#317

JETLINER

TYPE C-102
1ST PROTOTYPE ONLY

Flight Manual

COMPILED TO
C.A.A. REQUIREMENTS
AND ISSUED BY
SALES AND SERVICE
DIVISION

AVRO

A. V. ROE CANADA LIMITED MALTON ONTARIO

AVRO C-102

FLIGHT MANUAL

FOR

FIRST PROTOTYPE

JETLINER

CF-EJD-X

EQUIPPED WITH

ROLLS ROYCE DERWENT
TYPE 5/17 ENGINES

A COPY OF THIS MANUAL MUST BE CARRIED IN THE AIRCRAFT AT ALL TIMES. IT SHOULD BE LOCATED IN A POSITION CONVENIENT TO ALL CREW MEMBERS

THE AIRCRAFT MUST BE OPERATED IN ACCORDANCE WITH THE OPERATING LIMITATIONS AND PROCEDURES AS OUTLINED IN SECTIONS 2 AND 3 OF THIS MANUAL.

APPROVED.....

D. O. T./OTTAWA

DATE.....

FOREWORD

This manual is produced to Department of Transport and CAA requirements and is intended for the use of flight and ground crews.

The manual contains a complete general description of the aircraft and in addition, provides the necessary handling data and information on Operating Limitations and Procedures.

Information supplied covers a detailed description of all functional services, including recommended operating procedures, their relative control and management.

LIST OF AMENDMENTS

A. L. No.	AMENDED BY	DATE	A. L. No.	AMENDED BY	DATE

LEADING PARTICULARS

GENERAL DESCRIPTION:

The aircraft is a four engined, all metal, low wing monoplane of conventional stressed skin construction. The wing and tail surfaces are fully cantilever and a tricycle landing gear is fitted. The cabin is sealed for pressurization and air conditioning.

BASIC TYPE:

High speed, high altitude, commercial passenger and cargo transport.

CREW:

Pilot, co-pilot and one attendant.

PASSENGERS:

Provision is made for accommodating a passenger load of 36 persons.

MAXIMUM FUEL CAPACITY:

MAIN TANKS	2353 Imp. Gals.
AUX. FUSELAGE TANKS	1044 Imp. Gals.

PERFORMANCE:

GROSS TAKE-OFF WEIGHT	65,000 Lbs.
GROSS LANDING WEIGHT	55,000 Lbs.
AVERAGE CRUISING SPEED AT 30,000 FEET	420 mph.

BALANCE: (UNTIL CERTIFIED)

FORWARD C OF G LIMIT	15% S.M.C.
AFT C OF G LIMIT	25% S.M.C.

PLACARD SPEEDS: 35,000 - 65000 Lbs.Wt.

FLAPS FULLY DOWN	176 m.p.h.
FLAPS 30° DOWN	250 m.p.h.
LANDING GEAR DOWN	200 m.p.h.
FLAPS AND LANDING GEAR UP	Refer to Fig.101

FUSELAGE:

LENGTH	82 ft. 5 ins.
HEIGHT	14 ft.3.6 ins.
DIAMETER	10 ft.
NET VOLUME HEIGHT	7 ft. 10 ins.
CABIN CEILING HEIGHT	6 ft.10.2 ins.
PASSENGER DOOR SILL HEIGHT	7 ft. 11 ins.

WING GROUP:

WING AREA	1,157 sq.ft.
SPAN (OVERALL)	98 ft. 1 in.
ROOT CHORD	14.25 ft.
TIP CHORD	7.11 ft.
S.M.C.	11.78 ft.

TAPER RATIO	2:1
ASPECT RATIO	8.3
INCIDENCE	2.5°
DIHEDRAL	6°
AIRFOIL -ROOT	N.A.C.A. 23016.5
TIP	N.A.C.A. 23012

FLAPS:

INBOARD FLAPS:

AREA - BOTH	22.3 sq.ft.
MOVEMENT	60°DOWN

OUTBOARD FLAPS:

AREA - BOTH	81.616 sq.ft.
MOVEMENT	60°DOWN

AILERONS:

AREA (AFT OF HINGE LINE) BOTH	52.6 sq.ft.
MOVEMENT	17° ± 1° UP 17° ± 1° DOWN

TAIL GROUP:

TAILPLANE:

SPAN	37 ft.
TOTAL AREA (INCLUDING FIXED POWER ELEVATORS)	202.5 sq.ft.
AIRFOIL	N.A.C.A. 0012.8
DIHEDRAL	0°
INCIDENCE (RELATIVE TO FUSELAGE DATUM)	2.5°
S.M.C.	6.48 ft.
ASPECT RATIO	5.81

ELEVATORS:

AREA (AFT OF HINGE LINE) BOTH	37.2 sq.ft.
MOVEMENT	30° ± 1° UP 20° ± 1° DOWN

VERTICAL TAIL SURFACES:

TOTAL AREA (INCLUDING DORSAL & RUDDER)	199.7 sq.ft.
AIRFOIL	N.A.C.A. 0012.8

FIN AND DORSAL:

AREA (INCLUDING FIXED POWER RUDDER)	182.87 sq.ft.

RUDDER:

AREA (AFT OF HINGE LINE)	16.86 sq.ft.
MOVEMENT	24° ±1°LEFT 24° ±1°RIGHT

LANDING GEAR:

TRACK - NOMINAL	22 ft. 6 ins.
WHEELBASE	31 ft. 9.5 ins.

ENGINE:

NAME	Rolls Royce Derwent Mk. 5/17
TYPE	Centrifugal flow, turbo-jet with a double entry, single stage centrifugal compressor, 9 combustion chambers and a single stage axial flow turbine.

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CHAPTER 1

GENERAL DESCRIPTION

1-1 GENERAL:

(1) The Avro Jetliner is a four engined short-to-medium range, passenger transport aircraft with a pressurized and air conditioned fuselage. Provision can be made to carry a maximum of 36 passengers.

(2) The aircraft is a low wing monoplane of all metal stressed skin construction. The four Rolls-Royce Derwent 5/17 turbo-jet engines are housed in pairs in two underslung wing nacelles carrying the main wheels of the tricycle landing gear. The stabilizing surfaces consist of conventional single vertical fin and rudder with tailplane and elevators mounted high on the fin.

1-2 FUSELAGE:

(1) The fuselage is of aluminum alloy semi-monocoque construction incorporating circular transverse formers giving a cross-section 10 feet in diameter. The formers are spaced by longitudinal stiffeners and the complete structure has a stressed skin covering designed to withstand pressurization.

(2) The fuselage is divided into four principal sections, comprising nose, centre, rear centre and rear sections. The centre wing is an integral part of the fuselage centre section while the fuselage rear section is included in the empennage structure.

(3) In order to achieve the highest strength for the fuselage formers, the stringers are attached to the outside former flanges. This arrangement eliminates cut-outs in the formers and preserves strength and continuity.

(4) Butt joints have been used at all transverse skin joints to reduce drag and aerodynamic noise. All skin sheets are pre-formed and stretched, and all external riveting is flush. The fuselage is sealed by special compounds applied between the faying surfaces and skin joints.

(5) The forward pressure dome is located immediately forward of the main instrument panel and forms the forward wall of the flight compartment, isolating the nose landing gear compartment from the pressurized region. The rear pressure dome is clamped between the angles of the rear centre section and rear section transport joints.

(6) All floor sections are of robust construction to withstand indentation and possible damage from freight. They are secured to the structure by screws. The centre floor panels between the seats are secured by Dzus fasteners which are easily removable for servicing purposes.

(7) Due to the provision of only four escape hatches in this aircraft, the C.A.A. regulations limit the maximum number of passengers which may be carried

to 36. Subsequent aircraft however, will be provided with five escape hatches to permit a greater passenger load. The hatches open inwards and are quickly removable from inside or outside the cabin. An additional hatch is fitted to the roof of the flight compartment for crew escape.

(8) The windscreen structure is a high strength aluminum alloy casting incorporating three centre panels of laminated glass construction. The vinyl core of the centre panels ensures extra protection in the cabin in the event of windscreen damage. The centre panels incorporate the NESA system of electrical de-icing. Each of the two direct vision nose windows and the two aft nose windows are of Plexiglas. Non-misting circular double windows are provided in the passenger cabin.

1-3 WING:

(1) The wing is of conventional stressed skin construction and consists of the centre wing (which is integral with the fuselage centre section) and right and left outer wings with detachable wing tips.

(2) Two heavily flanged, solid web spars form the basis of the complete wing structure. The spars carry the wing loads which are transmitted through the stressed skin and the truss-type ribs.

(3) The spars of each outer wing are attached to the centre wing by continuous butt-strap joints. Heavy gauge skin and stringers take the place of concentrated spar booms, and provide the high torsional stiffness necessary.

(4) The areas between the spars and ribs in the inner section of the outer wing form the integral fuel tanks. There are four tanks, two in each outer wing, the inboard tank extending between ribs 1 and 7, and the outboard tank between ribs 7 and 13.

(5) Plasticizers and synthetic resins are added to the thickol-based sealant which is used to close the joints and rivets of the integral wing tanks. The faying surfaces are not sealed. The finished tank inner surfaces are sprayed with a cyclohexanone solvent to bond the complete inner surface.

(6) The ailerons extend approximately half way along the outer wing span and are of all-metal construction. They are aerodynamically unbalanced, and power-assisted hydraulically. The ailerons are hinged to the upper surfaces of the outer wings by piano type hinges. This arrangement provides for a sealed upper wing surface. Internal mass balance weights are employed and manual trim tabs are fitted.

(7) Hydraulically operated, inboard split flaps on the centre wing extend from the wing root to the nacelle. Outboard flaps of the split type are fitted on each outer wing between the nacelle and the aileron.

1-4 NACELLES:

(1) Each pair of engines and jet pipes is housed in a single structure. This structure is built up as a unit and encloses the engine mountings which are attached to the front spar of the centre wing. The jet pipes are supported in spherical trunnions and swinging suspension links attached to the structure.

(2) The nacelle structure consists of transverse frames of pressed channel section, and channel section longitudinal intercostals.

(3) Each nacelle is divided into two main engine bays which are further divided by fireproof bulkheads into a number of zones or compartments (Fig. 32). The front compartment (Zone 1) constitutes the plenum chamber and extends from the air intake to the distributor casing of each engine. The rear compartment is subdivided into two zones, Zone 2 which envelopes the engine combustion chambers and turbine casings, and Zone 3 which contains the jet pipes and extends aft from the rear of the turbine casings to the jet pipe outlets. Zone 3 is lined with steel sheet to form insulating envelopes isolating the jet pipes from the nacelle structure.

(4) A compartment between the engine bays, and separated from them by fireproof bulkheads, contains the accessories gear box and hydraulic and electrical equipment.

1-5 EMPENNAGE:

(1) The empennage includes the fuselage rear section and lower fin, the upper fin, tail cone, rudder, tailplane and elevators.

(2) The lower fin structure consists principally of two fuselage formers which are spaced by stiffeners and web plates. These formers extend upwards to form the fin posts. The tailplane is mounted on the top of the lower fin, while a recess in the tailplane accommodates the upper fin. The complete assembly is secured by attachment bolts.

(3) The rudder extends to almost the full length of the fin. It is hinged to an intermediate section attached to the rear post of the fin structure. The rudder is aerodynamically unbalanced and is provided with a trim tab.

(4) Each elevator is hinged to an intermediate section attached to the rear of the tailplane rear spar. The elevators are internally mass balanced and are fitted with trim tabs.

NOTE

(1) The "intermediate sections" mentioned in the preceding paragraphs were originally designed as power operated sections of the rudder and

elevator assemblies. This arrangement is now eliminated, the sections being locked to the vertical fin and tailplane structure respectively by means of straps riveted in position. For all practical purposes the intermediate sections (or power rudder and elevator) now form an integral part of the vertical fin and tailplane assemblies.

(5) The tailcone is fabricated from three formers which extend to form the rudder fairing posts. The structure has a stressed skin covering and a semi-spherical transparent end cap.

1-6 LANDING GEAR:

(1) An electro-hydraulically operated tricycle landing gear, incorporating twin wheels on both main and nose units, is fitted to the aircraft. The steerable nosewheel of the nose unit is attached to a single liquid sprung shock strut which is self-centering and fully castoring and incorporates shimmy damping. The unit retracts forward into the nosewheel well in front of the forward pressure bulkhead. The main units are also single struts incorporating liquid springing and are pivoted from strengthened ribs at a point adjacent to the centre wing rear spar. These units are housed in wheel well frames and retract forward between the spars in their relative nacelles.

(2) A split type main unit door is hinged to either side of the wheel well frame in each nacelle. The individual door sections are actuated by a common hydraulic jack and, on retraction of the landing gear, close inwards. The nose unit door is also of the split type and is hinged to the sides of the nosewheel well frame. Both sections of the nosewheel door are closed on retraction by a trip mechanism fitted to the nose unit.

1-7 CREW ACCOMMODATION:

(1) This aircraft is operated by a crew consisting of pilot, co-pilot and one attendant. The pilot and co-pilot are accommodated in the flight compartment and are seated in the conventional side-by-side arrangement. An attendant's station is provided on the left hand side of the cabin, forward of the rear cabin bulkhead.

1-8 PASSENGER ACCOMMODATION:

(1) The aircraft will eventually be provided with seating for 36 passengers, also a buffet, coat racks, washroom and cargo space for operation on domestic routes. However, in the present state of trials, seating is provided for six passengers only. An Elsan chemical toilet is carried in the rear cargo compartment.

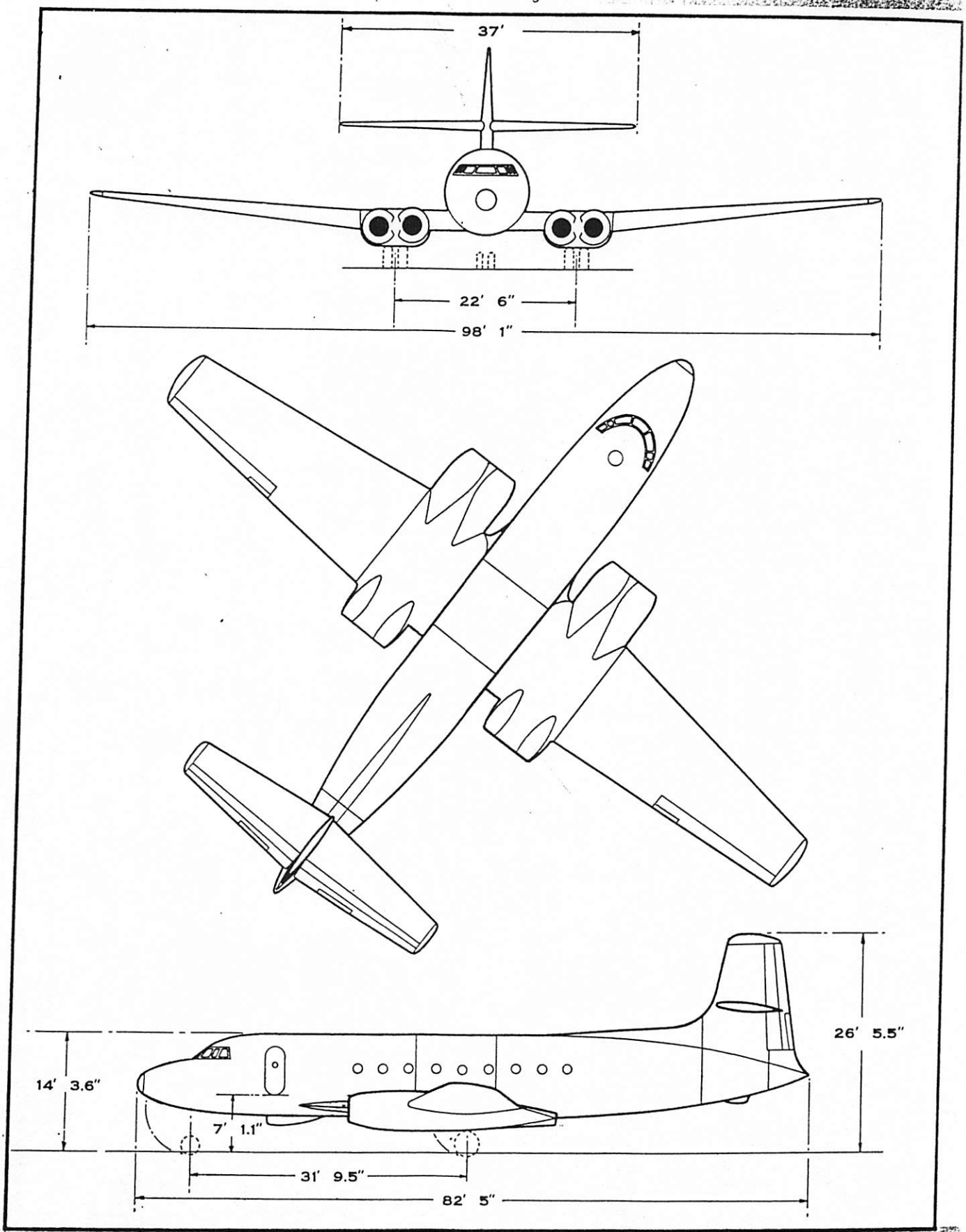
1-9 ENTRANCE DOORS

(1) Entrance doors are provided for access from the passenger cabin to the flight compartment and to the aft luggage compartment.

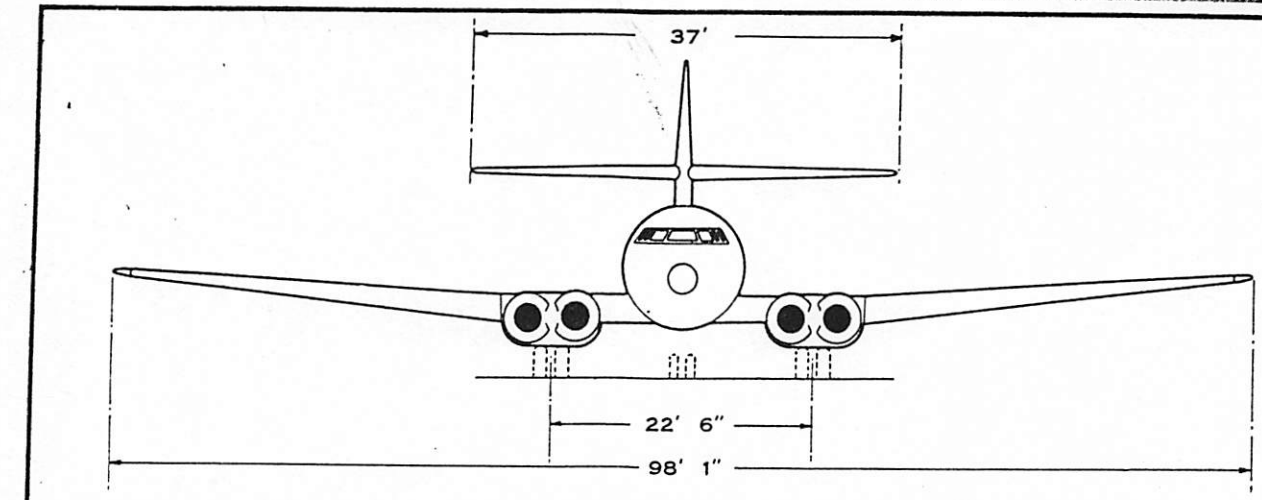
(2) All doors (with the exception of the main entrance door) are inside the fuselage in the pressurized area and are of light construction. The main entrance door is of robust construction and incorporates a rubber seal around its periphery.

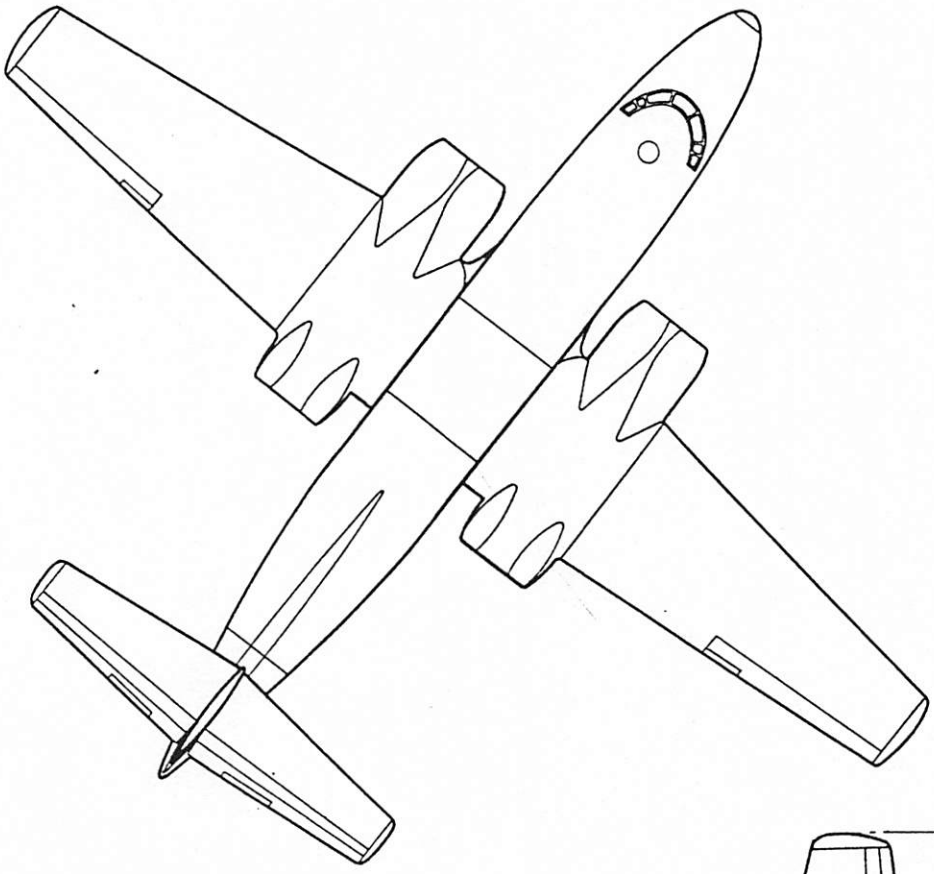
(3) Further entrances are provided for access to the radio, accessory and hydraulic compartments. Access to the radio and accessory compartments is gained through quick release panels. Entrance to the hydraulic compartment is through a conventional swinging door.

