, makes the Double Wasp two stage engine unusually free from icing tendencies. However, ice can form in the induction system ahead of the auxiliary stage when the outside air temperature is below 0°C (32°F) and free moisture is present. Under icing conditions mentioned above, the air control should be shifted to "PROTECTED AIR".

CAUTION

Do not use an intermediate position of this control.

1-23. WATER INJECTION SYSTEM CONTROL. The WEP switch which actuates the water injection fluid pump is located on the left hand shelf just outboard of the engine control quadrant. Early model airplanes are equipped with a quantity gage for the fluid tank (16 gallons), located to the right of the main instrument panel and connected electrically to a float type transmitter in the tank.

a. Move WEP switch AFT to "OFF".

b. Move WEP switch FORWARD to "ON".

1-24. Before the water injection system is put into operation by moving the throttle FULL FORWARD, the switch must be placed in the "ON" position so that the water pump will be allowed to build up the required pressure necessary for the operation of the system.

1-25. WATER REGULATOR. A water regulator is located in the engine accessory compartment. A line extends from the regulator to the carburetor spray nozzle unit. An electrical solenoid valve on the regulator controls the flow of water through this line. A micro-switch located on the engine control box controls the valve. A micro-switch is actuated by a tab attached to the throttle control rod. This tab can be adjusted, thereby allowing control over the water regulator at various manifold pressures.

1-26. MANIFOLD PRESSURE SWITCH. Airplane BUAERO #10801 and subsequent, are equipped with an electric manifold pressure switch connected in parallel with the engine throttle micro-switch. This manifold pressure switch closes the water regulator solenoid valve circuit when the manifold pressure reaches 54 inches and opens the circuit when pressure falls below 51 inches; thus allowing a gradual reduction in horsepower when the throttle is moved AFT from the "FULL FORWARD" position.

Note

In the F6F-3 and -3N when the water pump switch is turned "ON", it immediately begins the operation of the pump but the water regulator solenoid valve does not open until the throttle is FULL FORWARD. However, in the F6F-5 and -5N the water pump switch and the throttle actuating switch are wired in series. When the water pump switch is turned "ON", the water pump will not operate until the actuating switch is closed by moving the throttle FULL FORWARD, then operation of the water pump and opening of the water regulator solenoid are effected.

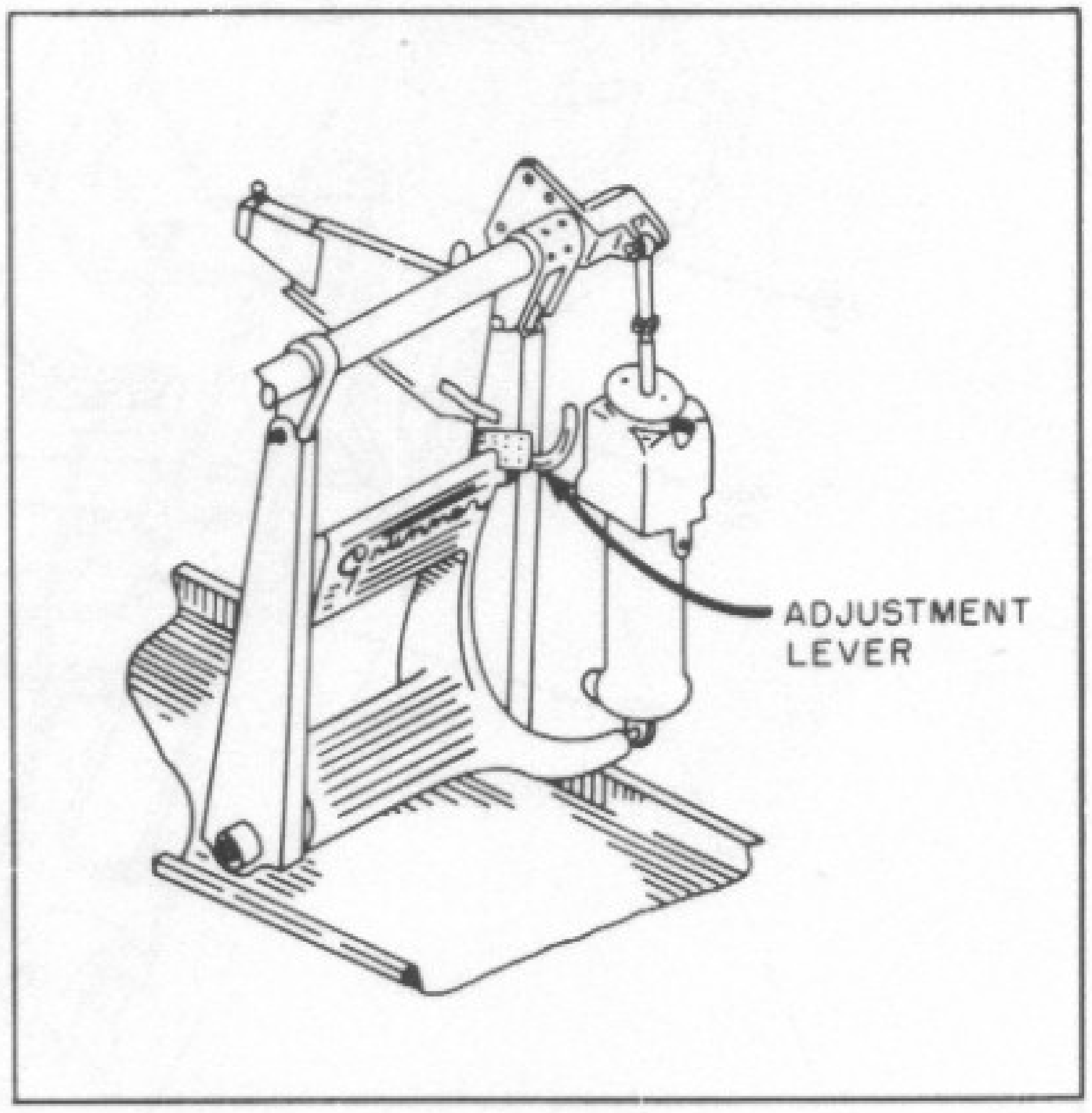


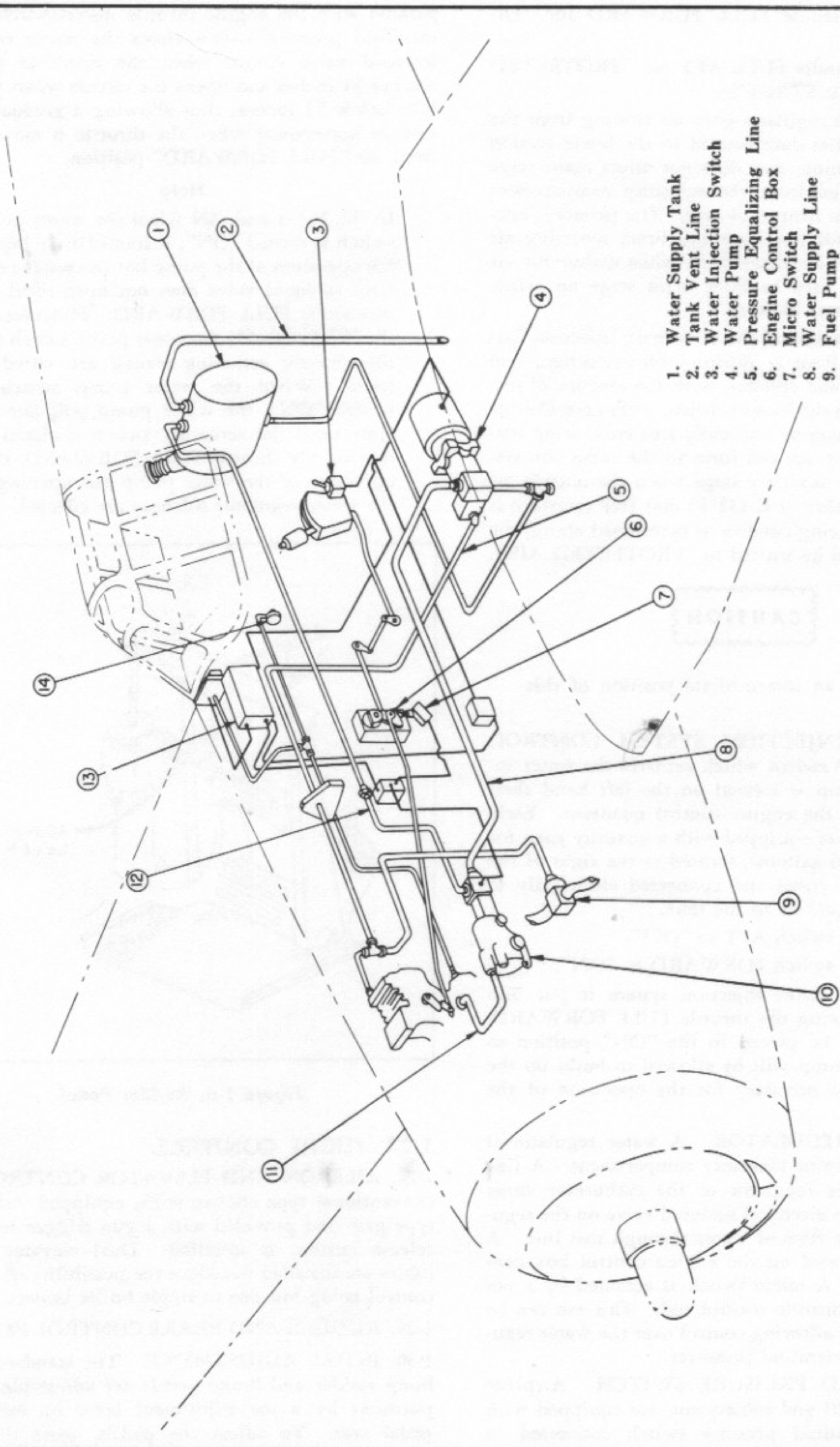
Figure 1-6. Rudder Pedal

1-27. FLIGHT CONTROLS.

1-28. AILERON AND ELEVATOR CONTROLS. A conventional type control stick, equipped with pistol type grip and provided with a gun trigger and bomb release button, is installed. Dual elevator control cables are installed to reduce the possibility of elevator control being lost due to single bullet impact.

1-29. RUDDER AND BRAKE CONTROL PEDALS.

1-30. PEDAL ADJUSTMENT. The standard underhung rudder and brake pedals are adjustable to four positions by a toe adjustment lever on each outer pedal arm. To adjust the pedals, press the levers



1. Water Supply Tank
2. Tank Vent Line
3. Water Injection Switch
4. Water Pump
5. Pressure Equalizing Line
6. Engine Control Box
7. Micro Switch
8. Water Supply Line
9. Fuel Pump
10. Water Regulator and Solenoid
11. Water Line to Fuel Spinner Nozzle
12. Vapor Vent Line
13. Distribution Box
14. Manifold Pressure Switch

Figure 1-7. Water Injection System Diagram (F6F-5 and -5N) down and push the pedals full forward with the toes; then put the toes under the pedals and pull aft one notch at a time to the desired position. Make certain that each pedal has ratcheted past the same number of notches.

1-31. Pedal motion is transmitted to the rudder by cables running aft to a bellcrank which drives two push-pull rods which deliver force differentially to the rudder quadrant.

1-32. AILERON, ELEVATOR AND RUDDER TRIM TAB CONTROLS. The trim tab controls for the left aileron, elevators and rudder are mounted as a unit on the left hand side of the cockpit. The operation of the controls is standard.

Note

> On the F6F-5 and -5N airplanes, the ailerons are equipped with spring tabs which lighten the control forces. Their operation is completely automatic. The left aileron tab also operates as a trim tab in the conventional manner.

1-33. AILERON TAB CONTROL (knob on forward side of unit). Rotate knob CLOCKWISE for "LEFT WING DOWN"—Rotate knob COUNTERCLOCKWISE for "RIGHT WING DOWN".

1-34. ELEVATOR TAB CONTROL (wheel on inboard side of unit). Rotate wheel CLOCKWISE for "NOSE DOWN"—Rotate wheel COUNTERCLOCKWISE for "NOSE UP".

1-35. RUDDER TAB CONTROL (knob on top of unit). Rotate knob CLOCKWISE for "NOSE RIGHT"

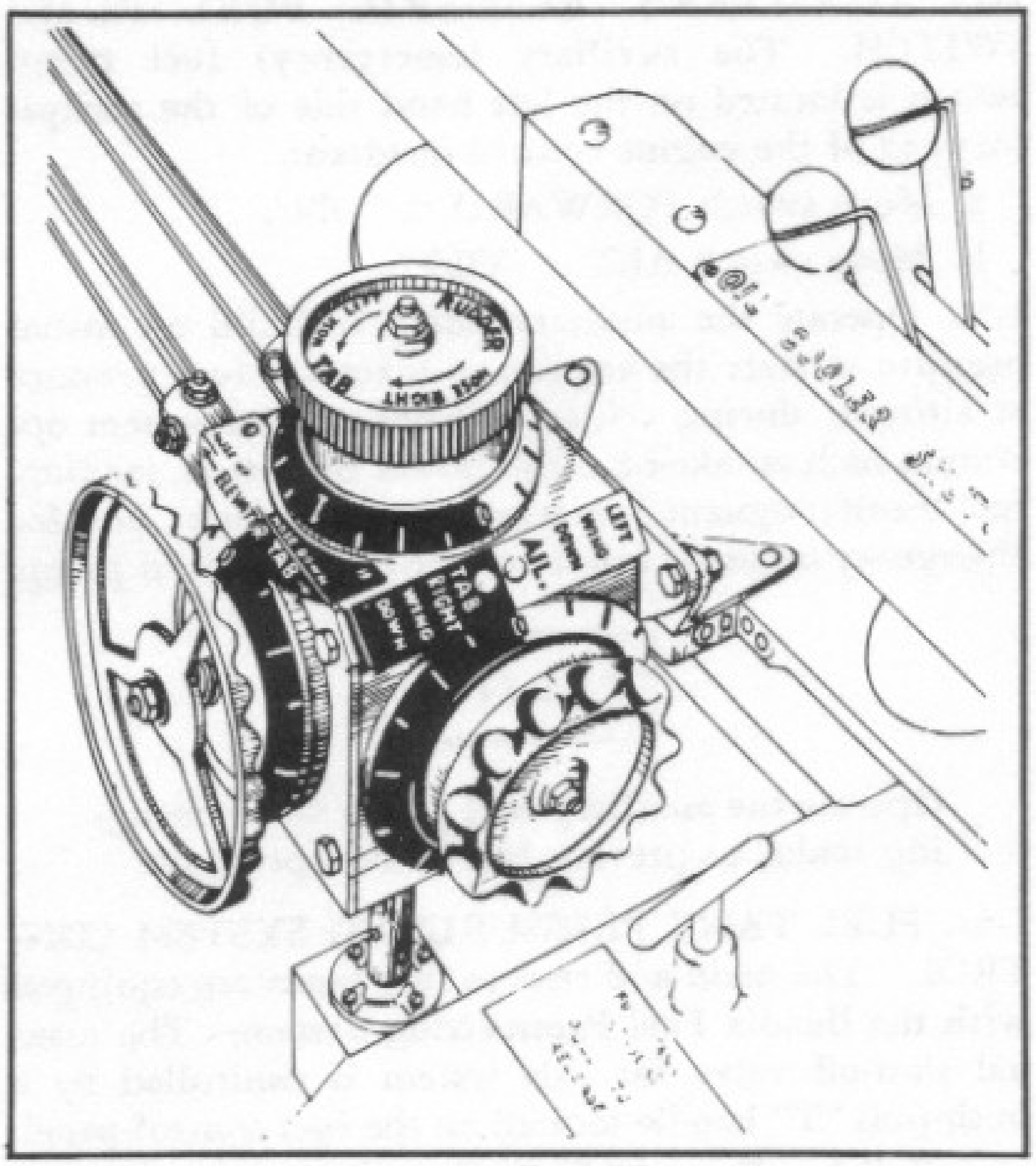


Figure 1-8. Tab Control Wheels

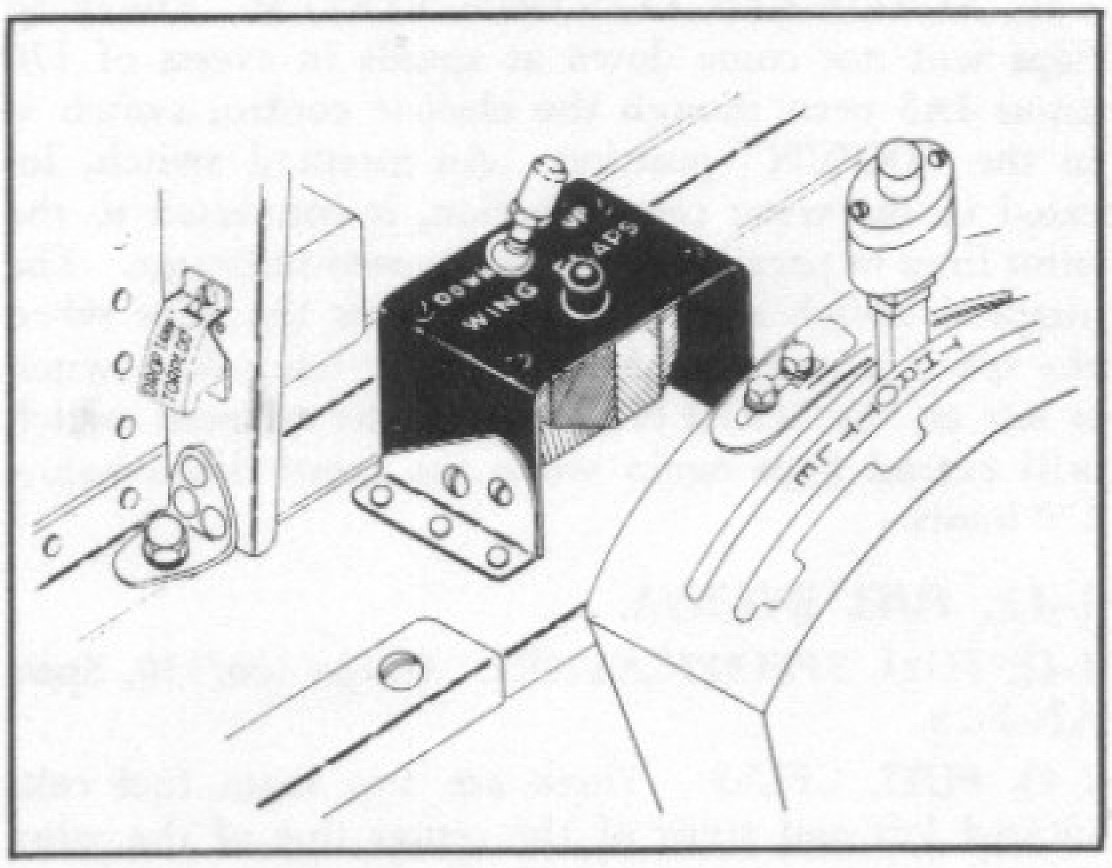


Figure 1-9. Wing Flaps Control

—Rotate knob COUNTERCLOCKWISE for "NOSE LEFT".

1-36. WING FLAPS CONTROL.

1-37. GENERAL. The wing flaps, of the low drag type, are actuated by four hydraulic cylinders controlled from the cockpit. Two cylinders are installed in each wing, one inboard and one outboard. A combined landing gear and wing flap position indicator is installed outboard of the landing gear control lever.

1-38. OPERATION. During normal conditions the wing flap hydraulic control valve, which actuates the four cylinders, is operated by an electric servo motor controlled by a toggle switch on the left hand shelf adjacent to the engine control quadrant. (For emergency operation of the wing flaps, refer to section IV.)

a. Move switch FORWARD for "FLAPS UP".

b. Move switch AFT for "FLAPS DOWN"

CAUTION

> The flaps are held down only by hydraulic pressure remaining constant—there is no other lock. In an emergency when loss of pressure or leaks are indicated in the system, lower the flaps last. If the flaps are lowered first, the force of the airstream may overcome the pressure and force them up, and there may not be enough fluid remaining in the system to lower them again.

1-39. OPERATION OF COMPRESSION SPRING UNITS. Four blow-up spring units, one connected to each flap, are installed to allow the flaps to "blow-up" with increasing airspeed. These spring units automatically control the flap angle in flight when the flaps are "DOWN". The spring units are not controllable from the cockpit and are entirely independent of the hydraulic cylinders. The range of flap angle is from 50° at 93 knots IAS to 15° at 150 knots IAS.

1-40. AUTOMATIC CONTROL SYSTEM. The wing flaps will not come down at speeds in excess of 170 knots IAS even though the electric control switch is in the "DOWN" position. An airspeed switch, located in the wing center section, is connected to the pitot lines in parallel with the airspeed indicator. The airspeed switch automatically retracts the flaps when the speed exceeds 170 knots. If the wing flap switch is left on the "DOWN" position, the airspeed switch will extend flaps again when the speed drops below 170 knots.

1-41. FUEL SYSTEM.

1-42. FUEL SPECIFICATION. Grade 100/130, Spec. AN-F-28.

1-43. FUEL CELLS. There are two main fuel cells located left and right of the center line of the wing center section. A reserve fuel cell is located in the fuselage under the cockpit floor. A 150 gallon droppable tank can be carried under the belly of the airplane, and two 100 or 150 gallon droppable tanks under the wing center section bomb racks.

1-44. TANK CAPACITIES.

	Gallons	
Left Main	87.5 U.S.	73 Imp.
Right Main	87.5 U.S.	73 Imp.
Reserve	75 U.S.	62 Imp.
Fuselage Droppable	150 U.S.	125 Imp.
Wing Droppable	100 U.S.	83 Imp.
	or 150 U.S.	125 Imp.

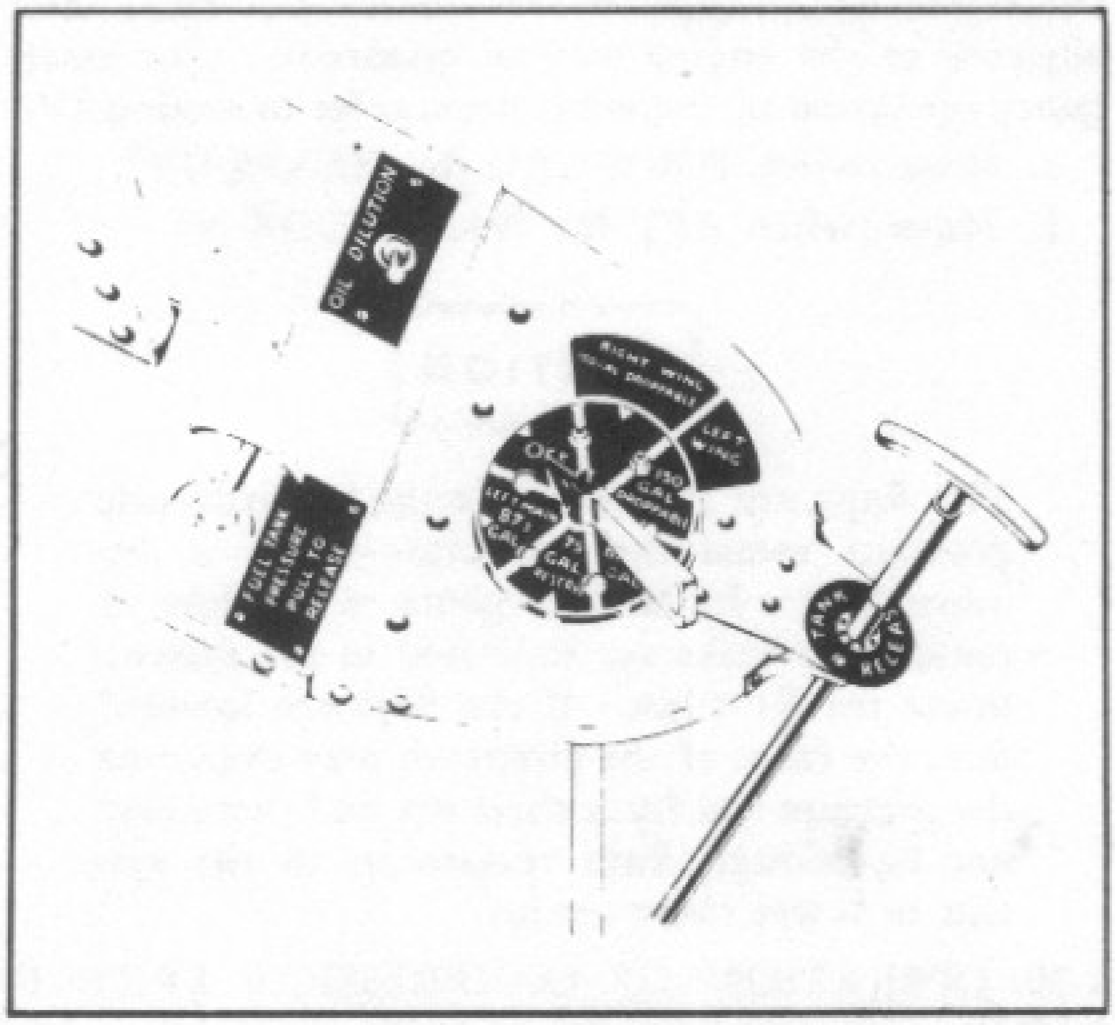


Figure 1-10. Fuel Control Panel

1-45. FUEL SYSTEM CONTROLS.

1-46 FUEL QUANTITY GAGE. The fuel quantity gage is located on the lower right hand instrument panel. Three individual pointers register for right and left main and reserve tanks.

Note

The vapor return line from the carburetor leads to the right main tank, and discharges approximately one to two gallons of fuel per hour.

1-47. RESERVE TANK WARNING LIGHT. Early model airplanes are equipped with a fuel warning light for the reserve tank located on the fuel control panel. The light will glow when the reserve tank contains 50 gals. or less. The fuel quantity gage should then be closely watched and the airplane flown at the most economical speed tactically feasible. Leave lamp on BRIGHT—Rotate to DIM.

1-48. TANK SELECTOR VALVE. The fuel tank selector valve dial and handle are located on the center of the fuel control panel.

a. Rotate pointer to FORWARD-OUTBOARD position for "OFF".

b. Rotate pointer to AFT-OUTBOARD position for "LEFT MAIN".

c. Rotate pointer to AFT position for "RESERVE".

d. Rotate pointer to AFT-INBOARD position for "RIGHT MAIN".

e. Rotate pointer to FORWARD-INBOARD position for "150 GAL. DROPPABLE" or "LEFT WING".

f. Rotate pointer to forward position for "RIGHT WING DROPPABLE".

Note

Be certain pointer is centered on selected tank.

1-49. AUXILIARY ELECTRIC FUEL PUMP SWITCH. The auxiliary (emergency) fuel pump switch is located on the left hand side of the cockpit forward of the engine control quadrant.

a. Move switch FORWARD to "ON".

b. Move switch AFT to "OFF".

1-50. Operate the auxiliary pump to build up initial pressure to start the engine, to maintain fuel pressure at altitude, during critical periods of fuel system operation such as take-off, high power operation, landing, fuel transfer system operation, changing tanks, and for emergency in case of failure of the engine-driven pump.

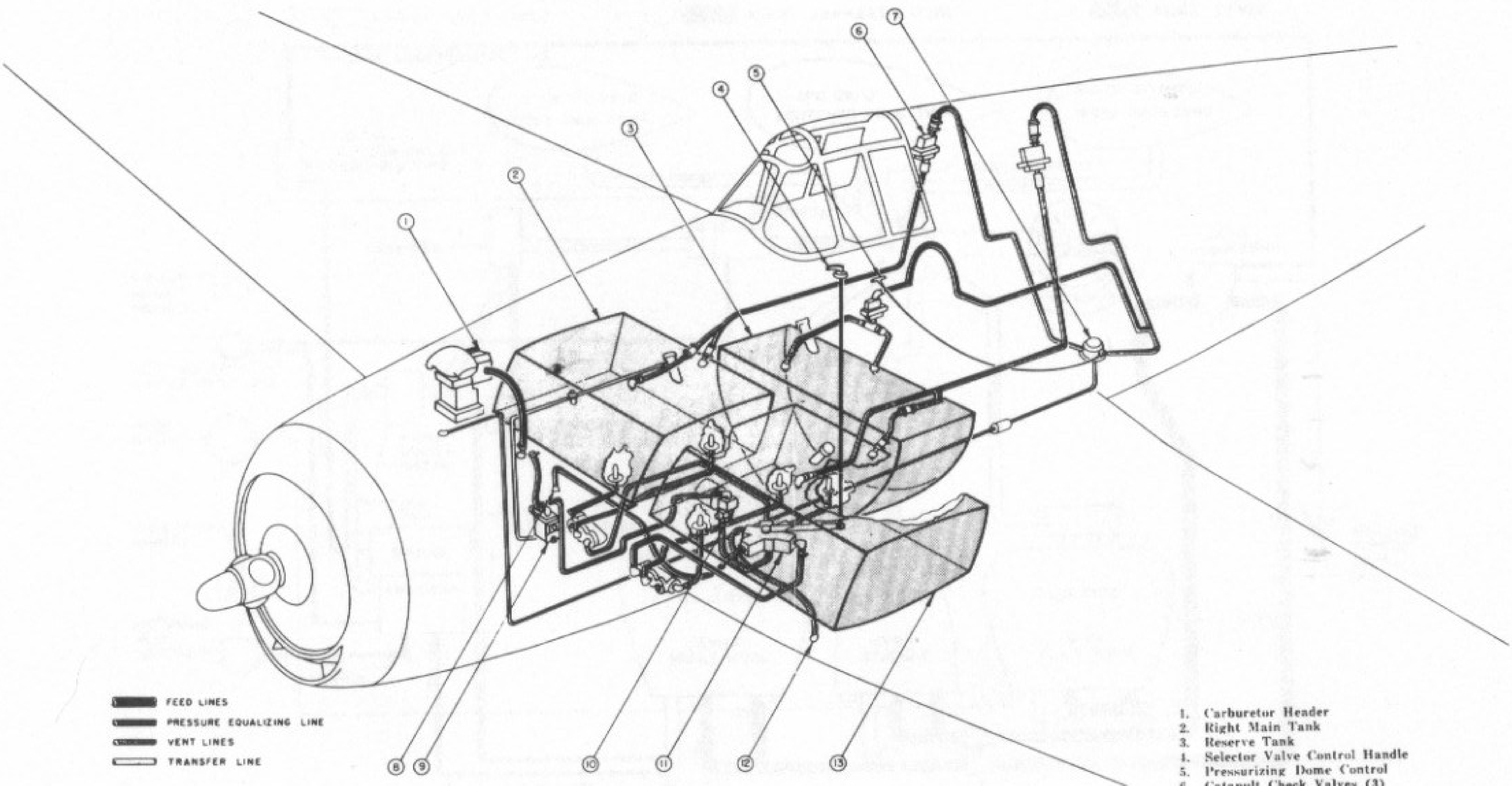
CAUTION

Operate the auxiliary fuel pump when changing tanks, to prevent loss of fuel pressure.

1-51. FUEL TANK PRESSURIZING SYSTEM CONTROL. The main and reserve fuel tanks are equipped with the Bendix Fuel Pressurizing System. The manual shut-off valve for this system is controlled by a push-pull "T" handle located on the fuel control panel.

a. Pull handle to "RELEASE PRESSURE".

b. Push handle to "RESTORE PRESSURE".



1. Carburetor Header
2. Right Main Tank
3. Reserve Tank
4. Selector Valve Control Handle
5. Pressurizing Dome Control
6. Catapult Check Valves (3)
7. Pressure Regulator Dome
8. Line to Right Wing Droppable Tank
9. Engine-Driven Fuel Pump
10. Auxiliary Fuel Pump (Electric)
11. Selector Valve
12. Line to Left Wing (or Fuselage) Droppable Tank
13. Left Main Tank

Figure 1-11. Fuel System Diagram

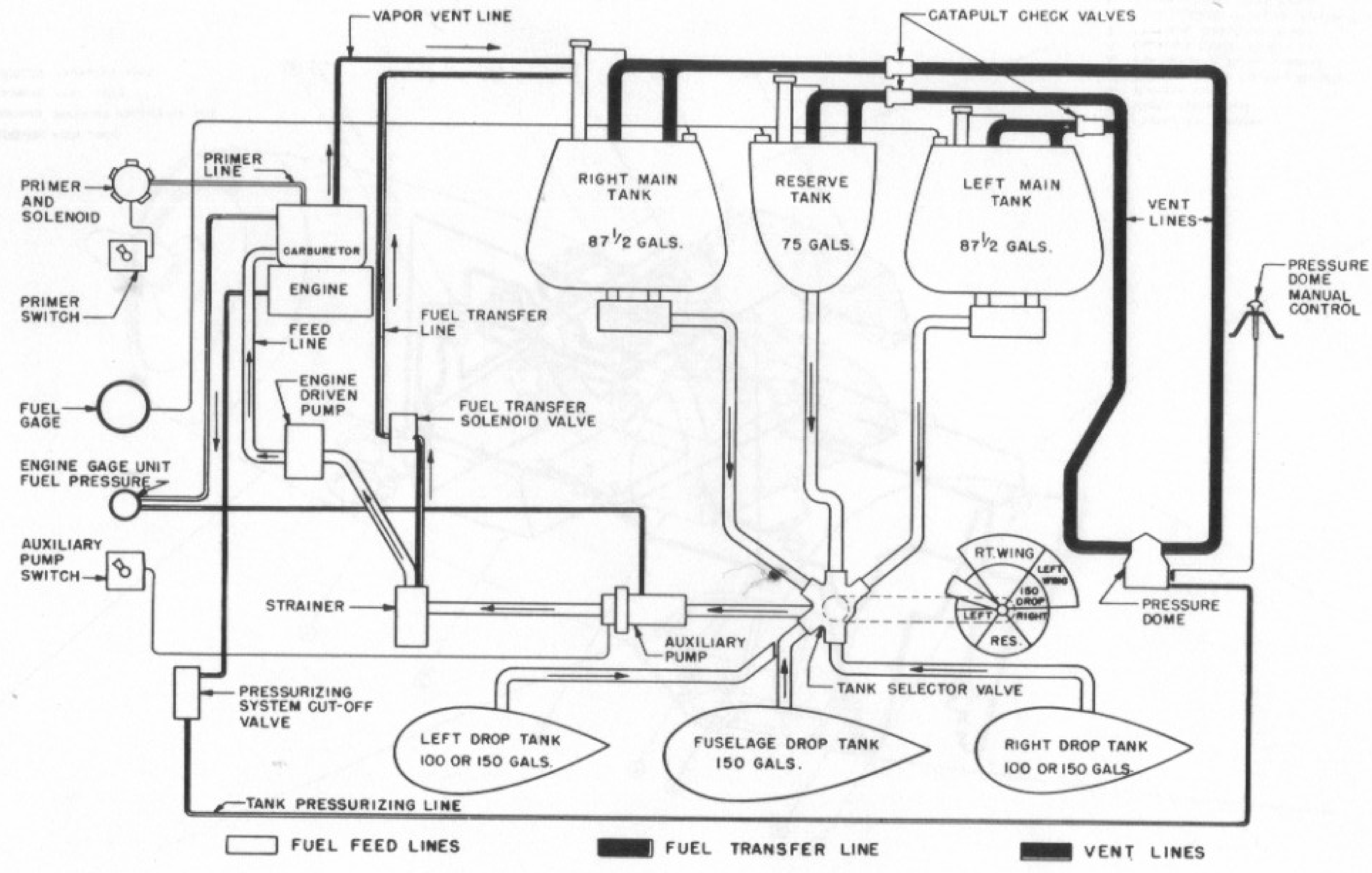


Figure 1-12. Fuel System Control Diagram

Note

> The pressurizing system is effective at altitudes above approximately 12000 ft and is used for operations at higher altitudes than can otherwise be realized with the use of the auxiliary fuel pump only, and for high altitude economical cruising.

1-52. FUEL PRESSURE. The fuel pressure gage is located on the right hand instrument panel.

Desired	—17 psi
Allowable	—16-18.5 psi
Minimum Idling—	7 psi

1-53. FUEL TRANSFER SYSTEM (F6F-5 AND -5N). The fuel transfer system replaces fuel consumed from the right main tank with that from any selected droppable tank. The fuel transfer solenoid valve is energized through a cam on the fuel tank selector when any droppable tank position is selected. This solenoid valve opens for fuel transfer only when all three of the following conditions are in effect, thus completing the electrical circuit:

a. A droppable tank is selected.

b. Fuel level in the right main tank drops below 81 gallons.

c. The auxiliary fuel pump switch is turned to "ON" position.

1-54. When the fuel level of the right main tank reaches 81 gallons, the system stops transferring automatically.

1-55. DROPPABLE TANKS RELEASE CONTROLS.

1-56. FUSELAGE DROPPABLE TANK RELEASE. A release switch with safety cap is located on the left hand side of the cockpit just above the cowl flaps control. A manual release is provided in the form of a "T" handle control located on the inboard side of the fuel control panel. To release the fuselage droppable tank, proceed as follows:

a. Turn auxiliary fuel pump switch—"ON".

b. Turn fuel tank selector valve handle to a full tank.

c. Pull the manual release "T" handle to the "UP" position then rotate clockwise.

d. Press spring-loaded release switch on left hand side of cockpit.

e. If electrical switch fails to function, pull the manual release "T" handle to "FULL UP" position.

1-57. WING DROPPABLE TANKS RELEASE. The left and right wing droppable tanks are released by the armament controls. A manual release for each tank is provided in the form of a "T" handle control located to the left of the lower control panel. To release the wing tanks, proceed as follows:

a. Turn auxiliary fuel pump switch—"ON".

b. Turn fuel tank selector valve handle to a full tank.

c. Bomb—RP selector switch—"BOMB".

d. Switch for tank or tanks (left—right shackle) to be released—"ON".

e. Armament master switch—"ON".

f. Press bomb button on stick grip.

1-58. OIL SYSTEM.

1-59. OIL SPECIFICATION. Grade 1100 or 1120, Spec. AN-O-8.

1-60. TANK AND COOLER. The oil tank, located in upper part of the engine accessory compartment, has a capacity of 19 U.S. (15.8 Imperial) gallons with a three gallon foaming space. The tank is provided with a warm-up compartment. The oil cooler, containing an automatic oil temperature control valve, is located in the bottom of the fuselage just aft of the firewall. The control valve causes the oil to by-pass the cooler when the oil-from-engine temperature is below 54°C (130°F), directing the outlet oil-from-engine back to the bottom of the oil tank for warm-up. Consequently, the tank supply of oil is by-passed when starting the engine, until the oil-from-engine temperature reaches approximately 54°C (130°F). The oil is then passed through the core of the cooler and returned to the top of the oil tank.

1-61. OIL SYSTEM CONTROLS.

1-62. OIL COOLER AND INTERCOOLER SHUTTERS. The oil cooler and intercooler shutters control the flow of cooling air to the oil cooler and the intercoolers, and are operated by a hydraulic control lever. On the F6F-3 and -3N airplanes one control lever, located on the left hand cockpit shelf, controls both the intercooler and oil cooler shutters.

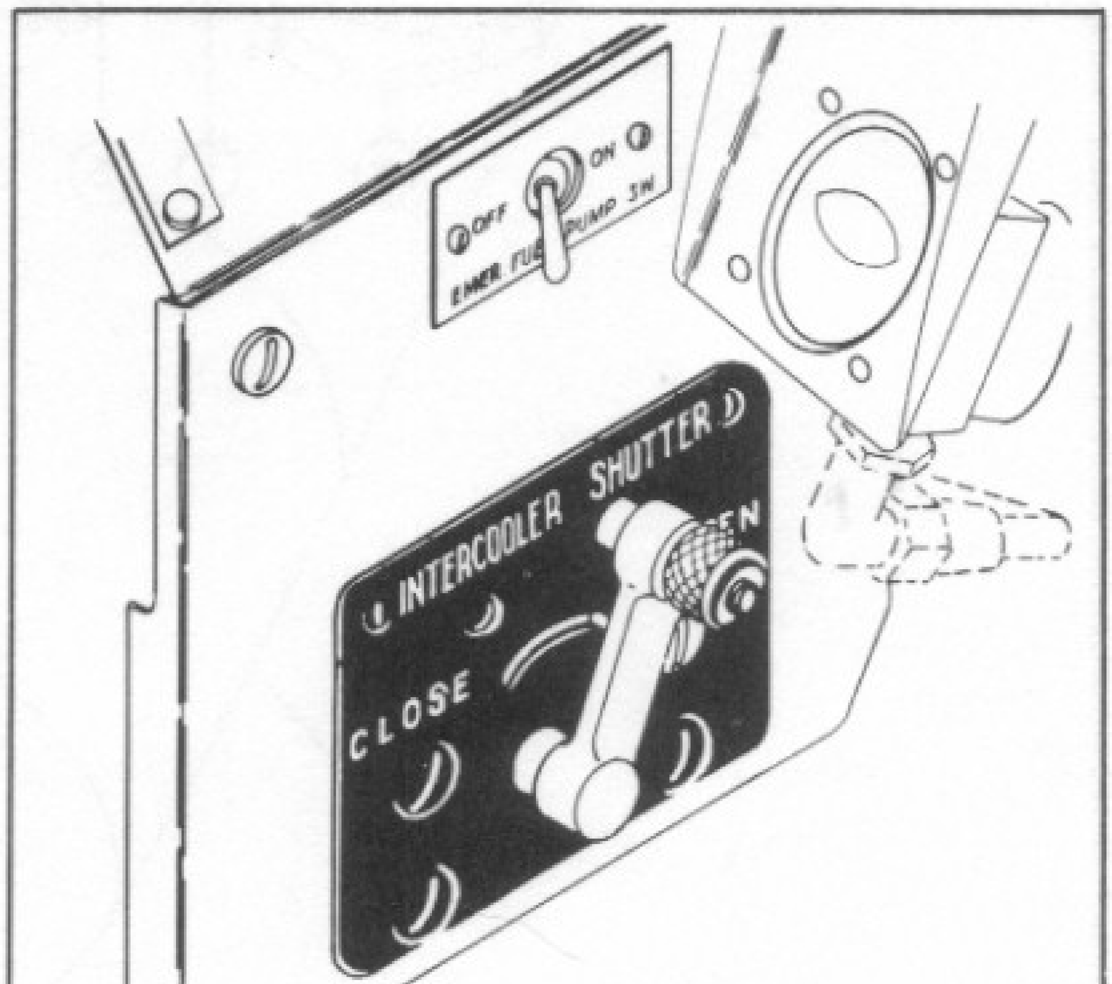


Figure 1-13. Intercooler Shutters Control

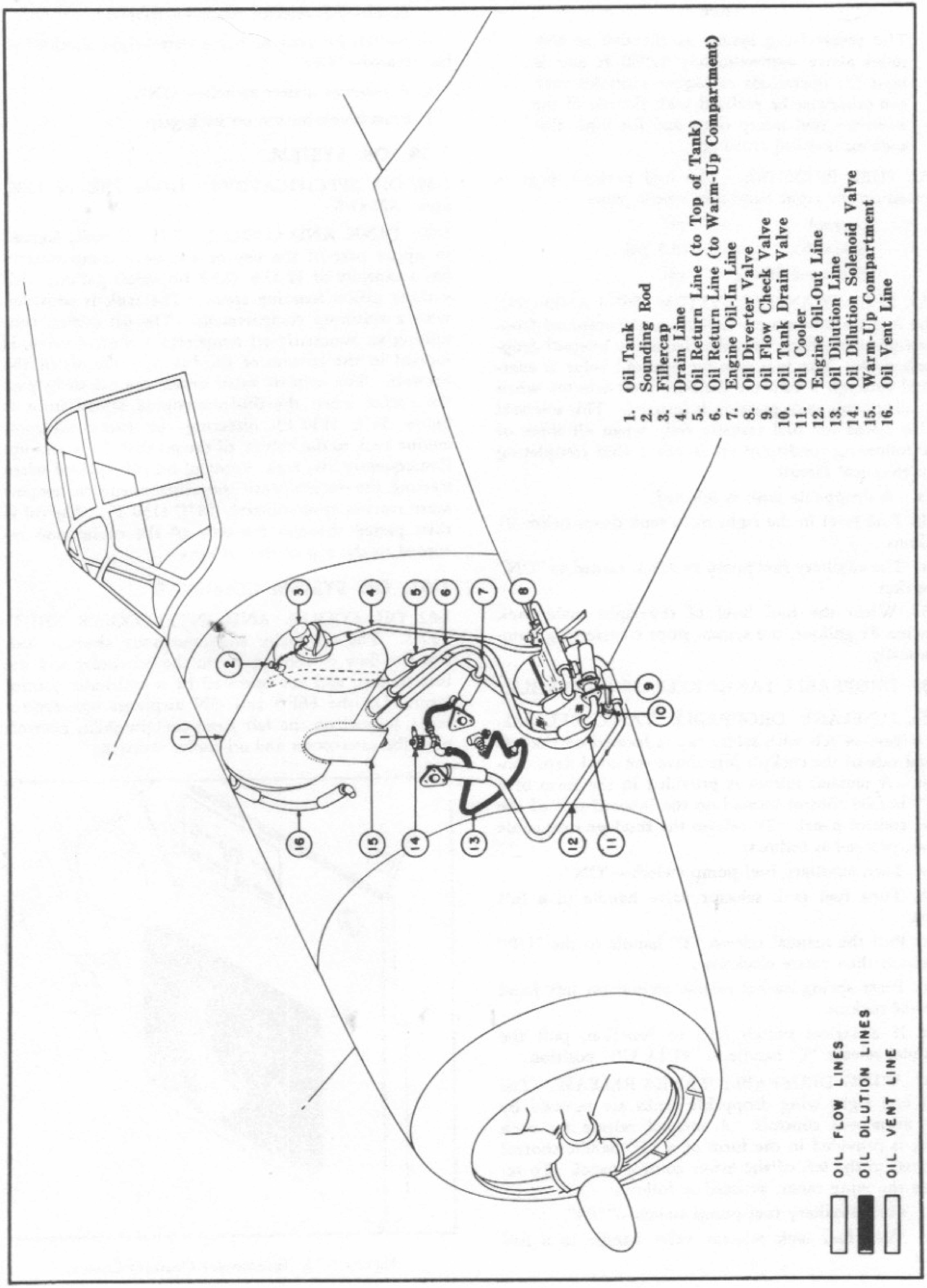


Figure 1-14. Oil System Diagram

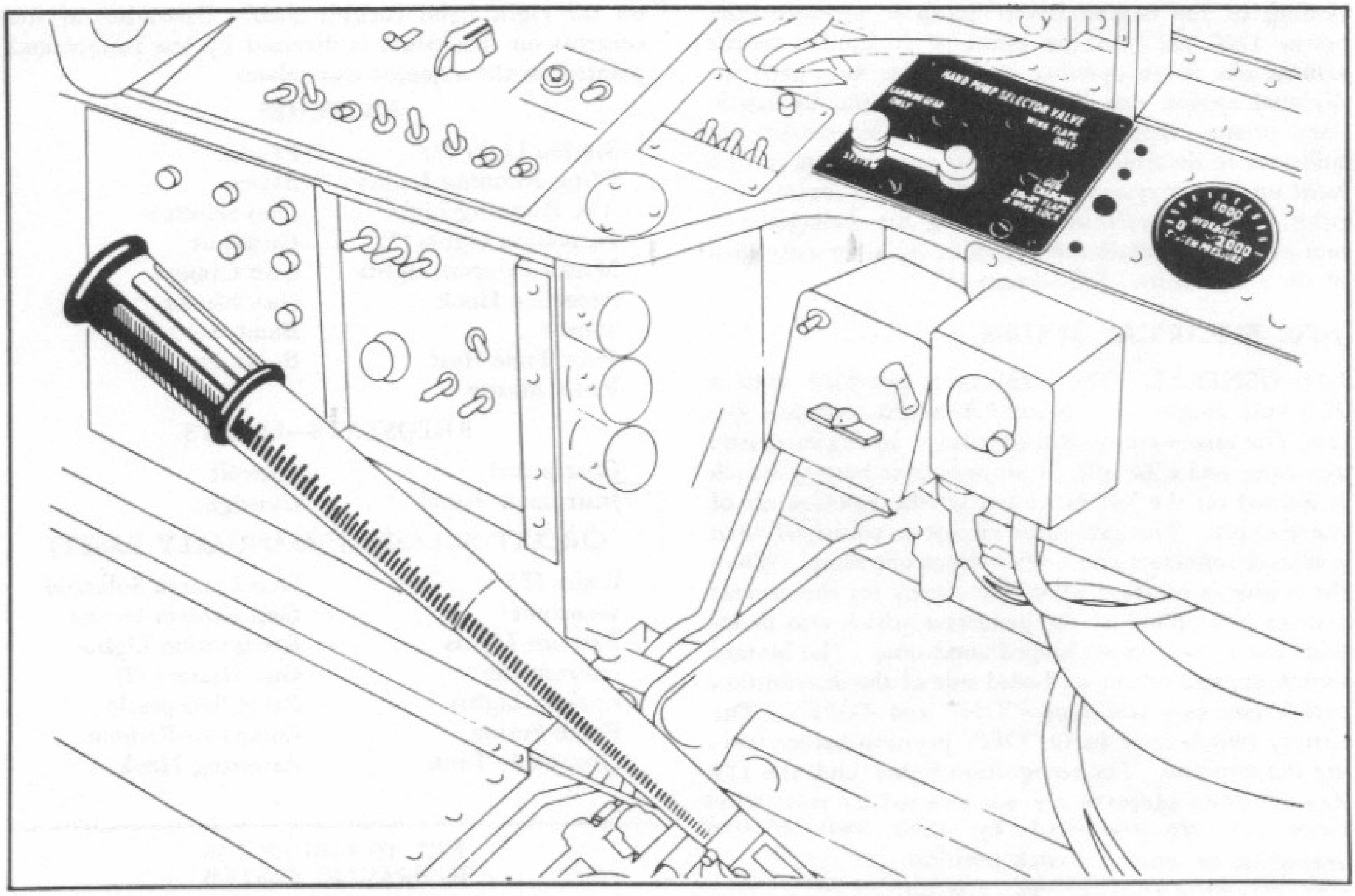


Figure 1-15. Hydraulic Hand Pump, Hand Pump Selector Valve, and Hydraulic Pressure Gage

a. Move lever FORWARD to "OPEN".

b. Move lever AFT to "CLOSE".

1-63. INTERCOOLER SHUTTERS. On the F6F-5 and -5N airplanes there are separate control levers for each unit. The intercooler shutters control lever is located on the left hand side of the cockpit forward of the fuel control panel.

a. Move lever FORWARD to "OPEN".

b. Move lever AFT to "CLOSE".

1-64. OIL DILUTION SWITCH. The oil dilution switch, that controls the oil dilution cut-off valve and diverter valve, is located on the fuel control panel. For operation of oil dilution, refer to Section II.

1-65. HYDRAULIC SYSTEM.

1-66. GENERAL. The hydraulic system, containing approximately four U.S. gallons (3.32 Imperial gallons) of fluid, Spec. AN-VV-O-366 (red color), operates the landing gear, wing flaps, cowl flaps, oil cooler and intercooler shutters, gun chargers, and wing hinge lockpins.

1-67. NORMAL OPERATION. The hydraulic system is normally operated by the engine-driven hydraulic pump located in the engine accessory compartment. The various functions of the system are controlled by hydraulic selector control valves located in the cockpit. The system pressure gage is located on the right hand cockpit shelf. The normal hydraulic system operating pressure is 1500 psi.

1-68. HAND PUMP OPERATION. A hydraulic hand pump is located on the cockpit floor to the right of the pilot's seat. This pump is used to operate the hydraulic system when the engine-driven pump is not functioning. The hand pump selector valve control, located on the right hand cockpit shelf, governs the systems to be operated by the hand pump. The selector valve handle has four positions—reading clockwise, they are: "SYSTEM", "LANDING GEAR ONLY", "WING FLAPS ONLY", "GUN CHARGING—ENGINE FLAPS AND WING LOCK".

CAUTION

When the hydraulic hand pump is not being used, the hand pump selector valve control handle should be kept in the "SYSTEM" position.

1-69. CHECKING INDIVIDUAL SYSTEMS. Under normal operation, when the hand pump selector valve control handle is moved from point to point, the system pressure gage indicates the pressure in that particular system selected. Thus, if the control handle is set on "SYSTEM", the gage will indicate malfunctioning of the engine-driven pump if pressure falls below 1200 psi. In the event of hydraulic system failure, due to an opening in a line or unit, each individual system can be checked with the hydraulic hand pump. Approximately eight to ten strokes are sufficient to determine whether or not pressure can be built up in that system. When the leak is located, the pilot shall then refrain from using the damaged system in order to retain the hydraulic fluid for operation of the other units. See Section IV.

1-70. ELECTRICAL SYSTEM.

1-71. GENERAL. The airplane is provided with a 28.0 volt, single wire, grounded return electrical system. The power equipment consists of an engine-driven generator and a 24 volt, 11 ampere-hour battery which is located on the left hand side of the fuselage aft of the cockpit. The generator circuit is equipped with a voltage regulator and a reverse current relay. When the engine is running, all of the energy for the various systems is supplied by the generator which also maintains the battery in a charged condition. The battery switch, located on the outboard side of the distribution panel, has two positions—"ON" and "OFF". The battery switch must be in "OFF" position before leaving the airplane. The recognition lights and the **IFF destructor circuits** are not affected by this switch since they are energized by their own controls regardless of battery switch position.

1-72. SYSTEM UNITS. The following units are operated electrically:

- Wheel & Flap Position
- Starter
- Fuel Transfer System
- Water Injection System
- Compass—Remote
- Wing Flaps Control
- Arresting Hook
- Primer
- Auxiliary Fuel Pump
- Lights
- Fuel Quantity Gage
- Cockpit Heater (early model airplanes)
- Pitot Tube Heater
- Gun Selector & Master Switches
- Gun Trigger Switch
- Gun Camera
- Gunsight
- Gun Heating
- Droppable Fuel Tank Releases
- Fusing & Selecting of Bombs
- Bomb Release
- Rocket Distributor
- Generator Warning Light (F6F-5N)
- Oil Dilution

1-73. DISTRIBUTION PANEL AND SWITCH BOX. The electrical system of the airplane is controlled mainly by switches, rheostats and circuit breakers on the electrical distribution panel and switch box, located on the right hand cockpit shelf. Operation of the controls on this panel is directed by the instructions printed on the adjacent nameplates.

SWITCHES

Section Light (2)	Primer
Wing Running Lights	Battery
Tail Running Light	Gun Selectors
Formation Lights (2)	Gunsight
Master Exterior Lights	Gun Camera
Arresting Hook	Gun Master
Starter	Bomb Selector
Pitot Tube Heat	Bomb Fusing
Radio Master	

RHEOSTATS—LIGHTS

Chartboard	Cockpit
Instrument Panel	Gunsight

CIRCUIT BREAKERS (MANUALLY RESET)

Radio IFF	Gun Camera Solenoid
Gunsight	Gun Camera Heater
Exterior Lights	Recognition Lights
Instruments	Gun Heaters (2)
Cockpit Lights	Panel Receptacle
Bomb Fusing	Compass—Remote
Droppable Tank	Arresting Hook

KEY TO FIGURE 1-16
HYDRAULIC SYSTEM

1. Wing Flap Cylinder
2. Restrictor
3. Gun Charging Cylinder
4. Wing Lock Cylinder
5. Wing Folding Timer Check Valve
6. Landing Gear Cylinder
7. Shuttle Valve
8. Reservoir
9. Accumulator
10. Filter (Engine-Pump Line)
11. Unloader Valve
12. Engine-Driven Pump
13. Intercooler Flap Cylinder
14. Cowl Flap Cylinder
15. Restrictor
16. Gun Charging Valve
17. Landing Gear Dump Valve
18. Landing Gear Vent Valve
19. Landing Gear Selector Valve
20. Cowl Flap Selector Valve
21. Oil Cooler Selector Valve
22. Wing Flap Selector Valve
23. Air Bottle
24. Check Relief Manifold
25. Wing Lock Selector Valve
26. Air Pressure Gage
27. Air Bottle Filling Valve
28. System Pressure Gage
29. Hand Pump Selector Valve
30. Hand Pump
31. Oil Cooler Flap Cylinder
32. Tail Wheel Cylinder
33. Relief Valve
34. Intercooler Selector Valve
35. Filter (Hand Pump Line)
36. Pressure Snubber

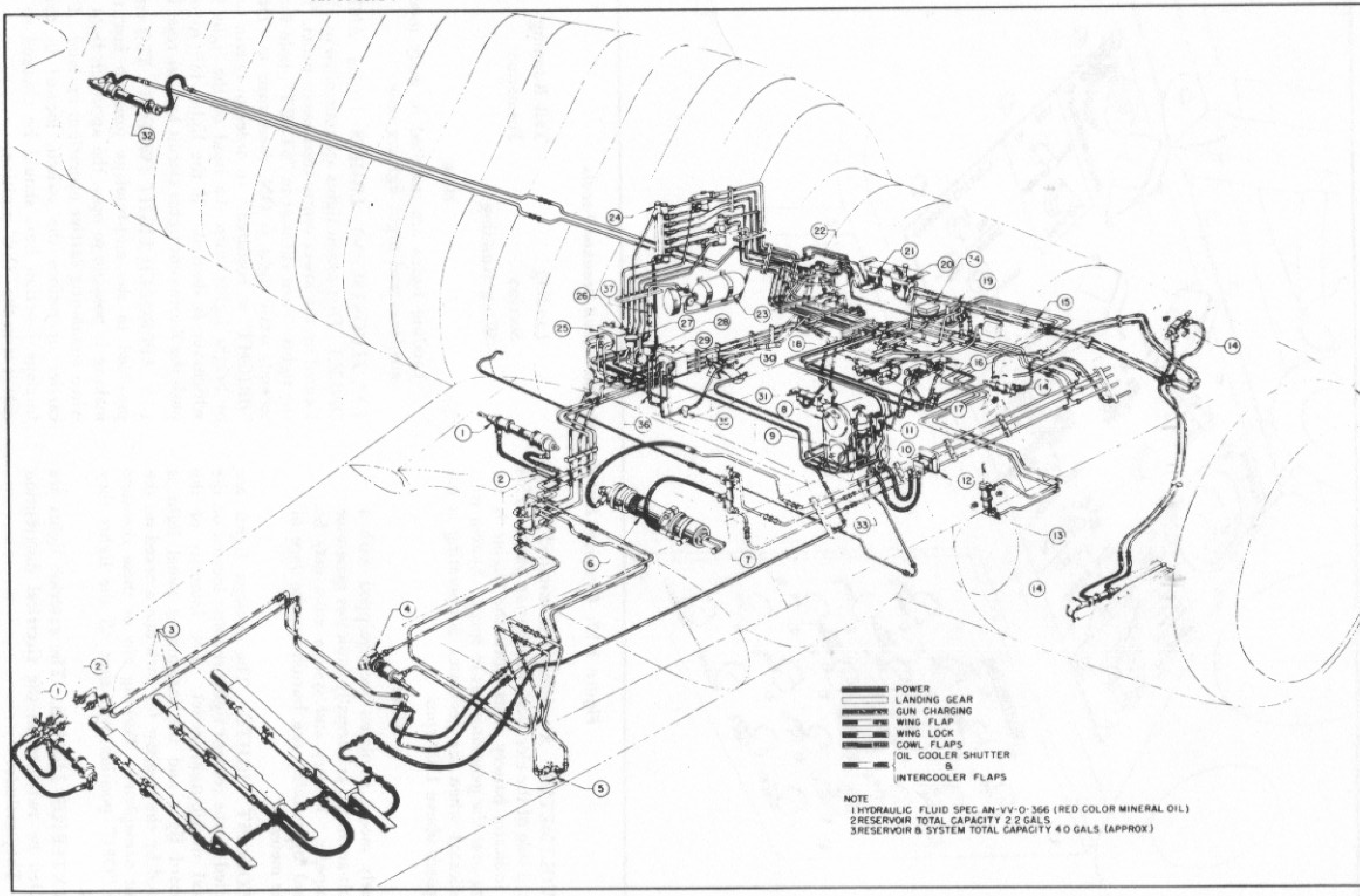


Figure 1-16. Hydraulic System Diagram

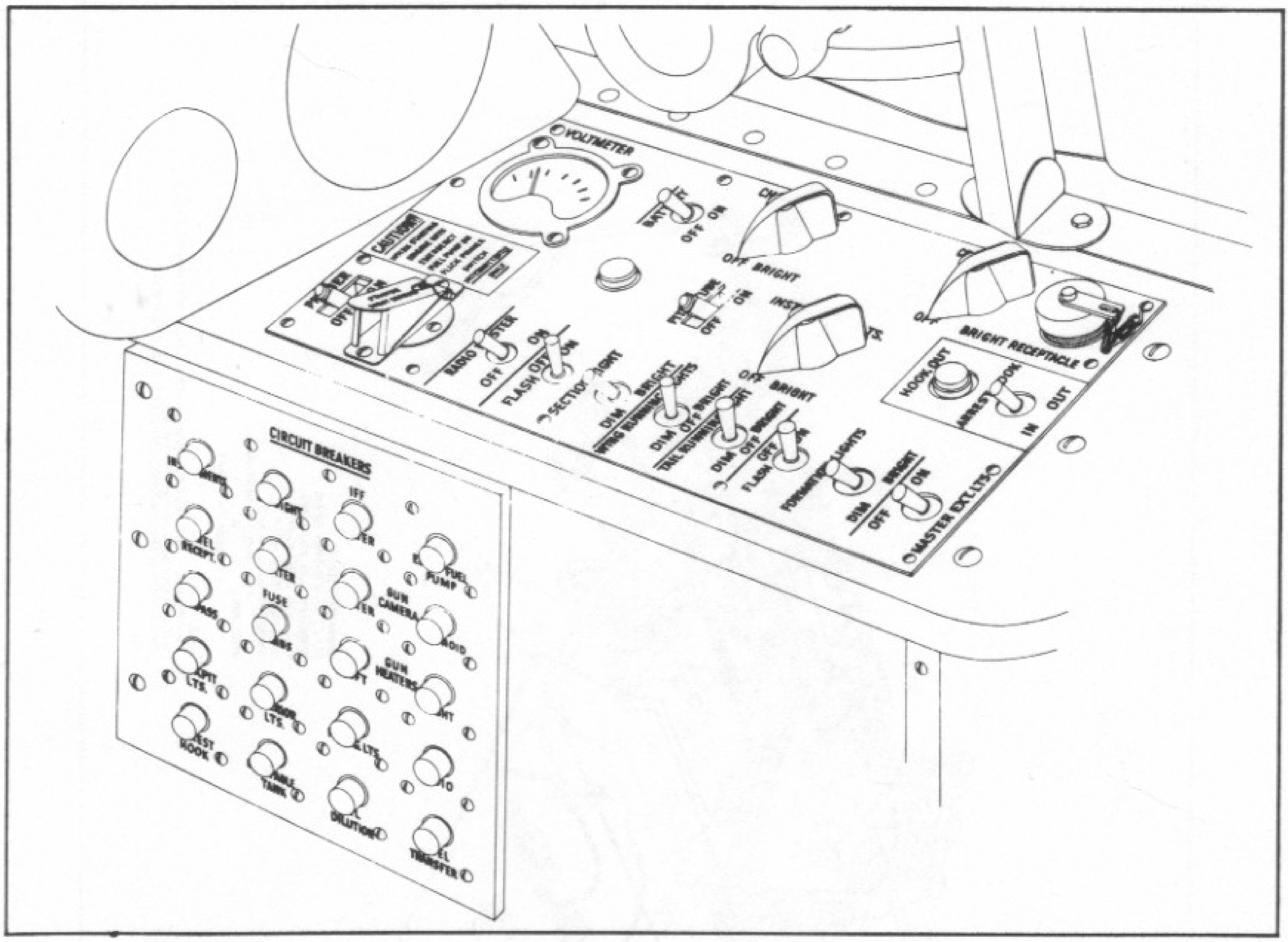


Figure 1-17. Electrical Distribution and Circuit Breaker Panels

1-74. VOLTMETER. A voltmeter is installed on the outboard side of the electrical distribution panel. This meter indicates battery volts when the engine is not running over the generator cut-in speed. System volts are indicated when the generator is operating at engine speeds above 1300 rpm.

Note

Early model airplanes are equipped with a volt-ammeter. It normally indicates generator amperes. Battery and system volts may be read by pushing the button on the face of the meter.

1-75. COCKPIT LIGHTS. The cockpit lights are controlled by the cockpit lights rheostat located on the electrical distribution panel. The intensity of the chartboard light and the instrument panel light is regulated by the rheostat for each one marked on the adjacent nameplates. Turning any of these rheostats to the "OFF" position will shut off the lights they control.

1-76. EXTERIOR LIGHTS. The exterior lights are controlled by switches on the electrical distribution panel. They include:

Landing	Tail Running
Section	Formation
Wing Running	

Note

Landing lights are installed in early model airplanes and night fighters only.

1-77. FLUORESCENT LIGHTS (F6F-3 AND -3N ONLY.) Fluorescent lights are controlled by rheostats located on the lower center instrument panel. To start the lights: Turn rheostat to "START", hold for a few seconds until light is ON, then turn to "DIM" or "BRIGHT" as required. In order to obtain ordinary or bright lights, turn the head of the light 90°, to whichever is desired. If the lights fail to operate, push the fluorescent lights circuit breaker reset button.

1-78. APPROACH LIGHT SWITCH. This switch is provided in the mid-fuselage junction box thereby making it possible to open the approach light circuit when simulating carrier operations on land. Prior to carrier operations the switch, located in the mid-fuselage junction box, should be checked to insure that it is in the "ON" position.

1-79. SPARE LIGHT BULBS. A spare bulbs container is provided in the fuselage at Station #127. A spare bulb for the gun sight is held in place by a clip mounted on gun sight mount bracket. On the F6F-5 and -5N airplanes, spare instrument panel bulbs are located on the left hand side of the instrument panel.

1-80. RECOGNITION LIGHTS SWITCHES. The recognition lights and keying switches are located on the right hand shelf just aft of the distribution panel. A switch is provided to select each light as desired. Throwing any switch to "STEADY" position will not cause the light to glow until the recognition lights keying switch is operated. If any light fails to operate, push the circuit breaker reset button, for this circuit.

1-81. PITOT TUBE HEATER. The pitot tube heater switch is located on the distribution panel. This switch should be turned to "ON" when icing conditions are encountered. If apparently incorrect airspeed is indicated during icing conditions, check the position of the switch to be sure it is in the "ON" position.

1-82. AUXILIARY CONTROLS.

1-83. LANDING GEAR CONTROL. The main and tail wheels are retracted and extended by double-acting hydraulic cylinders. The operating pressure is normally supplied by the engine-driven hydraulic pump or by the hand pump. The cylinders are controlled by the two position control lever located at the base of the left hand instrument panel. This square knob control lever is distinct in appearance, so designed to prevent inadvertant operation of the landing gear control. (For auxiliary and emergency operation of landing gear, refer to Section IV.)

a. Move lever UP for wheels "UP".

b. Move lever DOWN for wheels "DOWN".

CAUTION

Make certain that landing gear control lever is in "DOWN" position before take-off or landing.

1-84. LANDING GEAR LOCK.

1-85. The mechanical interconnector, between the landing gear square knob control lever and the nutcracker arm on the left hand shock strut, prevents landing gear retraction on the ground. The control lever cannot be moved to the "UP" position unless the left oleo is fully extended which occurs in flight only. (Operation is completely automatic.)

1-86. On the ground, a mechanical lock prevents the drag strut knuckle from breaking under any loading condition. In flight, as the wheels retract, this lock is released during the initial motion of the hydraulic cylinders. The position of this lock is indicated electrically by a micro-switch operated by the lock itself and connected to the position indicator in the cockpit.

1-87. LANDING GEAR POSITION INDICATOR. The position of the main and tail wheels is shown on the combination flap and landing gear indicator, located on the left hand instrument panel. This indicator, in addition to showing the position of each wheel, also shows whether or not they are locked up or down.

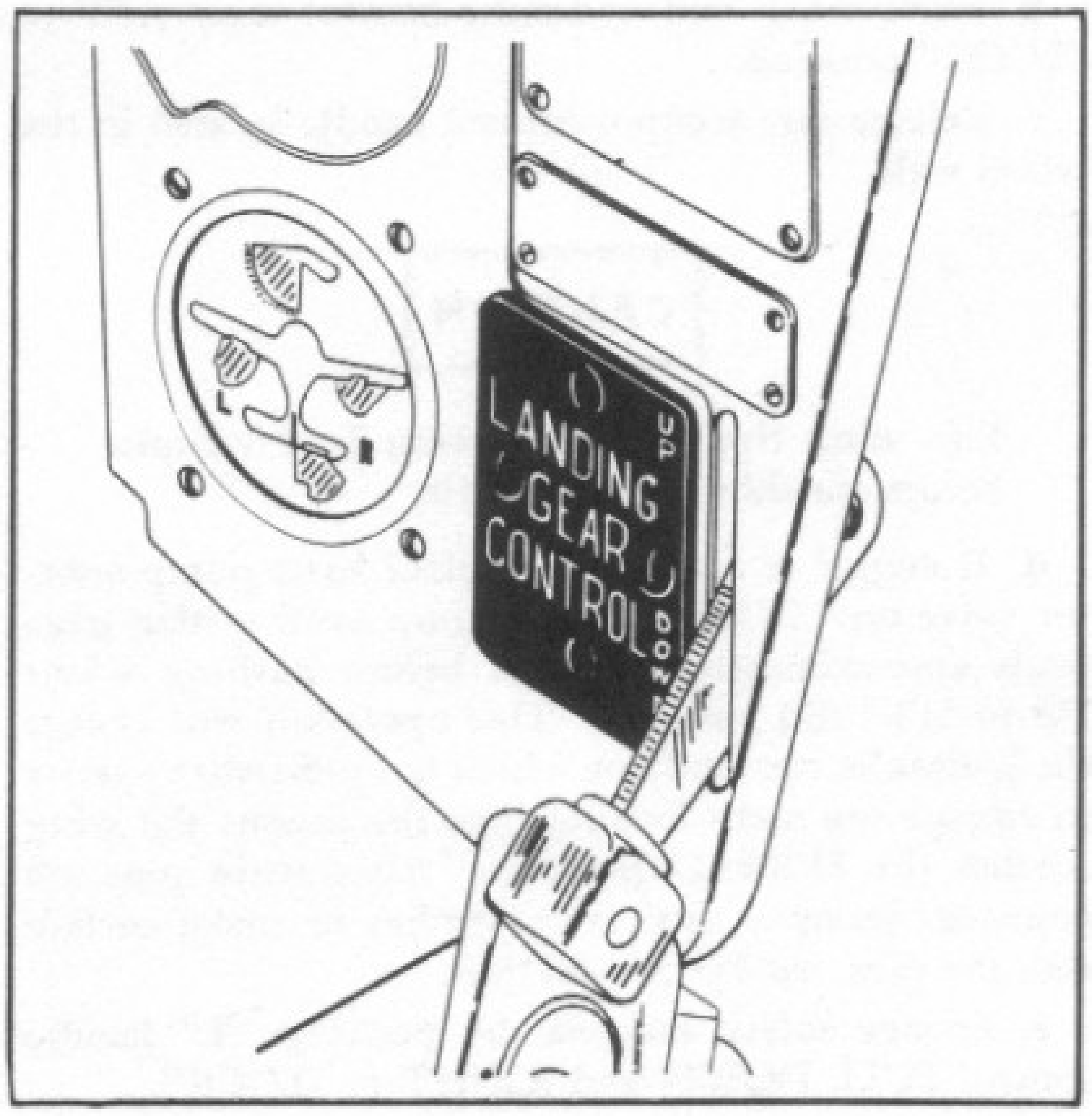


Figure 1-18. Landing Gear Control and Position Indicator

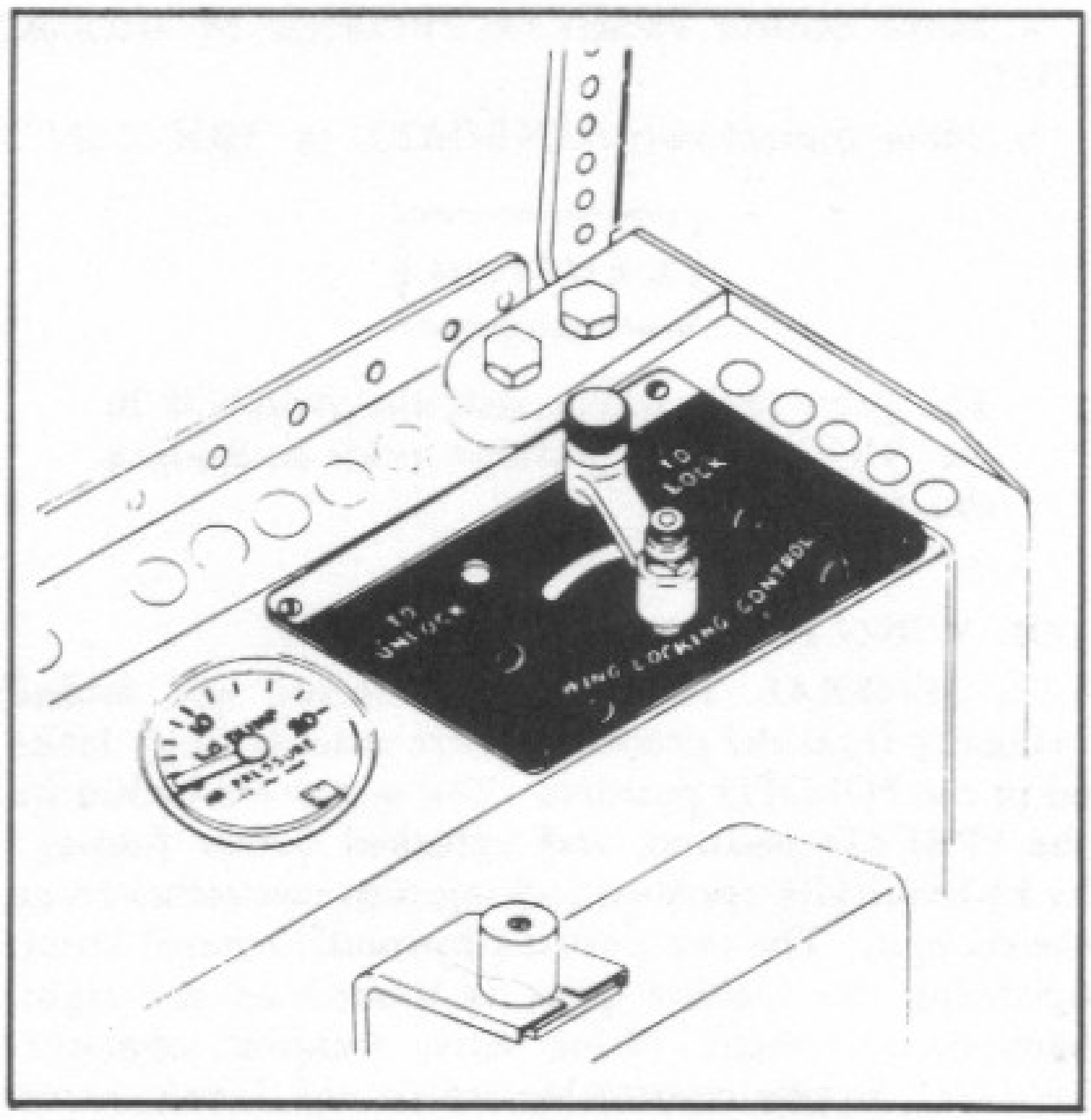


Figure 1-19. Wing Lock Control

Figure 1-20. Automatic Pilot ON-OFF Control

1-88. ARRESTING HOOK CONTROL.

1-89. The arresting hook is extended and retracted electrically. The control switch and circuit breaker are located on the electrical distribution panel on the right hand side of the cockpit.

1-90. When the hook is FULL OUT, the small light adjacent to the switch will glow. The running OUT of the hook will also turn on the approach light in the left wing leading edge. (For emergency operation of the arresting hook, refer to Section IV.)

a. Move control switch OUTBOARD to "HOOK OUT".

b. Move control switch INBOARD to "HOOK IN".

CAUTION

The pilot shall insure that the switch is in the "HOOK OUT" position prior to landing aboard a carrier.

1-91. WING FOLDING.

1-92. GENERAL. The wings are spread and folded manually from the ground and are automatically locked in the FOLDED position. The wings are locked in the SPREAD position, and unlocked before folding, by hydraulically operated locking pins controlled from the cockpit. The two position hydraulic control lever, operating the locking pins, is located on the right hand cockpit shelf. Wing safety lockpins, operated by a "T" handle control located on the lower center control panel, are used to safety the main locking pins after the outer panels are moved to the SPREAD position and the main locking pins are FULLY HOME. These safety lockpins, when engaged, prevent the main locking pins from disengaging, regardless of hydraulic pressure.

1-93. As the safety lockpins are withdrawn during the folding operation, red warning cylinders appear through the upper surface on each side of the wing center section. Before take-off, check the cylinders which will be flush with the wing surface if the wings are spread and locked properly.

1-94. TO FOLD WINGS.

a. Place wing flaps in "UP" position.

b. Disengage safety lockpins by pushing "T" handle control LEFT and FULL UP to "UNLOCK".

c. Move wing fold hydraulic control lever FORWARD to "UNLOCK" position. (If engine is not running, operate hydraulic hand pump, with hand pump selector valve on "SYSTEM" or "WING LOCK".

d. Push wing outer panel aft until folded lock engages automatically.

WARNING

The clearance between part of the wing and cockpit enclosure during folding is small; therefore, do not fold wings with anyone standing on walkway, or with arms or any part of the body projecting outside of the cockpit.

1-95. TO SPREAD WINGS

a. Place wing flaps in "UP" position.

b. Move wing fold hydraulic control lever AFT to "LOCK" position.

c. Release jury lockpin control handle located in the wheel well.

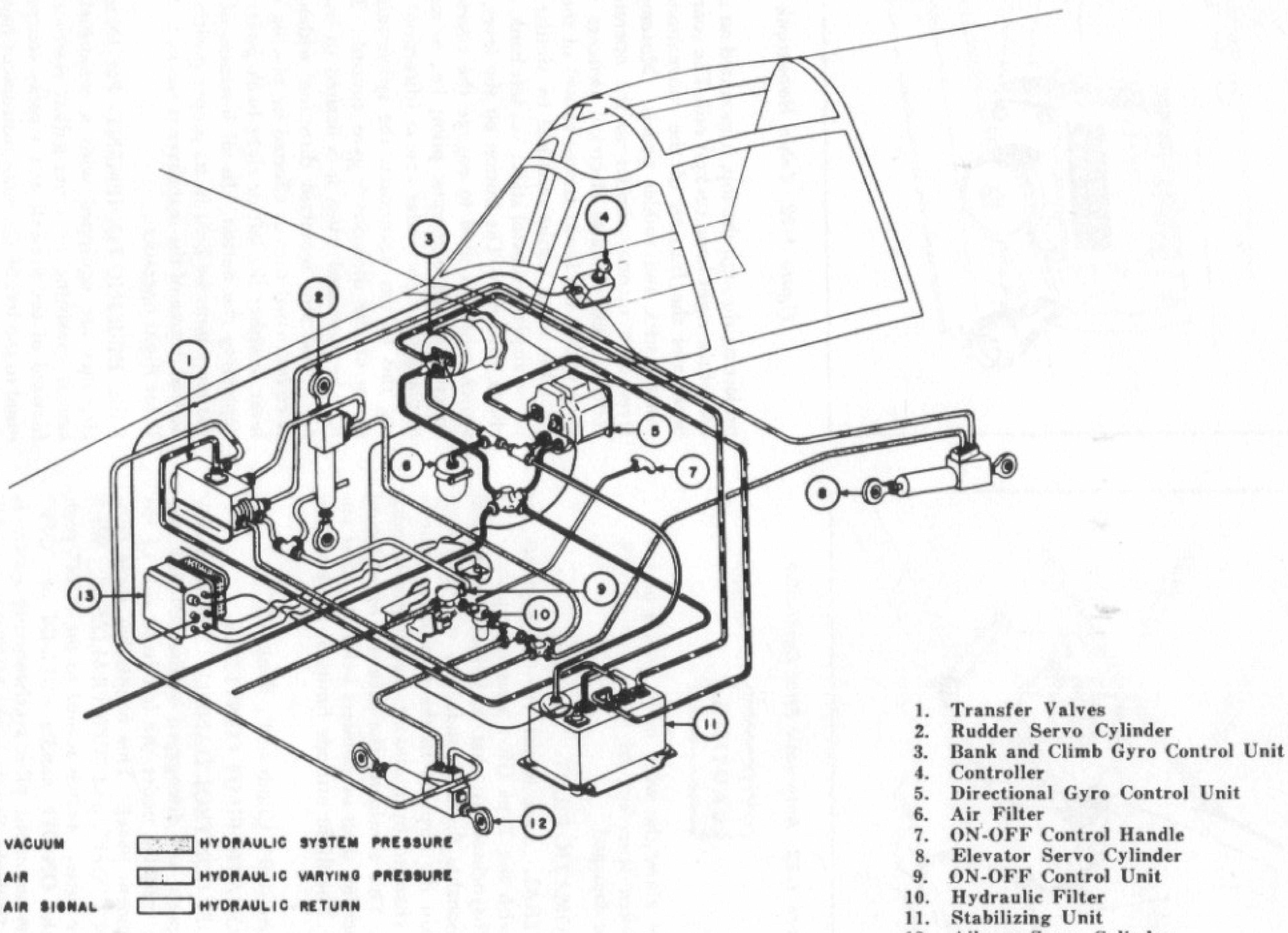
CAUTION

Lift wing tips when releasing jury lockpin before spreading outer panels.

d. If engine is not running, place hand pump selector valve on "SYSTEM" and pump until system gage reads approximately 1500 psi, before pushing wings UP to SPREAD position. This operation will charge the hydraulic accumulator which has sufficient capacity to engage the main locking pins the instant the wing reaches the SPREAD position. After main pins are engaged, pump a few extra strokes to make certain that the pins are FULLY HOME.

e. Engage safety lockpins by pushing "T" handle control FULL DOWN and RIGHT to "LOCK".

f. Check red warning flags on upper surface of wing center section to make certain that they are flush with upper surface of wing.