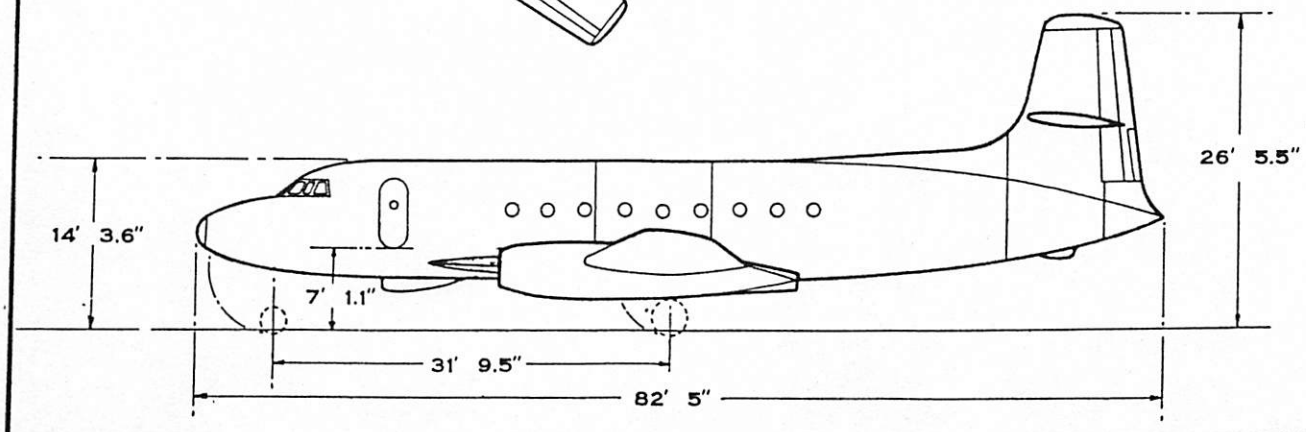
THREE VIEW
DIMENSIONAL G.A.
FIG. 1



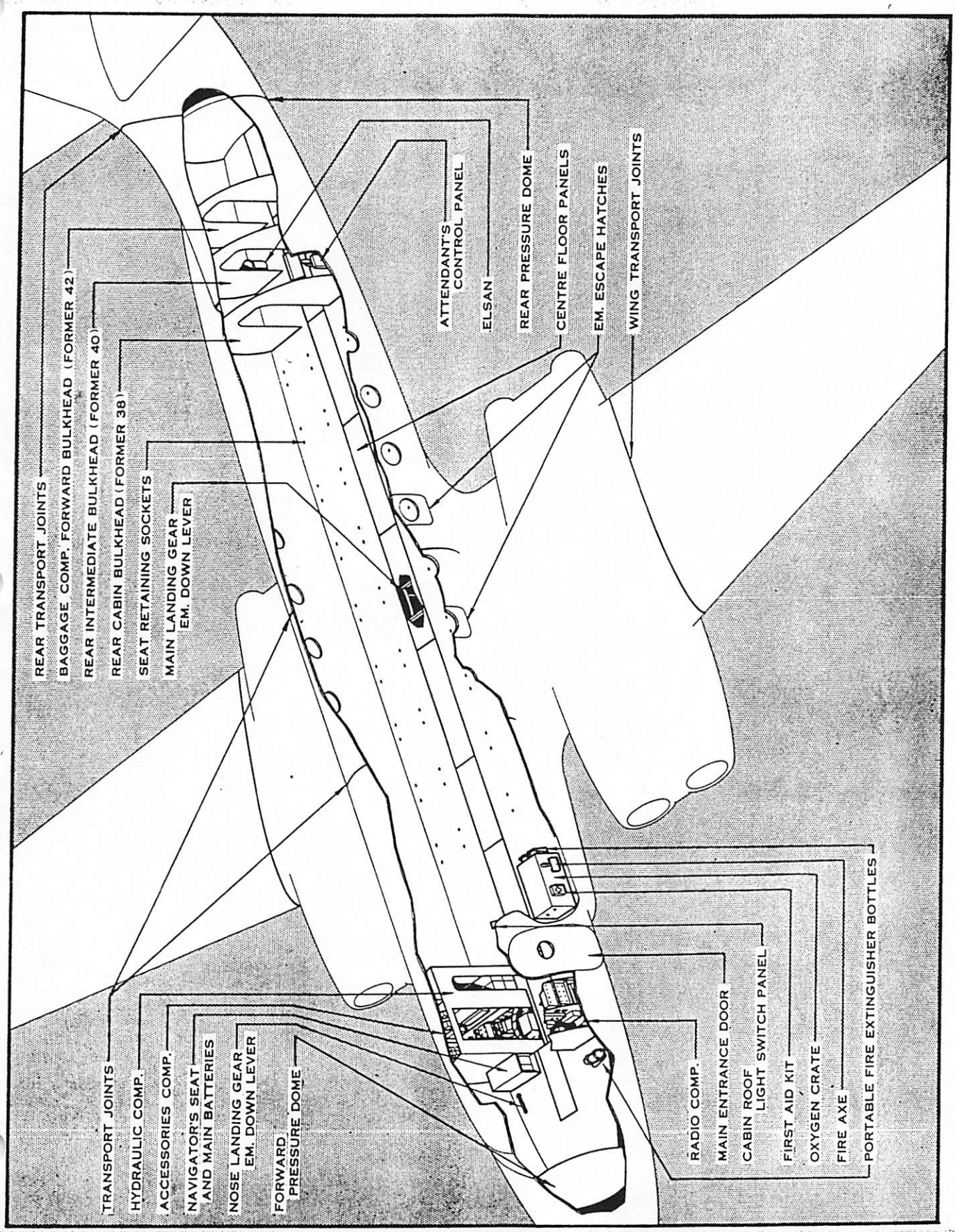
THREE VIEW
DIMENSIONAL G.A.
FIG. 1

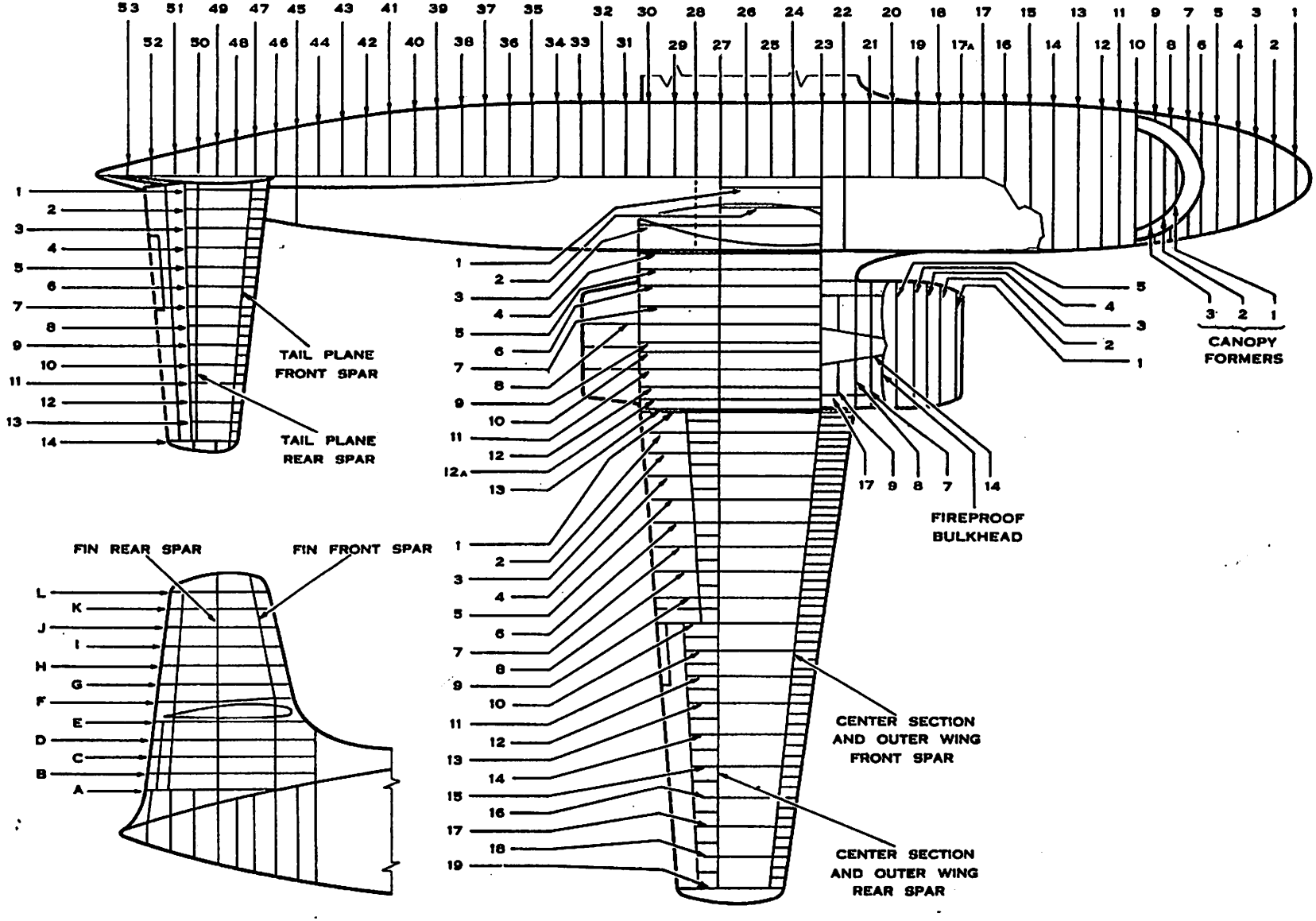




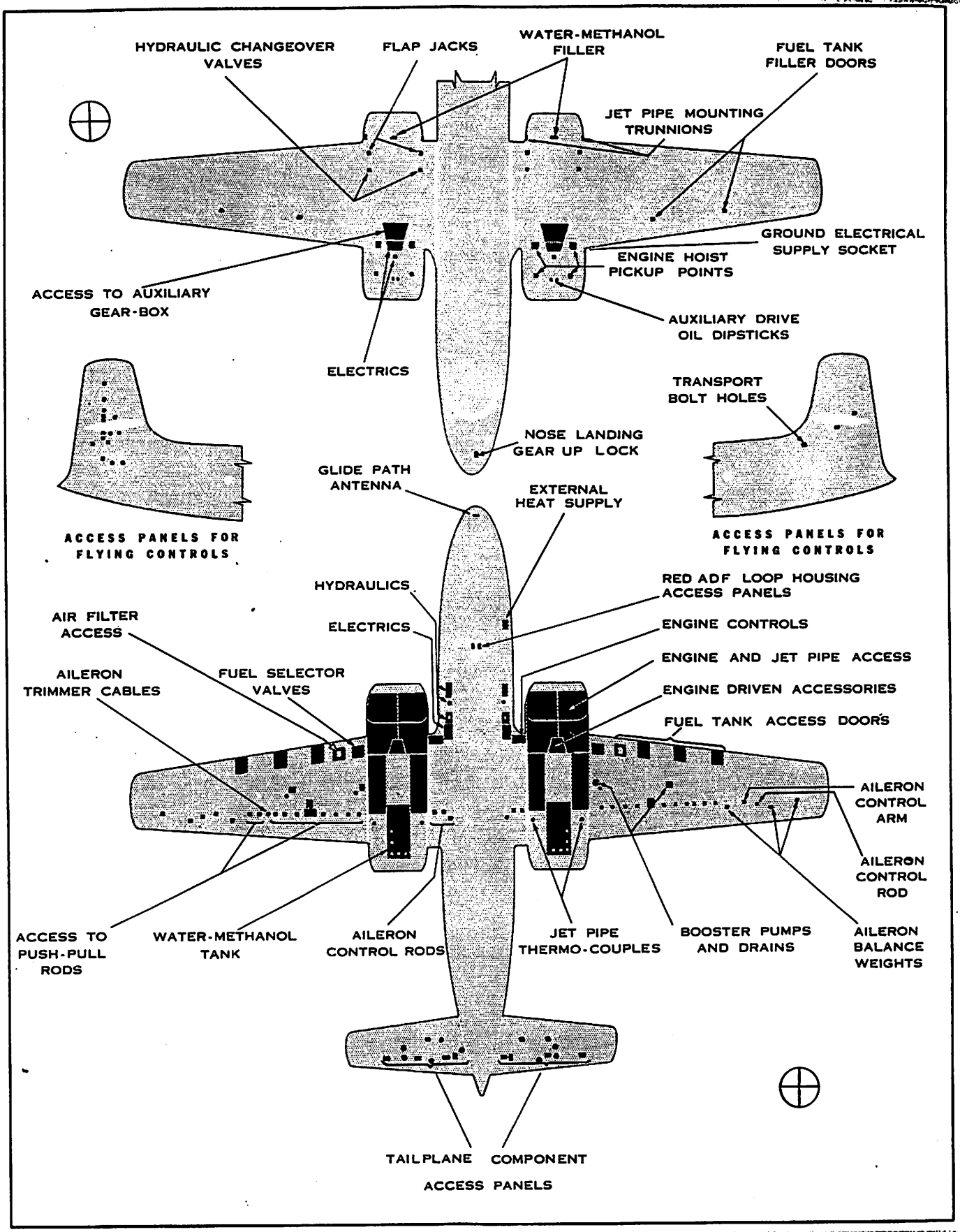


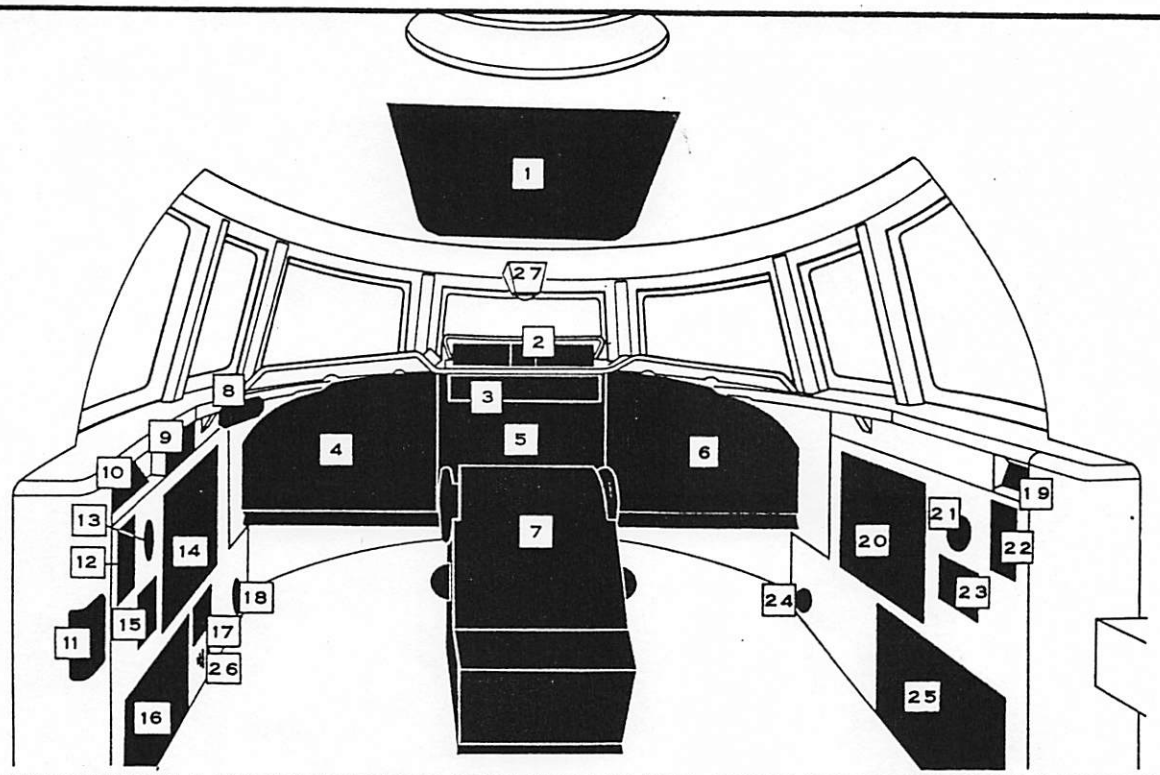
INTERIOR
G.A. OF AIRCRAFT
FIG. 2




53 51 49 47 45 43 41 39 37 35 32 30 28 26 24 22 20 18 17 15 13 11 9 7 5 3 1
52 50 48 46 44 42 40 38 36 34 33 31 29 27 25 23 21 19 17A 16 14 12 10 8 6 4 2
1 2 3 4 5 6 7 8 9 10 11 12 13 14
TAIL PLANE FRONT SPAR
TAIL PLANE REAR SPAR
1 2 3 4 5 6 7 8 9 10 11 12 12A 13
5 4 3 2 1
3 2 1
CANOPY FORMERS
17 9 8 7 14
FIREPROOF BULKHEAD
FIN REAR SPAR
FIN FRONT SPAR
L K J I H G F E D C B A
1 2 3 4 5 6 7 8 9 10 11 12 13 14 15 16 17 18 19
CENTER SECTION AND OUTER WING FRONT SPAR
CENTER SECTION AND OUTER WING REAR SPAR

ACCESS AND INSPECTION PANELS FIG. 4





FOR DETAILS OF PANELS AND COMPARTMENTS, INCLUDING REAR WALL PANELS, NOT SHOWN IN THIS ILLUSTRATION, SEE RELATIVE ILLUSTRATIONS INCLUDED IN THIS CHAPTER.

1 ROOF CONTROL PANEL
2 RADIO COMPASS LOOP CONTROL UNITS
3 FUEL SYSTEM BOOSTER PUMP CONTROL PANEL
4 PILOT'S FLIGHT INSTRUMENT PANEL
5 ENGINE INSTRUMENT PANEL
6 CO-PILOT'S FLIGHT INSTRUMENT PANEL
7 THROTTLE CONTROL PEDESTAL
8 ANTI-SKID BRAKE CONTROL AND INDICATOR PANEL
9 PRESSURIZATION SWITCH PANEL
10 PILOT'S JACK BOX
11 ACCESS TO OXYGEN BOTTLE CONTROL VALVE
12 PILOT'S HEADPHONE STOWAGE
13 OXYGEN REGULATOR
14 PRESSURIZATION CONTROL PANEL
15 OXYGEN MASK STOWAGE
16 HINGED MAP STOWAGE
17 GLOVE COMPARTMENT
18 CONTROLLABLE COLD AIR LOUVRE
19 CO-PILOT'S JACK BOX
20 AIR CONDITIONING CONTROL PANEL
21 OXYGEN REGULATOR
22 CO-PILOT'S HEADPHONE STOWAGE
23 CO-PILOT'S OXYGEN MASK STOWAGE
24 CONTROLLABLE COLD AIR LOUVRE
25 HINGED MAP STOWAGE
26 AUDIBLE WARNING DEVICES OUTLET (LANDING GEAR AND STALL WARNING)
27 MAGNETIC COMPASS

CHAPTER 2

CONTROLS AND EQUIPMENT AT CREW STATIONS

1-10 GENERAL:

(1) This chapter with its relative illustrations is intended as a general guide to the location of controls and equipment at crew stations. Functional details on the use of equipment and the necessary handling data and operating procedure may be found in the relevant sections within this publication.

1-11 INSTRUMENT PANELS AND CONTROLS:

(1) All control and instrument panels, with the exception of the attendant's control panels and a limited number of isolated component controls, are housed in the flight deck.

(2) The general configuration of the flight deck panel layout (Fig.5) is based on the main instrument panel, the throttle control pedestal, the roof control panel, electrical and radio power control panels, and the right and left side wall groups of auxiliary control panels.

(3) The main instrument panel is attached to the structure below the windshield and is divided into three sections as follows:-

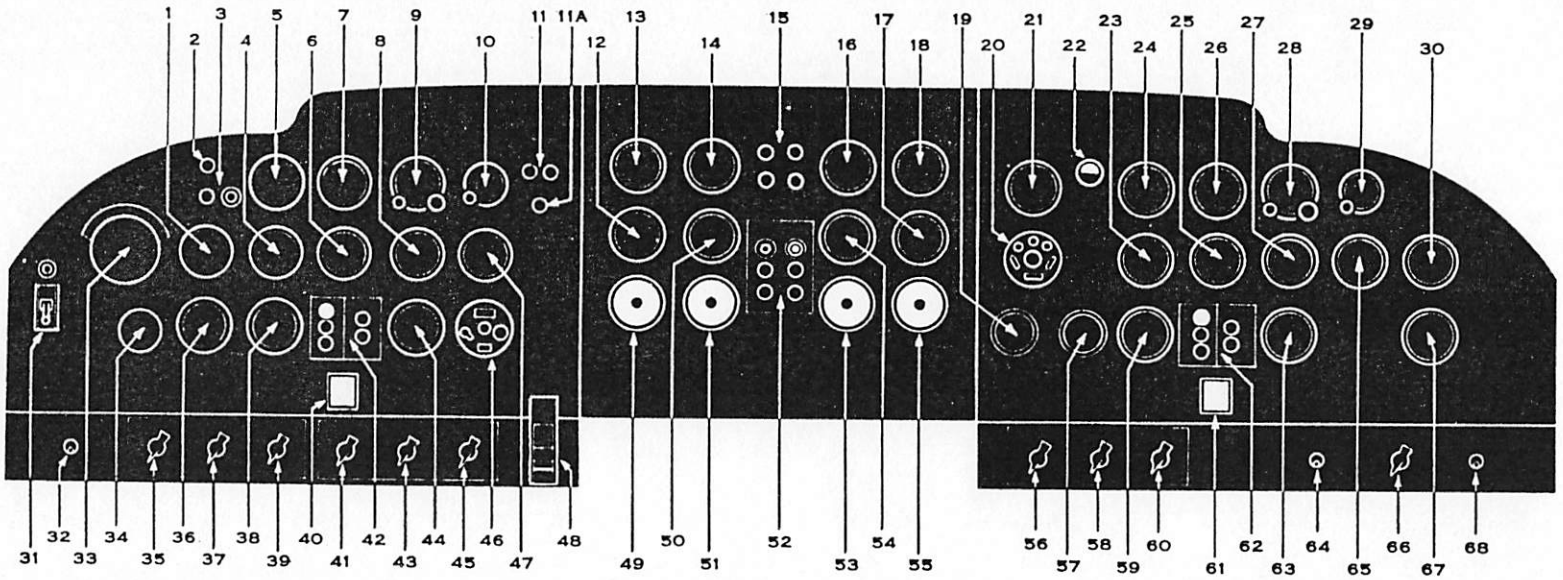
(a) Pilot's flight instrument panel (left side).
(b) Co-pilot's flight instrument panel (right side).
(c) Engine instrument panel (centrally disposed).

(4) In addition to the above, a centrally disposed auxiliary panel mounted directly in front of the engine instrument panel carries the booster pump control switches for the fuel tanks. A schematic diagram of the fuel system is engraved on this panel.

(5) Located immediately above the booster pump control panel, are two radio compass loop control units complete with switch panel. These are housed in an independent casing which is attached to the main instrument panel shroud.

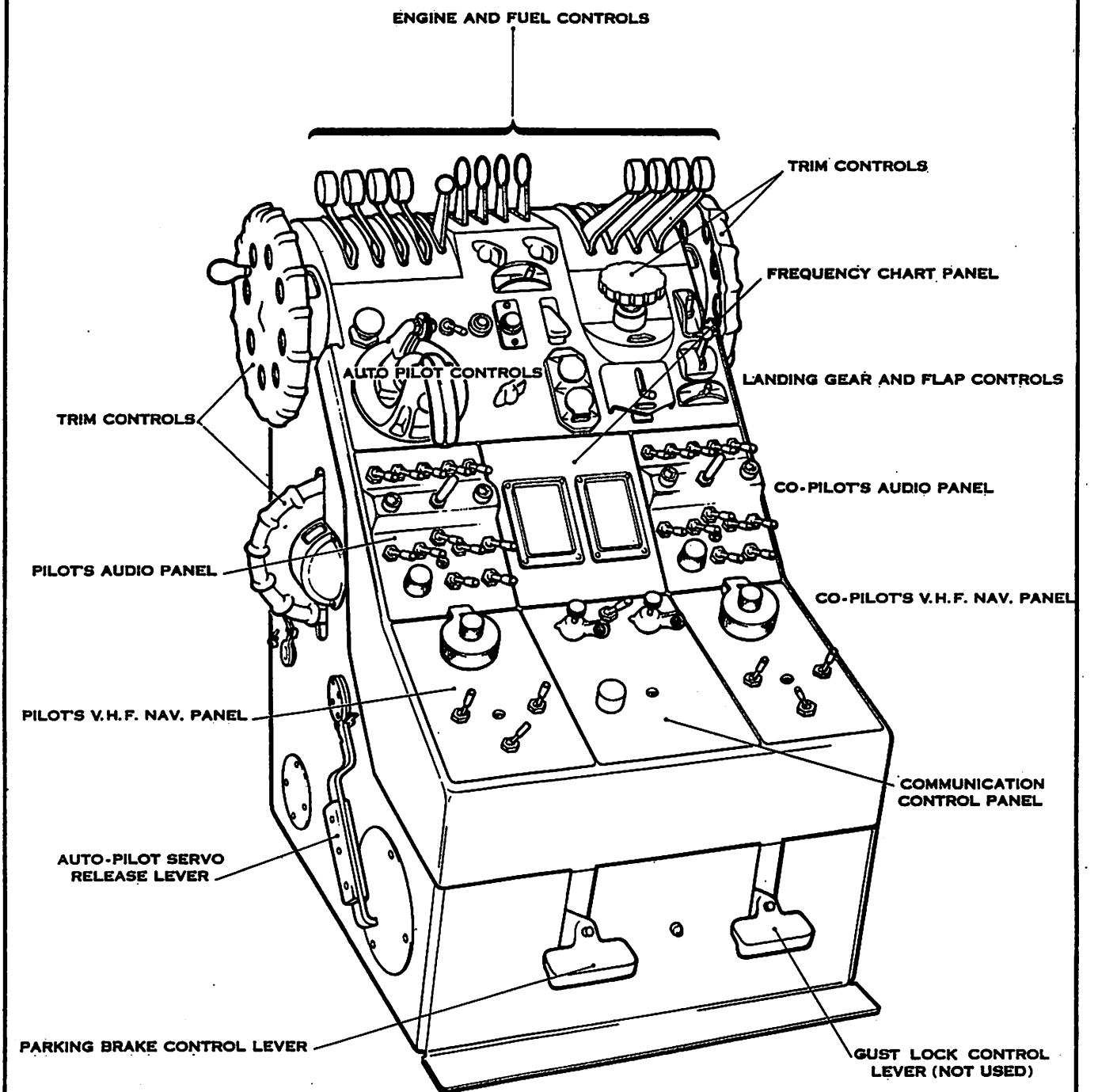
(6) The throttle control pedestal is attached to the floor of the flight deck at a point adjacent to the centrally disposed engine instrument panel, and carries the following equipment:

(a) Pilot's and Co-pilot's throttle control levers.
(b) Fuel cock controls and tank selectors.
(c) Auto-pilot controls.
(d) Landing gear controls.

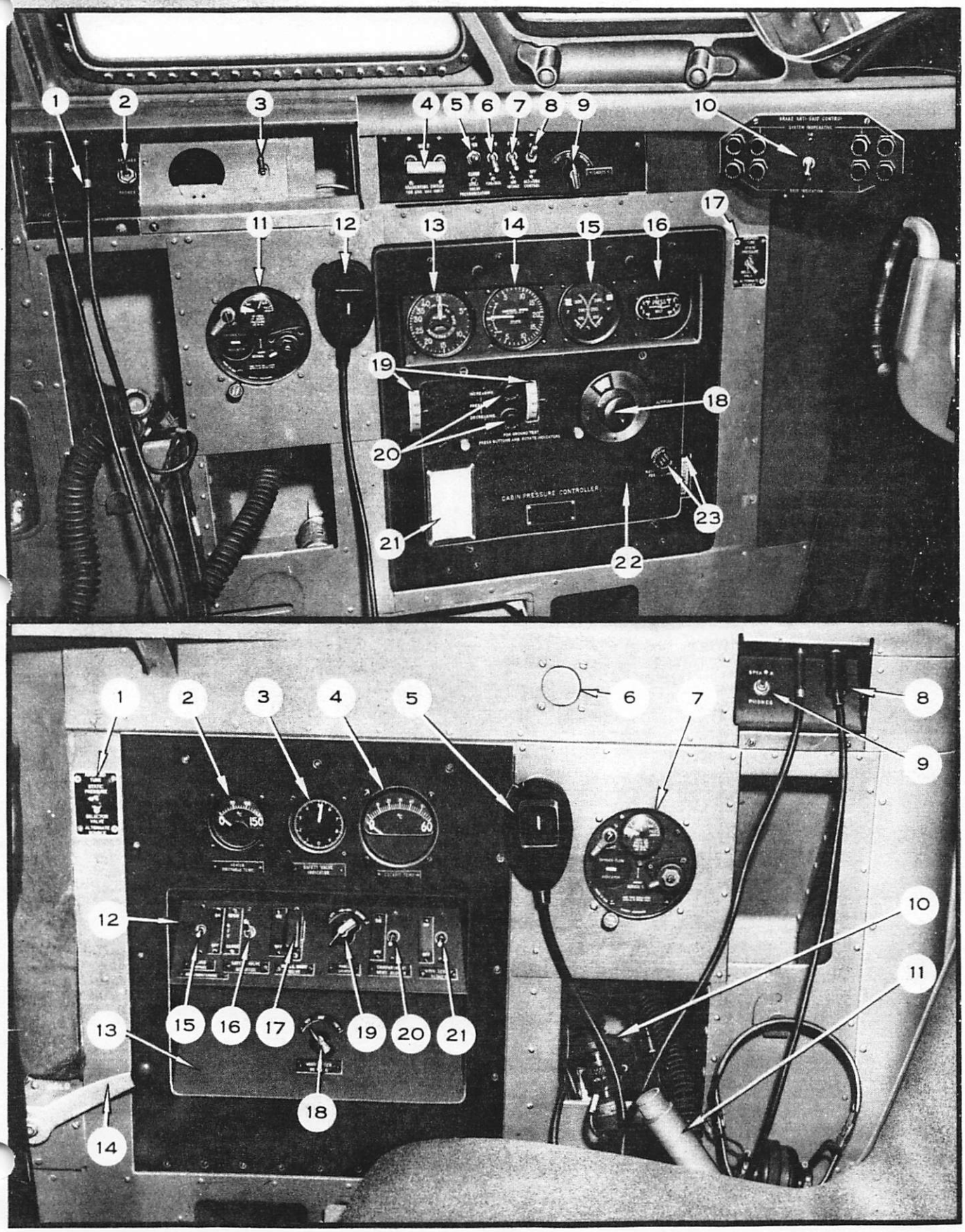


1 ACCELEROMETER
2 ATTENDANT CALL LIGHT
3 STALL WARNING LIGHT AND SWITCH
4 TURN AND BANK INDICATOR
5 ALTIMETER
6 FLUX GATE COMPASS INDICATOR
7 AIR SPEED INDICATOR
8 I. L. S. INDICATOR
9 GYRO HORIZON 10 CLOCK
11 LANDING GEAR DOOR INDICATORS
11A AILERON BOOST CUT-OUT INDICATOR
12 OIL PRESSURE GAUGE
13 R. P. M. INDICATOR
14 R. P. M. INDICATOR
15 LOW PRESSURE OIL WARNING LIGHTS
16 R. P. M. INDICATOR
17 OIL PRESSURE GAUGE
18 R. P. M. INDICATOR
19 HYDRAULIC BRAKE PRESSURE INDICATOR
20 LANDING GEAR INDICATOR
21 FLAP INDICATOR
22 INVERTER FAILURE INDICATOR
23 TURN AND BANK INDICATOR
24 ALTIMETER
25 GYROSYN COMPASS
26 AIR SPEED INDICATOR
27 I. L. S. INDICATOR
28 GYRO HORIZON
29 CLOCK
30 MACH INDICATOR
31 NOSE WHEEL STEERING SWITCH AND INDICATOR LIGHT
32 WINDSHIELD WIPER SWITCH
33 NOSE WHEEL STEERING CONTROL
34 (NOT IN USE)
35 LEFT PANEL LIGHT SWITCH (FLOOD)
36 ELEVATOR INDICATOR
37 LEFT PANEL LIGHT SWITCH (WHITE)
38 AUTO PILOT TRIM INDICATOR
39 LEFT PANEL LIGHT SWITCH (RED)
40 GYRO FLUX GATE COMPASS (P. B. 10) CORRECTOR CARD
41 CENTRE PANEL LIGHT SWITCH (FLOOD)
42 I. L. S. INDICATING LIGHTS
43 CENTRE PANEL LIGHT SWITCH (WHITE)
44 A. D. F. INDICATOR
45 CENTRE PANEL LIGHT SWITCH (RED)
46 OMNI BEARING SELECTOR
47 RATE OF CLIMB INDICATOR
48 POWER ELEVATOR CONTROL
49 FUEL CONTENTS GAUGE
50 JET PIPE TEMPERATURE INDICATOR
51 FUEL CONTENTS GAUGE
52 WATER METHANOL CONTROL PANEL
53 FUEL CONTENTS GAUGE
54 JET PIPE TEMPERATURE INDICATOR
55 FUEL CONTENTS GAUGE
56 RIGHT PANEL LIGHT SWITCH (FLOOD)
57 HYDRAULIC PRESSURE SYSTEM INDICATOR
58 RIGHT PANEL LIGHT SWITCH (WHITE)
59 OUTSIDE AIR TEMPERATURE INDICATOR
60 RIGHT PANEL LIGHT SWITCH (RED)
61 GYRO FLUX GATE COMPASS (P. B. 10) CORRECTOR CARD
62 I. L. S. INDICATING LIGHTS
63 A. D. F. INDICATOR
64 FLOOR LIGHT SWITCH
65 RATE OF CLIMB INDICATOR
66 MAGNETIC COMPASS SWITCH
67 CABIN PRESSURE INDICATOR
68 WINDSHIELD WIPER SWITCH

THROTTLE CONTROL
PEDESTAL
FIG. 7



PILOT'S & CO-PILOT'S
SIDE PANELS
FIG. 8



KEY TO FIG. 8 PILOT'S AND CO-PILOT'S SIDE PANELS

PILOT'S SIDE PANELS

1. Interphone Panel
2. Interphone Switch
3. Not used
4. Spill Valve Sequencing Switch
5. Spill Valve Control Switch
6. Pressurization Control Switch
7. Air Intake Control Switch
8. Altitude Control Switch
9. Panel Lights Selector Switch
10. Anti-skid Control and Indicating Panel
11. Oxygen Regulator
12. Interphone
13. Altitude Indicator
14. Vertical Speed Indicator
15. Cabin Supercharger Outlet Temperature Indicator
16. Mass Flow Indicator
17. Airspeed Tube Static Source Selector Switch
18. Altitude Selector Knob
19. Cabin Pressure Indicators
20. Cabin Pressure Indicator Lights
21. Pressure Controller Data Card
22. Cabin Pressure Controller
23. Cabin Rate of Change Control Knob and Indicator

CO-PILOT'S SIDE PANELS

1. A.S.I. Static Source Selector Switch
2. Heater Discharge Temperature Indicator
3. Safety Valve Indicator
4. Cockpit Temperature Indicator
5. Interphone
6. Flight Deck Temperature Bulb
7. Oxygen Regulator
8. Interphone Panel
9. Interphone Switch
10. Oxygen Mask and Tube Storage Compartment
11. Seat Adjusting Lever
12. Air Conditioning Control Panel
13. Windscreen De-Icer Panel
14. Flight Deck Heat Control Lever
15. Air Conditioning Master Control Switch
16. Safety Valve Actuating Switch
17. Dump Valve Switch
18. Windshield De-Icing Control Switch
19. Panel Lights Control Switch
20. Radio Compartment Ventilating Blower Switch
21. Windshield Blower Switch

(e) Radio controls.
(f) Flap and Trim Controls.
(g) Parking brake control lever.

(7) Side panels are provided at both pilot's and co-pilot's positions. The pilot's side panels are mounted on the left side wall adjacent to the seat and the co-pilot's side panels are located on the right side wall of the flight deck.

(8) The left side panels (Fig. 8) are grouped just forward of the pilot's seat and comprise the following:-

(a) A pressurization control panel, incorporating relative selector knobs, indicators and switches.
(b) An automatic anti-skid brake control panel with master control switch, operating and warning lights.
(c) An interphone panel assembly incorporating a switch and plug-in sockets for the interphone head set.

(9) The right side panels (Fig. 8) are grouped adjacent to the co-pilot's seat and comprise the following:-

(a) An Air Conditioning control panel with the relative control switches and indicators.
(b) Interphone panel identical to that on the pilot's side panel layout.
(c) Anti-icing control panel.

(10) In addition to this equipment, there are a number of installations common to both right and left side panels. These are:-

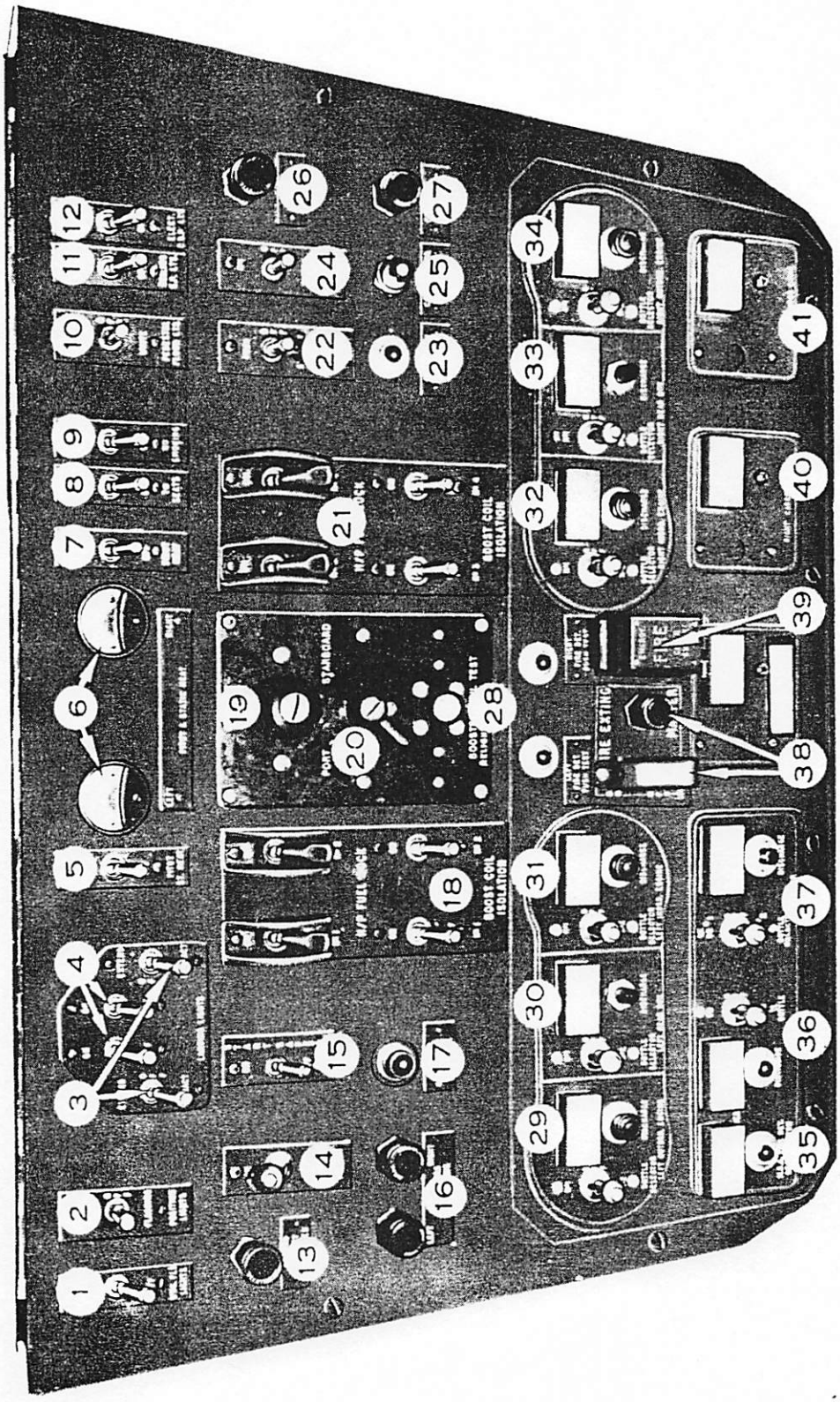
(a) Selector switches for control of the A.S.I. alternate static source.
(b) Oxygen regulators.
(c) Stowage compartments for oxygen masks and tubes.
(d) Hinged stowage compartments for maps and personal effects.

(11) Equipment mounted in the roof of the flight deck consists of the following:-

(a) A roof control panel centrally disposed over the windshield carrying the starting controls, external and internal light switches, fire extinguishing controls and fire detection indicators.
(b) Pilot's and co-pilot's interphone loudspeakers.
(c) Air conditioning air intake louvres.
(d) Escape hatch seal inflator compartment.
(e) Dome light and switch panel.

(12) A number of control panels are mounted on the left and right sides of the flight deck rear wall. The following are located on the left side of the rear wall.

(a) Radio power supply panel, incorporating circuit breakers, fuses and pilot lights. The lower section of the panel is covered by a detachable


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STARBOARD
PORT
TEST
FIRE
FIRE EXTING

ROOF PANEL

FIG. 9

1. Wing Flood Lights Switch
2. Position Lights Selector Switch
3. Landing Light Extend-Retract Switch
4. Landing Lights Switch
5. Pitot Static Heat Switch
6. Pitot and Static Heat Ammeters
7. Pitot Heat Switch
8. 'To Seats' Indicator Switch
9. 'No Smoking' Indicator Switch
10. Control Pedestal Floodlights Control Switch
11. Ceiling Starter Panel Lights Switch
12. Electrical Circuit Breaker Panel Lights Switch
13. 'Ground Battery Connected' Indicator Light
14. Battery Master Switch
15. Start Safety Switch
16. Generator Overheat Warning Lights
17. Starter Current Flow Indicator
18. Boost Coil Isolation Switches
19. Starter Button
20. Engine Selector Lever
21. H.P. Fuel Cock Switches
22. Main Inverter Switch
23. Ground Crew Call System Button
24. Inverter Control Switch
25. Cabin Attendant's Call System Button
26. Ice Warning Light
27. Forward Entrance Door Warning Light
28. Booster Coil Test - Relight Button

FIRE EXTINGUISHING CONTROLS

29. No. 1 Engine Bottle Selector and Discharge Button
30. Left Auxiliary Gear Box Bottle Selector and Discharge Button
31. No. 2 Engine Bottle Selector and Discharge Button
32. No. 3 Engine Bottle Selector and Discharge Button
33. Right Auxiliary Gear Box Bottle Selector and Discharge Button
34. No. 4 Engine Bottle Selector and Discharge Button
35. M.S.A. Reset Button
36. Bottle Selector and Discharge Button for Heater (Inoperative)
37. Bottle Selector and Discharge Button for Accessories Compartment
38. Master Fire Control Switch and Indicator Light
39. Emergency Crash Switch
40. Warning Right Cargo Compartment
41. Warning Aft Cargo Compartment

panel.
(b) Dome light switch.

(13) The right side of the rear wall accommodates the following:-

(a) Main electrical power supply panel incorporating Circuit breakers and fuses.
(b) Generator and alternator control panel.
(c) Fire detection system fire warning bell.

(14) Extra equipment for a flight observer is carried in the flight deck. This equipment consists of the following:-

(a) A box seat, covering the standard aircraft batteries located at the rear of the co-pilot's seat.
(b) An additional oxygen regulator, located on the right side wall aft of the co-pilot's seat.
(c) An interphone head set and jack box.

(15) The cabin attendant's equipment is mounted on the left wall of the cabin. The major portion of this equipment comprises a cabin heating control panel, interphone equipment and the electrical system aft power junction box. These components are located on the left side of the fuselage (Fig. 10) adjacent to the rear baggage compartment forward bulkhead. An additional panel for the control of hall and roof lighting is mounted on the cabin roof adjacent to the main entrance door.

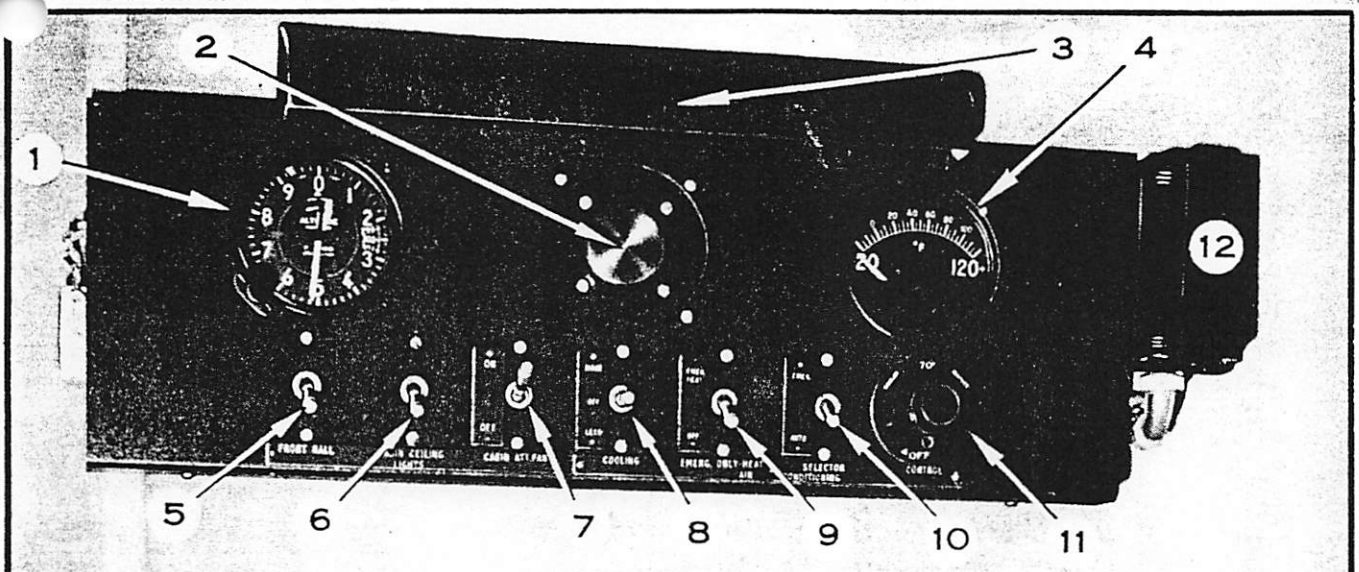
1-12 CREW SEATS:

(1) The pilot and co-pilot are seated in the conventional side by side arrangement in the flight deck. The seats are mounted on slide rails attached to the flight deck and can be adjusted in a fore and aft direction by operating the relative lever on the inboard side of the seat. Each seat is provided with hinged armrests and can be adjusted for height by the lever located on the outboard side of the seat.

1-13 RUDDER PEDAL ADJUSTMENT:

(1) The rudder pedals can be adjusted for both height and distance to suit the requirements of crew members. Adjustments are made through the depressing of two small levers, conveniently mounted on each brake pedal control rod and on the pivot quadrants. Once the pedals have been set, the adjusting levers must be returned to the original locked position.

CABIN ATTENDANT'S CONTROL PANEL FIG. 10



1 CABIN ALTIMETER

2 CABIN TEMPERATURE BULB

3 FLOODLIGHT

4 CABIN TEMPERATURE INDICATOR

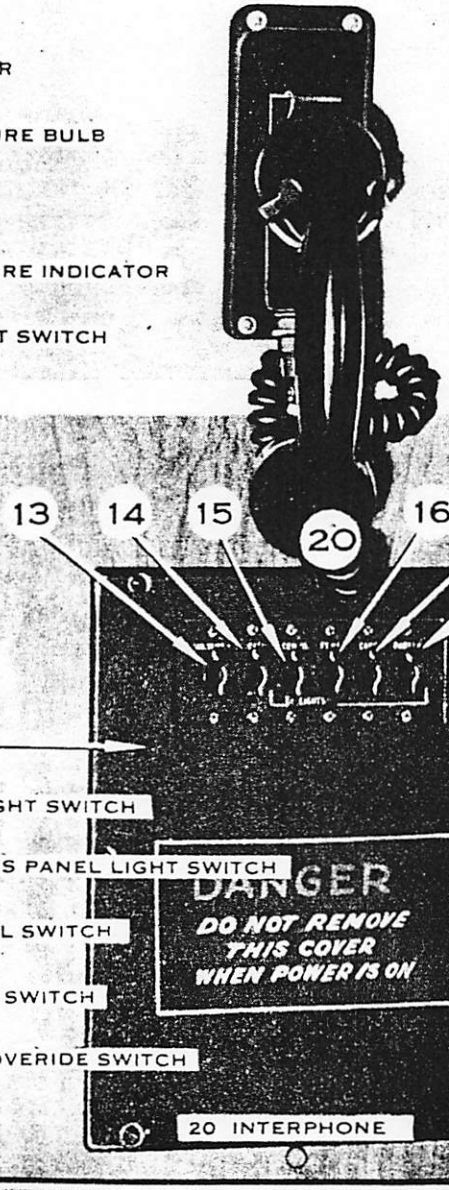
5 FRONT HALL LIGHT SWITCH

11 AIR CONDITIONING TEMPERATURE SELECTOR

12 CABINSTAT

13 CABIN TEMPERATURE CONTROL CIRCUIT BREAKER

14 INTERPHONE "CALL" CIRCUIT BREAKER



6 CABIN CEILING LIGHT SWITCH

7 CABIN ATTENDANTS PANEL LIGHT SWITCH

8 COOLING CONTROL SWITCH

9 EMERGENCY HEAT SWITCH

10 AUTO CONTROL OVERIDE SWITCH

15 CEILING LIGHT CIRCUIT BREAKER

16 FRONT HALL LIGHT CIRCUIT BREAKER

17 CARGO COMPARTMENT LIGHT CIRCUIT BREAKER

18 PANEL LIGHT CIRCUIT BREAKER

19 AFT POWER JUNCTION BOX

20 INTERPHONE

CHAPTER 3.

FLYING CONTROLS

1-14 GENERAL:

(1) Conventional dual controls are installed in the flight deck. Each set comprises a pair of fully adjustable rudder pedals and an aileron handwheel mounted on a cranked elevator control column. These three controls are linked directly to their respective movable control surfaces by a system of levers and push-pull rods, (Fig. 11). The system enables backlash and friction to be kept to a minimum and is unimpaired by severe temperature change. The rods are of aluminum alloy and are supported by self-lubricating guide bearings which are equally disposed around the rods in sets of three at each support. Seals are fitted at the pressurized bulkheads. Connections to servos operated by a Pioneer PB10 autopilot are incorporated in the control runs. Manually-operated trimmer controls are fitted (Fig. 12).

(2) The outer and inner landing flaps are electro-hydraulically actuated by means of switches on the control pedestal (Fig. 7).

(3) Aileron control is assisted by a Fairey hydraulic booster unit located on the rear spar in the centre of the fuselage. Use of this device reduces the pilot's manual efforts by 80%. Even with the hydraulic booster inoperative, the stick forces are not unusually high. If preferred, the pilot can disengage the unit by means of a manually-operated control lever located on the pedestal. Each aileron is mass-balanced internally and extends for about 53% of the outer wing span.

(4) The rudder pedals are pivoted below the floor and are fully adjustable to give complete support for the feet. Each pedal is articulated for differential or equal brake application. The single rudder surface is unbalanced and is carried on piano hinges secured to the left of the fin.

(5) Fore and aft movement of the column in the usual way controls the elevators, which are externally mass-balanced.

NOTE

(1) The movement ranges and tolerances of the main control surfaces are as follows:

		Angular Movement	Angular Tolerance
Rudder:	To Right	24°	± 1°
	To Left	24°	± 1°
Elevators:	UP	30°	± 1°
	DOWN	20°	± 1°
Ailerons:	UP	17°	± 1°
	DOWN	17°	± 1°

FLYING CONTROLS AND FLAPS

FIG. 11

TRIM CONTROLS
FIG. 12

1-15 TRIM CONTROLS:

(1) Except for the off-set rudder trim tab control wheel the manually-operated trim controls are duplicated on each side of the engine control pedestal. The trim surfaces are cable-operated by means of handwheels which function in the conventional manner. One trim tab is fitted to each of the five main movable control surfaces. In addition to the manually-operated elevator trim control, an electrically-operated thumb switch is fitted to each aileron control wheel for adjustment in flight.

NOTE

(1) Trim tab movement ranges and tolerances are as follows:

Tabs		Angular Movement	Angular Tolerance
Rudder Tabs:	To Right	15°	± 1°
	To Left	15°	± 1°
Elevator Tabs:	UP	15°	± 1°
	DOWN	15°	± 1°
Aileron Tabs:	UP	13°	± 1°
	DOWN	13°	± 1°

1-16 FLAPS:

(1) Hydraulically-operated outboard and inboard split type trailing edge flaps are fitted respectively to the outer and centre section wing assemblies. The inboard flaps are incorporated for use as dive flaps.

(2) The landing flaps are operated by means of a flap selector and an actuator switch mounted on the control pedestal (Fig.7). The latter is an ON-OFF switch, for both inboard and outboard flaps. The selector enables the pilot to preselect the desired flap angles for both inboard and/or outboard flaps. An additional switch is used for independent operation of the inboard dive flaps and functions only when the NORMAL actuator switch is in the OFF position.

(3) The inboard or outboard landing flaps should not be lowered at indicated air speeds in excess of 250 m.p.h. Flaps setting at this speed is restricted to 30°.

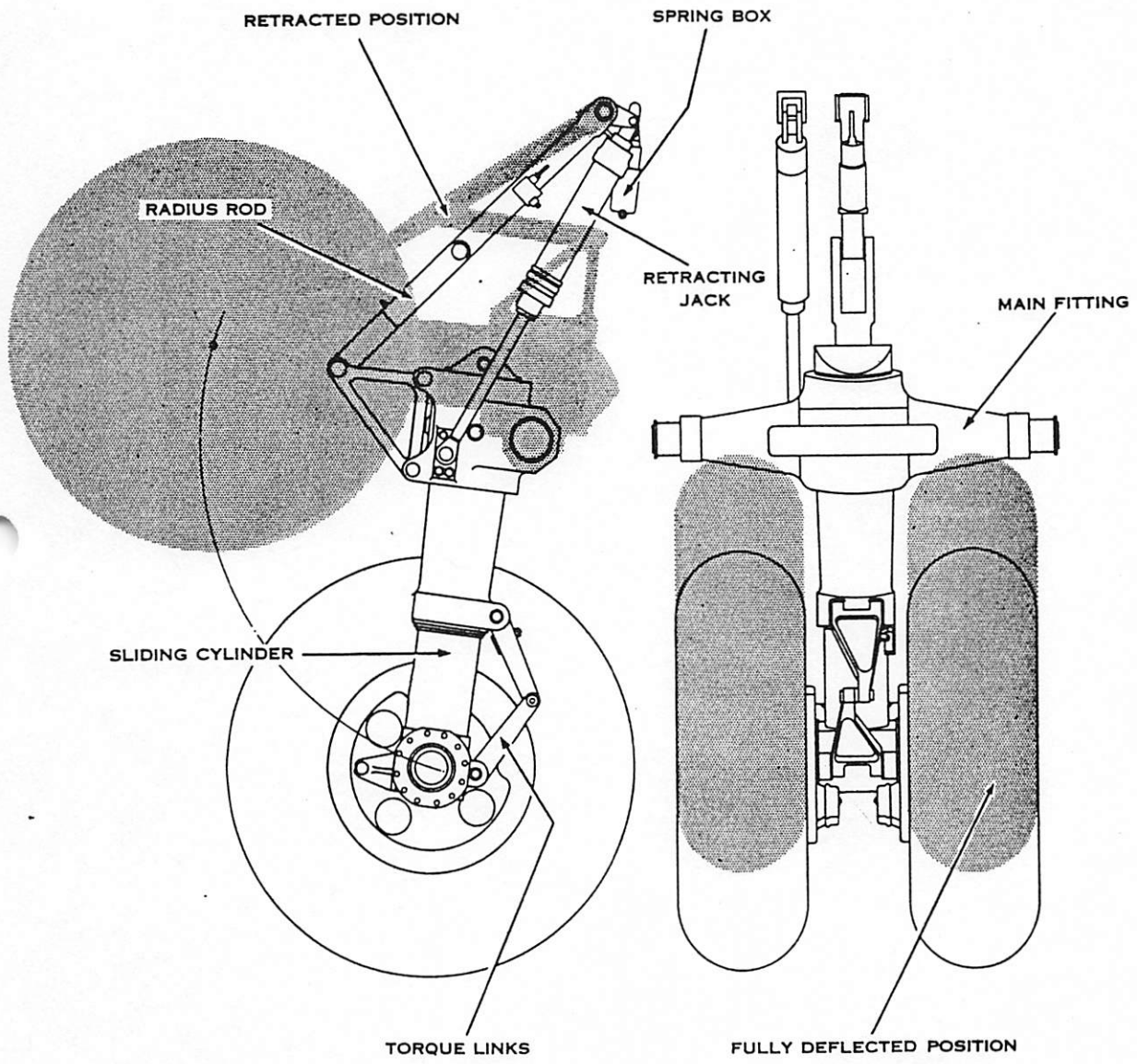
1-17 CONTROL LOCKS:

(1) Provision is made on the aircraft to incorporate cable controlled gust

locks. This system is at present inoperative, but when operating, will be controlled by a push-pull mechanism located at the base of the flight deck control pedestal.

(2) The flying controls can also be secured by means of a set of locks stowed in a canvas bag in the rear baggage compartment. The rudder locks comprise a pair of wooden bars which clamp over the rudder pedals when the latter are set in their neutral position. The elevator lock assembly comprises a single leather strap passed round the rudder lock and the control column to secure the latter in its most forward position. An additional strap is used to lock the aileron handwheel in the mid-position.

OLEO AEC PRESSURES		NOSE	-	MAIN
	SUMMER -	1500	-	2600
	WINTER -	1700	-	3200



CHAPTER 4.

LANDING GEAR.

1-18 GENERAL:

(1) The tricycle type landing gear is composed of three retractable units. Two main landing gears equipped with dual wheels and hydraulic brakes are fitted to the right and left nacelle assemblies, while a steerable nosewheel landing gear, with twin wheels and hydraulic steering, is fitted to the nose section of the fuselage. The nosewheel steering is controlled from an independant wheel mechanism in the flight deck compartment.

(2) Lowering and raising of the gear mechanism is accomplished by hydraulic actuators controlled electrically from the flight deck pedestal.

(3) In the event of NORMAL hydraulic or electrical system failures, the gear may be lowered by means of an EMERGENCY electrical power pack, or by the emergency manual release system.

(4) Accidental UP selection on the ground is prevented by a micro-switch, which operates when more than 5% of the aircraft weight is on the wheels.

1-19 MAIN LANDING GEAR:

(1) As shown on Fig. 13, each main Dowty Type T1301 landing gear unit consists fundamentally of a shock-absorber unit carrying twin wheels at its lower end and pivoting at its upper end about a main fitting secured to the nacelle structure. A hydraulic actuator is used to raise and lower the unit in conjunction with an elbow-joint type drag strut incorporating a DOWN lock. A spring box assembly on the upper end of the drag strut gives the final impulse to lock the unit in the DOWN position.

(2) The upper main housing carries the shock-absorber cylinder which is filled with hydraulic fluid under pressure. The lower sliding cylinder houses the shock-absorber piston and rod. Four holes are drilled through the piston head but two can be sealed by the upward movement of a plate valve against the underside of the piston. The main cylinders are interconnected externally by a torque link assembly.

(3) When the shock absorber is compressed, the ram is thrust upward into the cylinder, forcing some of the fluid through the four holes and tending to compress the remainder of the fluid. This action produces the springing medium for taxying and landing loads. As the ram reaches the top of its stroke, pressures are equalized above and below the piston head, but due to the greater area of the upper surface there is a resultant downward force which returns the ram to its extended position. During the recoil stroke the reversed flow of fluid to the plate valve closes two of the holes in the piston head to damp the recoil action.

1-20 OPERATION:

(1) To raise or lower the landing gear in flight (Fig. 7), the pilot presses the appropriate UP or DOWN shielded push-switch on the right side of the control pedestal. When UP selection is made, and the gear is fully retracted the UP lock micro-switch actuates the landing gear door UP valve. Once the doors are closed, the DOORS CLOSED de-energizing switch is depressed, and the system is electrically de-energized, the initial movement of the operating cylinder unlocks the drag strut at the elbow joint and simultaneously breaks the contacts of a micro-switch on the drag strut upper member. The landing gear units are then raised as the drag strut retracts and are finally locked in the UP position. The completion of each stage is signalled by an indicator mounted on the instrument panel.

(2) When a landing gear DOWN selection is made, the up-locks are released and the landing gear is lowered. As the two members of the hinged drag strut straighten the spring box assembly assists in the final straightening of the elbow joint until the joint is locked and the micro-switch contacts depressed.

1-21 MAIN LANDING GEAR DOORS:

(1) The main landing gear doors are hinged from the longitudinal walls of the main wheel wells. The latter are positioned between the right and left nacelle groups.

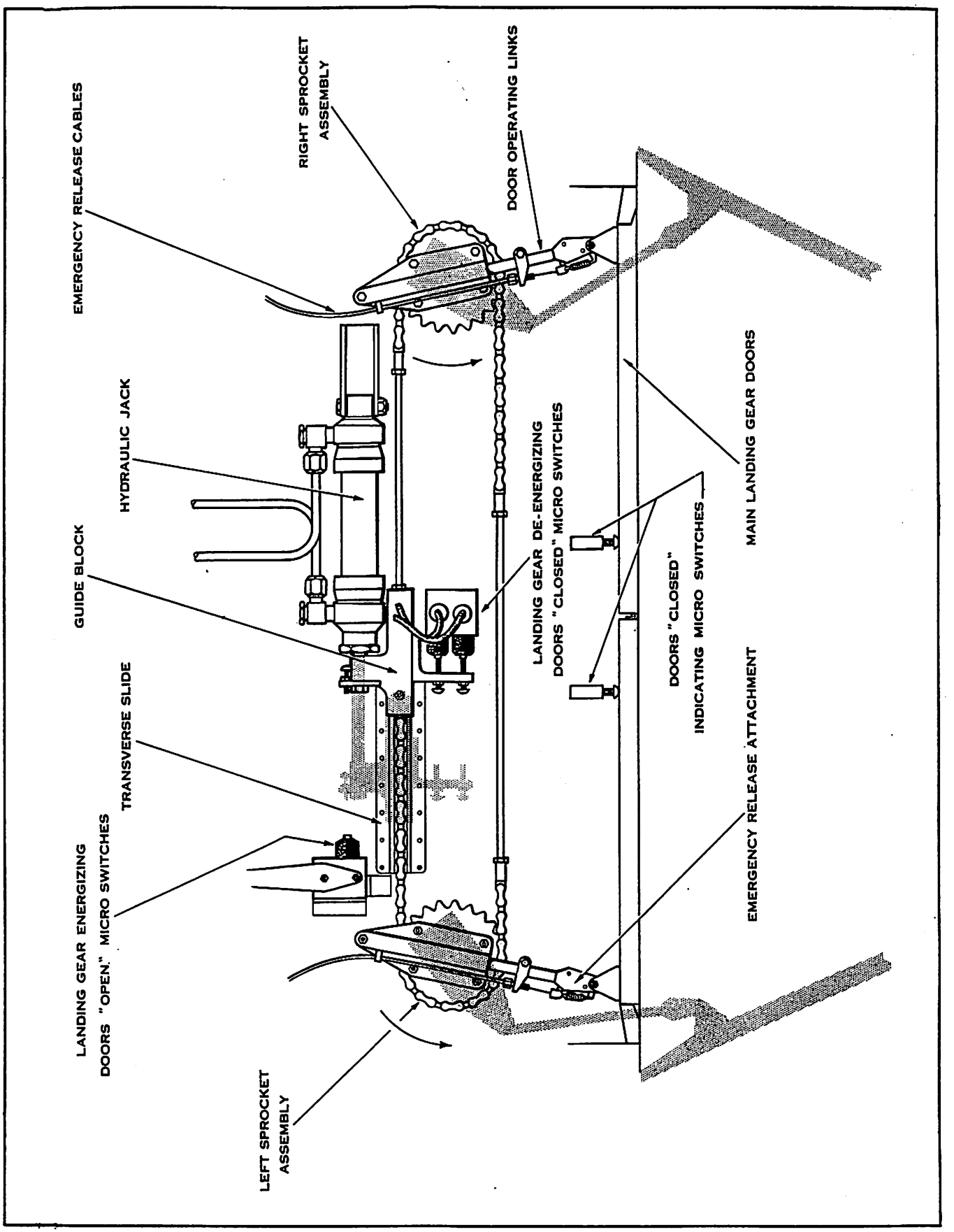
(2) Each assembly comprises two doors hinged at front and rear and controlled through a mechanical linkage and a hydraulic actuator. The actuator and linkage assembly is mounted on the rear face of the wheel well forward cross frame and screened at the rear by a splash guard. The latter is retained in position by means of camlock fasteners.

(3) The mechanism (Fig. 14) is controlled through a hydraulic jack secured by brackets to the forward wall of the wheel well frame. An attachment on the piston end of the jack cylinder is bolted to a guide block operating on a transverse slide. The slide is also attached to the forward wall and centrally disposed between two gears attached to the door operating links. Chains and push pull rods interconnect the guide block with the right and left sprocket assemblies.

(4) The lower end of each door operating link carries a slotted cam incorporating a spring loaded catch, the catch pivoting about a retaining rivet providing an attachment for the door retaining bracket on the forward end of each door. In the event of hydraulic failure, a cable attached to the catch and controlled by a lever pivoted on the rear face of the main spar below the cabin floor releases the door and allows it to fall open under its own weight.

(5) Four micro-switches bracketed to the forward wall and controlled by adjustable stops on the guide block, control the operation of the main landing gear. Two additional micro-switches, centrally disposed over the rear of the wheel well frame, control the operation of the door indicating lights on the flight deck.

MAIN GEAR DOOR
MECHANISM
FIG. 14



(6) OPENING CYCLE: When the landing gear is selected DOWN, the main landing gear door selector valves are energized, and hydraulic fluid is forced into the operating jack and the piston is extended. This movement of the piston forces the guide block outwards, the resulting movement rotating the gears in an anti-clockwise direction, pushing the door links down and the doors to the open position. Once the doors are open, the DOORS OPEN energizing switches are depressed (Fig. 14) to energize the main gear DOWN selector valve.

(7) CLOSING CYCLE: As the landing gear reaches the final stage of the retracting cycle, the landing gear UP micro-switch energizes the DOORS UP selector valve, and fluid is directed to the door operating jack and the piston is retracted. This movement pulls the attached guide block inwards, rotating the gears in a clockwise direction, and directing the links upwards to retain the doors in the closed position. As the doors are CLOSED, the de-energizing micro-switches are depressed and the circuit is broken.

1-22 NOSE GEAR:

(1) The Dowty type T1300 nose landing gear unit comprises a levered suspension shock-absorber strut incorporating liquid springing, a radius rod assembly, a retracting jack, and a steering control assembly attached to the strut. A Dunlop wheel unit is mounted on either side of the axle. UP and DOWN locks are incorporated in the elbow type joint of the drag strut in the assembly. The major components of the unit are attached to structural members in the nose wheel bay, forward of the pressure bulkhead. Figure 15 shows the general arrangement of the complete unit and indicates the retraction geometry.

1-23 OPERATION:

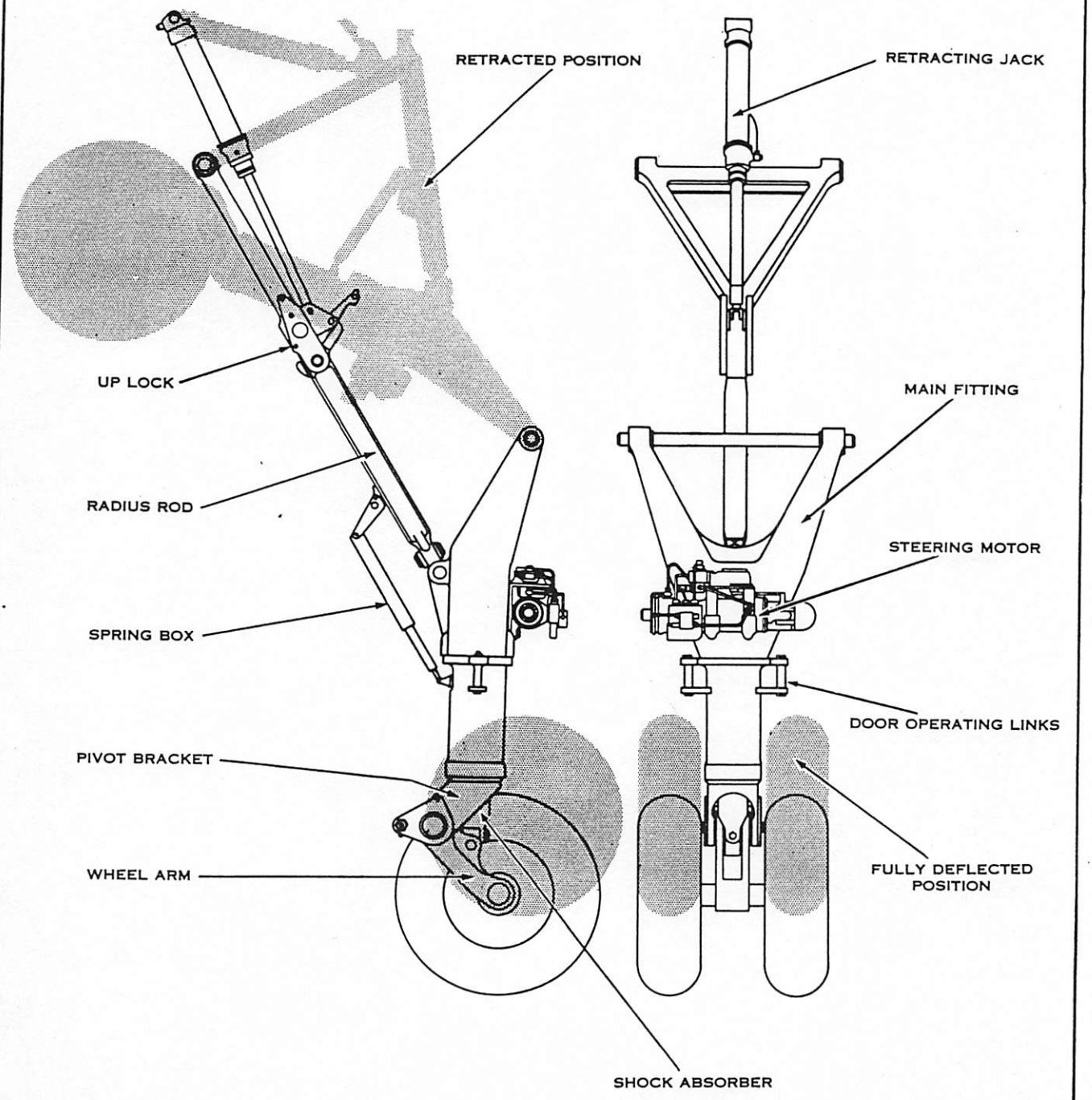
(1) When landing gear UP is selected, the initial movement of the hydraulic jack actuates the mechanism which releases the DOWN latch and breaks the contacts of the micro-switch connected to the flight deck indicator. Further jack movement breaks the elbow joint so that the drag strut folds and retracts the landing gear. Continued upward movement pushes the UP latch on the drag strut over the latch bar on the aircraft structure and secures the landing gear in the UP and LOCKED position.

(2) When landing gear DOWN is selected, the initial movement of the hydraulic operating jack disengages the UP latch from its locating bar on the aircraft structure. The continued expanding movement of the jack forces the landing gear to the DOWN position, the final straightening of the elbow joint and the locking of the unit in the DOWN position being assisted by a combination of gravity and drag forces. The mechanism is so designed that the gear can drop to the DOWN and LOCKED position without the aid of fluid pressure from the hydraulic operating jack.

1-24 NOSE GEAR DOORS:

(1) The two nose wheel well doors are hinged front and rear to the nosewheel

NOSE LANDING GEAR
FIG. 15



landing gear.

(2) Each door mechanism comprises a cam plate pivoted on a bracket attached to its respective longitudinal member of the wheel well frame (Fig.16). Integral with the bracket is a 'Doors Open' stop. Attached to the cam plate, by means of clevis pins, is the door operating link and two tension springs. The lower ends of the springs are attached to a bracket on the operating link while the lower end of the operating link pivots on a clevis pin secured to the door bracket.

(3) OPENING CYCLE: During the initial 10° angular downward movement of the nose landing gear, the strut contact rollers engage with the slots in their relative cam plates. This engagement forces the cam plates to rotate about their axes, and at the same time, push the connecting links down and pivot the hinged doors to the open position. The outward travel of the doors is limited by the stops contained in the cam brackets. The tension springs are so arranged that the doors will be retained in the open position.

(4) CLOSING CYCLE: As the landing gear reaches the approximate final 10° of travel in the retracting cycle, the contact rollers of the strut engage with the inner extremities of the slots in the door cams and pull the door upwards. As the cams rotate toward the closed position, the strut rollers disengage from the cam slot and permit the tension springs to hold the doors closed.

1-25 WARNING DEVICES:

(1) Visual and audible warning devices are incorporated.