1. Transfer Valves
2. Rudder Servo Cylinder
3. Bank and Climb Gyro Control Unit
4. Controller
5. Directional Gyro Control Unit
6. Air Filter
7. ON-OFF Control Handle
8. Elevator Servo Cylinder
9. ON-OFF Control Unit
10. Hydraulic Filter
11. Stabilizing Unit
12. Aileron Servo Cylinder
13. Relay Box

Figure 1-21. Automatic Pilot Installation Diagram (Late Model F6F-5N Airplanes)

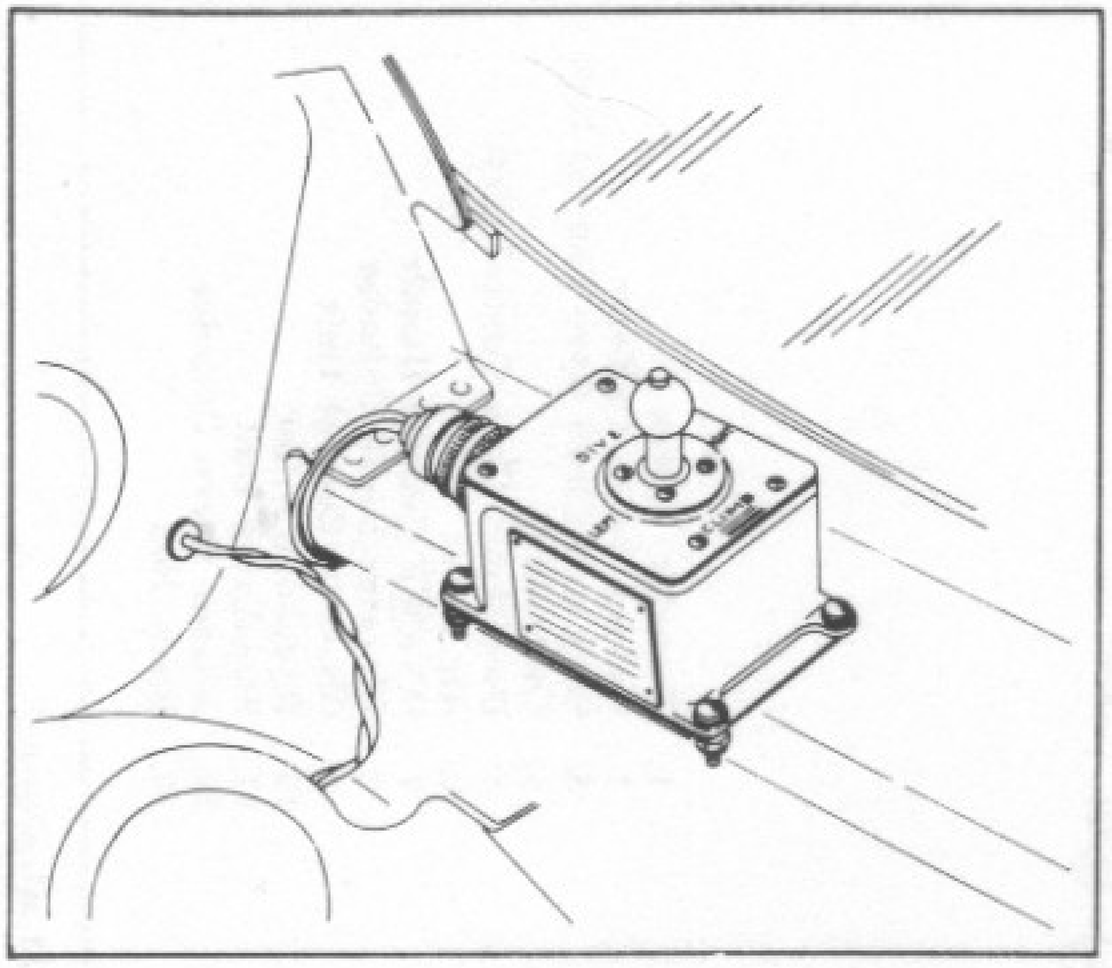


Figure 1-22. Automatic Pilot Controller

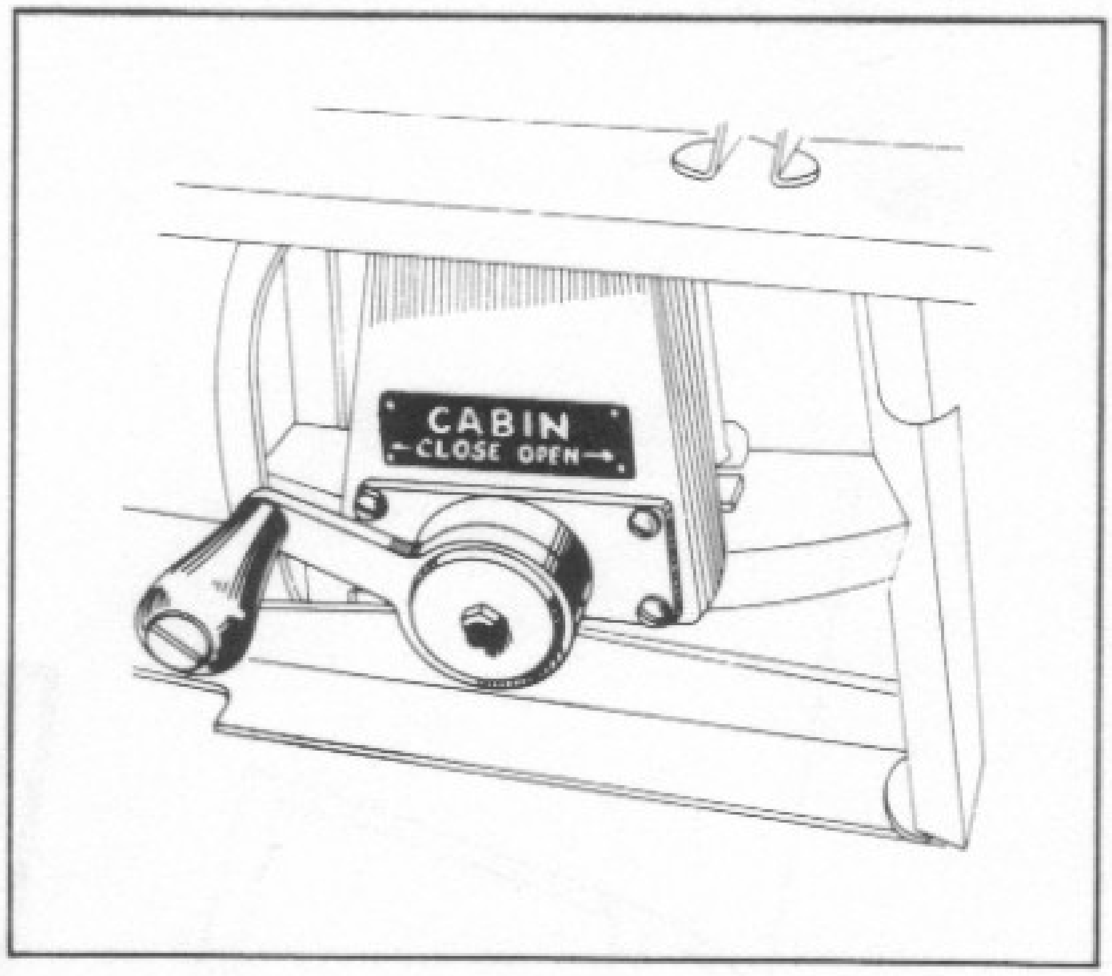


Figure 1-23. Cabin Handcrank

CAUTION

Do not allow the wing outer panels to fall free when spreading as wing folding axis may be damaged.

1-96. AUTOMATIC PILOT.

1-97. GENERAL. Late model F6F-5N airplanes are equipped with the Type GR-1 automatic pilot. The pilot is self-synchronous; that is, it may be engaged in any reasonable flight attitude and will take over and maintain the airplane in the prevailing attitude until it is changed by means of the automatic pilot controller. This eliminates the necessity of adjusting knobs, or setting dials or pointers before or after engagement. The flight attitude limits of the pilot are as follows:

Dive—50°, Climb—30°, Bank—45°

1-98. AUTOMATIC PILOT CONTROLS.

1-99. ON-OFF CONTROL HANDLE. The automatic pilot is engaged and disengaged by the ON-OFF control handle installed under the left hand side of the main instrument panel. This handle has three positions: "ON", "OFF", and "CENTRALIZED". With the airplane battery switch turned to the "ON" position and the ON-OFF handle FULL-IN or "OFF" position, the automatic pilot synchronizing system is operative. With the handle in the MIDDLE or "ON" position, synchronization ceases for the bank and climb gyro units and the automatic pilot takes control and maintains the prevailing flight attitude. When the ON-OFF handle is pulled out and held FULL-OUT in the "CENTRALIZED" position, the airplane returns to straight and level flight.

1-100. CONTROLLER. The automatic pilot controller, consisting of a small case with a controller lever projecting through the top, is mounted on the forward end of the right hand cockpit rail. The controller lever duplicates the features of the conventional airplane control stick and rudder pedals. Movement of this lever in a manner similar to the operation of the control stick, operates electrical switches which produce within the limits (noted above) of the automatic pilot the same results obtained by similar movement of the airplane control stick; i.e., left bank, right bank, climb, dive etc. The button on the lever, controls a switch which is used to engage the course holding function of the automatic pilot; i.e., to maintain the airplane on a particular course (direction). Depressing this button disconnects the synchronizing mechanism of the directional gyro control. The button may be depressed when it is desired to bank the airplane and still maintain direction without turning. A co-ordinated turn is effected by placing the control lever in either the left or right bank position without depressing the button. In all instances of operation, the lever must be held in its proper position until the desired degree of the maneuver is reached. See Section II for flight operation.

1-101. DE-ICING EQUIPMENT. F6F-3N and F6F-5N airplanes are equipped with a windshield de-icing system consisting of a one-gallon reservoir mounted forward of the firewall and a spray control knob located to the left of the main instrument panel.

1-102. MISCELLANEOUS CONTROLS AND EQUIPMENT.

1-103. COCKPIT ENCLOSURE. The cockpit enclosure consists of three pieces of Plexiglas attached to an aluminum alloy frame which rides by means of four rollers on the sections of track surrounding the cockpit. The self-locking handcrank mechanism, located

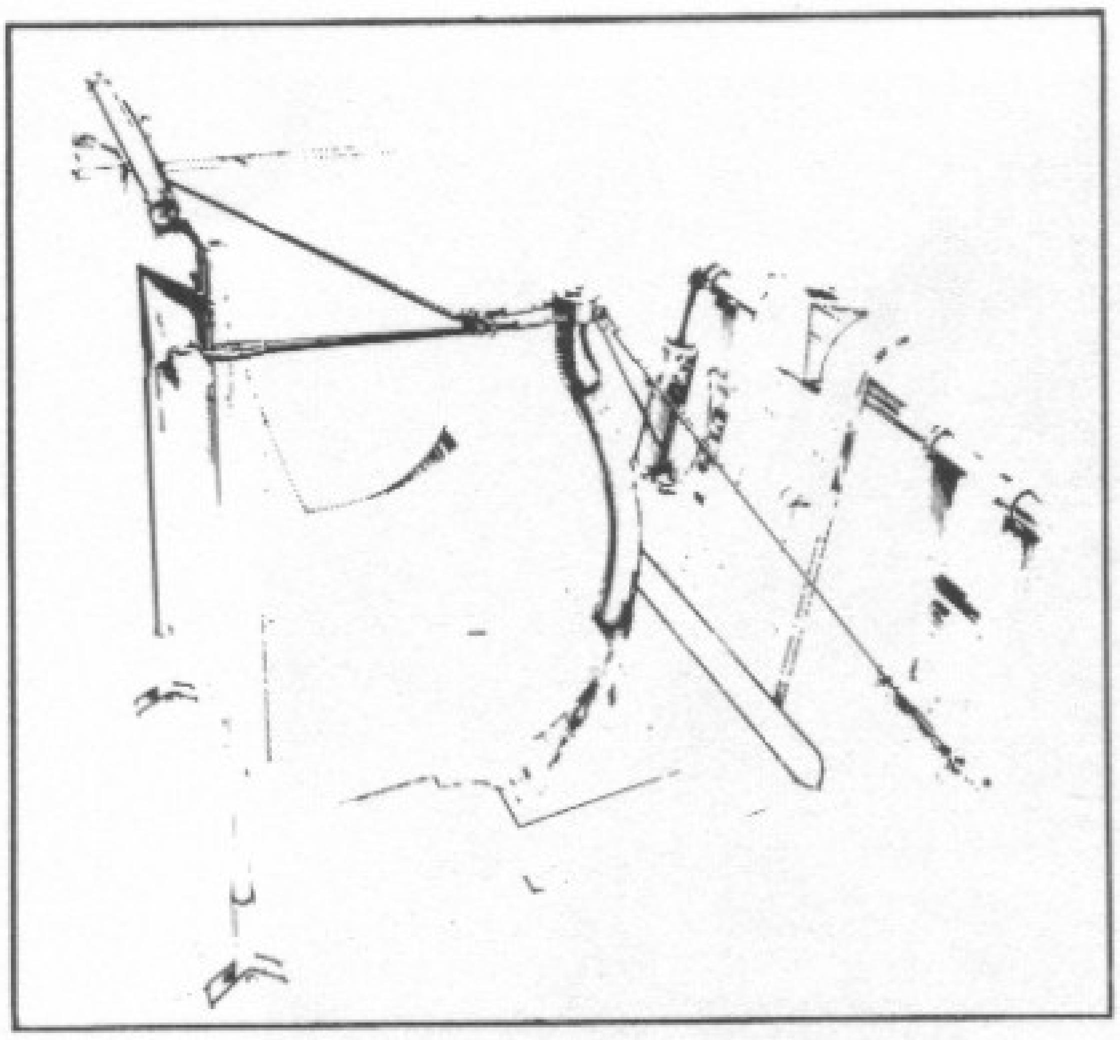


Figure 1-24. Surface Controls Lock

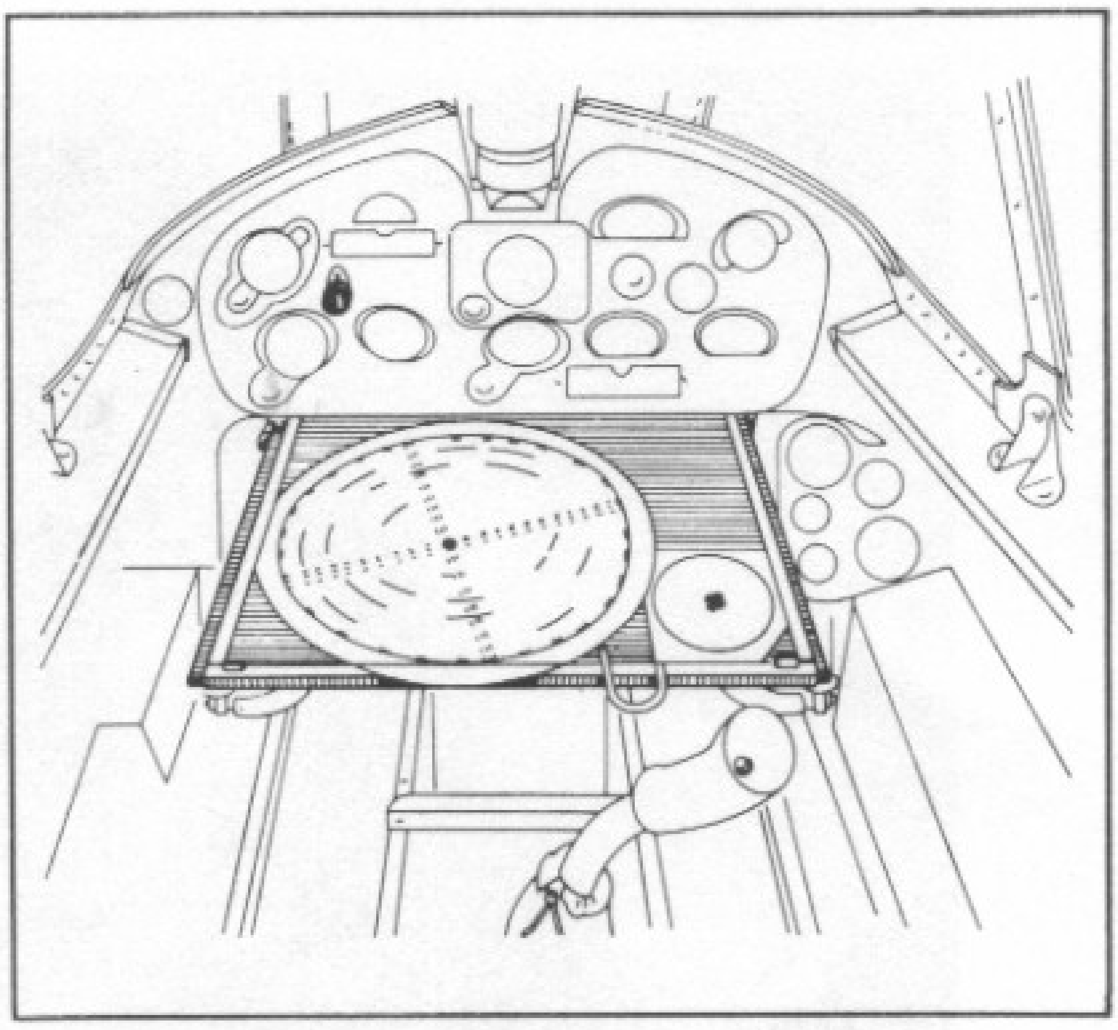


Figure 1-25. Chartboard

above the right hand cockpit shelf, propels the hood fore and aft by means of a chain and flexible cable assembly. A hold-back block installed on the right hand cockpit rail, can be swung across the track to prevent the hood from slamming closed when the airplane is brought to a sudden stop. The hood may be opened or closed from outside the airplane by pushing the button, located on the right hand cockpit exterior, and manually sliding the hood to the desired position. When the button is released the hood is automatically locked. (For emergency jettisoning of cockpit enclosure, refer to Section IV.) For normal operation, proceed as follows:

a. Rotate handwheel CLOCKWISE to "OPEN".

b. Rotate handwheel COUNTERCLOCKWISE to "CLOSE".

1-104. COCKPIT VENTILATOR. A fresh air valve, located at the base of the lower center control panel, regulates the flow of air taken from the wing leading edge ducts to the cockpit. The pilot can open or close the valve by pushing his foot against the knob located on the right hand side of the valve.

a. Apply foot to UPPER side of knob to "CLOSE".

b. Apply foot to LOWER side of knob to "OPEN".

1-105. COCKPIT HEATER. The cockpit heating and ventilating system consists of a hot air combustion type heater. An electric blower unit forces hot air up through the windshield duct and back to the rear of the cockpit. A switch, which controls the blower and gas ignitor simultaneously, is located on the electrical distribution panel. (On the F6F-3 airplane, the switch is located on the lower center control panel.) A heat control lever, located on the lower center control panel, directs the flow of warm air.

a. Move lever FULL-UP for windshield "DEFROSTER".

b. Move lever to CENTER for "DEFROSTER AND FEET".

c. Move lever FULL-DOWN for "FEET ONLY".

Note

The cockpit heating and windshield defroster system was eliminated on later model airplanes.

1-106. SURFACE CONTROLS LOCK. The control stick and rudder pedals are secured by the lashing device provided with each airplane. The device consists of a metal cap which slides over the top of the pistol grip and four cables, two of which connect to the base of the rudder pedals and two connect aft to the hooks provided at the rear of the cockpit rail.

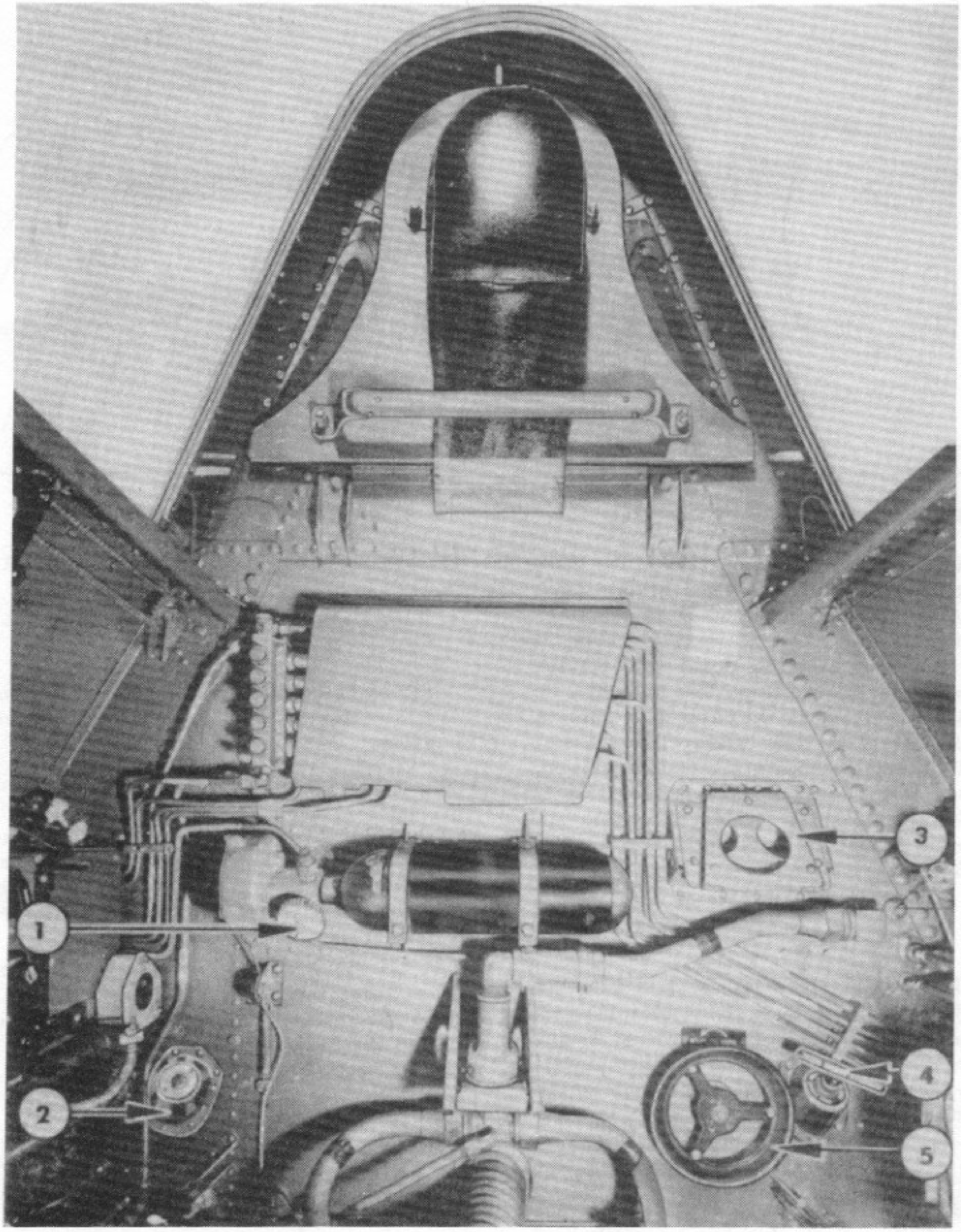
1-107. CHARTBOARD. The pilot's chartboard is stowed under the main instrument panel. Depress latch located on lower left corner of instrument panel and pull chartboard AFT.

1-108. MAP CASE. A map case with pad and pencil holder is located on the cockpit floor to the left of the pilot's seat.

1-109. RELIEF TUBE. A relief tube is stowed in a clip under the pilot's seat.

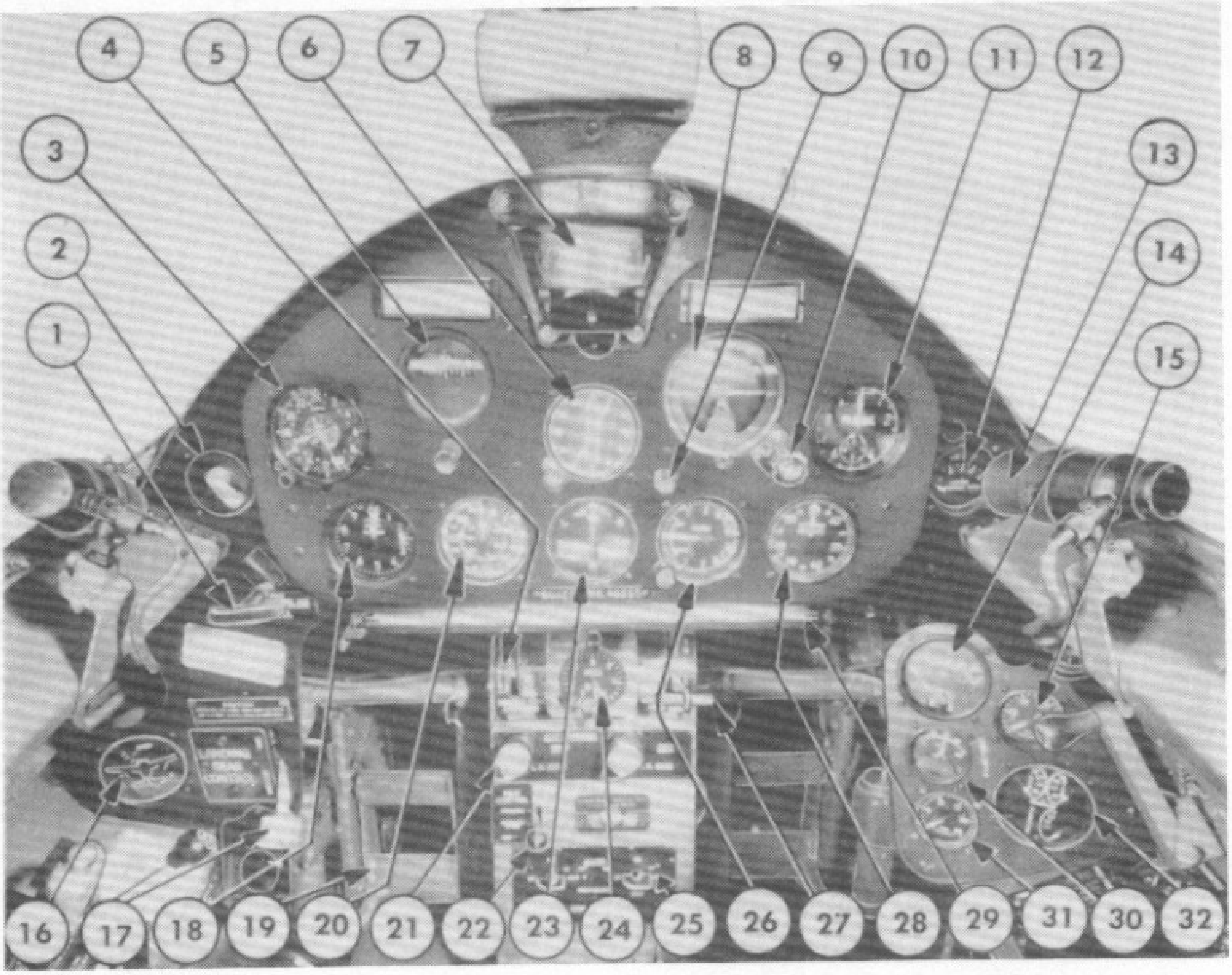
1-110. PILOT'S SEAT AND HARNESS. The pilot's seat can be moved up or down (adjustment six in.) by the control lever located on right hand side of the seat. Pull lever UP to change adjustment either way. A shoulder harness adjustment lever is located on the left hand side of the pilot's seat. Push lever AFT to change adjustment.

1-111. ANTI-BLACKOUT EQUIPMENT. Provisions are made for the attachment of anti-blackout equipment. A quick-disconnect assembly, for attachment of the anti-blackout suit, is installed on the cockpit floor on the left hand side of the pilot's seat.



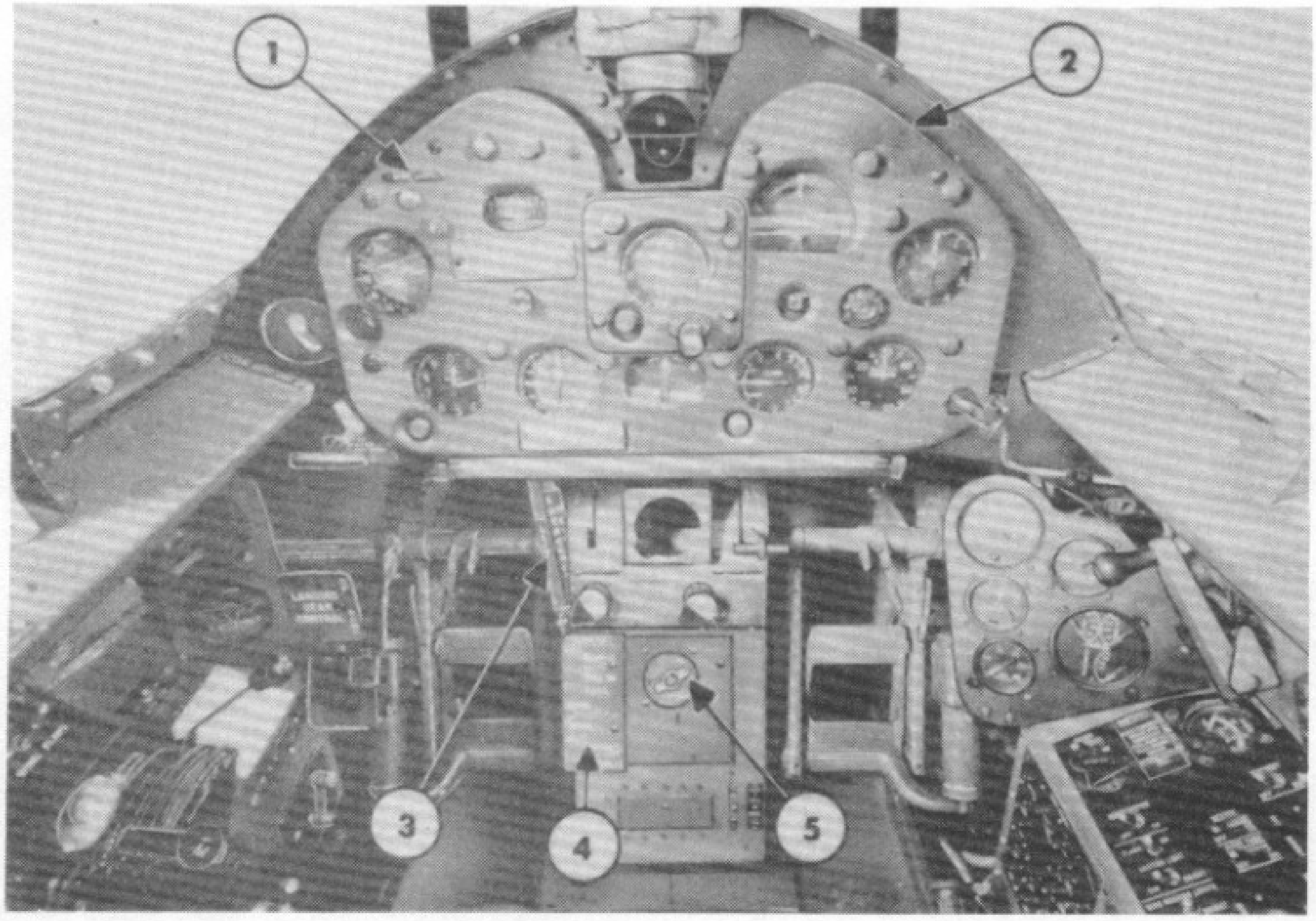
1. Landing Gear Emergency Dump Test Valve
2. Tow Target Release Control
3. Oxygen Regulator Support
4. Arresting Hook Emergency Control
5. Oxygen Cylinder Control Handle

Figure 1-26. Cockpit—Rear View



1. Carburetor Protected Air Control (Aux. Stage Only)
2. Ignition Switch
3. Clock
4. Landing Gear Emergency Lowering Control
5. Directional Gyro
6. Compass
7. Gunsight
8. Attitude Gyro
9. Chartboard Light
10. Attitude Gyro Caging Knob
11. Tachometer
12. Water Quantity Gage—A.D.I. System
13. Instrument Panel Fluorescent Light
14. Cylinder Head Temperature Gage
15. Oil Pressure Gage
16. Landing Gear & Wing Flap Position Indicator
17. Landing Gear Control
18. Altimeter
19. Rudder Pedals
20. Airspeed Indicator
21. Gun Charging Controls
22. Cockpit Heater Control
23. Turn and Bank Indicator
24. Ammunition Rounds Counter
25. Fluorescent Lights Control
26. Rate of Climb Indicator
27. Wing Lock Safety Control Handle
28. Manifold Pressure Gage
29. Chartboard
30. Oil-In Temperaure Gage
31. Fuel Pressure Gage
32. Fuel Quantity Gages

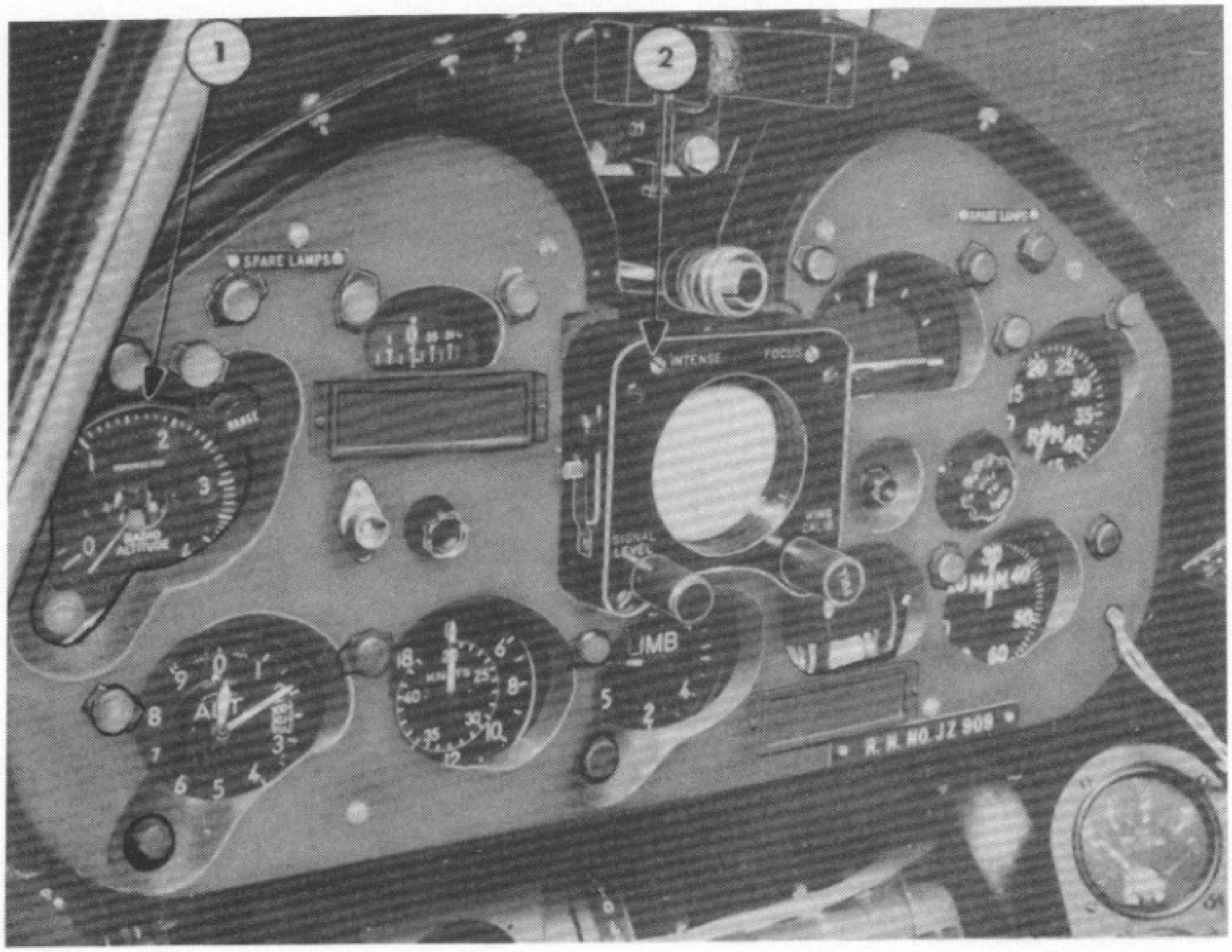
Figure 1-27. Cockpit—Forward View (F6F-3)



The F6F-5 differs from the F6F-3 in the following respects:

1. Spare Lamps Container
2. Reflector Panel
3. Wing Bomb Rack Manual Release
4. Check-Off Card
5. MK I Rocket Selector
6. Removal of Fluorescent Lights and Control
7. Removal of Cockpit Heat Control Switch to Main Electrical Distribution Panel

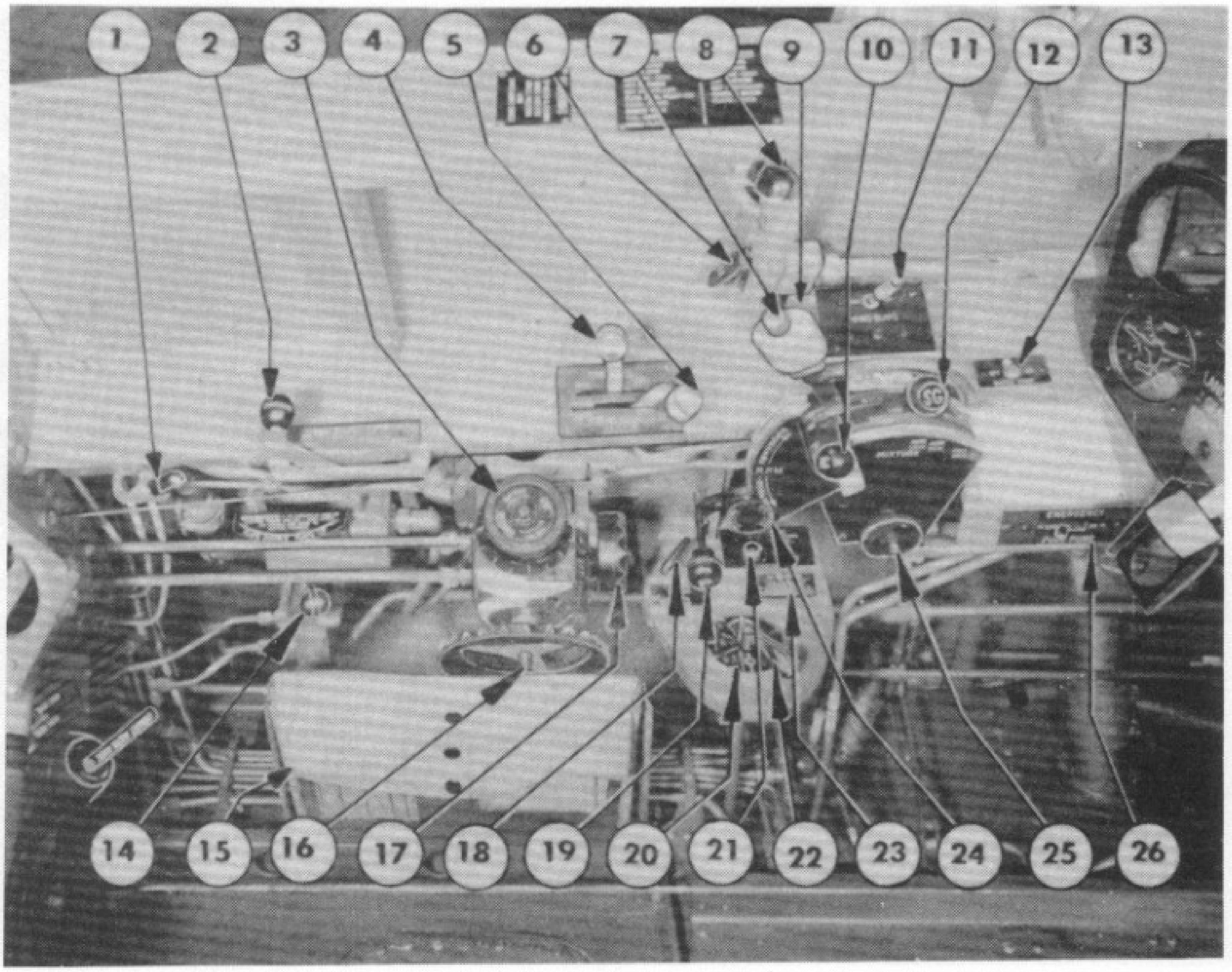
Figure 1-28. Cockpit—Forward View (F6F-5)



The F6F-5N differs from F6F-5 in the following respects:

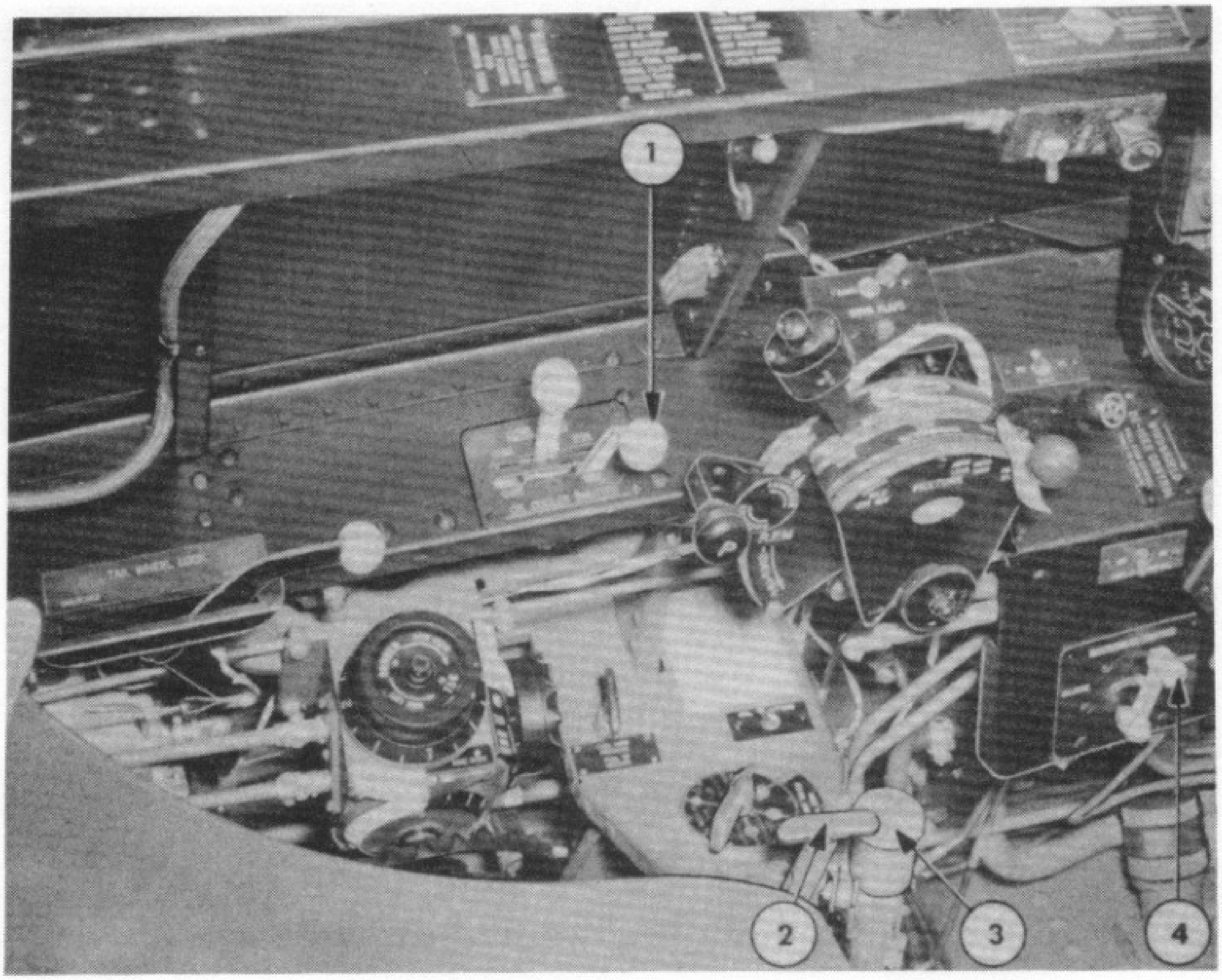
1. Radio Altimeter Indicator
2. Radar Scope

Figure 1-29. Cockpit—Forward View (F6F-5N)



1. Lower Left Cockpit Light
2. Tail Wheel Lock Control
3. Rudder Trim Tab Control
4. Cowl Flaps Control
5. Oil Cooler-Intercooler Shutters Control
6. Droppable Fuel Tank Release Switch
7. Mask Microphone Switch
8. Upper Left Cockpit Light
9. Throttle Control
10. Mixture Control
11. Wing Flap Electrical Switch
12. Supercharger Control
13. Water Injection Control Switch
14. Wing Flap Manual Control
15. Map Case
16. Elevator Trim Tab Control
17. Aileron Trim Tab Control
18. Fuel Tank Pressurizing Control
19. Propeller Pitch Control
20. Fuel Selector Valve Dialface
21. Reserve Fuel Tank Pressurizing Control
22. Fuel Tank Selector Valve Control
23. Oil Dilution Switch
24. Propeller Pitch Vernier Control
25. Engine Control Quadrant Friction Knob
26. Auxiliary Electric Fuel Pump Switch

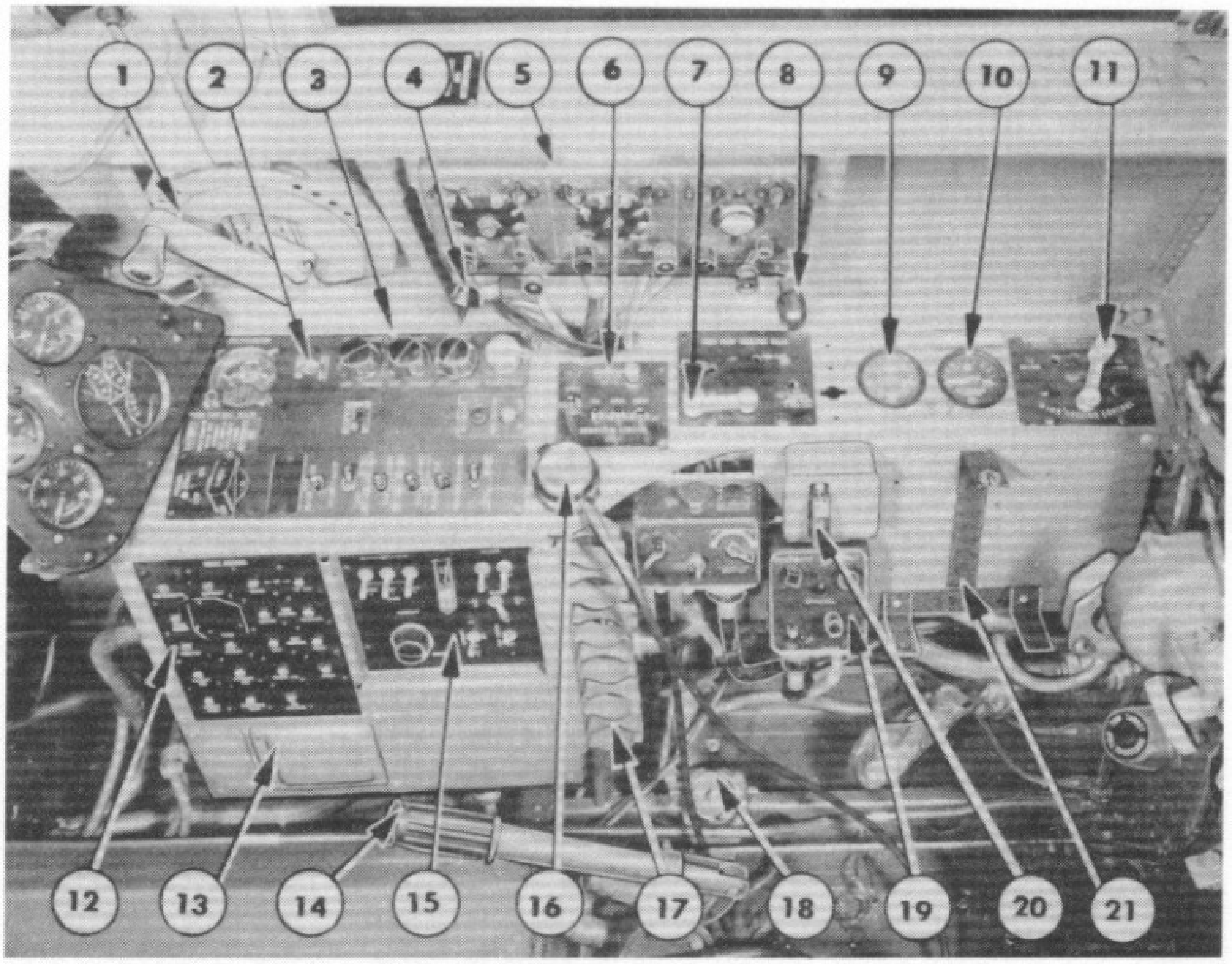
Figure 1-30. Cockpit—Left View (F6F-3)



The F6F-5 differs from the F6F-3 in the following respects:

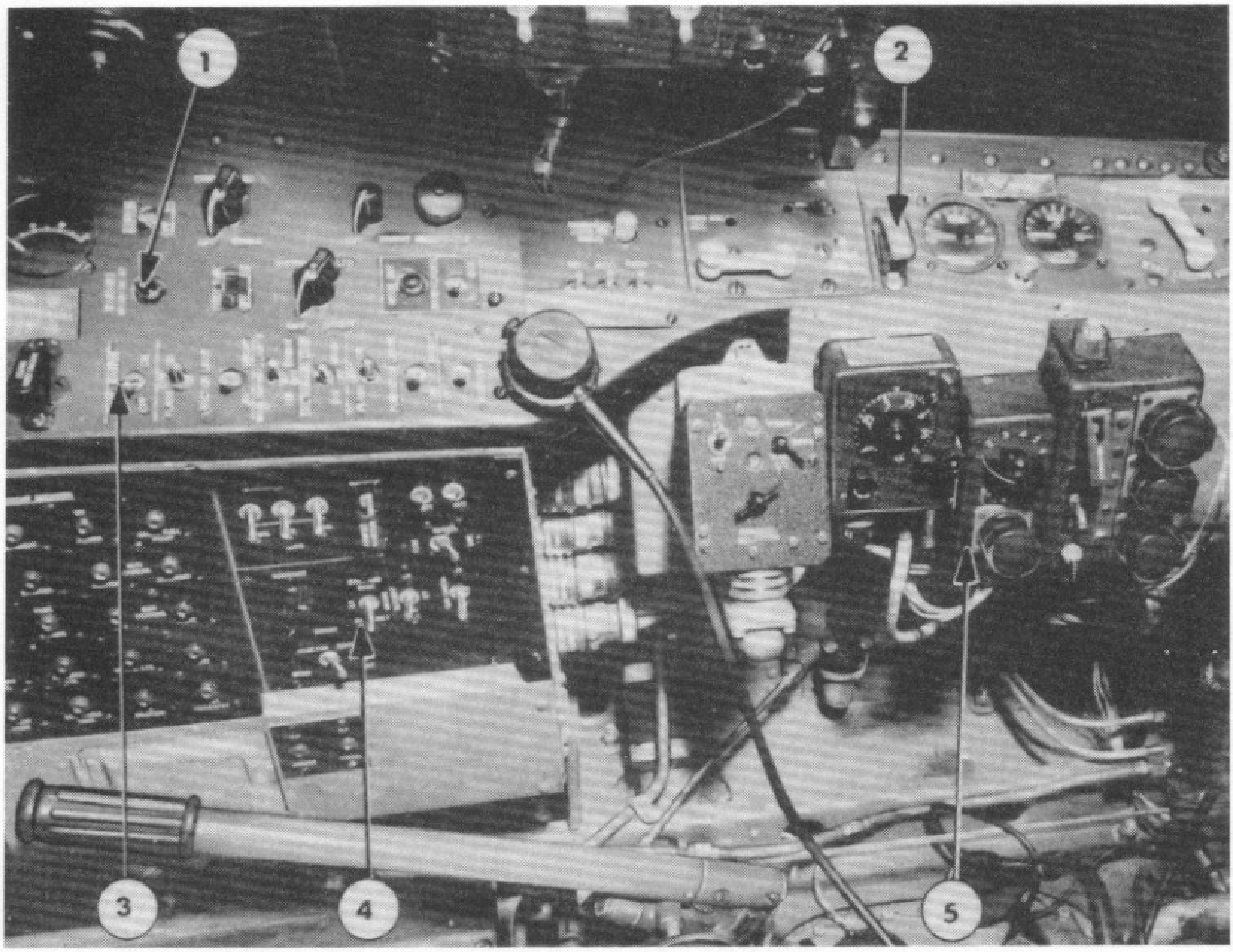
1. Oil Cooler Shutter Control
2. Fuselage Droppable Tank Manual Release Control
3. Anti-Blackout Regulator
4. Intercooler Shutter Control
5. Removal of the Fuel Level Warning Light

Figure 1-31. Cockpit—Left View (F6F-5)



1. Cabin Sliding Hood Control
2. Battery Switch
3. Main Electrical Distribution Panel
4. Electrical Panel Light
5. Radio Controls
6. Recognition Lights
7. Hand Pump Selector Valve
8. Aft Right Cockpit Shelf Light
9. Hydraulic System Pressure Gage
10. Landing Gear Emergency Dump Pressure Gage
11. Wing Locking Hydraulic Control
12. Manual Reset Circuit Breaker Panel
13. Access to Reverse Current Relay
14. Hydraulic Hand Pump
15. Armament Panel
16. Hand Microphone
17. Pyrotechnic Cartridge Clips
18. Pyrotechnic Pistol Retainer
19. Radio Controls
20. IFF Destruction Switch
21. IFF Equipment Support

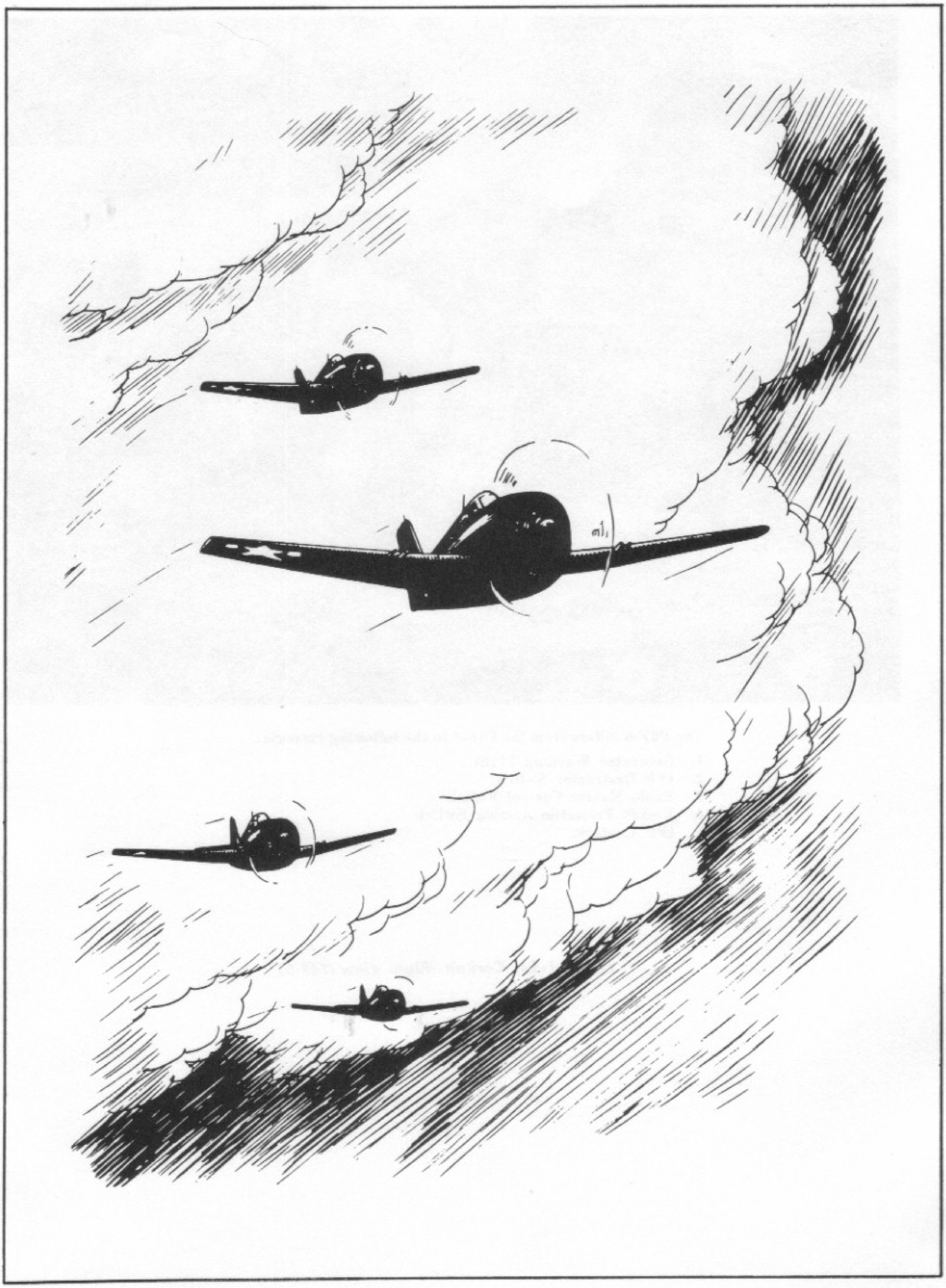
Figure 1-32. Cockpit—Right View (F6F-3)



The F6F-5 differs from the F6F-3 in the following respects:

1. Generator Warning Light
2. IFF Destructor Switch
3. Radio Master Control Switch
4. Rocket Projectile Arming Switch
5. IFF Controls

Figure 1-33. Cockpit—Right View (F6F-5)



SECTION II
NORMAL OPERATING INSTRUCTIONS

2–1. BEFORE ENTERING THE COCKPIT.

2-2. Note the following flight limitations and restrictions:

MANEUVERS

All standard fighter maneuvers are permitted with any internal loading arrangement except as follows:

Inverted flight—for entering dives or gunnery runs.

Inverted spins—not over one turn.

Normal spins—not over two turns.

When carrying a torpedo, bombs, smoke tank, or droppable fuel tank, the permissible maneuvers are the following:

Wing-over.

Vertical turn.

Inverted flight—for entering a dive.

Aileron roll—for entering a dive.

DIVING SPEED

F6F-3 and -3N

Do not exceed 415 knots IAS below 10,000 feet. Refer to paragraph 2-96, Diving, and figure A-5, for weights, speeds, altitudes and accelerations.

F6F-5 and -5N

With airplane Bureau No. 71098 and subsequent and any F6F-5 or -5N with Model F6 Service Change #75 incorporated do not exceed 440 knots IAS below 10,000 feet. If this change has not been incorporated, the airplane is restricted to limits .5g less or 15 knots less than the limits shown on figure A-6. Refer to paragraph 2-96, Diving, and figure A-6, for weights, speeds, altitudes and accelerations.

USE OF AILERONS

The maximum permissible speed for unlimited use of the ailerons is 260 knots IAS. Accelerations of +5.0g and −2.0g must not be exceeded. At higher airspeeds, use of the ailerons is restricted to the same control force required for full aileron operation at 260 knots IAS.

LANDING GEAR

Maximum speed—raising or lowering—135 knots IAS. If lowered at speeds above 135 knots IAS, the gear will trail.

Maximum speed—gear trailing or extended—300 knots IAS.

Emergency lowering—do not exceed 90 knots IAS.

WING FLAPS

If the electrical flap control system is not operating satisfactorily, reduce speed below 170 knots IAS before lowering flaps. With flaps deflected accelerations of +5.0g and −2.0g shall not be exceeded.

Flaps blow up automatically to the following angles regardless of the cockpit control setting:

50°— 90 knots IAS

15°—150 knots IAS

ENGINE SPEED

Maximum diving rpm—3060 for 30 seconds.

These limitations may be supplemented or superseded by instructions included in Service Publications.

2-3. Check gross weight and center of gravity location for take-off and for anticipated landing condition. Loading data is furnished in the Handbook of Weight and Balance Data, AN 01-1B-40.

2-4. ENTRANCE TO AIRPLANE. The cockpit is accessible from either side of the airplane—a step is located on the fuselage aft of the wing trailing edge, and a hand grip above and aft of the wing.

2-5. To open the cockpit, push and hold in the release button on the fuselage skin below the windshield (right hand side) then push the enclosure aft.

2-6. ON ENTERING THE PILOT'S COCKPIT.

2-7. STANDARD CHECK FOR ALL FLIGHTS.

a. Wheels chocked.

b. Control's locking device—OFF.

c. Landing gear control—"DOWN".

d. Ignition switch—"OFF".

e. Wings—SPREAD and LOCKED. Warning signal flags flush with wing skin.

f. Adjust seat height and rudder pedals. Check stick and pedals for freedom of movement and full throw—watch control surfaces.

g. Adjust shoulder harness and rear view mirror.

h. Mixture control—"IDLE CUT-OFF".

i. Battery switch—"ON"

j. Check fuel and oil supply.

k. If rockets are carried and a MK. 3 distributor is installed, remove rocket safety plug located on distributor box. (Refer to Section V.)

l. Check gun sight illumination and ammunition loading. (Refer to Section V.)

m. Gun charging controls—"SAFE". All armament switches—"OFF". (Refer to Section V.)

n. Check communication equipment. (Refer to Section V.)

o. Arresting hook—"IN". Red light OFF.

p. Fuel pressurizing handle—"PRESSURE" position.

q. Check hydraulic system pressure—1250-1500 psi.

r. Check landing gear dump bottle air pressure—1800-1950 psi.

s. Check enclosure emergency release pins for security.

t. Uncage directional gyro and artificial horizon.

u. Set altimeter to correct barometric pressure.

v. Check clock.

w. GR-1 automatic pilot (late model F6F-5N)—DISENGAGED.

x. Oil dilution switch—"OFF".

2-8. SPECIAL CHECK FOR NIGHT FLIGHTS.

a. Battery switch—"ON".

b. Test operation of landing light (night fighters).

c. Chartboard, instruments, and cockpit light rheostats—turn "ON" and adjust to best light.

d. Test operation of the following switches and lights:

Master exterior lights.
Formation lights.
Tail running lights
Wing running lights.
Section light.

e. Test operation of recognition lights. (10 seconds only.)

f. Lower arresting hook to test approach light. Arresting hook—"IN".

2-9. FUEL AND OIL SYSTEM MANAGEMENT.

2-10. OPERATION OF FUEL SYSTEM. The fuel system is managed primarily with the fuel tank selector and auxiliary (emergency) fuel pump switch.

2-11. After warm-up, take-off and a safe altitude has been reached, rotate the tank selector from the right main to wing droppable (left or right) or fuselage droppable depending on which tank or tanks are carried. If fuel pressure drop is indicated on the engine gage unit (fuel pressure dial), it will be accompanied by a drop in engine performance. More than likely the tank being used is empty or nearly so, providing the remainder of the system is functioning normally. Select another tank immediately and turn the auxiliary (emergency) fuel pump switch to "ON". Keep this pump "ON" until the fuel pressure becomes steady.

2-12. The carburetor bleed-back (vapor vent) line returns approximately two gallons per hour to the right main tank. Also, while the fuel transfer system is in operation, fuel is fed to this tank until it contains 81 gallons. Refer to Section I for further information.

2-13. OPERATION OF OIL SYSTEM. The operation of the oil system is automatic except for oil dilution and the setting of the oil cooler shutter. This shutter is controlled by a hydraulic lever on the left hand shelf. Refer to paragraph —.

2-14. During flight, oil temperature can be reduced more rapidly by reducing rpm than by throttling only. Oil temperature and pressure limits are as follows:

Scramble take-off—40°C (104°F) min.
Ground test—40°C (104°F) min.
Desired—60°-85°C (140°-185°F), 75-95 psi
Maximum—95°C (203°F), 100 psi

2-15. OIL DILUTION. During cold weather operation, hard starting of the engine is minimized by using gasoline to thin the oil during the previous stopping of the engine. This is accomplished by operating the oil dilution system, consisting mainly of electrically operated solenoid and diluter valves. A manual shutoff valve, in the oil dilution line, provides protection against the possibility of inadvertent operation of the oil dilution system. This valve is safety-wired in the closing position whenever weather conditions do not require oil dilution. The oil dilution control switch is located on the fuel control panel.

2-16. STARTING ENGINE.

2-17. On starting the engine, proceed as follows:

a. With ignition and battery switches "OFF", manually rotate propeller four or five times. Never rotate engine backwards to clear it.

b. Mixture—"IDLE CUT-OFF".

c. Fuel tank selector—"RIGHT MAIN".

d. Propeller control—FULL "INCREASE RPM".

e. Throttle—ONE INCH OPEN (Approx.)

f. Supercharger—"NEUTRAL".

g. Carburetor air control—FULL IN.

h. Cowl flaps—"OPEN".

i. Oil cooler shutter—position dependent on temperature.

j. Battery switch—"ON".

k. Auxiliary (emergency) fuel pump switch—"ON".

l. Primer switch—"ON" (three to five seconds).

m. Ignition switch—on "BOTH".

n. Starter switch—"ON".

o. Mixture control—advance to "AUTO-RICH" as engine fires. If engine fails to continue running, return to "IDLE CUT-OFF".

p. Idle engine at or below 1000 rpm until oil pressure begins steadying out.

WARNING

If the oil pressure gage does not register within 30 seconds, stop the engine and investigate. Never run oil pressure over 200 psi during warm-up.

2-18. WARM-UP AND GROUND TEST.

2-19. Open the throttle to 1200 rpm until oil-in temperature reaches 40°C range. An increase in oil pressure when the throttle is opened, indicates that further warm-up is required. If the oil dilution system has been used prior to stopping the engine, warm up the engine sufficiently before take-off so that the oil system is operating efficiently.

2-20. CHECK MAGNETOS. Open the throttle to 30 in. hg which will give 2100-2200 rpm in low pitch. Check the functioning of the magnetos by putting the ignition switch in "LEFT". The normal drop-off in rpm is 50 to 75 and should not exceed 100 rpm. Return the switch to "BOTH" until rpm stabilizes then turn to "RIGHT" and note rpm drop-off. Do not operate longer than 15 seconds on a single magneto. This test should be made in as short a time as practicable.

Note

Make this check prior to the IDLE MIXTURE CHECK to make certain that the plugs are not fouled.

2-21. SUPERCHARGER CHECK AND DESLUDGING PROCEDURE. The supercharger check should never be made nor the clutches desludged until the oil temperature has reached 40°C (104°F), and it is preferable to wait until the oil temperature has reached 60°C (140°F). If there is not enough time to complete a regular supercharger check, desludge the clutches twice as directed below.

CAUTION

Always shift from one position to another without hesitation, or at least without dwelling between positions.

a. Adjust the throttle to obtain 1400 rpm with the propeller control in take-off position, then shift from "NEUTRAL" to "LOW".

b. After 30 to 60 seconds in "LOW", shift to "HIGH".

c. After 30 to 60 seconds in "HIGH", open the throttle to 30 in. manifold pressure and note rpm.

d. Shift back to "LOW" and after the manifold pressure has stabilized readjust the throttle to obtain 30 in. manifold pressure. Note rpm.

e. Shift from "LOW" to "NEUTRAL" and again adjust the throttle to obtain 30 in. manifold pressure. Note rpm.

Note

An increase in the engine rpm when shifting from a higher to a lower blower ratio, while maintaining a constant manifold pressure, indicates that the clutches are operating correctly. Less power is required to drive the supercharger in the lower ratios thereby releasing more power to drive the propeller.

f. Desludge the couplings by shifting as directed in paragraphs a., b., and c. above. After a minimum of 30 seconds operation in high blower, move the supercharger control to "NEUTRAL" without any hesitation at the "LOW" position.

2-22. CHECK PROPELLER CONTROL. With the engine turning at approximately 2000 rpm, move the propeller control toward "DECREASE RPM" until a 300 rpm drop is indicated then return control to full "INCREASE RPM". When returned to "INCREASE RPM" position, rpm should be steady and free from surging. The constant speed range of the governor is between 1200 and 2700 rpm.

2-23. If the oil dilution system has been used, take care to see that the propeller pitch charging mechanism is operating properly as the oil operating the hydromatic propeller is not diluted.

2-24. CHECK CARBURETOR IDLE MIXTURE. Set the throttle to obtain 600 rpm—propeller control in full "INCREASE RPM" position. Move the mixture control lever momentarily, but with a smooth steady pull, to the "IDLE CUT-OFF" position and observe the tachometer for any change in rpm. A momentary rise above 10 rpm indicates too rich a mixture; no change indicates too lean a mixture. A five to ten rpm rise is desired to prevent spark plug fouling at idling speeds and at the same time to afford good acceleration characteristics.

Note

Auxiliary fuel pump should be "ON" for this check.

2-25. CHECK INSTRUMENTS, RUNNING AT 2000 RPM.

a. Oil temperature—60-85°C (140-185°F) desired. (40°C (104°F) minimum).

b. Oil pressure —75-95 psi.

c. Fuel pressure—16-18½ psi. Auxiliary fuel pump switch "OFF" during this check.

d. Cylinder head temperature 100°C (212°F) (desired minimum). Note manifold pressure as a reference for future checks.

e. Hydraulic system pressure—1250-1500 psi.

2-26. GENERATOR SYSTEM CHECK. If an external power source is being used, have the ground crew disconnect it and make the following check:

a. With the engine idling and the battery switch "ON", turn on some electrical load such as lights, instruments, radio, etc.

b. Check the closing of the reverse current cut-out by slowly increasing the engine rpm. The voltmeter reading should increase to the value at which the reverse current relay cut-out closes, which will be indicated by a dip in voltage. This will occur at approximately 26.5 volts.

c. As the engine rpm is further increased, the voltage should rise to about 28.0 and then remain at this value regardless of a further increase in rpm.

d. If the reverse current cut-out fails to close between 26.0 and 27.0 volts or the regulated voltage is not maintained between 27.5 and 28.5 volts, there is trouble in the generator system. The trouble must be corrected before taking off.

2-27. PITOT TUBE HEAT. "ON" if icing conditions prevail.

2-28. DROPPABLE TANKS. With auxiliary fuel pump switch "ON", check fuel flow from each drop tank. Return fuel tank selector to "MAIN" at end of check and leave fuel pump switch "ON".

WARNING

Do not change tanks just before take-off. Allow several minutes ground run after change.

2-29. AUTOMATIC PILOT CHECK.

2-30. In order to insure that it is functioning properly, check its operation in the following manner:

a. Run engine at 1500 rpm (minimum) for full vacuum.

b. Bank and climb gyro—"UNCAGED".

c. Directional gyro—"UNCAGED".

d. Center stick and pedals then engage automtatic pilot by pulling the ON-OFF control "T" handle to "ON". The controls should jar slightly as the unit engages, indicating proper synchronization. The aileron and elevator controls should remain in position. The rudder can be centered with the directional gyro caging knob. As the airplane is not level (three-point position), the horizon bar of the bank and climb gyro will move slowly toward the correct indication of the attitude of the airplane and cause the elevator and aileron controls to follow.

e. Check for direction of control movement by manipulating the controller lever. Observe the direction of the control stick and surface controls movement to ascertain that each control surface moves in the proper direction.

CAUTION

Be careful that the tail of the airplane does not rise from the ground when checking the dive control.

f. Depress straight-course button on controller lever, then cage directional gyro and turn caging knob slowly to the left and right—rudder should move accordingly. After this check, reset and uncage gyro.

2-31. CENTRALIZED POSITION CHECK (ELECTRICAL SYSTEM CHECK). Move the control stick to a position other than "NEUTRAL" then pull the ON-OFF control "T" handle to "CENTRALIZED". The electrical system should return the control stick to a neutral position.

2-32. OVER POWER CHECK. Check to be sure that the automatic pilot can be over powered with the ON-OFF control handle in the "ON" position. After this check, disengage the automatic pilot by pushing ON-OFF control handle to "OFF".

2-33. SCRAMBLE TAKE-OFF.

2-34. An emergency take-off may be made in accordance with the regular take-off procedure provided that:

a. Oil pressure is steady.

b. Oil temperature—at least 40°C (104°F).

c. Throttle may be advanced without causing the engine to cough or cut-out. Engine acceleration should be smooth.

2-35. TAXIING INSTRUCTIONS.

2-36. Taxi with the tail wheel unlocked except in strong cross winds. A steady run of the engine is preferable to repeated short bursts of power. Set cowl flaps in FULL "OPEN" position. Use the brakes to help guide the airplane but avoid riding them to prevent overheating.

2-37. TAKE-OFF.

2-38. CLEAR ENGINE. It is necessary to clear the engine at high power immediately before take-off.

2-39. CHECK-OFF LIST.

a. Obtain traffic clearance.

b. Enclosure—"OPEN".

c. Shoulder harness—TIGHT.

d. Wings—SPREAD and "LOCKED". Red warning cylinders flush with wing skin.

e. Cowl flaps—½ "OPEN" (approx).

f. Oil cooler and intercooler shutters—"OPEN" (F6F-3 and -3N). Oil cooler shutter—"OPEN" (F6F-5 and-5N). Intercooler shutters—"CLOSED" (F6F-5 and -5N).

g. Carburetor air control FULL IN—"DIRECT". If icing conditions exist, use "PROTECTED AIR" to clear engine, then switch to "DIRECT" for take-off.

h. Propeller control — FULL "INCREASE RPM" (2700).

i. Mixture control—"AUTO RICH".

j. WEP switch—"OFF".

k. Supercharger control — "NEUTRAL" (regardless of airport altitude).

l. Fuel tank selector—"RIGHT MAIN".

m. Auxiliary fuel pump—"ON".

n. Tab control settings: aileron—0; elevator—0; rudder—2 marks "NOSE RIGHT". (With normal airplane loading).

o. Wing flaps—"UP" or "DOWN" as required.

p. Tail wheel—"LOCKED", land—"UNLOCKED," carrier.

q. Cockpit heater switch—"OFF" (installed in early model airplanes).

r. Automatic pilot—DISENGAGED (late model F6F-5N).

s. Throttle—open smoothly to 54 in. hg max.

t. Raise landing gear immediately after becoming airborne.

u. Raise flaps.

v. Adjust power plant according to the Power Plant Chart, Section III.

2-40. CATAPULT CHECK-OFF LIST.

a. Enclosure—"OPEN" and STOPPED.

b. Shoulder harness—TIGHT.

c. Place back and head firmly against seat and headrest.

d. Place feet against rudder pedals with legs stiff.

e. Brace right arm.

f. Wing flaps—FULL "DOWN".

g. Check tab control settings.

h. Throttle and propeller levers friction should be sufficient to prevent controls from moving if hand is removed.

i. Use full take-off power—54 in. hg and 2700 rpm.

2-41. ENGINE FAILURE DURING TAKE-OFF.

2-42. If the engine fails during the take-off, proceed as follows:

a. Nose down to maintain flying speed.

b. Shoulder harness—TIGHT.

c. Jettison external load items. Safe armament units before release.

d. Check that landing gear control is in "UP" position, if landing gear does not have time to retract it will collapse upon landing.

e. After making certain that the selected field can be reached, lower the wing flaps, and if time permits:

f. Ignition switch—"OFF".

g. Battery switch—"OFF".

h. Mixture control—"IDLE CUT-OFF"

i. Fuel selector control—"OFF".

2-43. CLIMB.

2-44. Refer to Section III and Appendix for flight operation instruction charts, ranges, and recommended power settings.

2-45. RATED POWER CLIMB.

a. Mixture control "AUTO LEAN" (leave in "AUTO RICH" after take-off until climbing IAS is attained).

b. Cowl flaps—⅓ "OPEN" (approx).

c. Do not exceed 260°C (500°F) cylinder head temperature.

2-46. SEA LEVEL TO 7000 FEET.

a. "NEUTRAL" blower.

b. 2550 rpm.

c. Use 44 in. manifold pressure (but not more), below full throttle.

d. Hold IAS constant at 140 knots.

2-47. 7000 TO 22,000 FEET.

a. "LOW" blower.

b. 2550 rpm.

c. Use 49.5 in. manifold pressure (but not more) below full throttle.

d. Hold IAS constant at 140 knots.

2-48. 22,000 FEET AND UP.

a. "HIGH" blower.

b. 2550 rpm.

c. Use 49.5 in. manifold pressure (but not more) below full throttle.

d. Hold IAS constant at 140 knots.

2-49. The IAS may vary five knots over or under 140 knots IAS without appreciably affecting the rate of climb. If cooling is not adequate, increase IAS as much as 10 knots before opening cowl flaps further. Increase IAS as required to maintain adequate cooling rather than open cowl flaps more than one-half. If stubborn overheating is encountered, and it refuses to yield to other methods of control, shift the mixture control to "AUTO RICH."

2-50. When maximum rate of climb is not essential, better cooling will be obtained if the IAS is increased 10 to 20 knots over the IAS for maximum rate of climb as the resulting loss in rate of climb will be small.

2-51. CRUISING CLIMB.

a. Mixture control "AUTO LEAN".

b. Cowl flaps "CLOSED".

c. Do not exceed 232°C (450°F) cylinder head temperature.

2-52. SEA LEVEL TO 12,000 FEET.

a. "NEUTRAL" blower.

b. 2050 rpm.

c. Use 34 in. manifold pressure (but not more) below full throttle.

d. Maintain 150 knots IAS with both clean and overloaded fighters.

2-53. 12,000 TO 25,000 FEET.

a. "LOW" blower.

b 2050 rpm.

c. Use 34 in. manifold pressure (but not more) below full throttle.

d. Maintain 150 knots IAS with both clean and overloaded fighters.

2-54. 25,000 FEET UP.

a. "HIGH" blower.

b. 2050 rpm.

c. Use 34 in. manifold pressure (but not more) below full throttle.

d. Maintain 150 knots IAS (with both clean and overload fighters).

2-55. Normally no opening of the cowl flaps should be necessary during climbs at cruising power. If cooling is inadequate try increasing IAS before resorting to use of cowl flaps. If stubborn overheating is encountered, and it refuses to yield to other methods of control; shift the mixture control to "AUTO RICH". Use of "AUTO RICH" should be avoided whenever possible because of the increased fuel consumption.

2-56. GENERAL FLYING CHARACTERISTICS.

2-57. STABILITY. The airplane is stable at all normal loadings.

a. Lowering of the landing gear tends to make the airplane slightly nose-heavy.

b. Lowering of the wing flaps tends to make the airplane nose-heavy.

2-58. AIRSPEED AND ACCELERATION RESTRICTIONS.

2-59. The speed and acceleration limits in the Appendix, figure A-5 or A-6, are largely determined by the fact that as these limits are approached, a buffeting or shaking of the airplane is likely to be encountered. If this becomes appreciable, immediately reduce speed or acceleration or both. Exceeding these limits will very probably be evidenced by permanent distortion or possibly by failure of the horizontal stabilizer. It is therefore, of the utmost importance that pilots avoid exceeding these limits. In general, the shaking phenomenon occurs at lower indicated airspeeds at higher altitudes as shown in figure A-5 or A-6. Pilots must also be on the alert to avoid steep dives at extreme altitudes because such dives will result in excessive speeds, which will in turn produce compressibility effects with their attendant dangers.

2-60. USE OF AILERONS. The maximum speed for unlimited use of the ailerons is 270 knots (F6F-5 and -5N) and 260 knots (F6F-3 and -3N) IAS. At higher speeds, use of the ailerons is restricted to the same control force required for full aileron operation at the above speeds for the respective models. Also maximum use of the ailerons is limited to accelerations of not over 5.0 g positive and 2.0 g negative.