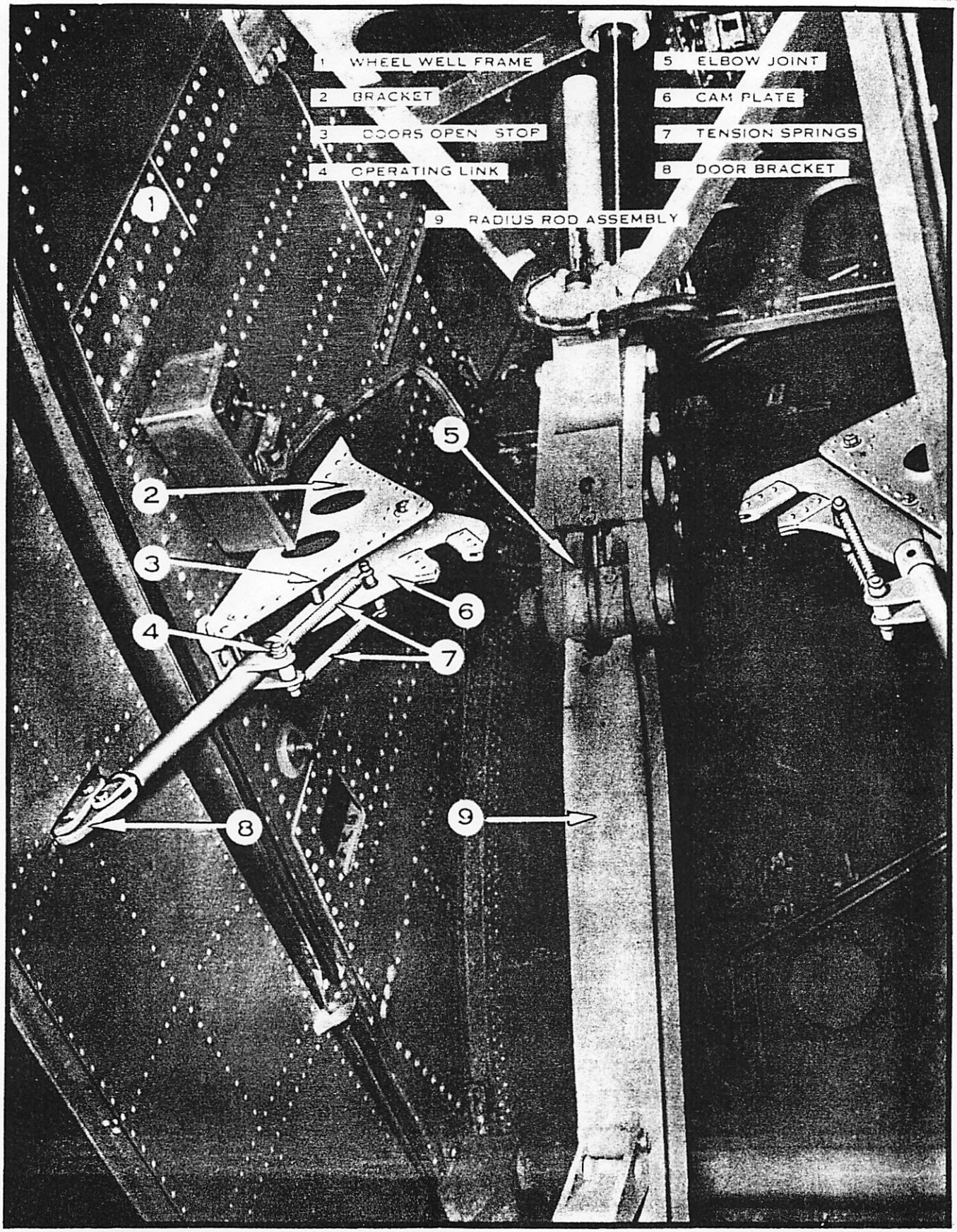
(2) The visual indicator comprises three red warning and three green safety lights, suitably labelled, for the nose and each of the main landing gear assemblies. The GREEN lights will register when the landing gear is in the DOWN and locked position. Any intermediate positioning of the landing gear between the UP and locked, or DOWN and locked positions will be indicated by the illumination of RED warning lights.

(3) The audible system comprises a horn electrically interconnected with the landing gear down locks and a pressure capsule incorporated in the pitot and static lines of the A.S.I. system (Paras. 1-75). When the airspeed falls below 130 m.p.h., the differential pressure felt on the pressure capsule closes two electrical contacts and power is relayed to operate the horn; providing the landing gear is not in the DOWN and LOCKED position. The horn is located behind a grill on the left side of the flight deck adjacent to the pilot's seat.

1-26 EMERGENCY MANUAL RELEASE: (Main Gear)

(1) The main landing gear release lever is located aft of the rear spar in the fuselage centre section below floor level. Access to the lever is gained through the centre floor panel immediately over the rear spar.

NOSE GEAR DOOR
MECHANISM
FIG. 16



(2) The mechanism comprises a lever and cable assembly, the cables extending through either side of the fuselage and terminating at the main landing gear door pins, UP lock levers and hydraulic system jettison valves.

(3) The cable in each main wheel well is so rigged, that when the lever is operated, the door link pins will be released on the initial movement of the lever, the releasing of the landing gear UP lock and the jettisoning of the fluid back to the hydraulic system return line following immediately afterwards.

(4) As the landing gear drops, the combined forces of gravity and air pressure force the unit to the DOWN and locked position.

NOTE

(1) For information on the EMERGENCY system, incorporating the use of the electrically driven hydraulic power pack, refer to Para. 1-58 of this manual.

1-27 EMERGENCY MANUAL RELEASE: (Nose Gear)

(1) The release handle for the nosewheel unit is located on the right side of the flight deck floor adjacent to the co-pilot's seat.

(2) This mechanism comprises a control handle attached to a flexible steel cable, the latter passing through the nose of the fuselage to a series of levers interconnected with the nose landing gear UP lock and the hydraulic system jettison valve in the nosewheel well.

(3) When the emergency lever is operated, the nose gear UP lock is broken and the hydraulic fluid is released from the UP side of the operating jack and returned to the hydraulic system. The landing gear then drops, and through a combination of gravity and air pressure, it is forced backward to the DOWN and locked position.

1-28 NOSE GEAR STEERING:

(1) Steering control through a range of 100 degrees is provided by means of the Dowty type T1305-50 nosewheel steering unit.

(2) A shielded switch to the left of the main instrument panel controls the operation of the steering motor. An indicator light above the switch lights up to indicate that the steering control is engaged. The handwheel to the right of the switch functions in the conventional manner to control the steering mechanism.

(3) When the steering motor is not engaged, anti-shimmy castoring through 50 degrees is obtainable. Anti-shimmy damping is provided by the by-pass valve in the steering unit, and is supplemented by the steering jack being automatically

self-centering.

(4) Nosewheel steering can be used for taxying, take-off up to V1 speeds, and for the latter part of the landing run. It is especially useful on cross wind take-offs and landings and in the case of engine failures on take-off.

1-29 GROUND LOCKS:

(1) The ground locks are fabricated steel plates which are fitted over the elbow joints of the main and nose landing gear radius rod assemblies when the aircraft is on the ground. The locks are painted red and have red warning streamers attached.

(2) Each lock assembly comprises two plates which are placed one on each side of the drag strut immediately below the elbow joint of their respective units.

1-30 MAIN WHEEL BRAKE MECHANISM:

(1) The action of the rudder pedals govern the brake control valve which in turn directs hydraulic fluid under pressure to the hydraulic cylinders in each brake assembly.

(2) A multiple plate type brake unit is incorporated in each of the four wheel assemblies of the main landing gear. The action of each unit is dependent upon the hydraulic operation of two radially-spaced friction pads, the latter bearing on the rotating surfaces of vertically-mounted concentric brake plate rings.

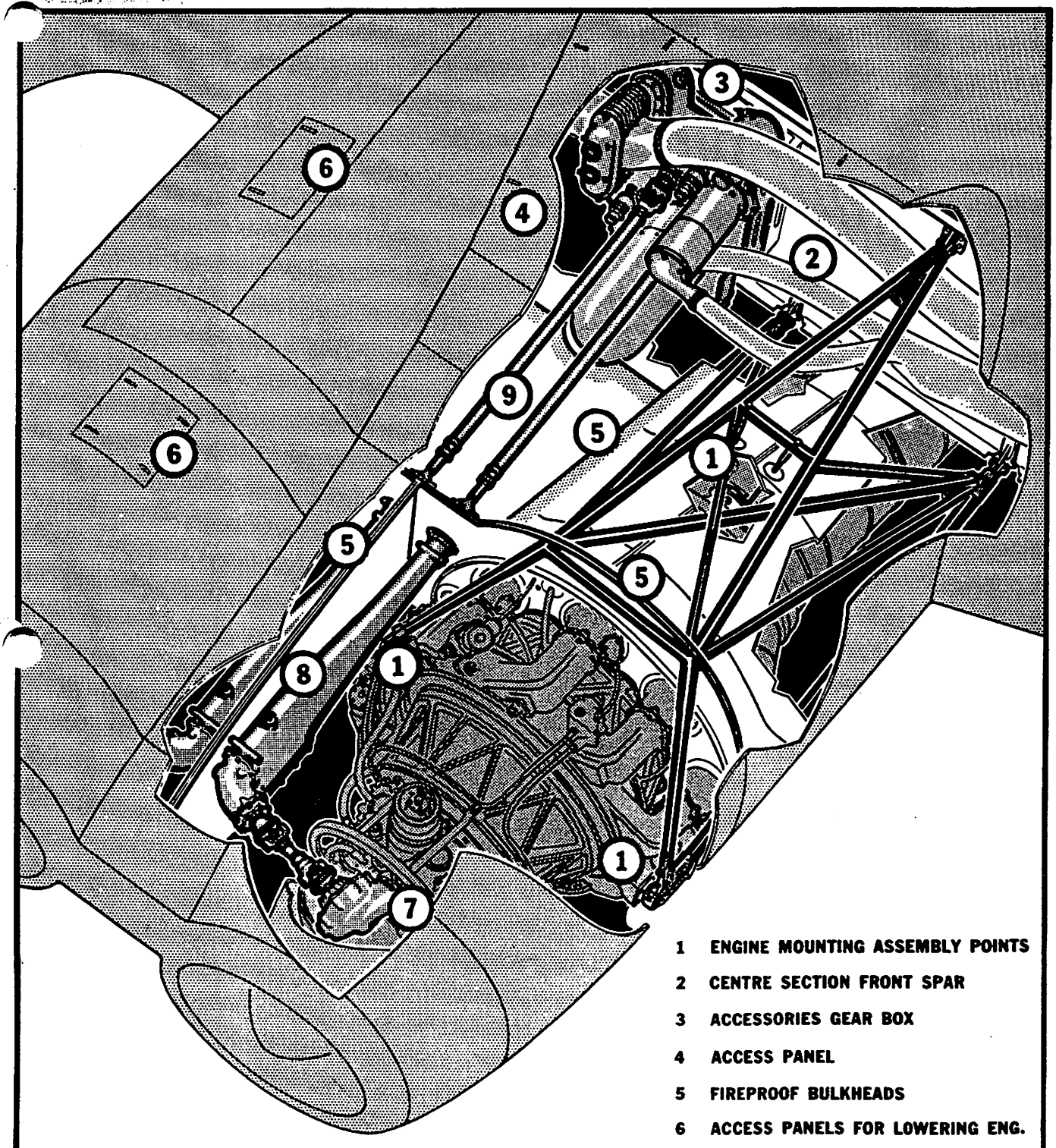
(3) The brakes operate at a maximum differential pressure of 700 p.s.i.

1-31 WHEELS AND TIRES:

(1) Each wheel is of the bolted flange type, the flanges being knurled to prevent tire creep.

(2) Dunlop 33 x 11.75-17 tires are fitted to the main wheels and are of the nylon cord type. Creep marks are painted on the walls. For correct operating tire pressures, refer to Figure 109.

(3) Dunlop 26 x 7.75-13 tires are fitted to the twin nosewheels. These tires are of the electrically conducting type with cotton cord. Creep marks are painted on the walls. For correct operating tire pressures, refer to Figure 109.



1 ENGINE MOUNTING ASSEMBLY POINTS
2 CENTRE SECTION FRONT SPAR
3 ACCESSORIES GEAR BOX
4 ACCESS PANEL
5 FIREPROOF BULKHEADS
6 ACCESS PANELS FOR LOWERING ENG.
7 BEVEL GEAR DRIVE HOUSING
8 INTERMEDIATE DRIVE HOUSING
9 INPUT DRIVE SHAFT

CHAPTER 5

POWER PLANT INSTALLATION

1-32 GENERAL:

(1) Four Rolls Royce Derwent 5/17 engines complete with detachable jet pipes are mounted in pairs in the two underslung nacelles (Fig. 17).

(2) An engine driven accessories gear box is attached to the front spar of the wing between each pair of engines. The gear box casing houses two entirely independent gear systems, one being driven by each engine. Each inboard engine drives a cabin blower and a tachometer transmitter. A vacuum pump is driven by the left inboard engine only. Each outboard engine drives a 50 Kw. alternator, a 9 Kw. generator, a hydraulic pump and a tachometer transmitter. Grease lubricated flexible couplings, and a system of drive shafts connect each engine to its accessories gear box. Each engine, the associated bevel gear boxes, and the accessories gear boxes, have independent integral oil lubricating systems.

(3) The main controls for each engine consist of a throttle, which varies the flow of fuel to the burners and consequently the engine r.p.m., and a high pressure fuel cock for controlling the fuel supply to the engine. An additional low pressure fuel cock for each engine is part of the aircraft fuel system. Figure 7 shows the location of the controls on the throttle control pedestal.

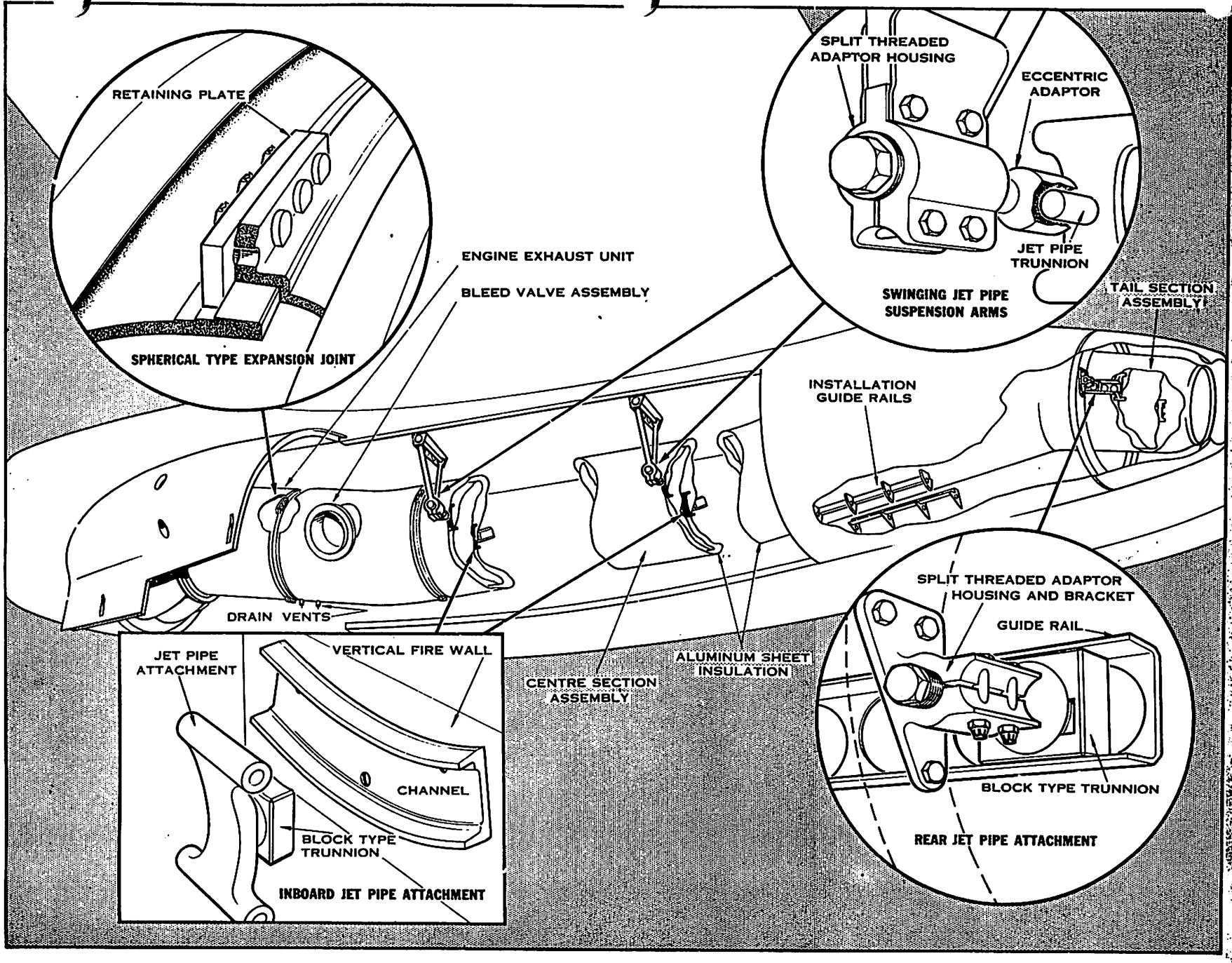
(4) A water methanol system is incorporated and has an independent control system.

(5) Each jet pipe is located in a stainless steel tunnel riveted to adjacent nacelle formers. The tunnel is installed for the purpose of deflecting the heat of the jet pipes away from the nacelle structure.

(6) The jet pipes are located in position by means of suspension arms and trunion guide rails. Each jet pipe assembly comprises three sections which are joined together by means of expansion joint rings and retaining plates. The forward section of each assembly is fitted to the engine exhaust outlet, while the after end of the tail section is located within a detachable fairing.

(7) Figure 18 illustrates the assembly and component parts of each installation. These are, the forward or bleeder valve section (now inoperative as such), a centre section and a tail section. The outer surface of the centre and tail sections are covered with aluminum sheeting to provide additional insulation, while four thermocouples arranged in pairs are located around the circumference of each tail section.

(8) Each jet pipe assembly (Fig. 18) is suspended from the nacelle at six points. Located on the inboard side of the nacelle just aft of the front spar and forward of the rear spar, are two curved channel section guides. The guides



JET PIPE INSTALLATION
FIG. 18

TYPE C-102

are bolted to the firewall and accommodate the female members of the jet pipe trunnions. This arrangement permits fore and aft movement of the jet pipe to allow for expansion and contraction of the pipe under operating conditions.

(9) Located adjacent to the outboard side of the nacelle and suspended from the top of the jet pipe tunnel, are two swinging jet pipe suspension arms. These arms are positioned to face the curved channel sections.

(10) Each jet pipe suspension arm carries at its base a split threaded adaptor housing. Each adaptor is secured by two clamping bolts. The adaptor which is threaded into this housing, accommodates the outboard trunnions on the side of the jet pipe. On the aft jet pipe suspension arm, the trunnion end of the adaptor is eccentric to its centre line. This eccentricity permits a screw adjustment to be employed in order to engage the adaptor with the jet pipe trunnion.

(11) Tail pipe centering is provided by similar attachments to those employed on the suspension arms. These attachments consist of two split threaded brackets secured to the rear of the nacelle structure. Screwed to these brackets, are block type trunnions which are accommodated in guide rails at the rear of the jet pipe. Each trunnion block is eccentric to the threaded portion and provides vertical adjustment for the rear portion of the jet pipe.

(12) The swinging suspension arms permit longitudinal expansion of the jet pipe, the trunnions at the rear position sliding within the jet pipe rail. Diametrical expansion is permitted by the provision of a .125" cold clearance at each attachment joint.

(13) In order to permit access to the jet pipe at the rear of the nacelle, the tail cone fairing is detachable. This fairing is secured to the nacelle structure by means of quick release fasteners which locate on attachment pins secured to the rear face of the jet pipe housing, and to the trailing edge section of the wing between the nacelles.

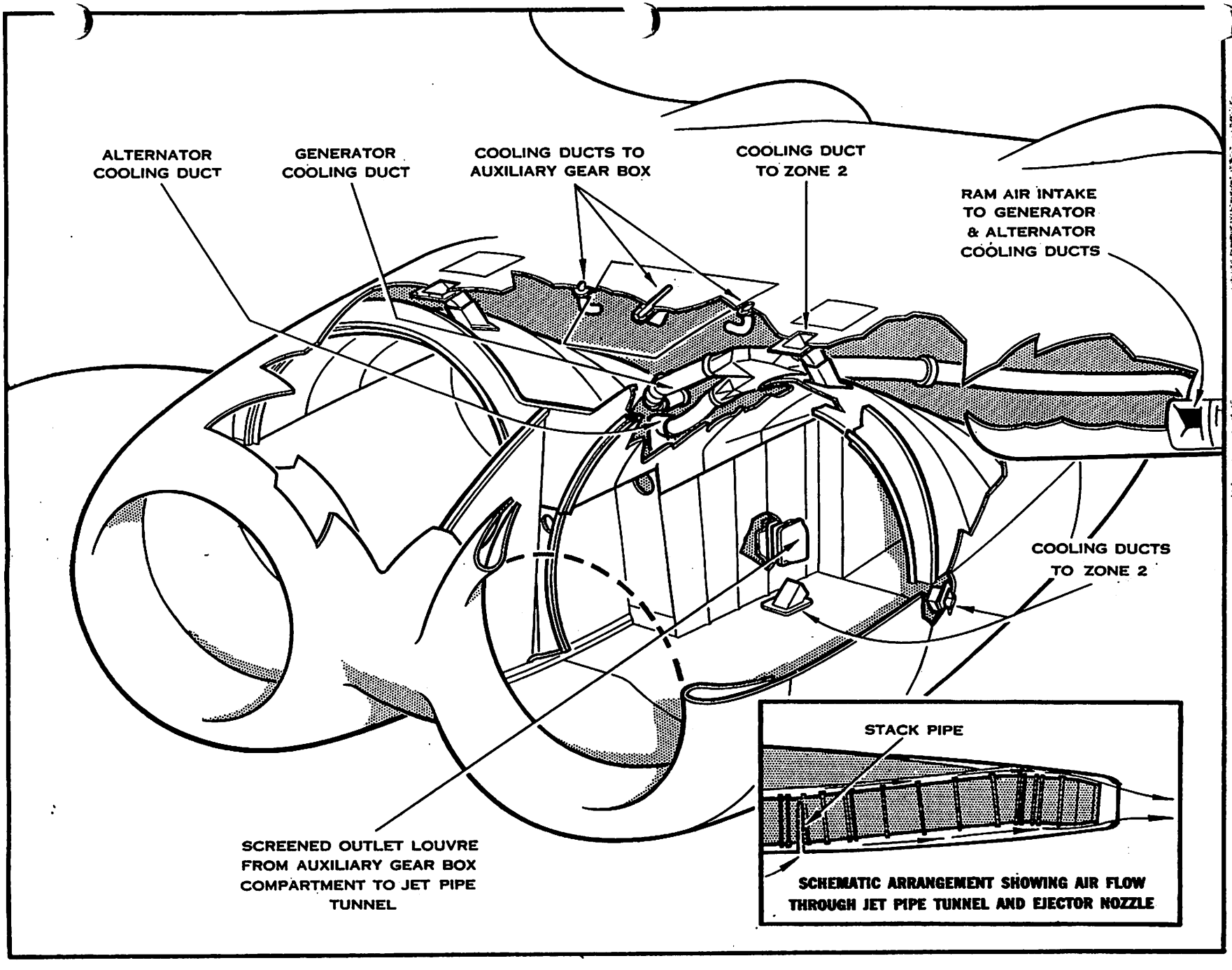
(14) Additional access panels to facilitate inspection and removal of the jet pipe assembly are shown in Figure 4.

(15) Gas turbine engines, including the jet pipe assemblies, require a minimum of external cooling. In general, all external cooling is provided mainly for the protection of adjacent aircraft structure.

(16) Internal cooling maintains engine and jet pipe temperatures within safe operating limits. Combustion gases, combined with the excess air within the engine, provides the necessary cooling medium.

(17) External cooling is incorporated to supply a blanket of insulating air between the engine, the jet pipe and adjacent aircraft structure.

(18) Air entering the engine air intakes supplies the necessary cooling to the forward engine compartment (Zone 1), while the rear engine compartment


ALTERNATOR COOLING DUCT
GENERATOR COOLING DUCT
COOLING DUCTS TO AUXILIARY GEAR BOX
COOLING DUCT TO ZONE 2
RAM AIR INTAKE TO GENERATOR & ALTERNATOR COOLING DUCTS
COOLING DUCTS TO ZONE 2
SCREENED OUTLET LOUVRE FROM AUXILIARY GEAR BOX COMPARTMENT TO JET PIPE TUNNEL
STACK PIPE
SCHEMATIC ARRANGEMENT SHOWING AIR FLOW THROUGH JET PIPE TUNNEL AND EJECTOR NOZZLE

(Zone 2) and the jet pipe tunnel (Zone 3) is protected by cooling air drawn through a series of ducts and directed over the engine combustion chambers and through the jet pipe tunnel by the ejector nozzle (Fig. 18A). The latter supplies the venturi effect necessary to create a depression over the entire cooling area.

(19) Cooling air enters the nacelle through the three ducts adjacent to the engine combustion chambers (these are located immediately aft of the firewall on the engine cowlings) and passes over the rear of the engine and through the jet pipe tunnel to the ejector nozzle. To distribute the cooling air more evenly between the jet pipe tunnel and the after end of the dividing nacelle firewall, a further duct incorporating a slotted stack pipe is located in the lower nacelle skin adjacent to the forward end of the water methanol compartment. The stack pipe extends upwards into the jet pipe tunnel and supplements the cold air supply to that particular section.

(20) Cooling for the auxiliary gearbox compartment is arranged through three intake ducts located in the upper skin surface (Fig. 18A) of each nacelle assembly. Two outlet ducts or louvres, are located on the side walls of the compartment. These louvres allow the air to flow from the compartment and exhaust to atmosphere through the jet pipe tunnel.

(21) Additional cooling is provided for the generator and alternator assemblies which are mounted on each auxiliary gear box. This cooling is arranged through separate ducting which is interconnected with a ram air intake located in the leading edge of each outer wing. Integral cooling fans built into the generator and alternator assemblies, ensure a constant supply of cooling air to these units and provide the necessary flow when the aircraft is stationary on the ground.

NOTE

(1) All air ducts are strategically located, so that the supply of coolest air is initially directed to the most critical areas. All ducts are covered with wire mesh to prevent the entry of foreign matter into the cooling system.

1-35 FUEL VALVES AND COCKS:

(1) All valves and cocks which serve to control one side of the aircraft fuel system are conveniently grouped in a compact unit where all fuel lines meet. Each of the two units is mounted in a safe position outside the relevant nacelle where it can be efficiently drained and vented. The valve units are directly accessible for servicing and inspection.

(2) On each unit, three motor-operated Whittaker slide valves control the flow from the inboard and outboard tanks. One valve is mounted in the fuel line from each tank system and the third valve opens or closes the passage connecting the two systems. The three valves are inter-connected and are controlled by means of a four-position selector switch on the fuel panel in the flight deck. The switch and corresponding valve positions are shown diagrammatically on Fig. 52, which also indicates the two engine low-pressure fuel cocks and one of the cross-feed valves.

(3) A cross-feed valve is located at each end of a stainless steel pipe that passes through the fuselage and gives communication between the right and left wing tanks. The two cross-feed valves are motor-operated, both valves being controlled from one switch on the flight deck control panel.

(4) The low-pressure fuel cock for each engine is manually-operated so that the fuel supply to any engine can be shut off even if a complete electrical power failure occurs. In the event of an engine fire, these low-pressure cocks are closed to isolate the fuel supply from the engines concerned.

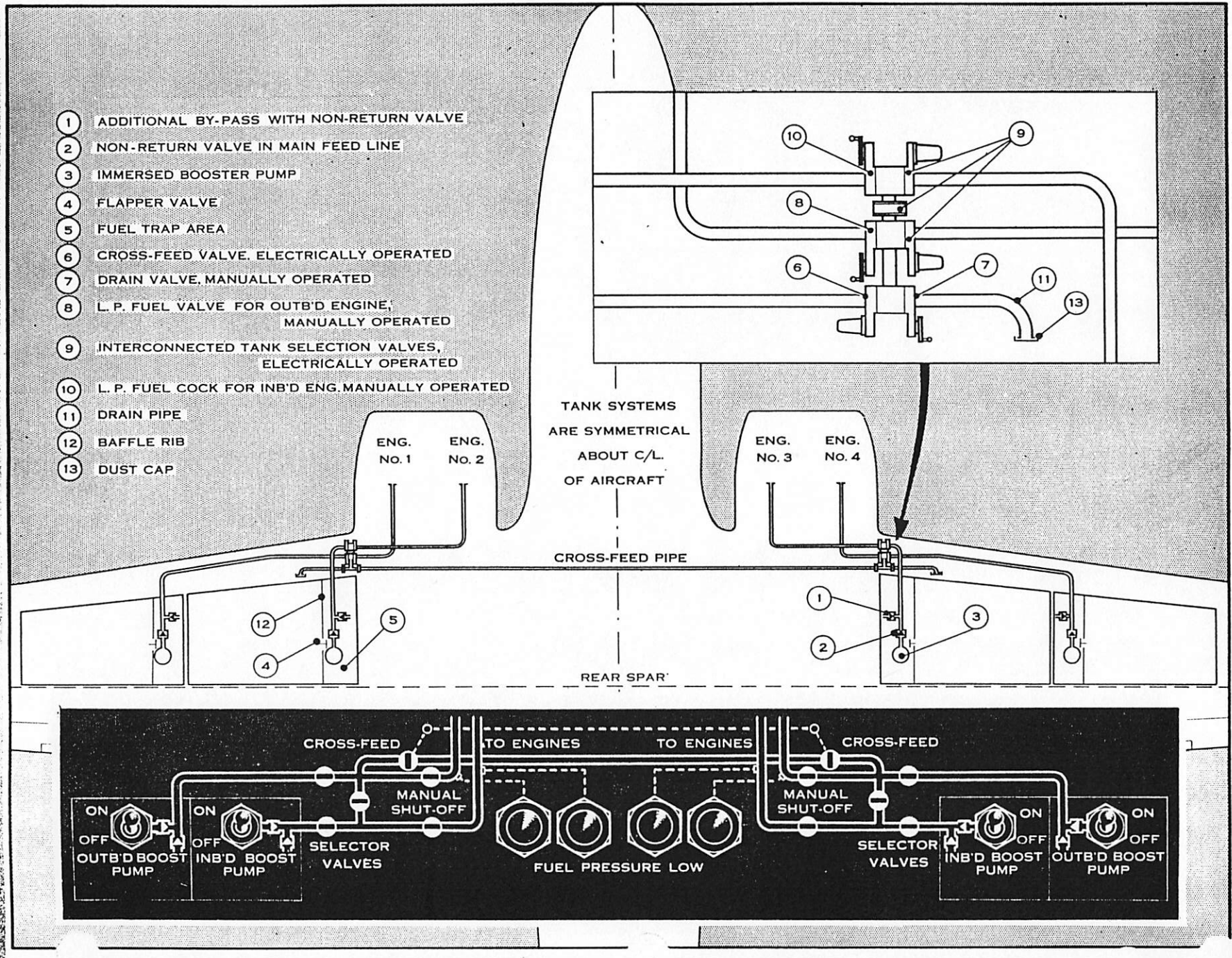
(5) A manually-operated drain valve is located at the base of each group of selector valves. These valves facilitate rapid draining of the system without disturbing any components or hose connections.

1-36 FUEL BOOSTER PUMPS:

(1) A Pesco electrically-operated booster pump is mounted at the inboard end of each wing tank. Its function is to provide a continuous supply of fuel under pressure to the engines. Each of these four pumps is controlled by an ON-OFF switch on a panel immediately above the engine throttles.

(2) In the event of booster pump failure, an adequate flow of fuel to the engines is maintained by the suction action of the high pressure engine-driven pumps.

(3) To allow fuel to flow freely from a tank without being restricted by an inoperative pump, a by-pass branch line incorporating a non-return valve is installed in the main supply line close to the pump. An additional non-return valve is fitted in the main feed pipe between the pump and the by-pass branch line of each tank. This arrangement allows fuel to flow only from the tank to the feed lines.



G.A. FUEL SYSTEM
FIG. 19

(2) The remaining tank ribs are of lattice construction. A centrally disposed rib in each tank has a baffle attached to prevent the surging of fuel. Baffles on ribs No.2 and 8 of each main plane, extend to approximately 50% of the rib height. This arrangement forms a fuel trap within the respective tanks. The baffles are riveted in position and incorporate water drain holes in their lower corners. In addition, each baffle incorporates a flap valve to ensure a sufficient supply of fuel to the booster pumps when the tank contents are low.

(3) All seams in the tank structures as well as the heads of all rivets are sealed with "Proseal" compound which is impervious to fuels and water. Access to the interior of each tank is gained through two detachable panels in the leading edge of the wing and two removable pressure-sealed panels in the front spar.

(4) Immersed booster pumps, one in each tank, are located on the lower wing skin on dished mounts and are covered with flush-fitting circular panels. The discharge pipes from the booster pumps are fitted inside the tanks and pass through the front spar to the wing nose section, where the fuel pipes and controls are installed.

(5) The booster pumps and additional bypass inlets are protected by coarse wire mesh, to prevent foreign matter from entering the fuel system.

(6) Due to the dihedral of the wing, the tanks take up a slanted position. A limited amount of fuel contained in the inboard portion of each tank, below the level of the booster pump inlet, cannot be drained through the pumps. This constitutes a tank sump. A water drain and a large drain plug are incorporated to facilitate the emptying of the tanks. These units are located at the lowest point of the sump.

(7) In the first aircraft where the underwing refueling system is not installed, the aircraft is refueled through overwing fillers located in the nose section of the wings. The filler wells are adequately drained and covered by a plate fitted flush with the wing surface. The position of the fillers is such that an expansion space in each tank of approximately 4% of the tank gross volume cannot be filled with fuel.

(8) Each tank is vented through a separate vent pipe of large diameter so situated that fuel cannot overflow through the vent pipe unless the aircraft is positioned in an abnormal attitude. The outlets of the vent pipe are located flush with the lower wing skin. To prevent syphoning of fuel through the vent pipes under certain flight conditions, an anti-splash box equipped with a flapper valve is installed in each tank at the inlet of the vent pipe.

(9) Four condenser units are fitted in each tank, one in each corner, as part of the "Pacitor" fuel measuring system. This system is described in Para.1-37.

CHAPTER 6

FUEL AND OIL SYSTEM

1-33 FUEL SYSTEM: (GENERAL)

(1) In the normal system, 2310.35 Imperial gallons of usable fuel are carried in four integral tanks, two located in each outer wing panel between the front and rear spars. For long-range operations, 1040 gallons of additional usable fuel can be carried in four tanks within the fuselage. These tanks utilize space normally available for passengers.

(2) The system is designed for flexibility of operation, so that normally, each tank supplies one engine through an entirely independent system of piping. The outboard engines are fed from the outboard tanks and the inboard engines from the inboard tanks. If necessary, however, any one engine, or pair of engines, can be fed from any tank by means of selector valves controlled by two 4-position switches in the cockpit and a separately-controlled cross-feed. (For further information refer to Para. 3-11).

(3) Fig. 19 shows the general arrangement of the system in simplified form. The booster pump in each tank pumps fuel, under pressure (3-15 p.s.i.) through a non-return valve to a block of selector valves and low-pressure cocks. From here, according to the position of the valves, fuel can be directed to either one or both of the adjacent engines and/or through the cross-feed to supply either one or both engines on the opposite side. This arrangement safeguards each individual engine against fuel starvation in the event of failure in its normal supply line. Any of the three remaining sources of supply can be selected at will. Failure of a booster pump will not cause the engines to be starved of fuel, as there is only a small pressure loss in the lines. The system will continue to function satisfactorily on suction from the high pressure engine-driven fuel pumps alone.

(4) Except for the manually-operated low-pressure cock for each engine, all equipment associated with the aircraft fuel system is electrically operated and controlled from panels in the cockpit. These are shown in Fig. 19 and described in paragraph 1-38. Further information on the main components in the system is given in the following paragraphs.

1-34 WING FUEL TANKS:

(1) The integral type fuel tanks, two in each wing, are located in the outer wing panels between the front and rear spars. Each inboard tank, of approximately 670 gallons fuel capacity extends from rib No. 1 to rib No. 7. Each outboard tank of approximately 506 gallons fuel capacity extends from rib No. 7 to rib No. 13. Ribs No. 1, 7 and 13 are developed as leak-proof bulkheads.

(4) Each booster pump is mounted on integral tank studs so that it can be easily removed without necessitating access to the interior of the tank. The latter must be completely drained before a pump is removed.

1-37 FUEL CONTENTS INDICATING SYSTEM:

(1) An electronic capacitor-type gauging system is used to measure the fuel contents of each tank continously and to register the amount in pounds on cockpit indicators.

(2) The basic principle of the Simmonds Pacitor system makes use of the change of electrical capacity of a condenser when the dielectric changes from liquid to air, these changes being recorded on a dial indicator. The fundamental circuit includes three elements: the tank unit, or condenser, which essentially comprises two tubular plates; the electronic power unit; and the cockpit indicator.

(3) The proportion of fuel or air between the two tubular plates of a tank unit directly affects the capacity of the unit. Hence the capacity of a unit is dependent on the fuel level at that particular location. As the fuel level varies, the changes in the electrical capacity of each unit are measured by a capacitance unit in terms of a small direct current. This small direct current is in turn fed to a conventional DC type ratiometer, or indicator, on the instrument panel.

(4) As there are four tank units in each tank, one mounted vertically in each corner, the integrated reading for any tank provides an indication of the contents accurate within 2% even during varying angles of incidence or bank. Fig. 19 shows the general arrangement of the system schematically, the cockpit indicator dials being shown on Fig. 6.

1-38 BOOSTER PUMP CONTROL PANEL:

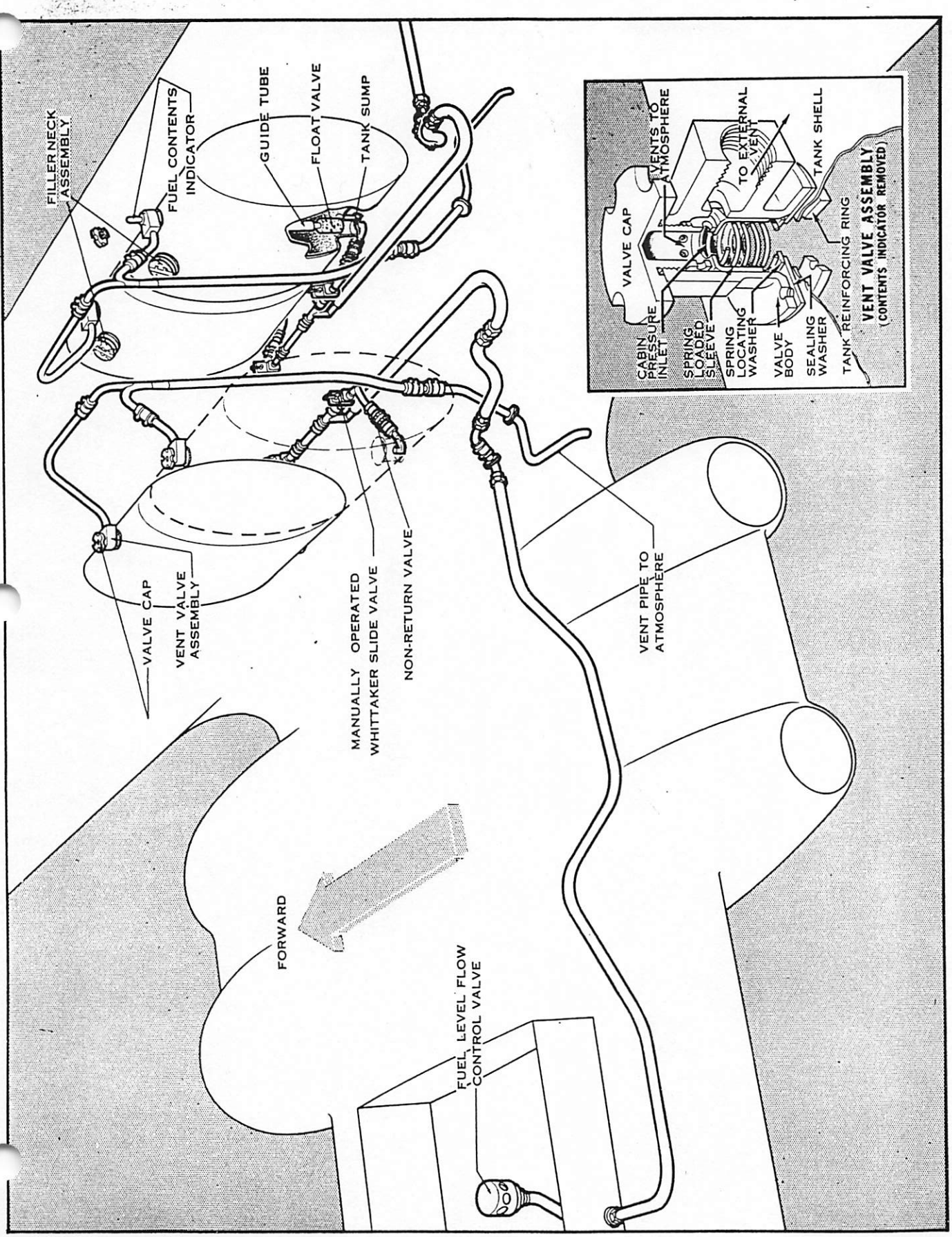
(1) The booster pump control panel (Fig.19) is mounted centrally above the engine instrument panel and carries the four booster pump switches and four low fuel pressure red warning lights. A simplified diagram of the complete fuel system between the tanks and engines is engraved on this panel.

(2) Two electrically-operated tank selector valve switches are situated immediately above the cross-feed switch on the throttle control pedestal. The four manually-operated low-pressure fuel cocks are located between the pilot's and co-pilot's throttles.

1-39 AUXILIARY FUSELAGE TANKS AND TRANSFER SYSTEM: (PROVISIONAL)

(1) The auxiliary fuel system (Fig. 20) comprises four additional long range aluminum alloy tanks and an independent piping service. The latter is arranged

AUXILIARY FUEL
SYSTEM
FIG. 20



below the cabin floor and is connected to the right and left inboard tanks of the main fuel system.

(2) Tanks are mounted in pairs on right and left cradle assemblies which are braced transversely and longitudinally by means of rods and cables tensioned by turnbuckles.

(3) Both assemblies are suspended from and bolted to fuselage formers at floor and roof locations. The bracketed inboard corners of each cradle assembly are bolted to formers 23 and 27 at approximately floor level. The inboard corners of each cradle incorporate attachments for two vertical tubes, the upper ends of which are attached to brackets bolted to formers 23 and 27 in the roof section of the fuselage.

(4) A filler assembly and an atmospheric vent assembly incorporating a spring loaded sleeve is fitted to each tank. The vents from each tank are interconnected through piping which passes to atmosphere in their respective wing fillets. The spring loaded sleeve is incorporated for the purpose of closing off the atmospheric vents when the system is being used. An integral dual-purpose fuel contents indicator comprising a float type shut-off valve and a guide rod and tube is fitted to each tank. When the tank filler cap is in position, the float valve seals the tank outlet, but when the cap is removed the float valve rises to approximately fuel level height and forces the dip stick to appear above the filler neck assembly.

(5) When the vent caps are removed from the selected tanks, the atmospheric vents will be automatically closed through the relative action of the spring loaded sleeve valves. Simultaneously, the fuel contents indicators with their relative shut-off valves will rise and allow fuel to flow to the manually operated Whittaker slide valves in each tank outlet. When the operating lever of the relative Whittaker valve is placed in the open position, fuel will flow through individual non-return valves to the main supply lines leading to the relative inboard wing tanks and their associated integral flow control valves.

(6) The flow control valve is designed so that fuel will not enter the wing tank from the auxiliary supply lines until the level has dropped to a certain height.

NOTE

(1) For information on the management of the auxiliary fuel system, refer to Para. 3-12.

1-40 OIL SYSTEM: (GENERAL)

(1) The lubrication system for each Derwent engine is of the dry sump type. A cast-magnesium tank mounted on the front of each wheelcase has a capacity

of 11 pints.

(2) From a gear-type pressure pump mounted on the base of the wheelcase, oil is fed to the wheelcase and main bearings through a "Purolotor" filter. Two scavenge gear-type pumps are mounted in line with and immediately above the pressure pump. Oil drained from the wheelcase and front main bearing is passed through a filter to the upper scavenge pump and returned to the tank. Oil from the centre and rear main bearings is similarly returned by the lower scavenge pump. Each of the three filters used in the system may be removed for cleaning purposes. The wheelcase breather is vented to atmosphere through the auxiliary gearbox hollow drive shaft. The negligible amount of oil which may escape past air-pressurized labyrinth glands on the shafts is drained to atmosphere through a pipe and passage in the combustion chamber drain valve. (For further particulars refer to the Derwent 5/17 Manual).

1-41 OIL GAUGES AND INDICATORS:

(1) Two dual reading oil pressure gauges and four low-pressure warning lamps are mounted near the top of the centrally disposed engine instrument panel. Each gauge is connected to two pressure transmitters, each one being located at the top of its corresponding engine wheelcase; the gauges provide continuous pressure readings for each engine oil system.

(2) The four warning lamps are arranged so that each one glows red when the pressure in its respective engine oil system falls below 5 p.s.i.

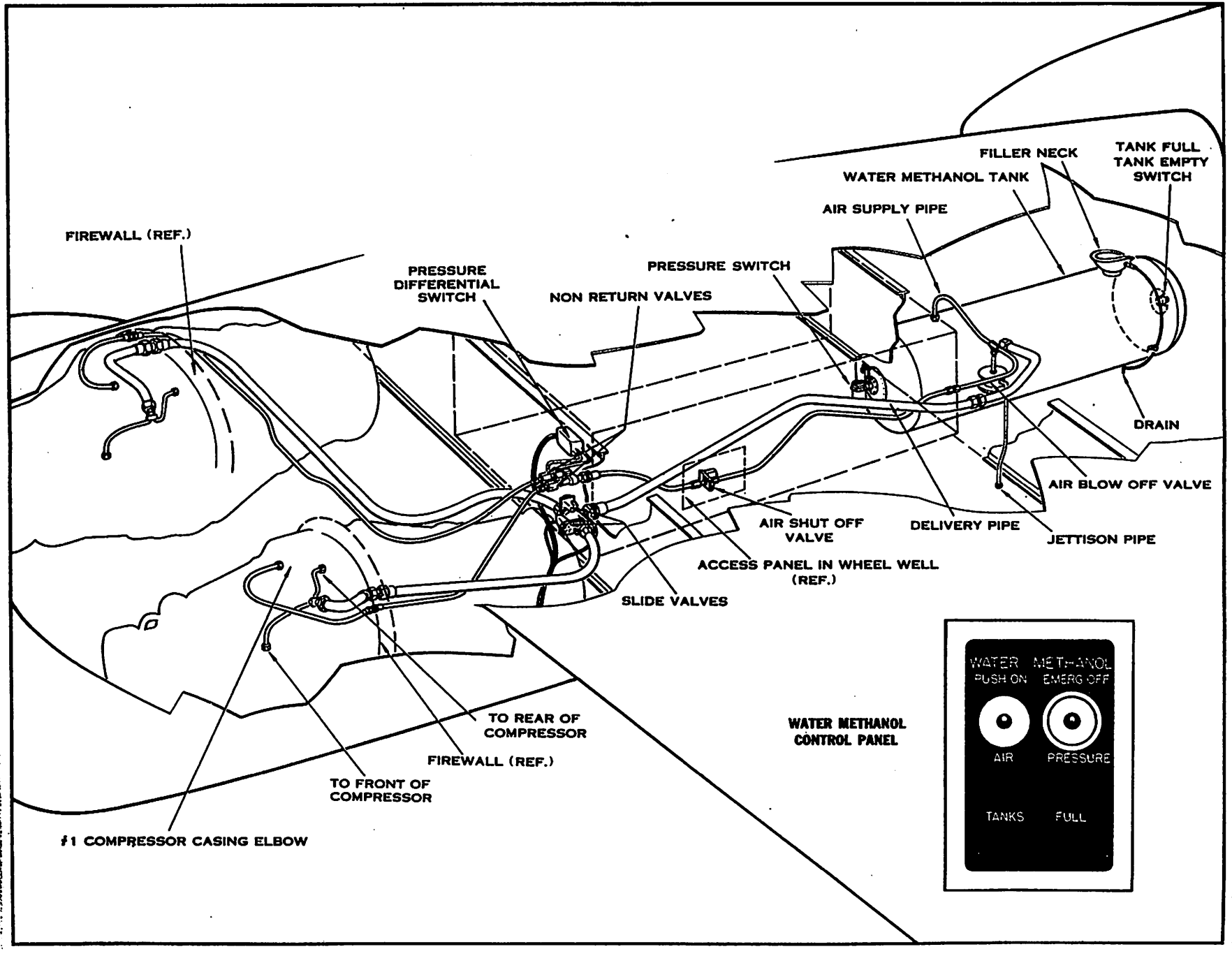
1-42 WATER METHANOL SYSTEM: (PROVISIONAL)

(1) The main purpose of the water methanol system is to compensate for the reduction of thrust caused through high ambient temperatures. It restores the thrust which would normally be available for take-off under standard I.C.A.N. conditions. The system should not be operated at temperatures below 20° C.

(2) Two separate installations are fitted. Each installation comprises an anodized aluminum alloy tank incorporating air pressure and float type fluid contents transmitting units. Individual tanks are pressurized from each engine compressor casing. Each tank has a capacity of 60 gallons and is suspended from cradles between the jet pipes.

(3) The system is controlled by a combination of switches mounted on the centrally disposed engine instrument panel in the flight deck, and by two sets of four microswitches incorporated in the throttle box.

(4) Two air pressure supply pipes (Fig. 21), one from each engine, are interconnected by means of a Cook pressure differential switch. The pipes supply air through individual non-return valves to a common supply line leading to the tank. Incorporated in the single air supply line is an air shut-off valve and a blow-off



WATER METHANOL SYSTEM
FIG. 21

valve. The air shut-off valve is controlled by the PUSH ON switch on the flight deck control panel, the blow-off valve is automatically controlled by line and tank pressure.

(5) Once the tank is pressurized, fluid is forced through the delivery pipe and branch lines, each of which incorporates a slide valve. These valves are electrically controlled by the micro-switches in the throttle control box. When all four throttles are in the FULLY OPEN position, the slide valves are electrically actuated and fluid is allowed to pass to the eye of each engine impeller.

(6) Should the air pressure drop in either of the related engines, a differential pressure is created between the separate air supply pipes. This differential pressure is immediately measured by the Cook differential relay switch which automatically stops the fluid delivery to the defective engine. Injection to the remaining engine is unaffected.

(7) The system normally operates at a pressure of between 12 to 15 p.s.i., any increase in pressure causes the blow-off valve in the air supply line to open and allow excessive air pressure to be dumped overboard through the jettison pipe.

(8) The control panel (Fig. 21) is centrally disposed on the engine instrument panel and comprises two switches and four indicating lights. The switch marked PUSH ON opens the air and fuel valves once the throttle controlled microswitches close the electrical circuit. The OFF switch is installed for the purpose of "closing off" the system. It is electrically interconnected with the air and fuel slide valves, and when pressed, closes all three valves in each installation irrespective of engine throttle setting. The AMBER air pressure lights are interconnected with the tank pressure transmitters, while the GREEN indicating lights are interconnected with the fluid contents transmitter. The latter are installed for the purpose of indicating that there is a sufficient quantity of fluid in the tanks for efficient operation of the system. The AMBER lights will function immediately operating pressure is obtained. When the tank contents fall below a predetermined level the GREEN lights will go OUT.

(9) The electrical circuit is arranged so that injection can take place only when the throttles are fully opened and the PUSH ON or master switch closes the circuit. The throttle levers must contact the forward set of micro-switches before the spring loaded PUSH ON button is pressed. This procedure must be strictly adhered to, otherwise the electrical circuits to the controlling valves would not be completed. The second set of throttle controlled micro-switches are arranged so that subsequent slight closing of any throttle lever will stop the fluid supply to that particular engine.

NOTE

(1) Normal thrust under prevailing conditions is increased by approximately 8% when the water methanol system is used.

(2) During injection, the jet pipe temperatures should not exceed 700°C.

is returned through the unpressurized return line to the reservoir, passing through the Vokes filter.

(4) Under normal operating conditions, the emergency and normal supply lines are pressurized at all times, so that in the event of normal system failure, the emergency supply will automatically take over through the piston valve in each wheel brake assembly.

(5) The emergency circuit to the aileron booster unit (37), comprises a branch supply line fed from the power pack (17) or the hand pump (12), the supply line passing aft along the fuselage below floor level to a non-return valve, where it enters the booster rotary control valve (33) situated aft of the rear spar in the fuselage centre section. Return fluid is passed from the booster unit directly back to the reservoir through the normal unpressurized return line.

1-58 EMERGENCY LANDING GEAR CIRCUIT:

(1) A branch of the emergency supply line connects with the hand pump (12) and power pack supply lines through the landing gear emergency control valve (14). The latter is located adjacent to the hand pump in the accessories compartment (Fig. 23).

(2) The emergency control valve comprises a single electro-magnetic unit with one inlet and two outlets, the flow of fluid being directed through the landing gear emergency pressure line, or by-passed directly through the normal pressurized return, the flow depending on valve setting. The landing gear side of the valve is normally open to return and emergency pressure cut off. When selecting DOWN the emergency pressure is directed to the landing gear down line. The valve setting may be controlled from the switch located on the pilot's throttle control pedestal, or from the operating lever on the valve housing. The emergency landing gear line is normally open to return when the valve is closed.

(3) The emergency supply line passes fluid from the control valve (14) to the main landing gear sequence valves (35) and the respective change over valves (3) and (25). The emergency supply lines leading to the nose landing gear and main door change over valves do not carry sequence valves. The latter are installed only in the main landing gear emergency supply line for the purpose of directing the initial supply of hydraulic fluid to the landing gear door change over valves. The doors are fully opened before the pressure builds up to 1800 p.s.i. and passes through the sequence valves and landing gear change over valves to lower the main landing gear wheel units.

NOTE

(1) The emergency landing gear valve incorporates a DOWN selection only.

1-59 EMERGENCY FLAP CIRCUIT:

(1) The emergency flap circuit is operated through the power pack (17) and comprises an emergency control valve (14), remotely controlled by a momentary contact switch on the flight deck throttle control pedestal. The control valve directs fluid through the flow divider (20) to the emergency flap UP and DOWN lines. Depending on switch selection, the fluid to the DOWN lines passing through a flow divider (20) which is incorporated to supply equal flow to the divided DOWN lines for the purpose of flap synchronization.

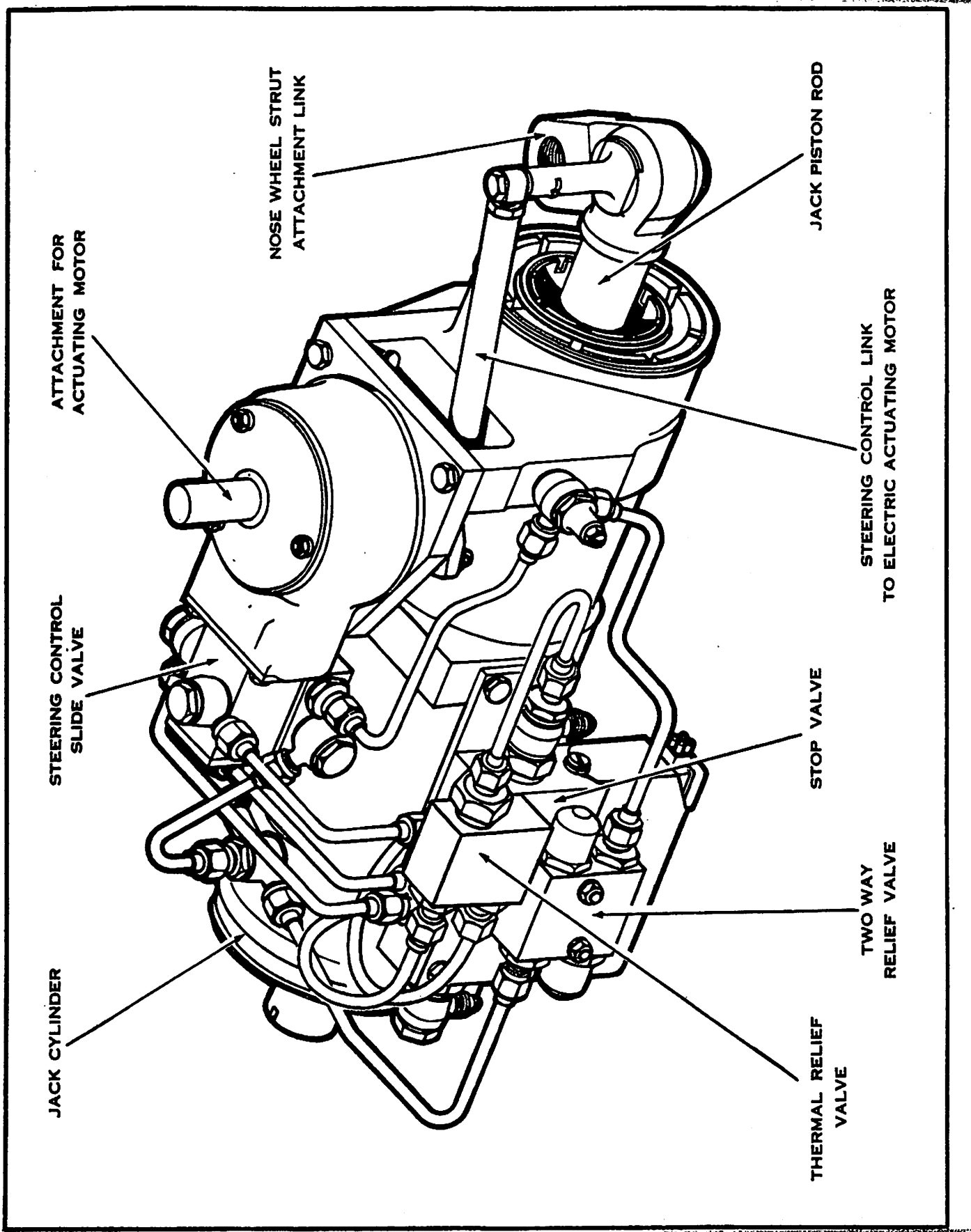
(2) The UP and DOWN lines pass from the flow divider (20) located in the right wing fillet, through the fuselage centre section and along the front spar of each wing to the right and left flap change over valves (25).

(3) Depending on flap selection, the emergency control valve (14) directs fluid under pressure through the emergency UP and DOWN lines, the alternative line acting as a pressure return to the emergency control valve (14).

(4) From the emergency control valve, fluid is returned to the pressurized return in the main hydraulic system.

NOSE WHEEL
STEERING MOTOR

FIG. 27



1-56 EMERGENCY POWER SUPPLY:

(1) An emergency supply of reserve fluid is maintained in the reservoir (16) below the level of the main circuit supply pipes, the emergency suction line branching to the hand pump (12) and the power pack (17), the line supplying each service with the necessary fluid.

(2) The feed line from the hand pump directs fluid to the two-way rotary control valve (13), one side of the valve directing fluid through a non-return valve to the main power circuit, the other line passing fluid through a non-return valve in the emergency pressure circuit to join the feed line from the power pack at the emergency landing gear control valve (14). Thus, through the medium of the rotary control valve, the hand pump can be used to direct fluid from the reservoir to the main system for test or emergency use, or to direct fluid from the feed line into the emergency pressure line.

(3) The power pack is energized through operation of the control switch located on the flight deck control pedestal. When the power pack is in operation, fluid is drawn from the reservoir and directed to the emergency pressure circuit, where it is maintained at a pressure of 2000 to 2250 p.s.i. by the unloading valve (18), excess pressure being by-passed to the pressurized return line. A small quantity of fluid also passes through the valve vent to the unpressurized return. Should the unloading valve fail, a pressure relief valve (9) installed between the pump and the unloading valve, by-passes fluid in excess of 2500 p.s.i. to the pressurized return circuit.

(4) A pressure accumulator (21) and transmitter unit (10) are also incorporated in the emergency pressure line, the former providing a reserve of pressure, sufficient to maintain emergency circuit pressure for a limited number of emergency operations. Fluid pressure is relayed from the transmitter to the dual pressure gauge located on the co-pilot's instrument panel.

1-57 EMERGENCY BRAKE AND AILERON BOOST CIRCUIT:

(1) This system is normally operated through the power pack supply, but pressure may be obtained from the hand pump (12) through the rotary control valve (13). Fluid enters the emergency brake power supply line through a non-return valve.

(2) The emergency brake power supply line incorporates two hydraulic pressure accumulators (9) located on the left side of the nosewheel well, and a pressure transmitter (10) interconnected with the dual brake pressure indicator on the co-pilot's instrument panel. Fluid passes from the supply line to the brake valve (11), where it is directed through the emergency brake pressure indicator on the co-pilot's instrument panel.

(3) Fluid passes from the supply line to the brake valve (11), where it is directed through the emergency brake pressure lines to the anti-skid solenoid valves (39) and the main wheel brake units (38). From the brake units, fluid

and isolate the system from the power circuit.

(3) Fluid displaced from the UP or DOWN side of the jacks, (depending on selection), returns through the change over valves (25) to the pressurized return line and to the reservoir through the back pressure valve and the Vokes filter.

1-53 NOSE WHEEL STEERING:

(1) The steering system is electrically controlled from the pilot's instrument panel. Steering controls, comprise a master ON-OFF switch, an indicator light and a Barber-Colman signaller steering control wheel with indicator, the latter being electrically connected to a Barber-Colman actuator on the nose wheel steering unit assembly. The master switch is connected in series with the nose safety micro-switch and the landing gear DOWN lock micro-switch, to eliminate the possibility of the nosewheel turning when in the retracted position.

(2) When the master switch is placed in the ON position, the indicator light should register, thus indicating the engagement of the actuating motor electro-magnetic clutch assembly with that of the steering valve.

(3) If the light fails to register with the master switch ON and the landing gear in the DOWN position with the weight of the aircraft on the ground, the steering control unit is not engaged and the nose wheel assembly will continue to castor. By rotating the cockpit control wheel to right and left, the clutch will engage and the indicating light register, to show that the nose wheel steering control is engaged.

1-54 OPERATION:

(1) Hydraulic fluid passes through the three-way control valve (8) in the power circuit to the electro-magnetic stop valve (7) of the steering control unit (5).

(2) When the system is operated, hydraulic fluid passes to the nose wheel steering unit assembly (5) mounted on the rear face of the nose wheel strut. The assembly comprising five separate units, operates in the following manner:

(a) An electro-magnetic stop valve (7) controlled by the cockpit master switch allows hydraulic fluid from the main power circuit to enter the nosewheel steering system.

(b) An electrical actuator or motor (5) incorporating right and left field windings (direction of rotation being defined by the position of the cockpit steering wheel) directs the flow of hydraulic fluid through the steering control valve (6).

(c) A steering control slide type valve (6) incorporating a by-pass to the stop valve (7), directs the flow of hydraulic fluid to either side of the double acting hydraulic jack, the control valve being actuated through its associated linkage connected with the actuator motor (5).

(d) The two-way relief valve is incorporated to prevent excessive pressure rise in the hydraulic circuit between the steering valve (6) and the operating jack, the relief valve eliminating excessive loads on turning, and acting as an anti-shimmy device. Pressure exceeding 2400 p.s.i. plus or minus 100 lbs., is by-passed through the low pressure side of the relief valve, back to the integral relief valve contained in the steering control unit (6) and through the pressurized return system to the reservoir, through the relief valve and the Vokes filter.