2-61. OPERATION WITH 11.75 IN. AR. The following operational data is the result of Navy Service Tests. The airplane should be flown in the following manner with these rockets installed:

WARNING

High wing loadings, high stalling speeds, low rates of climb, loss of maneuverability, and marginal rudder control in the carrier approach condition at weights above 16,000 lbs require that a very careful indoctrination period be followed.

a. All take-offs should be made with full flap deflection (16,000 lbs gross weight).

b. Trim tab settings for take-off: elevator—1° "NOSE DOWN"; ailerons—"NEUTRAL"; and rudder —30° "RIGHT".

c. Following take-off at high gross weights, it is necessary to hold hard right rudder pressure until the flaps are raised even though full right rudder tab is used.

d. At high gross weights above 16,000 lbs and low speeds, it becomes difficult to bring a wing up with the rudder. Also considerable rudder must be used to maintain straight flight.

e. Restrict all maneuvers to gentle turns until a safe speed and altitude are reached.

f. Stalling characteristics are satisfactory. However, with either a 100 or 150 gallon drop tank installed on the port wing and one 11.75 in. AR on the starboard wing, the power-on stall in the landing condition is followed by a roll to the right. Altitude required for recovery from a stall increases rapidly with increasing gross weight, becoming approximately 1000 feet at a gross weight of 17,164 lbs.

WARNING

Carrier landings with these rockets installed should be confined only to emergencies. Under no circumstances should the airplane be landed with a rocket installed on only the port wing rack.

g. The airplane may be landed in an emergency in the following configurations at gross weights up to 14,300 lbs.

One 11.75 in. AR—starboard wing rack.

Two 11.75 in. AR—one on each wing rack.

One 11.75 in. AR—fuselage rack.

h. With a rocket installed on the starboard wing rack, full left rudder trim and full left wing down trim are insufficient to maintain level flight at 80 knots IAS. Full left aileron is required from cut to landing.

i. With two rockets installed, one on each wing rack, the airplane is comfortable in the carrier approach down to a minimum of 80 knots IAS, but drops in hard after the cut.

2-62. CRUISING.

2-63. Unless there is an urgent reason for using higher powers, it is recommended that all cruising be done at "maximum cruising" power or lower. Use of higher powers will exact a penalty in the form of increased fuel consumption and reduced engine life and reliability. For maximum cruising power, use the rpm—manifold pressure combinations shown on the Engine Calibration Curves, figure A-7.

2-64. For lower cruising power—observe the following rules:

2-65. RULE FOR CRUISING. Maintain 34 in. manifold pressure (but not more) or full throttle, if above critical altitude. Control IAS by adjusting rpm. Do not exceed 2050 rpm in "NEUTRAL" "LOW" or in "HIGH" blowers. Do not use less than 1300 rpm. (If spark plug fouling occurs at 1300 rpm in cold weather, increase rpm enough to stop fouling.) If IAS obtained at 34 in. hg and minimum rpm is too high, reduce manifold pressure as necessary. Otherwise, do not use less than 34 in. below critical altitude. Use "AUTO LEAN" at all times unless shift to "AUTO RICH" is necessary to control head temperature. Do not exceed 232°C (450°F) cylinder head temperature. Do not open cowl flaps, or oil cooler and intercooler shutters unless temperatures approach limits.

2-66. RULE FOR OBTAINING MAXIMUM RANGE. Observe rule for cruising, and maintain 145 knots IAS (constant) regardless of airplane configuration. Fly at lowest feasible altitude. A small gain will result if IAS is changed to compensate for wind. If this refinement is desired, increase IAS five knots for each ten knots of headwind until IAS reaches 150 knots. Do not exceed 160 knots IAS, regardless of wind force. Do not compensate for tailwinds.

2-67. RULE FOR OBTAINING MAXIMUM ENDURANCE. Observe rule for cruising, and maintain 125 knots IAS (constant) regardless of airplane configuration. If mushiness is felt when gross weight is high, increase IAS slightly. Fly at lowest feasible altitude.

2-68. WAR EMERGENCY POWER. War emergency ratings have been established to permit operation at the highest power, within reasonable safety limits, that the structural limitations of the engine will permit.

2-69. When the throttle is advanced beyond the limit stop to full forward position, it closes a micro-switch which controls the water supply to the engine. When the micro-switch is closed, three things occur:

a. Water pressure shuts off a fuel jet in the carburetor, "de-riching" the mixture to approximately best-power.

b. Water pressure resets the auxiliary stage supercharger regulator for approximately 3.5 in. hg higher carburetor inlet pressure.

c. Water is metered and mixed with the fuel.

2-70. Before using WEP, close the water pump toggle switch located on the left hand cockpit shelf. The water pump will then clear the lines of air, and build up water pressure behind the micro-switch controlled solenoid valve.

2-71. With the power controls set for military power (see Power Plant Chart, figure 3-2), advance the throttle past the limit stop to full forward position. If in "NEUTRAL" blower, the manifold pressure should rise immediately to not over 60 in., and a second or two later, a surge of power should be felt. If in "LOW" or "HIGH" blower, the manifold pressure rise and power surge may occur more nearly at the same time.

CAUTION

Continuous operation at WEP should not exceed five minutes.

2-72. If "LOW" or "HIGH" blower is being used, the engine will automatically return to military power when the water supply is exhausted. When operating in "NEUTRAL" blower, the auxiliary stage regulator cannot control the manifold pressure *and the throttle must be brought back behind the limit stop immediately when the water has been exhausted.* Failure to observe this rule may result in serious engine damage, or total failure, within a few seconds. The water pump switch should be placed in "OFF" position as soon as the water supply is exhausted. The pump is not designed for dry running.

2-73. If the engine fails to respond within two or three seconds after the throttle is advanced for WEP, return the throttle to setting for military power immediately.

2-74. WEP is intended for combat use only, and should not be used at any other time, except during familiarization. The limited supply of water that can be carried should be conserved like ammunition, because it may last none too long when really needed. Although there should be no hesitation in using WEP for maximum effect during combat, pilots should be alert to the fact that engine life is shortened considerably by operation at this power, and should be on the lookout for signs of lessened performance or reliability.

Note

If the pilot is flying above the critical altitude for any blower setting, water injection will have a negligible effect on engine power.

2-75. MANIFOLD PRESSURE SWITCH. Airplanes BuAer No. 10801 and subsequent, are equipped with an electrically operated manifold pressure switch connected in parallel with the engine throttle microswitch. This manifold pressure switch closes the water regulator solenoid valve circuit when manifold pressure reaches 45 in. hg and opens the circuit when pressure falls below 51 in. hg thus allowing a gradual reduction in horsepower when the throttle is moved AFT from the FULL FORWARD position.

2-76. GENERATOR SYSTEM CHECK. At regular intervals during flight, the generator system should be checked by turning the battery switch "OFF". If the electrical loads remain in operation and the voltmeter readings are between 27.5-28.5 volts, the generator is functioning properly. Put battery switch back to "ON" position.

2-77. SUPERCHARGER OPERATION.

2-78. There are inherent characteristics of the auxiliary blower engaging system on two-stage engines that may sometimes cause cutting out or other irregular operation while the blowers are shifting. This condition is beyond the pilot's control, but the use of the procedure outlined below will result in a minimum of manifold pressure surge, rpm surge, and cutting out during blower shifts from a lower to a higher ratio.

a. Shift the blower selector lever to the desired new blower ratio position without reducing rpm.

b. Wait for the first indication of the manifold pressure to pick up.

c. Quickly retard the throttle to about ⅔ open. (This ⅔ open position will hereafter be referred to as the "surge control position".) After the blower has become fully engaged (about 15 to 20 seconds after repositioning the blower selector lever), the throttle may be opened to maintain the desired manifold pressure.

Note

If the engine is already operating with the throttle at or near the "surge control position" when the shift is made, it will not be necessary to retard the throttle further. If the engine is operating at a throttle position which is less than the "surge control position", it is best to open the throttle to the "surge" control position before shifting.

2-79. After a few practice shifts, the pilot will find himself familiar with the feel of the throttle while shifting so that he will be able to make the shift with a minimum of discomfort.

2-80. When operating at powers below 1900 rpm and 30 in. hg, it is not necessary to retard the throttle after repositioning the blower selector lever as the manifold pressure surge will be much smaller.

2-81. If the auxiilary stage is operating and it is desired to shift to a lower blower ratio, move the blower selector lever to the desired lower blower ratio position. It is not necessary to change either the throttle position or the engine speed during this process. Blowers may be shifted down from high to low, high to neutral, or low to neutral. Usually there is a smooth surgeless transition of power as the blower ratio decreases. However, it is possible that the engine may cut out momentarily if the blower ratio is shifted from high to neutral at altitudes above 20,000 ft. after considerable high power operation in high blower.

CAUTION

In flight do not shift into the same ratio at intervals of less than five minutes, except in an emergency. This allows sufficient time for dissipation of heat from the clutches which is generated during the shift. When in the higher blower ratios, avoid unduly high rates of change in engine speed to prevent excessive loads on the clutches and supercharger drive.

2-82. When climbing at WEP in neutral or low blower, it is not necessary to retard the throttle after shifting to a higher blower ratio. The whole auxiliary blower system is working at such a high percentage of its capacity at WEP that the manifold pressure does not surge beyond 60 in. hg.

CAUTION

Shifts at WEP shall be made only during an emergency.

2-83. CHANGING POWER CONDITIONS. In order to prevent excessive pressures within the cylinders when changing power, the following procedures shall be used:

a. INCREASING ENGINE POWER. Adjust propeller control to desired rpm, then adjust the throttle to obtain the desired manifold pressure.

b. DECREASING ENGINE POWER. Adjust the throttle to obtain the desired manifold pressure, then adjust the propeller control to the desired rpm.

2-84. AUTOMATIC PILOT ENGAGEMENT. Until thoroughly familiar with its operation, the automatic pilot should not be engaged before a reasonable altitude (2000 feet approximately) has been attained. After complete familiarization, the pilot may be engaged after take-off.

a. Trim airplane for hands-off level flight.

Note

The automatic pilot, being self-synchronous, will take control and maintain the existing flight attitude at the moment of engagement within its operating limits: Bank—45°, Climb—30°, and Dive—50°.

b. Set the direction gyro card for heading. Engage the pilot by pulling the ON-OFF control "T" handle to "ON". By holding the controls as the pilot is engaged, you will feel when it is flying the airplane.

c. At the time of application of the "T" handle to "ON", the pilot will take control and maintain the existing flight attitude of the airplane until an attitude change is effected by the controller lever; or until the automatic pilot is overpowered by operating the airplane controls.

d. To make coordinated turns, push controller lever to the left or right as desired and hold until the desired rate of turn is obtained then release. This automatically disengages the straight course feature. In order to resume a straight course again, momentarily press the controller lever button after the airplane is set on the new course.

e. To bank without turning, push the controller lever to the left or right while depressing the button. Hold lever and button in position only until desired angle of bank is obtained (limit—30°).

f. To climb, pull maneuvering lever aft. Hold lever in position until desired angle of climb is obtained (limit—30°).

g. To dive, push maneuvering lever forward. Hold lever in position until desired angle of dive is obtained (limit—50°).

h. To return to level flight automatically, pull ON-OFF control "T" handle to "CENTRALIZED". Hold until level flight or desired recovery angle is attained. The airplane will automatically return to straight and level flight from attitudes within the limits of the gyroscopes of the automatic pilot.

i. To overpower the automatic pilot, apply approximately twice the normal force on the airplane controls.

2-85. PERIODIC FLIGHT CHECK.

a. Check how closely the flight attitude is held. Check for hunt by inducing transients in the elevator and ailerons.

b. Hold a selected heading for a minimum of 10 minutes. Check for drift, left or right, maximum ±1°.

c. During banks to left and right, rate of roll should be approximately 5° per second. Check for smoothness of roll and rudder coordination.

d. Check operation in a climb to 100 knots and dive to 280 knots. Check combined climb and bank, and dive and bank.

e. "CENTRALIZE" from bank, climbing turn and diving turn. Limits after recovery are 800 feet per minute climb or glide.

f. Overpower manually all controls. Note return to original attitude. Also note any tendency toward oscillation of the controls.

Note

1

At periodic intervals, correct for directional gyro drift in the conventional manner.

2

Keep gyros uncaged at all times except when leveling the bank and climb control or resetting the directional control.

3

Keep the airplane approximately in trim. Check at such times when the airplane is being flown manually.

4

During turns, correct for loss of altitude due to bank and climb gyro turn error by using controller lever.

5

In icing conditions, disengage the pilot frequently and move the controls manually to see that they are free.

6

For corrections in excess of 2°, disengage the directional control by momentarily flicking the controller lever to the side. Rudder to the new heading and then depress directional (lock) button momentarily to maintain new heading.

2-86. STALLS.

2-87. The stalling characteristics of this airplane are very satisfactory. As the stall approaches, the right wing tends to drop slowly and a severe quivering is felt throughout the airplane. The stalling speeds are given on figure A-3, Appendix.

2-88. SPINS.

2-89. Normal spins of not over two turns and inverted spins of not over one turn are permitted. With the airplane loaded to 11,250 lbs. and with the cg position at 26.0%, spins of four turns have been investigated. A normal entry was made with ailerons one half against the spin.

2-90. RIGHT SPIN.

a. Nose drops to 50-60° angle.

b. Aileron forces negligible.

c. Nose oscillation same frequency as the rate of rotation of airplane.

2-91. RECOVERY.

a. After 4¼ turns, full rudder reversal was made followed about one second later by full elevator reversal. The spin steepened sharply and the rate of rotation appeared to double. Rudder forces were relatively light, elevator forces were moderately heavy, aileron forces were fairly heavy.

b. Recovery effected in 1½ turns and level flight attained.

c. Loss of altitude 5000 feet.

2-92. LEFT SPIN.

a. Nose drop not as steep as in right spin.

b. Aileron forces are heavier.

c. Nose oscillations are of greater amplitude.

2-93. RECOVERY.

a. After four turns recovery was effected in 1¾ turns.

b. Aileron and elevator forces about double those experienced in right spin; rudder forces about the same.

c. Loss of altitude 4400 feet.

2-94. PERMISSIBLE ACROBATICS.

2-95. All acrobatics with the exception of those listed in paragraph 2-2 may be performed. However, before starting any acrobatics or violent maneuvers, cage the directional gyro and the artificial horizon. Disengage the GR-1 automatic pilot (later model F6F-5N).

2-96. DIVING.

2-97. For ordinary short dives in maneuvers, the engine nose section will not load up nor will the engine cool off to any extent. The controls should be set as follows:

a. Canopy—CLOSED.

b. Supercharger control—"NEUTRAL" or "LOW" depending on altitude.

c. Set propeller control—2050-2250 rpm.

d. Cowl flaps—"CLOSED".

e. Oil cooler and intercooler shutters—"CLOSED".

2-98. For prolonged dives to avoid loading up the engine nose section or cooling the engine excessively:

a. Set the propeller control to maximum cruising—2250±100 rpm.

b. Set throttle—15 in. hg manifold pressure(minimum).

c. Mixture control—"AUTO RICH".

2-99. MAXIMUM DIVING RPM—3060 FOR 30 SECONDS.

2-100. In the event that overspeeding beyond the overspeed limit of the engine occurs, the following procedure is recommended:

a. Throttle to "CLOSED".

b. Propeller to "DECREASE RPM".

c. Reduce airspeed to minimum speed for safe glide.

2-101. NIGHT FLYING.

2-102. Proceed as follows:

a. Wear red goggles for one half hour before each flight.

b. Avoid all light (searchlights, flares, etc.) as much as possible, except red light.

c. Do not look at lighted instruments longer than necessary even though light is red.

d. Practice "blindfold drills" until all controls can be operated with ease in the dark.

e. Scan the sky systematically, moving the eyes over small areas at a time. Do not stare. Learn to look for night targets out of the corners of the eyes.

f. Use oxygen for all night flights.

g. Learn to look for and identify objects solely by contrast (light and shadow).

2-103. APPROACH AND LANDING.

2-104. The approach and landing may be made with or without power. Reduce speed during initial circuit to approximately 120 knots IAS and then prepare for landing as follows:

2-105. CHECK-OFF LIST.

a. Enclosure—"OPEN"—STOPPED.

b. Shoulder harness—TIGHT.

c. Tab control settings—as required.

d. Auxiliary fuel pump—"ON".

e. Fuel selector valve—BEST TANK.

f. Mixture control—"AUTO RICH".

g. Propeller—"INCREASE RPM"—2250-2450 rpm.

h. Supercharger—"NEUTRAL".

i. Cowl flaps—"OPEN".

j. Oil cooler and intercooler flaps—"OPEN".

k. Tail wheel caster—"LOCKED" for land operation and "UNLOCKED" for carrier operation.

l. Arresting hook extended (carrier operation).

m. Safety guns—master armament switch—"OFF".

n. Rocket safety plug (MK 3 unit)—REMOVE.

o. Wing flaps—"DOWN".

2-106. The landing characteristics of this airplane are excellent. The landing speeds will vary according to the loading conditions of the airplane. At the conclusion of the land run:

a. Wing flaps—"UP".

b. UNLOCK tail wheel before taxiing for land operation.

c. Cowl flaps "OPEN".

d. Oil cooler—intercooler flaps—as required.

2-107. STOPPING ENGINE.

2-108. The following procedure shall be followed when stopping the engine:

a. Propeller control—full "INCREASE RPM".

b. Throttle set at approximately 1400 rpm.

c. Shift supercharger controls remaining in each position for approximately 30 seconds for de-sludging blowers (see paragraph 2-21).

d. Operate engine at 1000-1200 rpm to cool if operation has been set at high powers.

e. Auxiliary fuel pump—"OFF".

f. Put mixture control to "IDLE CUT-OFF".

g. Put ignition switch in "OFF" position after propeller stops rotating.

h. Place fuel selector in "OFF" position.

i. Battery switch—"OFF".

j. Leave cowl flaps—"OPEN" until engine cools.

k. Install surface controls lashing device if airplane is to remain grounded.

2-109. OIL DIULTION PROCEDURE. If cold weather starting (temperature below—2°C (-35°F) is anticipated, the oil dilution system should be operated as follows:

a. Oil dilution shut-off valve—OPEN.

b. Engine speed—1000 rpm.

c. Oil dilution switch "ON" for approximately two minutes.

d. Oil-in temperature not more than 40°C (140°F.)

Note

> When the solenoid valve is opened by the switch action, there willl be a sharp drop in indicated fuel pressure. Fuel pressure should return to normal immediately after closing the valve. If not, stop the engine at once and check for valve leakage.

d. Stop engine by moving mixture control to "IDLE CUT-OFF".

e. Ignition switch—"OFF".

f. When the cold engine is subsequently started, and after running a short time the oil pressure starts to fluctuate or drop, the dilution switch shall be held "ON" intermittently for intervals of a few seconds over a period of approximately fifteen seconds. If the oil pressure still does not steady out, stop the engine and wait for approximately five minutes before attempting another start.

2-110. BEFORE LEAVING PILOT'S COCKPIT.

2-111, Install the controls locking device. Refer to paragraph 1-106.

2-112. Check that all switches are in the "OFF" position except the generator switch.

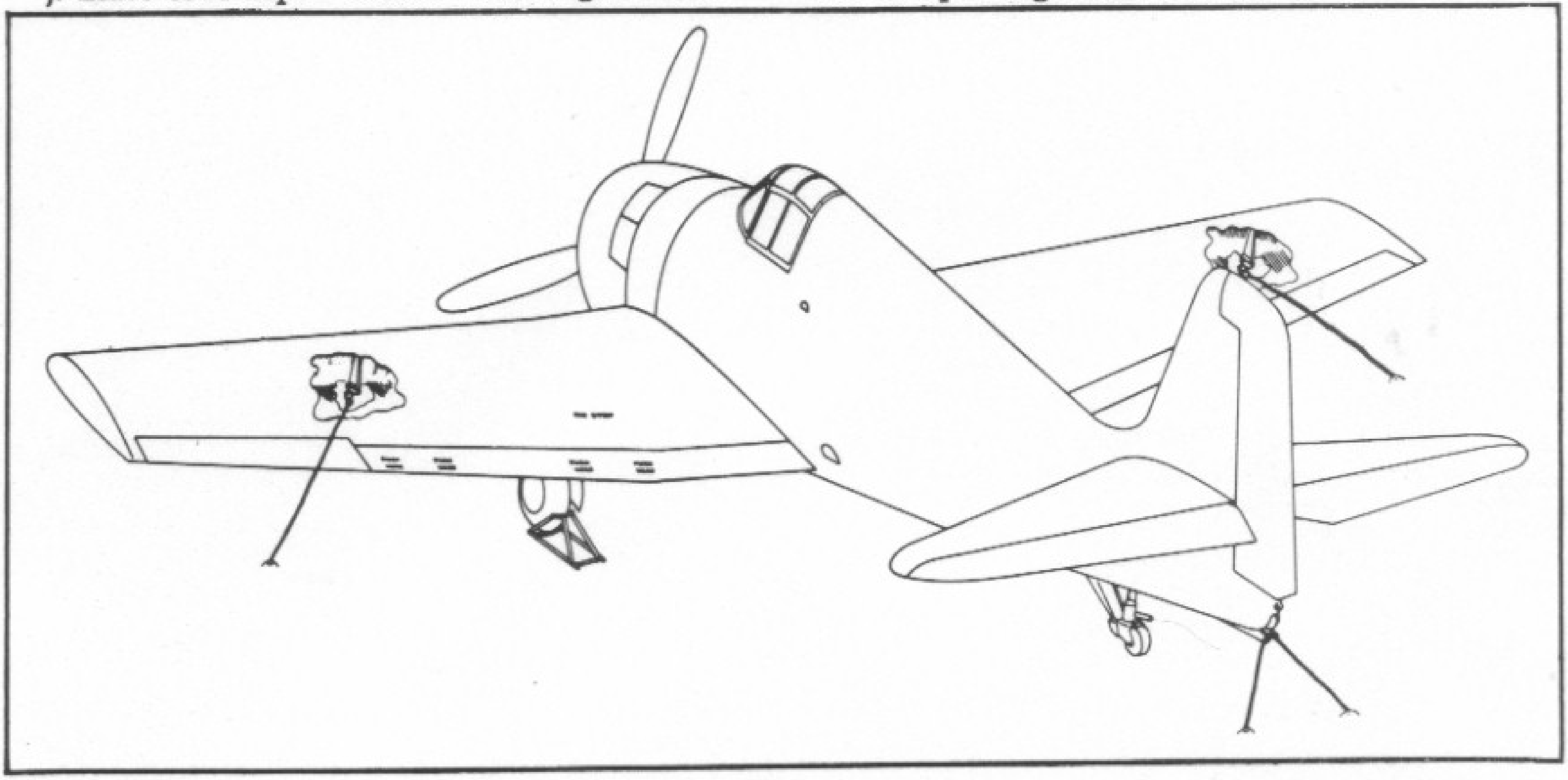
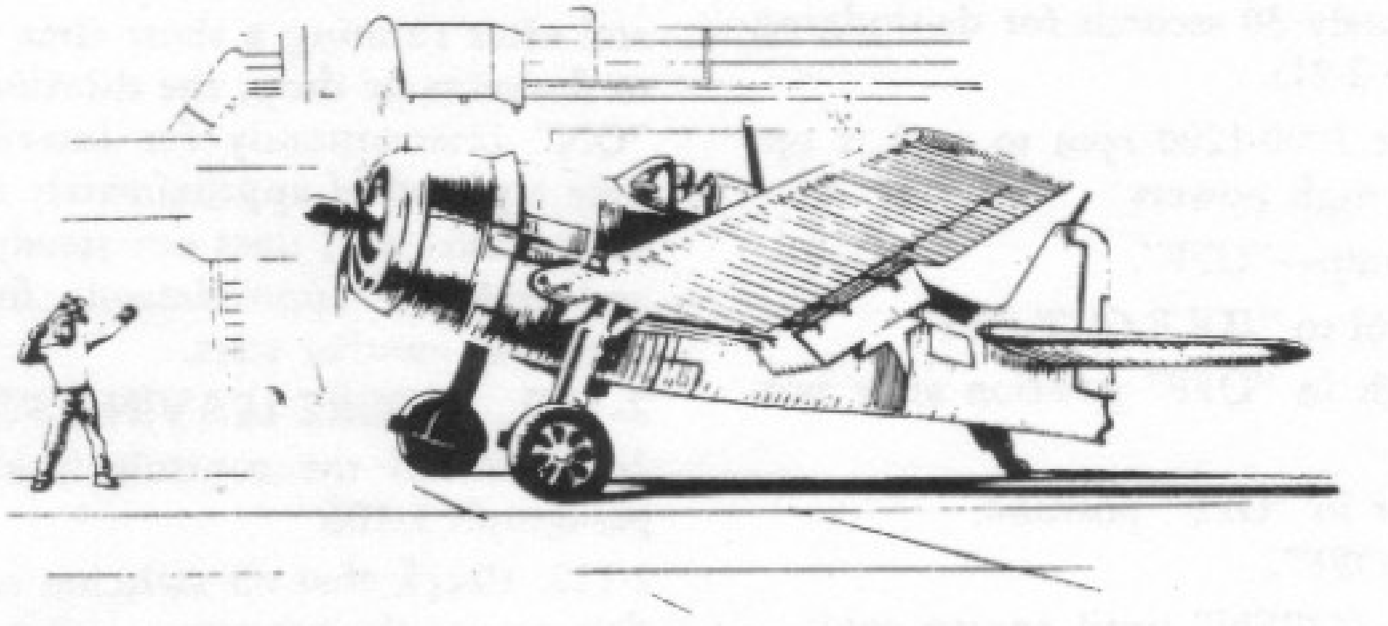


Figure 2-1. Mooring Diagram

2-113. MOORING.

2-114. The airplane may be moored with the wings in the spread position or in the folded position. To moor the airplane with wings spread, attach ropes to the tie-down rings located in the wing outer panels between Stations #159½ and #168. Secure the tail of the airplane by attaching a rope to the catapult hold-back fitting and place chocks at the main wheels, if available. If not, then secure main wheels by placing ropes through the nutcracker fittings. To moor the airplane on a carrier deck with the wings in folded position, attach three lines to each main shock strut just below the lowest wheel fairing clamp and secure one line forward, one aft, and one outboard to the carrier deck. Lash the tail of the airplane to the deck by attaching a line to the rear catapult fitting.



SECTION III
OPERATING DATA

PITOT STATIC ORIFICE LOCATED IN FUSELAGE SKIN AT STATION #97

IAS *(Knots)*		*Correction (Knots)*
FLAPS UP		
120	Deduct	2.5
140	Deduct	2.5
160	Deduct	4.5
180	Deduct	4.5
200	Deduct	4.5
220	Deduct	4.5
240	Deduct	4.5
260	Deduct	3.5
280	Deduct	3.0
300	Deduct	2.5
FLAPS DOWN		
100	Deduct	6.5
110	Deduct	8.0
120	Deduct	9.0
130	Deduct	9.5
140	Deduct	11.5

PITOT STATIC ORIFICE LOCATED AT WING TIP

IAS *(Knots)*		*Correction (Knots)*
FLAPS UP		
100	Add	4.5
120	Add	4.5
140	Add	6.0
160	Add	7.0
180	Add	7.0
200	Add	8.0
220	Add	9.0
240	Add	9.0
260	Add	10.5
280	Add	11.5
300	Add	11.5
FLAPS DOWN		
95	Add	6.5
100	Add	6.5
110	Add	7.0
120	Add	7.0
130	Add	7.5
140	Add	7.5

Figure 3-1. Airspeed Installation Correction Tables

POWER PLANT CHART

AIRCRAFT MODEL(S)
F6F-3, F6F-3N
F6F-5, F6F-5N

PROPELLER(S)
Hamilton Standard - 3 blade Hydromatic
6501A - 0/2E50 -495

ENGINE MODEL(S)
R-2800-10
R-2800-10W

GAUGE READING	FUEL PRESS.	OIL PRESS.	OIL TEMP.	COOLANT TEMP.		OIL(1) CONS.
DESIRED MAXIMUM	17 18.5	90 100	70 100			
MINIMUM IDLING	16 7	60				

MAXIMUM PERMISSABLE DIVING RPM:
MINIMUM RECOMMENDED CRUISE RPM:
MAXIMUM RECOMMENDED TURBO RPM:
OIL GRADE: (S)1120 (W) Spec-AN-V-06
FUEL GRADE: 100/130; Spec. AN-F-28

WAR EMERGENCY (COMBAT EMERGENCY)			MILITARY POWER (NON-COMBAT EMERGENCY)			OPERATING CONDITION			NORMAL RATED (MAXIMUM CONTINUOUS)			MAXIMUM CRUISE (NORMAL OPERATION)		
5 MINUTES 260°C			30 MINUTES 260°C			TIME LIMIT MAX. CYL. HD. TEMP.			UNLIMITED 260°C			UNLIMITED		
Auto Lean 2700(4)			Auto Lean 2700			MIXTURE R. P. M.			Auto Lean 25.50			Auto Lean 2200 N 2150 L & H		
MANIF. PRESS.	SUPER-CHARGER	FUEL(2) Gal/Min	MANIF. PRESS.	SUPER-CHARGER	FUEL(2) Gal/Min	STD. TEMP. °C	PRESSURE ALTITUDE	STD. TEMP. °F	MANIF. PRESS.	SUPER-CHARGER	FUEL GPH(3)	MANIF. PRESS.	SUPER-CHARGER	FUEL GPH(3)
						-56.0	40,000 FT.	-67.0						
						-56.0	38,000 FT.	-67.0						
						-56.0	36,000 FT.	-67.0						
						-52.4	34,000 FT.	-62.3						
						-48.4	32,000 FT.	-55.1						
						-44.4	30,000 FT.	-48.0	FT	H	122	Ft	H	85
FT	H	2.3	(Use normal rated power)			-40.5	28,000 FT.	-40.9	FT	H	150	FT	H	93
FT	H	2.7	See Note (4)			-36.5	26,000 FT.	-33.7	FT	H	190	FT	L	82
FT	H	3.0				-32.5	24,000 FT.	-26.5	FT	H	180	FT	L	89
FT	H	3.2	FT	L	3.3	-28.6	22,000 FT.	-19.4	49.5	H	178	36	L	96
FT	H(4)	3.5	FT	L	3.5	-24.6	20,000 FT.	-12.3	FT	L	200	36	L	95
58	H	3.6	53	L	3.7	-20.7	18,000 FT.	-5.2	49.5	L	208	36	L	94
FT	L	3.5	53	L	3.7	-16.7	16,000 FT.	2.0	49.5	L	208	36	L	93
58	L	3.7	53	L	3.7	-12.7	14,000 FT.	9.1	49.5	L	208	36	L	91
58	L	3.6	53	L	3.7	-8.8	12,000 FT.	16.2	49.5	L	208	36	L	90
58	L	3.6	53	L	3.7	-4.8	10,000 FT.	23.4	49.5	L	208	FT	N	88
58	L	3.6	53	L	3.7	-0.8	8,000 FT.	30.5	49.5	L	206	37	N	96
58	L	3.5	53	L	3.7	3.1	6,000 FT.	37.6	FT	N	170	37	N	92
58	L	3.6	FT	N	3.4	7.1	4,000 FT.	44.7	44	N	162	37	N	88
58	L	3.6	FT	N	3.5	11.0	2,000 FT.	51.8	44	N	156	37	N	84
FT	N	3.7	54	N	3.7	15.0	SEA LEVEL	59.0	44	N	150	37	N	80

GENERAL NOTES

(1) OIL CONSUMPTION: MAXIMUM U.S. QUART PER HOUR PER ENGINE.
(2) Gal/Min: APPROXIMATE U.S. GALLON PER MINUTE PER ENGINE
(3) GPH: APPROXIMATE U.S. GALLON PER HOUR PER ENGINE.
F.T.: MEANS FULL THROTTLE OPERATION.
VALUES ARE FOR LEVEL FLIGHT WITH RAM.

FOR COMPLETE CRUISING DATA SEE APPENDIX II
NOTE: TO DETERMINE CONSUMPTION IN BRITISH IMPERIAL UNITS, MULTIPLY BY 10 THEN DIVIDE BY 12. RED FIGURES ARE PRELIMINARY SUBJECT TO REVISION AFTER FLIGHT CHECK.

TAKE-OFF CONDITIONS: 2700 RPM
54" manifold pressure. Auto Rich mixture
260°C max. cyl. head temp.

CONDITIONS TO AVOID:

SPECIAL NOTES

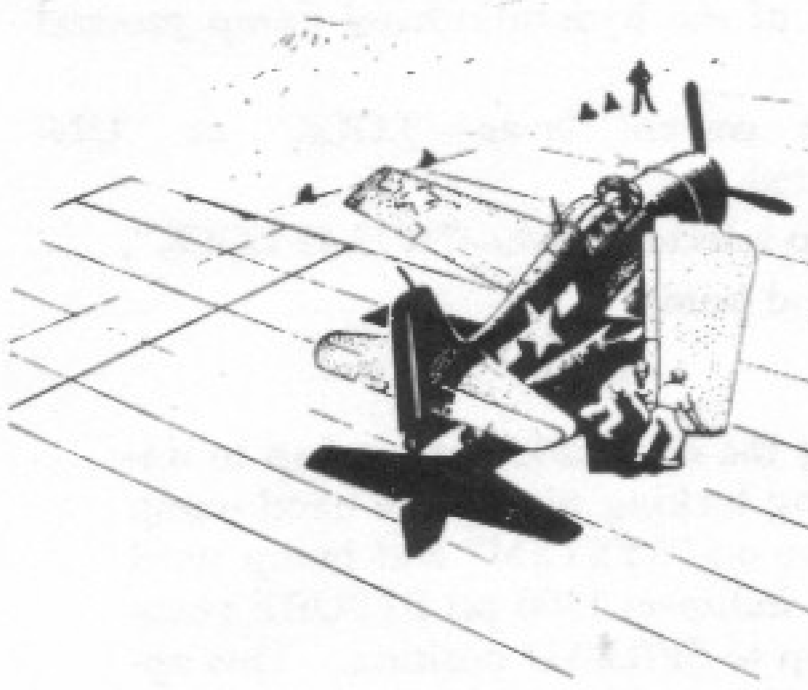
(4) Do not exceed 2550 RPM in high blower.

(5) Use Auto Rich mixture if cylinder head temperature exceeds above limits in Auto Lean.

DATA AS OF 7-11-45 BASED ON Naval Air Test Center, Flight Test.

AAFMC-326 8-1-44

Figure 3-2. Power Plant Chart



SECTION IV
EMERGENCY OPERATING INSTRUCTIONS

4-1. FIRE.

4-2. In case of fire, the controls should be operated as follows:

a. Fuel selector valve—"OFF".

b. Throttle—"OPEN" to consume fuel in carburetor.

c. Oil cooler and intercooler shutters—"OPEN".

d. Cowl flaps—"OPEN".

4-3. ENGINE FAILURE DURING FLIGHT.

4-4. Check tank selector valve, auxiliary fuel pump, and fuel quantity gages as failure may be due to an empty tank.

4-5. FORCED LANDING.

4-6. ON LAND. If it is necessary to make a forced landing on land, proceed as follows:

a. Nose down to maintain flying speed.

b. Jettison external load items—"SAFE" armament units before releasing.

c. Shoulder harness—TIGHT.

d. Cockpit hood—OPEN and STOPPED.

e. Wing flaps—"DOWN". If time permits:

f. Ignition switch—"OFF".

g. Battery switch—"OFF".

h. Mixture control—"IDLE CUT-OFF".

i. Fuel selector valve—"OFF"

j. Landing gear "UP"—if landing on soft or rough terrain.

4-7. ON WATER. The emergency procedure for landing on water is essentially the same as that for on land except that landing gear MUST be "UP"

a. Cockpit hood—OPEN and STOPPED.

b. Smooth sea—land into the wind.

c. Rough sea—land along trough of swell—across wind if necessary.

d. Make a power stall landing with the flaps full "DOWN".

4-8. EMERGENCY ESCAPE FROM AIRPLANE.

4-9. The cockpit hood is equipped with quick release latches, consisting of release pins with red finger rings attached, one on each side of the forward end of the track. To jettison the hood in flight:

a. Open hood one inch.

b. Turn red finger rings INBOARD and pull both lockpins AFT simultaneously.

c. Push forward end of hood UP into airstream and lower head to prevent injury in case hood does not clear properly.

4-10. The right and left hand side panels of the hood can be opened for emergency exit when the airplane is in an inverted position on the ground by means of the release lever (painted red) located on the center strip of the panel.

a. Pull release lever INBOARD and UP.

b. Push panel OUTBOARD.

4-11. EMERGENCY OPERATION OF ELECTRICAL SYSTEM.

4-12. HIGH VOLTAGE (over 30.0 volts). If the voltmeter reading is over 30.0 volts, turn "OFF" all switches for circuits not essential to flight. Turn "OFF" battery switch in order to prevent overcharging. Recharge the battery periodically by closing the battery switch for not more than five minutes at a time.

4-13. LOW VOLTAGE (below 26.0 volts). If the voltmeter reading is below 26.0 volts, immediately turn off all equipment requiring heavy electrical loads. Then turn the battery switch "OFF" to determine if the generator system is operating properly. If light electrical loads such as cockpit lights and instruments continue to operate, the reverse current cut-out has not opened and the generator is still supplying current to the battery. If the lights and instruments cease to operate it is an indication that the reverse current cut-out has opened and that the battery is the only source of electric power. In that case, conserve the battery by using electrically operated equipment as sparingly as possible.

4-14. ELECTRICAL FIRE. If the faulty circuit is known, turn off the proper switch. Otherwise, turn "OFF" the battery switch and all other electrical switches. Turn the electrical switches back on one at a time until the faulty circuit can be determined by a new outbreak of fire. Do not use the faulty circuit.

4-15. EMERGENCY OPERATION OF HYDRAULIC SYSTEM.

4-16. GENERAL. If there is insufficient hydraulic pressure to operate any of the various systems due to a line failure or malfunctioning of the engine-driven pump, the hydraulic hand pump should be used to supply pressure to the desired system.

4-17 LANDING GEAR. To operate the landing gear by means of the hydraulic hand pump, proceed as follows:

a. Move landing gear control "UP" or "DOWN" as desired.

b. Move hand pump selector valve control, located on the right hand cockpit shelf to "LANDING GEAR ONLY"

c. Operate hand pump through approx. 90 cycles (double strokes) to RAISE WHEELS and approx. 70 cycles (double strokes) to LOWER WHEELS.

Note

The landing gear will not LOCK DOWN above 135 knots IAS. When lowering by hand pump, considerably less effort will be required if the airspeed is reduced to 100 knots or less.

4-18. WING FLAPS. To operate the wing flaps by means of the hydraulic hand pump, proceed as follows:

a. Turn wing flap switch to "UP" or "DOWN" as desired.

b. Move hand pump selector valve to "FLAPS ONLY".

c. Operate hand pump through approx. 35 cycles (double strokes) to LOWER FLAPS, and approx. 25 cycles (double strokes) to RAISE FLAPS.

Note

The wing flaps are held down only by hydraulic pressure remaining constant—there is no other lock. In an emergency when loss of pressure is indicated in the hydraulic system, always lower the landing gear before the flaps. If the flaps are lowered first, the force of the airstream may overcome the hydraulic pressure (reduced by leaks) and force the flaps up, in which case there may not be sufficient pressure to lower them again.

4-19. GUN CHARGING. Refer to Section V.

4-20. COWL FLAPS. To operate the cowl flaps by means of the hydraulic hand pump, proceed as follows:

a. Cowl flaps control lever—"OPEN" or "CLOSED" as required.

b. Hand pump selector valve control—"ENGINE FLAP".

c. Operate hand pump.

4-21. WING LOCK. To operate the wing lock mechanism by means of the hydraulic hand pump proceed as follows:

a. Wing lock control lever—"LOCK" or "UNLOCK" as required.

b. Hand pump selector valve—"WING LOCK".

c. Operate hand pump.

Note

When using the hydraulic hand pump to unlock the wing locking pins, place hand pump selector valve on "SYSTEM" and pump until system gage indicates 1500 psi BEFORE pushing wings up to SPREAD position. This operation will charge the hydraulic accumulator which has sufficient capacity to engage the main locking pins the instant the wing reaches the SPREAD position. After pins are engaged, pump a few extra strokes to make certain pins are FULL HOME.

4-22. OIL COOLER SHUTTER. To operate the oil cooler shutter by means of the hydraulic hand pump, proceed as follows:

a. Oil cooler shutter control lever—"OPEN" or "CLOSED" as required.

b. Hand pump selector valve control—"SYSTEM".

c. Operate hand pump.

4-23. INTERCOOLER SHUTTERS. To operate the intercooler shutters by means of the hydraulic hand pump, proceed as follows:

a. Intercooler shutters control lever—"OPEN" or "CLOSED" as required.

b. Hand pump selector valve—"SYSTEM".

c. Operate hand pump.

4-24. EMERGENCY LANDING GEAR OPERATION.

4-25. GENERAL. If the landing gear fails to come down when the hydraulic control lever is operated (using either engine-driven or hand pump hydraulic pressure), the gear may be lowered and locked by means of the air bottle emergency system. The system is controlled by the emergency release "T" handle located on the lower center control panel.

Note

Do not use emergency landing gear release above 90 knots IAS.

4-26. TO LOWER LANDING GEAR. Pull "T" handle FULL DOWN and LOCK. Check landing gear indicator to make certain wheels are DOWN and LOCKED. Time required to LOWER wheels is approximately ten seconds.

4-27. When the "T" handle is pulled, the uplocks are released, the air system vent valve closes, the air bottle

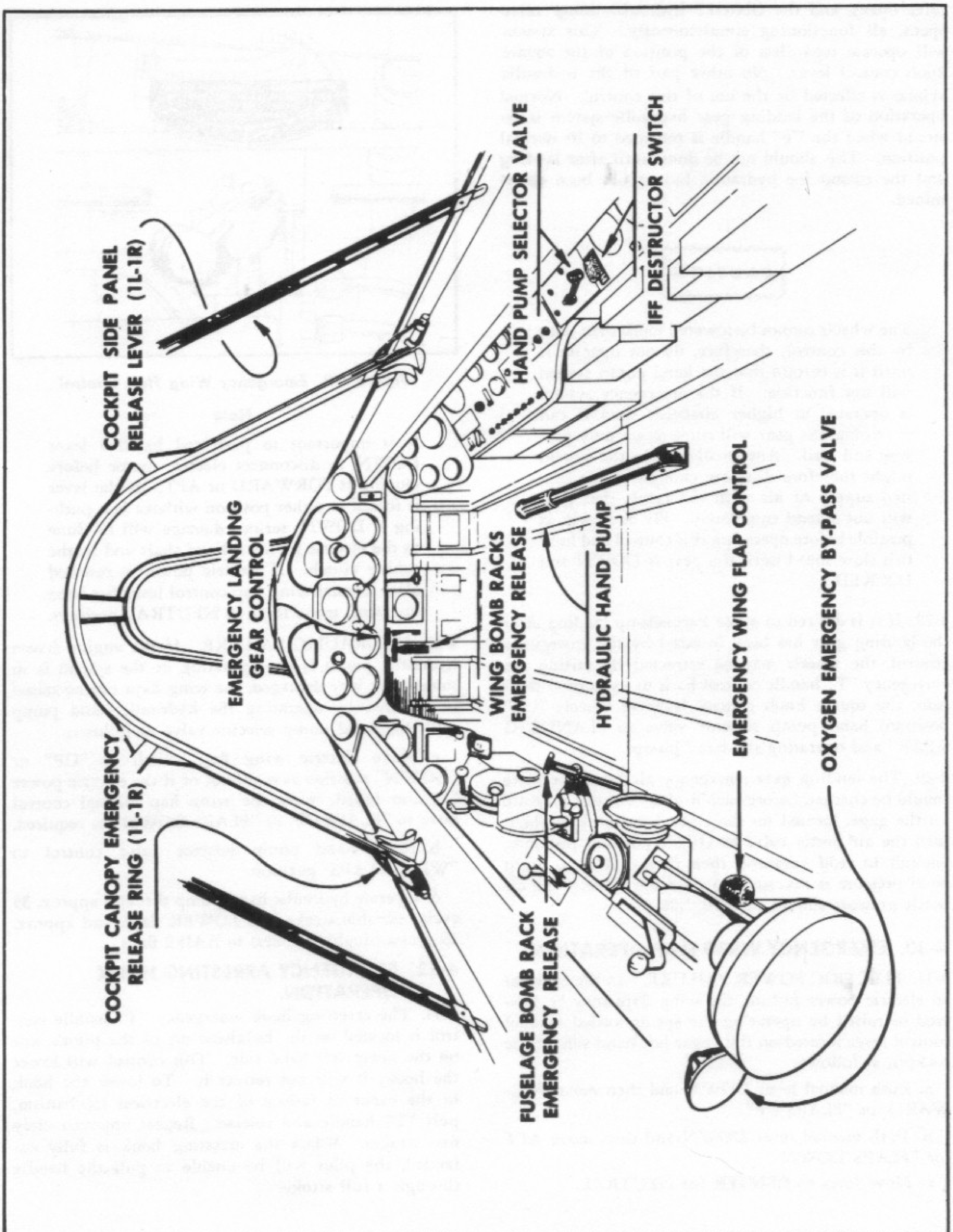


Figure 4-1. Emergency Controls Diagram

valve opens, and the ON-OFF hydraulic dump valve opens, all functioning simultaneously. This system will operate regardless of the position of the square knob control lever. No other part of the hydraulic system is affected by the use of this control. Normal operation of the landing gear hydraulic system is restored when the "T" handle is returned to its normal position. This should not be done until after landing and the reason for hydraulic failure has been determined.

CAUTION

The wheels cannot be lowered more than once by this control; therefore, do not operate it until it is certain that the hand pump system will not function. If the emergency system is operated at higher airspeeds than it can overcome, the gear will come down only part way and trail. Any small leak in the system might therefore dissipate enough of the limited supply of air with the result that gear will not extend completely. Fly as slowly as possible before operating this control and hold this slow speed until the gear is DOWN and LOCKED.

4-28. If it is desired to make a wheels-up landing after the landing gear has been lowered by the emergency control, the wheels may be retracted by raising the emergency "T" handle control back to its normal position, the square knob control lever to wheels "UP" position, hand pump selector valve to "LANDING GEAR" and operating the hand pump.

4-29. The landing gear emergency air bottle pressure should be checked before each flight. To read pressure on the gage, located on the right hand cockpit shelf, turn the air bottle valve to OPEN position just long enough to read pressure, then close. Only a slight hand pressure is necessary to close the valve. The air bottle pressure should be 1850 ±50 psi.

4-30. EMERGENCY WING FLAP OPERATION.

4-31. ELECTRIC POWER FAILURE. In the event of an electric power failure, the wing flaps may be lowered or raised by operating the spring-loaded manual control lever located on the lower left hand side of the cockpit, as follows:

a. Push manual lever DOWN and then move FORWARD for "FLAPS UP".

b. Push manual lever DOWN and then move AFT for "FLAPS DOWN".

c. Move lever to CENTER for NEUTRAL.

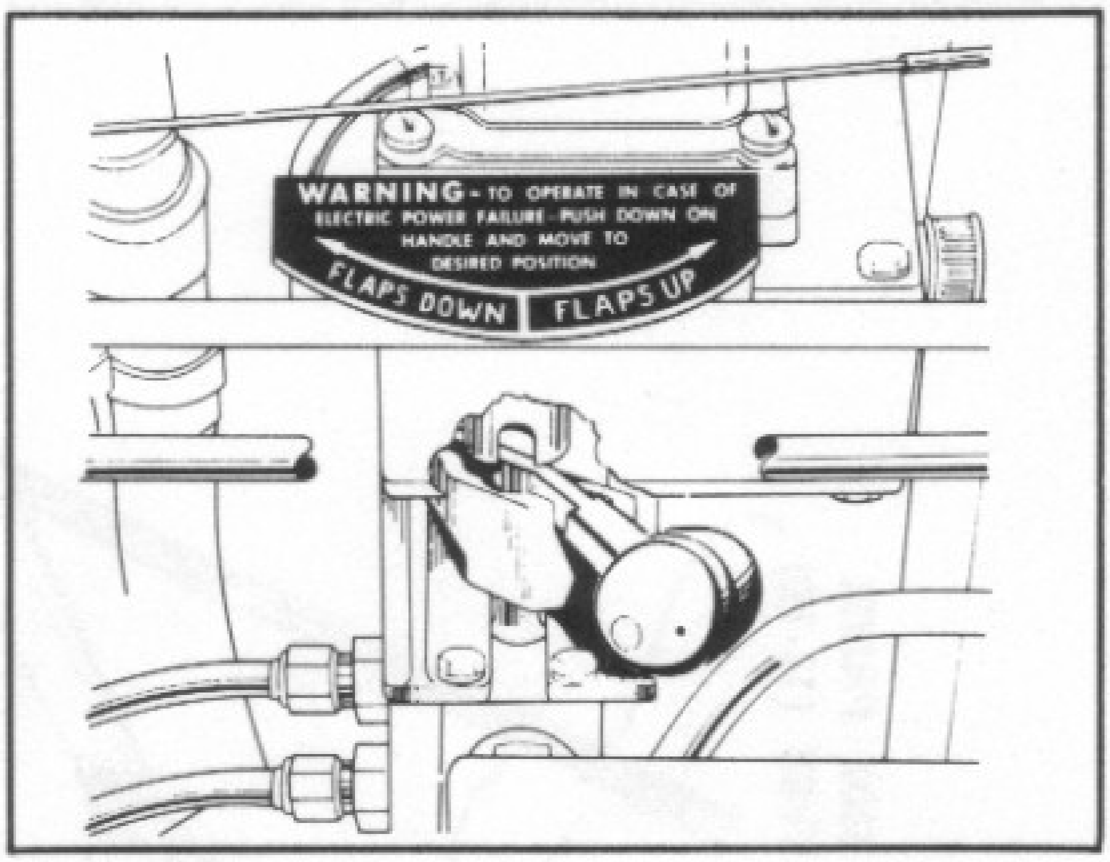


Figure 4-2. Emergency Wing Flap Control

Note

It is important to push red handled lever DOWN to disconnect electric motor before moving it FORWARD or AFT. If the lever is forced to either position without first pushing it DOWN, serious damage will be done to the electric motor splined shaft and to the rubber clutch. If electric power is restored after manual wing flap control lever has been operated, move lever to NEUTRAL position.

4-32. HYDRAULIC FAILURE. If the engine-driven hydraulic pump is not operating or the system is in some other way damaged, the wing flaps can be raised or lowered by operating the hydraulic hand pump with the hand pump selector valve as follows:

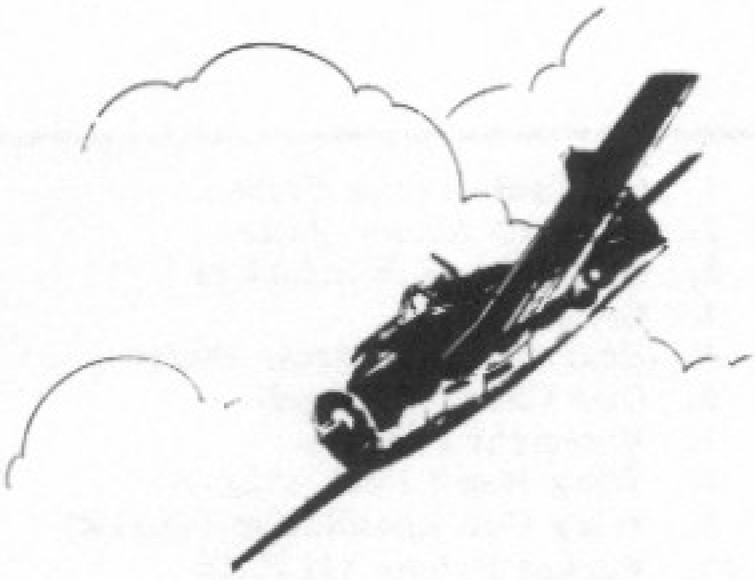
a. Move electric wing flap switch to "UP" or "DOWN" position as required; or if the electric power has also failed, move the wing flap manual control lever to "FLAPS UP' or "FLAPS DOWN" as required.

b. Move hand pump selector valve control to "WING FLAPS" position.

c. Operate hydraulic hand pump through approx. 35 cycles (double strokes) to LOWER flaps, and approx. 25 cycles (double strokes) to RAISE flaps.

4-33. EMERGENCY ARRESTING HOOK OPERATION.

4-34. The arresting hook emergency "T" handle control is located on the bulkhead aft of the pilot's seat on the lower left hand side. This control will lower the hook; it will not retract it. To lower the hook in the event of failure of the electrical mechanism, pull "T" handle and release. Repeat approximately five strokes. When the arresting hook is fully extended, the pilot will be unable to pull the handle through a full stroke.



SECTION V
OPERATING EQUIPMENT

5-1. ARMAMENT

5-2. WING GUNS. The F6F-3 and -3N airplanes are equipped with six .50 cal machine guns installed in the outer panels, three left and three right, with a maximum of 2400 rounds of ammunition.

5-3. The F6F-5 and -5N airplanes are designed to accomodate either six .50 cal machine guns or a mixed battery in which a 20 mm cannon replaces the inboard .50 cal machine gun, one left and one right. All late model F6F-5N airplanes are provided with mixed batteries. Three ammunition boxes, for each gun, are installed outboard of the guns through hinged, folding doors in the wing upper surface.

Note

Airplanes equipped with mixed batteries will carry a maximum of 1800 rounds of .50 cal ammunition and 250 rounds of 20 mm ammunition.

5-4. The guns are charged hydraulically by the two control handles located on the lower center control panel and fired electrically by a switch installed on the control stick grip. The armament master switch and the gun selector switches are located on the armament control panel on the right hand side of the cockpit. Provision is made for the installation of gun heaters.

Note

The gun heater circuit, connected directly to the generator through circuit breakers, will be energized only when the generator is operating. The battery will not energize the gun heaters. If it is desired to prevent the heaters from operating when the engine is running, the plug at the heater must be disconnected.

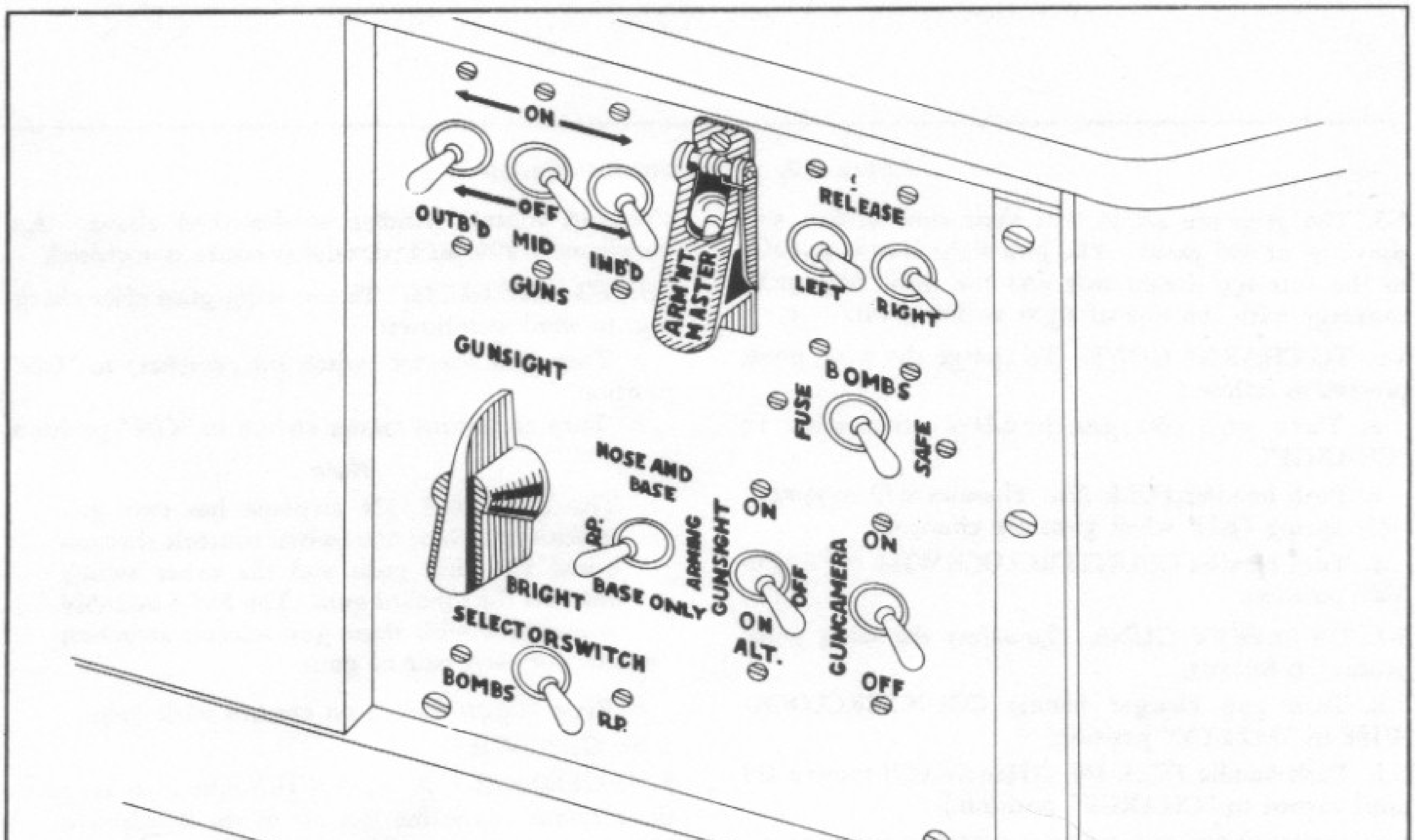


Figure 5-1. Armament Control Panel

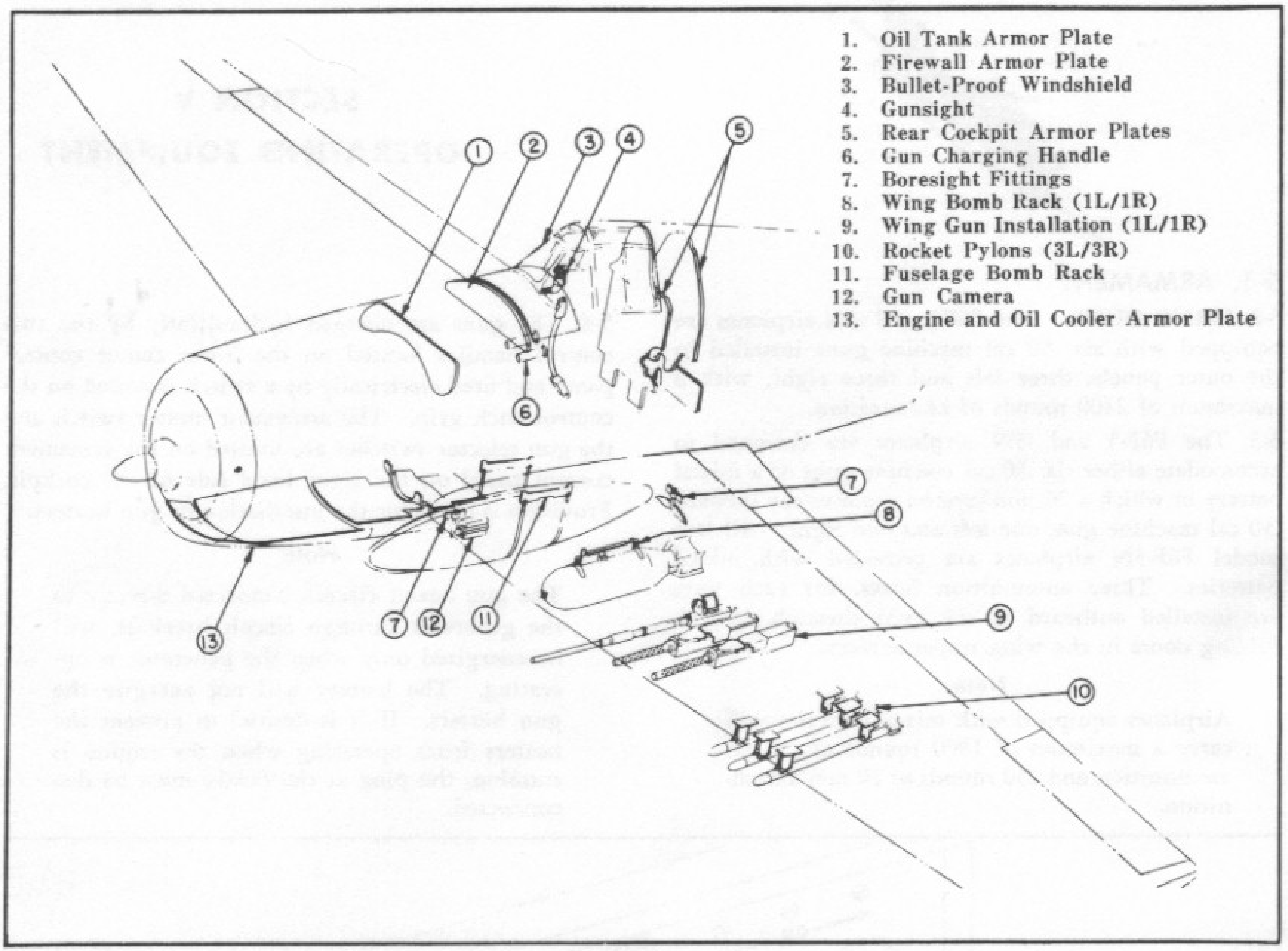


Figure 5-2. Armament Installation

5-5. The guns are set so that their cone of fire will converge at 300 yards. The gun sight line is parallel to the fuselage datum line and the guns fire up to converge with the line of sight at 300 yards.

5-6. TO CHARGE GUNS. To charge the wing guns, proceed as follows:

a. Turn gun charger handles clockwise to "CHARGE".

b. Push handles FULL IN. Handles will automatically spring OUT when guns are charged.

c. Turn handles COUNTERCLOCKWISE to "SAFETY" position.

5-7. TO SAFETY GUNS. To safety the wing guns, proceed as follows:

a. Turn gun charger handle COUNTERCLOCKWISE to "SAFETY" position.

b. Push handle FULL IN. (Handle will remain IN until turned to "CHARGE" position.)

5-8. AUXILIARY CHARGING OPERATION. If the engine-driven hydraulic pump is not operating, the guns may be charged by setting the hand pump selector valve control at "GUN CHARGING" and operating the hand pump while operating the gun charging control handles as described above. Approximately 800 psi hydraulic pressure is required.

5-9. TO FIRE GUNS. To fire wing guns after charging, proceed as follows:

a. Turn gun selector switch (or switches) to "ON" position.

b. Turn armament master switch to "ON" position.

Note

The F6F-3 and -3N airplane has two gun selector switches; one switch controls the outboard and mid guns and the other switch controls the inboard gun. The F6F-5 and -5N is provided with three gun selector switches; one for each pair of guns.

c. Press trigger switch on control stick grip.

5-10. GUNSIGHT.

5-11. GENERAL. A Mark 8 gunsight is mounted on the airplane centerline just aft of the windshield. A two filament lamp within the gunsight illuminates and projects the deflection rings and cross hairs on the bullet-resistant glass of the windshield. A spare lamp is clipped to the gunsight mount and may be installed in flight.

5-12. OPERATION. The two filament lamp is controlled by a switch which may be placed in "ON" or "ALTERNATE" positions enabling the pilot to switch to the second filament if one burns out in flight. The intensity of the lamp is controlled by a rheostat. Both switch and rheostat are located on the armament panel.

5-13. BOMBING EQUIPMENT.

5-14. GENERAL. A MK 51-7 bomb rack is installed on each side of the wing center section. A 1000 lb bomb can be carried on each wing rack. Shackle selector, bomb selector and arming switches are located on the armament control panel. The bombs release switch button is installed on the top of the control stick grip.

5-15. Late model airplanes are equipped with emergency manual release handles for wing bombs. The control handles are located at the upper left hand side of the lower center control panel.

5-16. TO RELEASE BOMBS. To release bombs from wing bomb rack, proceed as follows:

a. BOMBS—RP selector switch—"BOMBS".

b. Switch for bomb or bombs to be released—"ON".

c. Fusing switch—"FUSE".

d. Armament master switch—"ON".

e. Press bomb release button on control stick grip.

f. For emergency release, pull AFT on manual release handle.

5-17. ROCKETS.

5-18. GENERAL. Late model airplanes are equipped with six MK V rocket launchers installed on the underside of the wing outer panel, three left and three right. The rockets are controlled by the selector switch on the armament control panel and a station distributor installed on the lower center control panel.

5-19. OPERATION — MK 1 STATION DISTRIBUTOR. Airplanes Bu. No. 94449 and subsequent, are equipped with MK 1 station distributors; to fire rockets proceed as follows:

a. BOMBS RP selector switch—"RP".

b. Rockets are set for impact burst by moving RP arming switch, located on armament control panel, to "BASE and NOSE", and for delayed burst by moving arming switch to "BASE ONLY".

c. The number of rockets to be fired and the sequence is controlled by the MK 1 station distributor, located on the lower center control panel. To fire rockets in pairs, from outboard to inboard, rotate distributor handle counterclockwise to "2-2-2". To fire the outboard rockets in pairs and then the mid and inboard rockets together, rotate distributor handle to "2-4".

d. Armament master switch—"ON".

e. Depress bomb release button on control stick grip.

WARNING

Do NOT push SALVO PRESET as the instantaneous firing of all rockets places an excessive strain on wings. When this unit is altered so that a ripple salvo can be fired, instructions will be issued.

5-20. OPERATION—MK 3 STATION DISTRIBUTOR. Airplanes up to and including Bu. No. 94448 are equipped with MK 3 station distributors. To fire rockets, proceed as follows:

a. BOMBS—RP selector switch—"RP".

b. Station distributor ON-OFF switch—"ON".

c. Arming switch—"ARM".

d. To fire all rockets in SALVO, set SINGLE-AUTO switch to "AUTO". To fire rockets in pairs, set SINGLE-AUTO switch to "SINGLE" and rotate a selector knob to desired position—"1" is outboard pair, "2" is mid pair, and "3" is inboard pair.

e. Armament master switch—"ON".

f. Depress bomb release button on control stick grip.

5-21. GUN CAMERA. An AN-N4 gun camera is installed in the leading edge of the left wing inboard the guns. The gun camera control switch is located on the armament control panel. To operate gun camera, proceed as follows:

a. Gun camera switch—"ON".

b. Armament master switch—"ON".

c. Depress either bomb release button or gun trigger on stick grip.

5-22. OXYGEN SYSTEM.

5-23. CYLINDER AND CONTROL. A standard 514 cu in. capacity shatter-proof cylinder is installed in the fuselage aft of cockpit rear bulkhead. Normal cylinder pressure is 1800 psi. The shut-off valve handwheel, connected to the cylinder, projects through cockpit rear bulkhead to the left of the pilot's seat. Turn handwheel COUNTERCLOCKWISE to "OPEN".

5-24. REGULATOR. A diluter-demand type regulator, together with a cylinder pressure gage is mounted to the left of the pilot's seat adjacent to the cylinder shut-off valve handwheel. An oxygen flow indicator is installed just below the landing gear control.

5-25. The diluter-demand regulator is designed to meet the demands of the inhalation phase of the breathing cycle and deliver either a properly proportioned mixture of air and oxygen or 100% oxygen dependent upon the setting of the adjustable air-valve lever. With the air-valve set to the "ON" (normal oxygen) position, air is drawn into the breathing system and is automatically mixed with oxygen from the supply cylinder to give the total needed oxygen required up to approximately 30,000 ft beyond, which 100% cylinder oxygen is delivered. With the air-valve set to the "OFF" (100% oxygen) position, 100% oxygen is delivered at all altitudes. With the air-valve of the diluter-demand regulator set to the "ON" (normal oxygen) position, a relatively small inhalation suction (one inch of water suction) is sufficient to deliver a flow of 150 liters of oxygen per minute. This characteristic assures the user an adequate oxygen flow and ease of breathing.

5-26. The regulator is attached directly to the high pressure oxygen supply through 3/16 inch OD copper tubing connected to the cylinder; the pressure in the cylinder may decrease from 1800 or 2000 psi to 50 psi without effecting the normal operation of the regulator.

5-27. PREFLIGHT CHECK LIST. The following items should be checked while the plane is on the ground prior to flight in which oxygen is to be used, or is likely to be used, to assure proper functioning of the oxygen system:

a. Emergency valve—"CLOSED".

b. Open cylinder valve and allow at least ten seconds for pressure in line to equalize. Pressure gage should read 1800 ±50 psi, if the cylinder is fully charged.

c. Close cylinder valve. After a few minutes, observe pressure gage and simultaneously open cylinder valve. If gage pointer jumps—leakage is indicated.

d. If leakage was found by paragraph (3) above—test further. Open cylinder valve, carefully noting pressure gage reading—then close cylinder valve. If gage pointer drops more than 100 psi in five minutes there is excessive leakage, and such an oxygen system must be repaired prior to use.

e. Check mask fit by placing thumb over end of mask tube and inhale lightly. If there is no leakage, mask will adhere tightly to face due to suction created. If mask leaks—tighten mask suspension straps and/or adjust nose wire.

CAUTION

DO NOT USE A MASK THAT LEAKS.

f. Couple mask securely to breathing tube by means of quick disconnect coupling. IMPORTANT: Mating parts of coupling must be fully engaged, not "COCKED".

g. Open cylinder valve. Depress diaphragm knob through hole in center of regulator case, and feel flow of oxygen into the mask—then release diaphragm knob. Breathe several times observing oxygen flow indicator (if installed) for "blink" verifying the positive flow of oxygen.

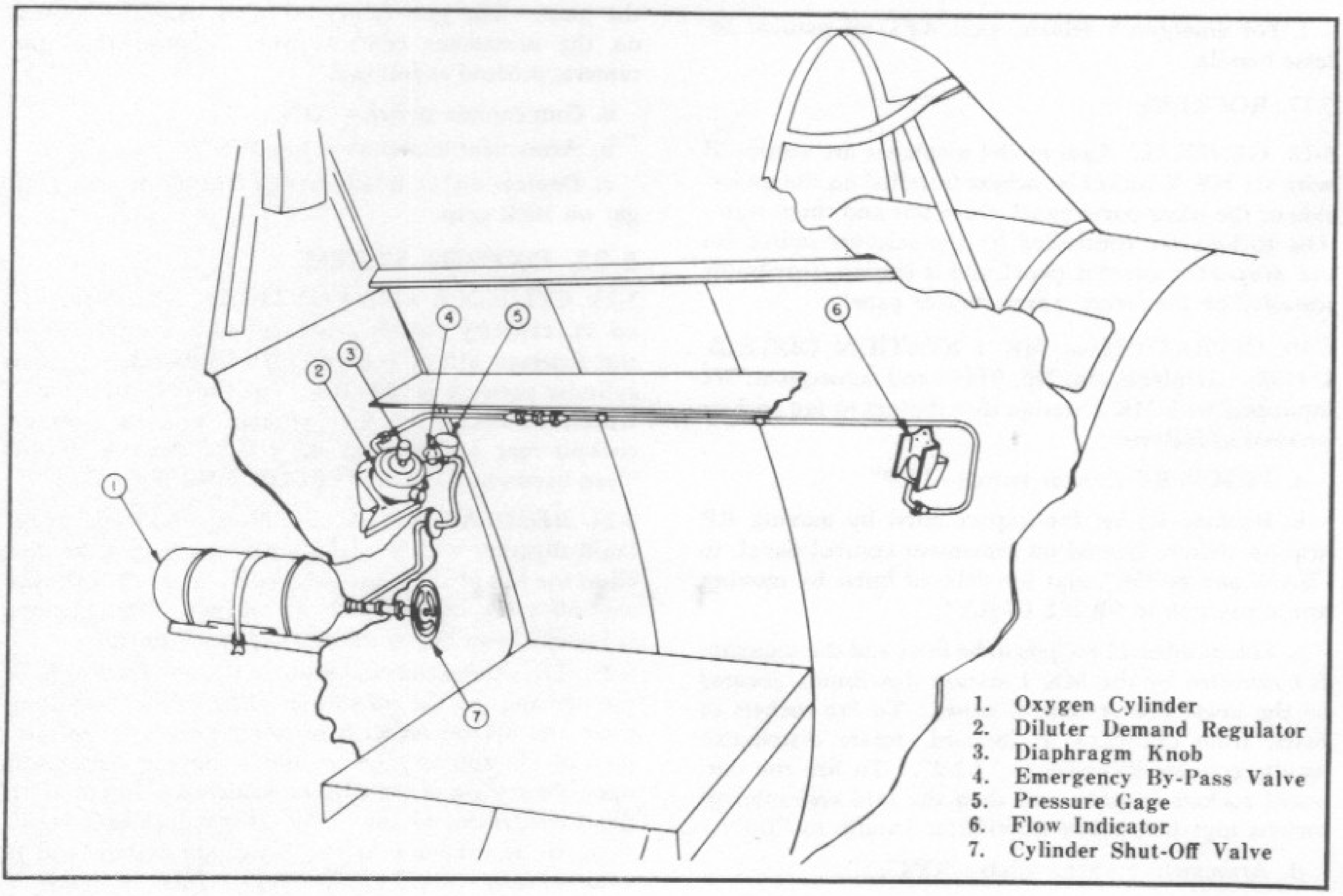


Figure 5-3. Oxygen System Installation

Note

Since the amount of added oxygen is very small at sea level, the oxygen flow meter may not operate while plane is on the ground. In this case turn air-valve to "OFF" (100% oxygen) and test again. If oxygen flow indicator operation is now satisfactory, reset air-valve to "ON" in which setting adequate oxygen flow and "blinker" operation will be assured at oxygen-use altitudes.

h. Check emergency valve by turning counterclockwise slowly until oxygen flows vigorously into mask—then close emergency valve.

i. Upon completion of oxygen flight—close cylinder valve.

5-28. OPERATING INSTRUCTIONS.

a. Open oxygen cylinder valve. Pressure gage should read 1800 ±50 psi, if cylinder is fully charged.

b. Set air-valve to "ON" (normal oxygen) position—except when the presence of excessive carbon monoxide is suspected—then set to "OFF" position.

c. Put on oxygen mask. Be sure that quick disconnect coupling is fully engaged.

d. Check mask fit by squeezing mask tube and inhaling lightly. Mask will adhere tightly to face due to suction, if there is no leakage. If mask leaks, tighten mask suspension straps.

CAUTION

Never check mask fit by squeezing mask tube while emergency valve is ON.

e. Breathe normally and observe oxygen flow indicator (if installed) for "blink", verifying positive flow of oxygen.

f. Frequently check cylinder pressure gage for state of available oxygen supply, and oxygen flow indicator for flow of oxygen to mask.

g. Upon completion of oxygen flight—close cylinder valve by rotating handle CLOCKWISE.

CAUTION

Keep oxygen equipment free from oil, grease and easily oxidized materials.

5-29. COMMUNICATING EQUIPMENT.

WARNING

The operation of this communication equipment involves the use of high voltages which are dangerous to life. Operating personnel must at all times observe all safety regulations. Do not change tubes or make adjustments inside the equipment with the high voltage supply turned on. Always shut down dynamotors or other associated power equipment, and open the main switch in the power supply circuit. Under certain conditions dangerous potentials may exist in circuits with the power controls in the off position due to charges retained by the capacitors. To avoid casualties, always discharge and ground the circuits prior to touching them. To avoid the possibility of fire, no installed radio transmitter should be tested or operated in any way with the dynamotor running until all parts of the antenna system are at least one foot removed from any object other than the airplane itself.

5-30. GENERAL. The communication system is divided into four groups: radio equipment, IFF equipment, radio altimeter and radar equipment. Radio altimeters and radar equipment are installed only in NIGHT FIGHTER airplanes F6F-3N and F6F-5N. The installation of the units follows conventional practice with control units within easy reach of the pilot on the right hand side of the cockpit, and in general, all transmitters and receivers located in the fuselage behind the cockpit.

5-31. RADIO EQUIPMENT.

Note

Per Service Change No. 82 now in effect, all radio equipment in F6F-3, -3N, -5, and -5N will include only AN/ARC-5, AN/ARC-1 and AN/ARR-2 equipment.

5-32. The AN/ARC-5 hf and AN/ARC-1 vhf radio equipment is used primarily for airplane-to-airplane or airplane-to-carrier communication. The AN/ARC-5 equipment consists of an mf transmitter and a range receiver. The AN/ARC-1 equipment consists of a 10 channel transmitter-receiver. The AN/ARR-2 receiver and the lf radio range receiver are used for navigation. Three of the following receivers (hf and lf) may be installed, depending upon the mission of the airplane: hf communication receiver, navigation receiver, and radio range receiver. The radio range receiver and its associated control are used primarily for ferrying purposes.

5-33. The following radio controls are installed in the cockpit (see figure 5-4):

a. RADIO MASTER SWITCH. Located on top of electrical control panel.

b. MASK MICROPHONE "PRESS-TO-TALK" SWITCH. Located on throttle handle.

c. HAND MICROPHONE "PRESS-TO-TALK" SWITCH. Located on hand microphone which is stowed on right hand cockpit shelf.

d. C-38/ARC-5 RECEIVER CONTROL UNIT. Located on right hand side of cockpit.

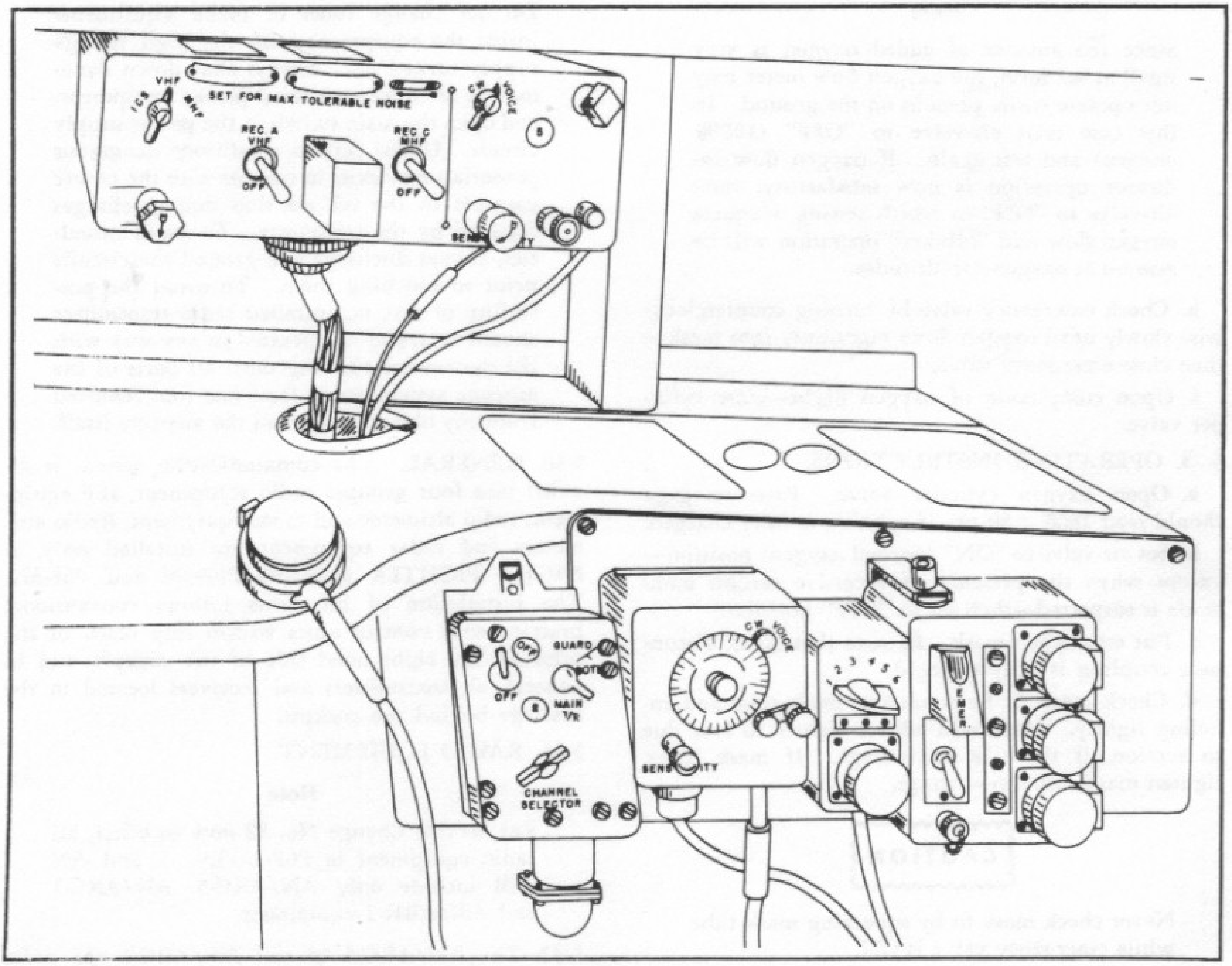


Figure 5-4. Radio Controls

e. C-26/ARC-5 RADIO RANGE CONTROL. Located on right hand side of cockpit.

f. C-45/ARC-1 TRANSMITTER CONTROL. Located on right hand side of cockpit.