(e) A thermal relief valve (40) is incorporated in the nose steering assembly to relieve excessive pressures caused through thermal expansion of the fluid. The valve is set to operate at 3,000 p.s.i. plus or minus 100 lbs., excess fluid being by-passed to the return line of the main hydraulic system. The valve operates only when the steering mechanism is inoperative.

(3) The steering jack or motor is a normal two-way operating jack (Fig. 27), the piston rod being linked to the rocker and connecting rod linkage of the nose wheel strut. An additional link passing to the steering control linkage and the actuator motor, returns the steering control slide valve to the neutral position when the desired setting is reached. The linkage converts the reciprocating action of the jack, to a rotary movement of the nosewheel pivot bracket. The steering jack is spring loaded to the neutral position, to provide automatic self centering, and to supplement the anti-shimmy damping provided by the by-pass valve. The latter is an integral part of the steering control slide valve.

1-55 EMERGENCY SYSTEMS: (GENERAL)

(1) The following paragraphs describe the emergency operation for each individual hydraulic service.

(2) A separate emergency pressure circuit is provided, the power being supplied through an electro-hydraulic power pack (17) mounted on the auxiliaries panel in the right wing fillet (Fig. 25), or through a hand pump system, operated from the bottom left corner of the accessories compartment forward bulkhead. A detachable pump handle (Fig. 23) stowed adjacent to the hydraulic reservoir provides the means of operation. In addition, a further emergency is supplied for the lowering of the landing gear only. This is a manual release and gravity extension system, depending on gravity and drag for operation.

1-50 AILERON CONTROL BOOSTER:

(1) A power assist booster (37), manufactured by the Fairey Aviation Company, is incorporated in the aileron control system.

(2) The unit comprises a double-acting hydraulic jack integral with the control system, the jack being located aft of the rear spar in the fuselage centre section below floor level. The jack is interconnected with the aileron push-pull rods, and is hydraulically controlled through a sensitive slide valve, the latter directing fluid to the appropriate side of the jack as determined by movement of the pilot's controls. Control movement is assisted at a 5 to 1 ratio by the booster unit.

(3) Hydraulic fluid from the main power circuit is directed through a pressure reducing valve (24) assembly, incorporating a non-return valve. The pressure reducing valve is set to pass fluid at a maximum of 2300 p.s.i. to the booster control pressure line, where a pressure accumulator (23) is installed. The accumulator (23) and pressure reducing valve (24) are integral with the hydraulic power circuit, and are located in the left wing fillet.

(4) From the pressure reducing valve (24) fluid is piped through the fuselage centre section and a line fuse (32) to the rotary control valve (33), where it is directed to the aileron booster unit (37). These units are located aft of the rear spar below floor level.

(5) The line fuse (32) automatically prevents excessive fluid loss from the emergency system should the booster line break between the fuse (32) and the main accumulator (23). In the event of normal system failure, fluid loss is prevented by the non-return effect of the pressure reducer (24).

(6) The rotary control valve (33), remotely controlled from the cockpit through a Teleflex control, provides a means of isolating the booster unit (37) from the hydraulic system. It is used as an ON-OFF control. With the rotary valve (33) in the OFF position the booster unit remains inoperative while fluid is retained in the accumulator and the booster pressure line. With the valve (33) in the ON position, fluid is directed through the valve to the booster control unit.

(7) The accumulator (23) maintains and provides hydraulic pressure for several control operations in the event of power circuit failure or drop in system pressure due to diversion of fluid pressure to other services. Return circuit fluid is passed from the booster unit to the unpressurized return line, and returned to the reservoir through the Vokes filter, by-passing the back pressure relief valve.

1-51 FLAP SYSTEM: (GENERAL)

(1) The flaps are controlled electrically and operated hydraulically through a combination of two actuator switches and a pre-selector potentiometer. The

units are located on the throttle control pedestal.

(2) Control is through four drum switch potentiometers, the latter controlling the flow of fluid through the electro-hydraulic control valves (4) and through the change over valves (25) to each relative flap operating jack (28) and (29). When the pre-selector switch is positioned to the flap angle required and the relative actuating switch is in the ON position, the current is directed to the pre-selector potentiometer and the drum switch synchronous potentiometers. From the latter, the current is directed to the UP or DOWN magnetic valves which are housed in their respective hydraulic control valve assemblies (4). From the hydraulic control valves, the fluid passes to the individual change over valves (25) and operating jacks (28) and (29) of each individual flap. Flap synchronization is controlled electrically to within a degree, by the synchronous potentiometers on the four individual drum switches.

(3) In addition to the actuator switches and the pre-selector potentiometer, an emergency Bendix overide valve (34) is incorporated to maintain outboard flap synchronization in the event of hydraulic or electrical failure. Operation of the overide valve (34) is automatically controlled by a Teleflex cable balance, the latter comprising two cable assemblies supported in conduit. Each assembly passes outboard from the valve along the trailing edge of the wings to their respective outboard flaps. The outer end of each cable is attached to the root end of its respective flap operating tube, the inboard end of the cables terminating at a sliding bracket located over the valve actuating rod.

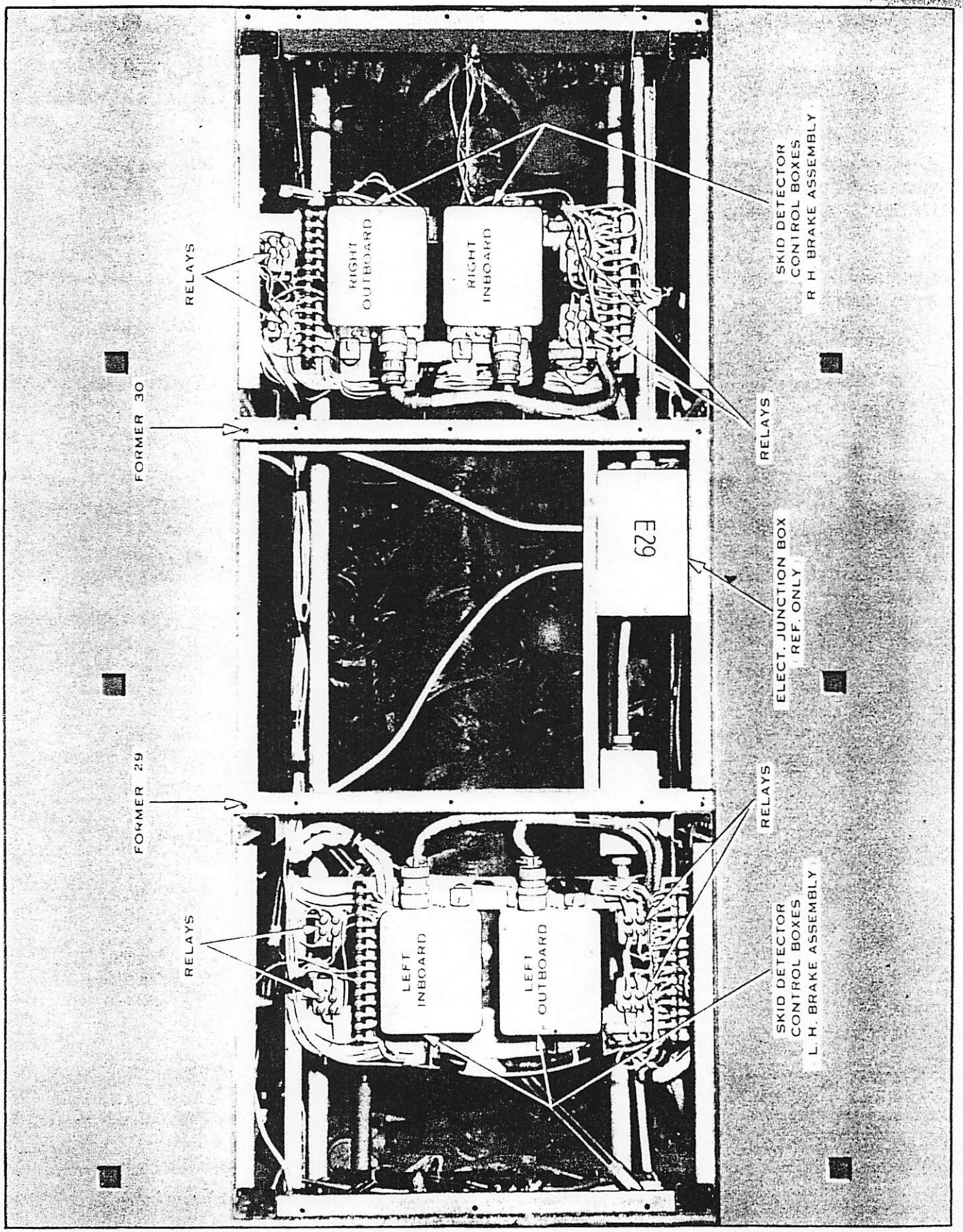
(4) The Bendix overide valve is essentially a three-way pressure balance valve, the valve controlling fluid pressure variations between the two outboard flap jack DOWN lines. Any variation between outboard flap settings, is corrected through the pressure balance lines communication with the overide valve, excess fluid from the lower flap supply line being passed from the valve (34) to the unpressurized return circuit of the main hydraulic system. The positioning of the valve ports is controlled by the Teleflex follow-up system described in sub-paragraph 3.

1-52 FLAP OPERATION:

(1) Fluid from the main hydraulic power circuit passes through a restrictor to the selector valves (4). From the selector valves, fluid is fed to the UP or DOWN lines of the change over valves (25) and their relative actuating jacks (28) and (29). The restrictor valve (31) located in the power circuit, restricts the flow of fluid to the selector valve for the purpose of slowing down extension or retracting sequences. Two additional restrictor valves (31), one in each of the UP and DOWN lines of the inboard flap jacks and one in the UP line of each outer wing flap jack, govern the rate of flap travel and assist in the synchronization of their respective services. A non-return valve is fitted to the pressure bypass in the UP line of each outer wing jack.

(2) When the flaps reach the selected angle, the potentiometers controlled by the pre-selector, break the electrical circuits through the control valves (4)

SKID DETECTOR
CONTROL BOX
FIG. 26



through the solenoid valves, is returned to the suction lines of the main hydraulic system.

(5) Normal braking action is carried out by the pilot regardless of whether the anti-skid control is operating. The anti-skid assembly is incorporated into the brake system for protective purposes only. Consequently, failure of the assembly does not affect the pilot's normal braking action.

(6) The indicating and control panel is located adjacent to the pilot's instrument panel on the left side of the flight deck. The panel carries four GREEN operating lights, four RED warning lights, and a master ON-OFF switch. The operating characteristics of each brake assembly is indicated by a relative GREEN and RED light.

(7) Each RED warning light indicates a malfunctioning or failure of a particular detector unit. When all four RED lights are ON the MASTER SWITCH will be OFF.

(8) The purpose of the GREEN operating lights is to indicate when the brake pressure is being released and/or re-applied by the anti-skid mechanism. These lights will flicker ON and OFF intermittently when the service is operating. (The period of illumination is only momentary.) Each light is electrically interconnected with its relative solenoid brake valve, and registers only when SKID CORRECTION is taking place.

(9) This service is armed only when the landing gear is in the DOWN position. Arming of the service is indicated by the illumination of the four GREEN indicating lights.

(10) Once the service is armed, the GREEN lights will remain ON until after the initial touch down on landing. Approximately FOUR seconds after the initial touch down, the landing gear wheels will have reached synchronous speed and the service will be fully armed. When the brakes are applied, the GREEN lights will flicker OFF and ON intermittently to indicate that SKID CORRECTIONS are taking place.

1-49 PARKING BRAKE:

(1) The parking brake is controlled through a spring loaded push-pull lever located on the left rear corner of the throttle control pedestal. The lever is connected to the brake control valve (11) through a linkage.

(2) The brake is placed in the ON position, by pulling the lever vertically upwards, the upward movement of the lever opening the control valve ports and allowing fluid pressure to be transmitted through each brake supply line to the individual wheel brake assemblies (38).

selector valves.

1-46 LANDING GEAR DOWN SELECTION:

(1) The landing gear lowering sequence is selected through the DOWN switch on the throttle control pedestal. On DOWN selection, the door unit of the twin electro-magnetic selector valves of the main landing gear doors, and the single electro-magnetic selector valve of the nose landing gear unit are energized and the electrical circuits completed.

(2) Hydraulic pressure from the power circuit is then directed through the change-over valves (25) to the DOWN side of the door jacks (26). When the doors are fully opened, the micro-switches are tripped to complete the electrical circuits to the selector valves (4) of the landing gear assembly. Hydraulic fluid from the power circuit is then directed through the landing gear change-over valves (25) to the DOWN side of the landing gear jacks (27). Restrictor valves (30), (excluding the nose assembly) are located in the UP supply lines to the jacks (27). These valves (30) restrict the flow of exhaust fluid to prevent the landing gear dropping too rapidly.

(3) Each change-over valve (25) incorporates a bleed to allow fluid returned from the UP side of the jack (27) to be diverted to the pressure side of the circuit. This valve (25) facilitates the extension sequence and reduces the amount of fluid required from the power circuit. This condition applies to the extension sequence only.

1-47 BRAKE SYSTEM:

(1) The brake system is designed so that the left and right pedals control the operation of their respective wheel brake assemblies through an automatic anti-skid control system. The dual brake pedals are toe actuated, dual movement being transmitted through a series of levers and cross tubes to the Dunlop brake control valve (11), the latter being located below the flight deck floor adjacent to the control pedestal. Pedal movement opens and closes the relative ports within the control valve (11), and directs hydraulic fluid under pressure through the aluminum alloy pipe lines and their relative dual flexible hose connections. These lines provide both normal and emergency system fluid to the brake piston assemblies located in the main wheel hubs. Operation is governed by the circuit delivering the greater pressure, the latter actuating the brake piston assembly (38). Brake pressure is proportionate to the amount of travel and pedal force exerted. This arrangement provides a dual brake system for both normal or emergency operation.

(2) The brake system receives fluid from the main power circuit at 2100 p.s.i. through a non-return valve located in the supply line, fluid pressure being maintained by the brake pressure accumulator (9). This pressure will supply sufficient power to provide at least 40 complete brake operations, independent of the normal power circuit, or emergency power pack supply.

(3) Two pressure transmitters (10) for registering normal and emergency fluid pressure between the accumulators and the Dunlop brake control valve (11) are located adjacent to the pressure accumulators (9) on the left vertical wall of the nosewheel well. Pressure from the transmitters (10) is directed to the dual gauge mounted on the co-pilot's instrument panel.

(4) Depending on the travel of the brake pedals, fluid leaves the brake control valve (11) at pressures up to a maximum of 700 p.s.i. (under full travel) and passes through the normal and emergency supply lines, to the brake piston assemblies (38) in the main landing gear wheels.

(5) Each brake piston assembly comprises a housing incorporating a piston valve with a fixed and sliding head. Normal brake pressure forces the sliding head along the piston valve to the fixed head. Should emergency pressure be greater, fluid will force the sliding valve away from the fixed head. Therefore, depending on the greater pressure, fluid will pass from either the emergency or normal lines and exert pressure against the fixed head. The latter controls the movement of the two plate, double spot disc of the wheel unit.

(6) When the brakes are released, return fluid from the system passes from the brake lines through the control valve return ports, the unpressurized return circuit, and to the reservoir (16) through the Vokes filter (15).

1-48 ANTI-SKID BRAKE CONTROL:

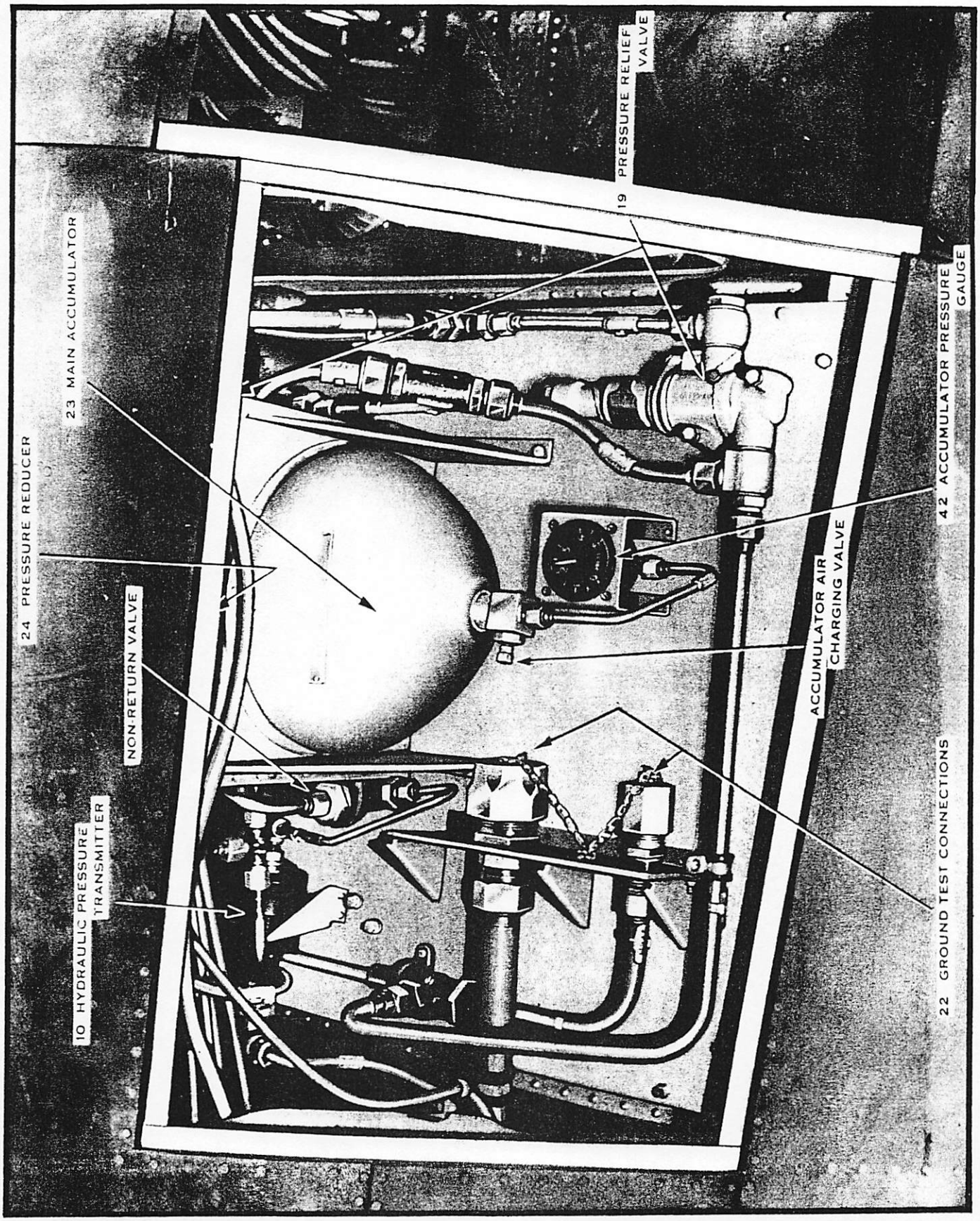
(1) The primary function of this system is to prevent the locking of a wheel in the event of excessive brake pressure being applied by the pilot during any phase of the landing roll. This requirement is accomplished through mechanical skid detectors and electronic controls which govern the action of the solenoid valves in the brake lines.

(2) A detector, mounted on the hub of each main wheel, operates on the inertia flywheel principle. Each detector consists essentially of a fixed assembly attached to the axle, and a rotating assembly which is located on the wheel. Depending on the relative motion between the two assemblies, a set of electrical contacts deliver either a skid or recovery signal to respective control boxes (Fig. 26).

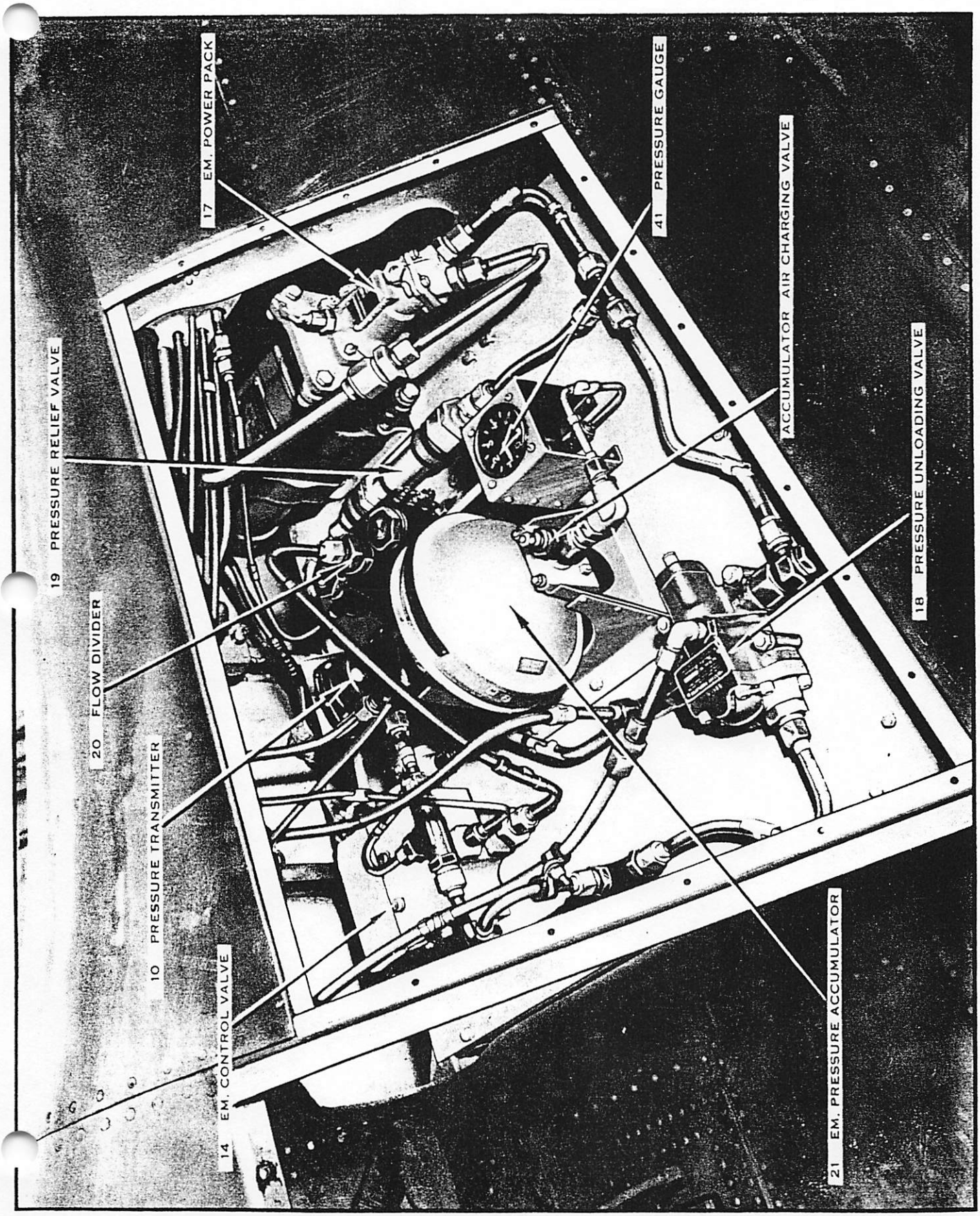
(3) Each control box comprises a number of relays and resistors which control the application of the skid and recovery signals to the solenoid valves (39) in both the main and emergency hydraulic brake lines (Fig. 22). The main and emergency solenoid valves in each wheel assembly are electrically interconnected, so that the relative brake units (38) on both inner and outer wheels are controlled by either skid detector.

(4) The solenoid valves (39) are essentially three-way valves which automatically control the release and re-application of hydraulic pressure to the main and emergency brake valve assemblies (38). Fluid by-passed from the brake lines

HYDRAULIC COMPONENTS
LEFT WING FILLET
FIG. 24



EMERGENCY HYDRAULIC
POWER SUPPLY PANEL
RIGHT WING FILLET
FIG. 25



HYDRAULIC SYSTEM
COMPONENT LAYOUT
FIG. 23

relief valves (19) by-passing pressure in excess of 2700 p.s.i. directly back to the reservoir (16) through the pressurized return circuit. From the relief valves (19) fluid is directed to the power circuit services, or through the pressurized return lines, where it is maintained at approximately 90 p.s.i. by an additional back pressure relief valve (19); the fluid then passing into the unpressurized return line through a Vokes filter(15) to the hydraulic reservoir. Return flow from the engine driven pumps, the brake control valve (11), and the aileron booster unit (37), is directed through the unpressurized return circuit to the filter and the reservoir, by-passing the back pressure relief valve.

(3) Two ground test 'quick disconnect' fittings (22) are located in the left wing fillet, one in the pressure line and one in the suction line of the power circuit. By connecting an external test rig.to these fittings, it is possible to ground test the complete hydraulic system.

(4) The engine driven pumps draw fluid from the reservoir through the suction lines, to supply the required services with a constant operating pressure. When the services have been operated, and no further pressure is required, the fluid enters the unpressurized return line and is fed directly back to the reservoir.

1-45 LANDING GEAR UP SELECTION:

(1) On selecting landing gear UP, electrical circuits are completed through the electro-magnetic selector valves (4), one for each main landing gear leg, and a single electro-magnetic selector valve for the nosewheel leg. As the appropriate coils in the selector valves are energized, the hydraulic fluid from the power circuit is directed through the selector valves (4) to the change over valves (25) and the UP side of the main wheel jacks (27). This movement places the landing gear in the UP and locked position. Fluid displaced from the DOWN side of the jacks (27) during an UP selection, is returned through the DOWN supply lines to their respective change-over and selector valves and to the pressurized return circuit.

(2) The main wheel landing gear doors are closed by separate hydraulic jacks (26). As the main landing gear reaches the UP and locked position, the relative micro-switches are actuated and the electrical circuits completed through the remaining section of the twin electro-magnetic selector valves. Hydraulic fluid from the power circuit, is then directed through the main landing gear door change-over valves (25) to the UP side of the door jacks (26). Fluid displaced from the DOWN supply lines flows to the change over valves (25) and the pressurized return circuit, then to the reservoir through the back pressure relief valve and the Vokes filter. When the doors reach the closed position, the micro-switches break the electrical circuits through the selector valves (4) and isolate the hydraulic power circuit from the landing gear system.

NOTE

(1) The change over valves are installed in the system so that emergency circuit operation of the landing gear may by-pass the electro-magnetic.

CHAPTER 7

HYDRAULIC SYSTEM

1-43 GENERAL:

(1) A high pressure, closed centre hydraulic system (Figures 22 and 23) provides power for operation of the following services:

(a) Landing Gear
(b) Flaps
(c) Aileron Booster
(d) Nose Wheel Steering
(e) Brakes

(2) Pressure is supplied by two constant pressure variable displacement, engine driven, Dowty Mk. 5 'Live Line' pumps operating at a normal working pressure of 1800 lbs. p.s.i. with a maximum cut-out pressure of 2100 p.s.i.

(3) The system is, in general, electro-hydraulically operated. There is a solenoid operated directional control valve for the operation of each jack. Actuation of the solenoid valves is accomplished electrically from the flight deck. This layout provides a minimum of high pressure hydraulic lines, particularly in the fuselage.

(4) Hydraulic fluid to Specification AN-O-366 is drawn from the reservoir by two hydraulic pumps, one pump being mounted on each auxiliary engine driven gear box. Both pumps feed a common power circuit, so that in the event of failure of one pump, the remaining pump can supply the necessary pressure to ensure efficient operation of the system.

(5) The hydraulic reservoir is located in the pressurized hydraulics compartment in the fuselage and gives a positive head on the pumps. The reservoir contains an emergency volume which is available to the electrically driven power pack and the hand pump only.

(6) The landing gear is electrically controlled from the flight deck by means of a twin push-button switch. The opening of the doors and the sequencing of the landing gear is accomplished electrically through micro-switch actuation of the solenoid selector valves.

(7) Flaps are operated by an electrical infinite-position selector lever and an actuator switch. Synchronization of the right and left side flaps is accomplished through position potentiometer control of the solenoid valves. The flaps are not mechanically interconnected across the fuselage.

(8) The nosewheel steering is electrically controlled from a switch and a steering control wheel on the pilot's instrument panel. The switch energizes a solenoid-operated stop valve which supplies hydraulic fluid to the steering motor unit.

(9) The aileron boost unit is integrated mechanically into the aileron control rod circuit, while the booster jack is operated through a follow-up type sensitive slide valve. The boost ratio is approximately 5 to 1 with the booster engaged. Disengagement is by means of mechanical control operating through a hydraulic shut off valve.

(10) A separate hydraulic emergency system, operated by an electrically driven pump, supplies fluid through an independent piping system to actuate the following services:

(a) Landing Gear Extension
(b) Flap Extension and Retraction
(c) Aileron Booster Control
(d) Brake Control

(11) The landing gear and flaps incorporate separate solenoid controlled selector valves for actuation of their circuits. These valves by-pass the normal system control valves.

(12) For the aileron booster, a separate line feeds directly into the unit from the Emergency System.

(13) Two cylinders of the quadruple brake valve are reserved for emergency operation. A separate feed supplies emergency fluid through the valve and duplicate lines to the actuating cylinders in the brake assembly. Duplicate pistons in the brake cylinders automatically provide for either normal or emergency operation.

(14) There are four accumulators in the system, one for the aileron booster, one for the power pack, and one each for the normal and emergency brake systems.

(15) All pressure lines are stainless steel with steel fittings, and all return lines are of aluminum alloy tubes and fittings.

1-44 POWER CIRCUIT:

(1) A hydraulic reservoir (16) comprising a cylindrical tank fitted with a filler cap assembly and a visual fluid level window, is located on the rear wall of the fuselage accessories compartment, and supplies fluid by a combination of gravity flow and pump suction to the Dowty 'Live Line' pumps (36).

(2) Fluid from the delivery side of the pumps is directed through non-return valves, past the two pressure relief valves housed in the left wing fillet; the

CHAPTER 8.

OPERATIONAL EQUIPMENT

1-60 AIR CONDITIONING & PRESSURIZATION SYSTEMS: (GENERAL)

(1) The air conditioning and pressurization units form a completely automatic system to supply fresh air at controlled temperatures and pressures under all conditions of flight. The system is automatically controlled from panels located on the right and left sides of the cockpit and in the attendant's compartment.

(2) When the system is in operation and the relevent control switches are selected, automatic control is governed by pressure regulation and thermostatic control. The pressure is controlled from the pressure control panel mounted on the left side of the flight deck, while thermostatic control is handled through the medium of the cabinstat, a series of cold and warm air ductstats strategically located throughout the air ducts, and an outside air temperature thermometer located on the left side of the fuselage nose section.

(3) The cabinstat, ductstats and thermometer (Fig. 28), transmit differential signals to a calibration unit housed on the accessories compartment inner wall. The calibrator acts as an electronic brain and operates the three control valves regulating the airflow through the system.

(4) Air enters the system from filtered or ram air intakes in each wing and is ducted to blowers mounted on the auxiliary gearboxes in the engine nacelles. Depending on cockpit selection, air from either blower is directed to a common supply duct in the fuselage, the air passing through either one or two coolers and refrigerating unit, or an electric heater, to the various discharge ducts and outlets located in crew compartments and in the cabin luggage racks. Provision is also made to include a supply of cold air, this supply being ducted from the coolers direct to controllable outlets (one for each passenger) in the base of the cabin luggage racks.

(5) The discharge valve assembly located in the rear fuselage below floor level provides an outlet for exhaust air.

(6) If the automatic system fails, an emergency system incorporating a dump valve, controls the release of cabin pressure, while two override switches have direct control over cabin heating. The dump valve is controlled from the flight deck. The override switches are mounted on the attendant's control panel.

(7) In general, the system is divided into three sections, Air Conditioning, Cabin Heating and Pressurization. Descriptions of each section are detailed in the following paragraphs.