5-34. OPERATION OF RADIO EQUIPMENT.

5-35. With the battery switch in the "ON" position, the engine running and the generator charging, turn ON the radio master switch located on the MASTER control unit. Set volume control to maximum and plug the headset into the pilot's jack box located on the side of the right hand panel. Plug either the hand or mask microphone into the jack box.

5-36. TRANSMISSION.

a. The position of the microphone selector on the receiver control unit (C-38/ARC-5) determines which component is used for transmission. The stop on the selector switch is arranged to permit turning to vhf and mhf only.

b. On the receiver control unit (C-38/ARC-5), throw toggle switch REC. C (mhf) to "OFF", and turn the navigation SENSITIVITY control to minimum; on the radio range control unit (C-26/ARC-5) turn SENSITIVITY to minimum.

c. For mhf, set microphone selector switch to "MHF", press microphone switch and talk.

d. For vhf, set microphone selector switch to 'VHF", set rotary switches on the transmitter control unit (C-45/ARC-1) as desired. Press microphone switch and talk.

e. The transmitter control unit (C-45/ARC-1) control switches are operated as follows: for transmission and reception any one of nine main channels use position "MAIN T/R"; for transmission and reception on any main channel and simultaneous reception on the GUARD channel use position "BOTH", for transmission and reception on the GUARD channel use position "GUARD".

5-37. RECEPTION.

a. For vhf, turn toggle switch REC. A on the receiver control unit (C-38/ARC-5) to the "UP" position and operate the transmitter control unit (C-45/ARC-1) as described under transmission.

Note

When the airplane is in a location relatively free from electrical noise and the engine is not running, no noise should be heard from the vhf receiver unless a signal is present. A noticeable hiss would indicate that the sensitivity is set too high. When the engine is running, a strong smooth hiss would indicate that the sensitivity is set too high. If ignition noise is present, it is an indication that the engine is producing excessive noise or the sensitivity is set too high. (See Navy Engine Bulletin No. 113 and model AN/ARC-1 Maintenance Instructions AN-8-30 ARC 1-3.

b. For mhf, turn toggle switch REC. C on the receiver control unit (C-38/ARC-5) to the "UP" position. Set for maximum tolerable noise. (This should be done with receiver A (vhf) OFF and navigation SENSITIVITY at a minimum. Both of these may be turned on again after the maximum tolerable noise is adjusted.)

c. For simultaneous reception on vhf & mhf, turn switches REC. A and REC. C to the "UP" position.

5-38. TO RECEIVE NAVIGATION SIGNALS.

a. On the receiver control unit (C-38/ARC-5) control turn switches REC. A and REC. C to the "OFF" position.

b. Operate the crank unit until the assigned channel number appears in the window.

c. Set NAV-VOICE selector switch to NAV. Set the OUTPUT control to produce a usably weak signal, or if the desired signal cannot be heard, to a fairly strong background hiss. VOLUME control should be left at maximum when using the AN/ARR-2 receiver.

d. Adjust the BEAT-NOTE control to produce a pleasing audible tone.

e. Readjust SENSITIVITY control to produce a usably weak signal. (If the signal is too strong, a clear-cut indication cannot be obtained.)

f. Turn the SENSITIVITY to a minimum.

5-39. TO RECEIVE RANGE SIGNALS.

Note

The following applies if a range receiver rather than a lock-tuned hf receiver is installed.

a On the range receiver control unit (C-26/ARC-5), make sure the rotary switch is set on "VOICE".

b. Advance the SENSITIVITY control until normal background noise is heard.

c. Tune in the desired radio range station and readjust the SENSITIVITY control for normal operation.

d. Rotate the SENSITIVITY control counterclockwise to a minimum.

5-40. RADIO CHECK-OFF LIST. Perform the following operations.

5-41. BEFORE TAKE-OFF.

a. Plug in headset and microphone.

b. Battery switch—"ON".

c. Radio master switch—"ON".

d. Test vhf receiver.

e. Perform operations applicable to the mission of the airplane—test navigation receiver—test range receiver.

f. Set controls for simultaneous reception of communication and navigation receivers.

g. Select desired transmission channel, and if activity instructions permit, make test transmission with base station.

5-42. AFTER LANDING.

a. Radio master switch—"OFF".

b. Range receiver controls—"OFF".

c. Battery switch—"OFF".

5-43. OPERATING NOTES AND PRECAUTIONS. Observe the following operating notes and precautions:

a. Observe radio silence regulations.

b. Reliable operation of vhf equipment is generally confined to approximately line-of-sight distance as determined by the height of the transmitting and receiving antennae, but since transmission at these frequencies is affected by meteorological conditions, large deviations from the line-of-sight distance may occur in certain areas.

5-44. IFF EQUIPMENT.

5-45. AN/APX-1 equipment consisting of a transmitter-receiver unit, a pilot's control unit located on the right hand side of the cockpit, and an impact switch is installed in all F6F-5 and late model F6F-3 airplanes. AN/APX-2 equipment is installed in all night fighter airplanes—F6F-5N and F6F-3N.

5-46. RADIO ALTIMETER (NIGHT FIGHTER ONLY).

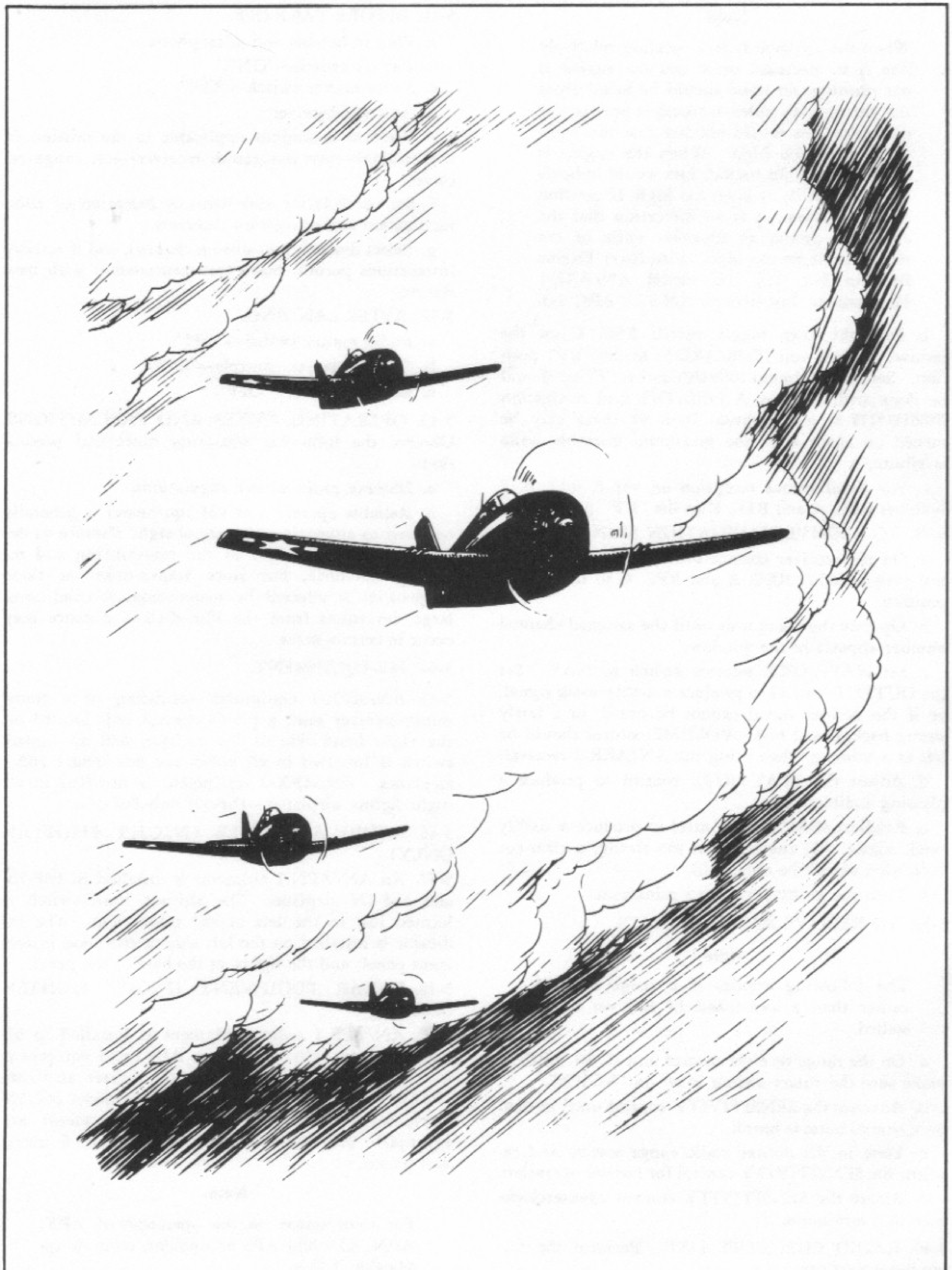
5-47. An AN/APN-1 altimeter is installed in F6F-5N and F6F-3N airplanes. The altitude limit switch is located just to the left of the pilot's seat. The indicator is installed on the left side of the main instrument panel; and the lights, at the base of the panel.

5-48. RADAR EQUIPMENT (NIGHT FIGHTER ONLY).

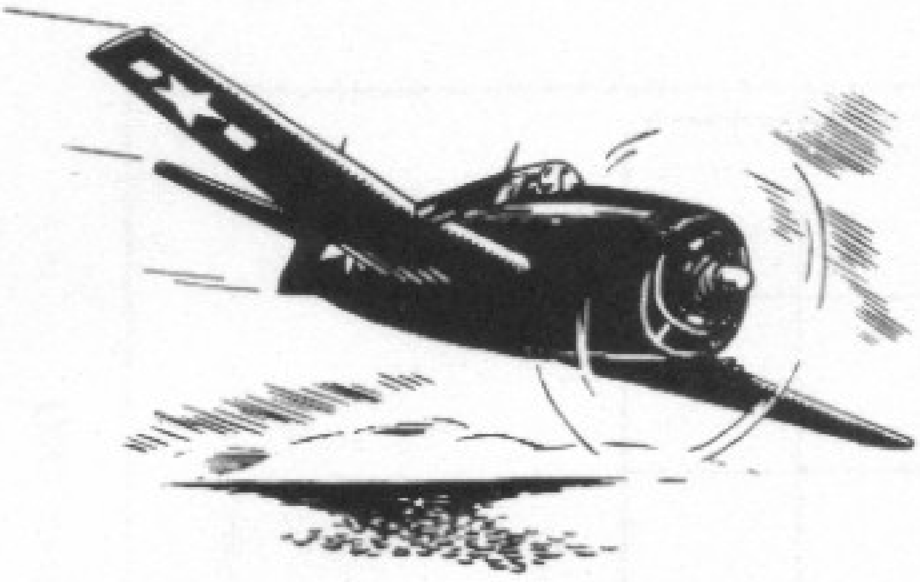
5-49. AN/APS-4 radar equipment is installed in all F6F-3N night fighter airplanes. AIA radar equipment is installed in early F6F-5N night fighter airplanes (BuAer No. 58004-No. 58989), and late model F6F-5N airplanes (BuAer No. 70038 and subsequent) are equipped with AN/APS-6A or AN/APS-6 radar equipment.

Note

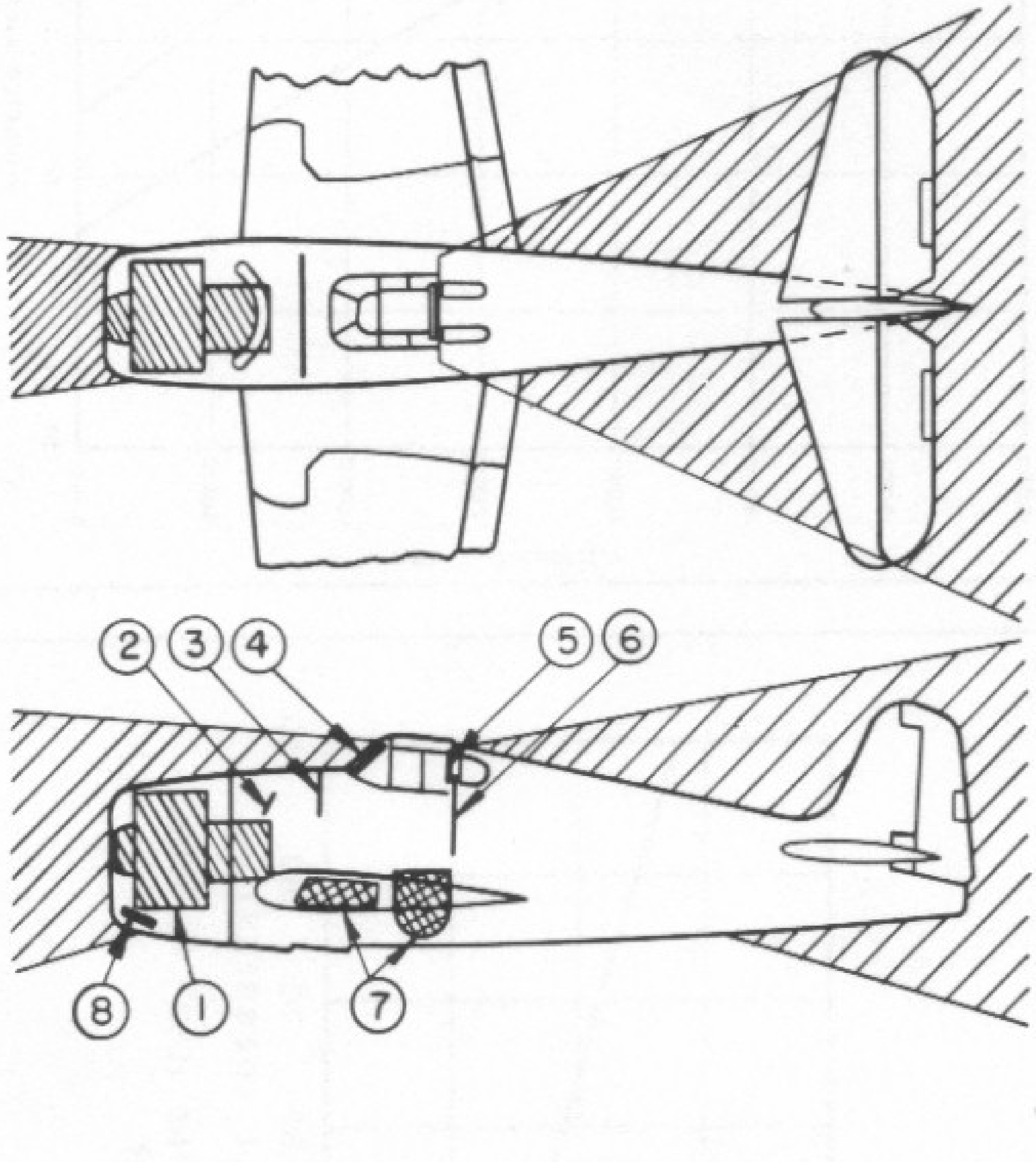
For information on the operation of APX, APN, AIA and APS equipment, refer to applicable T.O.'s.



Appendix I of this publication shall not be carried in aircraft on combat missions or when there is a reasonable chance of its falling into the hands of the enemy.



APPENDIX



1. Engine
2. Homogeneous Armor Plate—Oil Tank
3. Homogeneous Armor Plate
4. Bullet Resistant Glass
5. Face Hardened Armor Plate—Head
6. Homogeneous Armor Plate
7. Fuel Tanks—Secondary Protection
8. Homogeneous Armor Plate Oil Cooler

Figure A-1. Protection from Gunfire Diagram

Appendix I of this publication shall not be carried in aircraft on combat missions or when there is a reasonable chance of its falling into the hands of the enemy.

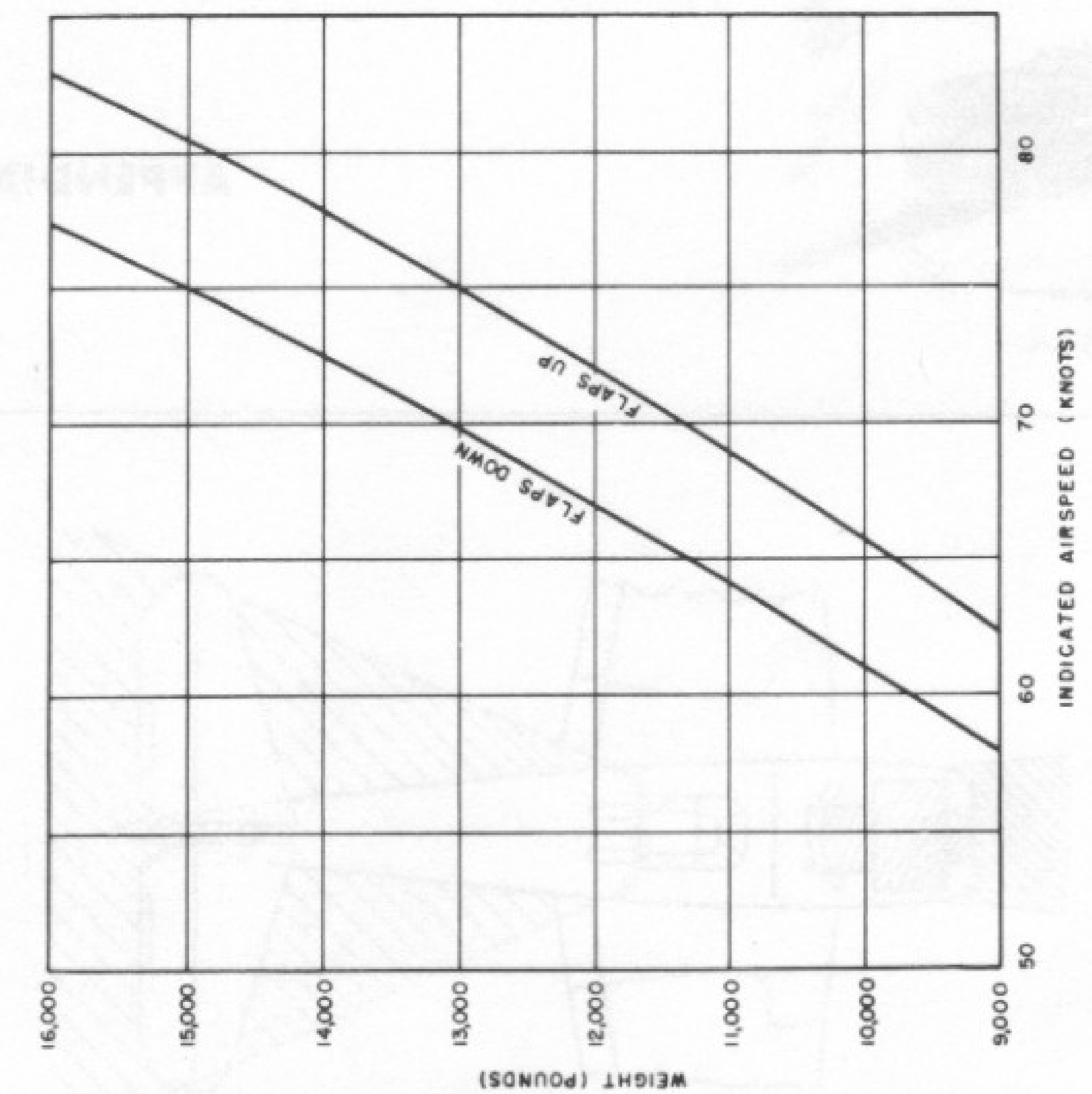


Figure A-3. Stalling Speed (Power Off)

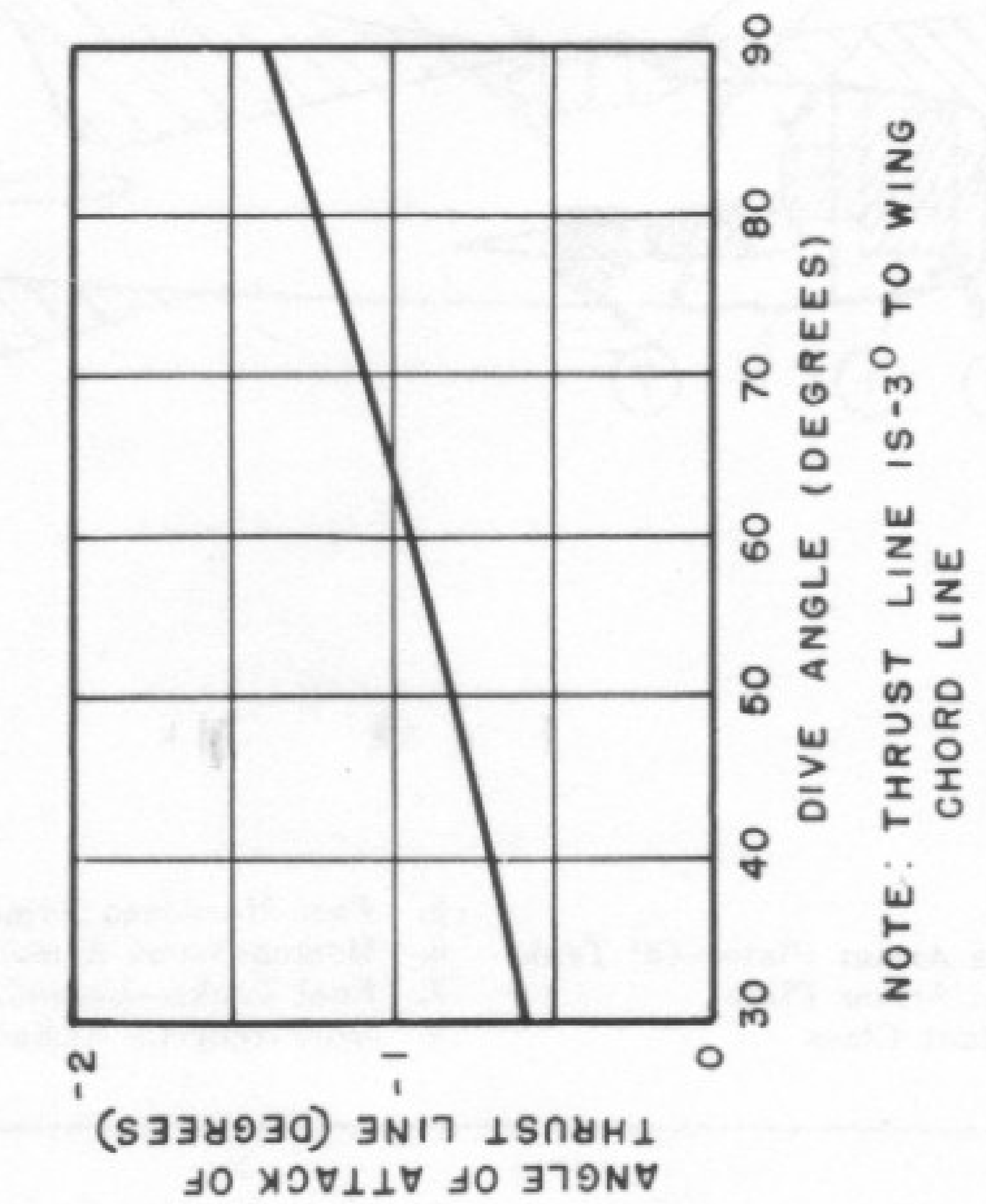


Figure A-2. Dive Angle vs. Angle of Attack of Thrust Line

Appendix I of this publication shall not be carried in aircraft on combat missions or when there is a reasonable chance of its falling into the hands of the enemy.

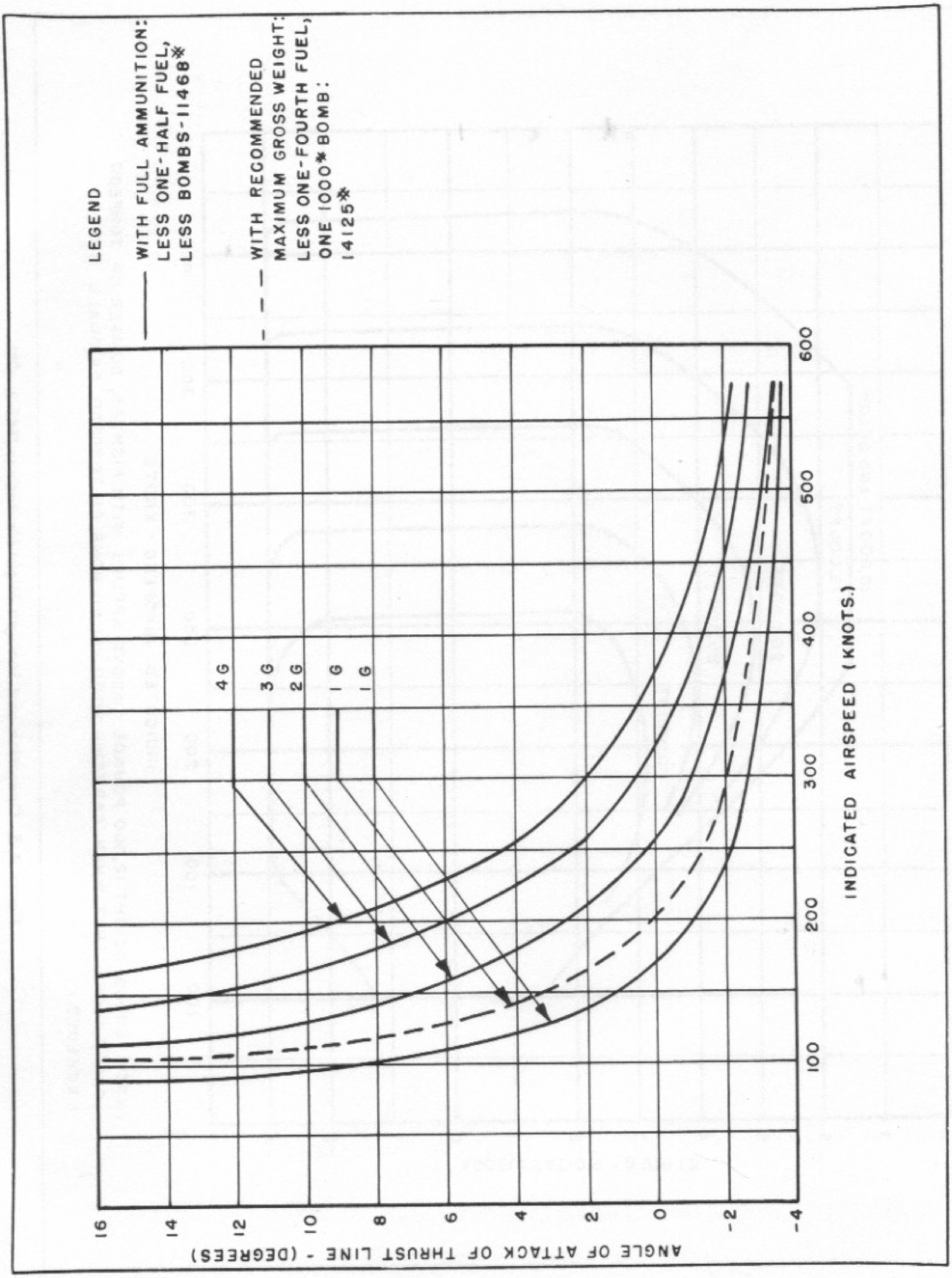
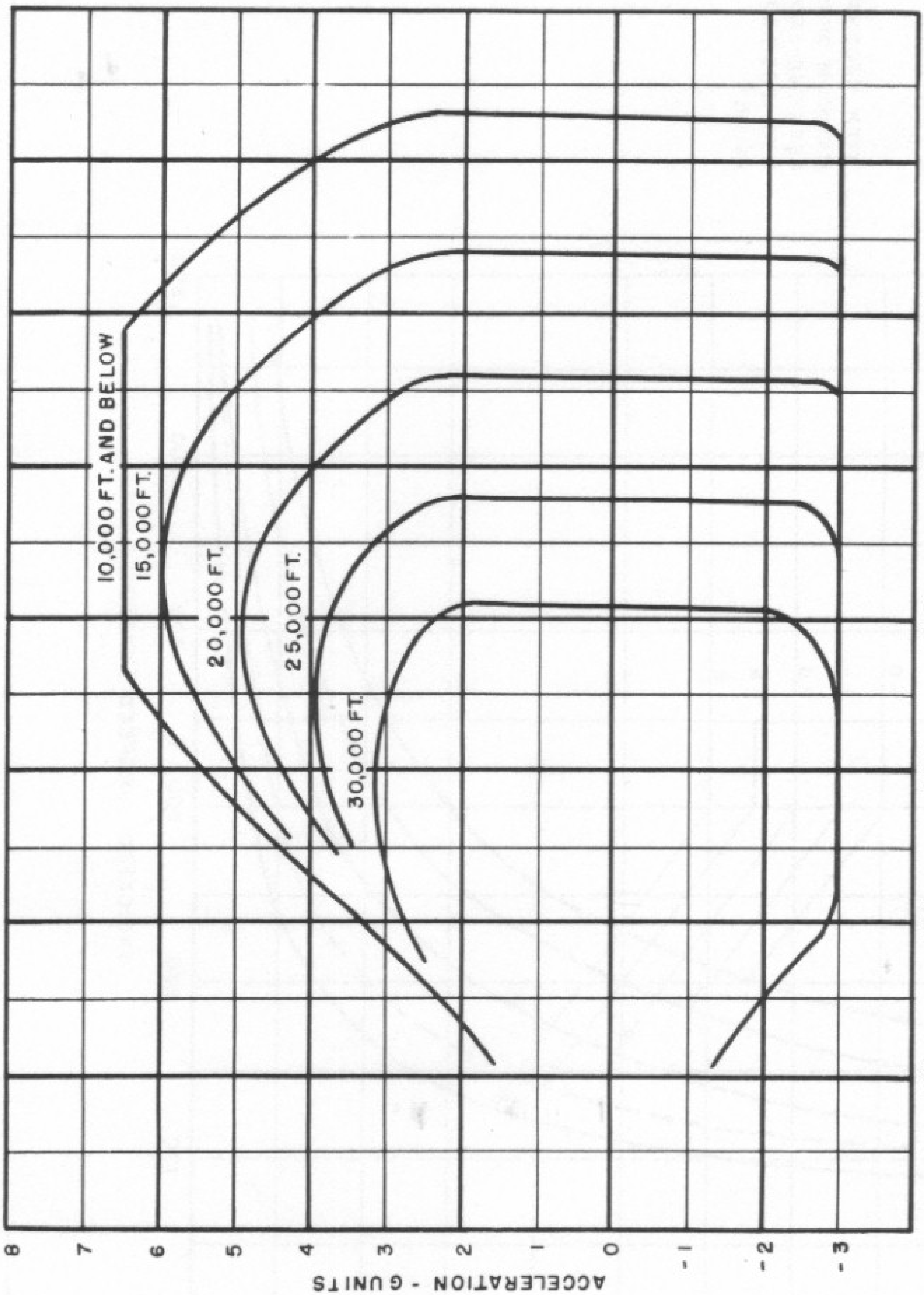


Figure A-4. Angle of Attack vs. Indicated Airspeed Curves

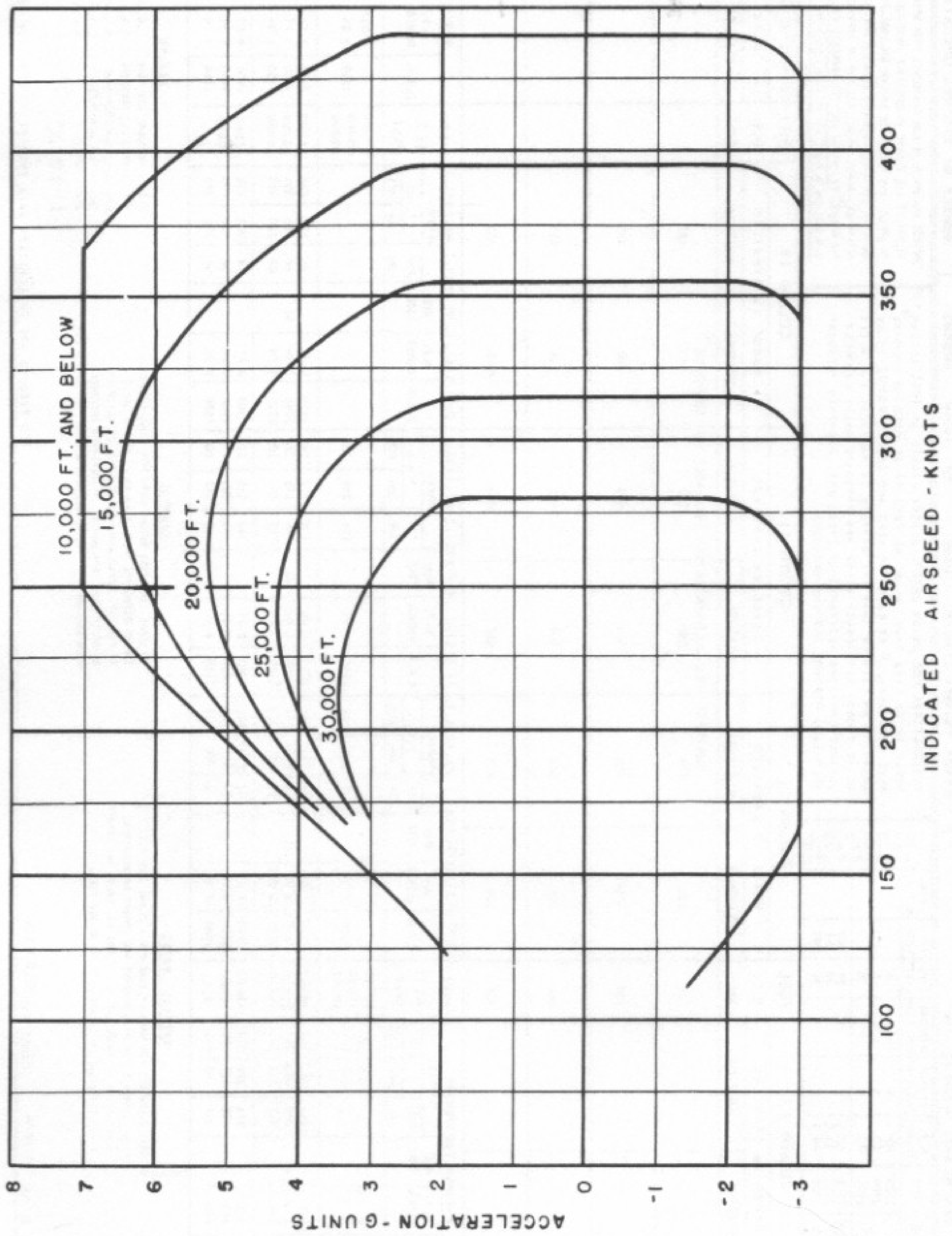
Appendix I of this publication shall not be carried in aircraft on combat missions or when there is a reasonable chance of its falling into the hands of the enemy.



(NOTE: GROSS WEIGHT 12,000 POUNDS - ABOVE APPLIES WITH FIGHTER, BOMBER OR TORPEDO LOADS, EXCEPT THAT WHEN CARRYING 2000 POUND BOMB OR TORPEDO, 5G SHALL NOT BE EXCEEDED.)

Figure A-5. Operation and Strength Flight Limitations (F6F-3, -3N)

Appendix I of this publication shall not be carried in aircraft on combat missions or when there is a reasonable chance of its falling into the hands of the enemy.



(NOTE: GROSS WEIGHT 12,000 POUNDS - ABOVE APPLIES WITH FIGHTER, BOMBER OR TORPEDO LOADS, EXCEPT THAT WHEN CARRYING 2000 POUND BOMB OR TORPEDO, 5G SHALL NOT BE EXCEEDED.)

Figure A-6. Operation and Strength Flight Limitations (F6F-5, -5N)

AAFMC-52A
4-1-44

AIRCRAFT MODEL (S)	FLIGHT OPERATION INSTRUCTION CHART	EXTERNAL LOAD ITEMS
F6F-3, F6F-3N, F6F-5, F6F-5N		1-150 GALLON DROP TANK
ENGINE (S): R-2800-10, -10W	CHART WEIGHT LIMITS: 10900 TO 13540 POUNDS	NUMBER OF ENGINES OPERATING:

LIMITS	RPM	M.P. IN.HG.	BLOWER POSITION	MIXTURE POSITION	TIME LIMIT	CYL. TEMP.	TOTAL G.P.H.
WAR EMERG.	2700 2700 2700	60 60 60	H L H		5 5 5		
MILITARY POWER	2700 2700 2700	52.5 53.0 49.5	H L H		30 30 30	260 260 260	

FOR DETAILS SEE POWER PLANT CHART (FIG. 39 SECT. III)

INSTRUCTIONS FOR USING CHART: SELECT FIGURE IN FUEL COLUMN EQUAL TO OR LESS THAN AMOUNT OF FUEL TO BE USED FOR CRUISING (1) MOVE HORIZONTALLY TO RIGHT OR LEFT AND SELECT RANGE VALUE EQUAL TO OR GREATER THAN THE STATUTE OR NAUTICAL AIR MILES TO BE FLOWN. VERTICALLY BELOW AND OPPOSITE VALUE NEAREST DESIRED CRUISING ALTITUDE (ALT.) READ RPM, MANIFOLD PRESSURE (M.P.) AND MIXTURE SETTING REQUIRED.

NOTES: COLUMN I IS FOR EMERGENCY HIGH SPEED CRUISING ONLY. COLUMNS II, III, IV AND V GIVE PROGRESSIVE INCREASE IN RANGE AT A SACRIFICE IN SPEED. AIR MILES PER GALLON (MI./GAL.) (NO WIND), GALLONS PER HR. (G.P.H.) AND TRUE AIRSPEED (T.A.S.) ARE APPROXIMATE VALUES FOR REFERENCE. RANGE VALUES ARE FOR AN AVERAGE AIRPLANE FLYING ALONE (NO WIND)(1). TO OBTAIN BRITISH IMPERIAL GAL. (OR G.P.H.): MULTIPLY U.S. GAL. (OR G.P.H.) BY 10 THEN DIVIDE BY 12.

COLUMN I		FUEL	COLUMN II		COLUMN III		COLUMN IV		FUEL	SEE NOTE (2) COLUMN V	
RANGE IN AIRMILES		U.S.	RANGE IN AIRMILES		RANGE IN AIRMILES		RANGE IN AIRMILES		U.S.	(2) RANGE IN AIRMILES	
STAUTE	NAUTICAL	GAL.	STATUTE	NAUTICAL	STATUTE	NAUTICAL	STATUTE	NAUTICAL	GAL.	STATUTE	NAUTICAL
			SUBTRACT FUEL ALLOWANCES NOT AVAILABLE FOR CRUISING (1)								
		75	200	174	207	180	215	187			
		150	400	348	414	359	429	373			
		250	668	580	690	599	715	621			
		350	935	812	966	840	1002	670			

MAXIMUM CONTINUOUS						PRESS	(2.67 STAT. (2-32 NAUT.) MI./GAL.)						(2.76 STAT. (2.40 NAUT.) MI./GAL.)						(2.86 STAT. (2.48 NAUT.) MI./GAL.)						PRESS	MAXIMUM AIR RANGE					
R.P.M.	M.P. INCHES	MIX-TURE	APPROX. TOT. GPH	T.A.S. MPH	T.A.S. KTS.	ALT. FEET	R.P.M.	M.P. INCHES	MIX-TURE	APPROX. TOT. GPH	T.A.S. MPH	T.A.S. KTS.	R.P.M.	M.P. INCHES	MIX-TURE	APPROX. TOT. GPH	T.A.S. MPH	T.A.S. KTS.	R.P.M.	M.P. INCHES	MIX-TURE	APPROX. TOT. GPH	T.A.S. MPH	T.A.S. KTS.	ALT. FEET	R.P.M.	M.P. INCHES	MIX-TURE	APPROX. TOT. GPH	T.A.S. MPH	T.A.S. KTS.
	H FT		137	308	267	40000																			40000		SEE NOTE (2)				
2550	H FT		175	347	301	35000	2300	H 34		111	299	260	2250	L 34		105	290	252							35000	2250	H 34		96	273	237
						30000																			30000						
	H49.5		213	351	305	25000	2300	L 36		106	283	246	2250	L 34		100	277	241	2250	L 34		95	272	236	25000	2100	L FT		90	263	228
2550	L49.5	AL	220	335	291	20000	2200	L 36	AL	100	265	230	2200	L 34	AL	96	253	228	2050	L 34	AL	88	255	221	20000	1850	L FT	AL	76	226	196
	L49.5		217	311	271	15000	2200	L 36.5		89	245	213	2100	L 34		87	241	209	2000	L 34		83	236	205	15000	1750	L 34		74	218	189
	L49.5		215	290	252	10000	2100	H 34		87	233	203	2000	H FT		81	224	195	1900	H FT		73	209	182	10000	1850	H FT		67	195	169
2550	H44		172	269	234	5000	2000	H 34		80	212	184	1900	H 34		75	206	179	1650	H 34		65	185	161	5000	1550	H 34		62	177	154
	H44		156	248	216	S.L.	2000	H 34		73	195	169	1700	H 34		66	183	159	1500	H 34		59	170	148	S.L.	1350	H 34		53	153	133

SPECIAL NOTES

(1) MAKE ALLOWANCE FOR WARM-UP, TAKE-OFF & CLIMB (SEE FIG.) PLUS ALLOWANCE FOR WIND, RESERVE AND COMBAT AS REQUIRED.
(2) THESE VALUES OF RANGE ARE FOR 20,000 FT. ONLY. THE NUMBER OF MILES PER GALLON IS GREATEST AT 20,000 FEET.

EXAMPLE

AT 13000 LB. GROSS WEIGHT WITH 100 GAL. OF FUEL (AFTER DEDUCTING TOTAL ALLOWANCES OF 300 GAL.) TO FLY 300 STAT. AIRMILES AT 8000 FT. ALTITUDE MAINTAIN 2100 RPM AND 34 IN. MANIFOLD PRESSURE WITH MIXTURE SET: AL IN NEUTRAL BLOWER

LEGEND

ALT. : PRESSURE ALTITUDE
M.P. : MANIFOLD PRESSURE
GPH : U.S.GAL. PER HOUR
TAS : TRUE AIRSPEED
KTS. : KNOTS
S.L. : SEA LEVEL
N : NEUTRAL
L : LOW
F.R. : FULL RICH
A.R. : AUTO-RICH
A.L. : AUTO-LEAN
C.L. : CRUISING LEAN
M.L. : MANUAL LEAN
F.T. : FULL THROTTLE
H : HIGH

DATA AS OF 2/12/46 BASED ON: FLIGHT TEST

RED FIGURES ARE PRELIMINARY DATA, SUBJECT TO REVISION AFTER FLIGHT CHECK

Figure A-7. Flight Operation Chart

Appendix I of this publication shall not be carried in aircraft on combat missions or when there is a reasonable chance of its falling into the hands of the enemy.

AAFMC-527
6-1-44

AIRCRAFT MODEL(S)
F6F-3, F6F-3N
F6F-5, F6F-5N

ENGINE MODEL(S)
R-2800-10, -10W

TAKE-OFF, CLIMB & LANDING CHART

TAKE-OFF DISTANCE FEET

GROSS WEIGHT LB.	HEAD WIND		HARD SURFACE RUNWAY						SOD-TURF RUNWAY						SOFT SURFACE RUNWAY					
			AT SEA LEVEL		AT 3000 FEET		AT 6000 FEET		AT SEA LEVEL		AT 3000 FEET		AT 6000 FEET		AT SEA LEVEL		AT 3000 FEET		AT 6000 FEET	
	M.P.H.	KTS.	GROUND RUN	TO CLEAR 50' OBJ.	GROUND RUN	TO CLEAR 50' OBJ.	GROUND RUN	TO CLEAR 50' OBJ.	GROUND RUN	TO CLEAR 50' OBJ.	GROUND RUN	TO CLEAR 50' OBJ.	GROUND RUN	TO CLEAR 50' OBJ.	GROUND RUN	TO CLEAR 50' OBJ.	GROUND RUN	TO CLEAR 50' OBJ.	GROUND RUN	TO CLEAR 50' OBJ.
11338	0	0	768	1203	939	1496	1043	1738	791	1226	962	1522	1088	1783	843	1278	1039	1599	1170	1866
	17.3	15	502	786	627	1000	714	1191	516	800	644	1019	745	1220	551	834	696	1070	801	1278
	34.5	30	280	439	362	577	428	712	289	448	371	587	446	731	308	466	401	617	480	766
	46.1	40	166	259	224	358	273	455	170	264	230	364	285	467	181	276	248	382	306	489
13508 1-150 Gallon Fuselage Drop Tank	0	0	1145	1778	1495	2513	1830	3287	1190	1823	1543	2561	1925	3382	1290	1923	1688	2706	2100	3557
	17.3	15	782	1214	1188	2000	1473	2643	813	1247	1226	2040	1550	2720	881	1313	1340	2150	1690	2862
	34.5	30	473	734	643	1080	824	1480	441	754	663	1100	866	1520	533	795	726	1163	945	1600
	46.1	40	305	472	426	716	567	1020	316	485	440	730	597	1050	343	512	481	770	651	1100

NOTE: INCREASE CHART DISTANCES AS FOLLOWS: 75°F + 10%; 100°F + 20%; 125°F + 30%; 150°F + 40%
OPTIMUM TAKE-OFF WITH RPM, IN.HG. & DEG.FLAP IS 80% OF CHART VALUES
DATA AS OF BASED ON:

CLIMB DATA

GROSS WEIGHT LB.	AT SEA LEVEL				AT 5000 FEET					AT 10,000 FEET					AT 15,000 FEET					AT 20,000 FEET					AT 25,000 FEET				
	BEST I.A.S.		RATE OF CLIMB	GAL. OF FUEL	BEST I.A.S.		RATE OF CLIMB	FROM SEA LEVEL		BEST I.A.S.		RATE OF CLIMB	FROM SEA LEVEL		BEST I.A.S.		RATE OF CLIMB	FROM SEA LEVEL		BEST I.A.S.		RATE OF CLIMB	FROM SEA LEVEL		BEST I.A.S.		RATE OF CLIMB	FROM SEA LEVEL	
	MPH	KTS	F.P.M.	USED	MPH	KTS	F.P.M.	TIME MIN.	FUEL USED	MPH	KTS	F.P.M.	TIME MIN.	FUEL USED	MPH	KTS	F.P.M.	TIME MIN.	FUEL USED	MPH	KTS	F.P.M.	TIME MIN.	FUEL USED	MPH	KTS	F.P.M.	TIME MIN.	FUEL USED
13508 (1-150 Gallon Drop Tank)	150	130	2070	14	150	130	1890	2.8	27	150	130	1560	5.7	42	150	130	1340	9.0	55	150	130	1120	12.8	68	150	130	760	17.6	81

POWER PLANT SETTINGS: (DETAILS ON FIG. SECTION III):
DATA AS OF 2-14-46 BASED ON: FLIGHT TEST
FUEL USED (U.S.GAL.) INCLUDES WARM-UP & TAKE-OFF ALLOWANCE

LANDING DISTANCE FEET

GROSS WEIGHT LB.	BEST IAS APPROACH				HARD DRY SURFACE						FIRM DRY SOD						WET OR SLIPPERY					
	POWER OFF		POWER ON		AT SEA LEVEL		AT 3000 FEET		AT 6000 FEET		AT SEA LEVEL		AT 3000 FEET		AT 6000 FEET		AT SEA LEVEL		AT 3000 FEET		AT 6000 FEET	
	MPH	KTS	MPH	KTS	GROUND ROLL	TO CLEAR 50' OBJ.	GROUND ROLL	TO CLEAR 50' OBJ.	GROUND ROLL	TO CLEAR 50' OBJ.	GROUND ROLL	TO CLEAR 50' OBJ.	GROUND ROLL	TO CLEAR 50' OBJ.	GROUND ROLL	TO CLEAR 50' OBJ.	GROUND ROLL	TO CLEAR 50' OBJ.	GROUND ROLL	TO CLEAR 50' OBJ.	GROUND ROLL	TO CLEAR 50' OBJ.

DATA AS OF BASED ON:
OPTIMUM LANDING IS 80% OF CHART VALUES

REMARKS:

NOTE: TO DETERMINE FUEL CONSUMPTION IN BRITISH IMPERIAL GALLONS, MULTIPLY BY 10, THEN DIVIDE BY 12

RED FIGURES ARE PRELIMINARY DATA, SUBJECT TO REVISION AFTER FLIGHT CHECK

LEGEND
I.A.S. : INDICATED AIRSPEED
M.P.H. : MILES PER HOUR
KTS. : KNOTS
F.P.M. : FEET PER MINUTE

Figure A-8. Take-off, Climb and Landing Chart

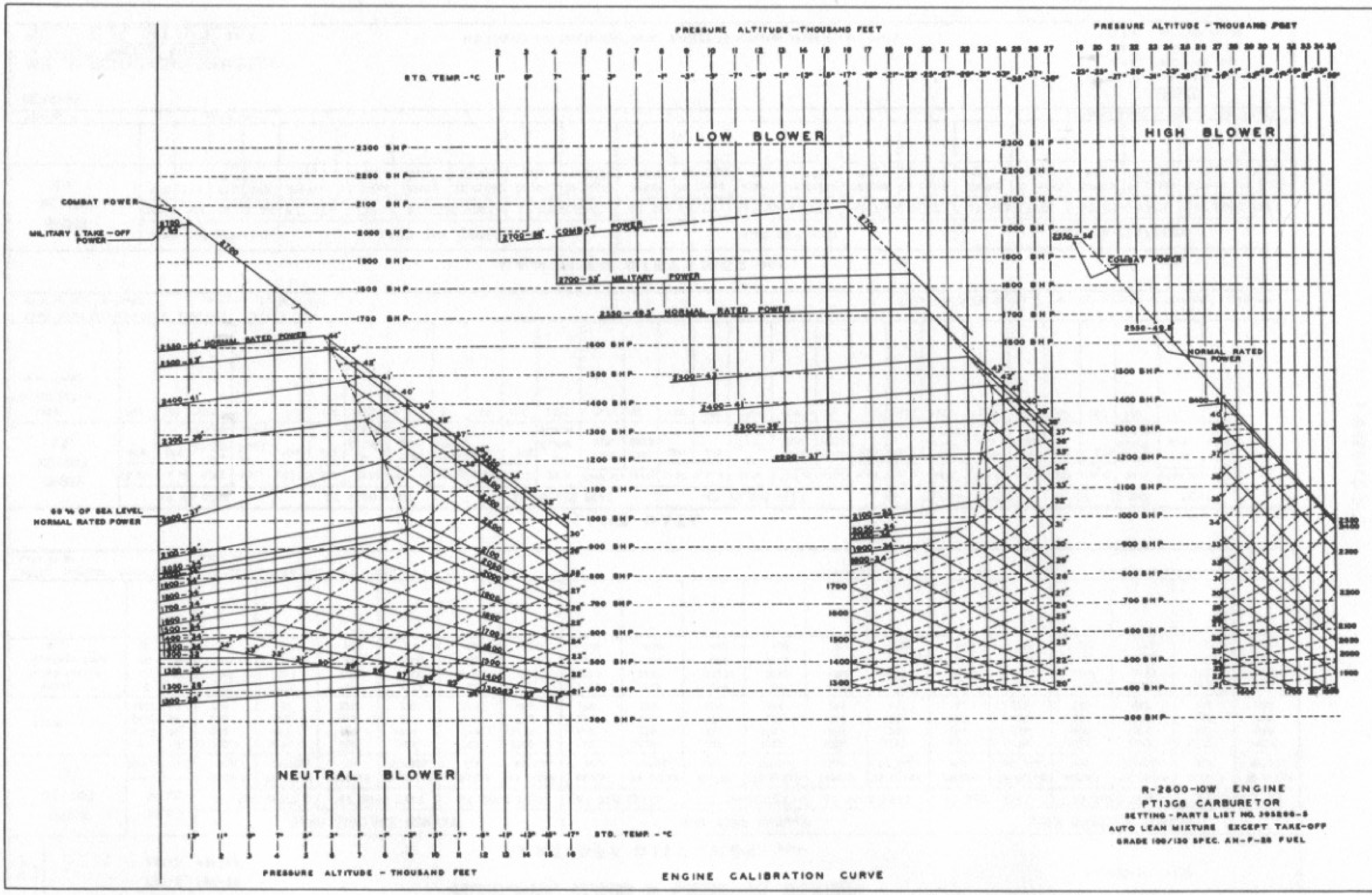


Figure A-9. Engine Calibration Curves

II. Fighter Tactics and Techniques

This section contains a brief discussion of the most basic tactics used by fighter pilot's. Topics include best attack positioning and dive bombing. The **User's Manual** covers the application of these tactics in **Hellcats.**

Fighter Tactics and Techniques

Air to Air

Basic fighter tactics were developed in the first world war when the role of the airplane changed from strictly surveillance to mainly offensive. Two German pilots, Oswald Boelcke and Max Immelmann, were highly successful in securing every advantage during aerial combat, known as "dogfighting."

Fundamental aerial combat tactics consist of:

1. gain a position behind and above your opponent;

2. conceal your aircraft in the sun or clouds if possible;

3. maneuver to descend in a long, slanting dive toward your opponent;

4. hold your fire until in close proximity to your opponent;

5. fire a series of short bursts while continuing to close.

Surprise and attack from the rear are elements always employed by the most effective fighter pilots. Surprise enables you to line-up more easily on the target allowing quick infliction of damage. Attacking from the rear keeps the target in firing range while limiting its ability to return fire.

Diving down onto the target utilizes gravity to gain increased speed for quicker maneuvering during aerial combat. Immelmann made good use of this by effectively getting two passes at an opponent with one dive. The "Immelmann Turn" began after the initial pass at a target. Immelmann would pull up into what appeared to be a loop, when he reached a vertical position, he performed a half roll, leaving him upright and headed the opposite direction. The net effect of this maneuver was to convert excess speed gained from the downward dive into altitude, another pass at the target, and a surprised victim.

Fighter pilots of WWII used the same basic

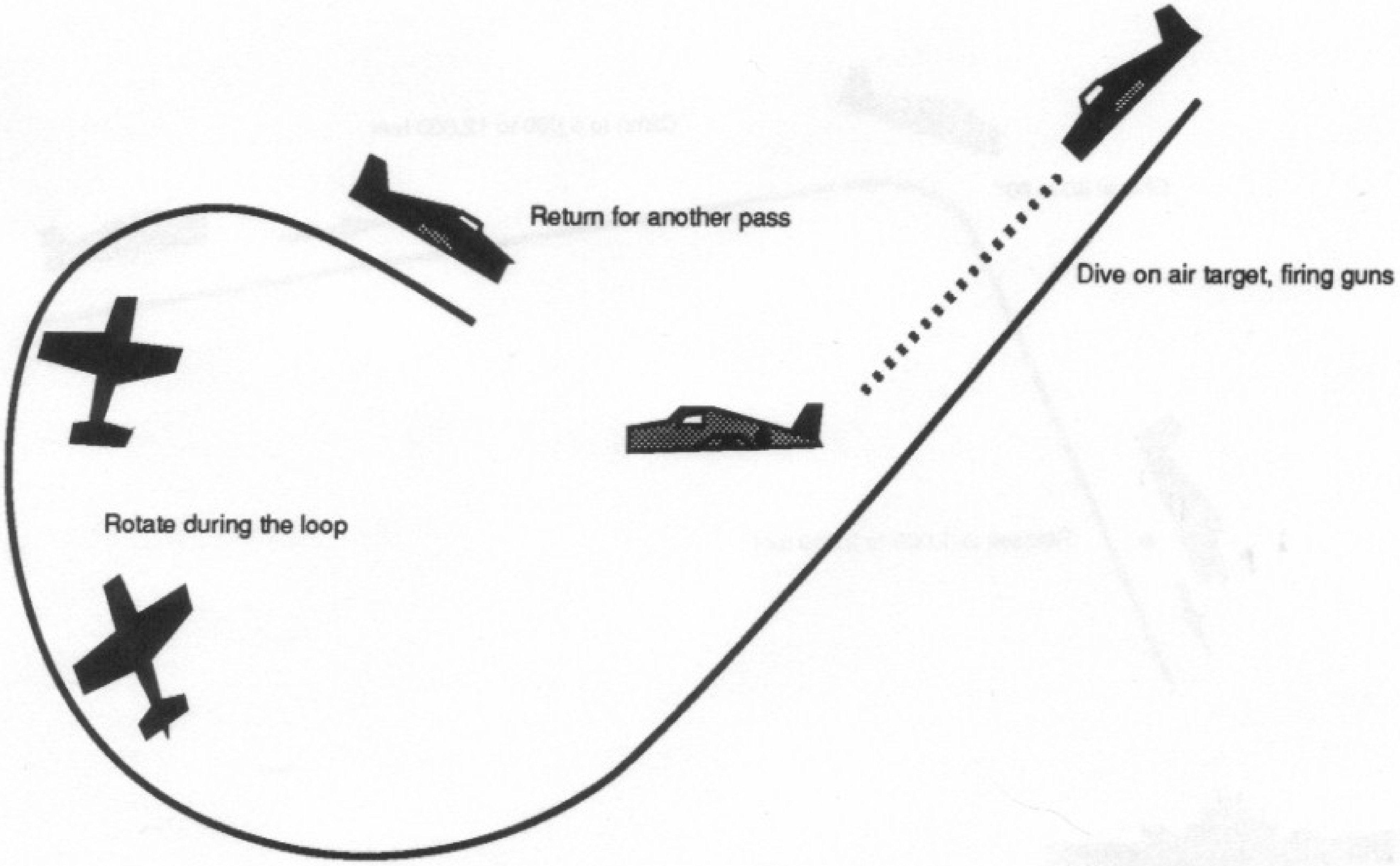


Figure 2-1 Immelmann Turn

tactics as were employed in WWI. However, the technique of "deflection shooting" was developed and polished. Deflection shooting involved lining up behind and to one side of a target. It exposed more of the target's surface area and, consequently, more critical parts of the plane than shooting directly from the rear. Being at the side of the target, the target moves across the guns line of fire, making it is necessary to "lead" the target. Leading involves judging where the bullets and target will intersect and aiming the sights so that the bullets have time to travel to the desired strike point on the moving target.

Air to Land

Dive Bombing

The most accurate method of delivering free-falling ordnance to its intended target is dive bombing. Developed in the latter stages of WWI, dive bombing effectively guides the bomb to the target using momentum gained from the speed of the airplane in a steep dive. To be considered a classic "dive bomb" the angle of descent during the dive must be between 60° and 80°.

A dive bombing run is executed by:

1. climbing to an altitude of between 8,000 and 12,000 feet above and in front of the target;

2. begin the dive at a distance away from the target to allow for an appropriate descent angle (between 60° and 80°);

3. aim the airplane directly at the intended ground target;

4. continue to dive to an altitude of between 1,000 to 2,000 feet above the target;

5. release the ordnance;

6. pull out of the dive and execute a 90° turn.

The straight line of the dive leaves the airplane somewhat vulnerable to ground attack. However, speed of escape and increased accuracy of dive bombing make it a very valuable tactic.

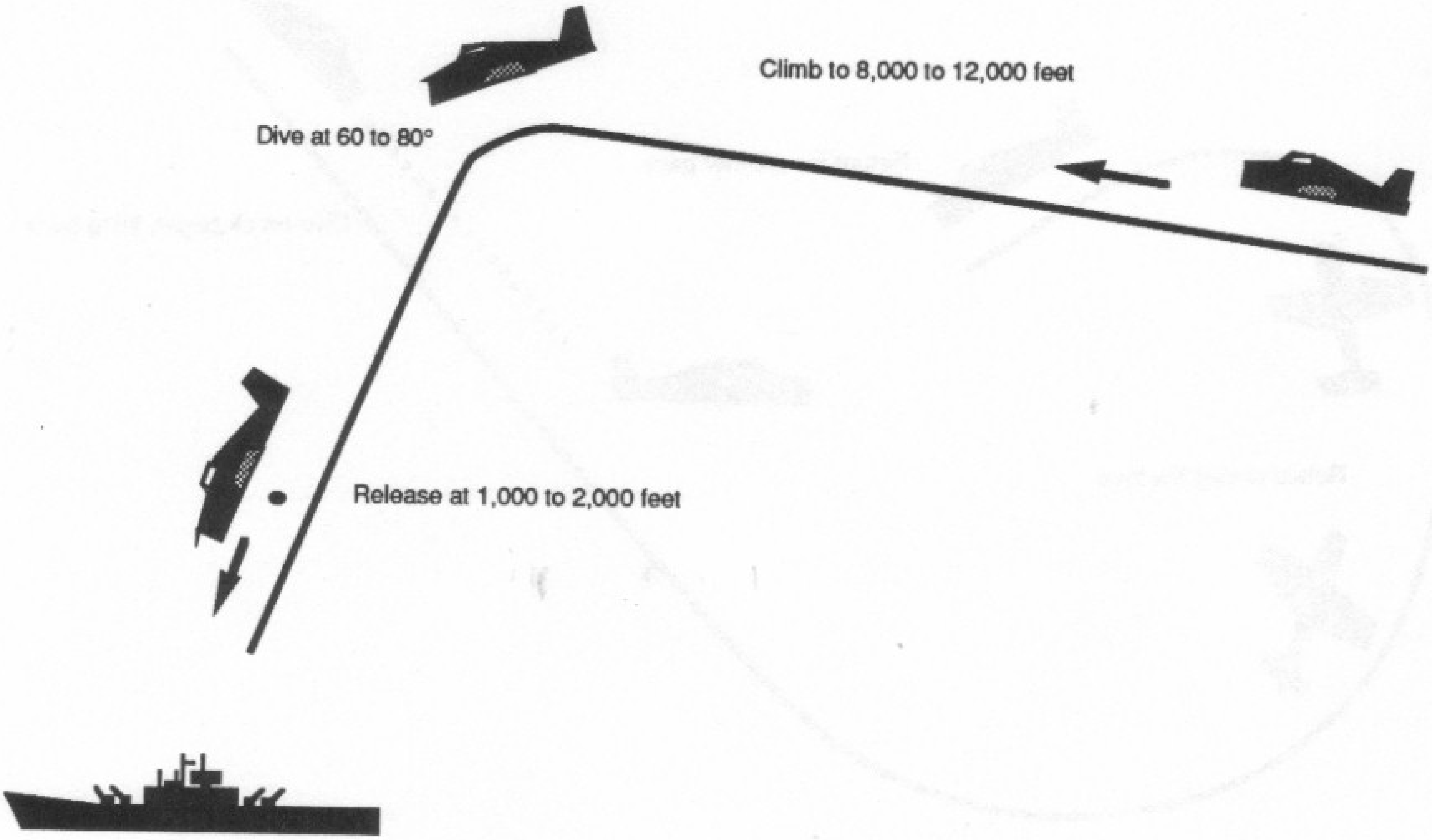


Figure 2-2 Dive Bombing Run

III. Principles of Flight and Performance Characteristics

This section contains material written by the Federal Aviation Administration as a tutorial of basic flight principles. This information is primary and essential for proper command of any type of airplane.

Principles of Flight and Performance Characteristics

This chapter discusses the fundamental physical laws governing the forces acting on an airplane in flight, and what effect these natural laws and forces have on the performance characteristics of airplanes. To competently control the airplane, the pilot must understand the principles involved and learn to utilize or counteract these natural forces.

Modern general aviation airplanes have what may be considered high performance characteristics. Therefore, it is increasingly necessary that pilots appreciate and understand the principles upon which the art of flying is based. The first men to fly learned *why* they were able to fly, before they were successful in their attempts to do so. In other words, they started on the ground and worked up. It is equally essential that the newcomer to today's realm of powered flight do likewise.

The principles of flight spelled out in this section represent the essence of the topics selected, and about which pilots should be well informed. Such information, when understood, can make one a safer and more effective pilot. It must be understood though that the material presented is not intended to cover *all* the parameters of powered flight.

Structure of the Atmosphere

The atmosphere in which we fly is an envelope of air which surrounds the earth and rests upon its surface. It is as much a part of the earth as the seas or the land. However, air differs from land and water inasmuch as it is a mixture of gases. It has mass weight and indefinite shape.

Air, like any other fluid, is able to flow, and change its shape when subjected to even minute pressures because of the lack of strong molecular cohesion. For example, gas will completely fill any container into which it is placed, expanding or contracting to adjust its shape to the limits of the container.

The atmosphere is composed of 78 percent nitrogen, 21 percent oxygen, and 1 percent other gases, such as argon, helium, etc. As some of these elements are heavier than others, there is a natural tendency of these heavier elements, such as oxygen, to settle to the surface of the earth, while the lighter elements are lifted up to the region of higher altitude. This explains why most of the oxygen is contained below 35,000 feet altitude.

Because air has mass and weight, it is a *body*, and as a body, it reacts to the scientific laws of bodies in the same manner as other gaseous bodies. This body of air resting upon the surface of the earth has weight (Fig. 3-1) and at sea level develops an average pressure of 14.7 pounds on each square inch of surface, or 29.92 inches of mercury,— but as its thickness is limited, the higher we go the less air there is above us. For this reason, the weight of the atmosphere at 18,000 feet is only one-half what it is at sea level.

Atmospheric Pressure

Though there are various kinds of pressure, we are mainly concerned with *atmospheric pressure*. It is one of the basic factors in weather changes, helps to lift the airplane, and actuates some of the important flight instruments in the airplane. These instruments are the altimeter, the airspeed indicator, the rate-of-climb indicator, and the manifold pressure gauge.

Though air is very light, it has *mass* and is affected by the attraction of gravity. Therefore, like any other substance, it has weight, and because of its weight, it has force. Since it is a fluid substance, this force is exerted equally in all directions, and its effect on bodies within the air is called *pressure*. Under standard conditions at sea level, the average pressure exerted on the human body by the weight of the atmosphere around it is approximately 14.7 lb./in. The density of air in which we fly has significant effects on the airplane's capability. As air becomes less dense it reduces (1) power because the engine takes in less air, (2) thrust because the propeller is less efficient in thin air, and (3) lift because the thin air exerts less force on the airfoils.

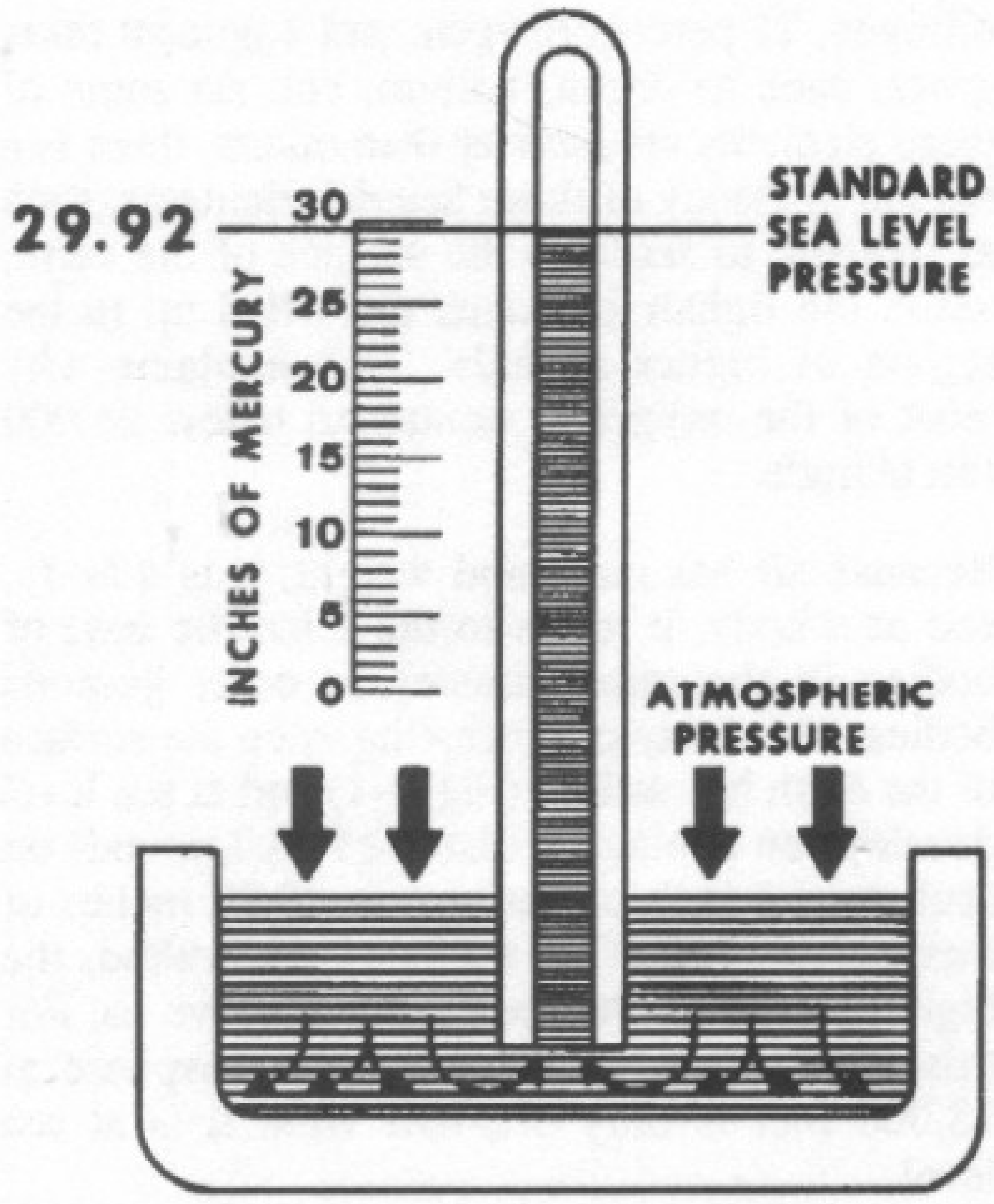


Figure 3-1 Standard Sea Level Pressure

Effect of Pressure on Density

Since air is a gas it can be compressed or expanded. When air is compressed a greater amount of air can occupy a given volume. Conversely, when pressure on a given volume of air is decreased, the air expands and occupies a greater space. That is, the original column of air at a lower pressure, contains a smaller mass of air. In other words, the density is decreased. In fact, density is directly proportional to pressure. If the pressure is doubled the density is doubled, and if the pressure is lowered, so is the density. *This statement is true, only at a constant temperature.*

Effect of Temperature on Density

The effect of increasing the temperature of a substance is to decrease its density. Conversely, deceasing the temperature has the effect of increasing the density. Thus, the density of air varies inversely as the absolute temperature varies. *This statement is true only at a constant pressure.*

In the atmosphere, both temperature and pressure decrease with altitude, and have conflicting effects upon density. However, the fairly rapid drop in pressure as altitude is increased usually has the dominating effect. Hence, we can expect the density to decrease with altitude.

Effect of Humidity on Density

The preceding paragraphs have assumed that the air was perfectly dry. In reality, it is never completely dry. The small amount of water vapor suspended in the atmosphere may be almost negligible under certain conditions, but in other conditions humidity may become an important factor in the performance of an aircraft. Water vapor is lighter than air; consequently *moist air is lighter than dry air*. It is lightest or least dense when, in a given set of conditions, it contains the maximum amount of water vapor. The higher the temperature, the greater amount of water vapor that the air can hold. When comparing two separate air masses, the first warm and moist (both qualities tending to lighten the air) and the second cold and dry (both qualities making it heavier), the first necessarily must be less dense than the second. Pressure, temperature, and humidity have a great influence on airplane performance, because of their effect upon density.

Newton's Laws of Motion and Force

In the 17th century, a philosopher and mathematician, Sir Isaac Newton, propounded three basic laws of motion. It is certain that he did not have the airplane in mind when he did so, but almost everything we know about motion goes back to his three simple laws. These laws, named after Newton, are as follows:

Newton's first law states, in part, that:

> A body at rest tends to remain at rest, and a body in motion tends to remain moving at the same speed and in the same direction.

This simply means that, in nature, nothing starts or stops moving until some outside force causes it to do so. An airplane at rest on the ramp will remain at rest unless a force strong enough to overcome its inertia is applied. Once it is moving, however, its inertia keeps it moving, subject to the various other forces acting on it.

These forces may add to its motion, slow it down, or change its direction.

Newton's second law implies that:

> When a body is acted upon by a constant force, its resulting acceleration is inversely proportional to the mass of the body and is directly proportional to the applied force.

What we are dealing with here are the factors involved in overcoming Newton's First Law of Inertia. It covers both changes in direction and speed, including starting up from rest (positive acceleration) and coming to a stop (negative acceleration, or deceleration).

Newton's third law states that:

> Whenever one body exerts a force on another, the second body always exerts on the first, a force which is equal in magnitude but opposite in direction.

The firing of a shotgun or large caliber rifle provides a "bang" in more ways than one. It may not be the most pleasant way to get your "kicks," but the recoil of a gun as it is fired is a graphic example of Newton's Third Law (Fig. 3-2). The champion swimmer who pushes against the side of the pool during the turn-around, or the infant learning to walk—both would fail but for the phenomena expressed in this law. The charge leaving the gun, or any push against some surface which "pushes back," illustrates this mutual action-reaction. In an airplane, the propeller moves and pushes back the air; consequently the air pushes the propeller (and thus the airplane) in the opposite direction—forward. In a jet airplane, the engine pushes a blast of hot gases backward; the force of equal and opposite reaction pushes against the engine and forces the airplane forward. The movement of all vehicles is a graphic illustration of Newton's Third Law.

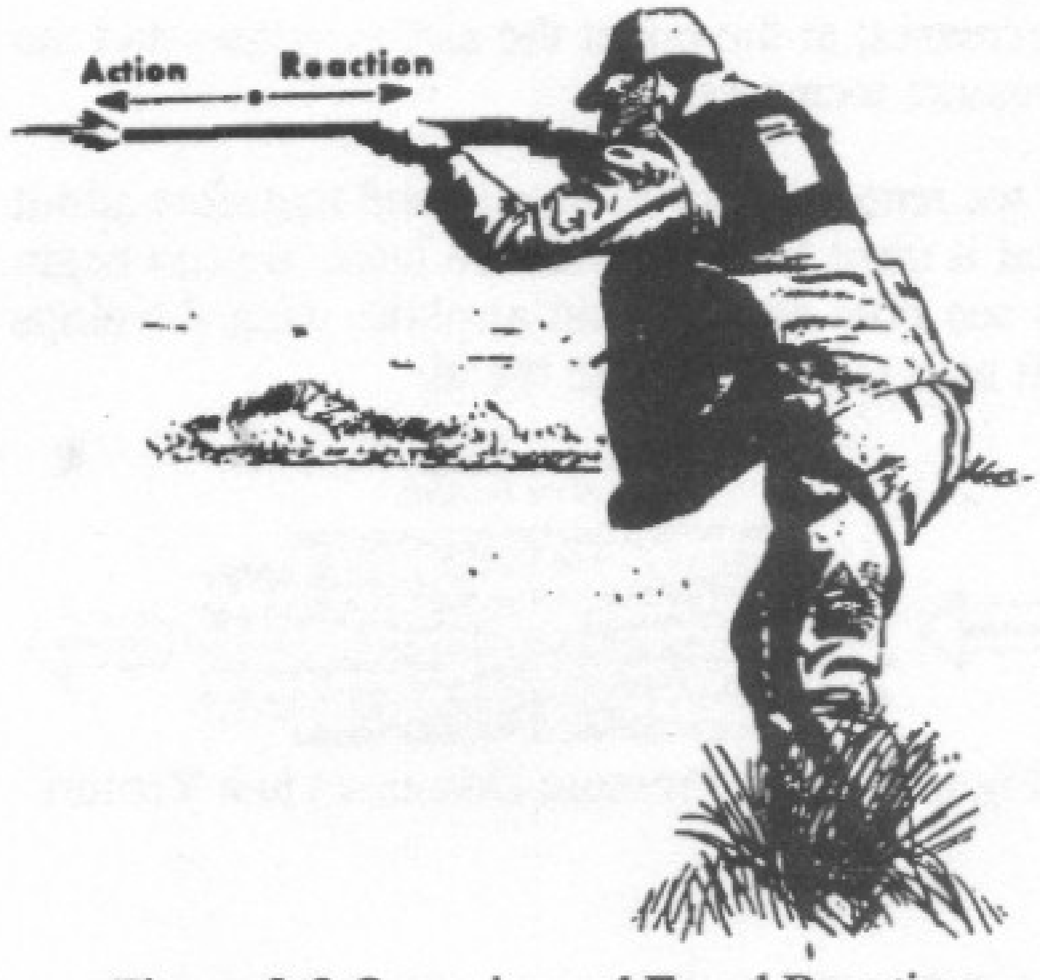


Figure 3-2 Opposite and Equal Reaction

Bernoulli's Principle of Pressure

A half Century after Sir Newton presented his laws, Mr. Daniel Bernoulli, a Swiss mathematician, explained how the pressure of a moving fluid (liquid or gas) varies with its speed of motion. Specifically, he stated that an increase in the speed of movement or flow would cause a decrease in the fluid's pressure. This is exactly what happens to air passing over the curved top of the airplane wing.

An appropriate analogy can be made with water flowing through a garden hose. Water moving through a hose of constant diameter exerts a uniform pressure on the hose; but if the diameter of a section of the hose is increased or decreased, it is certain to change the pressure of the water at that point. Suppose we were to pinch the hose, thereby constricting the area through which the water flows. Assuming that the same volume of water flows through the constricted portion of the hose in the *same period of time* as before the hose was pinched, it follows that the speed of flow must increase at that point.

Therefore, if we constrict a portion of the hose, we not only *increase* the speed of the flow, but we also *decrease*, the pressure at that point. We could achieve like results if we were to introduce streamlined solids (airfoils) at the same point in the hose. This same principle is the basis for the measurement of airspeed (fluid flow) and for analyzing the airfoil's ability to produce lift.

A practical Application of Bernoulli's theorem is the venturi tube (Fig. 3-3). The venturi tube has an air inlet which narrows to a throat (constricted point) and an outlet section which increases in diameter toward the rear. The diameter of the outlet is the same as that of the inlet. At the throat the airflow speeds up and the pressure decreases; at the outlet the airflow slows and the pressure increases.

If we recognize air as a body and therefore admit that it must follow the above laws, we can begin to see how and why an airplane wing develops lift as it moves through the air.

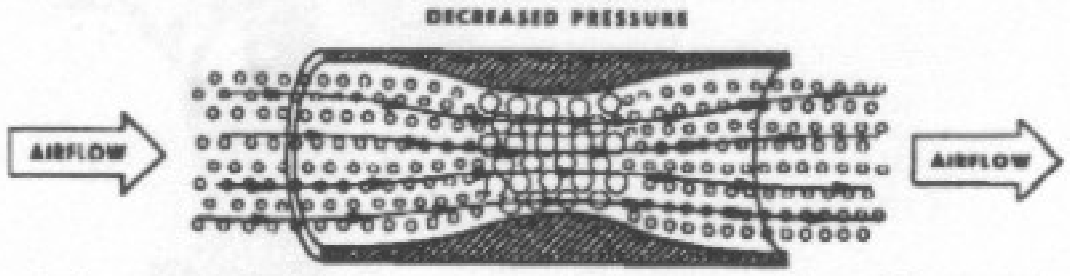


Figure 3-3 Air Pressure Decreases in a Venturi

Airfoil Design

It is far from accidental that there is a basic similarity between the wings of birds and the wings of airplanes. A bird's wing is nothing more than an airfoil, and man has merely copied its shape, modified its design and structure, and developed mechanical power sources as substitutes for his own inadequacies in this area—and so he too flies.

In the sections devoted to Newton's and Bernoulli's discoveries, we have already discussed in general terms the question of how a bird's wings or the wings of man's flying machine sustain flight when both the bird and the machine are heavier than air. Perhaps the explanation to the whole riddle can best be reduced to its most elementary concept by stating that lift (flight) is simply the result of fluid flow (air) about an airfoil—or in everyday language, the result of moving an airfoil (wing), by whatever means, through the air.

Since it is the airfoil which harnesses the force developed by its movement through the air, we will further discuss and explain this "magic" structure, as well as some of the material presented in previous discussions on Newton's and Bernoulli's laws. We can, in this way emphasize the principles that are basic to an understanding of airfoils and airflow.

An airfoil is a structure designed to obtain reaction upon its surface from the air through which it moves or that moves past such a structure. Air acts in various ways when submitted to different pressures and velocities; but we will confine this discussion to the parts of an airplane that we are most concerned with in flight; namely, the airfoils designed to produce lift. By looking at a typical airfoil profile such as the cross section of a wing you can see several obvious characteristics of design (Fig. 3-4). Notice that there is a difference in the curvatures of the upper and lower surfaces of the airfoil (the curvature is called camber). The camber of the upper surface is more pronounced than that of the lower surface, which is somewhat flat in most instances.

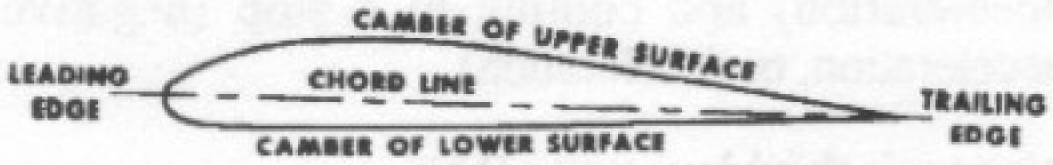


Figure 3-4

In Fig. 3-4, note that the two extremities of the airfoil profile differ in appearance. The end which faces forward in flight is called the *leading edge*, and is rounded, while the other end, the *trailing edge*, is quite narrow and tapered.

A reference line often used in discussing the airfoil is the *chord line*, a straight line drawn through the profile connecting the extremities of the leading and trailing edges. The distance from this chord line to the upper and lower surfaces of the wing denotes the magnitude of the upper and lower camber at any point. Another reference line, drawn from the leading edge to the trailing edge, is the "mean camber line." This mean line is equidistant at all points from the upper and lower contours.

The construction of the wing so as to provide actions greater than its weight, is done by shaping the wing (Fig. 3-4) so that advantage can be taken of the air's response to certain physical laws, and thus develop two actions from the air mass; a positive-pressure lifting action from the air mass below the wing, and a negative-pressure lifting action from lowered pressure above the wing.

As the airstream strikes the relatively flat lower surface of the wing when inclined at a small angle to its direction of motion, the air is forced to rebound downward and therefore causes an upward reaction in positive lift while at the same time airstream striking the upper curved section of the "leading edge" of the wing is deflected

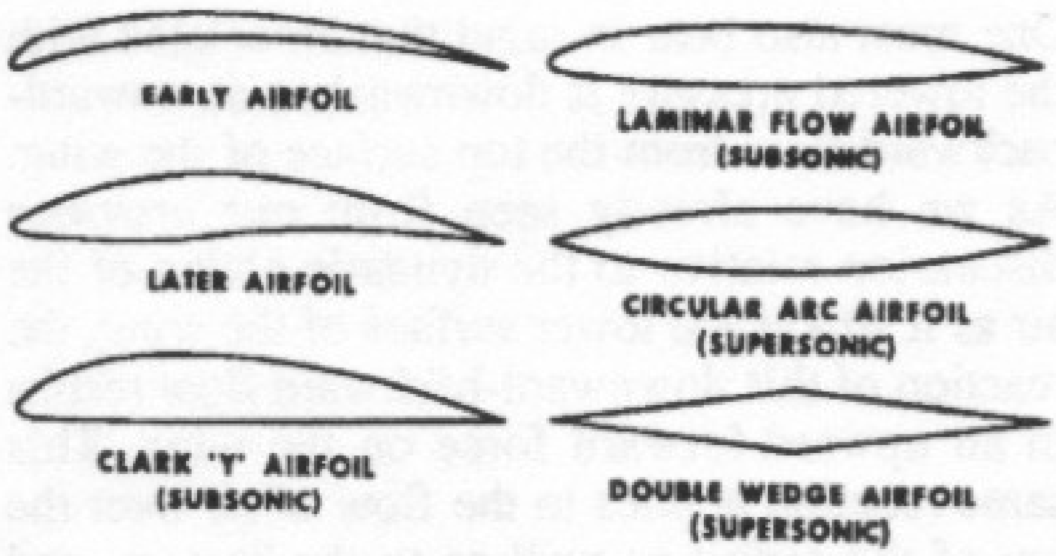


Figure 3-5 Airfoil Designs

upward. In other words, a wing shaped to cause an action on the air, and forcing it downward, will provide an equal reaction from the air, forcing the wing upward. If a wing is constructed in such form that it will cause a lift force greater than the weight of the airplane, the airplane will fly.

Probably you have held your flattened hand out of the window of a moving automobile. As you inclined your hand to the flow of air, the force of air against it pushed it up. Without realizing it you were demonstrating how an airplane gets a portion of its lift. The "wing," in this case your hand, pushed against the air, the air pushed back—remember Newton's third law? As we have already learned, the upward component of this force is called *lift*.

However, if all the lift required were obtained merely from the deflection of air by the lower surface of the wing, an airplane would need only a flat wing like a kite. This, of course, is not the case at all; under certain conditions disturbed air currents circulating at the trailing edge of the wing could be so excessive as to make the airplane lose speed and lift. The balance of the lift needed to support the airplane comes from the flow of air above the wing. Herein lies the key to flight. The fact that most lift is the result of the airflow's downwash from above the wing, must be thoroughly understood in order to continue further in the study of flight. It is neither accurate nor does it serve a useful purpose, however, to assign specific values to the percentage of lift generated by the upper surface of an airfoil versus that generated by the lower surface. These are not constant values, and will vary, not only with flight conditions, but with different wing designs.

It should be understood that different airfoils have different flight characteristics. Many thousands of airfoils have been tested in wind tunnels and in actual flight, but no one airfoil has been found that satisfies every flight requirement. The weight, speed, and purpose of each airplane dictate the shape of its airfoil. It was learned many years ago that the most efficient airfoil for producing the greatest lift was one that had a concave, or "scooped out" lower surface. Later it was also learned that as a fixed design, this type of airfoil sacrificed too much speed while producing lift and, therefore, was not suitable for high speed flight. It is interesting to note, however, that through advanced progress in engineering, today's high speed jets can again take advantage of the concave airfoil's high lift characteristics. Leading edge (Kreuger) flaps and trailing edge (Fowler) flaps, when extended from the basic wing structure, literally change the airfoil shape into the classic concave form, thereby generating much greater lift during slow flight conditions.

On the other hand, an airfoil that is perfectly streamlined and offers little wind resistance sometimes does not have enough lifting power to take the airplane off the ground. Thus, modern airplanes have airfoils which strike a medium between extremes in design, the shape varying according to the needs of the airplane for which it is designed. Figure 3-5 shows some of the more common airfoil sections.

Low Pressure Above

In a wind tunnel or in flight, an airfoil is simply a streamlined object inserted into a moving stream of air. If the airfoil profile were in the shape of a teardrop, the speed and the pressure changes of the air passing over the top and bottom would be the same on both sides. But if the teardrop-shaped airfoil were cut in half lengthwise, a form resembling the basic airfoil (wing) section would result. If the airfoil were then inclined so the airflow strikes it at an angle (angle of attack), the air molecules moving over the upper surface would be forced to move faster than would the molecules moving along the bottom of the air foil, since the upper molecules must travel a greater distance due to the curvature of the upper surface. This increased velocity reduces the pressure above the airfoil.

Bernoulli's principle of pressure by itself does not explain the distribution of pressure over the upper surface of the airfoil. To understand more fully, let us investigate the influence of momentum of the air as it flows in various curved paths near the airfoil (Fig. 3-6). Momentum is the resistance a moving body offers to having its direction or amount of motion changed. When a body is forced to move in a circular path it offers resistance in the direction away from the center of the curved path. This is "centrifugal force." While the particles of air move in the curved path AB, centrifugal force tends to throw them in the direction of the arrows between A and B and hence, causes the air to exert more than normal pressure on the leading edge of the airfoil. But after the air particles pass B (the point of reversal of the curvature of the path) the centrifugal force tends to throw them in the direction of the arrows between B and C (causing reduced pressure on the airfoil). This effect is held until the particles reach C, the second point of reversal of curvature of the airflow. Again the centrifugal force is reversed and the particles may even tend to give slightly more than normal pressure on the trailing edge of the airfoil, as indicated by the short arrows between C and D.

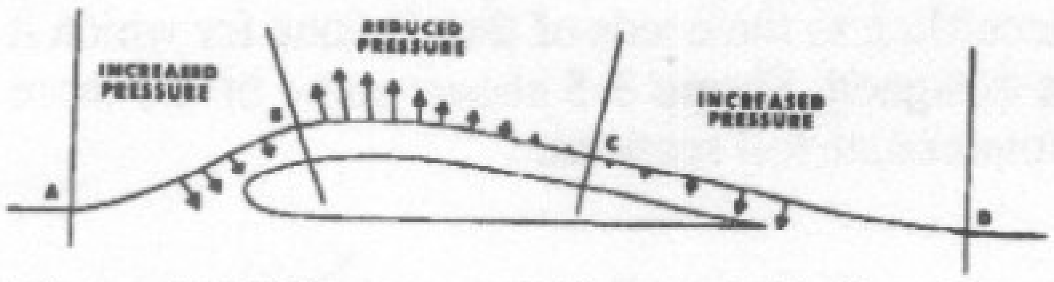


Figure 3-6 Momentum Influences Airflow Over an Airfoil

Therefore, the air pressure on the upper surface of the airfoil is distributed so that the pressure is much greater on the leading edge than the surrounding atmospheric pressure, causing strong resistance to forward motion; but the air pressure is *less* than surrounding atmospheric pressure over a large portion of the top surface (B to C).

As we have seen in the application of Bernoulli's theorem to a venturi, the speedup of air on the top of an airfoil produces a drop in pressure. This lowered pressure is a component of total lift. It is a mistake, however, to assume that the pressure difference between the upper and lower surface of a wing alone accounts for the total lift force produced.

One must also bear in mind that associated with the lowered pressure is downwash; a downward-backward flow from the top surface of the wing. As we have already seen from our previous discussion relative to the dynamic action of the air as it strikes the lower surface of the wing, the reaction of this downward-backward flow results in an upward-forward force on the wing. This same reaction applies to the flow of air over the top of the airfoil as well as to the bottom, and Newton's third law is again in the picture.

High Pressure Below

In the section dealing with Newton's laws as they apply to lift we have already discussed how a certain amount of lift is generated by pressure conditions underneath the wing. Because of the manner in which air flows underneath the wing, a positive pressure results, particularly at higher angles of attack. But there is another aspect to this airflow which must be considered. At a point close to the leading edge, the airflow is virtually stopped (stagnation point) and then gradually increases speed. At some point near the trailing edge it has again reached a velocity equal to that on the upper surface. In conformance with Bernoulli's principles, where the airflow was slowed beneath the wing, a positive upward pressure was created against the wing; i.e., as the fluid speed decreases, the pressure must increase. In essence, this simply "accentuates the positive," since it increases the pressure differential between the upper and lower surface of the airfoil, and therefore increases total lift over that which would have resulted had there been no increase of pressure at the lower surface. Both Bernoulli's principle and Newton's laws are in operation whenever lift is being generated by an airfoil.

Fluid flow or airflow then, is the basis for flight in airplanes, and is a product of the velocity of the airplane. The velocity of the airplane is very important to the pilot since it affects the lift and drag forces of the airplane, as we shall see later in the section on "Forces Acting on an Airplane." The pilot uses the velocity (airspeed) to fly at a minimum glide angle, at maximum endurance, and for a number of other flight maneuvers. airspeed is the velocity of the airplane relative to the air mass through which it is flying.

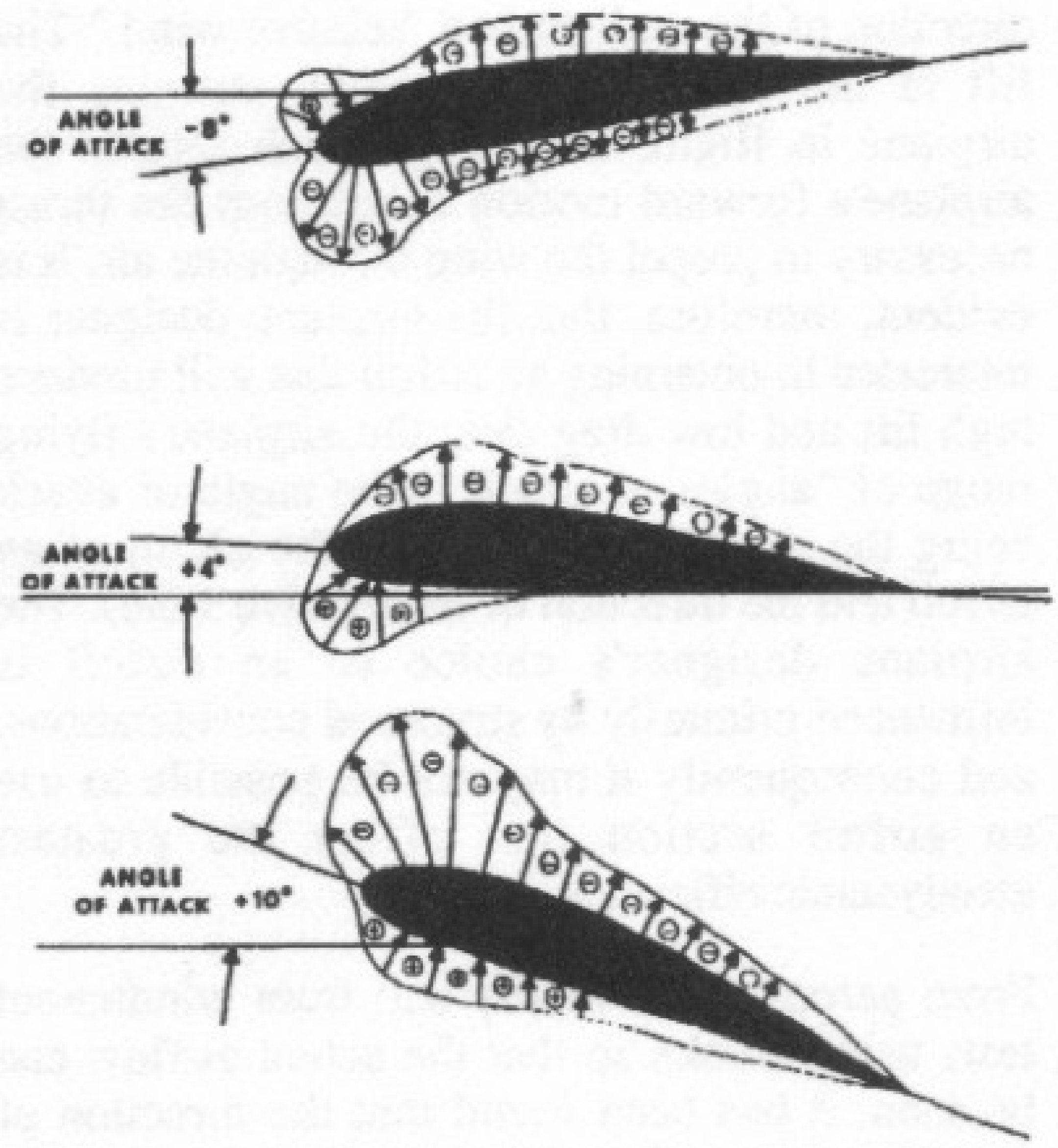


Figure 3-7 Pressure Distribution on an Airfoil

Pressure Distribution

From experiments conducted on wind-tunnel models and on full-size airplanes, it has been determined that as air flows along the surface of a wing at different angles of attack there are regions along the surface where the pressure is negative, or less than atmospheric, and regions where the pressure is positive, or greater than atmospheric. This negative pressure on the upper surface creates a relatively larger force on the wing than is caused by the positive pressure resulting from the air striking the lower wing surface. Figure 3-7 shows the pressure distribution along an airfoil at three different angles of attack. In general, at high angles of attack the center of pressure moves forward, while at low angles of attack the center of pressure moves aft. In the design of wing structures this center of pressure travel is very important, since it affects the position of the airloads imposed on the wing structure in low angle-of-attack conditions and high angle-of-attack conditions. The airplane's aerodynamic balance and controllability are governed by changes in the center of pressure.

The center of pressure is determined through calculation and wind tunnel tests by varying the airfoil's angle of attack through normal operating extremes. As the angle of attack is changed, so

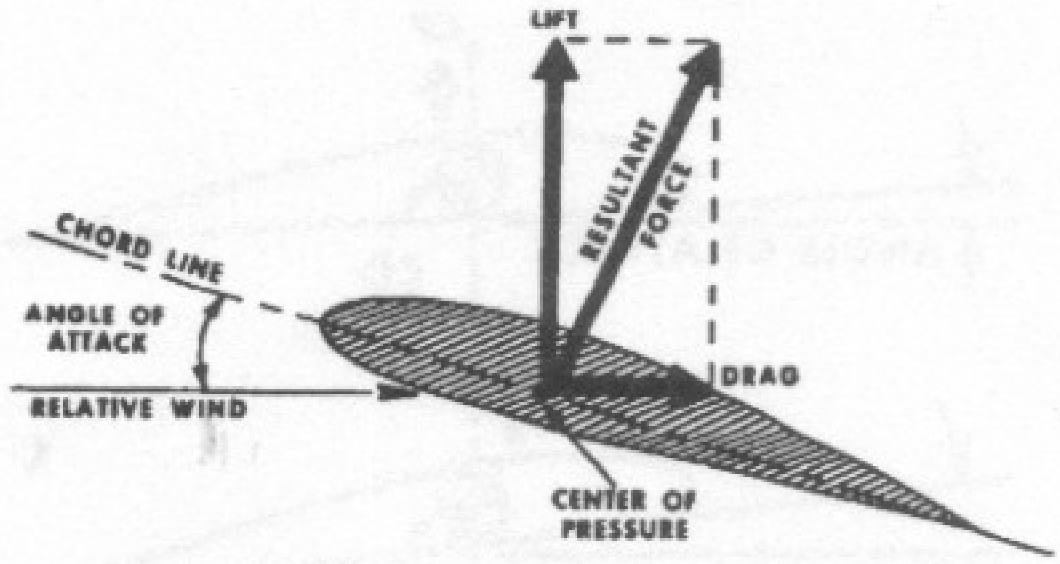


Figure 3-8 Force Vectors on an Airfoil

are the various pressure distribution characteristics (Fig. 3-7). Positive (+) and negative (—) pressure forces are totaled for each angle of attack and the resultant force is obtained. The total resultant pressure is represented by the resultant force vector shown in Fig. 3-8. The point of application of this force vector is termed the "center of pressure" (CP). For any given angle of attack, the center of pressure is the point where the resultant force crosses the chord line.

This point is expressed as a percentage of the chord of the airfoil. A center of pressure at 30 percent of a 60-inch chord would be 18 inches aft of the wing's leading edge. It would appear then that if the designer would place the wing so that its center of pressure was at the airplane's center of gravity, the airplane would always balance. The difficulty arises, however, that the location of the center of pressure changes with change in the airfoil's angle of attack (Fig. 3-9).

In the airplane's normal range of flight attitudes, if the angle of attack is increased, the center of pressure moves forward; and if decreased, it moves rearward. Since the center of gravity is fixed at one point, it is evident that as the angle of attack increases, the center of lift (CP) moves ahead of the center of gravity, creating a force which tends to raise the nose of the airplane or tends to increase the angle of attack still more. On the other hand, if the angle of attack is decreased, the center of lift (CP) moves aft and tends to decrease the angle a greater amount. It is seen then, that the ordinary airfoil is inherently unstable, and that an auxiliary device, such as the horizontal tail surface, must be added to make the airplane balance longitudinally. The balancing forces imparted by the tail surfaces will be discussed later.

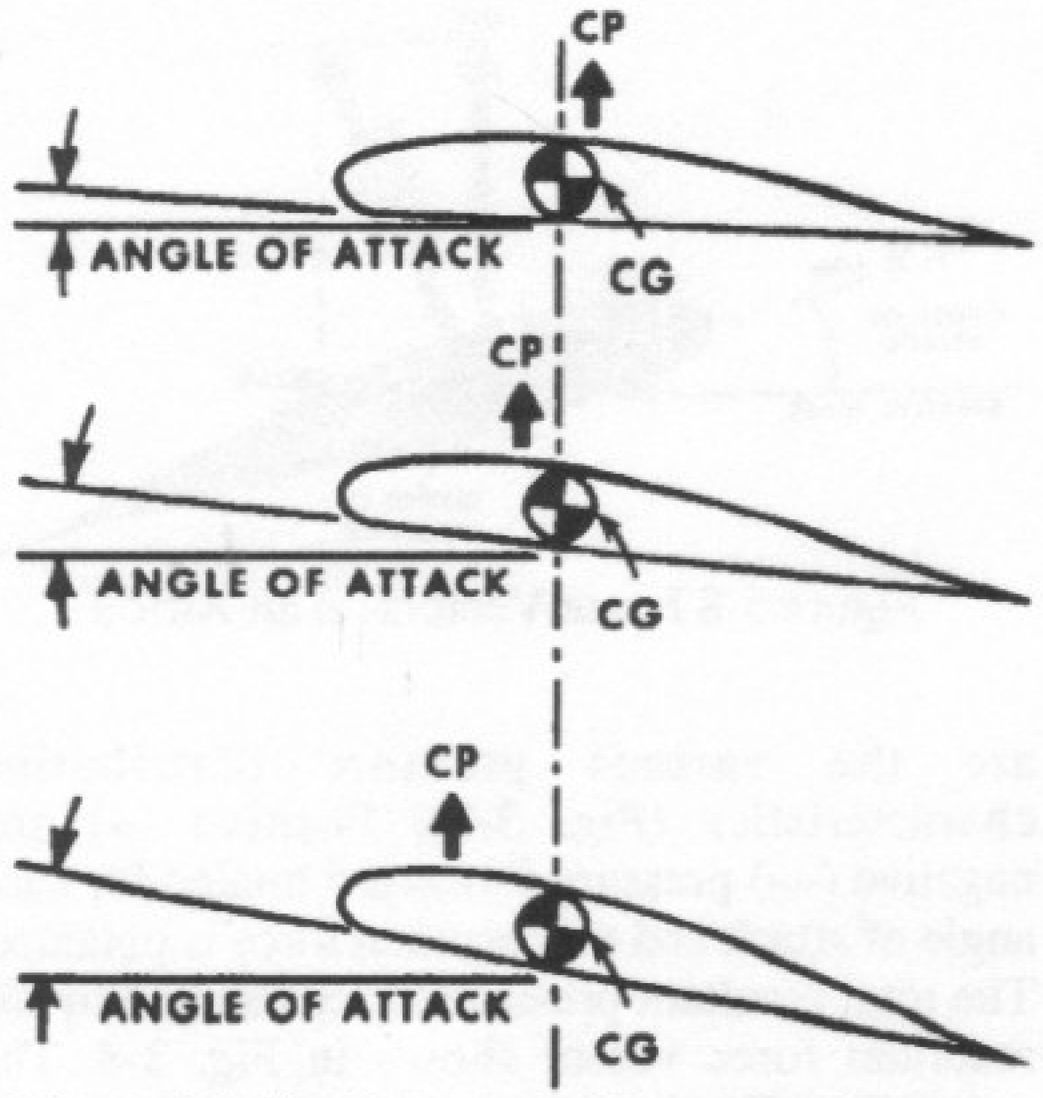


Figure 3-9 CP Changes With Angle of Attack

The balance of an airplane in flight depends, therefore, on the relative position of the center of gravity (CG) and the center of pressure (CP) of the airfoil. Experience has shown that an airplane with the center of gravity in the vicinity of 20 percent of the wing chord can be made to balance and fly satisfactorily.

The tapered wing presents a variety of wing chords throughout the span of the wing. It becomes necessary then, to specify some chord about which the point of balance can be expressed. This chord, known as the mean aerodynamic chord (MAC), usually is defined as the chord of an imaginary untapered wing which would have the same center of pressure characteristics as the wing in question.

Airplane loading and weight distribution also affect center of gravity and cause additional forces which in turn affect airplane balance. This will be discussed in a later section.

Forces on an Airfoil

Air through which the wing moves creates a force the components of which are referred to as "lift" and "drag" (Fig. 3-8). The resultant force, or "force vector," is resolved trigonometrically into these components of lift and drag, perpendicular and parallel, respectively, to the direction of the undisturbed "relative wind." The lift is the upward force which sustains the airplane in flight. The drag which retards the airplane's forward motion determines the thrust necessary to propel the wing through the air. It is evident, therefore, that the airplane designer is interested in obtaining an airfoil that will produce high lift and low drag over the airplane's flying range of "angles of attack," (the angle of attack being the acute angle between the chord of an airfoil and the direction of the relative wind). The airplane designer's choice of an airfoil is influenced primarily by structural considerations, and consequently it may not be possible to use an airfoil section that offers the greatest aerodynamic efficiency.

From aerodynamic theory and from windtunnel tests using smoke so that the actual airflow can be seen, it has been found that the direction of the airstream, after passing over a wing, is changed slightly. The air, as it leaves the trailing edge of the wing, has imparted to it a downward velocity; and this mass of air, therefore, has downward momentum (downwash). As we have learned from Newton's third law of motion that "for every action there is an equal and opposite reaction," an upward reaction of lift is imparted to the wing. Further, the air passing over the top of an airfoil is accelerated. This results in reduction of pressure on top of the wing. Bernoulli's principle, ". . . as velocity increases, the pressure is reduced . . ., ' explains this additional factor in the development of a lifting force.

The drag of a wing is a rearward force which acts opposite to the direction of the airplane's forward motion and is made up of two components, the profile drag and the induced drag.

The profile drag is the resistance, or skin friction, due to the viscosity (stickiness) of the air as it passes along the surface of the wing, in combination with a form drag which is due to the eddying and turbulent wake of air left behind. Profile drag may be thought of as a pure resistance, such as is always encountered when an object is pulled through a viscous medium.

Parasite drag, which applies to the entire airplane, is composed of forces caused by protuberances extending into the airstream which do not contribute to lift. Items such as radio

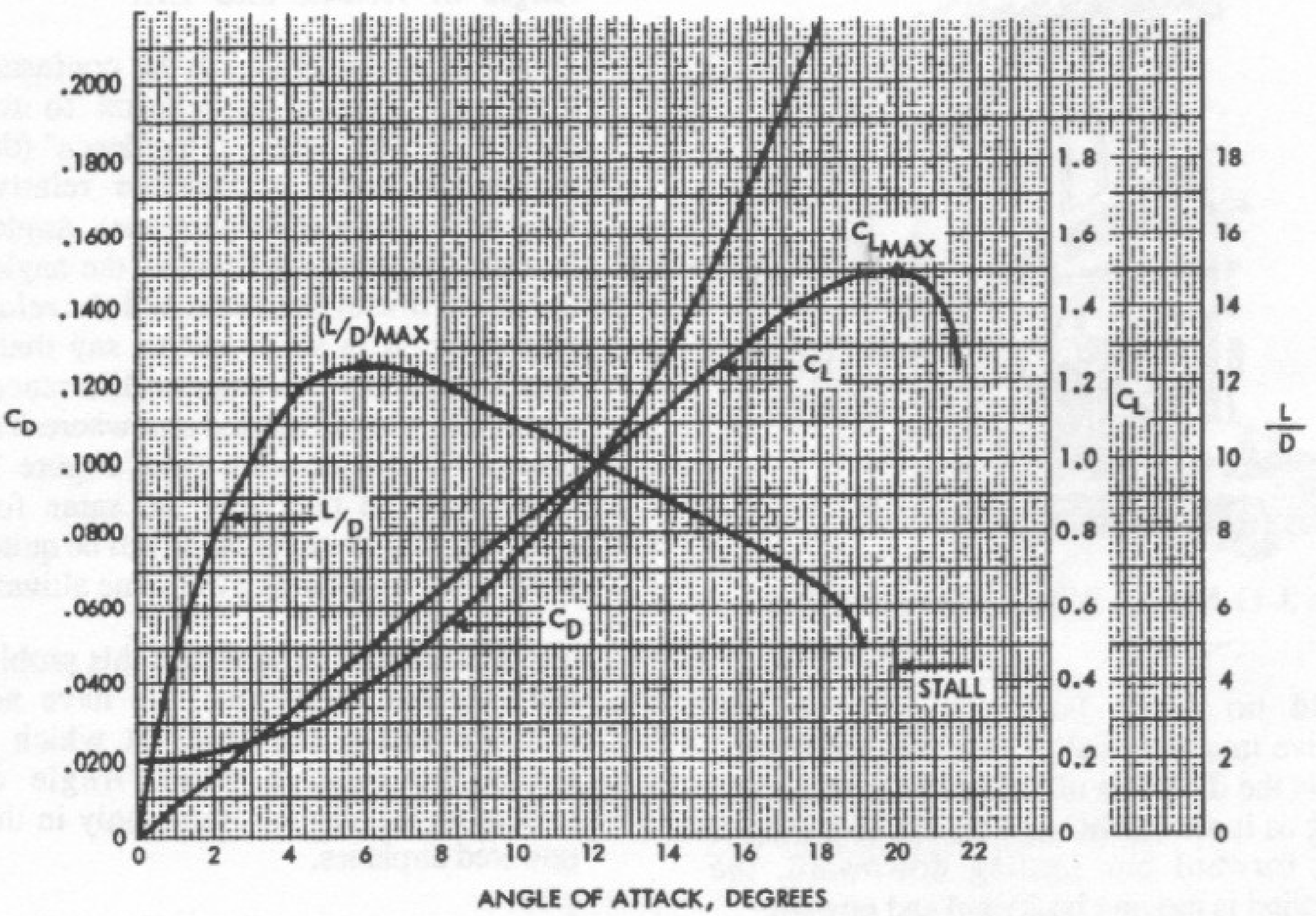


Figure 3-10 Lift Coefficients at Various Angles of Attack

antennas, struts, fittings, landing gear, etc., produce parasite drag.

Induced drag is the direct result of the aerodynamic force resulting from the downward velocity imparted to the air as discussed previously. It may be said induced drag is the result of creating lift.

These forms of drag are more fully explained in later sections of this chapter. The lift and drag of a wing are forces that will vary with the density of the air, the area of the wing, the square of the air velocity, and the angle of attack of the wing as well as other parameters.

The manner in which the lift and drag will vary with the air density, wing area, and velocity factors is the same for any airfoil, but the variation of lift and drag with different angles of attack is a definite characteristic of each individual airfoil section. Graphs which show the variation of lift and drag with angle of attack are drawn for each airfoil design. (Fig. 3-10).

The fact that these graphs give the lift and drag in terms of coefficients should not be confusing. Coefficients are numbers indicating the amount of some change under certain specified conditions, often expressed as a ratio. These coefficients include the mathematical calculations that make it possible to multiply the coefficient by the density, area, and velocity factors corresponding to any particular condition to determine the lift and drag forces in pounds.

Relative Wind

Some pilots seem to find the term "relative wind" difficult to understand. Perhaps Figure 3-11 will help clarify it. The smoke from the factory moves because of natural wind. The pennant on the car traveling along the highway near the smoke stack flutters in a "relative" wind—that is, a flow of air which is created, not by the natural wind but by the auto's forward speed. As can be seen the pennant on the car shows a "wind" whose direction is opposite to that of the smoke, though both are in the same natural wind. Relative wind is really not a wind at all in terms of our everyday understanding of the word. When dealing with principles of flight,

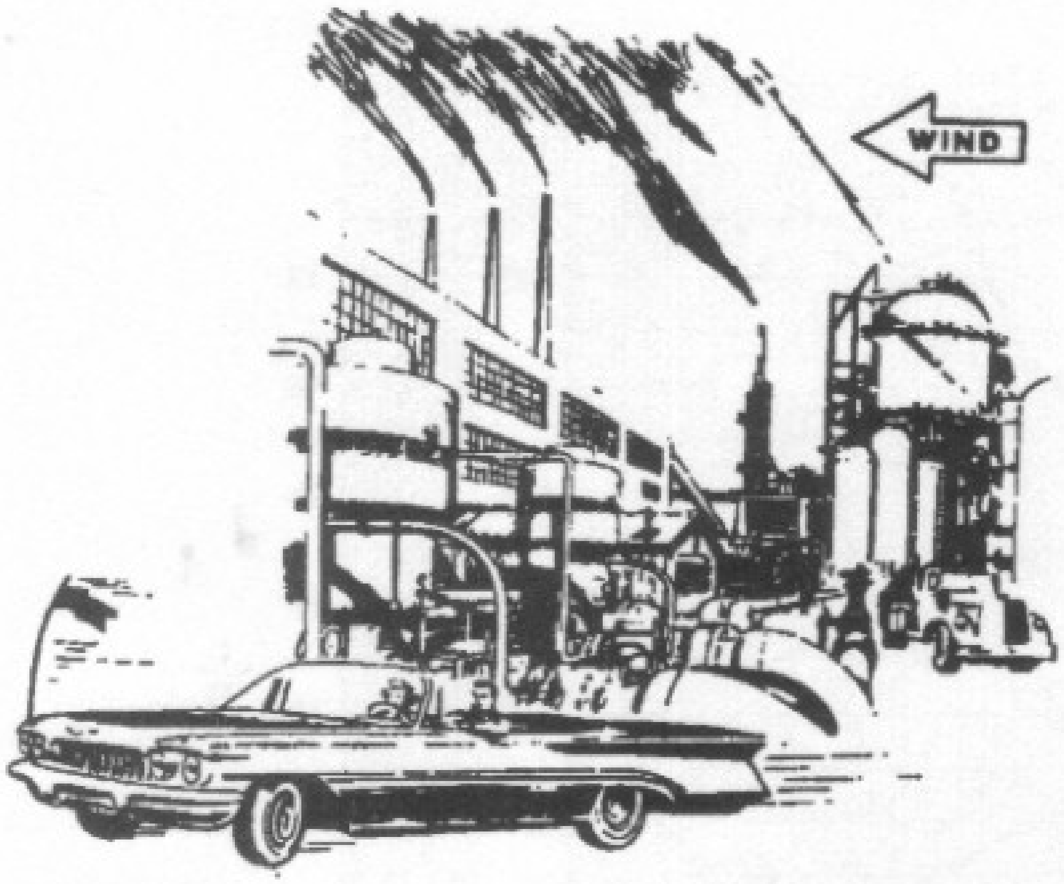


Figure 3-11 Natural Wind VS Relative Wind

it would no doubt be more accurate and descriptive to simply call it "the wind of flight," since it is the direction of airflow with respect to the wing as it moves through the air. If a wing is moving forward but settling downward, the relative wind is moving backward and upward.

Relative wind can be created by the motion of a body through the air, the motion of air past a stationary body, or the combined motions of the air and the body.

For example, an airplane in flight creates relative wind by virtue of its motion. Likewise, an airplane parked on the ramp with a mass of air flowing over its surfaces is subject to relative wind. Also, on a takeoff roll, an airplane is subject to a relative wind which is the resultant of two motions—that of the aircraft along the ground and that of the moving mass of air. It is for this reason that birds and airplanes, when given a choice, take off directly into the wind so that the relative wind flowing along the wings will be greatest and provide as much lift as possible.

When airborne, the actual flightpath of the airplane determines the direction of the relative wind. An understanding of the relative wind concept is fundamental to understanding "angle of attack."

Angle of Attack and Lift

Angle of attack must not be confused with an airplane's attitude in relation to the earth's surface, or with "angle of incidence" (the angle at which the wing is attached relative to the longitudinal axis of the airplane). Angle of attack is most frequently defined as the angle between the chord line of the wing, and the *relative wind.* Generally, it is sufficient to say that angle of attack is simply the angular difference between where the wing is headed and where it is actually going. As can be seen from Figure 3-12, this angle may be precisely the same for climbs, descents, and level flight, or can be quite different even when maintaining the same altitude.

The feature that complicates this problem is that with certain exceptions, we have no way of actually seeing the angle at which the wing meets the relative wind. Angle of attack indicators usually are found only in the turbojet powered airplanes.

In a very real sense the angle of attack is what flight in airplanes is all about. By changing the angle of attack the pilot can control lift, airspeed, and drag. Even the total load supported in flight by the wing, may be modified by variations in angle of attack; and when coordinated with power changes, and auxiliary devices such as flaps, slots, slats, etc.. is the essence of airplane control.

The angle of attack of an airfoil directly controls the distribution of pressure below and above it. When a wing is at a *low* but positive angle of attack, most of the lift is due to the wing's negative pressure (upper surface) and downwash. (Negative pressure is any pressure less than atmospheric, and positive pressure is pressure greater than atmospheric.)

From Fig. 3-7 it can be seen that the positive pressure below the wing at a low angle of attack is very slight, and it can be noted also that the negative pressure above the wing is quite strong by comparison.

At any angle of attack, other than the angle at zero lift, all the forces acting on the wing as a result of the pressure distribution surrounding it may he summed up and represented as one force—the center of pressure.

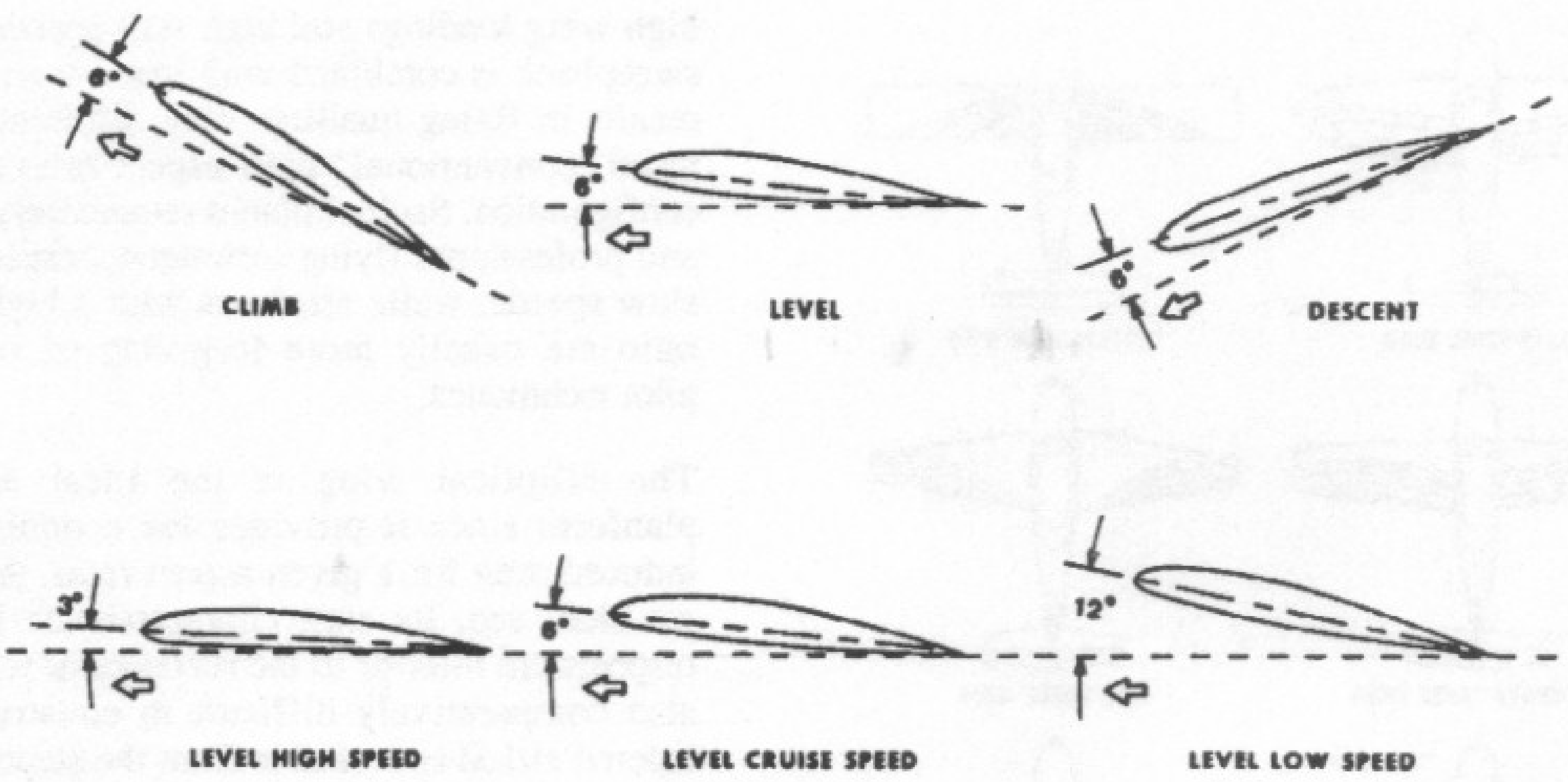


Figure 3-12 Angles of Attack VS Attitude and Speed

When the angle of attack increases to approximately 18° to 20° (on most wings), the air can no longer flow smoothly over the top wing surface. Because the airflow cannot make such a great change in direction so quickly it becomes impossible for the air to follow the contour of the wing. This is the *stalling or critical angle of attack*, and is often called the *burble point*. The burbling or turbulent flow of air which begins near the trailing edge of the wing, suddenly spreads forward over the entire upper wing surface. The negative pressure above the wing suddenly becomes almost equal to atmospheric pressure in value with a resulting loss of lift and a sudden increase in resistance or drag. These events show that Bernoulli's principle is true only in a streamline or smooth airflow—not in a turbulent airflow. The center of pressure at the point of stall is at its maximum forward position, and the resultant force tilts sharply backward.

One of the most important things a pilot should understand about angle of attack is that for any given airplane the stalling or critical angle of attack remains constant regardless of weight, dynamic pressure, bank angle, or pitch attitude. These factors certainly will affect the *speed* at which the stall occurs, but not the *angle*. The aerodynamicist may say that the stalling angle of attack is not always an absolute constant, but for our purposes here it is a valid, useful, and safe concept.

Wing Planform

The previous discussions on wings have dealt only with airfoil *section* properties and two-dimensional airflow. Wing planform—the shape of the wing as viewed from directly above—deals with airflow in three dimensions, and is very important to understanding wing performance and airplane flight characteristics. *Aspect ratio, taper ratio,* and *sweepback* are factors in planform design that are very important to the overall aerodynamic characteristic of a wing (Fig. 3-13).

Aspect ratio is the ratio of wing span to wing chord.

Taper ratio can be either in planform or thickness, or both. In its simplest terms, it is a decrease from wing root to wingtip in wing chord or wing thickness.

Sweepback is the rearward slant of a wing, horizontal tail, or other airfoil surface.

There are two general means by which the designer can change the planform of a wing, either of which will affect aerodynamic characteristics of the wing. The first is to effect a change in the aspect ratio. Aspect ratio is the primary factor in determining the three-dimensional characteristics of the ordinary wing and its lift/drag ratio. An increase in aspect ratio with constant velocity will decrease the drag, especially at high angles of attack, improving the

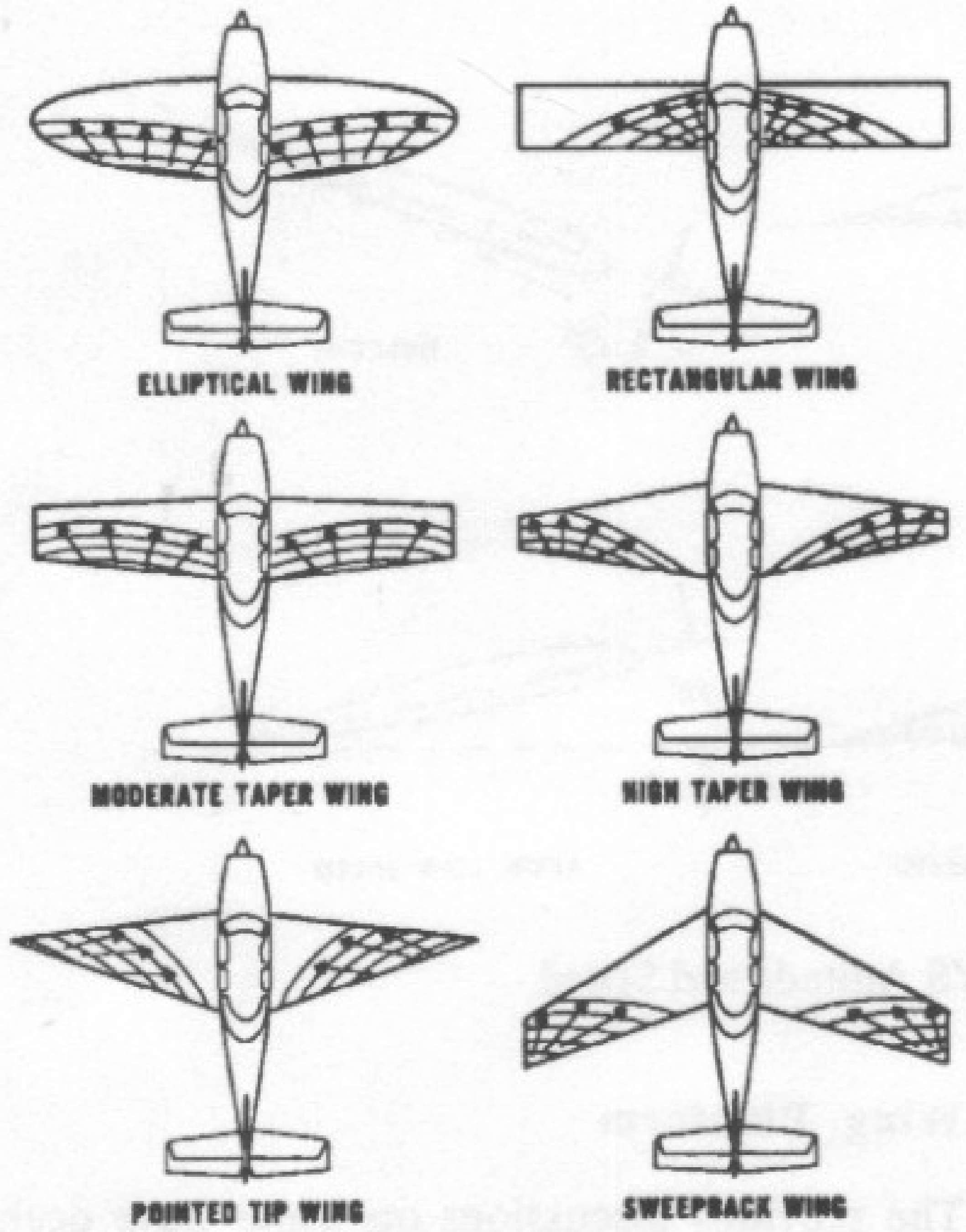


Figure 3-13 Wing Planforms (Exaggerated) and Stall Patterns

performance of the wing when in a climbing attitude. A decrease in aspect ratio will give a corresponding increase in drag. It should be noted, however, that with an increase in aspect ratio there is an increase in the length of span, with a corresponding increase in the weight of the wing structure, which means the wing must be heavier to carry the same load. For this reason, part of the gain (due to a decrease in drag) is lost because of the increased weight, and a compromise in design is necessary to obtain the best results from these two conflicting conditions. The second means of changing the planform is by "tapering" (decreasing the length of chord from the root to the tip of the wing). In general, tapering will cause a decrease in drag (most effective at high speeds) and an increase in lift. There is also a structural benefit due to a saving in weight of the wing.

Most training and general aviation type airplanes are operated at high lift coefficients, and therefore require comparatively high aspect ratios. Airplanes which are developed to operate at very high speeds demand greater aerodynamic cleanness, and greater strength—therefore low aspect ratios. Very low aspect ratios result in high wing loadings and high stall speeds. When sweepback is combined with low aspect ratio, it results in flying qualities very different from a more "conventional" high aspect ratio airplane configuration. Such airplanes require very precise and professional flying techniques, especially at slow speeds, while airplanes with a high aspect ratio are usually more forgiving of improper pilot techniques.

The elliptical wing is the ideal subsonic planform since it provides for a minimum of induced drag for a given aspect ratio, though as we shall see, its stall characteristics in some respects are inferior to the rectangular wing. It is also comparatively difficult to construct. The tapered airfoil is desirable from the standpoint of weight and stiffness, but again is not as efficient aerodynamically as the elliptical wing. In order to preserve the aerodynamic efficiency of the elliptical wing, rectangular and tapered wings are sometimes "tailored" through use of wing twist and variation in airfoil sections until they provide as nearly as possible the elliptical wing's lift distribution.

While it is true that the elliptical wing provides the best lift coefficients before reaching an incipient stall, it gives little advance warning of a complete stall, and lateral control may be difficult because of poor aileron effectiveness.

In comparison, the rectangular wing has a tendency to stall first at the wing root and provides adequate stall warning, adequate aileron effectiveness, and is usually quite stable. It is, therefore, favored in the design of low-cost, low-speed airplanes.

Stall progression patterns for various wing planforms are graphically depicted in Figure 3-13. Note that it is possible for the trailing edge of the inboard portion of the rectangular wing to be stalled while the rest of the wing is developing lift. This is a very desirable characteristic, and along with simplicity of construction is the reason why this type of wing is so popular in light airplanes, despite certain structural and aerodynamic inefficiencies.

Forces Acting on the Airplane

In some respects at least, how well a pilot performs in flight depends upon the ability to plan and coordinate the use of the power and flight controls for changing the forces of thrust, drag, lift, and weight. It is the balance between these forces which the pilot must always control. The better the understanding of the forces and means of controlling them, the greater will be the pilot's skill at doing so.

First, we should define these forces in relation to straight-and-level, unaccelerated flight.

Thrust is the forward force produced by the powerplant/propeller. It opposes or overcomes the force of drag. As a general rule it is said to act parallel to the longitudinal axis. However, this is not always the case as will be explained later.

Drag is a rearward, retarding force, and is caused by disruption of airflow by the wing, fuselage, and other protruding objects. Drag opposes thrust, and acts rearward parallel to the relative wind.

Weight is the combined load of the airplane itself, the crew, the fuel, and the cargo or baggage. Weight pulls the airplane downward because of the force of gravity. It opposes lift, and acts vertically downward through the airplane's center of gravity.

Lift opposes the downward force of weight, is produced by the dynamic effect of the air acting on the wing, and acts perpendicular to the flightpath through the wing's center of lift.

In steady flight the sum of these opposing forces is equal to zero. There can be no unbalanced forces in steady, straight flight (Newton's Third Law). This is true whether flying level or when climbing or descending. This is not the same thing as saying that the four forces are *all* equal. It simply means that the *opposing* forces are equal to, and thereby cancel the effects of, each other. Often the relationship between the four forces has been erroneously explained or illustrated in such a way that this point is obscured. Consider Figure 3-14, for example. In the upper illustration the force vectors of thrust, drag, lift, and weight appear to be equal in value.

The usual explanation states (without stipulating that thrust and drag *do not* equal weight and lift) that thrust equals drag and lift equals weight as shown in Figure 3-14. This basically true statement must be understood or it can be misleading. It should be understood that in straight, level, unaccelerated flight, it is true that the opposing lift/weight forces are equal but they are also *greater than* the opposing forces of thrust/drag that are equal only to each other; not to lift/weight. If we are to be quite correct about this matter, we must say that in steady flight

a. The sum of *all upward* forces (not just lift) equals the sum of *all downward* forces (not just weight).

b. The sum of *all forward* forces (not just thrust) equals the sum of *all backward* forces (not just drag).

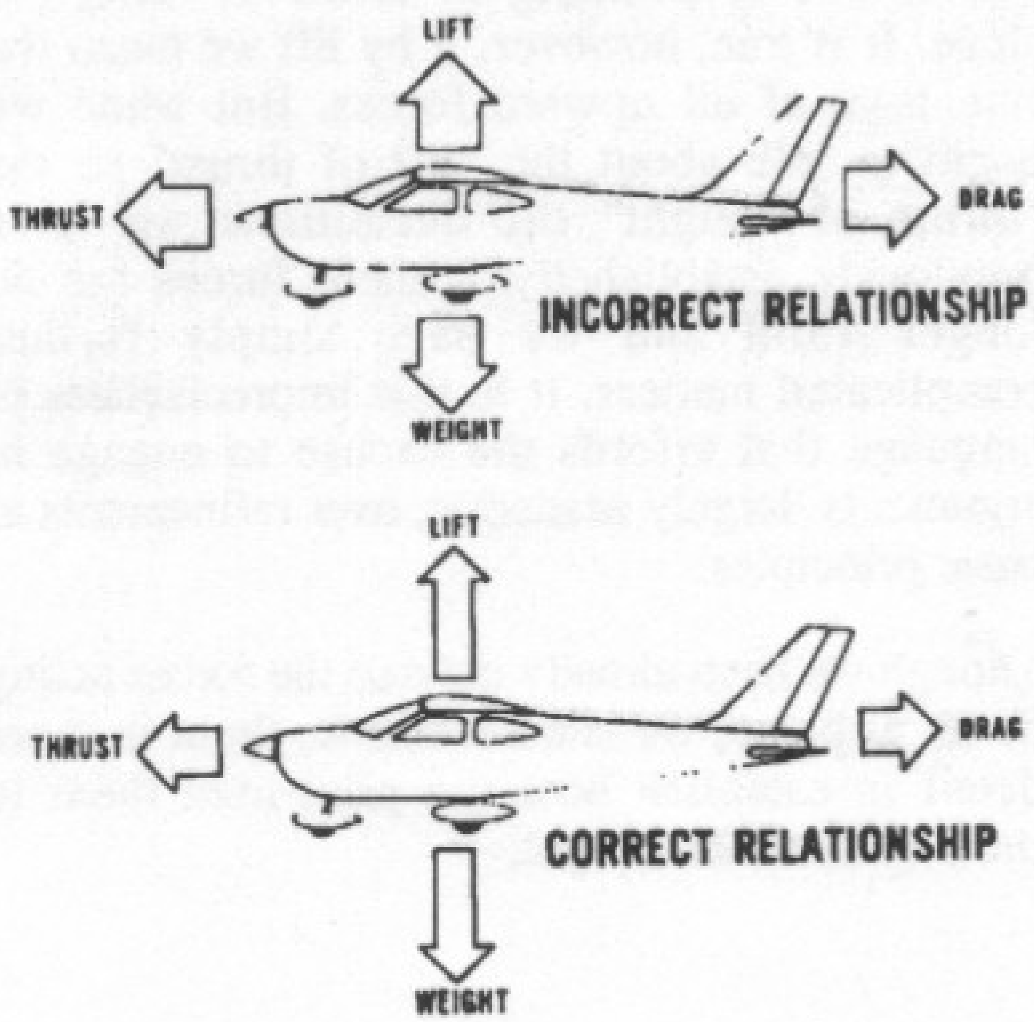


Figure 3-14 Relationship of Forces Acting on an Airplane

This refinement of the old "thrust equals drag; lift equals weight" formula takes into account the fact that in climbs a portion of thrust, since it is directed upward, acts as if it were lift, and a portion of weight since it is directed backward acts as if it were drag. In glides, a portion of the weight vector is directed forward, and therefore acts as thrust. In other words, any time the *flightpath* of the airplane is not horizontal, lift, weight, thrust, and drag vectors must each be broken down into two components (Fig. 3-15).

Discussions of the preceding concepts are frequently omitted in aeronautical texts/handbooks/manuals. The reason is not that they are of no consequence, but because by omitting such discussions, the main ideas with respect to the aerodynamic forces acting upon an airplane in flight can be presented in their most essential elements without being involved in the technicalities of the aerodynamicist. In point of fact, as long as we consider only level flight, and normal climbs and glides in a steady state, it is still true that wing lift is the really *important upward* force, and weight is the really *important downward* force.

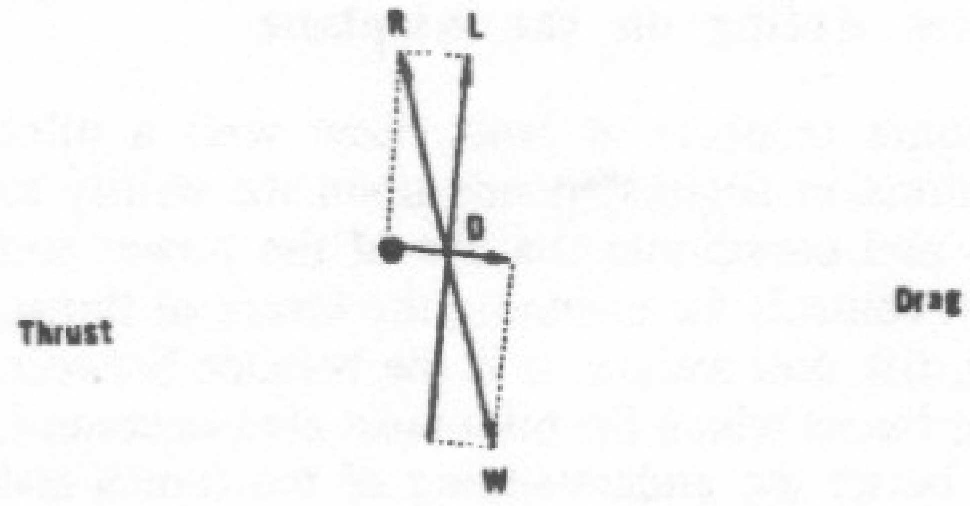
Frequently much of the difficulty encountered in explaining the forces that act upon an airplane is largely a matter of language and its meaning. For example, pilots have long believed that an airplane climbs because of *excess* lift. This is not true if one is thinking in terms of wing lift alone. It is true, however, if by lift we mean the sum total of all upward forces. But when we begin to talk about the "lift of thrust" or the "thrust of weight" the definitions we have previously established for these forces are no longer valid and we have simply further complicated matters. It is this impreciseness in language that affords the excuse to engage in arguments, largely academic, over refinements to basic principles.

Though we have already defined the forces acting on an airplane, we should discuss them in more detail to establish how the pilot uses them to produce controlled flight.

Thrust

Before the airplane begins to move, thrust must be exerted. It continues to move and gain speed until thrust and drag are equal. In order to maintain an *constant* airspeed, thrust and drag must remain equal, just as lift and weight must be equal to maintain a *constant* altitude. If in level flight, the engine power is reduced, the thrust is lessened and the airplane slows down. As long as the thrust is less than the drag, the airplane continues to decelerate until its airspeed is insufficient to support it in the air.

Likewise, if the engine power is increased, thrust becomes greater than drag and the airspeed increases. As long as the thrust continues to be greater than the drag, the airplane continues to accelerate. When drag equals thrust, the airplane flies at a constant airspeed.



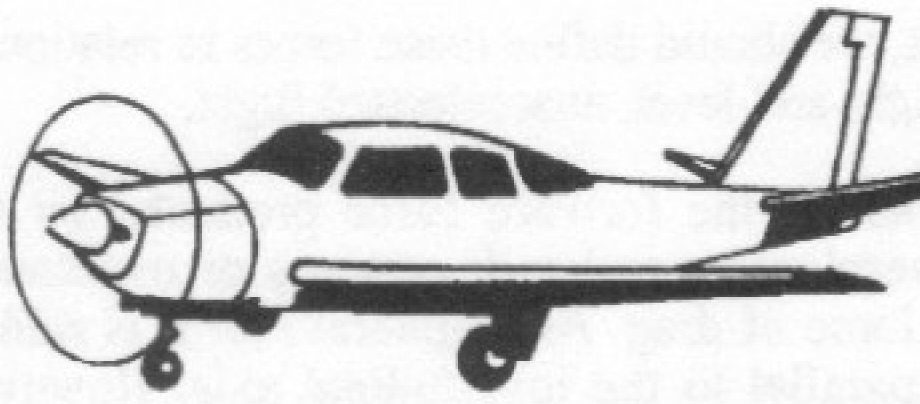


Figure 3-15 Force Vectors During Climb

Straight-and-level flight may be sustained at speeds from very slow to very fast. The pilot must coordinate angle of attack and thrust in all speed regimes if the airplane is to be held in level flight. Roughly, these regimes can be grouped in three categories; low-speed flight, cruising flight, and high-speed flight.

When the airspeed is low, the angle of attack must be relatively high to increase lift if the balance between lift and weight is to be maintained (Fig. 3-16). If thrust decreases and airspeed decreases, lift becomes less than weight and the airplane will start to descend. To maintain level flight, the pilot can increase the angle of attack an amount which will generate a lift force again equal to the weight of the airplane and while the airplane will be flying more slowly, it will still maintain level flight if the pilot has properly coordinated thrust and angle of attack.

Straight-and-level flight in the slow speed regime provides some interesting conditions relative to the equilibrium of forces, because with the airplane in a nose-high attitude there is a vertical component of thrust which helps support the airplane. For one thing, wing loading tends to be less than would be expected. Most pilots are aware that an airplane will stall, other conditions being equal, at a slower speed with the power on than with the power off. (Induced airflow over the wings from the propeller contributes to this, too.) However, if we restrict our analysis to the four forces as they are usually defined, one can say that in *straight-and-level slow speed flight* the thrust is equal to drag, and lift is equal to weight.

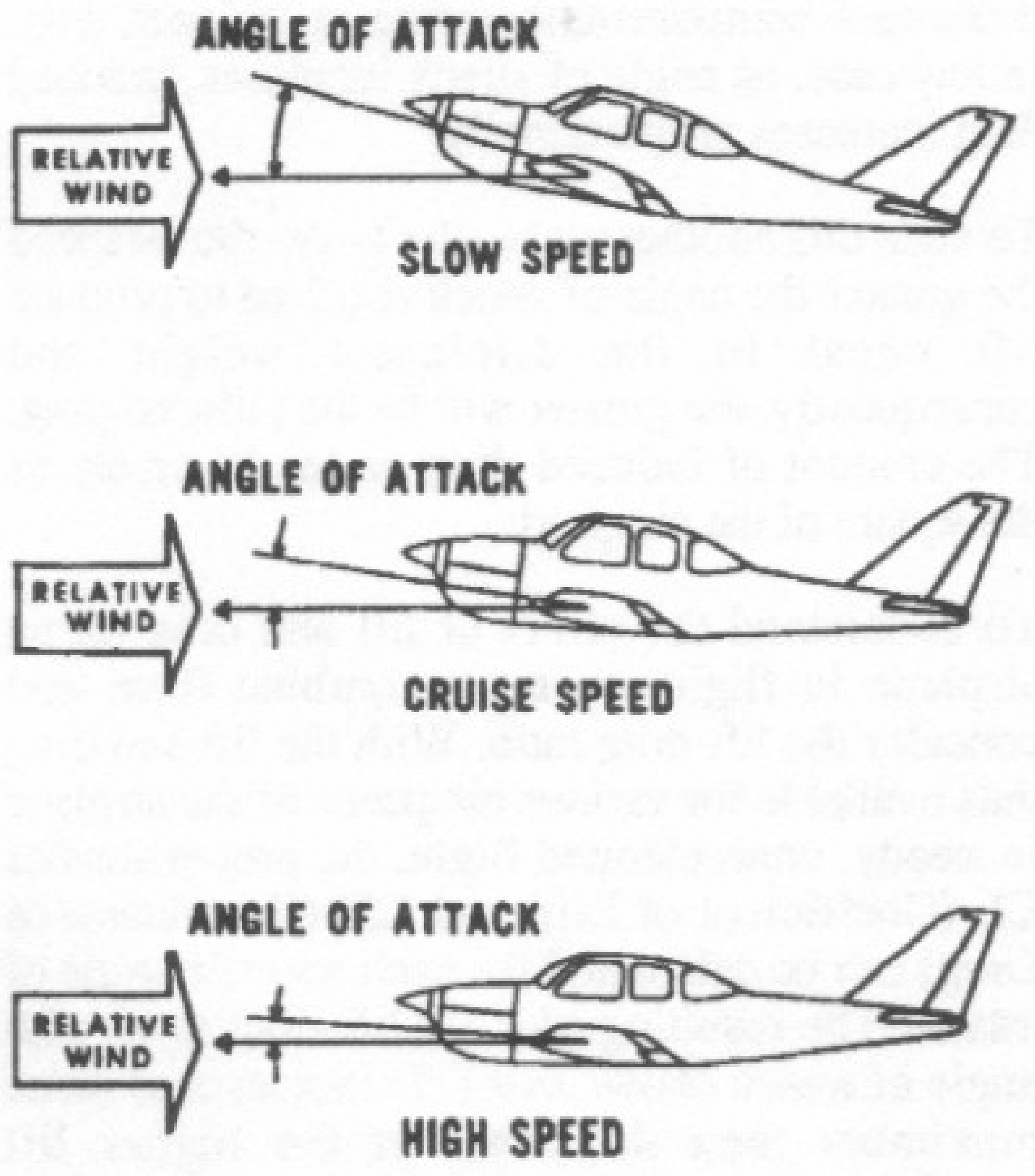


Figure 3-16 Angles of Attack at Various Speeds

During straight-and-level flight when thrust is increased and the airspeed increases, the angle of attack must be decreased. That is, if changes have been coordinated, the airplane will still remain in level flight but at a higher speed when the proper relationship between thrust and angle of attack is established.

If the angle of attack were not coordinated (decreased) with this increase of thrust the airplane would climb. But decreasing the angle of attack modifies the lift, keeping it equal to the weight, and if properly done, the airplane still remains in level flight. Level flight at even slightly negative angles of attack is possible at very high speed. It is evident then, that level flight can be performed with any angle of attack between stalling angle and the relatively small negative angles found at high speed.

Drag

Drag in flight is of two basic types: parasite drag and induced drag. The first is called parasite because it in no way functions to aid flight while the second is induced or created as a result of the wing developing lift.

Parasite drag is composed of two basic elements: form drag, resulting from the disruption of the streamline flow; and the resistance of skin friction.

Of the two components of parasite drag, form drag is the easier to reduce when designing an airplane. In general, a more streamlined object produces the best form to reduce parasite drag.

Skin friction is the type of parasite drag that is most difficult to reduce. No surface is perfectly smooth. Even machined surfaces, when inspected through magnification, have a ragged, uneven appearance. This rough surface will deflect the streamlines of air on the surface, causing resistance to smooth airflow. Skin friction can be minimized by employing a glossy, flat finish to surfaces, and by eliminating protruding rivet heads, roughness, and other irregularities.

Another element must be added to the consideration of parasite drag when designing an airplane. This drag combines the effects of form drag and skin friction and is called interference drag. If we place two objects adjacent to one another the resulting turbulence produced may be 50 to 200 percent greater than the parts tested separately.

The three elements, form drag, skin friction, and interference drag, are all computed to determine parasite drag on an airplane.

Shape of an object is a big factor in parasite drag. However, indicated airspeed is an equally important factor when speaking of parasite drag. The profile drag of a streamlined object held in a fixed position relative to the airflow increases approximately as the square of the velocity; thus, doubling the airspeed increases the drag four times, and tripling the airspeed increases the drag nine times. This relationship, however, holds good only at comparatively low subsonic speeds. At some higher airspeeds the rate at which profile drag has been increased with speed suddenly begins to increase more rapidly.

The second basic type of drag is *induced drag*. It is an established physical fact that no system which does work in the mechanical sense can be 100% efficient. This means that whatever the nature of the system, the required work is obtained at the expense of certain additional work that is dissipated or lost in the system. The more efficient the system the smaller this loss.

In level flight the aerodynamic properties of the wing produce a required lift, but this can be obtained only at the expense of a certain penalty. The name given to this penalty is *induced drag*. Induced drag is inherent whenever a wing is producing lift and, in fact, this type of drag is inseparable from the production of lift. Consequently, it is always present if lift is produced.

The wing produces the lift force by making use of the energy of the free airstream. Whenever the wing is producing lift we have seen that the pressure on the lower surface of the wing is greater than that on the upper surface. As a result, the air tends to flow from the high pressure area below the wingtip upward to the low pressure area above the wing. In the vicinity of the wingtips there is a tendency for these pressures to equalize, resulting in a lateral flow outward from the underside to the upper surface of the wing. This lateral flow imparts a rotational velocity to the air at the wingtips and trails behind the wing. Therefore, flow about the wingtips will be in the form of two vortices trailing behind as the wings moves on.

When the airplane is viewed from the tail, these vortices will circulate counterclockwise about the right wingtip and clockwise about the left wingtip (Fig. 3-17). Bearing in mind the direction of rotation of these vortices, it can be seen that they induce an upward flow of air beyond the wingtip, and a downwash flow behind the wing's trailing edge. This induced downwash has nothing in common with the downwash that is necessary to produce lift. It is, in fact, the source of induced drag. The greater the size and strength of the vortices and consequent downwash component on the net airflow over the wing, the greater the induced drag effect becomes.

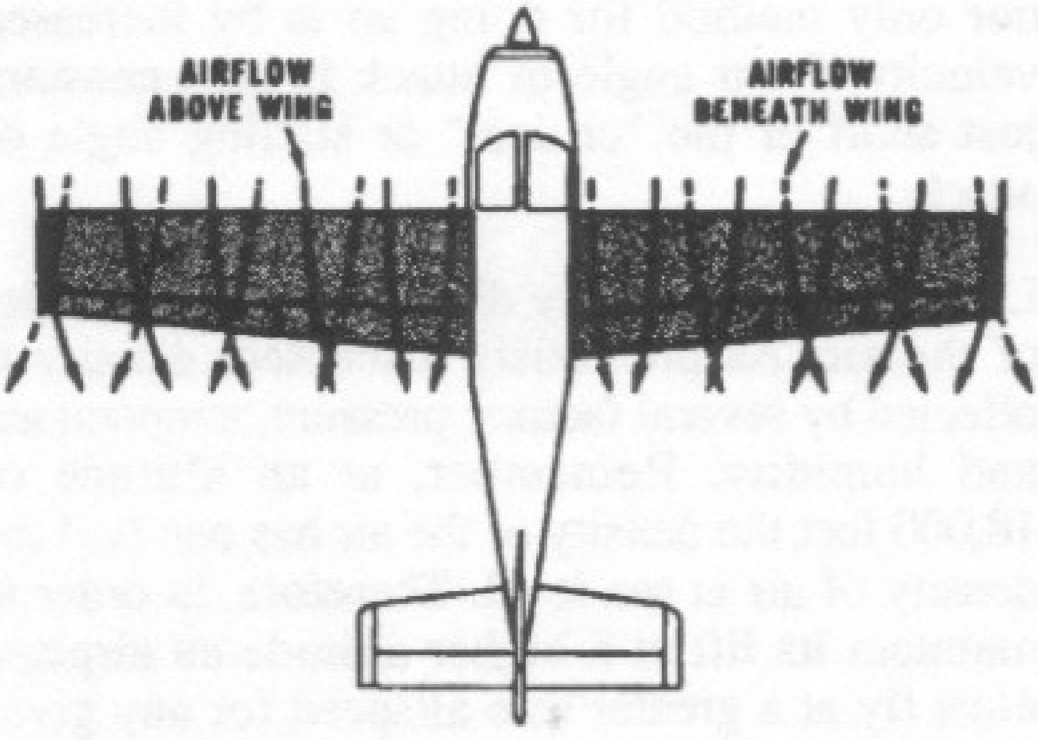
This downwash over the top of the wing at the tip has the same effect as bending the lift vector rearward; therefore, the lift is slightly aft of perpendicular to the relative wind, creating a rearward lift component. This is induced drag.

It should be remembered also that in order to create a greater negative pressure on the top of the wing, the wing can be inclined to a higher angle of attack; also, that if the angle of attack of an asymmetrical wing were zero, there would be no pressure differential and consequently no downwash component; therefore, no induced drag. In any case, as angle of attack increases, induced drag increases proportionally.

To state this another way—the lower the airspeed the greater the angle of attack required to produce lift equal to the airplane's weight and consequently, the greater will be the induced drag. The amount of induced drag varies inversely as the square of the airspeed.

To understand the effect of lift and drag on an airplane in flight we must combine them and consider the lift-drag ratio. With the lift and drag data available for various airspeeds of the airplane in steady, unaccelerated flight, the proportions of CL (Coefficient of Lift) and CD (Coefficient of Drag) can be calculated for each specific angle of attack. The resulting plot for lift-drag ratio with angle of attack shows that L/D increases to some maximum then decreases at the higher lift coefficient and angles of attack, as shown in Fig. 3-10. Note that the maximum lift-drag ratio, (L/D max) occurs at one specific angle of attack and lift coefficient. If the airplane is operated in steady flight at L/D max, the total drag is at a minimum. Any angle of attack lower or higher than that for L/D max reduces the lift-drag ratio and consequently increases the total drag for a given airplane's lift.

The location of the center of gravity (CG) is determined by the general design of each particular airplane. The designers determine how far the center of pressure (CP) will travel. They then fix the center of gravity forward of the center of pressure for the corresponding flight speed in order to provide an adequate restoring moment to retain flight equilibrium.



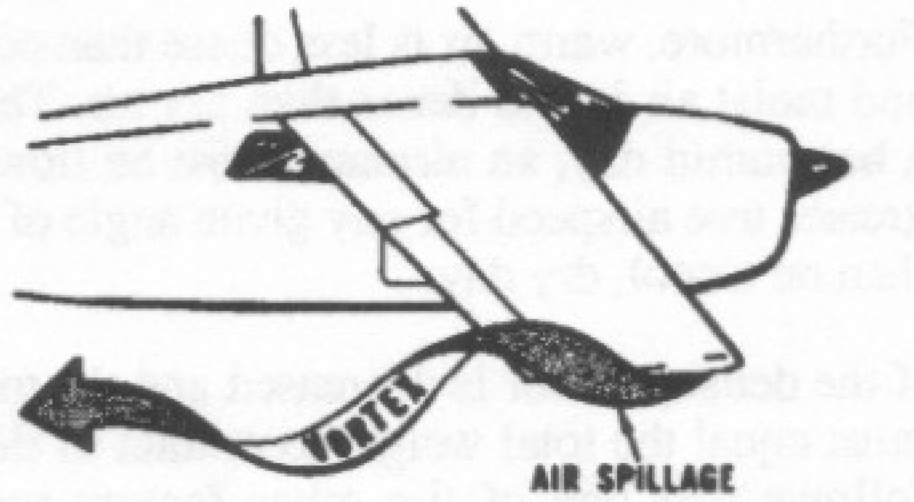


Figure 3-17 Wingtip Vortices

The configuration of an airplane has a great effect on the lift-drag ratio. The high performance sailplane may have extremely high lift-drag ratios. The supersonic fighter may have seemingly low lift-drag ratios in subsonic flight, but the airplane configurations required for supersonic flight (and high L/D's at high Mach numbers) cause this situation.

Weight

Gravity is the pulling force that tends to draw all bodies to the center of the earth. The center of gravity may be considered as a point at which all the weight of the airplane is concentrated. If the airplane were supported at its exact center of gravity, it would balance in any attitude. It will be noted that center of gravity is of major importance in an airplane, for its position has a great bearing upon stability.

The location of the center gravity (CG) is determined by the general design of each particular airplane. The designers determine how far the center of pressure (CP) will travel. They then fix the center of gravity forward of the center of pressure for the corresponding flight speed in order to provide an adequate restoring moment to retain flight equilibrium.

From the foregoing discussion we have seen that parasite drag increases as the square of the airspeed, and induced drag varies inversely as the square of the airspeed. It can be seen that as airspeed decreases to near the stalling speed the total drag becomes greater, due mainly to the sharp rise in induced drag. Similarly, as the airspeed reaches the terminal velocity of the airplane, the total drag again increases rapidly, due to the sharp increase of parasite drag. As we can see in Figure 3-18 at some given airspeed, total drag is at its maximum amount. This is very important in figuring the maximum endurance and range of airplanes. For when drag is at a minimum, power required to overcome drag is also at a minimum. The drag factors will play an important part in a later section on airplane performance.

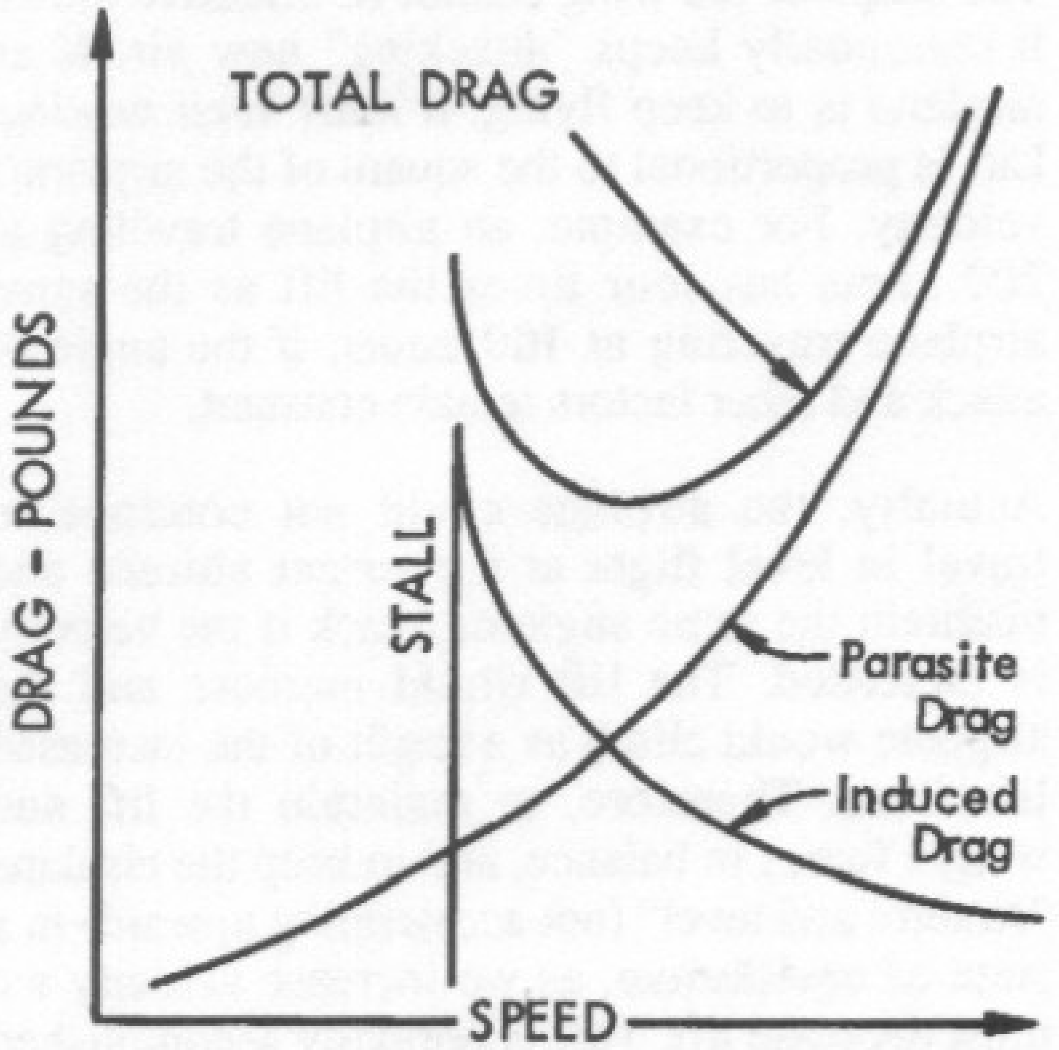


Figure 3-18 Drag Versus Speed

Weight has a definite relationship with lift, and thrust with drag. This relationship is simple, but important in understanding the aerodynamics of flying. As stated previously, lift is the upward force on the wing acting perpendicular to the relative wind. Lift is required to counteract the airplane's weight (which is caused by the force of gravity acting on the mass of the airplane). This weight (gravity) force acts downward through the airplane's center of gravity. In stabilized level flight, when the lift force is equal to the weight force, the airplane is in a state of equilibrium and neither gains nor loses altitude. If lift becomes less than weight, the airplane loses altitude. When the lift is greater than weight, the airplane gains altitude.

Lift

The pilot can control the lift. Any time the control wheel is more fore or aft, the angle of attack is changed. As angle of attack increases, lift increases (all other factors being equal). When the airplane reaches the maximum angle of attack, lift begins to diminish rapidly. This is the stalling angle of attack, or burble point.

Before proceeding further with lift and how it can be controlled, we must now interject velocity. The shape of the wing cannot be effective unless it continually keeps "attacking" new air. If an airplane is to keep flying, *it must keep moving*. Lift is proportional to the square of the airplane's velocity. For example, an airplane traveling at 200 knots has four times the lift as the same airplane traveling at 100 knots, *if* the angle of attack and other factors remain constant.

Actually, the airplane could not continue to travel in level flight at a constant altitude and maintain the same angle of attack if the velocity is increased. The lift would increase and the airplane would climb as a result of the increased lift force. Therefore, to maintain the lift and weight forces in balance, and to keep the airplane "straight and level" (not accelerating upward) in a state of equilibrium, as we increase velocity we must decrease lift. This is normally accomplished by reducing the angle of attack; i.e., lowering the nose. Conversely, as we slow the airplane, the decreasing velocity requires increasing the angle of attack to maintain lift sufficient to maintain flight. There is, of course, a limit to how far we can go in this direction, if a stall is to be avoided.

Therefore, it may be concluded that for every angle of attack there is a corresponding indicated airspeed required to maintain altitude in steady, unaccelerated flight all other factors being constant. (Bear in mind this is only true if we are maintaining "level flight.") Since an airfoil will always stall at the same angle of attack, if we increase weight we must also increase lift, and our only method for doing so is by increased velocity if our angle of attack is held constant just short of the "critical" or stalling angle of attack.

Lift and drag also vary directly with the density of the air. As previously discussed, density is affected by several factors: pressure, temperature, and humidity. Remember, at an altitude of 18,000 feet the density of the air has one-half the density of air at sea level. Therefore, in order to maintain its lift at a higher altitude an airplane must fly at a greater true airspeed for any given angle of attack.

Furthermore, warm air is less dense than cool air, and moist air is less dense than dry air. Thus, on a hot humid day, an airplane must be flown at a greater true airspeed for any given angle of attack than on a cool, dry day.

If the density factor is decreased and the total lift must equal the total weight to remain in flight, it follows that one of the other factors must be increased. The factors usually increased are the airspeed or the angle of attack, because these factors can be controlled directly by the pilot.