1-61 AIR CONDITIONING:

(1) Two engine driven blowers (Fig.28) provide the necessary air flow for the system. The air is fed through the wing and fuselage ducting, the latter containing the necessary controls for automatic regulation and conditioning of air to the flight deck and cabin. A constant flow of approximately 60 pounds of air per minute is maintained at altitudes by automatic regulation, either blower maintaining this flow up to 13,500 feet. Above this altitude, both blowers are required to maintain the necessary flow.

(2) Each blower has a delivery capacity of approximately 90 pounds of air per minute at sea level and 30 pounds of air per minute at an altitude of 30,000 feet against a cabin differential of 4.5 p.s.i.

1-62 OPERATION:

(1) Filtered or ram air, depending on the position of either wing air intake change-over valve, is ducted from the wing intake to each cabin blower, the latter being located in their respective power plant auxiliary gear boxes.

(2) Air from the blowers flows through its respective silencers and spill valves, the latter incorporating safety valves for the jettisoning of back pressures in excess of 11.4 lbs. to atmosphere. Each spill valve is remotely controlled through its relative actuator by the mass flow controller located in the fuselage supply duct. From either spill valve, air at a regulated flow of approximately 60 lbs. per minute flows through either non-return valve, mass flow indicator and venturi measuring head, the latter providing operating pressures for the mass flow controller and for the flow indicators. The air then passes to a common fuselage intake duct incorporating the mass flow controller and a silencer, and to a primary intercooler on the right side of the fuselage.

(3) Depending on control setting, the flow is supplied directly to the fuselage heating ducts after initial temperature correction in the primary intercooler, or if necessary, diverted through the refrigerating unit and secondary intercooler on the left side of the fuselage to the silencer containing a water separator on the right side of the fuselage. Before passing into the fuselage heating and discharge ducts, the flow of air from the refrigerating unit and the water separator is brought to the desired temperature (if necessary) by means of an electric heater. From the exhaust side of the heater, the conditioned air flows through the main supply duct to wall outlets, the warm air heating the insulated air space between the outer and inner walls and discharging to the cabin through grilles incorporated in the right and left luggage racks. A separate manually operated mixing valve, remotely controlled by a Teleflex actuator on the co-pilot's side wall, regulates the flow of tempered air in the branch duct leading to the two controllable louvres in the roof on the flight deck, and to two outlets at floor level.

(4) To supplement the conditioned air from the luggage racks, a branch duct from the intake side of the heater is directed to each side of the fuselage and supplies cool air to the individually controllable louvres housed in the lower skin of the cabin luggage racks.

(5) After circulating through the cabin, air is exhausted through floor level grills in the cabin walls and discharged to the atmosphere through the discharge valve located in the bottom of the fuselage aft of the rear spar. An exhaust fan located in the electronics compartment cools the radio rack and creates a low pressure area in this particular location for the purpose of diverting air to below floor level.

1-63 AIR CONDITIONING PANEL: (GENERAL)

(1) This panel (Fig. 8) is located on the right side of the fuselage adjacent to the co-pilot's seat. The assembly incorporates three indicators and four toggle switches relative to the pressurization and air conditioning systems, and a variable panel light control and components relative to other services.

1-64 PANEL INSTRUMENTATION:

(1) The Weston temperature gauge Model 727 registers the discharge temperature of warm air from the C.G.E. thermocouple located in the calrod electric heater.

(2) A Weston air temperature direct reading instrument Model 728-342-2, indicates cockpit temperatures.

(3) A Smith Desynn position indicator is connected to the high pressure safety valve housed in the discharge valve assembly. This instrument indicates whether the safety valve is in the OPEN or CLOSED position.

1-65 PANEL CONTROL SWITCHES:

(1) The ON-OFF Master Switch controls the air conditioning power supply.

(2) The three-position safety valve switch controls the OPEN-OFF-CLOSED positions of the high pressure safety valve actuator.

(3) The protected ON-OFF switch controls the emergency dump solenoid valve actuator, safety and spill valve assemblies.

(4) The ON-OFF switch controls the operation of the radio compartment vent blower.

1-66 CABIN HEATING: (GENERAL)

(1) Cabin heat is thermostatically controlled by a cabinstat and manually adjusted by a temperature control selector, both units being located in the attendant's compartment (Fig. 10). Working in conjunction with the cabinstat

are two ductstats and an outside air temperature thermometer located in the left side of the nose section. The warm air ductstats are housed in the warm air supply duct, one in the vicinity of the heater outlet, and one at the discharge outlet of the primary intercooler (Fig. 28).

(2) Through the medium of the cabinstat and the outside air temperature thermometer, differential signals are transmitted through the electrical components housed in the calibration unit located on the accessories compartment inner wall; the calibrator relaying signals to the heater control amplifier governing cabin temperature change, and a motor amplifier located on the radio rack, the latter transmitting power to each of the three valve actuating motors.

(3) Operation of the three control valves is initially governed by the positioning of the primary cooler modulating valve, the valve shaft incorporating cams controlling the operation of a series of micro-switches, which in turn control identical micro-switch settings on the ON-OFF valves of the secondary intercooler and the refrigerating unit. (For further information on electrical operation, refer to paragraph 1-102).

1-67 TEMPERATURE CONTROL PANEL:

(1) This assembly is mounted on the left side of the fuselage in the attendant's compartment (Fig. 10) and comprises a panel incorporating two indicators, a temperature bulb, six toggle switches and a rotary control valve. The following text describes their respective functions in the system.

(2) The Kollsman altitude indicator Part No. 671CK-1-01 is a direct reading instrument indicating cabin altitude.

(3) A Weston temperature indicator Model 728, is a direct reading instrument indicating cabin temperature.

(4) Reading fore and aft, the first three switches control the lights in the front hall, the cabin, and on the attendant's panel. The three-position cooling switch, the emergency heat switch and emergency selector switch override the automatic electronic controls in the calibration unit.

(5) The Minneapolis-Honeywell temperature selector Part No. G-1087A is an electronic switch incorporating a rheostat and two cam operated switches. This selector allows for adjustment of cabin temperatures, by altering the setting of the cabin control amplifier.

1-68 PRESSURIZATION: (GENERAL)

(1) Air entering the sealed fuselage from the air conditioning system (Fig.28) causes an increase in pressure. This pressure, is automatically governed by the pressure controller, the latter regulating the air pressure and the rate of pressure change by controlling the action of the discharge valve.

(2) The pressure controller forms an integral part of the pressurization panel assembly located on the left side of the cockpit. The panel provides a base for the mounting of the master control unit, relative indicators and switches. The assembly is suitably labelled with instructions for the setting and operation of the pressure controller for both flight and ground test conditions. The following breakdown provides a detailed description including operation of components.

1-69 PRESSURE CONTROLLER: (GENERAL)

(1) The pressure controller is an automatic device for controlling cabin pressure by regulating the action of one or more discharge valves direct or through the medium of pressure relay valves. In this installation it consists of a master unit mounted, with the control shaft uppermost, on a control panel.

(2) The master unit is duplicated to provide additional security against complete failure of the controlling mechanisms. In this installation the pressure controller is set to operate in conjunction with one Type 30 discharge valve and is set during manufacture to control to a maximum differential pressure of 4.5 p.s.i.

(3) The control panel (Fig. 29) carries the graduated dial of an altitude selector and is labelled with instructions for setting and operating the pressure controller. Slots in the panel allow for the protrusion of the master unit hand indicator wheels and the rate of change scale. The altitude selector shaft, the rate of change selector shaft, and the master unit clutch buttons, also protrude through clearance holes in the panel.

(4) The pipelines from the master unit capsule chamber connections, are connected to the unions on the discharge valve, and pipelines from unions below the atmosphere chamber are led away to two separate static outlets.

(5) The pressure controller is rigidly mounted to the pressurization control panel, the latter being flexibly mounted to the structure on the left side of the flight deck.

NOTE

(1) The terms left and right, upper and lower, top and bottom, etc., refer to the unit as viewed in the diagrams.

(6) The master control unit consists of a rectangular light alloy casting divided into two capsule chambers, independent control mechanisms being built into each chamber. This duplication provides a safeguard against the failure of any one mechanism. An electric motor, a variable speed reduction gear, a camshaft mechanism and associated components are located on the exterior of the main casting. Drillings in the upper and lower surfaces of the chambers provide communication ports and locational housings for the control mechanism

details.

(7) Each control mechanism consists essentially of two sets of metal capsules, a beam mechanism, and an assembly incorporating a Schrader type outlet valve. As each chamber is a duplicate of the other, the description which follows applies to both.

(8) The absolute capsule pack (31) consisting of a set of metal capsules from which the air has been exhausted, is secured to and mounted above, a single differential capsule (30) which is open to outside atmosphere. The base of the lower capsule is soldered to an attachment bolt which screws into a solid central stem (27) and the latter is pinned to an inner guide (26).

(9) The inner guide fits into a cylindrical outer guide (28) which screws in a boss in the capsule chamber floor and is secured in position by a lock nut. A coil spring, positioned between the stem and the internal guide, locates between the bottom of the guide and the under face of a key plate. The key plate is riveted to a flange on top of the external guide and has an extension tongue which engages with a vertical slot in the inner guide and so prevents rotation of the capsule assembly when the latter is moved vertically.

(10) The capsule attachment bolt is drilled so that the interior of the single differential capsule communicates with the atmosphere through a capillary tube which leads away to the atmosphere chamber (20).

(11) The capsule end fitting (32) is provided with a knife edge which bears against one end of a beam (36). A stop (32) for limiting the upward travel of the right hand end of the beam passes through a central hole in the top of the end fitting, between the knife edges.

(12) The differential capsule assembly (43) consists of a set of intercommunicating metal capsules. Drilled attachment fittings provide communication between the differential capsules and the atmosphere through a union on the atmosphere chamber (20). An arrangement of spacer rods (42) guide plate (41) and spider plate (19) ensures the correct linear action of the capsule assembly when the latter compresses or extends.

(13) The differential capsule end fitting (39) passes through a hole in the left hand end of the beam and a stop (50) limits the travel of the beam at this end.

(14) An assembly containing an outlet valve (35) is screwed into the floor of the capsule chamber between the two capsule packs. The valve communicates to atmosphere through the atmosphere chamber (20) and is spring loaded in the direction of closure, so that its stem is held against the beam mechanism by the valve spring.

(15) The beam is a simple lever positioned by a beam spring assembly (37). One end of the spring is pinned to the centre of the beam, and the two curved arms locate at the top of the left hand spacer rods. The beam spring positions

(23) Two micro-switches (47) are fitted to the mounting plate (52) by means of adjustable micro-switch brackets (57). The switches are in circuit with the electric motor. The micro switch operating links (50 and 60) pivot on a common axis pin and are connected by a spring (59). A swinging link assembly (55) also pivots about the same point. The swinging link terminates at one end in a pin (56) which, in the central position, bears against each micro-switch link. At the other end is a roller which, in the off or central position of the swinging link, registers in a slot (51) in the cam (49).

(24) The cam (49) is secured to the end of the solid shaft (50) which protrudes through a clearance hole in the mounting plate. At the front end of the solid shaft is the control knob and pointer of the cabin altitude selector. A friction clamp (13) is fitted around the solid shaft, just behind the control knob, to preload the shaft and so prevent altitude selector settings being disturbed accidentally, or by vibration.

(25) Immediately to the right of the helical gear an extractor collar fits over the camshaft, a coil spring being located between the face of the collar and the end of the helical gear sleeve. The arms of an operating lever assembly (15) are attached to the extractor collar so that depression of the clutch button (16) causes the operating lever assembly to rotate on its axis and move the collar to the left and compress its spring.

(26) A pin carrier (22) fits over the camshaft, adjacent to the extractor collar and is secured to the shaft by a grub screw locked with wire. A locating pin (21) with a flanged head passes through the body of the pin carrier and engages with a flange of the extractor collar.

(27) A hollow shaft fits over the camshaft and has a cam clutch plate (23) and a cam (24) formed on it. The cam operates a rocker arm (25) fitted with a tappet screw which bears against the underside of the stem and guide of the absolute capsule assembly. A hole in the cam clutch plate (23) receives the end of the locating pin (21) so that the hollow shaft is locked to the camshaft.

(28) When the clutch button is pressed the collar slides along the shaft, the spring is compressed and the locking pin is withdrawn from the clutch plate. The hollow shaft, the cam, and the hand indicator wheel (18) can then be rotated independent of the camshaft when it is required to carry out Ground Testing. When the hole is re-aligned with the pin, the spring-loaded collar causes the pin to engage the clutch plate and lock the hollow shaft to the camshaft.

NOTE

(1) Prior to flight the clutches must be engaged, and must not be disengaged during flight.

(29) The hollow shaft is supported by the pillar (17) which acts as a bearing for it. The hand indicator wheel (18) fits against a wide flange on the

hollow shaft and is secured to the latter by a friction lock nut. The indicator wheel is graduated in thousands of feet, from 1,000 feet below sea level to 8,000 feet above. A position for "Ground Test" is also marked on the wheel. To the right of the hand wheel the mechanism is duplicated, except for the helical gear and switch and selector mechanism. A collar is provided to locate the extractor collar spring. A splined projection on the extreme right hand end of the shaft is provided when duplicated master units are used. A special flexible coupling is employed to connect the splined end to a bush on the second master unit, which replaces the helical gear sleeve.

1-70 OPERATION OF PRESSURE CONTROLLER:

(1) The following sub paragraphs and the schematic diagram (Fig. 29) describes the operation of the Pressure Controller.

(2) Air can enter the capsule chamber through the fixed orifice D in the restrictor body, and can pass to atmosphere through the outlet valve E. The setting of the outlet valve, (i.e. the size of the exit orifice,) governs the pressure in the capsule chamber. As this latter pressure is transmitted to the discharge valve bellows through K, the setting of the outlet valve regulates the resistance the discharge valve offers to the air discharging from the cabin. Alternatively the pressure is transmitted to a pressure relay valve through K.

(3) The setting of the outlet valve is controlled by sensitive capsules. The evacuated metal capsules of the pack B respond to changes in absolute pressure while the capsules of the pack A have their interior connected to atmosphere and are therefore sensitive to differential pressure. Capsule movement is transmitted to the outlet valve by the beam mechanism.

(4) The absolute capsule assembly can be preset so that the cabin pressure is kept constant, at a figure corresponding to any predetermined altitude between 0 and 8,000 feet, while the aircraft ascends or descends. Alternatively the setting of the absolute capsule assembly can be varied over a period of time so that the cabin pressure changes at a given and acceptable rate, while the aircraft climbs or descends at speeds in keeping with its performance and its operating requirements. It is also possible for the cabin pressure to be varied, in flight, while the aircraft maintains a constant altitude.

(5) When the pressure in the cabin is altered the differential pressure (between the cabin and the atmosphere) will vary between zero and the maximum permitted. When the maximum permitted differential pressure is reached, the differential capsule assembly prevents the maximum differential pressure being exceeded.

1-71 ACTION OF PRESSURE CONTROLLER SWITCH AND SELECTOR MECHANISM:

(1) When the altitude selector knob is turned to register the pointer with the required cabin altitude marked on the dial, the solid cross shaft (50) rotates inside the hollow geared shaft, and the slotted cam (49) being fixed to the end of

the solid shaft, rotates with it. The roller (54) of the swinging link assembly (55) is forced out of the slot (51), and rides on the circumference of the cam.

(2) The ejection of the roller from the cam slot causes the swinging link to pivot so that the pin (56) bears against one of the micro switch levers and pivots it about its axis, placing tension on the spring (59). The resultant pull on the other micro switch operating lever causes a micro switch operating plunger to be depressed by the pin on the end of the link so that the electric motor starts.

(3) The electric motor rotates the mushroom drive wheel (6) through a reduction gear. The mushroom drive wheel engages with a friction drive wheel (9) which in turn, meshes with a worm drive (8) to drive the camshaft. As the camshaft rotates, the cam (24) varies its lift, and raises or lowers the absolute capsule assembly. The hand indicator wheels rotate with the camshaft and the geared hollow cross shaft is rotated by its engagement with the helical gear on the camshaft.

(4) The direction of rotation of the motor and camshaft depends upon the direction in which the altitude control is rotated when the pointer is set to the required figure. If the control is set to increase the cabin altitude, the camshaft and the cam operate in the direction required to decrease the cam lift, and allow the absolute capsule pack to be lowered bodily, so that the beam causes the capsule chamber outlet valve opening to be increased. This reduces the master unit pressure acting on the pressure relay valve or on the discharge valve bellows direct, with the result that the cabin pressure is reduced. If the cabin altitude selector is set to decrease the cabin altitude, the camshaft and the cam operate in the direction required to increase the cam lift and causes the absolute capsule assembly to be raised bodily, so that the beam causes the outlet valve opening to be reduced in size. This increases the master unit pressure acting on the pressure relay valve or on the discharge valve bellows direct, with the result that the cabin pressure increases.

(5) While the cabin pressure is being reduced and the cabin altitude is being increased, a red indicator lamp is alight. While the cabin pressure is being increased and the cabin altitude is being decreased, a green indicator lamp is alight. When the camshaft and cams are varying the setting of the absolute capsule assemblies, the helical gear on the camshaft, engaging with the corresponding gear on the hollow cross shaft, causes the latter to rotate around the now stationary solid central shaft. The mounting plate (52), and the micro switch mechanism mounted upon it, rotate with the hollow shaft, and the roller of the swinging link assembly, rolls around the periphery of the slotted cam.

(6) Rotation of the hollow shaft and the micro switch mechanism continues until the roller on the swinging link reaches the cam slot, when the roller enters the slot under the transmitted influence of the spring (59). The swinging link and the micro switch levers can now centralize and pressure on the micro switch plunger is relieved. The motor circuit is broken, the camshaft mechanism stops and the indicating light goes out.

NOTE

(1) For further information on the micro-switch mechanism and the internal wiring diagram, refer to Para. 1-107 of this manual.

(7) The foregoing operation causes the absolute pack of the associated capsule chamber to be non-effective, and when air is introduced to the cabin by means of blowers, or by a Ground Test truck, the discharge valve resistance is sufficient to allow pressure to build up in the cabin. Should the cabin pressure tend to exceed the maximum permitted differential pressure, the outlet valve is opened by the action of the differential capsule pack, and the cabin pressure is kept constant at the maximum permitted value. It is thus possible to check the differential setting of each individual control mechanism of the master unit.

1-72 PRESSURIZATION PANEL:

(1) This panel incorporates switches controlling the pressurization system. It is mounted on the left side of the cockpit, adjacent to the pilot's seat, (Fig. 8).

(2) The first four switches control the setting of the spill valve assemblies.

(3) Manual positioning of the control switch overrides the setting of the sequencing and spill valve switches.

1-73 PANEL CONTROL SWITCHES:

(1) When the 'automatic-manual' control switch is in the 'manual' position, the coupled sequence switches provide a means of pressure selection from the right or left blowers. This switch assembly is for ground use only.

(2) The three-position spill valve switch controls the positioning of the spill valves when the control switch is in the 'manual' position.

(3) The air intake switch controls the positioning of each wing duct change over valve through its relative actuator. Depending on the position of the switch, 'filtered' or 'ram' air is passed to the system.

(4) The master altitude control switch directs the supply of electrical current to the pressure controller.

1-74 PANEL INSTRUMENTATION:

(1) The altitude control panel carries four indicators in addition to those contained in the pressure controller. These instruments serve the system in the following manner:

(2) The Kollsman rate of climb indicator type 716-K-011 registers vertical speed with an UP-DOWN range of 0-2000 feet per minute. This is a direct reading instrument.

(3) A Koolsman dual altimeter and differential pressure gauge is a direct reading sensitive altimeter type 906B-4-016. This instrument registers the differential pressure in pounds per square inch relative to cabin and aircraft altitude. This pressure differential gauge functions as a dual altimeter, the individual aneroid chambers being vented independently through individual static lines.

(4) A Weston air temperature indicator Model 327 Special Z-2, registers each blower discharge temperature through their respective C.C.E. thermocouples. These units are located in each wing duct adjacent to its respective blower.

(5) A Smith Desynn indicator part no. EX 481444, registers the air flow from the right and left superchargers through their respective flow indicators located in each wing duct.

1-75 A.S.I. SYSTEM:

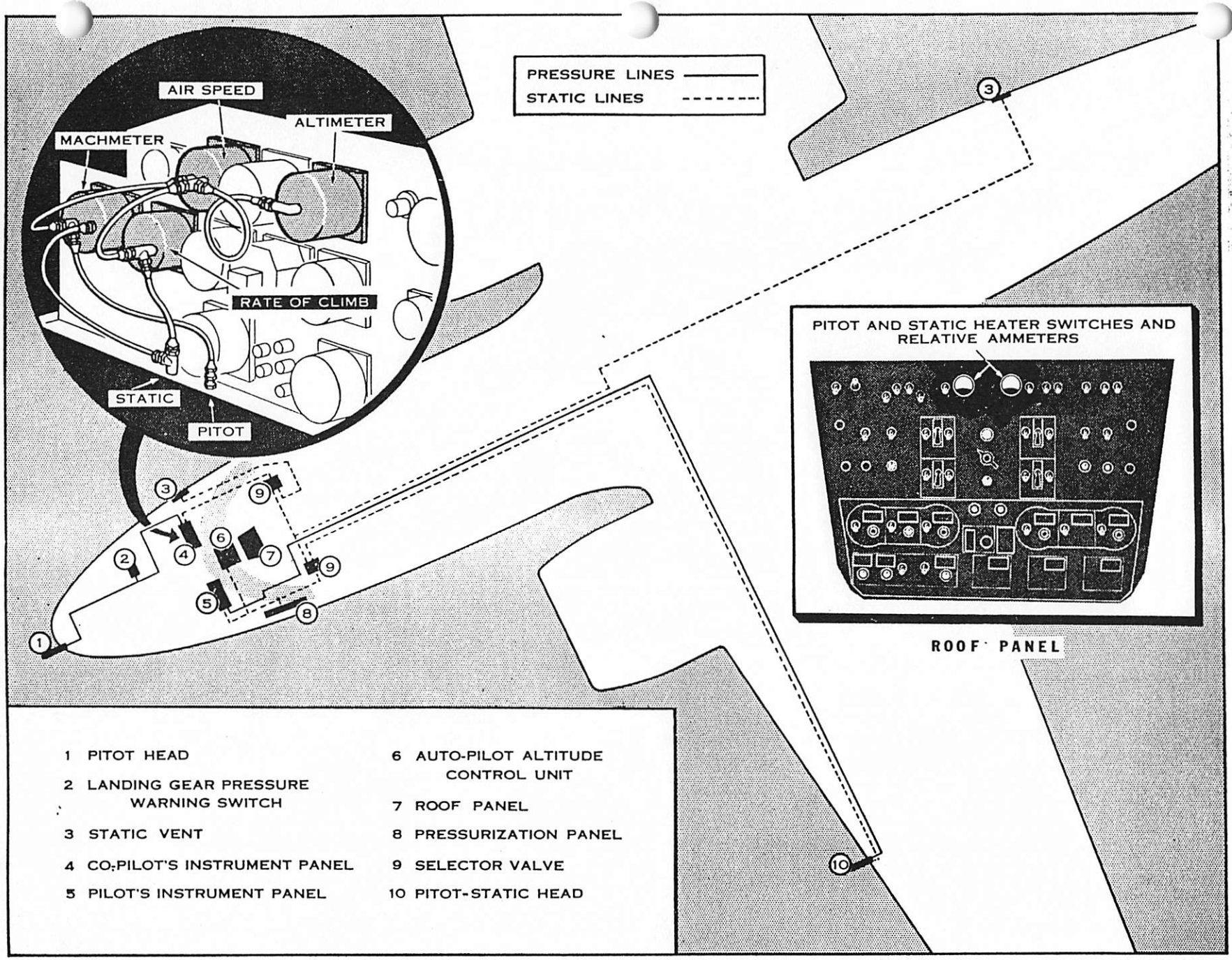
(1) Two airspeed indicators and a Machmeter are connected to the pressure heads and static lines of the system (Fig. 30). The system is designed so that the A.S.I. on the pilot's instrument panel is connected to a pressure head on the left wing, while the co-pilot's A.S.I. and a Machmeter are connected to a pressure head in the nose of the fuselage.

(2) The connecting pipes from the three static outlets are arranged so that each outlet may be connected to any one or both groups of instruments by means of selector valves and switches. The pilot's and co-pilot's selector switches are fitted to the side walls of the flight deck immediately aft of the main instrument panel.

(3) A static pressure line passes from the pressure head (Kollsman type 781-02) in the nose section of the fuselage and along the right side of the nosewheel bay. From the nosewheel bay the line passes through the pressurized forward bulkhead to terminate at the co-pilot's instrument panel. At this point a branch line is diverted to the landing gear pressure warning switch. (For information on the latter refer to Para. 1-25).

(4) The pitot-static head (Kollsman type 369-D-012) projects from the leading edge of the left wing and is connected to pressure and static lines attached to the face of the main spar. The pressure line passes to the pilot's instrument panel, while the static line terminates at the left side selector valve adjacent to the pilot's seat. (The pressure and static lines for the corresponding pitot-static head mounted on the right wing, have been 'capped off' inside the wing adjacent to the instrument).

(5) As shown diagrammatically (Fig. 30), pipes from two other static


PRESSURE LINES
STATIC LINES
AIR SPEED
ALTIMETER
MACHMETER
RATE OF CLIMB
STATIC
PITOT
PITOT AND STATIC HEATER SWITCHES AND RELATIVE AMMETERS
ROOF PANEL
1 PITOT HEAD
2 LANDING GEAR PRESSURE WARNING SWITCH
3 STATIC VENT
4 CO-PILOT'S INSTRUMENT PANEL
5 PILOT'S INSTRUMENT PANEL
6 AUTO-PILOT ALTITUDE CONTROL UNIT
7 ROOF PANEL
8 PRESSURIZATION PANEL
9 SELECTOR VALVE
10 PITOT-STATIC HEAD

outlets on the right side of the fuselage, one adjacent to the flight deck, the other just forward of the empennage, are connected through the selector valves to both the pilot's and co-pilot's instruments. The static lines serve the rate of climb indicators, the pressurization panel, auto pilot and altimeters as well as the relative A.S.I.'s and the Machmeter.

NOTE

(1) Pressure head covers must be removed prior to flight and replaced after landing.

(2) When icing conditions are prevalent, the pressure head and static heaters must be switched ON. These switches are centrally disposed on the upper portion of the roof control panel (Fig. 30).

1-76 AUTOMATIC PILOT:

(1) An Eclipse-Pioneer PB-10 Automatic Pilot system is installed in the aircraft to provide precise automatic flight control. The system is designed to maintain the aircraft on the selected magnetic heading and in the attitude of flight which applies at the instant of engagement. Longitudinal and lateral stability are automatically controlled. Trimming of the aircraft before engagement of the automatic pilot is not necessary, but is good practice.

(2) Manoeuvering is affected by means of the controller unit mounted on the pedestal (Fig. 31). The aircraft can be manoeuvred into a turn by moving the pistol-grip handle of the controller to the left or right. Correct bank for the turn is automatically introduced. The aircraft can be maintained in a dive, climb, or fixed angle of bank by operating the controller bank-trim and pitch-trim controls, and can be returned to the normal flight attitude when these controls are centered. When the pistol-grip handle is moved out of its detent, or its trigger grip is squeezed, the course control system is temporarily disconnected. This arrangement enables the aircraft to be set on a new heading with the auto-pilot still in control. This procedure is carried out by making a controller turn and by re-centering the pistol-grip handle at the desired heading.

(3) In addition to the basic installation, two refinements are included. One of these is an altitude control unit which automatically regulates the elevators to maintain the aircraft at a constant barometric pressure altitude. This arrangement does NOT fly the aircraft at a constant distance above the terrain over which it is passing but maintains it at a constant height relative to barometric pressure only. The other refinement is a trim tab servo unit which eliminates manual fore-and-aft trimming of the aircraft by automatic adjustment of the elevator trim tabs, whenever the necessity arises.

(4) The system consists of ten fundamental components, which are as follows:

(a) Gyro Flux Gate Transmitter

AUTOMATIC PILOT
FIG. 31

JETLINER Flight Manual
TYPE C-102

(b) Amplifier and Signal Generator
(c) Rudder Servo
(d) Elevator Servo
(e) Aileron Servo
(f) Trim Tab Servo
(g) Trim Tab Amplifier
(h) Master Direction Indicator (Flux Gate Compass Indicator)
(i) Pitch Trim Indicator
(j) Controller

NOTE

(1) In the following paragraphs, item (b) is referred to as the amplifier.

(2) In addition to items (a) to (j) there are four additional switches to control the complex functions of the equipment.

(5) The components work together as an "error-sensing" system which establishes flight references for the heading and attitude of the plane, senses any "errors" or deviation from these references, and applies control movements to keep the plane at the required heading and attitude. Except when the controller is in use, these functions are carried out entirely automatically. To maintain straight and level flight, a reference must be established for each axis of control, i.e., for yaw, roll and pitch.

(6) The earth's magnetic field is used as the reference for the yaw axis by means of the gyroscopically-mounted Flux Gate Transmitter which is located in the right wing tip. Impulses from this unit are transmitted to the Master Direction Indicator which converts them into dial movements to provide the human pilot with a constant visual indication of the magnetic heading of the aircraft. The same impulses, or signals, originate course signals in the Master Direction Indicator which transmits them to the amplifier. The amplifier, in turn, passes the modified signals to the servo motor which moves the rudder linkage in the desired direction and at the correct rate to apply the correct amount of rudder travel. No impulses or signals are originated by the Flux Gate transmitter unless there is a change in the heading of the aircraft. Consequently dial movement and rudder movement will only occur if air currents, or other forces, deflect the aircraft from the heading at which the automatic pilot was engaged. The Master Direction Indicator registers the magnetic heading of the aircraft even when the automatic pilot is not engaged.

(7) The reference for the roll and pitch axes, aileron and elevator control, are supplied by the Flight Gyro. This device, which is contained in the amplifier housing, is maintained in an upright position by gyroscopic inertia regardless of movements of the aircraft. If the plane is thrown off the established attitude, signals originated by the Flight Gyro are amplified and modified to form working signals which are in turn conveyed to the aileron and elevator servos. These servos apply control to return the aircraft automatically to the correct attitude in pitch and roll.

(8) It is important that the Flight Gyro should always be upright to provide the proper flight reference for the control concerned. A gyro beacon indicating light, located centrally at the rear base of the pedestal, is illuminated approximately once every 1-1/2 seconds, providing the gyro is erect and running at the correct speed. If for any reason the gyro is not erect, the beacon fails to glow and the pilot must operate the caging switch. The gyros will then be automatically caged to the erect position before being released. As a safety feature, the caging circuit includes an interlock, so that, if the caging switch is closed when the automatic pilot system is engaged, the clutch switch holding solenoid is energized. This arrangement releases the clutch switch, and provides for automatic disengagement of the auto-pilot system.

(9) To manoeuvre the aircraft when the auto-pilot is engaged, the human pilot must use the controller. By displacing the pistol-grip handle to the left or right precision turns can be made. Movement of the handle originates properly related signals for the rudder, elevators and ailerons to produce a co-ordinated level turn. Increased movement of the handle increases the rate of turn. To make small adjustments to the plane's heading, the trigger grip must first be squeezed. The aircraft is automatically held on the new heading when the handle is centered.

(10) To ascend or descend, the pilot rotates the pitch knob in the same way as a manual trimmer handwheel. Rotation of the bank knob to the left or right alters the angle of bank accordingly. As long as these controls are displaced from their centre position, the aircraft will continue the manoeuvre and will be automatically retained in the desired attitude of flight.

(11) To maintain the aircraft at a constant barometric pressure altitude, the altitude switch must be set to "ON". Expansion or contraction of an aneroid mounted in the amplifier then operates a device which originates an electric signal. This signal is applied to the elevator channel to return the plane to the reference barometric altitude at which the control was engaged. Unless there is a change in the barometric pressure, no movement of the aneroid takes place, consequently no signal is generated and no movement of the elevator takes place.

(12) The pitch trim indicator immediately registers the condition of the aircraft if fore-and-aft trimming is required. Automatic trimming to balance the aircraft correctly is accomplished by the trim tab servo installation. This servo, operating through a separate amplifier, responds to a sustained elevator signal which trims the aircraft in pitch and automatically cancels the signal.