It should also be pointed out that lift varies directly with the wing *area*, provided there is no change in the wing's *planform*. If the wings have the same proportion and airfoil sections, a wing with a planform area of 200 square feet lifts twice as much at the same angle of attack as a wing with an area of 100 square feet.

As can be seen, two major factors from the pilot's viewpoint are lift and velocity because these are the two that can be controlled most readily and accurately. Of course, the pilot can also control density by adjusting the altitude and can control wing area if the airplane happens to have flaps of the type that enlarge wing area. However, for most situations, the pilot is controlling lift and velocity to maneuver the airplane. For instance, in straight-and-level flight, cruising along at a constant altitude, altitude is maintained by adjusting lift to match the airplane's velocity or cruise airspeed, while maintaining a state of equilibrium where lift equals weight. In an approach to landing, when the pilot wishes to land as slowly as practical, it is necessary to increase lift to near maximum to maintain lift equal to the weight of the airplane.

Wingtip Vortices

As we have already learned the action of the airfoil that gives an airplane lift also causes induced drag. It was determined that when a wing is flown at a positive angle of attack, a pressure differential exists between the upper and lower surfaces of the wing; that is—the pressure above the wing is less than atmospheric pressure and the pressure below the wing is equal to or greater than atmospheric pressure. Since air always moves from high pressure toward low pressure, and the path of least resistance is toward the airplane's wingtips, there is a spanwise movement of air from the bottom of the wing outward from the fuselage and upward around the wingtips. This flow of air results in "spillage" over the wingtips, thereby setting up a whirlpool of air called a "vortex" (Fig. 3-17). At the same time the air on the upper surface of the wing has a tendency to flow in toward the fuselage and off the trailing edge. This air current forms a similar vortex at the inboard portion of the trailing edge of the wing, but because the fuselage limits the inward flow, the vortex is insignificant. Consequently, the deviation in flow direction is greatest at the wingtips where the unrestricted lateral flow is the strongest. As the air curls upward around the wingtip it combines with the wing's downwash to form a fast-spinning trailing vortex. These vortices increase drag because of energy spent in producing the turbulence. It can be seen, then, that whenever the wing is producing lift, induced drag occurs, and wingtip vortices are created.

Just as lift increases with an increase in angle of attack, induced drag also increases. This occurs because as the angle of attack is increased, there is a greater pressure difference between the top and bottom of the wing, and a greater lateral flow of air; consequently, this causes more violent vortices to be set up, resulting in more turbulence and more induced drag.

The intensity or strength of the wingtip vortices is directly proportional to the weight of the airplane and inversely proportional to the wingspan and speed of the airplane. The heavier and slower the airplane, the greater the angle of attack and the stronger the wingtip vortices. Thus, an airplane will create wingtip vortices with maximum strength occurring during the takeoff, climb, and landing phases of flight.

Ground Effect

It is possible to fly an airplane just clear of the ground (or water) at a slightly slower airspeed than that required to sustain level flight at higher altitudes. This is the result of a phenomenon which is better known than understood even by some experienced pilots.

When an airplane in flight gets within several feet from the ground surface, a change occurs in the three dimensional flow pattern around the airplane because the vertical component of the airflow around the wing is restricted by the ground surface. This alters the wing's upwash, downwash, and wingtip vortices (Fig. 3-19). These general effects due to the presence of the ground are referred to as "ground effect." Ground effect, then, is due to the interference of the ground (or water) surface with the airflow patterns about the airplane in flight.

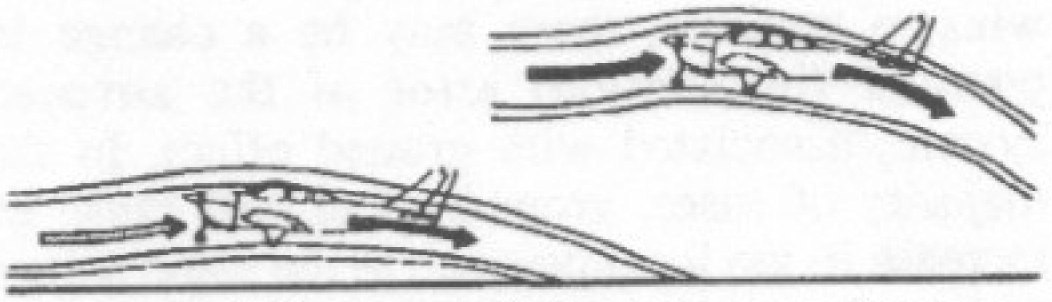


Figure 3-19 Ground Effect Changes Airflow

While the aerodynamic characteristics of the tail surfaces and the fuselage are altered by ground effects, the principal effects due to proximity of the ground are the changes in the aerodynamic characteristics of the wing. As the wing encounters ground effect and is maintained at a constant lift coefficient, there is consequent reduction in the upwash, downwash, and wingtip vortices.

We have learned earlier that induced drag is a result of the wing's work of sustaining the airplane and that the wing lifts the airplane simply by accelerating a mass of air downward. It is true that reduced pressure on top of an airfoil is essential to lift, but that is but one of the things that contributes to the overall effect of pushing an air mass downward. The more downwash there is, the harder the wing is pushing the mass of air down. At high angles of attack, the amount of induced drag is high and since this corresponds to lower airspeeds in actual flight, it can be said that induced drag predominates at low speed.

However, the reduction of the wingtip vortices due to ground effect alters the spanwise lift distribution and reduces the induced angle of attack and induced drag. Therefore, the wing will require a lower angle of attack in ground effect to produce the same lift coefficient or, if a constant angle of attack is maintained, an increase in lift coefficient will result (Fig. 3-20) .

Ground effect also will alter the thrust required versus velocity. Since induced drag predominates at low speeds, the reduction of induced drag due to ground effect will cause the most significant reduction of thrust required (parasite plus induced drag) at low speeds.

The reduction in induced flow due to ground effect causes a significant reduction in *induced drag* but causes no direct effect on parasite drag. As a result of the reduction in induced drag, the thrust required at low speeds will be reduced.

Due to the change in upwash, downwash, and wingtip vortices, there may be a change in position (installation) error of the airspeed system, associated with ground effect. In the majority of cases, ground effect will cause an increase in the local pressure at the static source and produce a lower indication of airspeed and altitude. Thus, the airplane may be airborne at an indicated airspeed less than that normally required.

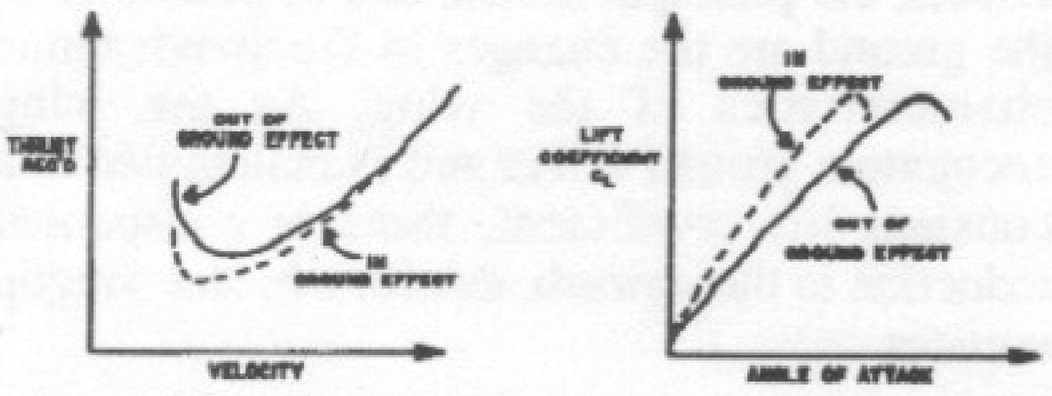


Figure 3-20 Ground Effect Changes Drag and Lift

In order for ground effect to be of significant magnitude, the wing must be quite close to the ground. As we have found, one of the direct results of ground effect is the variation of induced drag with wing height above the ground at an constant lift coefficient. When the wing is at a height equal to its span, the reduction in induced drag is only 1.4 percent. However, when the wing is at a height equal to one-fourth its span, the reduction in induced drag is 23.5 percent and, when the wing is at a height equal to one-tenth its span, the reduction in induced drag is 47.6 percent. Thus, a large reduction in induced drag will take place only when the wing is very close to the ground. Because of this variation, ground effect is most usually recognized during the liftoff for takeoff or just prior to touchdown when landing.

During the takeoff phase of flight, ground effect produces some important relationships. The airplane leaving ground effect after takeoff encounters just the reverse of the airplane entering ground effect during landing; i.e., the airplane *leaving* ground effect will (1) require an increase in angle of attack to maintain the same lift coefficient, (2) experience an increase in induced drag and thrust required, (3) experience a decrease in stability and a nose-up change in moment, (4) produce a reduction in static source pressure and increase in indicated airspeed. These general effects should point out the possible danger in attempting takeoff prior to achieving the recommended takeoff speed. Due to the reduced drag in ground effect the airplane may seem capable of takeoff well below the recommended speed. However, as the airplane rises out of ground effect with a deficiency of speed, the greater induced drag may result very marginal initial climb performance. In the extreme conditions such as high gross weight, high density altitude, and high temperature, a deficiency of airspeed during takeoff may permit the airplane to become airborne but be incapable of flying out of ground effect. In this case, the airplane may become airborne initially with a deficiency of speed, and then settle back to the runway. It is important that no attempt be made to force the airplane to become airborne with a deficiency of speed; the recommended takeoff speed is necessary to provide adequate initial climb performance. For this reason, it is imperative that a definite climb be established *before* retracting the landing gear or flaps.

During the landing phase of flight, the effect of proximity to the ground also must be understood and appreciated. If the airplane is brought into ground effect with a constant angle of attack, the airplane will experience an increase in lift coefficient and a reduction in the thrust required. Hence, a "floating" effect may occur. Because of the reduced drag and power-off deceleration in ground effect, any excess speed at the point of flare may incur a considerable "float" distance. As the airplane nears the point of touchdown, ground

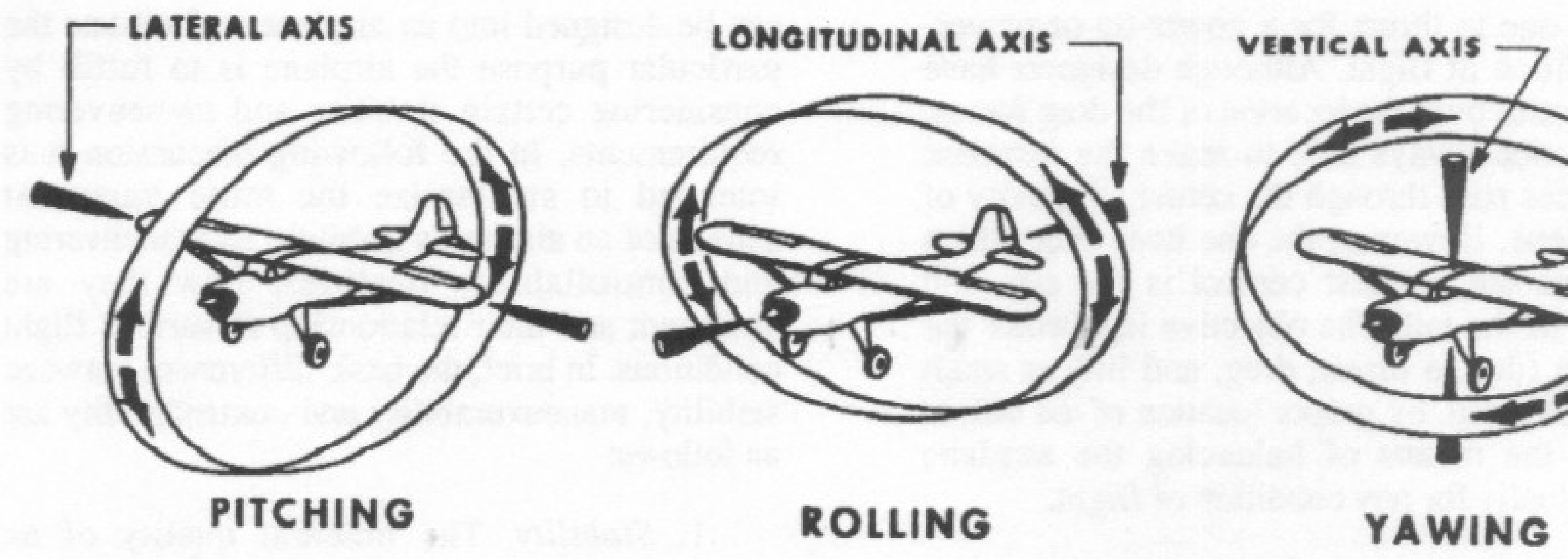


Figure 3-21 Axes of an Airplane

effect will be most realized at altitudes less than the wingspan. During the final phases of the approach as the airplane nears the ground, a reduced power setting is necessary or the reduced thrust required would allow the airplane to climb above the desired glidepath.

Axes of an Airplane

Whenever an airplane changes its flight attitude or position in flight, it rotates about one or more of three axes, which are imaginary lines that pass through the airplane's center of gravity. The axes of an airplane can be considered as imaginary axles around which the airplane turns, much like the axle around which a wheel rotates. At the point where all three axes intersect, each is at a 90° angle to the other two. The axis which extends lengthwise through the fuselage from the nose to the tail is called the longitudinal axis. The axis which extends crosswise, from wingtip to wingtip, is the lateral axis. The axis which passes vertically through the center of gravity, is called the vertical axis (Fig. 3-21).

The airplane's motion about its longitudinal axis resembles the roll of a ship from side to side. In fact, the names used in describing the motion about an airplane's three axes were originally nautical terms. They have been adapted to aeronautical terminology because of the similarity of motion between an airplane and the seagoing ship.

In light of the adoption of nautical terms, the motion about the airplane's longitudinal axis is called "roll"; motion along its lateral axis is referred to as "pitch." Finally, an airplane moves about its vertical axis in a motion which is termed "yaw"—that is, a horizontal (left and right) movement of the airplane's nose.

The three motions of the airplane—roll, pitch, and yaw—are controlled by three control surfaces. Roll is controlled by the ailerons; pitch is controlled by the elevators; yaw is controlled by the rudder.

Moments and Moment Arm

A study of physics shows that a body that is free to rotate will always turn about its center of gravity. In aerodynamic terms, the mathematical measure of an airplane's tendency to rotate about its center of gravity is called a "moment." A moment is said to be equal to the product of the force applied and the distance at which the force is applied. (A moment arm is the distance from a datum [reference point or line] to the applied force.) For airplane weight and balance computations, "moments" are expressed in terms of the distance of the arm times the airplane's weight, or simply, *inch pounds*.

As previously mentioned, airplane designers locate the fore and aft position of the airplane's center of gravity as nearly as possible to the 20 percent point of the mean aerodynamic chord (MAC). If the thrust line is designed to pass horizontally through the center of gravity, it will not cause the airplane to pitch when power is changed, and there will be no difference in moment due to thrust for a power-on or power-off condition of flight. Although designers have some control over the location of the drag forces, they are not always able to make the resultant drag forces pass through the center of gravity of the airplane. However, the one item over which they have the greatest control is the size and location of the tail. The objective is to make the moments (due to thrust, drag, and lift) as small as possible; and, by proper location of the tail, to provide the means of balancing the airplane longitudinally for any condition of flight.

The pilot has no direct control over the location of forces acting on the airplane in flight, except for controlling the center of lift by changing the angle of attack. Such a change however immediately involves changes in other forces. Therefore, the pilot cannot independently change the location of one force without changing the effect of others. For example, a change in airspeed involves a change in lift, as well as a change in drag and a change in the up or down force on the tail. As forces such as turbulence, gusts, etc., act to displace the airplane, the pilot reacts by providing opposing control forces to counteract this displacement.

Some airplanes are subject to changes in the location of the center of gravity with variations of load. Trimming devices are used to counteract the forces set up by fuel burnoff, and loading or off-loading of passengers, cargo, etc. Elevator trim tabs and adjustable horizontal stabilizers comprise the most common devices provided to the pilot for trimming for load variations. Over the wide ranges of balance during flight in large airplanes, the force which the pilot has to exert on the controls would become excessive and fatiguing if means of trimming were not provided.

Design Characteristics

Every pilot who has flown numerous types of airplanes has noted that each airplane handles somewhat differently—that is, each resists or responds to control pressures in its own way. A training type airplane is quick to respond to control applications, while a transport airplane usually feels heavy on the controls and responds to control pressures more slowly. These features can be designed into an airplane to facilitate the particular purpose the airplane is to fulfill by considering certain stability and maneuvering requirements. In the following discussion it is intended to summarize the more important aspects of an airplane's stability; its maneuvering and controllability qualities; how they are analyzed; and their relationship to various flight conditions. In brief, the basic differences between stability, maneuverability and controllability are as follows:

1. *Stability*. The inherent quality of an airplane to correct for conditions that may disturb its equilibrium, and to return or to continue on the original flightpath. It is primarily an airplane design characteristic.

2. *Maneuverability*. The quality of an airplane that permits it to be maneuvered easily and to withstand the stresses imposed by maneuvers. It is governed by the airplane's weight, inertia, size and location of flight controls, structural strength, and powerplant. It too is an airplane design characteristic.

3. *Controllability*. The capability of an airplane to respond to the pilot's control, especially with regard to flightpath and attitude. It is the quality of the airplane's response to the pilot's control application when maneuvering the airplane, regardless of its stability characteristics.

Basic Concepts of Stability

The flightpaths and attitudes in which an airplane can fly are limited only by the aerodynamic characteristics of the airplane, its propulsive system, and its structural strength. These limitations indicate the maximum performance and maneuverability of the airplane. If the airplane is to provide maximum utility, it must be safely controllable to the full extent of these limits without exceeding the pilot's strength or requiring exceptional flying ability. If an airplane is to fly straight and steady along any arbitrary flightpath, the forces acting on it must be in static equilibrium. The reaction of any body when its equilibrium is disturbed is referred to as stability. There are two types of stability; static and dynamic. We will first consider the static, and in this discussion the following definitions

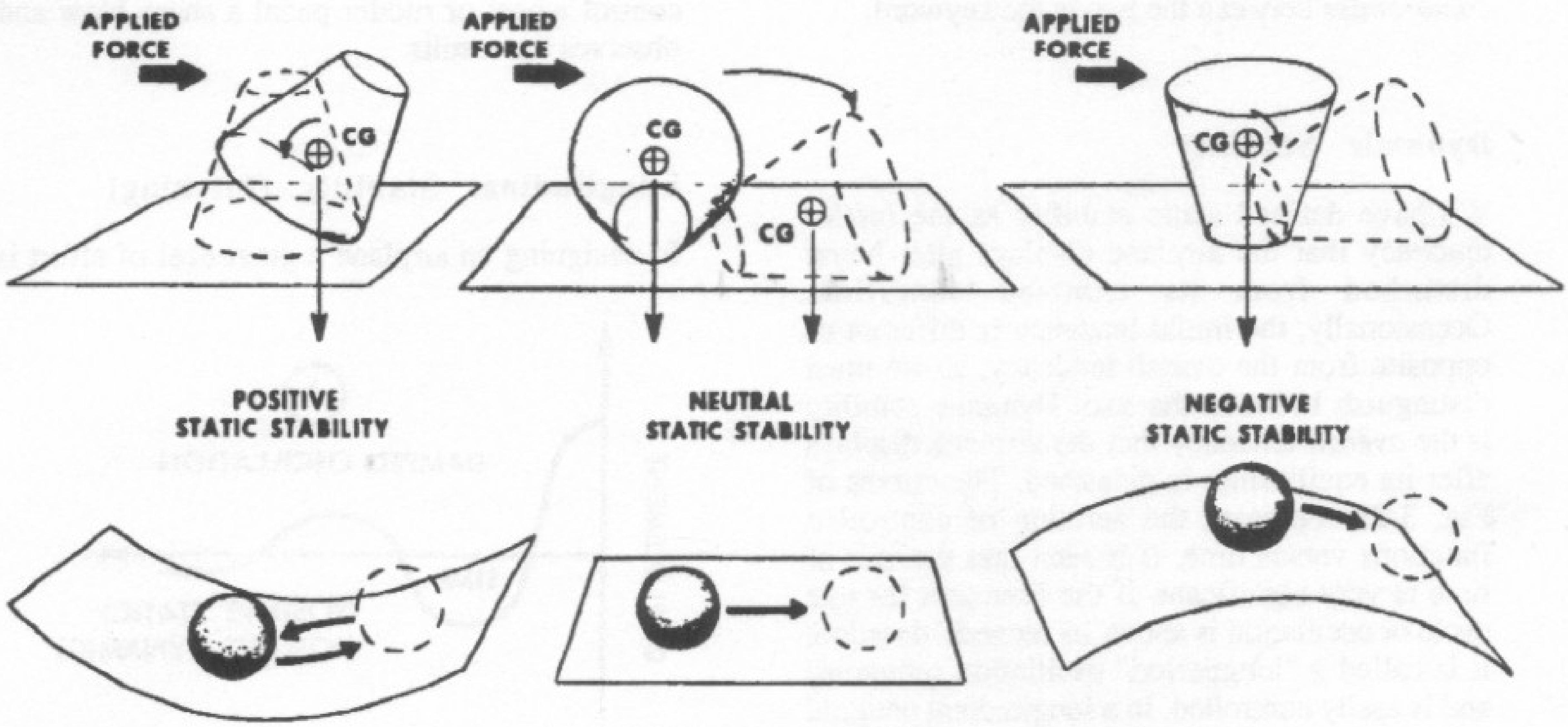


Figure 3-22 Types of Stability

will apply:

1. *Equilibrium*. All opposing forces acting on the airplane are balanced; (i.e.. steady unaccelerated flight conditions).

2. *Static Stability*. The initial tendency that the airplane displays after its equilibrium is disturbed.

3. *Positive Static Stability*. The initial tendency of the airplane to return to the original state of equilibrium after being disturbed (Fig. 3-22).

4. *Negative Static Stability*. The initial tendency of the airplane to continue away from the original state of equilibrium after being disturbed (Fig. 3-22).

5. *Neutral Static Stability*. The initial tendency of the airplane to remain in a new condition after its equilibrium has been distrurbed (Fig. 3-22).

Static Stability

Stability of an airplane in flight is slightly more complex than just explained, because the airplane is free to move in any direction and must be controllable in pitch, roll, and direction. When designing the airplane, engineers must compromise between stability, maneuverability, and controllability; and the problem is compounded because of the airplane's three-axis freedom.

By comparing this three-axis freedom with the limited flexibility of an automobile, the problem may be better understood. Pitch control of a car is limited to a shock absorbing and leveling problem while roll control is a matter of the lateral spacing of the wheels and the banking of highways around curves. Essentially, then, the car has only one degree of real freedom—directional; in many cases, though, the lack of pitch and bank control have caused it to lose directional control. Automobile designers provide "centering" in the steering mechanism or "toe-in ' on the wheels to provide directional stability so the car will maintain a straight path when the steering wheel is released. Too much centering or "toe-in" is, of course, objectionable because of the excessive effort required of the driver to turn the car. When power steering was introduced to the driving public, many considered it objectionable because of insufficient centering or "feel." Even now, most atomically models are offered either with or without power steering to satisfy the public.

The conclusions, then, are that too much stability is detrimental to maneuverability, and similarly, not enough stability is detrimental to controllability. In the design of airplanes, *compromise* between the two is the keyword.

Dynamic Stability

We have defined static stability as the *initial* tendency that the airplane displays after being disturbed from its trimmed condition. Occasionally, the initial tendency is different or opposite from the overall tendency, so we must distinguish between the two. Dynamic stability is the *overall* tendency that the airplane displays after its equilibrium is disturbed. The curves of Fig. 3-23 represent the aeration of controlled functions versus time. It is seen that the unit of time is very significant. If the time unit for one cycle or oscillation is above 10 seconds' duration, it is called a "longperiod" oscillation (phugoid) and is easily controlled. In a longitudinal phugoid oscillation, the angle of attack remains constant when the airspeed increases and decreases. To a certain degree a convergent phugoid is desirable but is not required. The phugoid can be determined only on a statically stable airplane, and this has a great effect on the trimming qualities of the airplane. If the time unit for one cycle or oscillation is less than one or two seconds, it is called a "short-period" oscillation and is normally very difficult, if not impossible, for the pilot to control. This is the type of oscillation that the pilot can easily "get in phase with" and reinforce.

A neutral or divergent, short-period oscillation is dangerous because structural failure usually results if the oscillation is not damped immediately. Short-period oscillations affect airplane and control surfaces alike and reveal themselves as "purposing" in the airplane, or as in "buzz" or "flutter" in the control surfaces. Basically, the short-period oscillation is a change in angle of attack with no change in airspeed. A short-period oscillation of a control surface is usually of such high frequency that the airplane does not have time to react. Logically, Federal Aviation Regulations require that short-period oscillations be heavily damped (i.e., die out immediately). Flight tests during the airworthiness certification of airplanes are conducted for this condition by inducing the oscillation in the controls for pitch roll, or yaw at the most critical speed (i.e., at VNE,, the never-exceed speed). The test pilot strikes the control wheel or rudder pedal a sharp blow and observes the results.

Longitudinal Stability (Pitching)

In designing an airplane a great deal of effort is

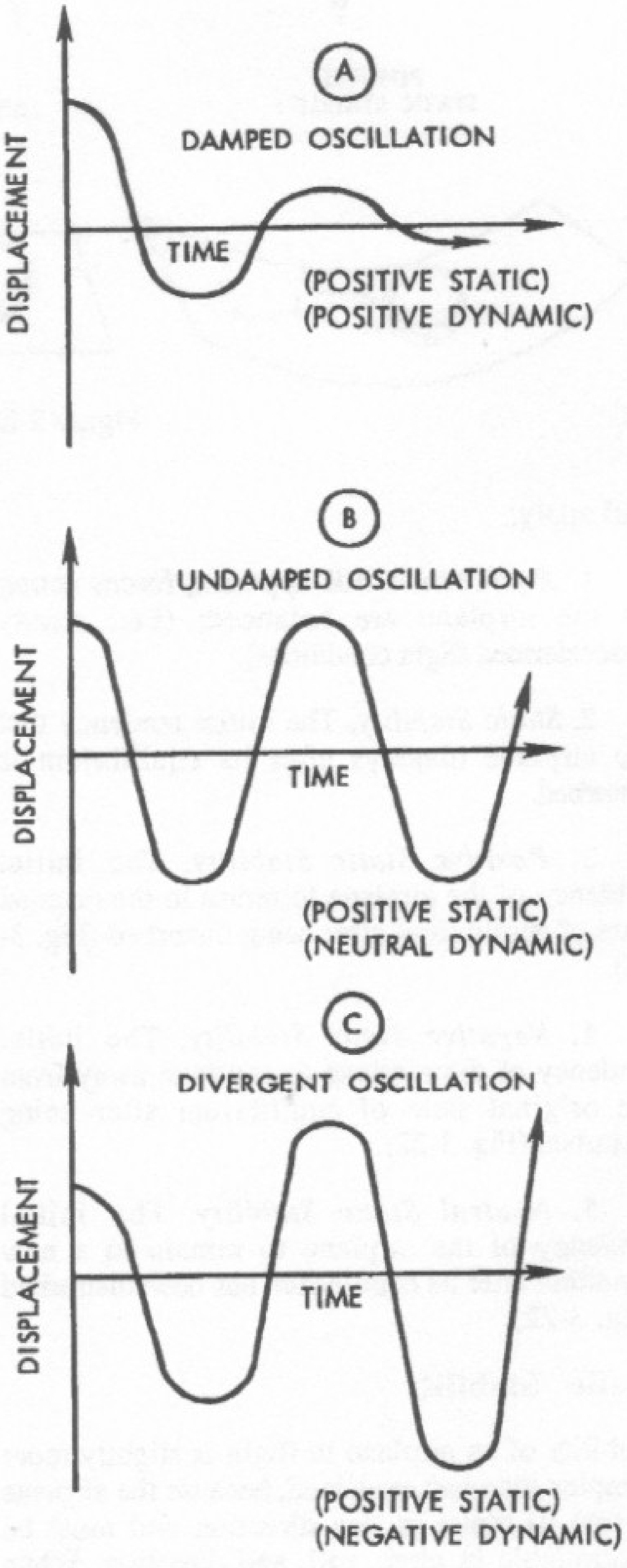


Figure 3-23 Damped Versus Undamped Stability

spent in developing the desired degree of stability around all three axes. But longitudinal stability about the lateral axis is considered to be the most affected by certain variables in various flight conditions.

As we learned earlier, longitudinal stability is the quality which makes an airplane stable about its lateral axis. It involves the pitching motion as the airplane's nose moves up and down in flight. A longitudinally *unstable* airplane has a tendency to dive or climb progressively into a very steep dive or climb, or even a stall. Thus, an airplane with longitudinal instability becomes difficult and sometimes dangerous to fly.

Static longitudinal stability or instability in an airplane, is dependent upon three factors:

1. Location of the wing with respect to the center of gravity;

2. Location of the horizontal tail surfaces with respect to the center of gravity; and

3. The area or size of the tail surfaces.

In analyzing stability it should be recalled that a body that is free to rotate will always turn about its center of gravity.

To obtain static longitudinal stability, the relation of the wing and tail moments must be such that, if the moments are initially balanced and the airplane is suddenly nosed up, the wing moments and tail moments will change so that the sum of their forces will provide an unbalanced but restoring moment which in turn, will bring the nose down again. Similarly, if the airplane is nosed down the resulting change in moments will bring the nose back up.

We have spoken of the airplane's center of gravity and the airfoil's center of lift in preceding sections. Now let us reexamine the center of lift or as it is sometimes called, the *center of pressure.*

As previously pointed out, the center of pressure in most unsymmetrical airfoils has a tendency to change its fore and aft position with a change in the angle of attack. The center of pressure tends to move *forward* with an increase in angle of attack and to move aft with a decrease in angle of attack. This means that when the angle of attack of an airfoil is increased, the center of pressure (lift) by moving forward, tends to lift the lending edge of the wing still more. This tendency gives the wing an inherent quality of ***instability***.

Figure 3-24, shows an airplane in straight-and-level flight. The line CG-CL-T represents the airplane's longitudinal axis from the center of gravity (CG) to a point T on the horizontal stabilizer. The center of lift (or center of pressure) is represented by the point CL.

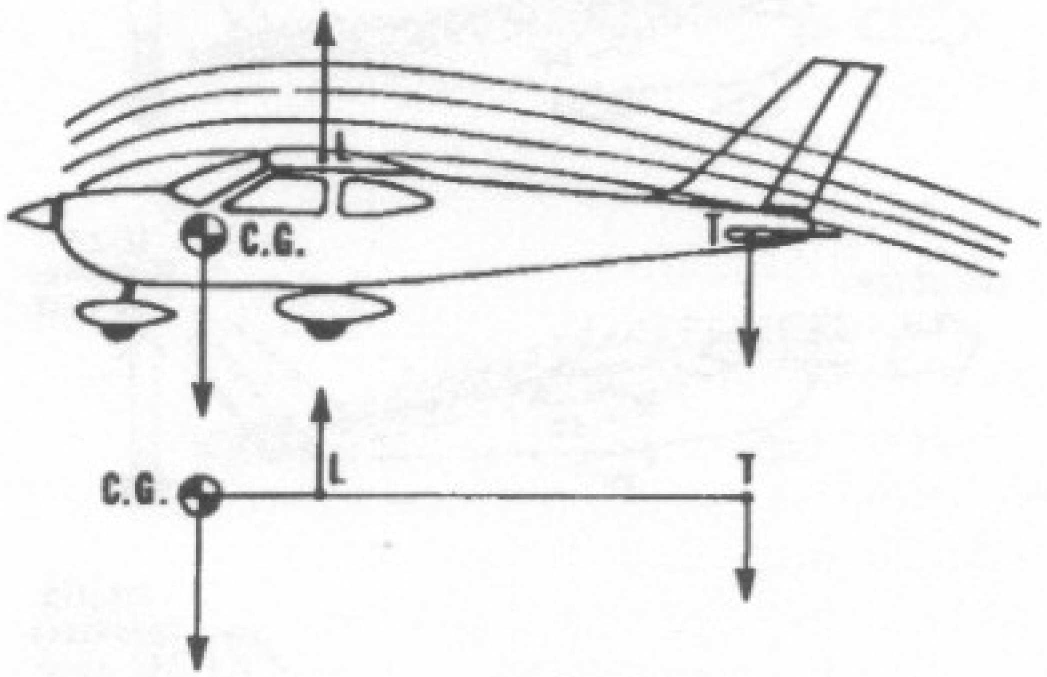


Figure 3-24 Longitudinal Stability

Most airplanes are designed so that the wing's center of lift (CL) is to the rear of the center of gravity. This makes the airplane "nose heavy" and requires that there be a slight downward force on the horizontal stabilizer in order to balance the airplane and keep the nose from continually pitching downward. Compensation for this nose heaviness is provided by setting the horizontal stabilizer at a slight negative angle of attack. The downward force thus produced, holds the tail down, counterbalancing the "heavy" nose. It is as if the line CG-CL-T was a lever with an upward force at CL and two downward forces balancing each other, one a strong force at the CG point and the other, a much lesser force, at point T (downward air pressure on the stabilizer). Applying simple physics principles, it can be seen that if an iron bar were suspended at point CL with a heavy weight hanging on it at the CG, it would take some downward pressure at point T to keep the "lever" in balance.

Even though the horizontal stabilizer may be level when the airplane is in level flight, there is a downwash of air from the wings. This downwash strikes the top of the stabilizer and produces a downward pressure which, at a certain speed, will be just enough to balance the "lever."

The faster the airplane is flying, the greater this downwash and the greater the downward force on the horizontal stabilizer (except "T" tails) (Fig.25). In airplanes with fixed position horizontal stabilizers, the airplane manufacturer sets the stabilizer at an angle that will provide the best stability (or balance) during flight at the design cruising speed and power setting (Fig.26).

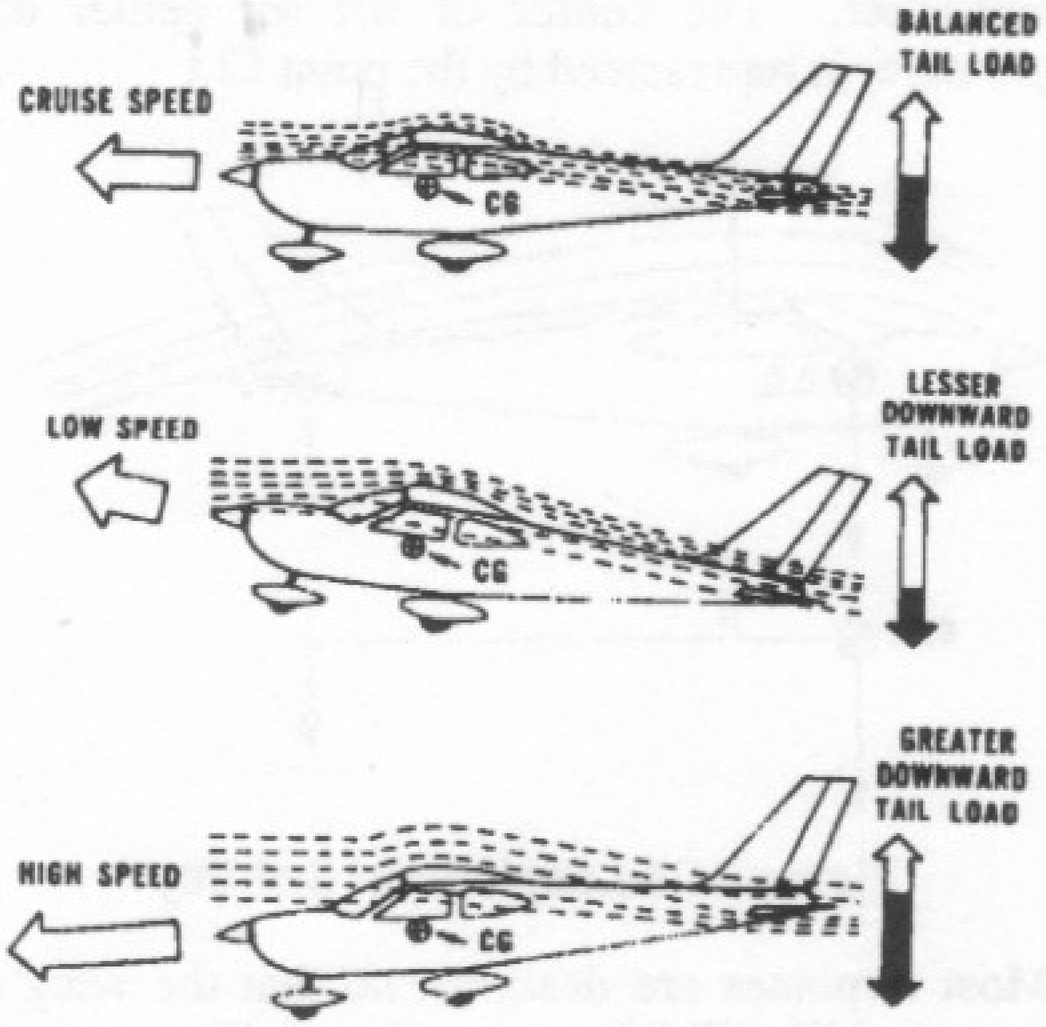


Figure 3-25 Effect of Speed on Downwash

If the airplane's speed decreases, the speed of the airflow over the wing is decreased. As a result of this decreased flow of air over the wing, the downwash is reduced, causing a lesser downward force on the horizontal stabilizer. In turn, the characteristic nose heaviness is accentuated, causing the airplane's nose to pitch down more. This places the airplane in a nose-low attitude, lessening the wing's angle of attack and drag and allowing the airspeed to increase. As the airplane continues in the nose-low attitude and its speed increases, the downward force on the horizontal stabilizer is once again increased. Consequently, the tail is again pushed downward and the nose rises into a climbing attitude.

As this climb continues, the airspeed again decreases, causing the downward force on the tail to decrease until the nose lowers once more. However, because the airplane is dynamically stable, the nose does not lower as far this time as it did before. The airplane will acquire enough speed in this more gradual dive to start it into

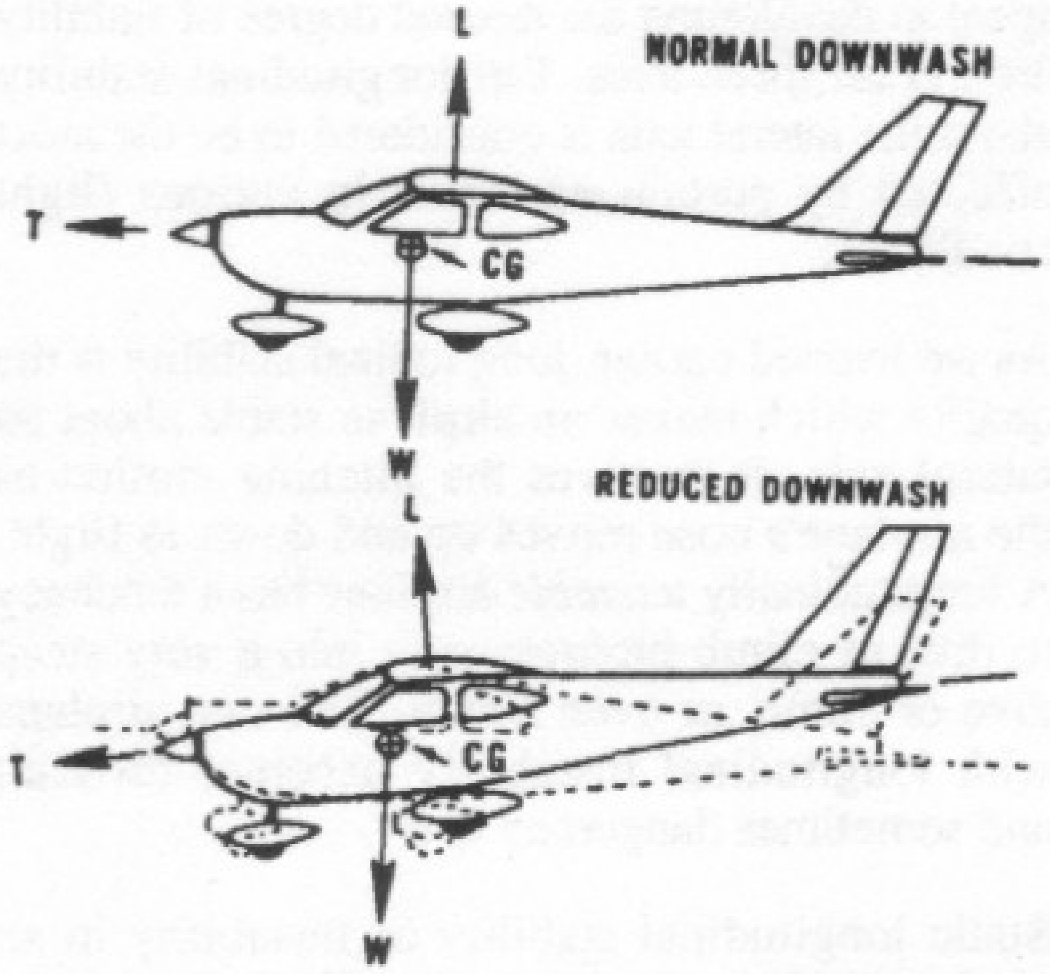


Figure 3-26 Reduced Power Allows Pitch Down

another climb, but the climb is not so steep as the preceding one.

After several of these diminishing oscillations, in which the nose alternately rises and lowers, the airplane will finally settle down to a speed at which the downward force on the tail exactly counteracts the tendency of the airplane to dive. When this condition is attained the airplane will once again be in balanced flight and will continue in stabilized flight as long as this attitude and airspeed are not changed.

A similar effect will be noted upon closing the throttle. The downwash of the wings is reduced and the force at T in Fig. 3-24 is not enough to hold the horizontal stabilizer down. It is as if the force at T on the lever were allowing the force of gravity to pull the nose down. This, of course, is a desirable characteristic because the airplane is inherently trying to regain airspeed and reestablish the proper balance.

Power or thrust can also have a destabilizing effect in that an increase of power may tend to make the nose rise. The airplane designer can offset this by establishing a "high thrustline" wherein the line of thrust passes above the center of gravity (Figs. 3-27, 3-28). In this case, as power or thrust is increased a moment is produced to counteract the down load on the tail. On the other hand, a very "low thrust line" would tend to add to the nose-up effect of the horizontal tail surface.

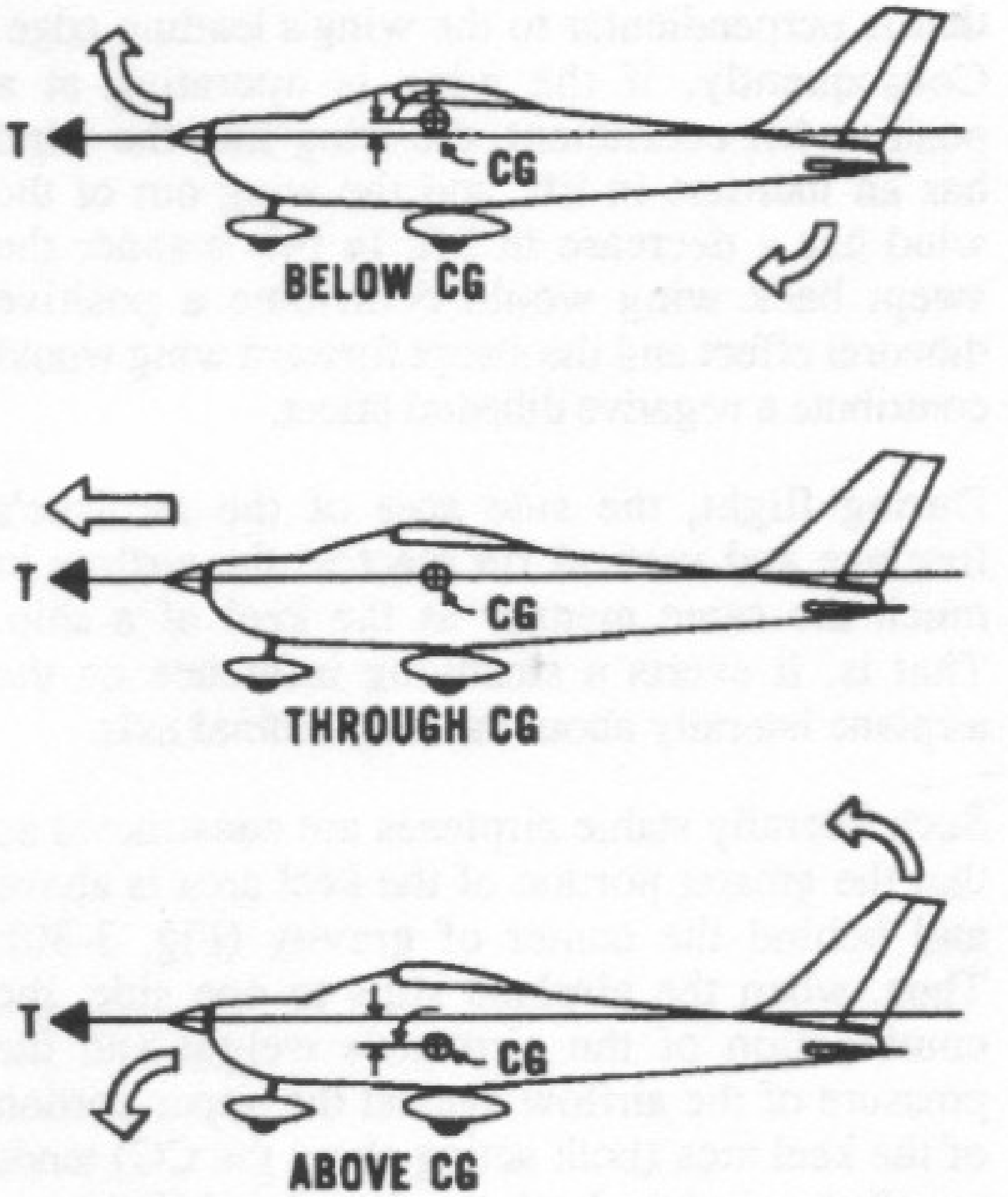


Figure 3-27 Thrust Line Affects Longitudinal Stability

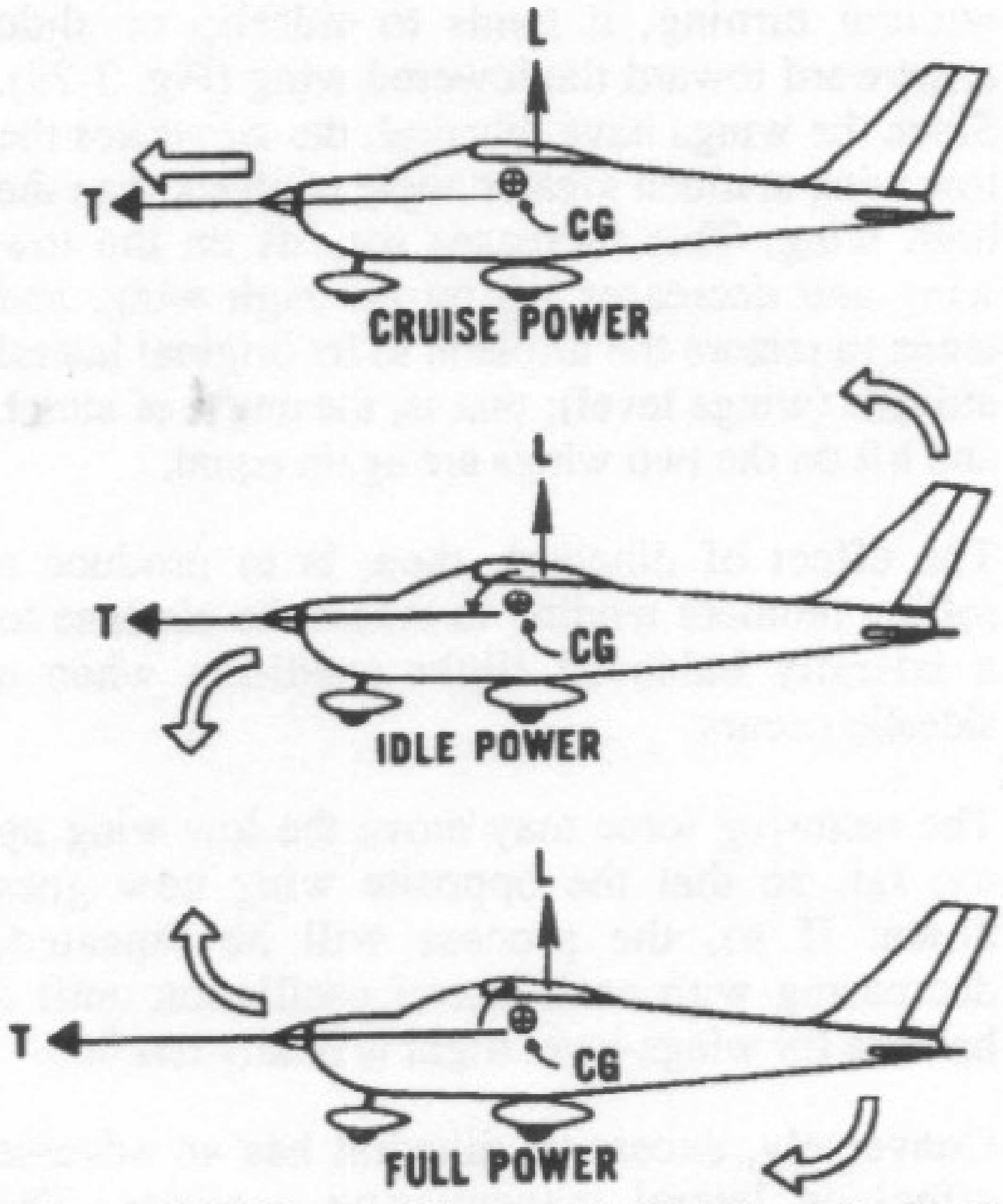


Figure 3-28 Power Changes Affect Longitudinal Stability

It can be concluded, then, that with the center of gravity forward of the center of lift, and with an aerodynamic tail-down force, the result is that the airplane always tries to return to a safe flying attitude.

A simple demonstration of longitudinal stability may he made as follows: Trim the airplane for "hands off" control in level flight. Then momentarily give the controls a slight push to nose the airplane down. If, within a brief period, the nose rises to the original position and then stops, the airplane is statically stable. Ordinarily, the nose will pass the original position (that of level flight) and a series of slow pitching oscillations will follow. If the oscillations gradually cease, the airplane has positive stability; if they continue unevenly the airplane has neutral stability; if they increase the airplane is unstable.

Lateral Stability (Rolling)

Stability about the airplane's longitudinal axis, which extends from nose to tail, is called lateral stability. This helps to stabilize the lateral or rolling effect when one wing gets lower than the wing on the opposite side of the airplane. There are four main design factors which make an airplane stable laterally—dihedral, keel effect, sweepback, and weight distribution. It will be seen in later discussions that these factors also aid in producing yawing or directional stability.

The most common procedure for producing lateral stability is to build the wings with a *dihedral angle* varying from one to three degrees. In other words, the wings on either side of the airplane join the fuselage to form a slight V or angle called "dihedral," and this is measured by the angle made by each wing above a line parallel to the lateral axis.

The basis of rolling stability is, of course, the lateral balance of forces produced by the airplane's wings. Any imbalance in lift results in a tendency for the airplane to roll about its longitudinal axis. Stated another way, dihedral involves a balance of lift created by the wings angle of attack on each side of the airplane's longitudinal axis.

If a momentary gust of wind forces one wing of the airplane to rise and the other to lower, the airplane will bank. When the airplane is banked without turning, it tends to sideslip or slide downward toward the lowered wing (Fig. 3-29). Since the wings have dihedral, the air strikes the low wing at much greater angle of attack than the high wing. This increases the lift on the low wing and decreases lift on the high wing, and tends to restore the airplane to its original lateral attitude (wings level); that is, the angle of attack and lift on the two wings are again equal.

The effect of dihedral, then, is to produce a rolling moment tending to return the airplane to a laterally balanced flight condition when a sideslip occurs.

The restoring force may move the low wing up too far, so that the opposite wing now goes down. If so, the process will be repeated, decreasing with each lateral oscillation until a balance for wings-level flight is finally reached.

Conversely, excessive dihedral has an adverse effect on lateral maneuvering qualities. The airplane may be so stable laterally that it resists any intentional rolling motion. For this reason, airplanes which require fast roll or banking characteristics usually have less dihedral than those which are designed for less maneuverability.

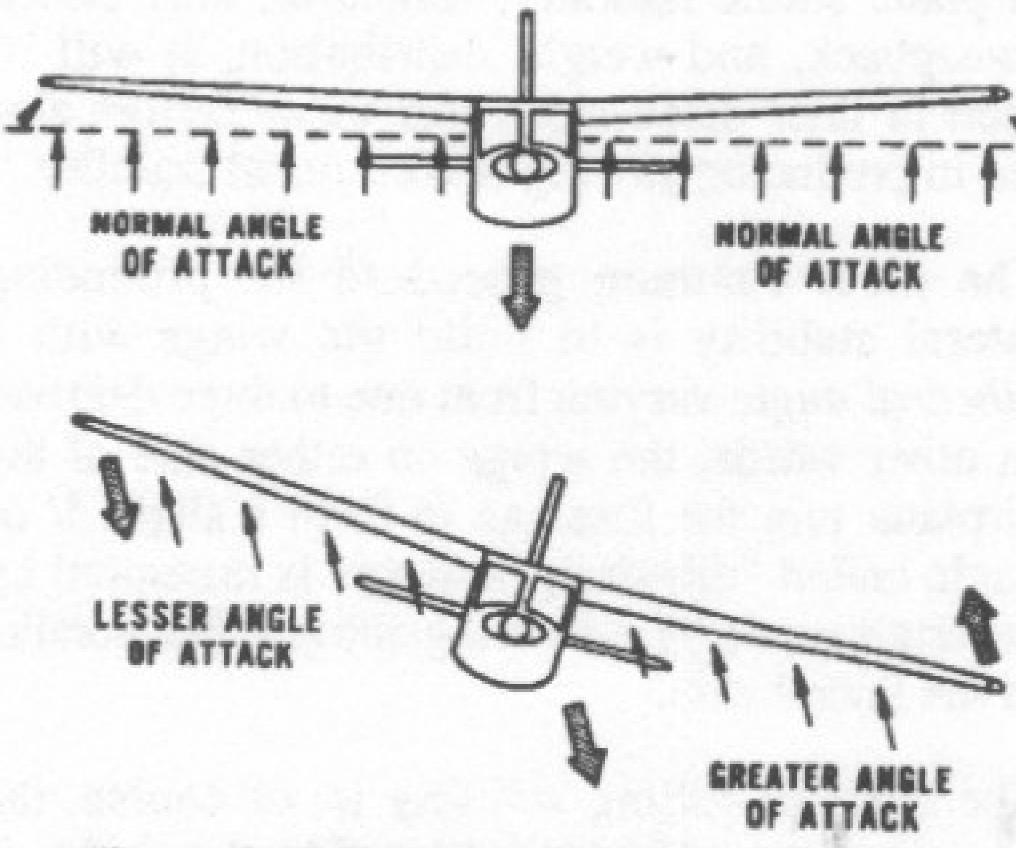


Figure 3-29 Dihedral for Lateral Stability

The contribution of *sweepback* to dihedral effect is important because of the nature of the contribution. In a sideslip, the wing into the wind is operating with an effective decrease in sweepback while the wing out of the wind is operating with an effective increase in sweepback. The reader will recall that the swept wing is responsive only to the wind component that is perpendicular to the wing's leading edge. Consequently, if the wing is operating at a positive lift coefficient, the wing into the wind has an increase in lift, and the wing out of the wind has a decrease in lift. In this manner the swept back wing would contribute a positive dihedral effect and the swept forward wing would contribute a negative dihedral effect.

During flight, the side area of the airplane's fuselage and vertical fin react to the airflow in much the same manner as the keel of a ship. That is, it exerts a steadying influence on the airplane laterally about the longitudinal axis.

Such laterally stable airplanes are constructed so that the greater portion of the keel area is above and behind the center of gravity (Fig. 3-30). Thus, when the airplane slips to one side, the combination of the airplane's weight and the pressure of the airflow against the upper portion of the keel area (both acting about the CG) tends to roll the airplane back to wings-level flight.

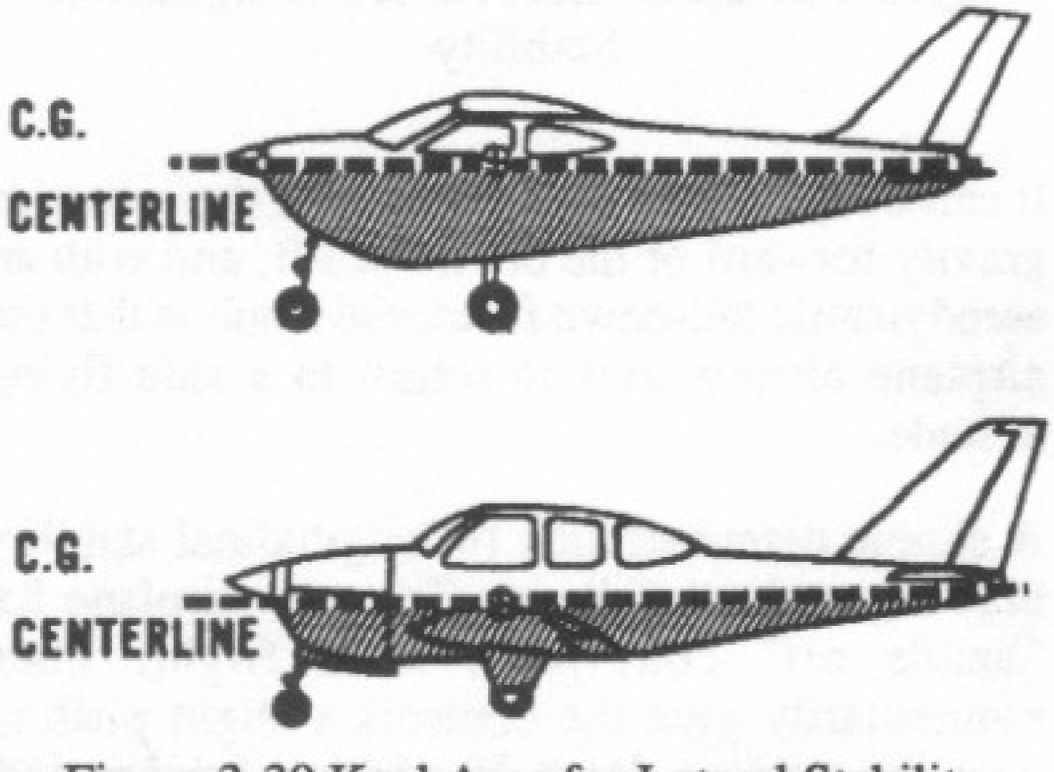


Figure 3-30 Keel Area for Lateral Stability

Vertical Stability (Yawing)

Stability about the airplane's vertical axis (the sideways moment), is called yawing or directional stability.

Yawing or directional stability is the more easily achieved stability in airplane design.The area of the vertical fin and the sides of the fuselage aft of the center of gravity are the prime contributors which make the airplane act like the well known weathervane or arrow, pointing its nose into the relative wind.

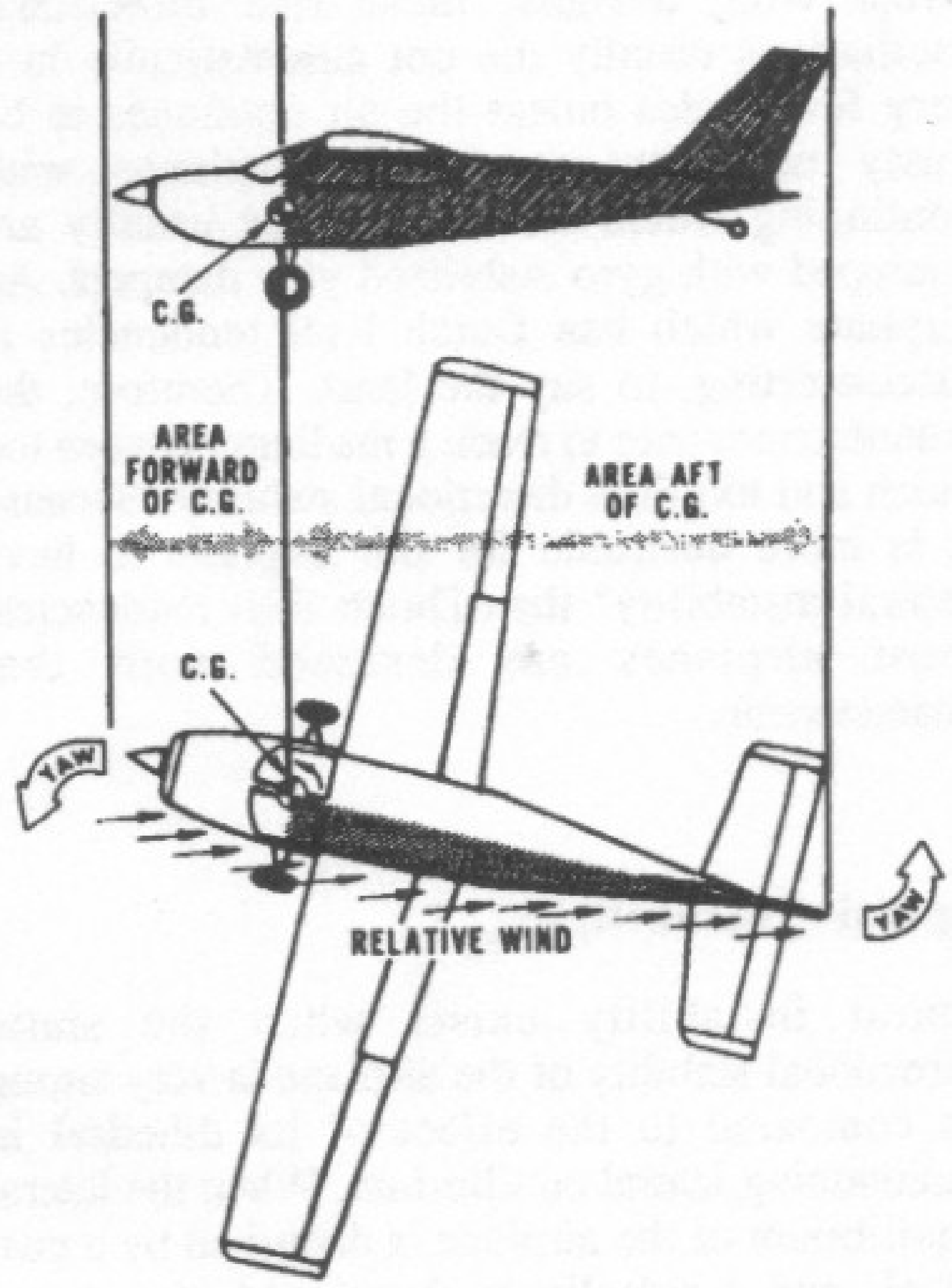


Figure 3-31 Fuselage and Fin for Vertical Stability

In examining a weathervane it can be seen that if exactly the same amount of surface were exposed to the wind in front of the pivot point as behind it, the forces fore and aft would be in balance and little or no directional movement would result. Consequently, it is necessary to have a greater surface aft of the pivot point than forward of it.

Similarly in an airplane, the designer must ensure positive directional stability by making the side surface greater aft than ahead of the center of gravity (Fig. 3-31) . To provide more positive stability aside from that provided by the fuselage, a vertical fin is added. The fin acts similar to the feather on an arrow in maintaining straight flight. Like the weathervane and the arrow the farther aft this fin is placed and the larger its size, the greater the airplane's directional stability.

If an airplane is flying in a straight line and a sideward gust of air gives the airplane a slight rotation about its vertical axis (i.e., the right) the motion is retarded and stopped by the fin because while the airplane is rotating to the right, the air is striking the left side of the fin at an angle. This causes pressure on the left side of the fin, which resists the turning motion and slows down the airplane's yaw. In doing so it acts somewhat like the weathervane by turning the airplane into the relative wind.

The initial change in direction of the airplane's flightpath is generally slightly behind its change of heading. Therefore, after a slight yawing of the airplane to the right, there is a brief moment when the airplane is still moving along its original path, but its longitudinal axis is pointed slightly to the right.

The airplane is then momentarily skidding sideways, and during that moment (since we assume that although the yawing motion has stopped, the excess pressure on the left side of the fin still persists) there is necessarily a tendency for the airplane to be turned partially back to the left. That is, there is a momentary restoring tendency caused by the fin.

This restoring tendency is relatively slow in developing and ceases when the airplane stops skidding. When it ceases, the airplane will be flying in a direction slightly different from the original direction. In other words, it will not of its own accord return to the original heading; the pilot must reestablish the initial heading.

A minor improvement of directional stability may be obtained through sweepback. Sweepback is incorporated in the design of the wing primarily to delay the onset of compressibility during high speed flight. In lighter and slower airplanes, sweepback aids in locating the center of pressure in the correct relationship with the center of gravity. As we have learned, a longitudinally stable airplane is built with the center of pressure aft of the center of gravity.

Because of structural reasons, airplane designers sometimes cannot attach the wings to the fuselage at the exact desired point. If they had to mount the wings too far forward, and at right angles to the fuselage, the center of pressure would not be far enough to the rear to effect the desired amount of longitudinal stability. By building sweepback into the wings, however, the designers can move the center of pressure toward the rear. The amount of sweepback and the position of the wings then place the center of pressure in the correct location.

The contribution of the *wing* to static directional stability is usually small. The swept wing provides a stable contribution depending on the amount of sweepback but the contribution is relatively small when compared with other components.

Free Directional Oscillations (Dutch Roll)

Dutch roll is a coupled lateral—directional oscillation which is usually dynamically stable but is objectionable in an airplane because of the oscillatory nature. The damping of the oscillatory mode may be weak or strong depending on the properties of the particular airplane.

Unfortunately all air is not smooth. There are bumps and depressions created by gusty updrafts and downdrafts, and by gusts from ahead, behind, or the side of the airplane.

The response of the airplane to a disturbance from equilibrium is a combined rolling—yawing oscillation in which the rolling motion is phased to precede the yawing motion. The yawing motion is not too significant, but the roll is much more noticeable. When the airplane rolls back toward level flight in response to dihedral effect, it rolls back too far and sideslips the other way. Thus, the airplane overshoots each time because of the strong dihedral effect. When the dihedral effect is large in comparison with static directional stability, the Dutch roll motion has weak damping and is objectionable. When the static directional stability is strong in comparison with the dihedral effect, the Dutch Roll motion has such heavy damping that it is not objectionable. However, these qualities tend toward spiral instability.

The choice is then the least of two evils, Dutch Roll is objectionable, and spiral instability is tolerable if the rate of divergence is low. Since the more important handling qualities are a result of high static directional stability and minimum necessary dihedral effect, most airplanes demonstrate a mild spiral tendency. This tendency would be indicated to the pilot by the fact that the airplane cannot be flown "hands off" indefinitely.

In most modern airplanes, excepting high speed swept wing designs, these free directional oscillations usually die out automatically in a very few cycles unless the air continues to be gusty or turbulent. Those airplanes with continuing Dutch Roll tendencies usually are equipped with gyro stabilized yaw dampers. An airplane which has Dutch Roll tendencies is disconcerting, to say the least. Therefore, the manufacturer tries to reach a medium between too much and too little directional stability. Because it is more desirable for the airplane to have "spiral instability" than Dutch Roll tendencies, most airplanes are designed with that characteristic.

Spiral Instability

Spiral instability exists when the static directional stability of the airplane is very strong as compared to the effect of its dihedral in maintaining lateral equilibrium. When the lateral equilibrium of the airplane is disturbed by a gust of air and a sideslip is introduced, the strong directional stability tends to yaw the nose into the resultant relative wind while the comparatively weak dihedral lags in restoring the lateral balance. Due to this yaw, the wing on the outside of the turning moment travels forward faster than the inside wing and as a consequence, its lift becomes greater. This produces an overbanking tendency which, if not corrected by the pilot, will result in the bank angle becoming steeper and steeper. At the same time, the strong directional stability which yaws the airplane into the relative wind is actually forcing the nose to a lower pitch attitude. We then have the start of a slow downward spiral which, if not counteracted by the pilot, will gradually increase into a steep spiral dive. Usually the rate of divergence in the spiral motion is so gradual that the pilot can control the tendency without any difficulty.

All airplanes are affected to some degree by this characteristic although they may be inherently stable in all other normal parameters. This tendency would be indicated to the pilot by the fact that the airplane cannot be flown "hands off" indefinitely.

Much study and effort has gone into development of control devices (wing leveler) to eliminate or at least correct this instability. Advanced stages

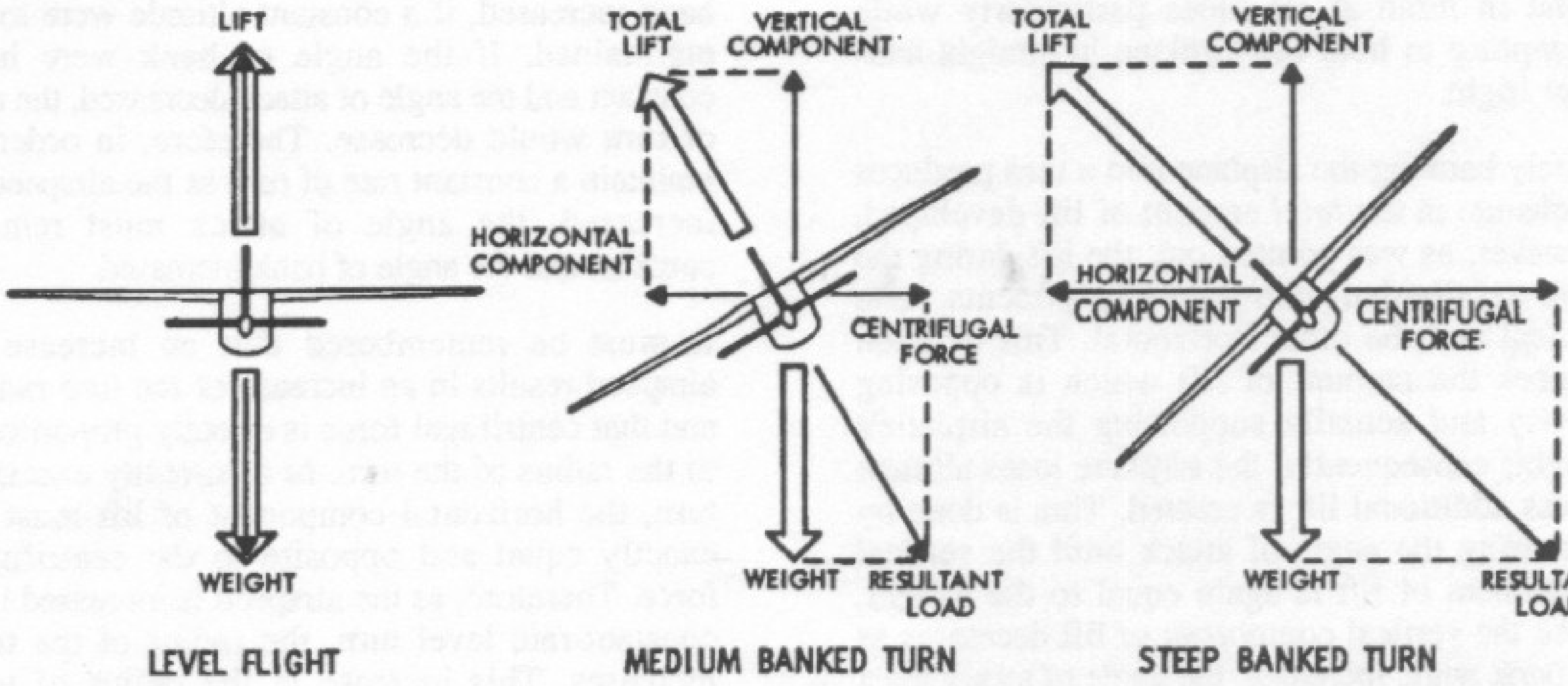


Figure 3-32 Forces During Normal Coordinated Turn

of this spiral condition demand that the pilot be very careful in application of recovery controls, or excessive loads on the structure may be imposed. Of the inflight structural failures that have occurred in general aviation airplanes, improper recovery from this condition has probably been the underlying cause of more fatalities than any other single factor. The reason is that the airspeed in the spiral condition builds up rapidly, and the application of back elevator force to reduce this speed and to pull the nose up only "tightens the turn," increasing the load factor. The results of the prolonged uncontrolled spiral are always the same; either inflight structural failure, crashing into the ground, or both. The most common causes on record for getting into this situation are: loss of horizon reference, inability of the pilot to control the airplane by reference to instruments, or a combination of both.

Forces in Turns

If an airplane were viewed in straight-and-level flight from the rear (Fig. 3-32), and if the forces acting on the airplane actually could be seen, two forces (lift and weight) would be apparent, and if the airplane were in a bank it would be apparent that lift did not act directly opposite to the weight—it now acts in the direction of the bank. The fact that when the airplane banks, lift acts inward toward the center of the turn, as well as upward, is one of the basic truths to remember in the consideration of turns.

As we learned earlier, an object at rest or moving in a straight line will remain at rest or continue to move in a straight line until acted on by some other force. An airplane, like any moving object, requires a sideward force to make it turn. In a normal turn, this force is supplied by banking the airplane so that lift is exerted inward as well as upward. The force of lift during a turn is separated into two components at right angles to each other. One component which acts vertically and opposite to the weight (gravity) is called the "vertical component of lift." The other which acts horizontally toward the center of the turn is called the "horizontal component of lift." The horizontal component of lift is the force that pulls the airplane from a straight flightpath to make it turn. Centrifugal force is the "equal and opposite reaction" of the airplane to the change in direction and acts equal and opposite to the horizontal component of lift. This explains why, in a correctly executed turn, the force that turns the airplane is not supplied by the rudder.

An airplane is not steered like a boat or an automobile; in order for it to turn, it must be banked. If the airplane is not banked, there is no force available that will cause it to deviate from a straight flightpath. Conversely, when an airplane is banked, it will turn, provided it is not slipping to the inside of the turn. Good directional control is based on the fact that the airplane will attempt to turn whenever it is banked. This fact should be borne in mind at all times particularly while attempting to hold the airplane in straight-and-level flight.

Merely banking the airplane into a turn produces no change in the *total* amount of lift developed. However, as was pointed out, the lift during the bank is divided into two components, one vertical and the other horizontal. This division reduces the amount of lift which is opposing gravity and actually supporting the airplane's weight; consequently, the airplane loses altitude unless additional lift is created. This is done by increasing the angle of attack until the vertical component of lift is again equal to the weight. Since the vertical component of lift decreases as the bank angle increases, the angle of attack must be progressively increased to produce sufficient vertical lift to support the airplane's weight. The fact that the vertical component of lift must be equal to the weight to maintain altitude is an important fact to remember when making constant altitude turns.

At a given airspeed, the rate at which an airplane turns depends upon the magnitude of the horizontal component of lift. It will be found that the horizontal component of lift is proportional to the angle of bank; that is, it increases or decreases respectively as the angle of bank increases or decreases. It logically follows then, that as the angle of bank is increased the horizontal component of lift increases, thereby increasing the rate of turn. Consequently, at any given airspeed the rate of turn can be controlled by adjusting the angle of bank.

To provide a vertical component of lift sufficient to hold altitude in a level turn, an increase in the angle of attack is required. Since the drag of the airfoil is directly proportional to its angle of attack, induced drag will increase as the lift is increased. This, in turn, causes a loss of airspeed in proportion to the angle of bank; a small angle of bank results in a small reduction in airspeed and a large angle of bank results in a large reduction in airspeed. Additional thrust (power) must be applied to prevent a reduction in airspeed in level turns; the required amount of additional thrust is proportional to the angle of bank.

To compensate for added lift which would result if the airspeed were increased during a turn, the angle of attack must be decreased, or the angle of bank increased, if a constant altitude were to be maintained. If the angle of bank were held constant and the angle of attack decreased, the rate of turn would decrease. Therefore, in order to maintain a constant rate of turn as the airspeed is increased, the angle of attack must remain constant and the angle of bank increased.

It must be remembered that an increase in airspeed results in an increase of the turn radius and that centrifugal force is directly proportional to the radius of the turn. In a correctly executed turn, the horizontal component of lift must be exactly equal and opposite to the centrifugal force. Therefore, as the airspeed is increased in a constant-rate level turn, the radius of the turn increases. This increase in the radius of turn causes an increase in the centrifugal force, which must be balanced by an increase in the horizontal component of lift. The horizontal component of lift can only be increased by increasing the angle of bank.

In a slipping turn the airplane is not turning at the rate appropriate to the bank being used, since the airplane is yawed toward the outside of the turning flightpath. The airplane is banked too much for the rate of turn, so the horizontal lift component is greater than the centrifugal force (Fig. 3-33). Equilibrium between the horizontal lift component and centrifugal force is reestablished either by decreasing the bank, increasing the rate of turn, or a combination of the two changes.

A skidding turn results from an excess of centrifugal force over the horizontal lift component, pulling the airplane toward the outside of the turn. The rate of turn is too great for the angle of bank. Correction of a skidding turn thus involves a reduction in the rate of turn, an increase in bank, or a combination of the two changes.

To maintain a given rate of turn, the angle of bank must be varied with the airspeed. This becomes particularly important in high-speed airplanes. For instance, at 400 miles per hour, an airplane must be banked approximately 44° to execute a standard rate turn (3° per second). At this angle of bank, only about 79 percent of the lift of the airplane comprises the vertical component of the lift; the result is a loss of

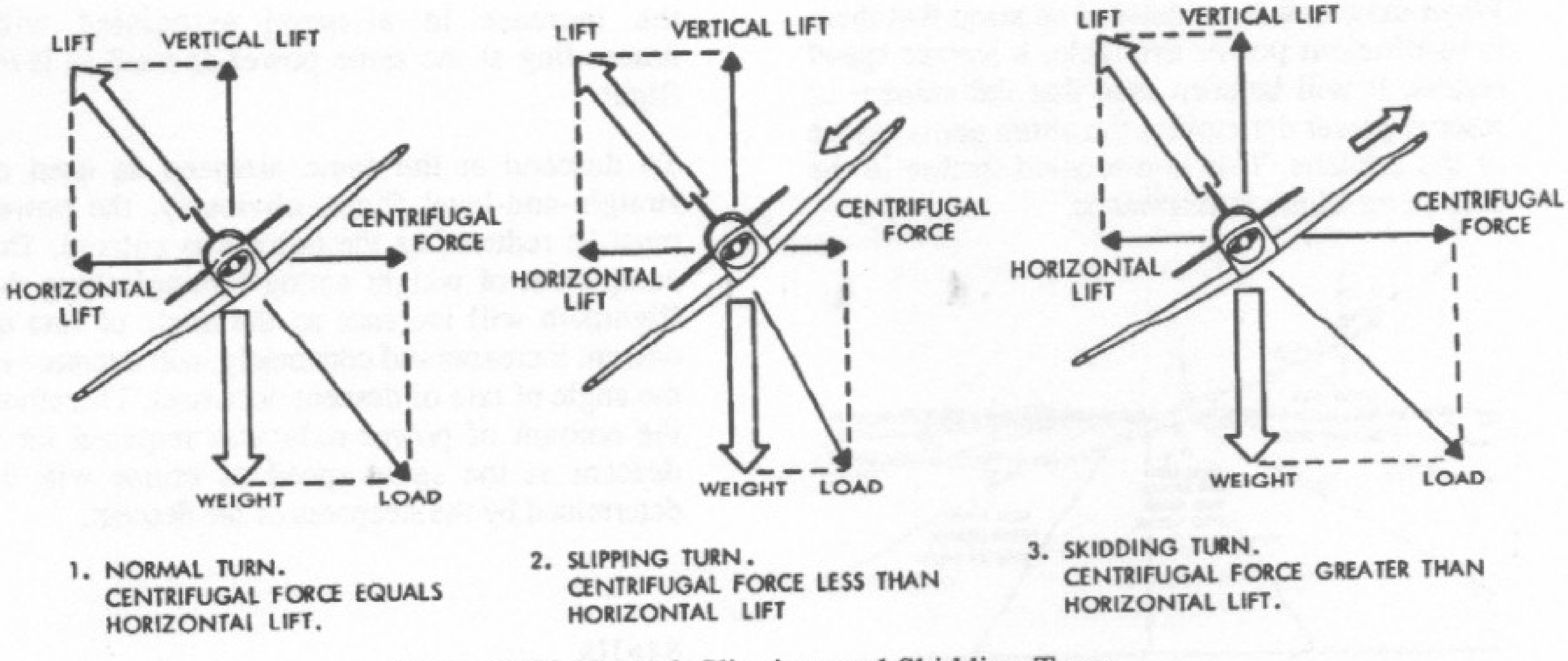


Figure 3-33 Normal, Slipping, and Skidding Turns

altitude unless the angle of attack is increased sufficiently to compensate for the loss of vertical lift.

Forces in Climbs

For all practical purposes, the wing's lift in a steady state normal climb is the same as it is in a steady level flight at the same airspeed. Though the airplane's flightpath has changed when the climb has been established, the angle of attack of the wing with respect to the inclined flightpath reverts to practically the same values, as does the lift. There is an initial momentary change, however, as shown in Fig. 3-34. During the transition from straight-and-level flight to a climb, a change in lift occurs when back elevator pressure is first applied. Raising the airplane's nose increases the angle of attack and momentarily increases the lift. Lift at this moment is now greater than weight and starts the airplane climbing. After the flightpath is stabilized on the upward incline, the angle of attack and lift again revert to about the level flight values.

If the climb is entered with no change in power setting, the airspeed gradually diminishes because the thrust required to maintain a given airspeed in level flight is insufficient to maintain the same airspeed in a climb. When the flightpath is inclined upward, a component of the airplane's weight acts in the same direction as, and parallel to, the total drag of the airplane, thereby increasing the total *effective* drag. Consequently, the total drag is greater than the power, and the airspeed decreases. The reduction in airspeed gradually results in a corresponding decrease in drag until the total drag (including the component of weight acting in the same direction) equal the thrust (Fig. 3-35). Due to momentum, the change in airspeed is gradual, varying considerably with differences in airplane size, weight, total drag, and other factors.

Generally speaking, the forces of thrust and drag, and lift and weight, again become balanced when the airspeed stabilizes but at a value lower than in straight-and-level flight at the same power setting. Since in a climb the airplane's weight is not only acting downward but rearward along with drag, additional power is required to maintain the same airspeed as in level flight. The amount of power depends on the angle of climb.

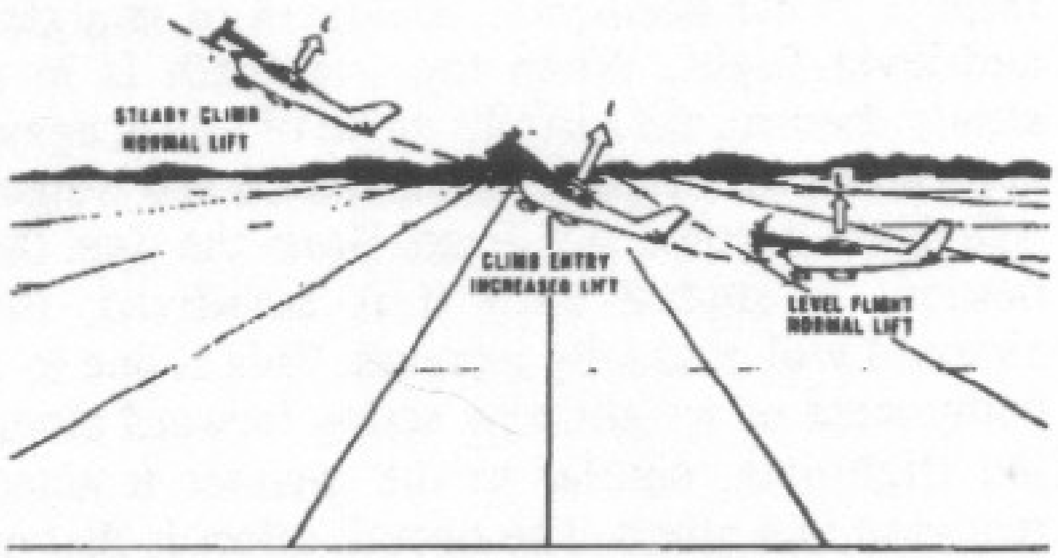


Figure 3-34 Changes In Lift During Climb Entry

When the climb is established so steep that there is insufficient power available, a slower speed results. It will be seen then that the amount of reserve power determines the climb performance of the airplane. This is discussed further in the section on Climb Performance.

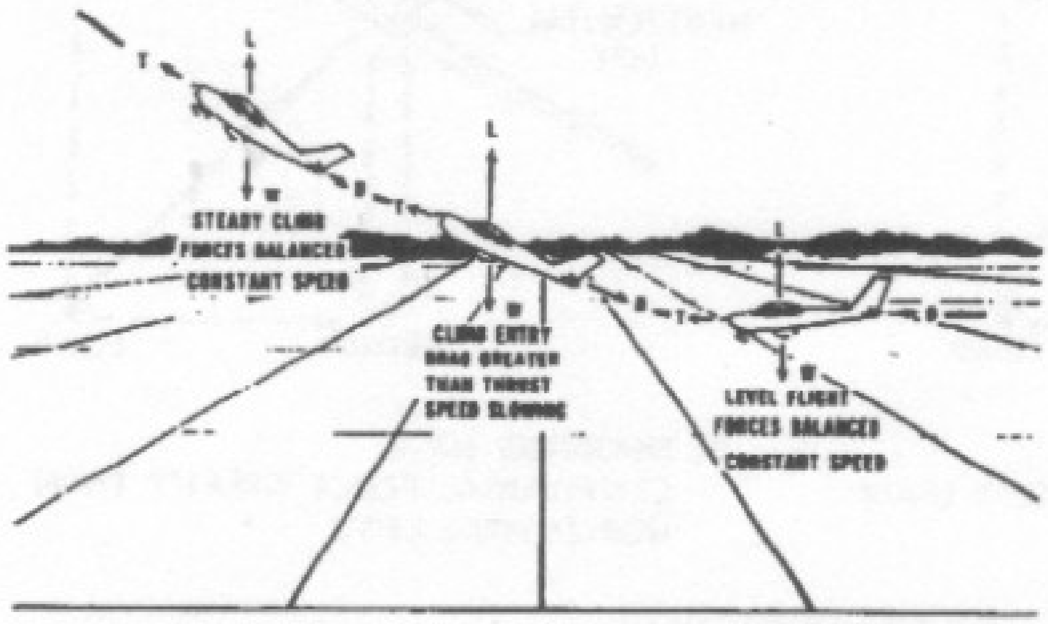


Figure 3-35 Changes In Speed During Climb Entry

Forces in Descents

As in climbs, the forces acting on the airplane go through definite changes when a descent is entered from straight-and-level flight. The analysis here is that of descending at the same power as used in straight-and-level flight.

When forward pressure is applied to the elevator control to start descending, or the airplane's nose is allowed to pitch down, the angle of attack is decreased and, as a result, the lift of the airfoil is reduced. This reduction in total lift and angle of attack is momentary and occur during the time the flightpath changes downward. The change to a downward flightpath is due to the lift momentarily becoming less than the weight of the airplane as the angle of attack is reduced. This unbalance between lift and weight causes the airplane to follow a descending flightpath with respect to the horizontal flightpath of straight-and-level flight. When the flightpath is in a steady descent, the airfoil's angle of attack again approaches the original value, and lift and weight will again become stabilized. From the time the descent is started until it is stabilized, the airspeed will gradually increase. This is due to a component of weight now acting forward along the flightpath, similar to the manner it acted rearward in a climb. The overall effect is that of increased power or thrust, which in turn causes the increase in airspeed associated with descending at the same power as used in level flight.

To descend at the same airspeed as used in straight-and-level flight, obviously, the power must be reduced as the descent is entered. The component of weight acting forward along the flightpath will increase as the angle of rate of descent increases and conversely, will decrease as the angle of rate of descent decreases. Therefore, the amount of power reduction required for a descent at the same speed as cruise will be determined by the steepness of the descent.

Stalls

In earlier discussions it was shown that an airplane will fly as long as the wing is creating sufficient lift to counteract the load imposed on it. When the lift is completely lost, the airplane stalls.

Remember, the direct cause of every stall is an excessive angle of attack. There are any number of flight maneuvers which may produce an increase in the angle of attack, but the stall does not occur until the angle of attack becomes excessive.

It must be emphasized that the *stalling speed* of a particular airplane is not a fixed value for all flight situations. However, a given airplane will always stall at the *same angle of attack* regardless of airspeed, weight, load factor, or density altitude. Each airplane has a particular angle of attack where the airflow separates from the upper surface of the wing and the stall occurs. This critical angle of attack varies from 16° to 20° depending on the airplane's design. But each airplane has only *one* specific angle of attack where the stall occurs.

There are three situations in which the critical angle of attack can be exceeded—in low speed flying, in high speed flying, and in turning flight.

The airplane can be stalled in straight-and-level flight by flying too slowly. As the airspeed is being decreased, the angle of attack must be increased to retain the lift required for maintaining altitude. The slower the airspeed becomes the more the angle of attack must be increased. Eventually an angle of attack is reached which will result in the wing not producing enough lift to support the airplane and it will start settling. If the airspeed is reduced further the airplane will stall, since the angle of attack has exceeded the critical angle and the airflow over the wing is disrupted.

It must be reemphasized here that low speed is not necessary to produce a stall. The wing can be brought into an excessive angle of attack at *any* speed. For example, take the case of an airplane which is in a dive with an airspeed of 200 knots when suddenly the pilot pulls back sharply on the elevator control (Fig. 3-36). Because of gravity and centrifugal force, the airplane could not immediately alter its flightpath but would merely change its angle of attack abruptly from quite low to very high. Since the flightpath of the airplane in relation to the oncoming air determines the direction of the relative wind, the angle of attack is suddenly increased, and the airplane would quickly reach the stalling angle at a speed much greater than the normal stall speed.

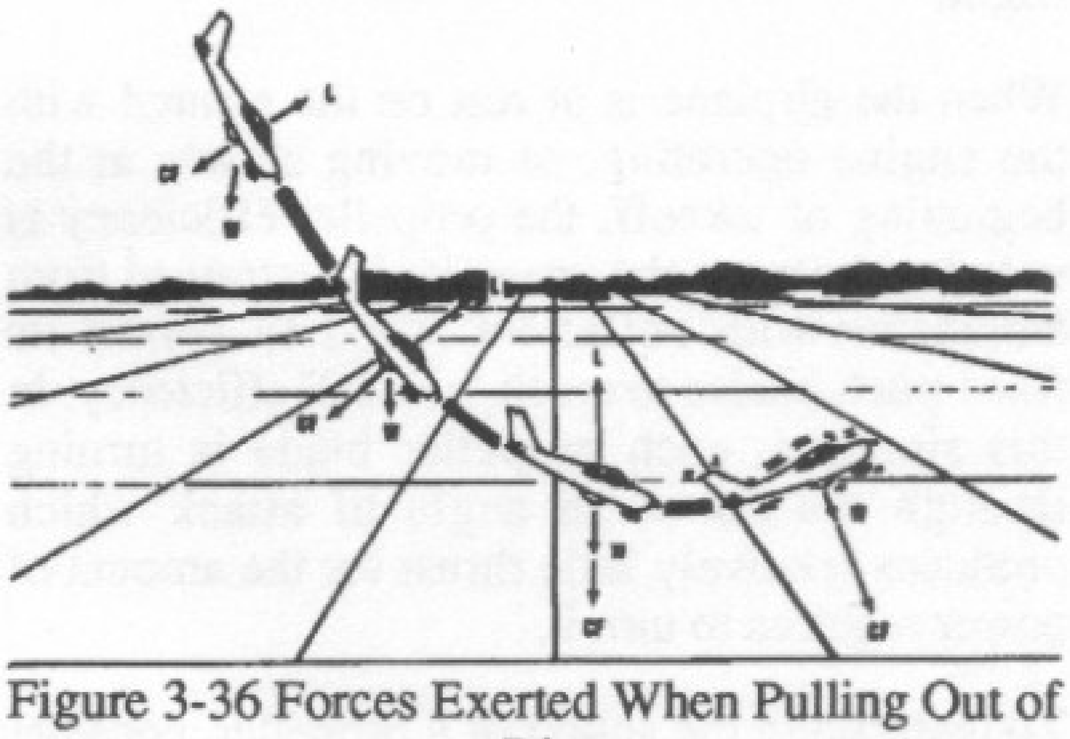


Figure 3-36 Forces Exerted When Pulling Out of a Dive

Similarly, the stalling speed of an airplane is higher in a level turn than in straight-and-level flight (Fig. 3-37). This is because centrifugal force is added to the airplane's weight, and the wing must produce sufficient additional lift to counterbalance the load imposed by the combination of centrifugal force and weight. In a turn, the necessary additional lift is acquired by applying back pressure to the elevator control. This increases the wing's angle of attack, and results in increased lift. As stated earlier, the angle of attack must increase as the bank angle increases to counteract the increasing load caused by centrifugal force. If at any time during a turn the angle of attack becomes excessive, the airplane will stall.

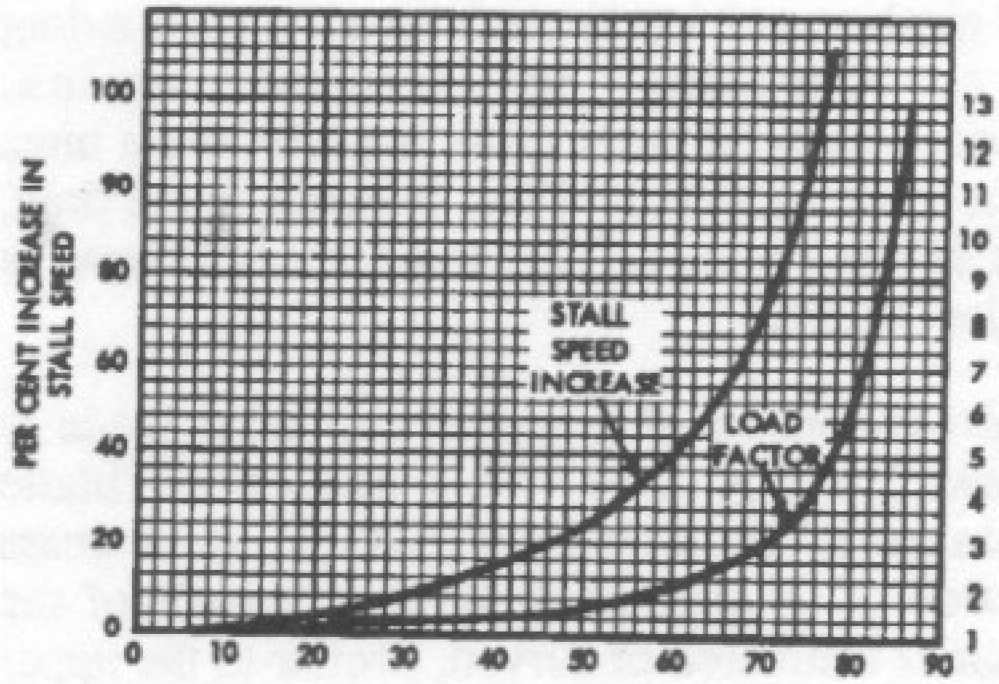


Figure 3-37 Increase in Stall Speed and Load Factor

At this point we should examine the action of the airplane during a stall. In our earlier discussion of pitching (longitudinal) stability, we learned that to balance the airplane aerodynamically, the center of lift is normally located aft of the center of gravity. It was also pointed out that although this made the inherently "nose heavy," downwash on the horizontal stabilizer counteracted this condition. It can he seen then, that at the point of stall when the upward force of the wing's lift and the downward tail force cease, an unbalanced condition exists. This allows the airplane to pitch down abruptly, rotating about its center of gravity. During this nose down attitude the angle of attack decreases and the airspeed again increases; hence, the smooth flow of air over the wing begins again, lift returns, and the airplane is again flying. However, considerable altitude may be lost before this cycle is complete.

Basic Propeller Principles

The airplane propeller consists of two or more blades and a central hub to which the blades are attached. Each blade of an airplane propeller is essentially a rotating wing. As a result of their construction, the propeller blades are like airfoils and produce forces that create the thrust to pull, or push, the airplane through the air.

The power needed to rotate the propeller blades is furnished by the engine. The engine rotates the airfoils of the blades through the air at high speeds, and the propeller transforms the rotary power of the engine into forward thrust.

An airplane moving through the air creates a drag force opposing its forward motion. Consequently, if an airplane is to fly, there must be a force applied to it that is equal to the drag, but acting forward. This force, as we know, is called "thrust."

A cross section of a typical propeller blade is shown in Fig. 3-38. This section or blade element is an airfoil comparable to a cross section of an airplane wing. One surface of the blade is cambered or curved, similar to the upper surface of an airplane wing, while the other surface is flat like the bottom surface of a wing. The chord line is an imaginary line drawn through the blade from its leading edge to its trailing edge. As in a wing, the leading edge is the thick edge of the blade that meets the air as the propeller rotates.

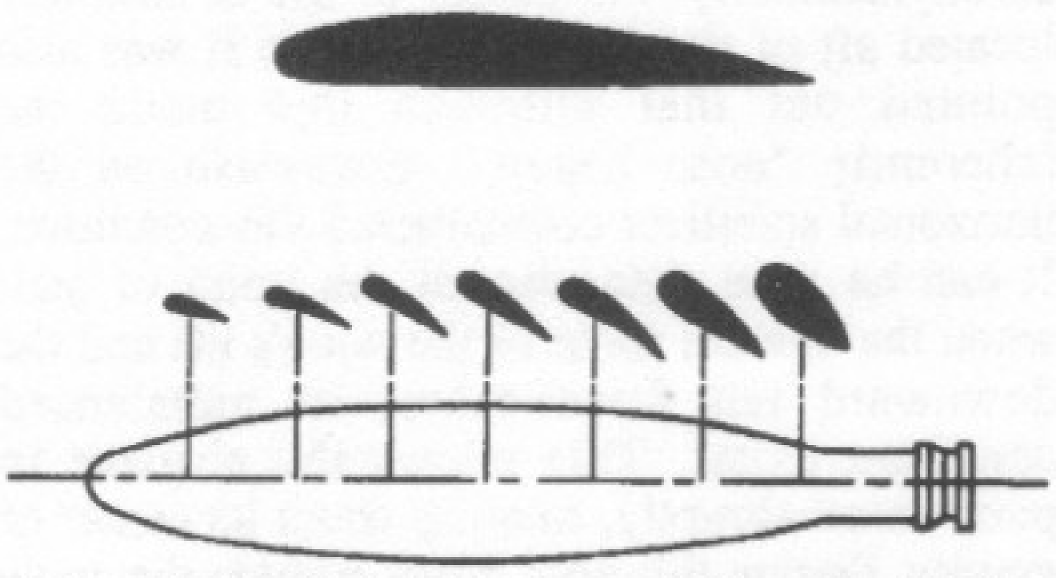


Figure 3-38 Airfoil Sections of Propeller Blade

Blade angle, usually measured in degrees, is the angle between the chord of the blade and the plane of rotation (Fig. 3-39) and is measured at a specific point along the length of the blade. Because most propellers have a flat blade "face," the chord line is often drawn along the face of the propeller blade. Pitch is not the same as blade angle, but because pitch is largely determined by blade angle, the two terms are often used interchangeably. An increase or decrease in one is usually associated with an increase or decrease in the other.

The pitch of a propeller may be designated in inches. A propeller designated as a "74-48" would be 74 inches in length and have an effective pitch of 48 inches. The pitch in inches is the distance which the propeller would screw through the air in one revolution if there were no slippage.

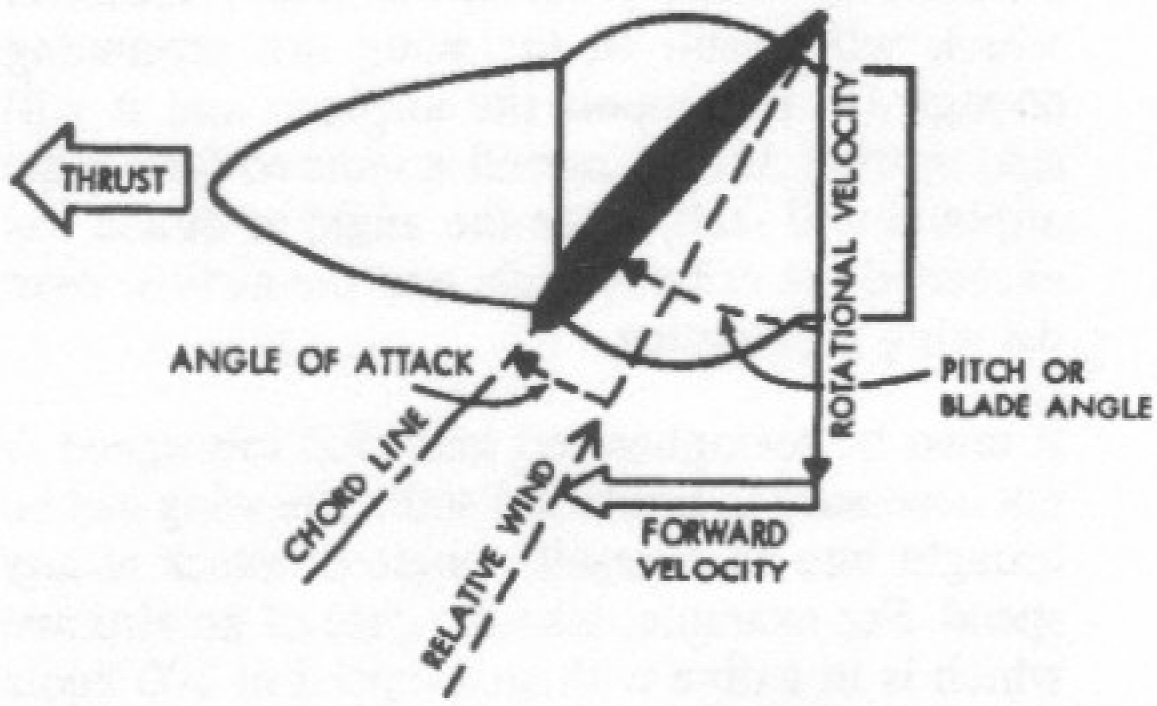


Figure 3-39 Propeller Blade Angle

When specifying a fixed-pitch propeller for a new type of airplane, the manufacturer usually selects one with a pitch likely will operate efficiently at the expected cruising speed of the airplane. Unfortunately, however, every fixed pitch propeller must be a compromise, because it can be efficient at only a given combination of airspeed and RPM. The pilot does not have it within his power to change this combination in flight.

When the airplane is at rest on the ground with the engine operating, or moving slowly at the beginning of takeoff, the propeller efficiency is very low because the propeller is restrained from advancing with sufficient speed to permit its fixed-pitch blades to reach their full efficiency. In this situation, each propeller blade is turning through the air at an angle of attack which produces relatively little thrust for the amount of power required to turn it.

To understand the action of a propeller, consider first its notion, which is both rotational and forward. Thus, as shown by the vectors of propeller forces in Fig. 3-39, each section of a propeller blade moves downward and forward. The angle at which this air (relative wind) strikes the propeller blade is its angle of attack. The air deflection produced by this angle causes the dynamic pressure at the engine side of the propeller blade to be greater than atmospheric, thus creating thrust.

The shape of the blade also creates thrust, because it is cambered like the airfoil shape of a wing. Consequently, as the air flows past the propeller, the pressure on one side is less than that on the other. As in a wing, this produces a reaction force in the direction of the lesser pressure. In the case of a wing, the air flow over the wing has less pressure, and the force (lift) is upward. In the case of the propeller, which is mounted in a vertical instead of a horizontal plane, the area of decreased pressure is in front of the propeller, and the force (thrust) is in a forward direction. Aerodynamically, then, thrust is the result of the propeller shape *and* the angle of attack of the blade.

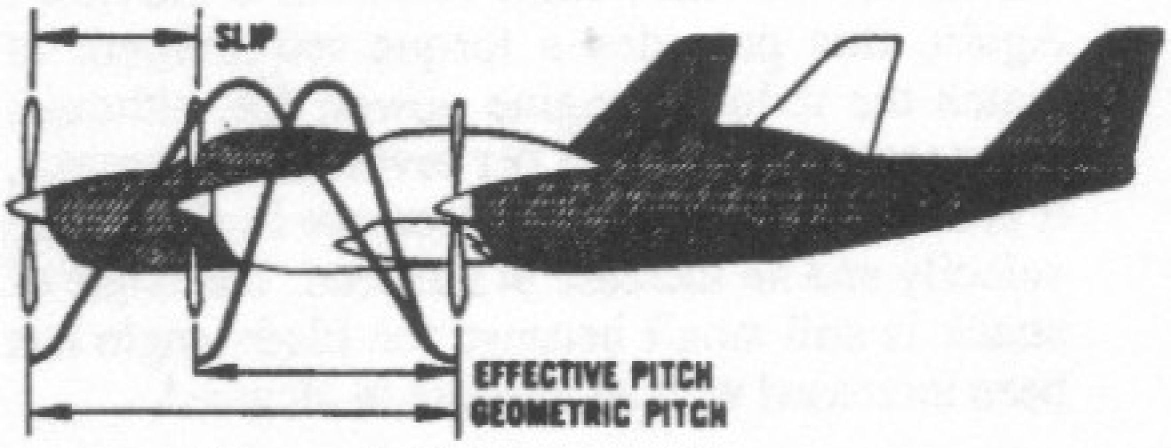


Figure 3-40 Propeller Slippage

Another way to consider thrust is in terms of the mass of air handled by the propeller. In these terms, thrust is equal to the mass of air handled times the slipstream velocity minus the velocity of the airplane. The power expended in producing thrust depends on the rate of air mass movement. On the average, thrust constitutes approximately 80% of the torque (total horsepower absorbed by the propeller). The other 20% is lost in friction and slippage. For any speed of rotation, the horsepower absorbed by the propeller balances the horsepower delivered by the engine. For any single revolution of the propeller, the amount of air handled depends on the blade angle, which determines how big a "bite" of air the propeller takes. Thus, the blade angle is an excellent means of adjusting the load on the propeller to control the engine RPM.

The blade angle is also an excellent method of adjusting the angle of attack of the propeller. On constant-speed propellers, the blade angle must be adjusted to provide the most efficient angle of attack at all engine and airplane speeds. Lift versus drag curves, which are dawn for propellers as well as wings, indicate that the most efficient angle of attack is a small one varying front 2° to 4° positive. The actual blade angle necessary to maintain this small angle of attack varies with the forward speed of the airplane.

Fixed-pitch and ground-adjustable propellers are designed for best efficiency at one rotation and forward speed. They are designed for a given airplane and engine combination. A propeller may be used that provides the maximum propeller efficiency for either takeoff, climb, cruise, or high speed flight. Any change in these conditions results in lowering the efficiency of both the propeller and the engine. Since the efficiency of any machine is the ratio of the useful power output to the actual power input, propeller efficiency is the ratio of thrust horsepower to brake horsepower. Propeller efficiency varies from 50% to 87%, depending on how much the propeller "slips."

Propeller slip is the difference between the geometric pitch of the propeller and its effective pitch (Fig. 3-40). Geometric pitch is the theoretical distance a propeller should advance in one revolution; effective pitch is the distance it actually advances. Thus, geometric or theoretical pitch is based on no slippage, but actual or effective pitch includes propeller slippage in the air.

If you wonder why a propeller is "twisted," the answer is that the outer parts of the propeller blades, like all things that turn about a central point, travel faster than the portions near the hub (Fig. 3-41). If the blades had the same geometric pitch throughout their lengths, at cruise speed the portions near the hub could have negative angles of attack while the propeller tips would be stalled. "Twisting," or variations in the geometric pitch of the blades, permits the propeller to operate with a relatively constant angle of attack along its length when in cruising flight. To put it another way, propeller blades are twisted to change the blade angle in proportion to the differences in speed of rotation along the length of the propeller and thereby keep thrust more nearly equalized along this length.

Usually 1° to 4° provides the most efficient lift/drag ratio, but in flight the propeller angle of attack of a fixed-pitch propeller will vary—normally from 0° to 15°. This variation is caused by changes in the relative airstream which in turn results from changes in aircraft speed. In short, propeller angle of attack is the product of two motions—propeller rotation about its axis and its forward motion.

A constant-speed propeller, however, automatically keeps the blade angle adjusted for maximum efficiency for most conditions encountered in flight. During takeoff, when maximum power and thrust are required, the constant-speed propeller is at a low propeller blade angle or pitch. The low blade angle keeps the angle of attack small and efficient with respect to the relative wind. At the same time, it allows the propeller to handle a smaller mass of air per revolution. This light load allows the engine to turn at high RPM and to convert the maximum amount of fuel into heat energy in a given time. The high RPM also creates maximum thrust; for, although the mass of air handled per revolution is small, the number of revolutions per minute is many, the slipstream velocity is high, and with the low airplane speed, the thrust is maximum.

After liftoff, as the speed of the airplane increases, the constant-speed propeller automatically chances to a higher angle (or pitch). Again, the higher blade angle keeps the angle of attack small and efficient with respect to the relative wind. The higher blade angle increases the mass of air handled per revolution. This decreases the engine RPM, reducing fuel consumption and engine wear, and keeps thrust at a maximum.

Figure 3-41 Propeller Tips Travel Faster Than Hubs

After the takeoff climb is established, in an airplane having a controllable pitch propeller, the pilot reduces the power output of the engine to climb power by first decreasing the manifold pressure and then increasing the blade angle to lower the RPM.

At cruising altitude, when the airplane is in level flight and less power is required than is used in takeoff or climb, the pilot again reduces engine power by reducing the manifold pressure and then increasing the blade angle to decrease the RPM. Again, this provides a torque requirement to match the reduced engine power; for, although the mass of air handled per revolution is greater, it is more than offset by a decrease in slipstream velocity and an increase in airspeed. The angle of attack is still small because the blade angle has been increased with an increase in airspeed.

Torque and P Factor

To the pilot, "torque" (the left turning tendency of the airplane) is made up of four elements which cause or produce a twisting or rotating motion around at least one of the airplane's three axes. These four elements are:

1. Torque Reaction from Engine and Propeller.

2. Corkscrewing Effect of the Slipstream.

3. Gyroscopic Action of the Propeller.

4. Asymmetric Loading of the Propeller (P Factor).

Torque Reaction

Torque reaction involves Newton's Third Law of Physics—for every action, there is an equal and opposite reaction. As applied to the airplane, this means that as the internal engine parts and propeller are revolving in one direction, an equal force is trying to rotate the airplane in the opposite direction (Fig. 3-42).

Figure 3-42 Torque Reaction

When the airplane is airborne, this force is acting around the longitudinal axis, tending to make the airplane roll. To compensate for this, some of the older airplanes are rigged a manner to create more lift on the wing which is being forced downward The more modern airplanes are designed with the engine offset to counteract this effect of torque.

> Note—Most United States built aircraft engines rotate the propeller clockwise, as viewed from the pilot's seat. The discussion here is with reference to those engines.

Generally, the compensating factors are permanently set so that they compensate for this force at cruising speed, since most of the airplane's operating life is at that speed. However, aileron trim tabs permit further adjustment for other speeds.

When the airplane wheels are on the ground during the takeoff roll, an additional turning moment around the vertical axis is induced by torque reaction. As the left side of the airplane is being forced down by torque reaction, more weight is then placed on the left main landing gear. This results in more ground friction, or drag on the left tire than on the right causing a further turning moment to the left. The magnitude of this moment is dependent on many variables. Some of these variables are: (1) size and horsepower of engine, (2) size of propeller and the RPM, (3) size of the airplane, and (4) condition of the ground surface.

This yawing moment on the takeoff roll is corrected by the pilot's proper use of the rudder or rudder trim.

Corkscrew Effect

The high speed rotation of an airplane propeller gives a corkscrew or spiraling rotation to the slipstream. At high propeller speeds and low forward speed (as in the takeoffs. approaches to power-on stalls, etc..), this spiraling rotation is very compact and exerts a strong sideward force on the airplane's vertical tail surface (Fig. 3-43).

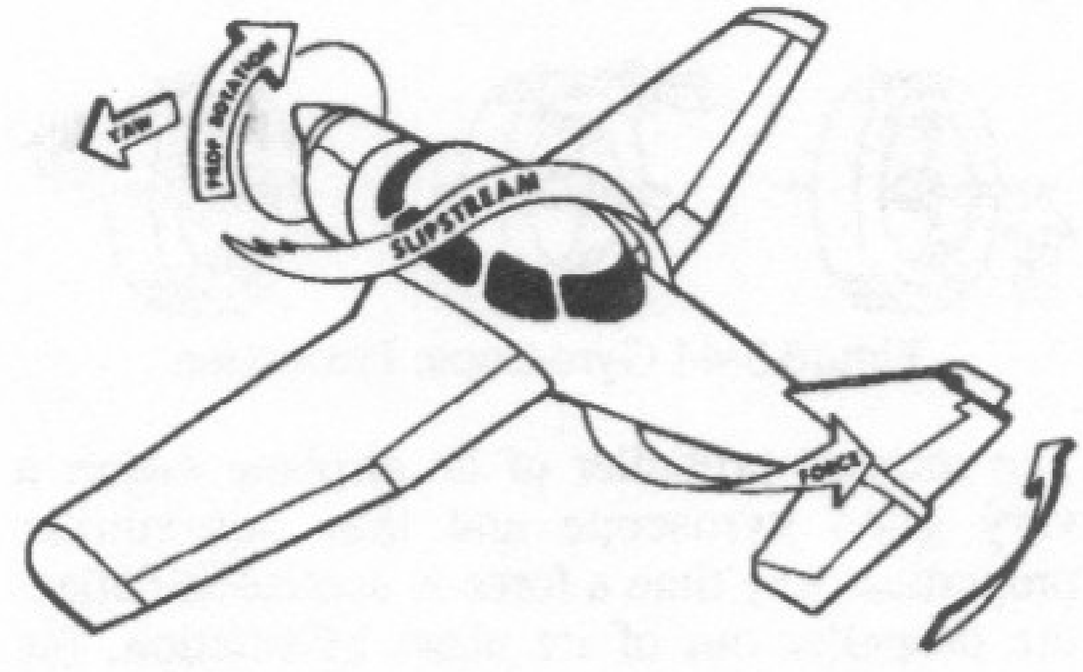


Figure 3-43 Corkscrewing Slipstream

When this spiraling slipstream strikes the vertical fin on the left, it causes a left turning moment about the airplane's vertical axis. The more compact the spiral, the more prominent this force is. As the forward speed increases, however, the spiral elongates and becomes less effective.

The corkscrew flow of the slipstream also causes a rolling moment around the longitudinal axis.

Note that this rolling moment caused by the corkscrew flow of the slipstream is to the right, while the rolling moment caused by torque reaction is to the left—in effect one may be counteracting the other. However, these forces vary greatly and it is up to the pilot to apply proper correction action by use of the flight controls at all times. These forces must be counteracted regardless of which is the most prominent at the time.

Gyroscopic Action

Before the gyroscopic effects of the propeller can be understood, it is necessary to understand the basic principles of a gyroscope.

All practical applications of the gyroscope are based upon two fundamental properties of gyroscopic action—rigidity in space, and precession. The one in which we are interested for this discussion is precession.

Precession is the resultant action, or deflection, of a spinning rotor when a deflecting force is applied to its rim. As can be seen in Fig. 3-44, when a force is applied, the resulting force takes effect 90° ahead of and in the direction of rotation.

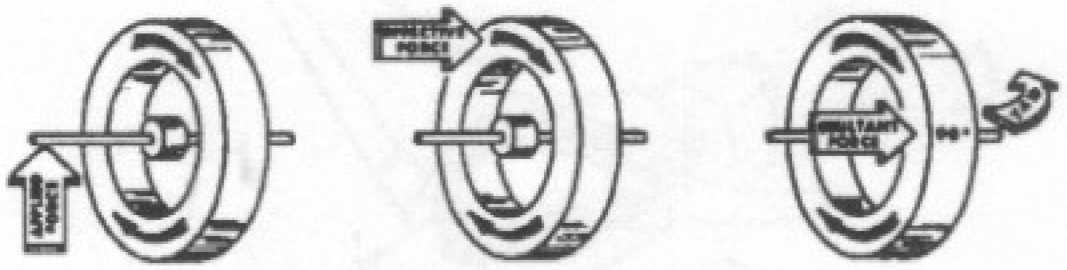


Figure 3-44 Gyroscopic Precession

The rotating propeller of an airplane makes a very good gyroscope and thus has similar properties. Any time a force is applied to deflect the propeller out of its plane of rotation, the resulting force is 90° ahead of and in the direction of rotation and in the direction of application, causing a pitching moment, a yawing moment, or a combination of the two depending upon the point at which the force was applied.

This element of torque effect has always been associated with and considered more prominent in tailwheel-type airplanes, and most often occurs when the tail is being raised during the takeoff roll (Fig. 3-45) . This change in pitch attitude has the same effect as applying a force to the top of the propeller's plane of rotation. The resultant force acting 90° ahead causes a yawing moment to the left around the vertical axis. The magnitude of this moment depends on several variables, one of which is the abruptness with which the tail is raised (amount of force applied). However, precession, or gyroscopic action, occurs when a force is applied to any point on the rim of the propeller's plane of rotation; the resultant force will still be 90° from the point of application in the direction of rotation. Depending on where the force is applied, the airplane is caused to yaw left or right, to pitch up or down, or a combination of pitching and yawing.

It can be said that as a result of gyroscopic action—any yawing around the vertical axis results in a pitching moment, and any pitching around the lateral axis results in a yawing moment.

To correct for the effect of gyroscopic action, it is necessary for the pilot to properly use elevator

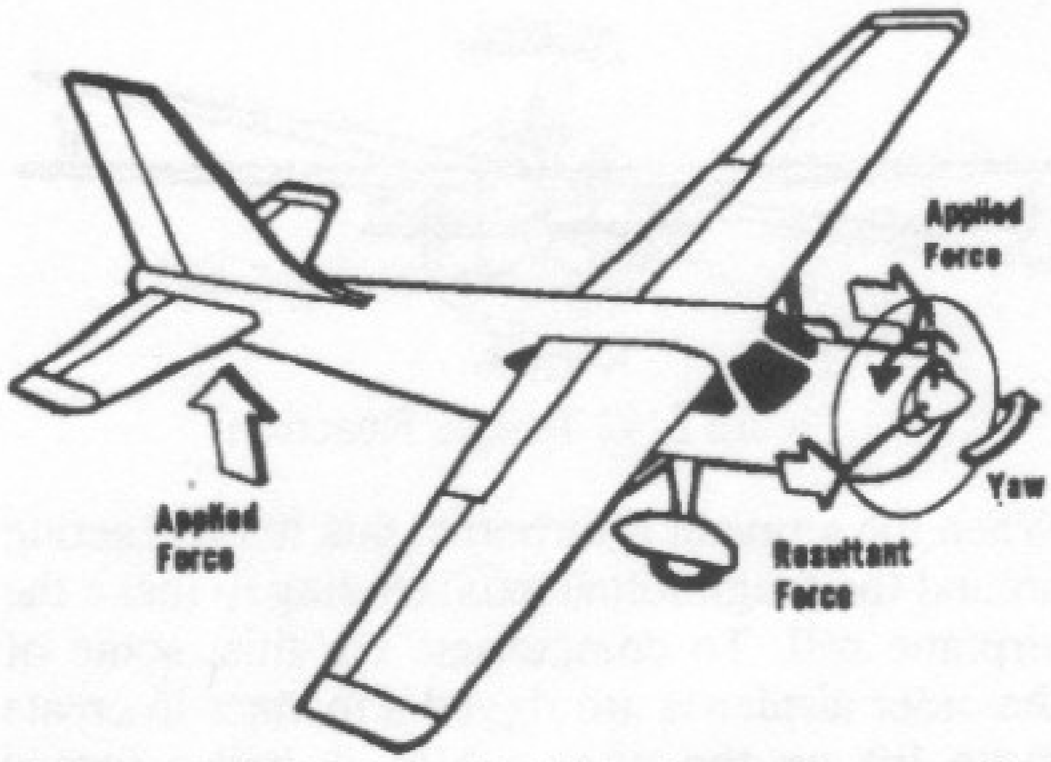


Figure 3-45 Raising Tail Produces Gyroscopic Precession

and rudder to prevent undesired pitching and yawing.

Asymmetric Loading (P Factor)

As in the past, it has been explained that when an airplane is flying with a high angle of attack, the "bite" of the downward moving blade is greater than the "bite" of the upward moving blade; thus moving the center of thrust to the right of the prop disc area—causing a yawing moment toward the left around the vertical axis. That explanation is correct; however, to prove this phenomenon, it would be necessary to work wind vector problems on each blade, which gets quite involved when considering both the angle of attack of the airplane and the angle of attack of each blade.

This asymmetric loading is caused by the resultant velocity which is generated by the combination of the velocity of the propeller blade in its plane of rotation and the velocity of the air passing horizontally through the propeller "disc." With the airplane being flown at positive angles of attack, the right (viewed from the rear) or downswinging blade, is passing through an area of resultant velocity which is greater than that affecting the left or upswinging blade. Since the propeller blade is an airfoil, increased velocity means increased lift. Therefore, the downswinging blade having more "lift" tends to pull (yaw) the airplane's nose to the left.

Simply stated, when the airplane is flying at a high angle of attack, the downward moving blade has a higher resultant velocity; therefore creating more lift than the upward moving blade (Fig. 3-46). This might be easier to visualize if we were to mount the propeller shaft perpendicular to the ground (like a helicopter). If there was no air movement at all, except that generated by the propeller itself, identical sections of each blade would have the same airspeed. But now, let's start air moving horizontally across this vertically mounted propeller. Now the blade proceeding forward into the flow of air will have a higher airspeed than the blade retreating with the airflow. Thus the blade proceeding into the horizontal airflow is creating more lift, or thrust, moving the center of thrust toward that blade. Now, let's visualize ROTATING the vertically mounted propeller shaft to shallower angles relative to the moving air (as on an airplane). This unbalanced thrust then will become proportionately smaller and continues getting smaller until it reaches the value of zero when the propeller shaft is exactly horizontal in relation to the moving air.

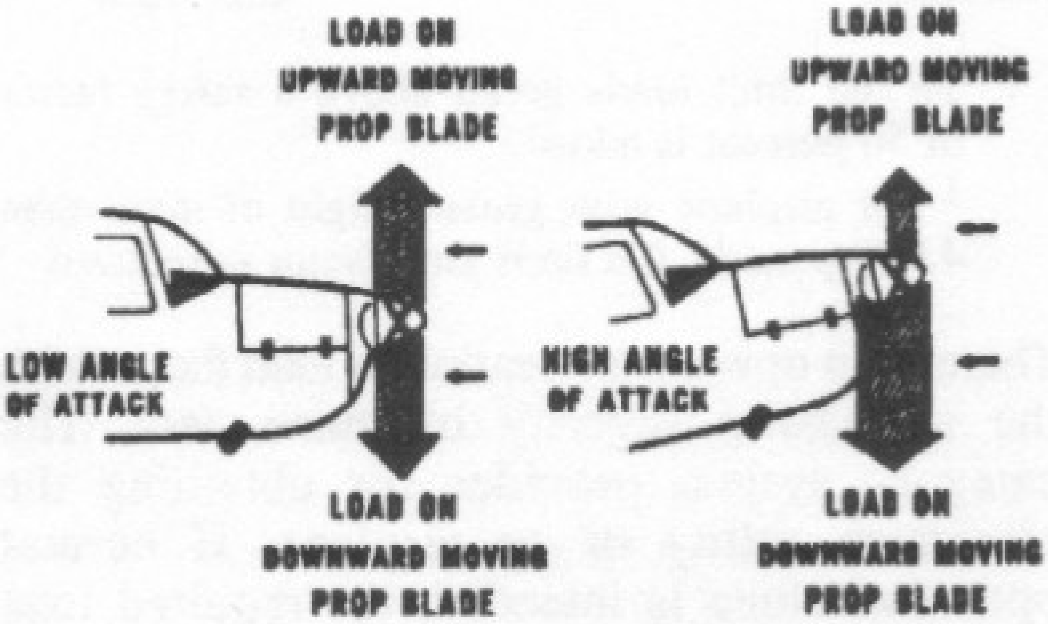


Figure 3-46 Asymmetrical Loading of Propeller (P-Factor)

Each of these four elements of torque effects vary in values with changes in flight situations. In one phase of flight, one of these elements may be more prominent than another; whereas in another phase of flight, another element may be more prominent. The relationship of these values to each other will vary with different airplanes—depending on the AIRFRAME, ENGINE, AND PROPELLER combinations as well as other design features.

To maintain positive control of the airplane in all flight conditions, the pilot must apply the flight controls as necessary to compensate for these varying values.

Load Factors

The preceding sections only briefly considered some of the practical points of the principles of flight. As we know to become a pilot, a detailed technical course in the science of aerodynamics is not necessary. However, with responsibilities for the safety of passengers, the *competent* pilot must have a well-founded concept of the forces which act on the airplane, and the advantageous use of these forces, as well as the operating limitations of the particular airplane. Any force applied to an airplane to deflect its flight from a straight line produces a stress on its structure; the amount of this force is termed "load factor."

A load factor is the ratio of the total airload acting on the airplane to the gross weight of the airplane. For example, a load factor of 3 means that the total load on an airplane's structure is three times its gross weight. Load factors are usually expressed in terms of "G"; that is, a load factor of 3 may be spoken of as 3 G's, a load factor of 4 as 4 G's, etc..

It is interesting to note that in subjecting an airplane to 3 G's in a pullup from a dive, one will be pressed down into the seat with a force equal to three times the person's weight. Thus, an idea of the magnitude of the load factor obtained in any maneuver can be determined by considering the degree to which one is pressed down into the seat. Since the operating speed of modern airplanes has increased significantly, this effect has become so pronounced that it is a primary consideration in the design of the structure for all airplanes.

With the structural design of airplanes planned to withstand only a certain amount of overload, a knowledge of load factors has become essential for all pilots. Load factors are important to the pilot for two distinct reasons:

1. Because of the obviously dangerous overload that is possible for a pilot to impose on the aircraft structures; and

2. Because an increased load factor increases the stalling speed and makes stalls possible at seemingly safe flight speeds.

Load Factors in Airplane Design

The answer to the question "how strong should an airplane be" is determined largely by the use to which the airplane will be subjected. This is a difficult problem, because the maximum possible loads are much too high for use in efficient design. It is true that any pilot can make a very hard landing or an extremely sharp pullup from a dive which would result in abnormal loads. However, such extremely *abnormal* loads must be dismissed somewhat if we are to build airplanes that will take off quickly, land slowly, and carry a worthwhile payload.

The problem of load factors in airplane design then reduces to that of determining the highest load factors which can be expected in normal operation under various operational situations. These load factors are called "limit load factors." For reasons of safety, it is required that the airplane be designed to withstand these load factors without any structural damage. Although Federal Aviation Regulations require that the airplane structure be capable of supporting one and one-half times these limit load factors without failure, it is accepted that parts of the airplane may bend or twist under these loads and that some structural damage may occur.

This 1.5 value is called the "factor of safety" and provides, to some extent, for loads higher than those expected under normal and reasonable operation. However, this strength reserve is not something which pilots should willfully abuse; rather it is there for their protection when they encounter unexpected conditions.

The above considerations apply to all loading conditions, whether they be due to gusts, maneuvers, or landings. The gust load factor requirements now in effect are substantially the same as those which have been in existence for years. Hundreds of thousands of operational hours have proven them adequate for safety. Since the pilot has little control over gust load factors (except to reduce the airplane's speed when rough air is encountered), the gust loading requirements are substantially the same for most general aviation type airplanes regardless of their operational use. Generally speaking, the gust load factors control the design of airplanes which are intended for strictly nonacrobatic usage.

An entirely different situation exists in airplane design with maneuvering load factors. It is necessary to discuss this matter separately with respect to: (1) Airplanes which are designed in accordance with the Category System (i.e., Normal, Utility, Acrobatic); and (2) Airplanes of older design which were built to requirements which did not provide for operational categories.

Airplanes designed under the category system are readily identified by a placard in the cockpit which states the operational category (or categories) in which the airplane is certificated. The maximum safe load factors (limit load factors) specified for airplanes in the various categories are as follows:

Category	*Limit Load*
Normal[1]	3.8—1.52
Utility (mild acrobatics, including spins)	4.4—1.76
Acrobatic	6.0—3.0

To the limit loads given above a safety factor of 50 percent is added.

[1] For airplane with gross weight of more than 4,000 pounds, the limit load factor is reduced.

There is an upward graduation in load factor with the increasing severity of maneuvers. The category system provides for obtaining the maximum utility of an airplane. If normal operation alone is intended, the required load factor (and consequently the weight of the airplane) is less than if the airplane is to be employed in training or acrobatic maneuvers as they result in higher maneuvering loads.

Airplanes which do not have the category placard are designs which were constructed under earlier engineering requirements in which no operational restrictions were specifically given to the pilots. For airplanes of this type (up to weights of about 4,000 pounds) the required strength is comparable to present-day utility category airplanes, and the same types of operation are permissible. For airplanes of this type over 4,000 pounds, the load factors decease with weight so that these airplanes should be regarded as being comparable to the normal category airplanes designed under the Category System, and they should be operated accordingly.

Load Factors in Steep Turns

In a constant altitude, coordinated turn in any airplane, the load factor is the result of two forces—centrifugal force and gravity (Fig. 3-47). For any given bank angle, the rate of turn varies with the airspeed; the higher the speed, the slower the rate of turn. This compensates for added centrifugal force, allowing the load factor to remain the same.

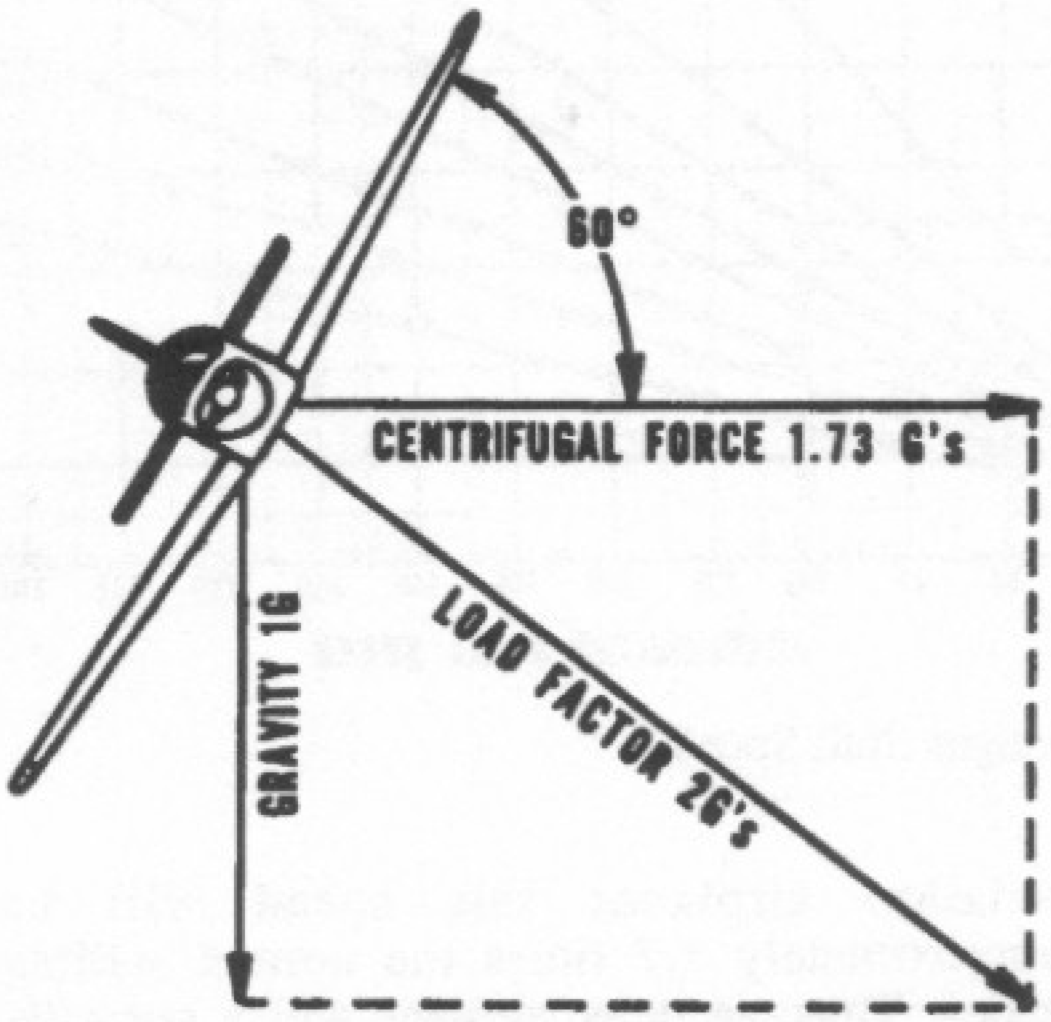


Figure 3-47 Two Causes Cause Load Factor During Turns

Fig. 3-48 reveals an important fact about turns—that the load factor increases at a terrific rate after a bank has reached 45° or 50°. The load factor for any airplane in a 60° bank is 2 G's. The load factor in an 80° bank is 5.76 G's. The wing must produce lift equal to these load factors if altitude is to be maintained.

It should be noted how rapidly the line denoting load factor rises as it approaches the 90° bank line, which it reaches only at infinity. The 90° banked, constant altitude turn mathematically is not possible. True, an airplane may be banked to 90° but not in a coordinated turn; an airplane which can be held in a 90° banked slipping turn is capable of straight knife-edged flight. At slightly more than 80° the load factor exceeds the limit of 6 G's, the limit load factor of an *acrobatic airplane*.

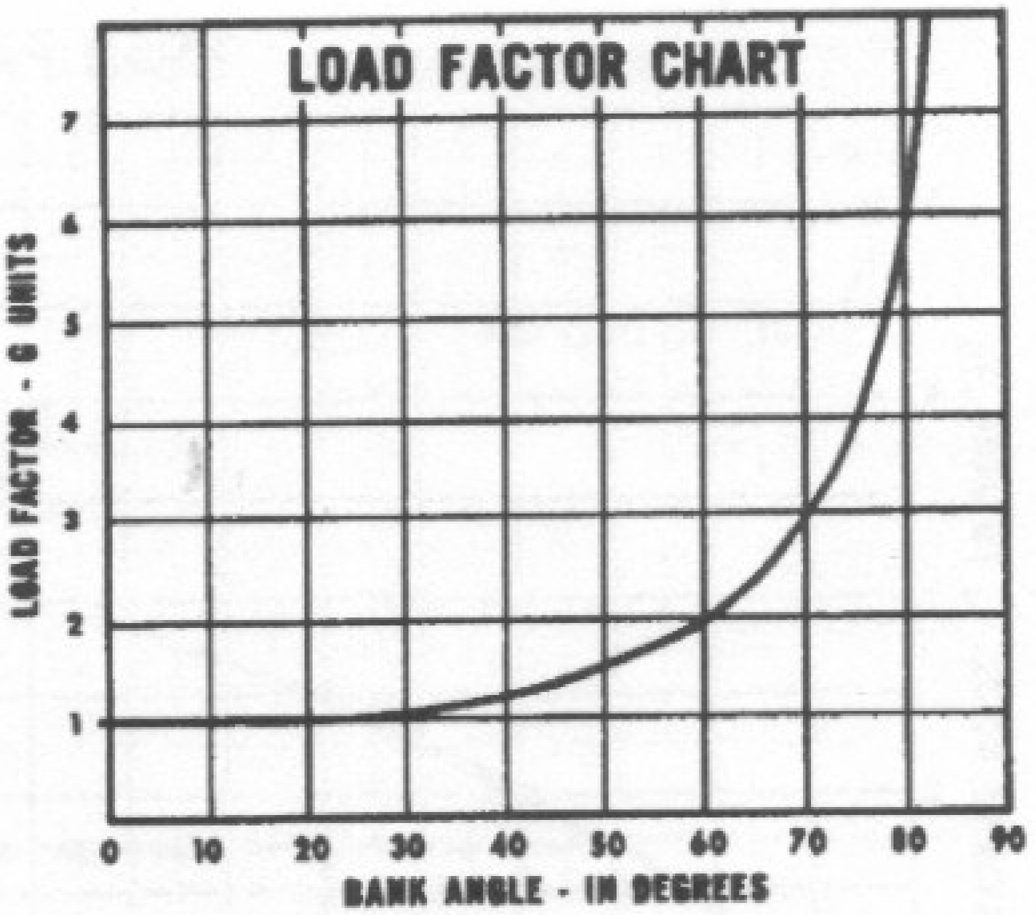


Figure 3-48 Angle of Bank Changes Load Factor

For a coordinated, constant altitude turn, the approximate maximum bank for the average general aviation airplane is 60°. This bank and its resultant necessary power setting reach the limit of this type of airplane. An additional 10° bank will increase the load factor by approximately 1G (Fig. 3-48), bringing it close to the yield point established for these airplanes.

Load Factors and Stalling Speeds

Any airplane, within the limits of its structure, may be stalled at any airspeed. When a sufficiently high angle of attack is imposed, the smooth flow of air over an airfoil breaks up and separates, producing an abrupt change of flight characteristics and a sudden loss of lift which results in a stall.

A study of this effect has revealed that the airplane's stalling speed increases in proportion to the square root of the load factor. This means that an airplane with a normal unaccelerated stalling speed of 50 knots can be stalled at 100 knots by inducing a load factor of 4 G's. If it were possible for this airplane to withstand a load factor of 9, it could be stalled at a speed of 150 knots. Therefore, a competent pilot should be aware of the following:

1. The danger of inadvertently stalling the airplane by increasing the load factor, as in a steep turn or spiral; and

2. That in intentionally stalling an airplane

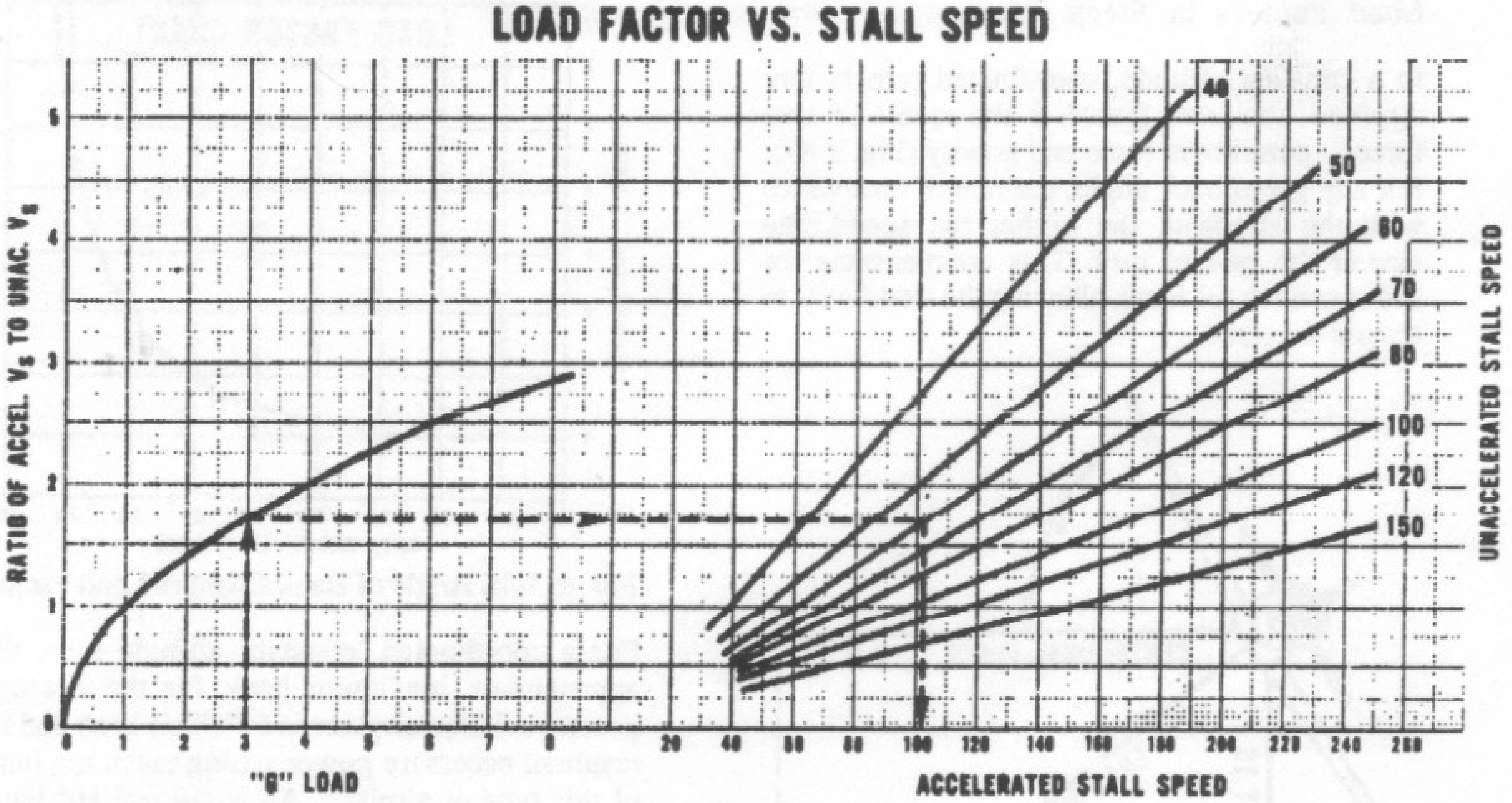


Figure 3-49 Load Factor Changes Stall Speed

above its design maneuvering speed, a tremendous load factor is imposed.

Reference to the charts in Figs. 3-48 and 3-49 will show that by banking the airplane to just beyond 72° in a steep turn produces a load factor of 3, and the stalling speed is increased significantly. If this turn is made in an airplane with a normal unaccelerated stalling speed of 45 knots, the airspeed must be kept above 90 knots to prevent inducing a stall. A similar effect is experienced in a quick pullup, or any maneuver producing load factors above 1 G. This has been the cause of accidents resulting from a sudden, unexpected loss of control, particularly in a steep turn or abrupt application of the back elevator control near the ground.

Since the load factor squares as the stalling speed doubles, it may be realized that tremendous loads may be imposed on structures by stalling an airplane at relatively high airspeeds.

The maximum speed at which an airplane may be stalled safely is now determined for all new designs. This speed is called the "design maneuvering speed," (VA) and is required to be entered in the FAA approved flight manual of all recently designed airplanes. For older general aviation airplanes this speed will be approximately 1.7 times the normal stalling speed. Thus, an older airplane which normally stalls at 60 knots must never be stalled at above 102 knots. (60 kts. X 1.7=102 kts.). An airplane with a normal stalling speed of 60 knots will undergo, when stalled at 102 knots, a load factor equal to the square of the increase in speed or 2.89 G's (1.7X1.7=2.89 G's). (These figures are an approximation to be considered as a guide and are not the exact answers to any set of problems. The design maneuvering speed should be determined from the particular airplane's operating limitations when provided by the manufacturer.)

Since the leverage in the control system varies with different airplanes and some types employ "balanced" control surfaces while others do not, the pressure exerted by the pilot on the controls cannot be accepted as an index of the load factors produced in different airplanes. In most cases, load factors can be judged by the experienced pilot from the feel of seat pressure. They can also be measured by an instrument called an "accelerometer," but since this instrument is not common in general aviation training airplanes, the development of the ability to judge load factors from the feel of their effect on the body is important. A knowledge of the principles outlined above is essential to the development of this ability to estimate load factors.

A thorough knowledge of load factors induced by varying degrees of bank, and the significance of design maneuvering speed (VA) will aid in the prevention of two of the most serious types of accidents:

1. Stalls from steep turns or excessive maneuvering near the ground; and

2. Structural failures during acrobatics or other violent maneuvers resulting from loss of control.

Load Factors and Flight Maneuvers

Critical load factors apply to all flight maneuvers except unaccelerated straight flight where a load factor of 1 G is always present. Certain maneuvers considered in this section are known to involve relatively high load factors.

Turns. Increased load factors are a characteristic of all banked turns. As noted in the section Load Factors in Steep Turns and particularly Figs. 3-48 and 3-49, load factors become significant both to flight performance and to the load on wing structure as the bank increases beyond approximately 45°.

The yield factor of the average light plane is reached at a bank of approximately 70°-75°, and the stalling speed is increased by approximately one-half at a bank of approximately 63°.

Stalls. The normal stall entered from straight level flight, or an unaccelerated straight climb, will not produce added load factors beyond the 1 G of straight-and-level flight. As the stall occurs, however, this load factor may be reduced toward zero, the factor at which nothing seems to have weight; and the pilot has the feeling of "floating free in space." In the event recovery is effected by snapping the elevator control forward, negative load factors, those which impose a down load on the wings and raise the pilot from the seat, may be produced.

During the pullup following stall recovery, significant load factors sometimes are induced. Inadvertently these may be further increased during excessive diving (and consequently high airspeed) and abrupt pullups to level flight. One usually leads to the other, thus increasing the load factor. Abrupt pullups at high diving speeds may impose critical loads on airplane structures and may produce recurrent or secondary stalls by increasing the angle of attack to that of stalling.

As a generalization, a recovery from a stall made by diving only to cruising or design maneuvering airspeed, with a gradual pullups as soon as the airspeed is safely above stalling, can be effected with a load factor not to exceed 2 or 2.5 G's. A higher load factor should never be necessary unless recovery has been effected with the airplane's nose near or beyond the vertical attitude, or at extremely low altitudes to avoid diving into the ground.

Spins. Since a stabilized spin is not essentially different from a stall in any element other than rotation, the same load factor considerations apply as those which apply to stall recovery. Since spin recoveries usually are effected with the nose much lower than is common in stall recoveries, higher airspeeds and consequently higher load factors are to be expected. The load factor in a proper spin recovery will usually be found to be about 2.5 G's.

The load factor during a spin will vary with the spin characteristics of each airplane but is usually found to be slightly above the 1 G of level flight. There are two reasons this is true:

1. The airspeed in a spin is very low; usually within 2 knots of the unaccelerated stalling speeds; and

2. The airplane pivots, rather than turns, while it is in a spin.

High-Speed Stalls. The average light plane is not built to withstand the repeated application of load factors common to high-speed stalls. The load factor necessary for these maneuvers produces a stress on the wings and tail structure, which does not leave a reasonable margin of safety in most light airplanes.

The only way this stall can be induced at an airspeed above normal stalling involves the imposition of an added load factor, which may be accomplished by a severe pull on the elevator control. A speed of 1.7 times stalling speed (about 102 knots in a light airplane with a stalling speed of 60 knots) will produce a load factor of 3 G's. Further, only a very narrow margin for error can be allowed for acrobatics in light airplanes. To illustrate how rapidly the load factor increases with airspeed, a high-speed stall at 112 knots in the same airplane would produce a load factor of 4 G's.

Chandelles and Lazy Eights. It would be difficult to make a definite statement concerning load factors in these maneuvers as both involve smooth, shallow dives and pullups. The load factors incurred depend directly on the speed of the dives and the abruptness of the pullups.

Generally, the better the maneuver is performed, the less extreme will be the load factor induced. A chandelle or lazy eight, in which the pullup produces a load factor greater than 2 G's will not result in as great a gain in altitude, and in low-powered airplanes it may result in a net loss of altitude.

The smoothest pullup possible, with a moderate load factor, will deliver the greatest gain in altitude in a chandelle and will result in a better overall performance in both chandelles and lazy eights. Further, it will be noted that recommended entry speed for these maneuvers is generally near the manufacturer's design maneuvering speed, thereby allowing maximum development of load factors without exceeding the load limits.

Rough Air. All certificated airplanes are designed to withstand loads imposed by gusts of considerable intensity. Gust load factors increase with increasing airspeed and the strength used for design purposes usually corresponds to the highest level flight speed. In extremely rough air, as in thunderstorms or frontal conditions, it is wise to reduce the speed to the design maneuvering speed. Regardless of the speed held, there may be gusts that can produce loads which exceed the load limits.

Most airplane flight manuals now include turbulent air penetration information. Operators of modern airplanes, capable of a wide range of speeds and altitudes, are benefited by this added feature both in comfort and safety. In this connection it is to be noted that the maximum "never exceed" placard dive speeds are determined for smooth air only. High-speed dives or acrobatics involving speed above the known maneuvering speed should never be practiced in rough or turbulent air.

In summary, it must be remembered that load factors induced by intentional acrobatics, abrupt pullups from dives, high-speed stalls, and gusts at high airspeeds all place added stress on the entire structure of an airplane.

Stress on the structure involves forces on any part of the airplane. There is a tendency for the uninformed to think of load factors only in terms of their effect on spars and struts. Most structural failures due to excess load factors involve rib structure within the leading and trailing edges of wings and tail group. The critical area of fabric-covered airplanes is the covering about one-third of the chord aft on the top surface of the wing.

The cumulative effect of such loads over a long period of time may tend to loosen and weaken vital parts so that actual failure may occur later when the airplane is being operated in a normal manner.

Vg Diagram

The flight operating strength of an airplane is presented on a graph whose horizontal scale is based on load factor (Fig. 3-19). The diagram is called a Vg diagram—velocity versus "g" loads or load factor. Each airplane has its own V-g diagram which is valid at a certain weight and altitude.

The lines of maximum lift capability (curved lines) are the first items of importance on the Vg diagram. The subject airplane in the illustration is capable of developing no more than one positive "g" at 62 m.p.h., the wing level stall speed of the airplane. Since the maximum load factor varies with the square of the airspeed, the maximum positive lift capability of this airplane is 2 "g" at 92 m.p.h., 3 "g" at 112 m.p.h., 4.4 "g" at 137 m.p.h., etc.. Any load factor above this line is unavailable aerodynamically; i.e., the subject airplane cannot fly above the line of maximum lift capability (it will stall).

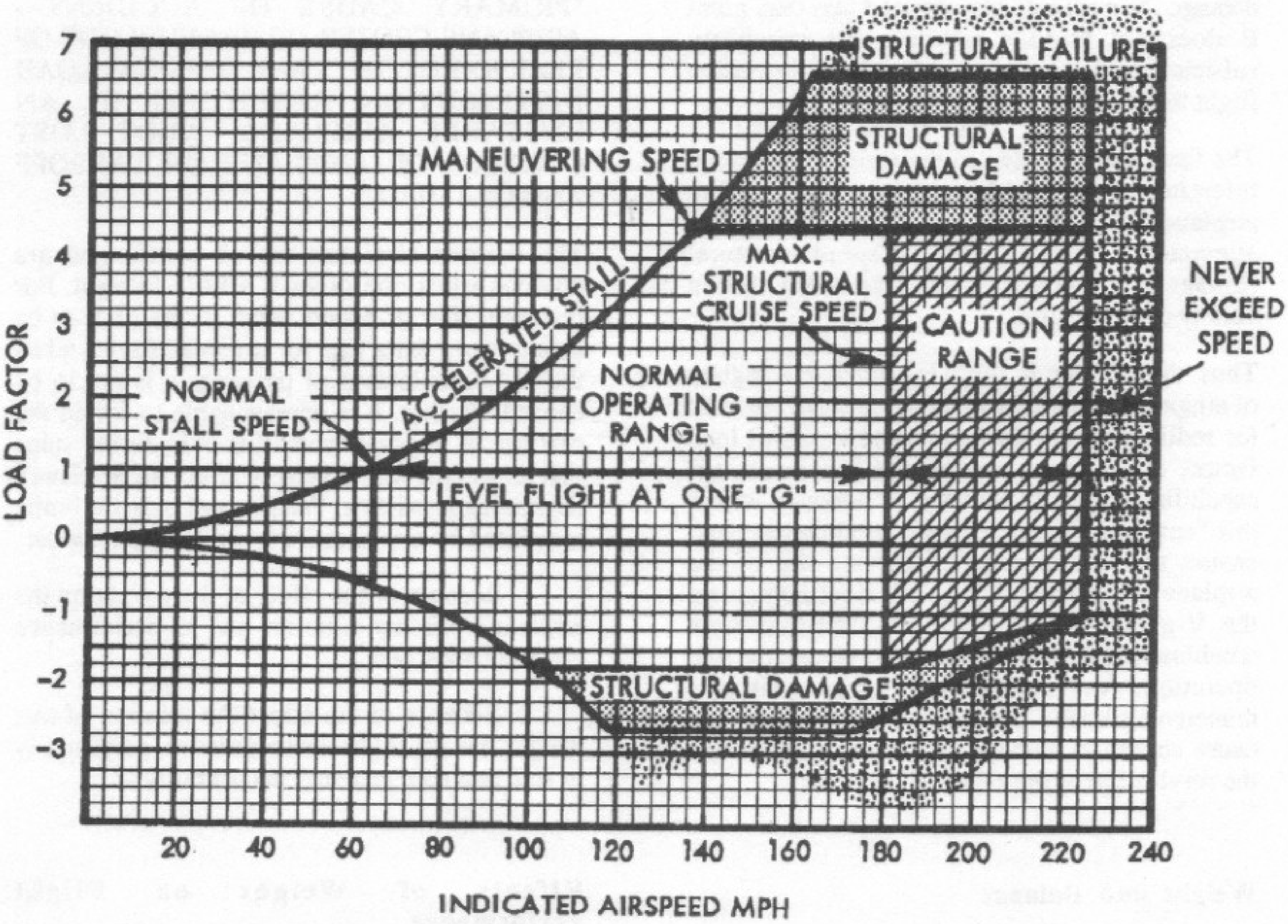


Figure 3-50 Typical Vg Diagram

Essentially the same situation exists for negative lift flight with the exception that the speed necessary to produce a given negative load factor is higher than that to produce the same positive load factor.

If the subject airplane is flown at a positive load factor greater than the positive limit load factor of 4.4, structural damage will be possible. When the airplane is operated in this region, objectionable permanent deformation of the primary structure may take place and a high rate of fatigue damage is incurred. Operation above the limit load factor must be avoided in normal operation.

There are two other points of importance on the Vg diagram. Point A is the intersection of the positive limit load factor and the line of maximum positive lift capability. The airspeed at this point is the minimum airspeed at which the limit load can be developed aerodynamically. Any airspeed greater than point A provides a positive lift capability sufficient to damage the airplane; any airspeed less than point A does NOT provide positive lift capability sufficient to cause damage from excessive flight loads. The usual term given to the speed at point A is the "maneuvering speed," since consideration of subsonic aerodynamics would predict minimum usable turn radius to occur at this condition. The maneuver speed is a valuable reference point since an airplane operating below this point cannot produce a damaging positive flight load. Any combination of maneuver and gust cannot create damage due to excess airload when the airplane is below the maneuver speed.

Point B is the intersection of the negative limit load factor and line of maximum negative lift capability. Any airspeed greater than point B provides a negative lift capability sufficient to damage the airplane; any airspeed less than point B does not provide negative lift capability sufficient to damage the airplane from excessive flight loads.

The limit airspeed (or redline speed) is a design reference point for the airplane—the subject airplane is limited to 225 m.p.h. If flight is attempted beyond the limit airspeed structural damage or structural failure may result from a variety of phenomena.

Thus, the airplane in flight is limited to a regime of airspeeds and g's which do not exceed the limit (or redline) speed, do not exceed the limit load factor, and cannot exceed the maximum lift capability. The airplane must be operated within this "envelope" to prevent structural damage and ensure that the anticipated service life of the airplane is obtained. The pilot must appreciate the V-g diagram as describing the allowable combination of airspeeds and load factors for safe operation. Any maneuver, gust, or gust plus maneuver outside the structural envelope can cause structural damage and effectively shorten the service life of the airplane.

Weight and Balance

Often a pilot regards the airplane's weight and balance data as information of interest only to engineers, dispatchers, and operators of scheduled and nonscheduled air carriers. Along with this idea, the reasoning is that the airplane was weighed during the certification process and that this data is valid indefinitely, regardless of equipment changes or modifications. Further, this information is mistakenly reduced to a workable routine or "rule of thumb" such as: "If I have three passengers, I can load only 100 gallons of fuel; four passengers—70 gallons." Admittedly, this rule of thumb is adequate in many cases, but as the subject "Weight and Balance" suggests, we are concerned not only with the weight of the airplane but also the location of its center of gravity. The importance of the C.G. should have become apparent in the discussions of stability, controllability, and performance. If all pilots understood and respected the effect of C.G. on an airplane, then one type of accident would be eliminated from our records:

"PRIMARY CAUSE OF ACCIDENT—AIRPLANE CENTER OF GRAVITY OUT OF REARWARD LIMITS AND UNEQUAL LOAD DISTRIBUTION RESULTING IN AN UNSTABLE AIRPLANE. PILOT LOST CONTROL OF AIRPLANE ON TAKEOFF AND CRASHED."

The reasons airplanes are so certificated are obvious when one gives it a little thought. For instance, it is of added value to the pilot to be able to carry extra fuel for extended flights when the full complement of passengers is not to be carried. Further, it is unreasonable to forbid the carriage of baggage when it is only during spins that its weight will adversely affect the airplane's flight characteristics. Weight and balance limits are placed on airplanes for two principal reasons:

1. Because of the effect of the weight on the airplane's primary structure and its performance characteristics; and

2. Because of the effect the location of this weight has on flight characteristics, particularly in stall and spin recovery and stability.

Effects of Weight on Flight Performance

The takeoff/climb and landing performance of an airplane are determined on the basis of its maximum allowable takeoff and landing weights. A heavier gross weight will result in a longer takeoff run and shallower climb, and a faster touchdown speed and longer landing roll. Even a minor overload may make it impossible for the airplane to clear an obstacle which normally would not have been seriously considered during takeoffs under more favorable conditions.

The detrimental effects of overloading on performance are not limited to the immediate hazards involving takeoffs and landings. Overloading has an adverse effect on all climb and cruise performance which leads to overheating during climbs, added wear on engine parts, increased fuel consumption, slower cruising speeds, and reduced range.

The manufacturers of modern airplanes furnish weight and balance data with each airplane produced. Generally, this information may be found in the FAA approved Airplane Flight Manual or pilot's operating handbook. With the advancements in airplane design and construction in recent years has come the development of "easy to read charts" for determining weight and balance data. Increased performance and load carrying capability of these airplanes require strict adherence to the operating limitations prescribed by the manufacturer. Deviations from the recommendations can result in structural damage or even complete failure of the airplane's structure. Even if an airplane is loaded well within the maximum weight limitations, it is imperative that weight distribution be within the limits of center of gravity location. The preceding brief study of aerodynamics and load factors points out the reasons for this precaution. The following discussion is background information into some of the reasons why weight and balance conditions are important to the safe flight of an airplane.

The pilot is often completely unaware of the weight and balance limitations of the airplane being flown and of the reasons for these limitations. In some airplanes it is not possible to fill all seats, baggage compartments, and fuel tanks, and still remain within approved weight or balance limits. As an example, in several popular four-place airplanes the fuel tanks may not be filled to capacity when four occupants and their baggage are carried. In a certain two-place airplane, no baggage may be carried in the compartment aft of the seats when spins are to be practiced.

Effects of Weight on Airplane Structure

The effect of additional weight on the wing structure of an airplane is not readily apparent. Airworthiness requirements prescribe that the structure of an airplane certificated in the normal category (in which acrobatics are prohibited) must be strong enough to withstand a load factor of 3.8 to take care of dynamic loads caused by maneuvering and gusts. This means that the primary structure of the airplane can withstand a load of 3.8 times the approved gross weight of the airplane without structural failure occurring. If this is accepted as indicative of the load factors which may be imposed during operations for which the airplane is intended, a 100-pound overload imposes a potential structural overload of 380 pounds. The same consideration is even more impressive in the case of utility and acrobatic category airplanes, which have load factor requirements of 4.4 and 6.0 respectively.

Structural failures which result from overloading may be dramatic and catastrophic, but more often they affect structural components progressively in a manner which is difficult to detect and expensive to repair. One of the most serious results of habitual overloading is that its results tend to be cumulative, and may result in structural failure later during completely normal operations. The additional stress placed on structural parts by overloading is believed to accelerate the occurrence of metallic fatigue failures.

A knowledge of load factors imposed by flight maneuvers and gusts will emphasize the consequences of an increase in the gross weight of an airplane. The structure of an airplane about to undergo a load factor of 3 G's, as in the recovery from a steep dive, must be prepared to withstand an added load of 300 pounds for each 100-pound increase in weight. It should be noted that this would be imposed by the addition of about 16 gallons of unneeded fuel in a particular airplane. The FAA certificated civil airplane has been analyzed structurally, and tested for flight at the maximum gross weight authorized and within the speeds posted for the type of flights to be performed. Flights at weights in excess of this amount are quite possible and often are well within the performance capabilities of an airplane. Nonetheless, this fact should not be allowed to mislead the pilot, as he may not realize that loads for which the airplane was not designed are being imposed on all or some part of the structure.

In loading an airplane with either passengers or cargo, the structure must be considered. Seats, baggage compartments, and cabin floors are designed for a certain load or concentration of load and no more. As an example, a light-plane baggage compartment may be placarded for 20 pounds because of the limited strength of its supporting structure even though the airplane may not be overloaded or out of center of gravity limits with more weight at that location.

Effect of Weight on Stability and Controllability

The effects that overloading has on stability also are not generally recognized. An airplane which is observed to be quite stable and controllable when loaded normally, may be discovered to have very different flight characteristics when it is overloaded. Although the distribution of weight has the most direct effect on this, an increase in the airplane's gross weight may be expected to have an adverse effect on stability, regardless of location of the center of gravity.

The stability of many certificated airplanes is completely unsatisfactory if the gross weight is exceeded.

Effect of Load Distribution

The effect of the position of the center of gravity on the load imposed on an airplane's wing in flight is not generally realized, although it may be very significant to climb and cruising performance. Contrary to the beliefs of some pilots, an airplane with forward loading is "heavier" and consequently, slower than the same airplane with the center of gravity further aft.

Fig. 3-51 illustrates the reason for this. With forward loading, "nose up" trim is required in most airplanes to maintain level cruising flight. Nose-up trim involves setting the tail surfaces to produce a greater down load on the aft portion of the fuselage, which adds to the wing loading and the total lift required from the wing if altitude is to be maintained. This requires a higher angle of attack of the wing, which results in more drag and, in turn, produces a higher stalling speed.

With aft loading and "nose-down" trim, the tail surfaces will exert less down load, relieving the wing of that much wing loading and lift required to maintain altitude. The required angle of attack of the wing is less, so the drag is less, allowing for a faster cruise speed. Theoretically, a neutral load on the tail surfaces in cruising flight would produce the most efficient overall performance and fastest cruising speed, but would also result in instability. Consequently, modern airplanes are designed to require a down load on the tail for stability and controllability.

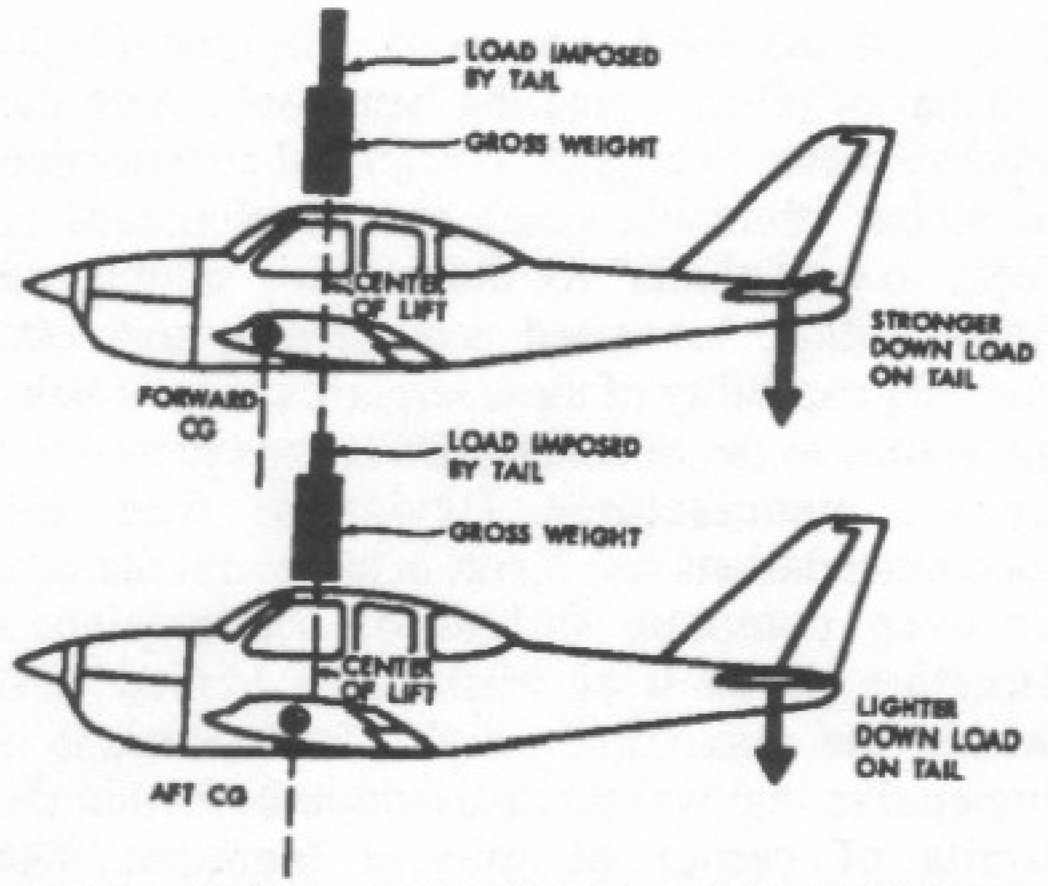


Figure 3-51 Load Distribution Affects Balance

Remember that a zero indication on the trim tab control is not necessarily the same as "neutral trim" because of the force exerted by downwash from the wings and the fuselage on the tail surfaces.

The effects of the distribution of the airplane's useful load have a significant influence on its flight characteristics, even when the load is within the center of gravity limits and the maximum permissible gross weight. Important among these effects are changes in controllability, stability, and the actual load imposed on the wing.

Generally speaking, an airplane becomes less controllable, especially at slow flight speeds, as the center of gravity is moved further aft. An airplane which cleanly recovers from a prolonged spin with the center of gravity at one position may fail completely to respond to normal recovery attempts when the center of gravity is moved aft by 1 or 2 inches.

It is common practice for airplane designers to establish an aft center of gravity limit that is within 1 inch of the maximum which will allow normal recovery from a one-turn spin. When certificating an airplane in the utility category to permit intentional spins, the aft center of gravity limit is usually established at a point several inches forward of that which is permissible for certification in the normal category.

Another factor affecting controllability which is becoming more important in current designs of large airplanes is the effect of long moment arms to the positions of heavy equipment and cargo. The same airplane may be loaded to maximum gross weight within its center of gravity limits by concentrating fuel, passengers, and cargo near the design center of gravity; or by dispersing fuel and cargo loads in wingtip tanks and cargo bins forward and aft of the cabin.

With the same total weight and center of gravity, maneuvering the airplane or maintaining level flight in turbulent air will require the application of greater control forces when the load is dispersed. This is true because of the longer moment arms to the positions of the heavy fuel and cargo loads which must be overcome by the action of the control surfaces. An airplane with full outboard wing tanks or tip tanks tends to be sluggish in roll when control situations are marginal, while one with full nose and aft cargo bins tends to be less responsive to the elevator controls.

The rearward center of gravity limit of an airplane is determined largely by considerations of stability. The original airworthiness requirements for a type certificate specify that an airplane in flight at a certain speed will dampen out vertical displacement of the nose within a certain number of oscillations. An airplane loaded too far rearward may not do this; instead when the nose is momentarily pulled up, it may alternately climb and dive becoming steeper with each oscillation. This instability is not only uncomfortable to occupants but it could even become dangerous by making the airplane unmanageable under certain conditions.

The recovery from a stall in any airplane becomes progressively more difficult as its center of gravity moves aft. This is particularly important in spin recovery, as there is a point in rearward loading of any airplane at which a "flat" spin will develop. A flat spin is one in which centrifugal force, acting through a center of gravity located well to the rear, will pull the tail of the airplane out away from the axis of the spin, making it impossible to get the nose down and recover.

An airplane loaded to the rear limit of its permissible center of gravity range will handle differently in turns and stall maneuvers and have different landing characteristics than when it is loaded near the forward limit.

The forward center of gravity limit is determined by a number of considerations. As a safety measure, it is required that the trimming device, whether tab or adjustable stabilizer, be capable of holding the airplane in a normal glide with the power off. A conventional airplane must be capable of a full stall, power-off landing in order to ensure minimum landing speed in emergencies. A tailwheel type airplane loaded excessively nose heavy will be difficult to taxi, particularly in high winds. It can be nosed over easily by use of the brakes, and it will be difficult to land without bouncing since it tends to pitch down on the wheels as it is slowed down and flared for landing. Steering difficulties on the ground may occur in nosewheel-type airplanes, particularly during the landing roll and takeoff.

1. The CG position influences the lift and angle of attack of the wing, the amount and direction of force on the tail, and the degree of deflection of the stabilizer needed to supply the proper tail force for equilibrium. The latter is very important because of its relationship to elevator control force.

2. The airplane will stall at a higher speed with a forward CG location. This is because the stalling angle of attack is reached at a higher speed due to increased wing loading.

3. Higher elevator control forces normally exist with a forward CG location due to the increased stabilizer deflection required to balance the airplane.

4. The airplane will cruise faster with an aft CG location because of reduced drag. The drag is reduced because a smaller angle of attack and less downward deflection of the stabilizer are required to support the airplane and overcome the nose-down pitching tendency.

5. The airplane becomes less and less stable as the CG is moved rearward. This is because when the angle of attack is increased it tends to result in additional increased angle of attack. Therefore, the wing contribution to the airplane's stability is now decreased, while the tail contribution is still stabilizing. When the point is reached that the wing and tail contributions balance, then neutral stability exists. Any CG movement further aft will result in an unstable airplane.

6. A forward CG location increases the need for greater back elevator pressure. The elevator may no longer be able to oppose any increase in nose-down pitching. Adequate elevator control is needed to control the airplane throughout the airspeed range down to the stall.

Weight and Balance Control

The aircraft manufacturer and the Federal Aviation Administration have major roles in designing and certificating the aircraft with a safe and workable means of controlling weight and balance. If the prototype aircraft has weight and balance control problems which are potentially dangerous or complicated, design changes must be made to ensure the airworthiness of the aircraft.

When an airplane is placarded requiring that it must be flown solo from a specified seat, that a given fuel tank is to be emptied first, or that a compartment or seat is to be left empty under certain conditions, the pilot may rest assured that the placard is necessary for some well-founded reason. Such placards must be maintained in the airplane and observed.

Weight and balance control is a matter of serious concern to all pilots as well as many other people who are involved in the flight. The pilot has to personally assume the responsibility because he has control over both the loading and the fuel management, the two variable factors which can change both total weight and CG location. Weight and balance information is available to the pilot in the form of aircraft records, operating handbooks, and placards in baggage compartments and on fuel tank caps. It is the aircraft owner or operator's responsibility to make certain that up-to-date information is available in the aircraft for the pilot's use.

It must be stressed that the empty weight and moment given in most manufacturers' handbooks are for the basic airplane prior to the installation of additional optional equipment. When the owner later adds such items as radio navigation equipment, auto pilot, deicers, etc.., the empty weight and the moment are changed. These changes must be recorded in the airplane's weight and balance data and used in all computations. In addition, the actual weight of occupants, baggage, fuel, and other useful load should be used rather than the sample weights given in the manufacturers' handbooks.

The owner or operator of the aircraft should ensure that maintenance personnel make appropriate entries in the aircraft maintenance records when repairs or modifications have been accomplished, and when optional equipment (radios, etc..) have been installed, or removed. Weight changes must be accounted for and proper notations made in the weight and balance records. Without such notations, the pilot has no foundation upon which to base his calculations and decisions.

The airplane's latest Weight and Balance Loading Form and/or its maintenance record will list the empty weight, the useful load, and the empty center of gravity location. If it is possible to load the airplane out of center of gravity limits, it will also include a specific listing of the most forward and the most rearward allowable limits. This information should be consulted when a pilot proposes to load and fly an airplane with which he or she is not thoroughly familiar.

Although seemingly complex at times, all weight and balance problems are based on the following moment equation:

```
Moment = Weight X Arm
```

This equation is the basic equation used to find the center of gravity location of an airplane and/or its components. By rearrangement of this equation to the forms, Weight=Moment÷Arm, and Arm=Moment÷Weight, with any two known values, the third value can be found.

Weight and balance computations are sometimes simplified by two graphic aids—the loading graph and the center of gravity moment envelope. The loading graph (Fig. 3-52) is typical of those found in many general aviation airplane owner's manuals. This graph, in effect, multiplies weight by arm giving moment, then divides the moment by a reduction factor, giving an index number. Weight values appear along the left side of the graph. The moment/1,000 or index numbers are along the bottom. In this chart, each line representing a useful load item is labeled. To determine the moment of any load item, find the weight along the left margin, then project a line directly to a point of intersection with the appropriate load item line. The CG moment envelope (Fig. 3-53) allows the pilot to bypass the computation of a CG number. It gives an acceptable range of index numbers for any airplane weight from minimum to maximum. If the lines from total weight and total moment intersect within the envelop, the airplane is within weight and balance limits.

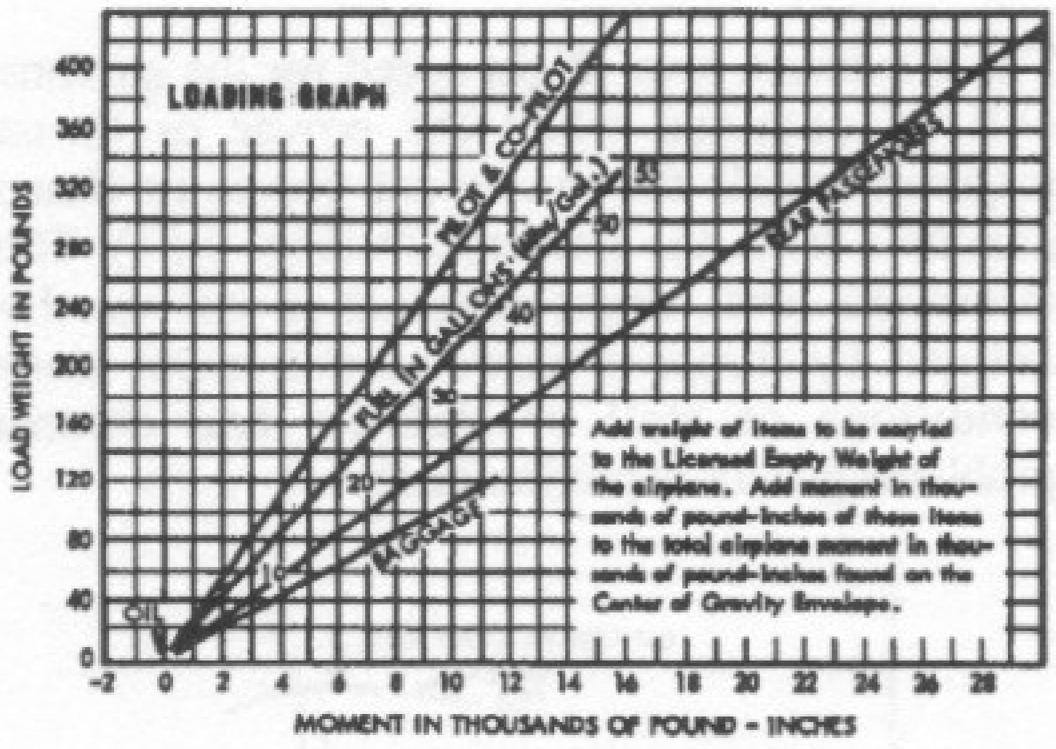


Figure 3-52 Sample Loading Graph

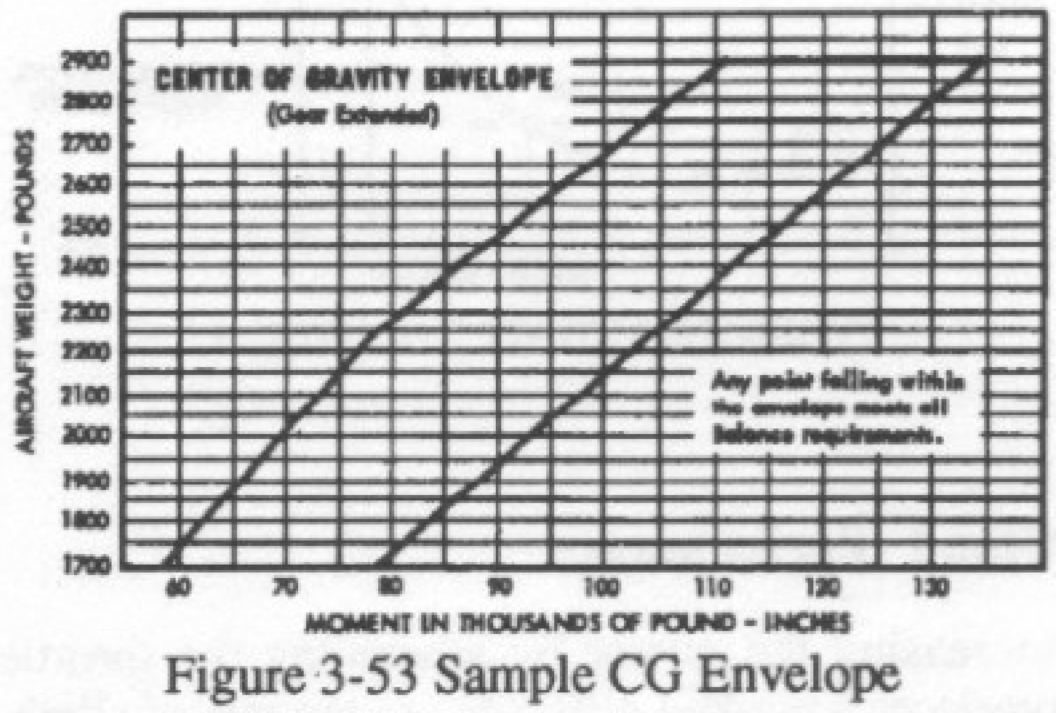


Figure 3-53 Sample CG Envelope

For those who have a need for greater appreciation of weight and balance control as it relates to safety of flight, a more comprehensive text on the subject is contained in the FAA Advisory Circular 91-23A, *Pilot's Weight and Balance Handbook*.

Airplane Performance

The clean, sleek appearance and the splendid performance characteristics of modern airplanes, even the most inexpensive training airplanes, reflect the demands of modern travel, business, and industry. The present-day airplane is clean and sleek because aerodynamic cleanness of line, and efficiency of design, result in greater range, speed, and payload at the least operating cost. Although this is especially important in the functions of large jet transports, it is also a principal factor in the operation of executive and personal type airplanes.

"Performance" is a term used to describe the ability of an airplane to accomplish certain things which make it useful for certain purposes. For example, the ability of the airplane to land and take off in a very short distance is an important factor to the pilot who operates in and out of confined fields. The ability to carry heavy loads, fly at high altitudes at fast speeds, or travel long distances is essential performance for operators of airline and executive type airplanes.

The chief elements of performance are the takeoff and landing distance, rate of climb, ceiling, payload, range, speed, maneuverability, stability, and fuel economy. Some of these factors are often directly opposed: for example, high speed versus shortness of landing distance; long range versus great payload; and high rate of climb versus fuel economy. It is the preeminence of one or more of these factors which dictates differences between airplanes and which explains the high degree of specialization found in modern airplanes.

The various items of airplane performance result from the combination of airplane and powerplant characteristics. The aerodynamic characteristics of the airplane generally define the power and thrust *requirements* at various conditions of flight while powerplant characteristics generally define the power and thrust *available* at various conditions of flight. The matching of the aerodynamic configuration with the powerplant is accomplished by the manufacturer to provide maximum performance at the specific design condition, e.g., range, endurance, climb, etc..

Straight-and-Level Flight

All of the principal items of flight performance involve steady-state flight conditions and equilibrium of the airplane. For the airplane to remain in steady level flight, equilibrium must be obtained by a lift equal to the airplane weight and a powerplant thrust equal to the airplane drag. Thus, the airplane drag defines the *thrust required* to maintain steady level flight.

All parts of the airplane that are exposed to the air contribute to the drag, though only the wings provide lift of any significance. For this reason, and certain others related to it, the total drag may be divided into two parts, the wing drag (induced) and the drag of everything but the wings (parasite).

The total power required for flight then can be considered as the sum of induced and parasite effects; that is, the total drag of the airplane. Parasite drag is the sum of pressure and friction drag which is due to the airplane's basic configuration and, as defined, is independent of lift. Induced drag is the undesirable but unavoidable consequence of the development of lift.

While the parasite drag predominates at high speed, induced drag predominates at low speed (Fig. 3-18). For example, if an airplane in a steady flight condition at 100 knots is then accelerated to 200 knots, the *parasite* drag becomes four times as great but the power required to overcome that drag is eight times the original value. Conversely, when the airplane is operated in steady level flight at twice as great a speed, the *induced* drag is one-fourth the original value and the power required to overcome that drag is only one-half the original value.

The wing or induced drag changes with speed in a very different way, because of the changes in the angle of attack. Near the stalling speed the wing is inclined to the relative wind at nearly the stalling angle, and its drag is very strong. But at ordinary flying speeds, with the angle of attack nearly zero, the wing cuts through the air almost like a knife, and the drag is minimal. After attaining a certain high speed, the angle of attack changes very little with any further increase in speed and the drag of the wing increases in direct proportion to any further increase in speed. This does not consider the factor of compressibility drag which is involved at speeds beyond the top speed of most general aviation airplanes.

To sum up these changes, for a typical, moderately powered airplane: As the speed increases from stalling speed to top speed, the induced drag decreases and parasite drag increases. As a result *the total drag decreases for the first part of the range and then increases again.*

When the airplane is in steady, level flight, the condition of equilibrium must prevail. The unaccelerated condition of flight is achieved with the airplane trimmed for lift equal to weight and the powerplant set for a thrust to equal the airplane drag.

The maximum level flight speed for the airplane will be obtained when the power or thrust required equals the maximum power or thrust available from the powerplant (Fig. 3-54). The minimum level flight airspeed is not usually defined by thrust or power requirement since conditions of stall or stability and control problems generally predominate.

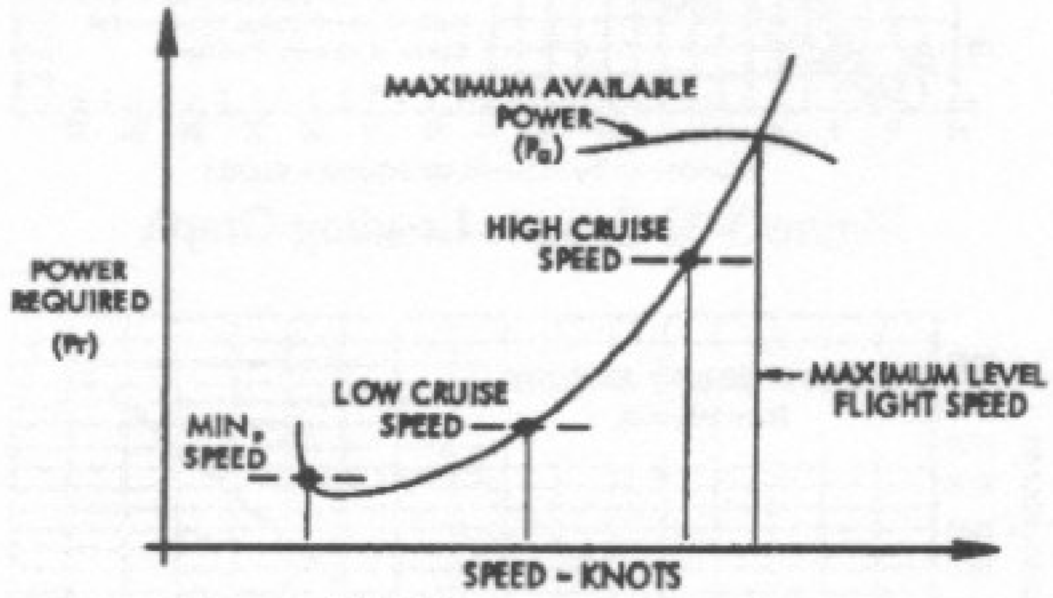


Figure 3-54 Power Versus Speed

Climb Performance

Increasing the power by advancing the throttle produces a marked difference in the rate of climb. Climb depends upon the *reserve* power or thrust. Reserve power is the available power over and above that required to maintain horizontal flight at a given speed. Thus, if an airplane is equipped with an engine which produces 200 total available horsepower and the airplane requires only 130 horsepower at a certain level flight speed, the power available for climb is 70 horsepower.

Although we sometimes use the terms "power" and "thrust" interchangeably, erroneously implying that they are synonymous, it is well to distinguish between the two when discussing climb performance. Work is the product of a force moving through a distance and is usually independent of time. Work is measured by several standards, the most common unit is called a "foot-pound." If a 1-pound mass is raised 1 foot, a work unit of 1 foot-pound has been performed. The common unit of mechanical power is horsepower; one horsepower is work equivalent to lifting 33,000 pounds a vertical distance of 1 foot in 1 minute. The term, "power," implies work *rate* or units of work per unit of time, and as such is a function of the speed at which the force is developed. "Thrust," also a function of work, means the force which imparts a change in the velocity of a mass. This force is measured in pounds but has no element of time or rate. It can be said then, that during a steady climb, the *rate* of climb is a function of excess thrust.

When the airplane is in steady level flight or with a slight angle of climb, the vertical component of lift is very nearly the same as the actual total lift. Such climbing flight would exist with the lift very nearly equal to the weight. The net thrust of the powerplant may be inclined relative to the flightpath but this effect will be neglected here for the sake of simplicity. Although the weight of the airplane acts vertically, a component of weight will act rearward along the flightpath (Fig. 3-55).

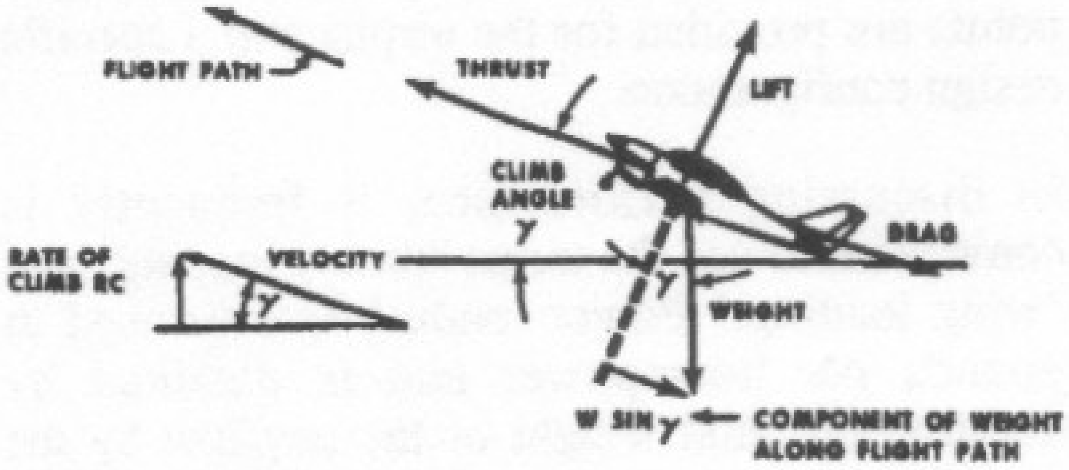


Figure 3-55 Weight Has Rearward Component

If it is assumed that the airplane is in a steady climb with essentially a small inclination of the flightpath, the summation of forces along the flightpath resolves to the following:

```
Forces forward=Forces aft
```

The *basic* relationship neglects some of the factors which may be of importance for airplanes of very high climb performance. (For example, a more detailed consideration would account for the inclination of thrust from the flightpath, lift not being equal to weight, a subsequent change of induced drag, etc..) However, this *basic* relationship will define the principal factors affecting climb performance.

This relationship means that, for a given weight of the airplane, *the angle of climb* depends on the difference between thrust and drag, or the excess thrust (Fig. 3-56). Of course, when the excess thrust is zero, the inclination of the flightpath is zero and the airplane will be in steady, level flight. When the thrust is greater than the drag, the excess thrust will allow a climb angle depending on the value of excess thrust. On the other hand, when the thrust is less than the drag, the deficiency of thrust will allow an angle of descent.

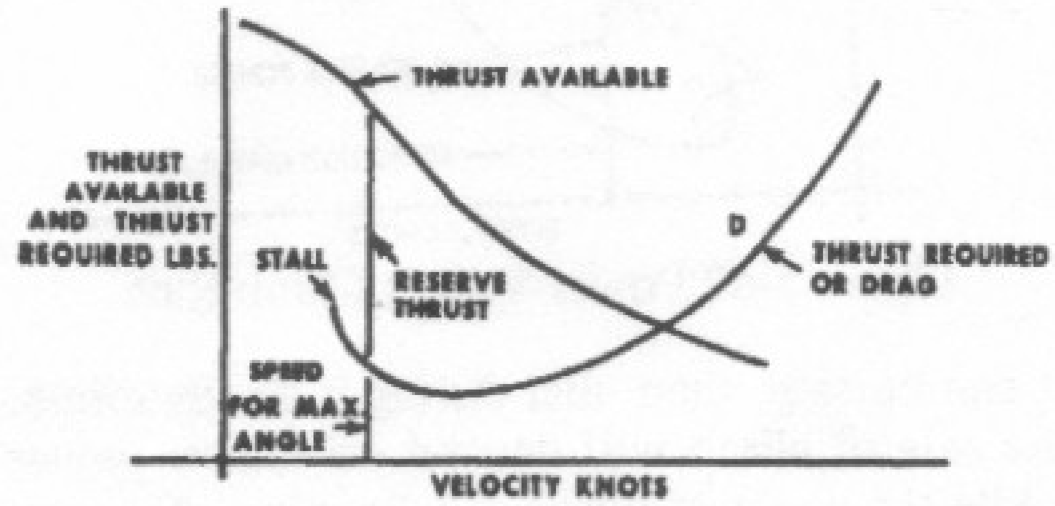


Figure 3-56 Thrust Versus Climb Angle

The most immediate interest in the climb angle performance involves obstacle clearance. The most obvious purpose for which it might be used is to clear obstacles when climbing out of short or confined airports.

The maximum *angle of climb* would occur where there exists the greatest difference between thrust available and thrust required; i.e., for the propeller powered airplane, the maximum excess thrust and angle of climb will occur at some speed just above the stall speed. Thus, if it is necessary to clear an obstacle after takeoff, the propeller powered airplane will attain maximum angle of climb at an airspeed close to—if not at—the takeoff speed.

Of greater general interest in climb performance are the factors which affect the *rate of climb*. The vertical velocity of an airplane depends on the flight speed and the inclination of the flightpath. In fact, the rate of climb is the vertical component of the flightpath velocity.

For rate of climb, the maximum rate would occur where there exists the greatest difference between power available and power required (Fig. 3-57).

This relationship means that, for a given weight of the airplane, the *rate of climb* depends on the difference between the power available and the power required, or the excess power. Of course, when the excess power is zero, the rate of climb is zero and the airplane is in steady level flight. When power available is greater than the power required, the excess power will allow a rate of climb specific to the magnitude of excess power.

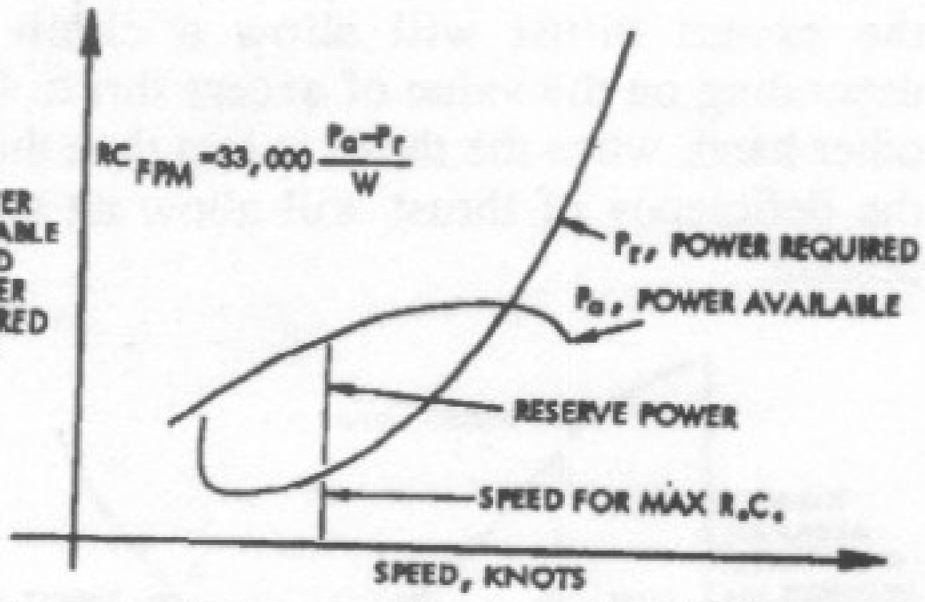


Figure 3-57 Power Versus Climb Rate

It can be said, then, that during a steady climb, the *rate* of climb will depend on *excess power* while the angle of climb is a function of *excess thrust*.

The climb performance of an airplane is affected by certain variables. The conditions of the airplane's maximum climb angle or maximum climb rate occur at specific speeds, and variations in speed will produce variations in climb performance. Generally, there is sufficient latitude in most general aviation airplanes that small variations in speed from the optimum do not produce large changes in climb performance, and certain operational considerations may require speeds slightly different from the optimum. Of course, climb performance would be most critical with high gross weight, at high altitude, in obstructed takeoff areas, or during malfunction of a powerplant. Then, optimum climb speeds are *necessary*.

Weight has a very pronounced effect on airplane performance. If weight is added to the airplane, it must fly at a higher angle of attack to maintain a given altitude and speed. This increases the induced drag of the wings, as well as the parasite drag of the airplane. Increased drag means that additional power is needed to overcome it, which in turn means that less reserve power is available for climbing. Airplane designers go to great effort to minimize the weight since it has such a marked effect on the factors pertaining to performance.

A change in the airplane's weight produces a twofold effect on climb performance. First. the weight affects both the climb angle and the climb rate. In addition, a change in weight will change the drag and the power required. This alters the reserve power available. Generally, an increase in weight will reduce the maximum rate of climb but the airplane must be operated at some increase of climb speed to achieve the smaller peak climb rate.

An increase in altitude also will increase the power required and decrease the power available. Hence, the climb performance of an airplane is affected greatly by altitude. The speeds for maximum rate of climb, maximum angle of climb, and maximum and minimum level flight airspeeds vary with altitude. As altitude is increased, these various speeds finally converge at the *absolute* ceiling of the airplane. At the absolute ceiling, there is no excess of power and only one speed will allow steady level flight. Consequently, the absolute ceiling of the airplane produces zero rate of climb. The *service ceiling is* the altitude at which the airplane is unable to climb at a rate greater than 100 feet per minute. Usually, these specific performance reference points are provided for the airplane at a specific design configuration.

In discussing performance, it frequently is convenient to use the terms "power loading" and "wing loading." Power loading is expressed in pounds per horsepower and is obtained by dividing the total weight of the airplane by the rated horsepower of the engine. It is a significant factor in the airplane's takeoff and climb capabilities. Wing loading is expressed in pounds per square foot and is obtained by dividing the total weight of the airplane in pounds by the wing area (including ailerons) in square feet. It is the airplane's wing loading that determines the landing speed. These factors are discussed in subsequent sections of this chapter.

Range Performance

The ability of an airplane to convert fuel energy into flying distance is one of the most important items of airplane performance. In flying operations, the problem of efficient range operation of an airplane appears in two general forms: (1) to extract the maximum flying distance from a given fuel load or (2) to fly a specified distance with a minimum expenditure of fuel. A common denominator for each of these operating problems is the "specific range"; that is, nautical miles of flying distance per pound of fuel. Cruise flight operations for maximum range should be conducted so that the airplane obtains maximum specific range throughout the flight.

The specific range can be defined by the following relationship:

$$\text{spec range} = \frac{\text{nautical miles}}{\text{lbs. of fuel}}$$

or

$$\text{spec range} = \frac{\text{nautical MPH}}{\text{lbs. of fuel/hr}}$$

or

knots+fuel flow

If maximum specific range is desired, the flight condition must provide a maximum of speed versus fuel flow.

The general item of *range* must be clearly distinguished from the item of *endurance* (Fig. 3-58). The item of range involves consideration of flying *distance*, while endurance involves consideration of flying *time*. Thus, it is appropriate to define a separate term "specific endurance."

$$\text{spec endurance} = \frac{\text{flight hours}}{\text{lbs. of fuel}}$$

or

$$\text{spec endurance} = \frac{\text{flight hrs/hr.}}{\text{lbs.of fuel/hr.}}$$

or

1+fuel flow

If maximum endurance is desired, the flight condition must provide a minimum of fuel flow.

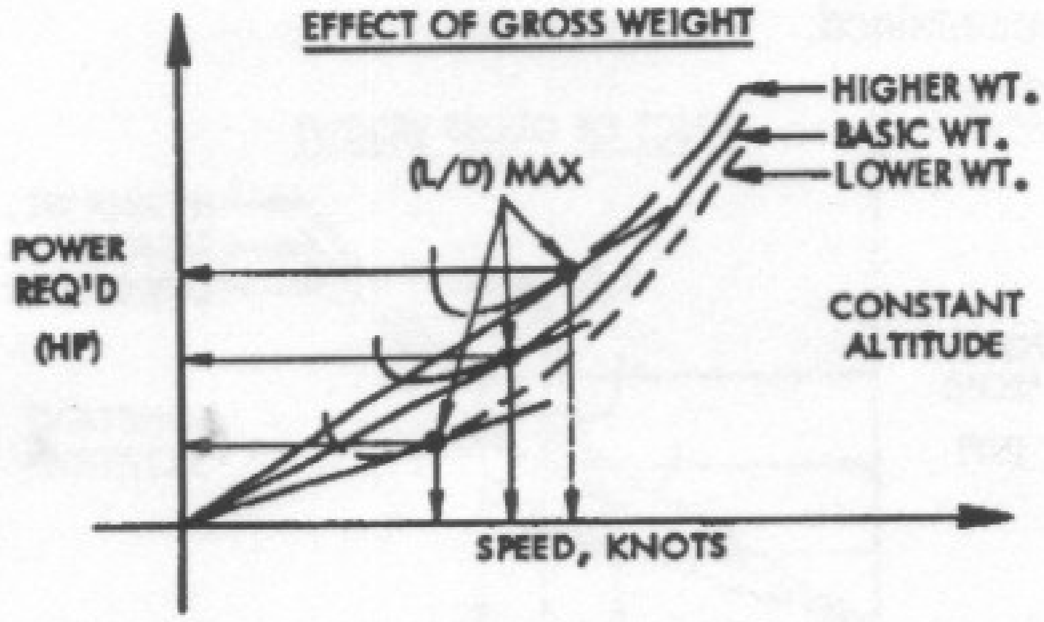


Figure 3-58 Power Versus Range

While the peak value of specific range would provide maximum range operation, long range cruise operation is generally recommended at some slightly higher airspeed. Most long range cruise operations are conducted at the flight condition which provides 99 percent of the absolute maximum specific range. The advantage of such operation is that 1 percent of range is traded for 3 to 5 percent higher cruise speed. Since the higher cruise speed has a great number of advantages, the small sacrifice of range is a fair bargain. The values of specific range versus speed are affected by three principal variables: (1) airplane gross weight, (2) altitude, and (3) the external aerodynamic configuration of the airplane. These are the source of range and endurance operating data included in the performance section of the airplane's flight handbook.

"Cruise control" of an airplane implies that the airplane is operated to maintain the recommended long—range cruise condition throughout the flight. Since fuel is consumed during cruise, the gross weight of the airplane will vary and optimum airspeed, altitude, and power setting can also vary. Generally,"cruise control" means the control of the optimum airspeed, altitude, and power setting to maintain the 99 percent maximum specific range condition. At the beginning of cruise flight, the relatively high initial weight of the airplane will require specific values of airspeed, altitude, and power setting to produce the recommended cruise condition (Fig. 3-59). As fuel is consumed and the airplane's gross weight decreases, the optimum airspeed and power setting may decrease, or, the optimum altitude may increase. In addition, the optimum specific range will increase. Therefore, the pilot must provide the proper cruise control technique to ensure that optimum conditions are maintained.

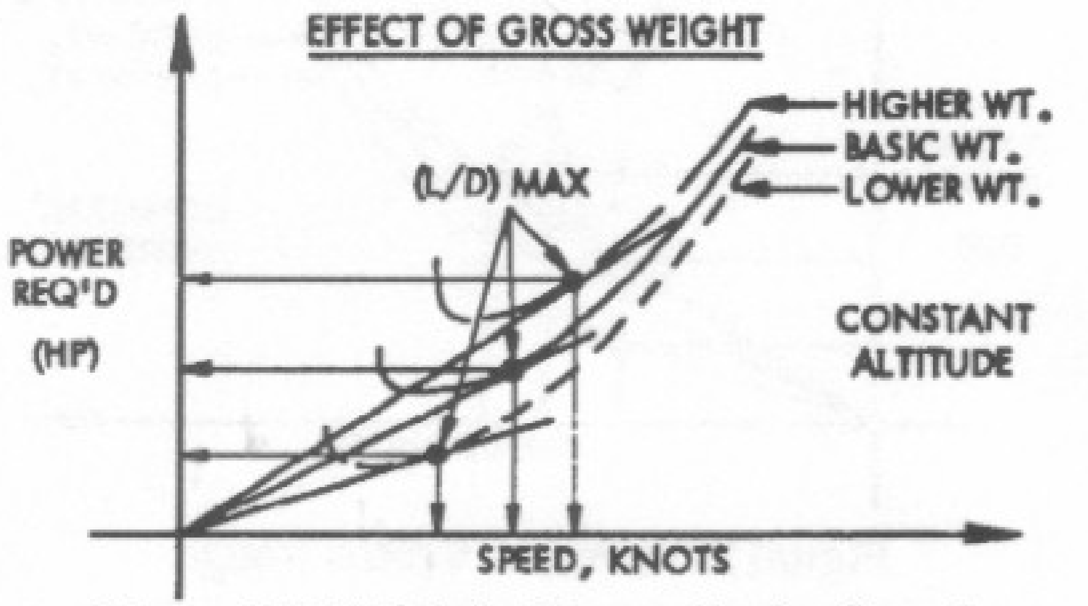


Figure 3-59 Weight Versus Cruise Speed

Total range is dependent on both fuel available and specific range. When range and economy of operation are the principal goals, the pilot must ensure that the airplane will be operated at the recommended long range cruise condition. By this procedure, the airplane will be capable of its maximum design operating radius, or can achieve flight distances less than the maximum with a maximum of fuel reserve at the destination.

The propeller driven airplane combines the propeller with the reciprocating engine for propulsive power. In the case of the reciprocating engine, fuel flow is determined mainly by the shaft *power* put into the propeller rather than *thrust*. Thus, the fuel flow can be related directly to the power required to maintain the airplane in steady, level flight. This fact allows for the determination of range through analysis of power required versus speed—variation of fuel flow versus speed.

The maximum *endurance* condition would be obtained at the point of minimum power required since this would require the lowest fuel flow to keep the airplane in steady, level flight. Maximum *range* condition would occur where the proportion between speed and power required is greatest (Fig. 3-58). The maximum *range* condition is obtained at maximum lift-drag ratio (L/D max) and it is important to note that for a given airplane configuration, the maximum lift-drag ratio occurs at a particular angle of attack and lift coefficient, and is unaffected by weight or altitude.

The flight condition of maximum lift-drag ratio is achieved at one particular value of lift coefficient for a given airplane configuration. Hence, a variation of gross weight will alter the values of airspeed, power required, and specific range obtained at the maximum lift-drag ratio.

The variations of speed and power required must be monitored by the pilot as part of the cruise control procedure to maintain the maximum lift-drag ratio. When the airplane's fuel weight is a small part of the gross weight and the airplane's range is small, the cruise control procedure can be simplified to essentially maintaining a constant speed and power setting throughout the time of cruise flight. On the other hand, the long range airplane has a fuel weight which is a considerable part of the gross weight, and cruise control procedures must employ scheduled airspeed and power changes to maintain optimum range conditions.

The effect of altitude on the range of the propeller driven airplane may be understood by inspection of Fig. 3-60. A flight conducted at high altitude will have a greater true airspeed and the power required will be proportionately greater than when conducted at sea level. The drag of the airplane at altitude is the same as the drag at sea level but the higher true airspeed causes a proportionately greater power required. Note that the straight line that is tangent to the sea level power curve is also tangent to the altitude power curve.

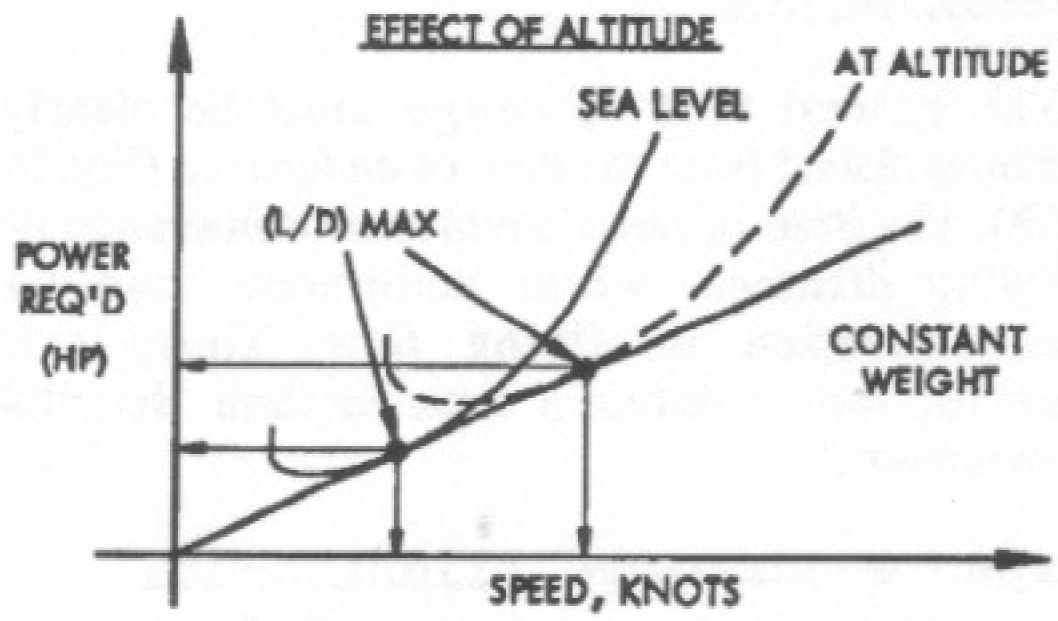


Figure 3-60 Altitude Versus Range

The effect of altitude on specific range also can be appreciated from the previous relationships. If a change in altitude causes identical changes in speed and power required, the proportion of speed to power required would be unchanged. The fact implies that the specific range of the propeller driven airplane would be unaffected by altitude. Actually, this is true to the extent that specific fuel consumption and propeller efficiency are the principal factors which could cause a variation of specific range with altitude. If compressibility effects are negligible, *any variation of specific range with altitude is strictly a function of engine-propeller performance.*

The airplane equipped with the reciprocating engine will experience very little, if any, variation of specific range with altitude at low altitudes. There is negligible variation of brake specific fuel consumption for values of brake horsepower below the maximal cruise power rating of the engine which is the lean range of engine operation. Thus, an increase in altitude will produce a decrease in specific range only when the increased power requirement exceeds the maximum cruise power rating of the engine. One advantage of supercharging is that the cruise power may be maintained at high altitude and the airplane may achieve the range at high altitude with the corresponding increase in true airspeed. The principal differences in the high altitude cruise and low altitude cruise are the true airspeeds and climb fuel requirements.

Takeoff and Landing Performance

The majority of pilot—caused airplane accidents occur during the takeoff and landing phase of flight. Because of this fact, the pilot must be familiar with all the variables which influence the takeoff and landing performance of an airplane and must strive for exacting, professional techniques of operation during these phases of flight.

Takeoff and landing performance is a condition of accelerated and decelerated motion. For instance, during takeoff the airplane starts at zero speed and accelerates to the takeoff speed to become airborne. During landing, the airplane touches down at the landing speed and decelerates to zero speed.

The important factors of takeoff or landing performance are:

1. The takeoff or landing *speed* which will generally be a function of the stall speed or minimum flying speed.

2. The rate of *acceleration* and *deceleration* during the takeoff or landing roll. The acceleration and deceleration experienced by any object vary directly with the unbalance of force and inversely as the mass of the object.

3. The takeoff or landing roll *distance* is a function of both acceleration/deceleration and speed.

Takeoff Performance

The minimum takeoff distance is of primary interest in the operation of any airplane because it defines the runway requirements. The minimum takeoff distance is obtained by taking off at some minimum safe speed which allows sufficient margin above stall and provides satisfactory control and initial rate of climb. Generally, the liftoff speed is some fixed percentage of the stall speed or minimum control speed for the airplane in the takeoff configuration. As such, the lift-off will be accomplished at some particular value of lift coefficient and angle of attack. Depending on the airplane characteristics, the lift-off speed will be anywhere from 1.05 to 1.25 times the stall speed or minimum control speed.

To obtain minimum takeoff distance at the specific lift-off speed, the forces which act on the airplane must provide the maximum acceleration during the takeoff roll. The various forces acting on the airplane may or may not be under the control of the pilot, and various techniques may be necessary in certain airplanes to maintain takeoff acceleration at the highest value.

The powerplant thrust is the principal force to provide the acceleration and, for minimum takeoff distance, the output thrust should be at a maximum. Lift and drag are produced as soon as the airplane has speed, and the values of lift and drag depend on the angle of attack and dynamic pressure.

In addition to the important factors of proper technique, many other variables affect the takeoff performance of an airplane. Any item which alters the takeoff speed or acceleration rate during the takeoff roll will affect the takeoff distance.

For example, the effect of *gross weight* on takeoff distance is significant and proper consideration of this item must be made in predicting the airplane's takeoff distance. Increased gross weight can be considered to produce a threefold effect on takeoff performance: (1) higher lift-off speed, (2) greater mass to accelerate, and (3) increased retarding force (drag and ground friction). If the gross weight increases, a greater speed is necessary to produce the greater lift necessary to get the airplane airborne at the takeoff lift coefficient. As an example of the effect of a change in gross weight, a 21 percent increase in takeoff weight will require a 10 percent increase in lift-off speed to support the greater weight.

A change in gross weight will change the net accelerating force, and change the mass which is being accelerated. If the airplane has a relatively high thrust-to-weight ratio, the change in the net accelerating force is slight and the principal effect on acceleration is due to the change in mass.

The takeoff distance will vary at least as the square of the gross weight. For example, a 10 percent increase in takeoff gross weight would cause:

1. a 5 percent increase in takeoff velocity,

2. at least a 9 percent decrease in rate of acceleration,

3. at least a 21 percent increase in takeoff distance.

For the airplane with a high thrust-to-weight ratio, the increase in takeoff distance might be approximately 21 to 22 percent, but for the airplane with a relatively low thrust-to-weight ratio, the increase in takeoff distance would be approximately 25 to 30 percent. Such a powerful effect requires proper consideration of gross weight in predicting takeoff distance.

The effect of *wind* on takeoff distance is large, and proper consideration also must be provided when predicting takeoff distance. The effect of a headwind is to allow the airplane to reach the lift-off speed at a lower groundspeed while the effect of a tailwind is to require the airplane to achieve a greater groundspeed to attain the lift-off speed.

A headwind which is 10 percent of the takeoff airspeed will reduce the takeoff distance approximately 19 percent (Fig. 3-63). However, a tailwind which is 10 percent of the takeoff airspeed will increase the takeoff distance approximately 21 percent. In the case where the headwind speed is 50 percent of the takeoff speed, the takeoff distance would be approximately 25 percent of the zero wind takeoff distance (75 percent reduction).

The effect of wind on landing distance is identical to the effect on takeoff distance. Figure 3-61 illustrates the general effect of wind by the percent change in takeoff or landing distance as a function of the ratio of wind velocity to takeoff or landing speed.

The effect of *proper takeoff speed is* especially important when runway lengths and takeoff distances are critical. The takeoff speeds specified in the airplane's flight handbook are generally the minimum safe speeds at which the airplane can become airborne. Any attempt to take off below the recommended speed could mean that the airplane may stall, be difficult to control, or have a very low initial rate of climb. In some cases, an excessive angle of attack may not allow the airplane to climb out of ground effect. On the other hand, an excessive airspeed at takeoff may improve the initial rate of climb and "feel" of the airplane, but will produce an undesirable increase in takeoff distance. Assuming that the acceleration is essentially unaffected, the takeoff distance varies as the square of the takeoff velocity.

Thus, 10 percent excess airspeed would increase the takeoff distance 21 percent. In most critical takeoff conditions, such an increase in takeoff distance would be prohibitive and the pilot must adhere to the recommended takeoff speeds.

The effect of *pressure altitude* and *ambient temperature* is to define primarily the *density altitude* and its effect on takeoff performance. While subsequent corrections are appropriate for the effect of temperature on certain items of powerplant performance, *density altitude* defines specific effects on takeoff performance. An increase in density altitude can produce a two-fold effect on takeoff performance: (1) greater takeoff speed and (2) decreased thrust and reduced net accelerating force. If an airplane of given weight and configuration is operated at greater heights above standard sea level, the airplane will still require the same dynamic pressure to become airborne at the takeoff lift coefficient. Thus, the airplane at altitude will take off at the same *indicated* airspeed as at sea level, but because of the reduced air density, the true airspeed will be greater.

The effect of density altitude on powerplant thrust depends much on the type of powerplant. An increase in altitude above *standard sea level,* will bring an immediate decrease in power output for the unsupercharged reciprocating engine. However, an increase in altitude above standard sea level will not cause a decrease in power output for the supercharged reciprocating engine until the altitude exceeds the critical operating altitude. For those powerplants which experience a decay in thrust with an increase in altitude, the effect on the net accelerating force and acceleration rate can be approximated by assuming a direct variation with density. Actually, this assumed variation would closely approximate the effect on airplanes with high

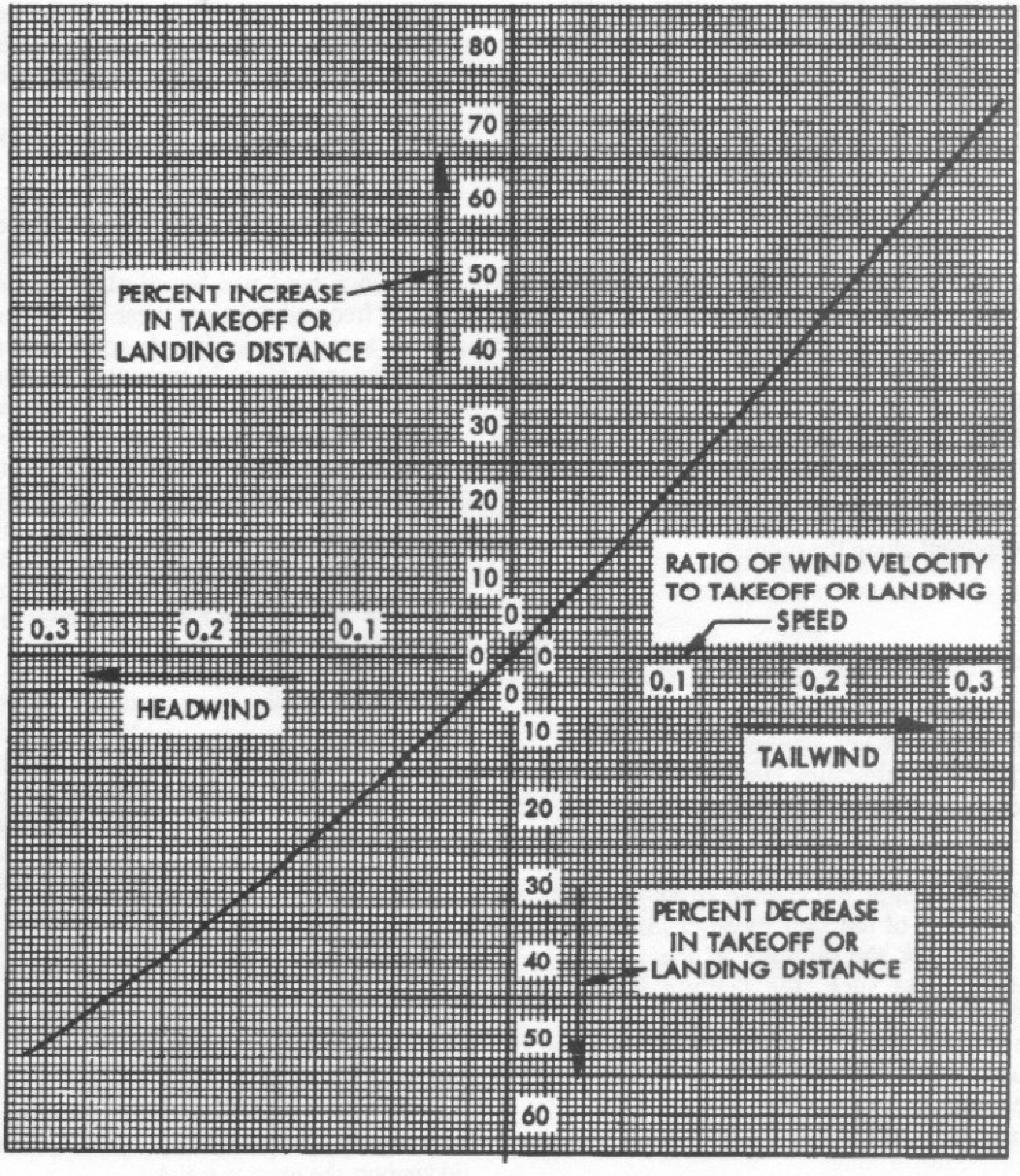


Figure 3-61 Effect of Wind on Takeoff and Landing

thrust-to-weight ratios.

Proper accounting of pressure altitude (field elevation is a poor substitute) and temperature, is mandatory for accurate prediction of takeoff roll distance.

The most critical conditions of takeoff performance are the result of some combination of high gross weight, altitude, temperature, and unfavorable wind. In all cases, it behooves the pilot to make an accurate prediction of takeoff distance from the performance data of the Airplane's Flight Handbook, regardless of the runway available, and to strive for a polished, professional takeoff technique.

In the prediction of takeoff distance from the handbook data, the following primary considerations must be given:

1. Pressure altitude and temperature—to define the effect of density altitude on distance.

2. Gross weight—a large effect on distance.

3. Runway slope and condition—the effect of an incline and retarding effect of snow, ice, etc.

4. Wind—a large effect due to the wind or wind component along the runway.

Landing Performance

In many cases, the landing distance of an airplane will define the runway requirements for flying operations. The minimum landing distance is obtained by landing at some minimum safe speed which allows sufficient margin above stall and provides satisfactory control and capability for a go-around. Generally, the landing speed is some fixed percentage of the stall speed or minimum control speed for the airplane in the landing configuration. As such, the landing will be accomplished at some particular value of lift coefficient and angle of attack. The exact values will depend on the airplane characteristics but, once defined, the values are independent of weight, altitude, wind, etc.

To obtain minimum landing distance at the specified landing speed, the forces which act on the airplane must provide maximum deceleration during the landing roll. The various forces acting on the airplane during the landing roll may require various techniques to maintain landing deceleration at the peak value.

A distinction should be made between the techniques for minimum landing distance and an ordinary landing roll with considerable excess runway available. Minimum landing distance will be obtained with the actual landing speed by creating a continuous peak deceleration of the airplane; that is, extensive use of the brakes for maximum deceleration. On the other hand, an ordinary landing roll with considerable excess runway may allow extensive use of aerodynamic drag to minimize wear and tear on the tires and brakes. If aerodynamic drag is sufficient to cause deceleration of the airplane, it can be used in deference to the brakes in the early stages of the landing roll; i.e., brakes and tires suffer from continuous hard use but airplane aerodynamic drag is free and does not wear out with use. The use of aerodynamic drag is applicable only for deceleration to 60 or 70 percent of the touchdown speed. At speeds less than 60 to 70 percent of the touchdown speed, aerodynamic drag is so slight as to be of little use, and braking must be utilized to produce continued deceleration of the airplane. Since the objective during the landing roll is to decelerate, the powerplant thrust should be the smallest possible positive value (or largest possible negative value in the case of thrust reversers).

In addition to the important factors of proper technique, many other variables affect the landing performance of an airplane. Any item which alters the landing speed or deceleration rate during the landing roll will affect the landing distance.

The effect of *gross weight* on landing distance is one of the principal items determining the landing distance of an airplane. One effect of an increased gross weight is that the airplane will require a greater speed to support the airplane at the landing angle of attack and lift coefficient.

As an example of the effect of a change in gross weight, a 21 percent increase in landing weight will require a 10 percent increase in landing speed to support the greater weight.

When minimum landing distances are considered, braking friction forces predominate during the landing roll and, for the majority of airplane configurations, braking friction is the main source of deceleration.

The minimum landing distance will vary in direct proportion to the gross weight. For example, a 10 percent increase in gross weight at landing would cause:

1. a 5 percent increase in landing velocity,

2. a 10 percent increase in landing distance.

A contingency of this is the relationship between weight and braking friction force.

The effect of *wind* on landing distance is large and deserves proper consideration when predicting landing distance. Since the airplane will land at a particular airspeed independent of the wind, the principal effect of wind on landing distance is due to the change in the groundspeed at which the airplane touches down. The effect of wind on deceleration during the landing is identical to the effect on acceleration during the takeoff.

A headwind which is 10 percent of the landing airspeed will reduce the landing distance approximately 19 percent but a tailwind which is 10 percent of the landing speed will increase the landing distance approximately 21 percent. Figure 3-61 illustrates this general effect.

The effect of *pressure altitude* and *ambient temperature* is to define *density altitude* and its effect on landing performance. An increase in density altitude will increase the landing speed but will not alter the net retarding force. Thus, the airplane at altitude will land at the same indicated airspeed—as at sea level but, because of the reduced density, the true airspeed (TAS) will be greater. Since the airplane lands at altitude with the same weight and dynamic pressure, the drag and braking friction throughout the landing roll have the same values as at sea level. As long as the condition is within the capability of the brakes, the net retarding force is unchanged and the deceleration is the same as with the landing at sea level. Since an increase in altitude does not alter deceleration, the effect of density altitude on landing distance would actually be due to the greater TAS (true airspeed).

The minimum landing distance at 5,000 feet would be 16 percent greater than the minimum landing distance at sea level. The approximate increase in landing distance with altitude is approximately 3 1/2 percent for each 1,000 feet of altitude. Proper accounting of density altitude is necessary to accurately predict landing distance.

The effect of proper *landing speed is* important when runway lengths and landing distances are critical. The landing speeds specified in the airplane's flight handbook are generally the minimum safe speeds at which the airplane can be landed. Any attempt to land at below the specified speed may mean that the airplane may stall, be difficult to control, or develop high rates of descent. On the other hand, an excessive speed at landing may improve the controllability slightly (especially in crosswinds), but will cause an undesirable increase in landing distance.

Thus, a 10 percent excess landing speed would cause a 21 percent increase in landing distance. The excess speed places a greater working load on the brakes because of the additional kinetic energy to be dissipated. Also, the additional speed causes increased drag and lift in the normal ground attitude and the increased lift will reduce the normal force on the braking surfaces. The deceleration during this range of speed immediately after touchdown may suffer and it will be more likely that a tire can be blown out from braking at this point.

The most critical conditions of landing performance are the result of some combination of high gross weight, high density altitude, and unfavorable wind. These conditions produce the greatest landing distance and provide critical levels of energy dissipation required of the brakes. In all cases, it is necessary to make an accurate prediction of minimum landing distance to compare with the available runway. A polished, professional landing technique is necessary because the landing phase of flight accounts for more pilot-caused airplane accidents than any other single phase of flight.

In the prediction of minimum landing distance from the handbook data, the following considerations must be given:

1. Pressure altitude and temperature—to define the effect of density altitude.

2. Gross weight—which defines the CAS or EAS for landing.

3. Wind—a large effect due to wind or wind component along the runway.

4. Runway slope and condition—relatively small correction for ordinary values of runway slope, but a significant effect of snow, ice, or soft ground.

Since the characteristics of the atmosphere in which the airplane operates has a predominant effect on performance, we should first review some of the dominating factors—pressure and temperature.

Atmospheric Pressure

Atmospheric pressure is the force exerted by the weight of the atmosphere above a unit area. The weight of the air exerts a force on the earth or on any object placed in the atmosphere. *Pressure* is a measure of this *force per unit area:* i.e., grams-per-square centimeter, pounds-per-square foot, etc.. Pressure at any point on the earth's surface is the weight per unit area of the column of air above that point. In other words, if the pressure is 14 pounds-per-square inch, a column of air having one square inch cross section extending to the top of the atmosphere weighs 14 pounds. (The average pressure at the surface of the earth is approximately 14.7 pounds per square inch.)

We have mentioned that at any point in the atmosphere, pressure is the weight of the air *above* that point. Since pressure is the weight of the air above, and since less and less air lies above a point as it moves upward through the atmosphere, pressure must decrease with increasing altitude. The greater pressure at low altitude compresses the air more than does the lesser pressure at higher altitude. Therefore, the rate of decrease (lapse rate) in pressure with height becomes less with increasing altitude. For example, from sea level to 1,000 feet, pressure drops about one inch of mercury; but from 19,000 to 20,000 feet, pressure drops only about six-tenths of an inch.

The rate of decrease of pressure with height, however, is not always constant. Like most substances, air contracts as it cools and expands as it becomes warmer. Therefore, when a sample of air cools, it occupies less space; when heated, it occupies more. As a result, the rate of pressure decrease with height in cold air is greater than in warm air.

Since air is a gas, it may be compressed or permitted to expand. When air is compressed, a given volume contains more air, hence its density, or weight, is increased. Conversely, when air is permitted to expand, a given volume contains less air, thus its density, or weight, is decreased.

Heat is a property of all matter. From early studies of science, we learned that heat is the motion of molecules. *Heat* is then defined as *the total energy of motion of molecules*. We also learned that dense air has more molecules than less dense air. The two might have the same average motion, and thus have the same temperature, but the total energy, and consequently the degree of heat is greater in the dense air with more molecules. We cannot measure heat directly, but we can measure temperature with the thermometer.

A general gas law defines the relationship of pressure, temperature, and density when there is no change of state or heat transfer. Simply stated, this would be "density varies directly with pressure, inversely with temperature." Consequently, the higher we fly the less dense the air becomes due to less pressure and temperature.

Atmospheric pressure is continually changing. It varies with both time and location. These pressure changes are caused primarily by changes in the air density (weight of air per unit volume) produced by variations in the distribution of temperature.

Standard Atmosphere

In order to provide a common denominator for comparison of the performance of various aircraft, a *standard* atmosphere has been adopted. The set of standard conditions presently used in the U.S. is known as the *International Standard Atmosphere* (ISA) and has been adopted by most of the nations and airlines of the world—the

International Civil Aviation Organization.

The standard atmosphere actually represents the mean or average properties of the atmosphere; that is, it represents the year-round average of the pressure-height temperature soundings observed over a period of years.

In the standard atmosphere, sea level pressure is 29.92" Hg and 15° C. (59° F.); the standard lapse rates (decrease) for pressure are approximately 1" Hg per 1,000 feet increase in altitude and 2° C. (3.5° F.) per 1,000 feet increase (up to the tropopause). Figure 3-62 illustrates the variation of the most important properties of the air throughout the standard atmosphere.

Since all airplane performance is compared and evaluated in the environment of the standard atmosphere, all of the airplane performance instrumentation is calibrated for the standard atmosphere. Thus, certain corrections must apply to the instrumentation, as well as the airplane performance, if the actual operating conditions do not fit the standard atmosphere. In order to account properly for the nonstandard atmosphere certain related terms must be defined.

STANDARD ATMOSPHERE

Altitude (ft)	Pressure (in. Hg)	Temp. (°C.)	Temp. (°F.)	Density-slugs per cubic foot
0	29.92	15.0	59.0	.002378
1,000	28.86	13.0	55.4	.002309
2,000	27.82	11.0	51.9	.002242
3,000	26.82	9.1	48.3	.002176
4,000	25.84	7.1	44.7	.002112
5,000	24.89	5.1	41.2	.002049
6,000	23.98	3.1	37.6	.001988
7,000	23.09	1.1	34.0	.001928
8,000	22.22	-0.9	30.5	.001869
9,000	21.38	-2.8	26.9	.001812
10,000	20.57	-4.8	23.3	.001756
11,000	19.79	-6.8	19.8	.001701
12,000	19.02	-8.8	16.2	.001648
13,000	18.29	-10.8	12.6	.001596
14,000	17.57	-12.7	9.1	.001545
15,000	16.88	-14.7	5.5	.001496
16,000	16.21	-16.7	1.9	.001448
17,000	15.56	-18.7	-1.6	.001401
18,000	14.94	-20.7	-5.2	.001355
19,000	14.33	-22.6	-8.8	.001310
20,000	13.74	-24.6	-12.3	.001267

Figure 3-62 Properties of Standard Atmosphere

The pressure altitude can be determined by either of two methods: (l) by setting the barometric scale of the altimeter to 29.92 and reading the indicated altitude, or (2) by applying a correction factor to the elevation according to the reported "altimeter setting".

Pressure Altitude

Pressure altitude is the altitude in the standard atmosphere corresponding to a particular pressure level. The airplane altimeter is essentially a sensitive barometer calibrated to indicate altitude in the standard atmosphere. If the altimeter is set for 29.92" Hg (Standard Datum Plane) the altitude indicated is the pressure altitude—the altitude in the standard atmosphere corresponding to the sensed pressure.

The Standard Datum Plane is a theoretical level where the weight of the atmosphere is 29.92" of mercury as measured by a barometer. As atmospheric pressure changes, the Standard Datum Plane may be below, at, or above sea level. Pressure altitude is important as a basis for determining aircraft performance as well as for assigning flight levels to aircraft operating at high altitude (above 18,000 feet).

Density Altitude

The more appropriate term for correlating aerodynamic performance in the nonstandard atmosphere is density altitude—the altitude in the standard atmosphere corresponding to a particular value of air density.

Density altitude is pressure altitude corrected for nonstandard temperature. Under standard atmospheric conditions, air at each level in the atmosphere has a specific density, and under standard conditions, pressure altitude and density altitude identify the same level. Density altitude, then, is the vertical distance above sea level in the standard atmosphere at which a given density is to be found.

Since density varies directly with pressure, and inversely with temperature, a given pressure altitude may exist for a wide range of temperature by allowing the density to vary. However, a known density occurs for any one temperature and pressure altitude. The density of the air, of course, has a pronounced effect on airplane and engine performance. Regardless of the actual altitude at which the airplane is operating, its performance will be as though it were operating at an altitude equal to the existing density altitude.

For example, when set at 29.92" the altimeter may indicate a pressure altitude of 5,000 feet. According to the airplane flight handbook, the ground run on takeoff may require a distance of 790 feet under standard temperature conditions. However, if the temperature is 20° C. above standard, the expansion of air raises the density level. Using temperature correction data from tables or graphs, or by deriving the density altitude with a computer, it may be found that the density level is above 7,000 feet, and the ground run may be closer to 1,000 feet.

The computation of density altitude must involve consideration of pressure (pressure altitude) and temperature. Since airplane performance data at any level is based upon air density under standard day conditions, such performance data apply to air density levels that may not be identical with altimeter indications. Under conditions higher or lower than standard, these levels cannot be determined directly from the altimeter.

Density altitude can be computed by applying the pressure altitude and outside air temperature at flight level to a navigation computer. Density altitude can also be determined by referring to the table and chart as shown in Fig. 3-64 and Fig. 3-63.

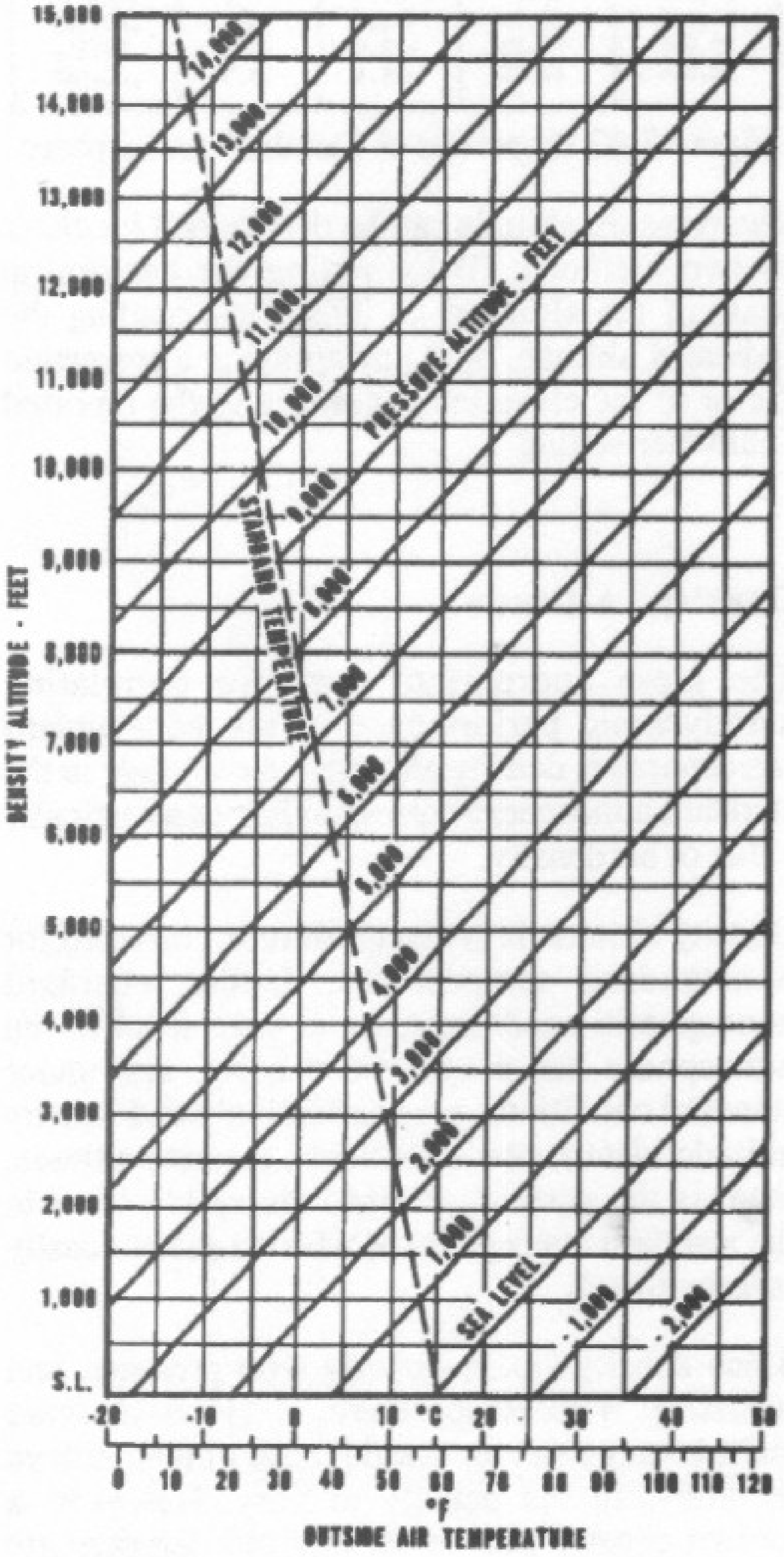
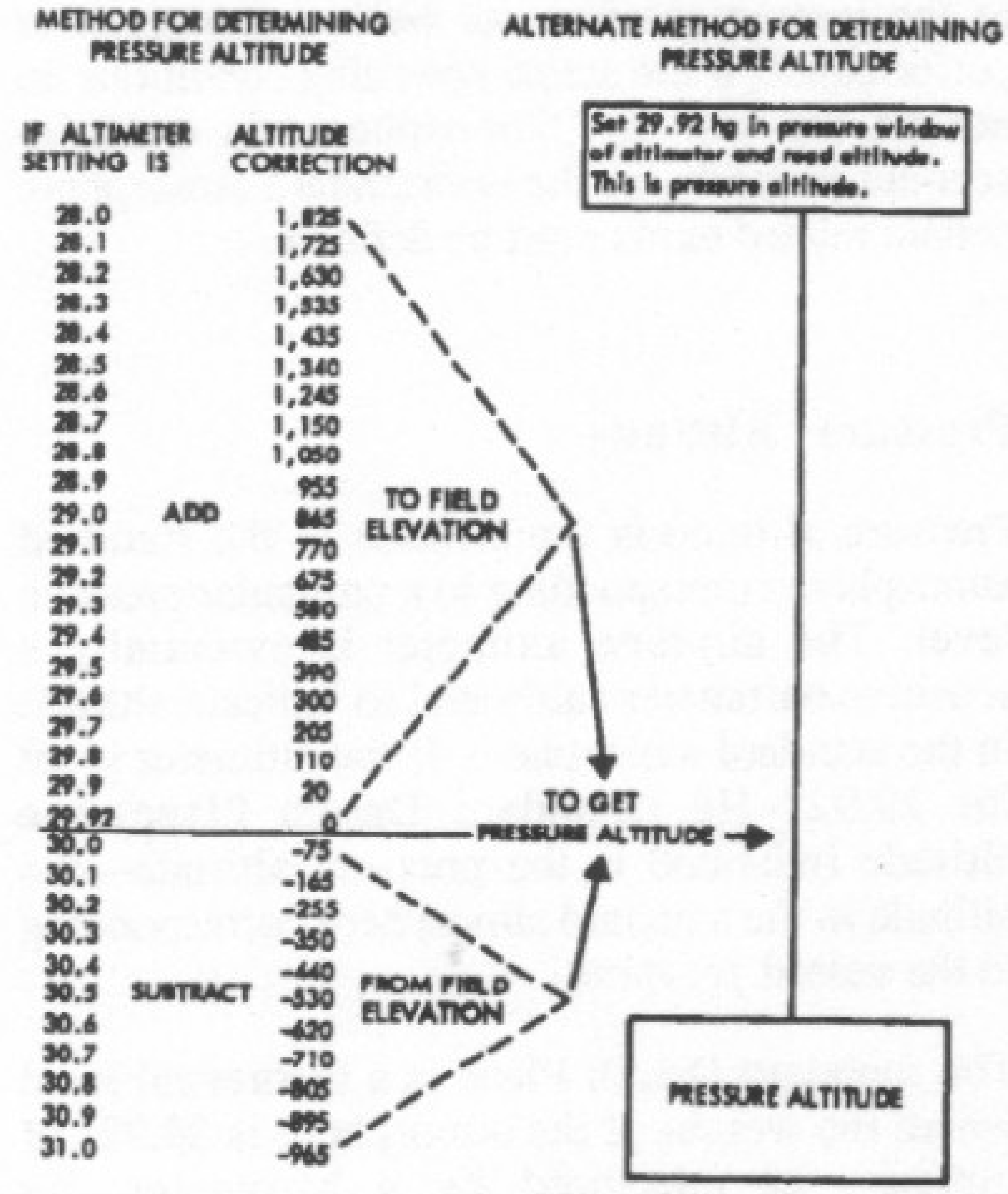


Figure 3-63 Density Altitude Chart



IF ALTIMETER SETTING IS	ALTITUDE CORRECTION
28.0	1,825
28.1	1,725
28.2	1,630
28.3	1,535
28.4	1,435
28.5	1,340
28.6	1,245
28.7	1,150
28.8	1,050
28.9	955
29.0	865
29.1	770
29.2	675
29.3	580
29.4	485
29.5	390
29.6	300
29.7	205
29.8	110
29.9	20
29.92	0
30.0	-75
30.1	-165
30.2	-255
30.3	-350
30.4	-440
30.5	-530
30.6	-620
30.7	-710
30.8	-805
30.9	-895
31.0	-965

Figure 3-64 Field Elevation Versus Pressure Altitude

Airplane Performance Speeds

True Airspeed (TAS)—the speed of the airplane in relation to the air mass in which it is flying.

Indicated Airspeed (IAS)—the speed of the airplane as observed on the airspeed indicator. It is the airspeed without correction for indicator, position (or installation), or compressibility errors.

Calibrated Airspeed (CAS)—the airspeed indicator reading corrected for position (or installation), and instrument errors. (CAS is equal to TAS at sea level in standard atmosphere.) The color coding for various design speeds marked on airspeed indicators may be IAS or CAS.

Equivalent Airspeed (EAS)—the airspeed indicator reading corrected for position (or installation), or instrument error, and for adiabatic compressible flow for the particular altitude. (EAS is equal to CAS at sea level in standard atmosphere.)

VSO—the calibrated power—off stalling speed or the minimum steady flight speed at which the airplane is controllable in the landing configuration.

VS1—The calibrated power—off stalling speed or the minimum steady flight speed at which the airplane is controllable in a specified configuration.

VY—the calibrated airspeed at which the airplane will obtain the maximum increase in altitude per unit of time. This best rate-of-climb speed normally decreases slightly with altitude.

VX—the calibrated airspeed at which the airplane will obtain the highest altitude in a given horizontal distance. This best angle-of-climb speed normally increases slightly with altitude.

VLE—the maximum calibrated airspeed at which the airplane can be safely flown with the landing gear extended. This is a problem involving stability and controllability.

VLO—the maximum calibrated airspeed at which the landing gear can be safely extended or retracted. This is a problem involving the air loads imposed on the operating mechanism during extension or retraction of the gear.

VFE—the highest calibrated airspeed permissible with the wing flaps in a prescribed extended position. This is a problem involving the air loads imposed on the structure of the flaps.

VA—the calibrated design maneuvering airspeed. This is the maximum speed at which the limit load can be imposed (either by gusts or full deflection of the control surfaces) without causing structural damage.

VNO—the maximum calibrated airspeed for normal operation or the maximum structural cruising speed. This is the speed at which exceeding the limit load factor may cause permanent deformation of the airplane structure.

VNE—the calibrated airspeed which should NEVER be exceeded. If flight is attempted above this speed, structural damage or structural *failure* may result.