(13) Disengagement of the automatic pilot can be accomplished by either electrical or mechanical means. Electrical release can be obtained by pressing the release button on the aileron handwheel, or by operating the mechanical manual release lever located on the side of the pedestal. The electrical release button disengages clutches in the servos and in the Master Direction Indicator and causes the clutch switch to disengage. The mechanical release lever disengages a separate clutch between each servo drive shaft and pulley.

1-77 NESA ANTI-ICING SYSTEM:

(1) The three transparent panels at the front of the windshield can be maintained at a constant controlled temperature by means of automatically-controlled electrical heating.

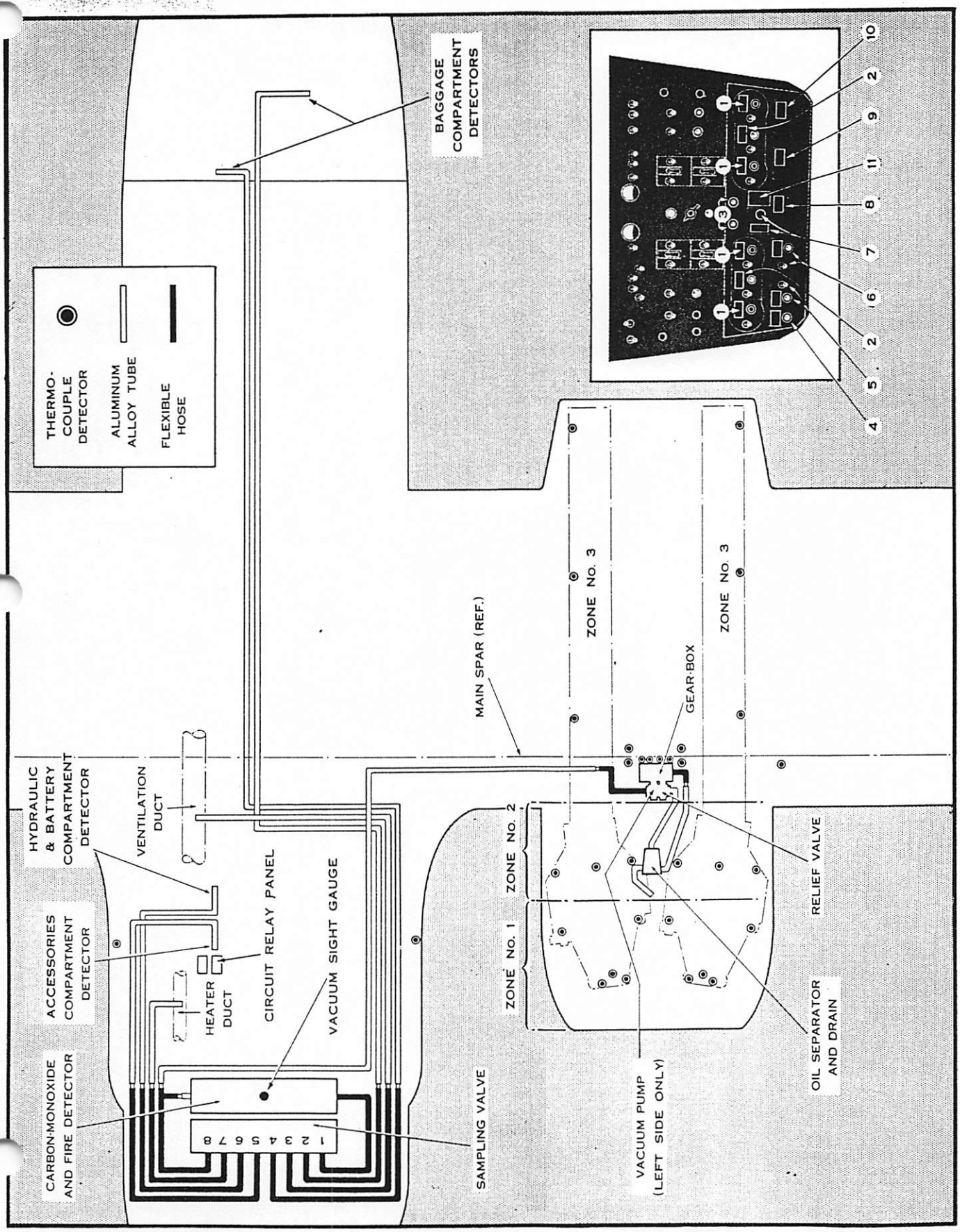
(2) Each panel is of laminated construction and consists of a vinyl core sandwiched between two layers of semi-tempered glass. The outer surface of the vinyl layer is coated with 'Nesa' compound, forming an electrical conductor which can absorb 3/4 to 3 watts per square inch. Terminals from the electrical power supply are located along the top and bottom of each panel. Sensing units are disposed in diagonal corners of the centre windshield panel. These units respond to temperature changes through the medium of an electronic amplifier, which maintains a constant temperature by regulating the current supplied.

(3) In addition to providing the required heat for de-misting and anti-icing, the system maintains the laminated panels in a condition offering maximum resistance to impact. If either of the three panels become severely damaged, the vinyl layers between the glass will expand to maintain the pressure seal. The vinyl layer is capable of withstanding more than twice the cabin differential pressure.

(4) When the windshield de-icing switch is set to the NORMAL HEAT position, an average windshield temperature of 90-95°F will be automatically maintained; providing normal climatic conditions exist. This temperature is sufficient to provide impact resistance and to keep the windshield perfectly clear, but will not necessarily prevent icing.

(5) Under icing conditions, the selector switch is set to EMERGENCY HEAT, so that the windshield temperature is increased to 120-140°F.

SCHEMATIC ARRANGEMENT
OF
EDISON AND M.S.A.
FIRE DETECTION SYSTEMS
FIG. 32



FIRE DETECTION AND EXTINGUISHING SYSTEMS

1-78 FIRE DETECTION SYSTEMS: (GENERAL)

(1) There are two fire detection systems incorporated in the aircraft (Fig. 32). These are the Edison rate of change thermocouple detector and the Mines Safety Appliance systems. Both installations incorporate visual and audible warning devices.

1-79 EDISON DETECTOR SYSTEM:

(1) The Edison rate of change detector system functions electrically through a series of thermocouples arranged throughout the engine bays and around the jet pipe installations (Zones 1,2 and 3). It also protects the engine driven auxiliary gear box compartments, the inner wing leading edges and the forward wing fillets.

(2) The thermocouple detectors are wired in series through a sensitive relay panel to relative dual indicating lights mounted in RED plastic boxes located on the lower half of the roof control panel (Fig. 34). The four circuit relay panel, is an enclosed unit located in the accessories compartment and comprises a sensitive relay, slave relay and a thermal test unit. The sensitive relay, consists of a moving element incorporating a small permanent magnet pivoted between two stationary field coils. Since the load carrying ability of the sensitive relay is limited, a slave relay is employed to permit operation of the warning devices. The thermal test unit consists of a thermocouple with two heating coils adjacent to its 'hot' junction. In order to secure uniformity of operation and to avoid the possibility of a fire hazard, these parts are sealed within a glass bulb.

(3) The system is automatically operated when an abnormal temperature rise occurs in the protected areas, the thermocouples generating an electrical current of sufficient strength to operate the sensitive relay and set off the alarm system.

(4) Two circuit test switches (Fig. 34), located in the lower section of the roof panel are installed for the purpose of making continuity checks and for the testing of warning lights and bells.

1-80 M.S.A. CO FIRE DETECTION SYSTEM:

(1) A Mines Safety Appliance fire warning and carbon monoxide detection system is vacuum operated and electrically controlled by a sampling valve located below the floor of the flight deck. The sampling valve has six suction pick-ups strategically located throughout the fuselage to analyze the carbon monoxide content of the atmosphere. These suction pick-ups take test samples from the accessories and hydraulic compartments from below the floor in the flight deck and cab-

in areas, from the ventilating ducts and the baggage compartment.

(2) The sampling valve is designed for continuous operation, the detector sampling air from each area for a period of 30 seconds. The time required to complete the cycle over the six protected areas is therefore three minutes, a delay of 2½ minutes taking place from the time the sampler leaves one area, until it returns to the same area.

(3) When the carbon monoxide concentration exceeds C.005%, a magnetic relay operates the respective warning lights on the control panel and the warning bell on the flight deck bulkhead. Both warnings remain in operation until the detector is manually reset by the depressing of the RESET button on the panel. As long as carbon monoxide is present, the warning devices will continue to operate, even after the RESET button has been pressed.

(4) The detector unit is contained in a shock mounted casing located below the floor of the flight deck. The case contains a sample analysis assembly and an alarm circuit panel assembly. The sample analysis assembly comprises a filter canister; a thermally insulated, thermostatically controlled sample heating chamber and a detector cell. An outlet from the latter is connected to the vacuum sight gauge visible through the sighting hole in the flight deck step. The alarm circuit panel assembly comprises a series of relays and other electrical components.

(5) The air sample from the six test locations in the fuselage (Fig. 32) is drawn by vacuum through the sampling lines to the filter canister in the sampling valve. The air is then filtered and heated and conveyed to the detector cell in the detector unit where a heat and chemical reaction takes place. If carbon monoxide is present in the air sample, a temperature increase will result and an electrical current will be generated and directed to the indicating meter and to the alarm circuit. When the expended air leaves the detector cell, it passes through the vacuum indicator and returns to the suction source.

NOTE

(1) A vacuum of at least 4 inches Hg. is required for correct operation of the M.S.A. fire detector.

(2) The vacuum pump operating the mechanism is driven from the left auxiliary gear box by No. 2 engine.

(3) The fire detection systems cannot be operated or checked until the MASTER fire control switch is in the ON position.

1-81 FIRE ALARM SYSTEM:

(1) The alarm system consists of a series of warning lights located on the flight deck roof panel, and a warning bell located adjacent to the electrical power supply panel on the flight deck bulkhead. The warning bell is interconnected with both the Edison and M.S.A. fire warning systems.

1-82 FIRE EXTINGUISHING SYSTEMS: (GENERAL)

(1) A series of selector switches and discharge button assemblies control the operation of the fire extinguishing systems (Fig. 33). The engine areas are protected by a Methyl Bromide system, while the gear box and accessories compartment are protected by means of a CO2 system.

1-83 METHYL BROMIDE SYSTEM:

(1) The Methyl Bromide system comprises four 36 lb. spherical bottles capable of supplying extinguishing fluid to their relative engine spray rings. (Fig. 32).

(2) The four bottles are arranged in pairs in the right and left nacelles and discharge through interconnected lines incorporating double check tees. This arrangement permits the bottles to be fired into either or both engines depending on cockpit selection.

(3) The individual three-position selector switches on the control panel determine whether bottle No. 1 or No. 2 is fired into the selected area. This system is so arranged, that two charges (one from each bottle) may be directed to separate engines.

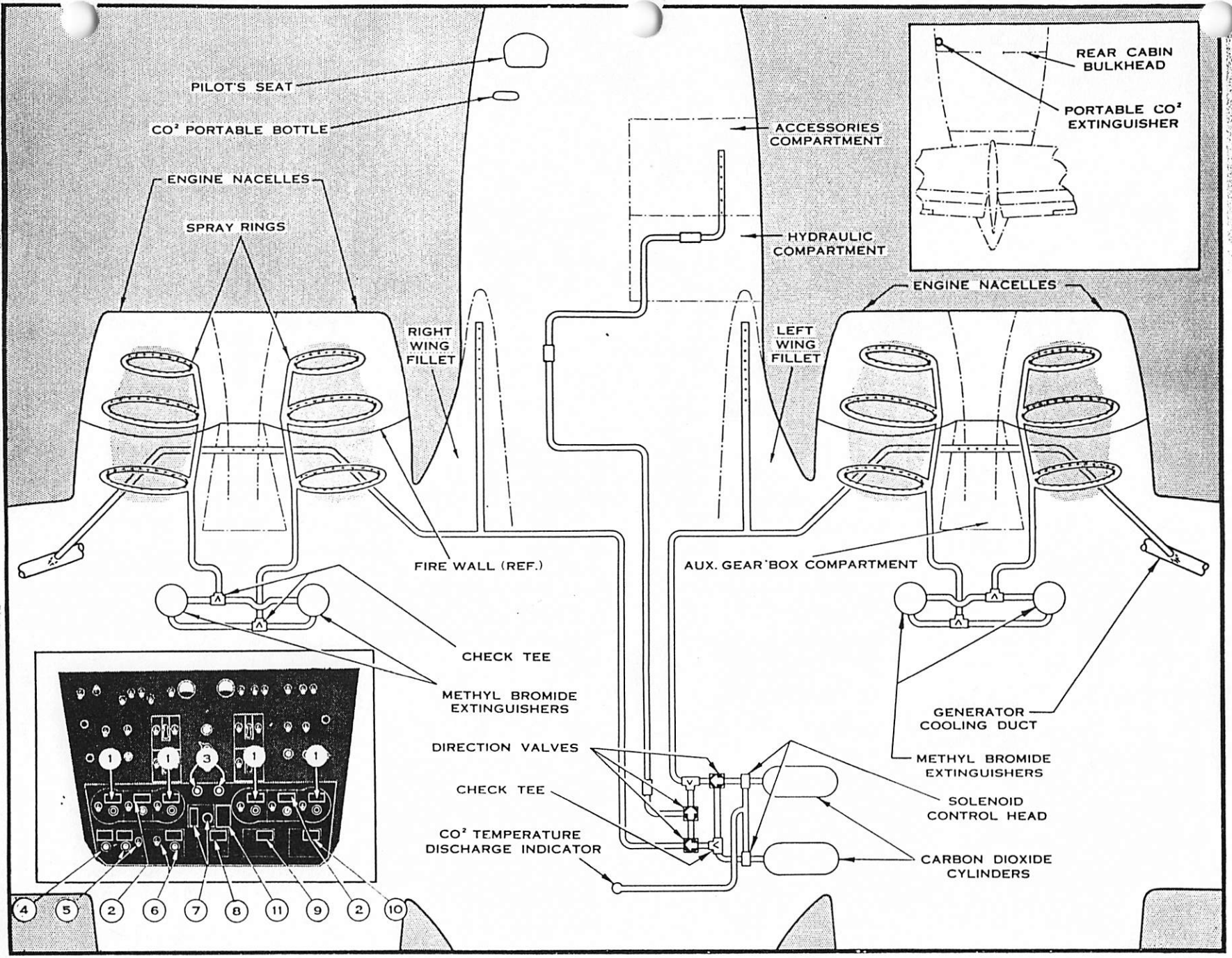
(4) The bottle discharge buttons for each engine are electrically interconnected with their respective selector switches through a time delay relay which operates in conjunction with its relative throttle lever. Once the throttle is closed, and the bottle to be discharged is selected, the bottle will discharge into the selected area after a period of 15 seconds. If the automatic discharge mechanism fails, the alternative method of pressing the discharge button may be used.

NOTE

(1) The reason for the 15 seconds time delay, is to allow the engine to reach idling r.p.m. before the extinguishing fluid is released. Failure to observe this precaution would result in insufficient saturation of the air due to high volume entry into the intake at higher r.p.m.

1-84 CARBON DIOXIDE SYSTEM:

(1) A battery comprising two cylinders, (each containing 12 lbs. of CO2) is located below the cabin floor at the rear of the main spar. The bottles are interconnected by pipe lines carrying solenoid operated two-way directional valves, double check tees, and a temperature discharge indicator. Both the



METHYL BROMIDE AND CARBON DIOXIDE FIRE EXTINGUISHING SYSTEMS
FIG. 33

directional valves and the double check tees permit the selection of a charge from each bottle to be directed to two separate areas, or alternately, to direct both charges to one area.

(2) Three supply lines from the battery, pass forward along the fuselage, one terminating in the accessories compartment, the remaining two passing outboard through the wing centre section to protect the right and left auxiliary gear box areas and generator cooling ducts. A non-return valve is fitted to the extremity of the accessory compartment supply line to prevent cabin pressure leaking back through the line to the auxiliary gear box spray deliveries.

(3) Firing of the cylinders requires relative SWITCH SELECTION and the DEPRESSING of the appropriate DISCHARGE button on the control panel (Fig. 34).

1-85 FIRE CONTROL PANEL:

(1) This panel comprises the bottom half of the flight deck roof control panel (Fig. 34) and carries all controls and warning devices relative to the fire detection and extinguishing systems.

(2) The MASTER FIRE CONTROL SWITCH must be in the ON position before any of the associated systems will operate. When this switch is in the OFF position, the adjacent indicating light will be illuminated. When the indicating light is OUT the circuits are armed.

(3) An emergency crash switch protected by a hinged flap, is centrally disposed on the lower half of the panel. When the switch is turned to the ON position, all extinguishers, both CO2 and Methyl Bromide are FIRED, the extinguishants being directed to all protected areas and compartments.

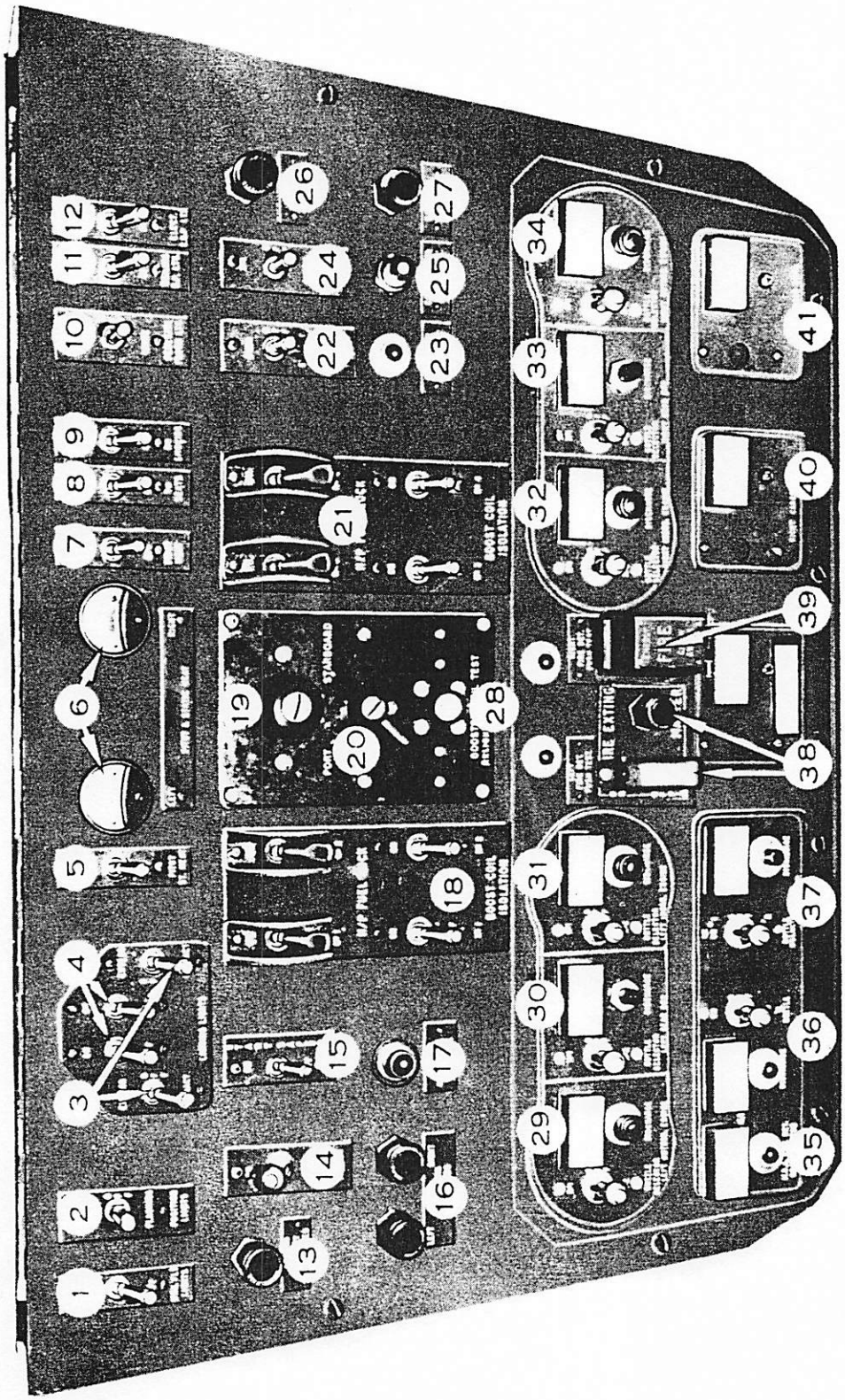
1-86 HAND FIRE EXTINGUISHERS:

(1) A number of 5 lb. portable hand fire extinguishers are located at strategic points throughout the fuselage. They are mounted in suitable brackets and located in the following positions:

(a) On the floor of the flight deck to the rear of the pilot's seat.
(b) On the wall of the main cabin adjacent to the attendant's station.

NOTE

(1) Provision will be made for the installation of four additional extinguishers.


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STANDBY
TEST
BOOST COIL ISOLATION
TRE EXTING

FIRE EXTINGUISHING CONTROLS

29. No. 1 Engine Bottle Selector and Discharge Button
30. Left Auxiliary Gear Box Bottle Selector and Discharge Button
31. No. 2 Engine Bottle Selector and Discharge Button
32. No. 3 Engine Bottle Selector and Discharge Button
33. Right Auxiliary Gear Box Bottle Selector and Discharge Button
34. No. 4 Engine Bottle Selector and Discharge Button
35. M.S.A. Reset Button
36. Bottle Selector and Discharge Button for Heater (Inoperative)
37. Bottle Selector and Discharge Button for Accessories Compartment
38. Master Fire Control Switch and Indicator Light
39. Emergency Crash Switch
40. Warning Right Cargo Compartment
41. Warning Aft Cargo Compartment

NOTES

OXYGEN SYSTEM

1-87 GENERAL:

(1) The oxygen system (Fig. 35) is a high pressure system incorporating demand type regulators which receive the supply from precharged cylinders. The system is for use with demand type C2 masks only. Additional type low pressure portable units are also provided.

1-88 OXYGEN CYLINDERS:

(1) The system supply consists of one main cylinder mounted on the left side of the flight compartment and a crate of fifteen cylinders bolted to the floor of the aircraft, aft of the entrance door. The total capacity of the system amounts to 12,328 litres, which is sufficient for five persons and at normal demand will last ten hours at a height of 30,000 feet, with the cabin unpressurized.

(2) The main cylinder has a capacity of 1,078 litres and is fitted with a hand operated stop valve. This valve is used to seal off the container whenever pipes are disconnected. A warning lable is placed near the container to remind personnel to close this valve before removal.

(3) The crate contains fifteen 1800 p.s.i. cylinders, each of 750 litres capacity. The cylinders are inter-connected and are coupled to a charging connection and an On/Off valve. All cylinders may be charged from the CHARGING connection on the forward end of the crate (Fig. 35).

(4) Portable 400 p.s.i. low pressure oxygen units of 360 litres capacity are used in the aircraft. Each unit consists of a cylinder with an integral regulator and pressure gauge.

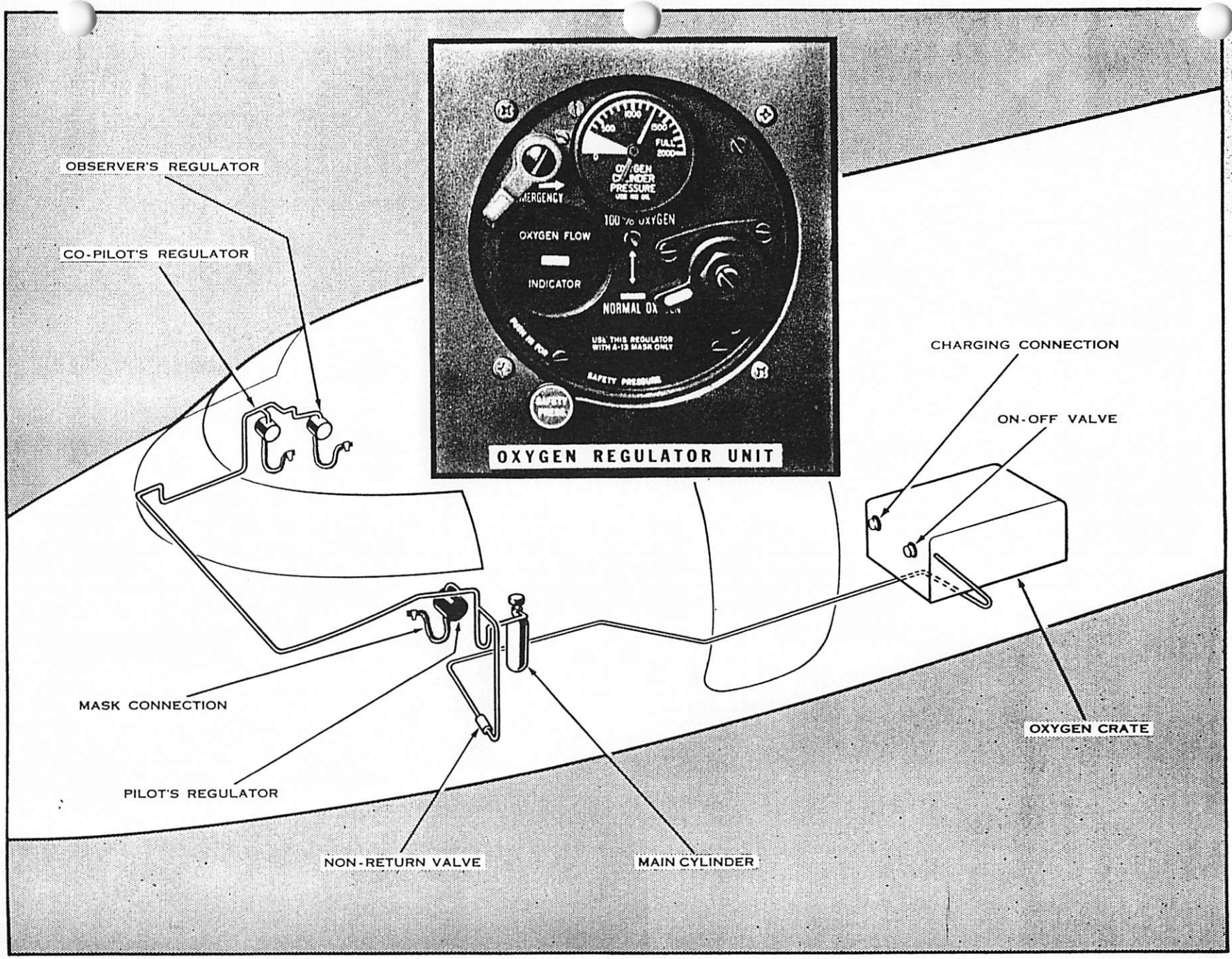
1-89 PRESSURE GAUGE:

(1) A pressure gauge is located on the face of the regulator unit and indicates the oxygen supply by denoting the pressure within the system. The dial is calibrated to show pounds per square inch pressure in 50 lb. graduations from 0 to 2000 lbs.

1-90 SUPPLY LINES:

(1) The supply from the main cylinder to the pilot's, co-pilot's and observer's regulator is by way of tubing into the main line from which the tappings are taken to the regulators.

(2) The supply from the crate to the main cylinders is piped from a connection on the outboard side of the crate, beneath the cabin floor, through a



OXYGEN SYSTEM
FIG. 35

non-return valve to the inlet of the main cylinder. The non-return valve is incorporated to prevent pressure from the main cylinder returning to the crate.

1-91 REGULATORS:

(1) The demand type regulator (Fig. 35) provides automatic mixing of oxygen and air in the required proportions for any altitude and permits an adequate supply to be maintained up to an effective pressure altitude of 40,000 feet. At altitudes higher than this figure, operation of an emergency lever on the regulator, will allow the oxygen supply to by-pass the regulator mechanism.

(2) Two major controls are provided on the unit, a NORMAL 100% OXYGEN valve and an emergency valve. Under ordinary operating conditions oxygen is permitted to enter the regulator at the top connection, where it is reduced in pressure to 30-40 p.s.i. and mixed with air according to the setting of the NORMAL 100% OXYGEN valve. When the control is in the NORMAL position an air valve within the control knob is fully open and is gradually closed by turning of the knob towards the 100% OXYGEN position.

(3) When the operator inhales, the reduced pressure in the line causes movement of a diaphragm in the regulator to operate a demand valve and thus permit a supply of regulated oxygen and air to be delivered to the mask. Compensation for altitude is made by an aneroid assembly so that the proportion of oxygen in the air is always correct for the cockpit altitude.

(4) Operation of the emergency lever permits a continuous supply of pure oxygen at demand pressure to by-pass the entire regulator mechanism. A SAFETY PRESS knob is provided for use in case of a leaking mask. When this control knob is operated, the flow of oxygen is increased to prevent dilution of the mixture.

(5) The flow is indicated by the oscillation of a blinker plate, located below the pressure indicator. The plate closes each time the operator inhales.

DITCHING CONTROLS

1-92 GENERAL:

(1) The purpose of this system is to prevent water entering the fuselage through the air conditioning and pressurization ducting when the aircraft is ditched.

(2) This system is operated from the floor of the flight deck. Opening of a hinged panel located immediately aft of the throttle control pedestal reveals the control ring. The panel incorporates a Hartwell quick release catch.

(3) The control ring is interconnected through cables and directional pulleys (along the centre line of the fuselage below floor level) to the ditching valve in the base of the pressurization discharge manifold, and to the non-

return valves in each wing duct.

(4) The cables are tensioned by bungee cords and the recoil action is dampened through two dashpots.

1-93 RELEASE MECHANISM:

(1) From the control ring an attached cable extends aft along the left side of the fuselage and is shackled to the trigger release mechanism.

(2) The release mechanism comprises a trigger assembly interconnected with the release cable. This trigger assembly pivots on a housing attached to the forward face of former 22, and registers with a cross head mounted on a shaft supported in bearings. The exposed portion of the shaft forms a guide for the cross head, the remaining portion concealed within the housing, screws through a mating bush and terminates in a control handle.

(3) The cross head is bracketed to receive two cable shackles, the cables passing forward in the fuselage through pulleys on former 21 and terminating at shackle plates between formers 20 and 21.

(4) From the shackle plates, two cables pass forward to their relative dashpots on former 18. The three remaining cables, two on the right shackle plate and one on the left shackle plate, are led aft to their respective control valve trigger mechanisms. All cables carry adjustable turnbuckles and are guided by pulleys located on fuselage formers.

(5) The dashpot cables are tensioned by the suspension of two bungee cords anchored at one end to a cross member on former 17, the alternate ends being attached to the dashpots.

1-94 OPERATION:

(1) When the ring is pulled the release mechanism trips its associated latch and allows the bungee cords to pull the assembly forward.

(2) As the latch is released, the cross head and operating cables are pulled forward. This action releases the trigger mechanisms of the ditching and non-return valves.

(3) The dashpots operate as dampers, and prevent shock loads being transferred throughout the controls.

1-95 CONTROL SETTING:

(1) The cords are pre-loaded to 280 p.s.i. by cranking the handle of the release mechanism. When the handle is cranked, the trip mechanism and relative

latches on the control valves are reset.

(2) This assembly is accessible from the centre floor panel between formers 21 and 22.

NOTE

(1) As a precaution to prevent inadvertent operation, the system is normally disconnected but can be made operative for over water flights.

ELECTRICAL SYSTEM

1-96 GENERAL:

(1) Direct and alternating current is provided for the operation of electrical equipment. The basic electrical system is a 24-28 volt, DC single wire system. The aircraft structure serves as a ground return for all circuits except where two-wire circuits are required to prevent instrument interference. DC power is supplied by two generators, the aircraft batteries, or by an outside source through an external power receptacle. Two inverters, one main and one emergency, operate on DC power and provide 400 cycle, 115 volts. In addition, a group of transformers supply 26 volt, 3 phase 400 cycle power for general instrumentation, while two engine driven alternators supply 208 volts, 3 phase, 400 to 800 cycle power for the windshield Nesa glass deicing system, wing and empennage anti-icing, cabin heating and galley equipment.

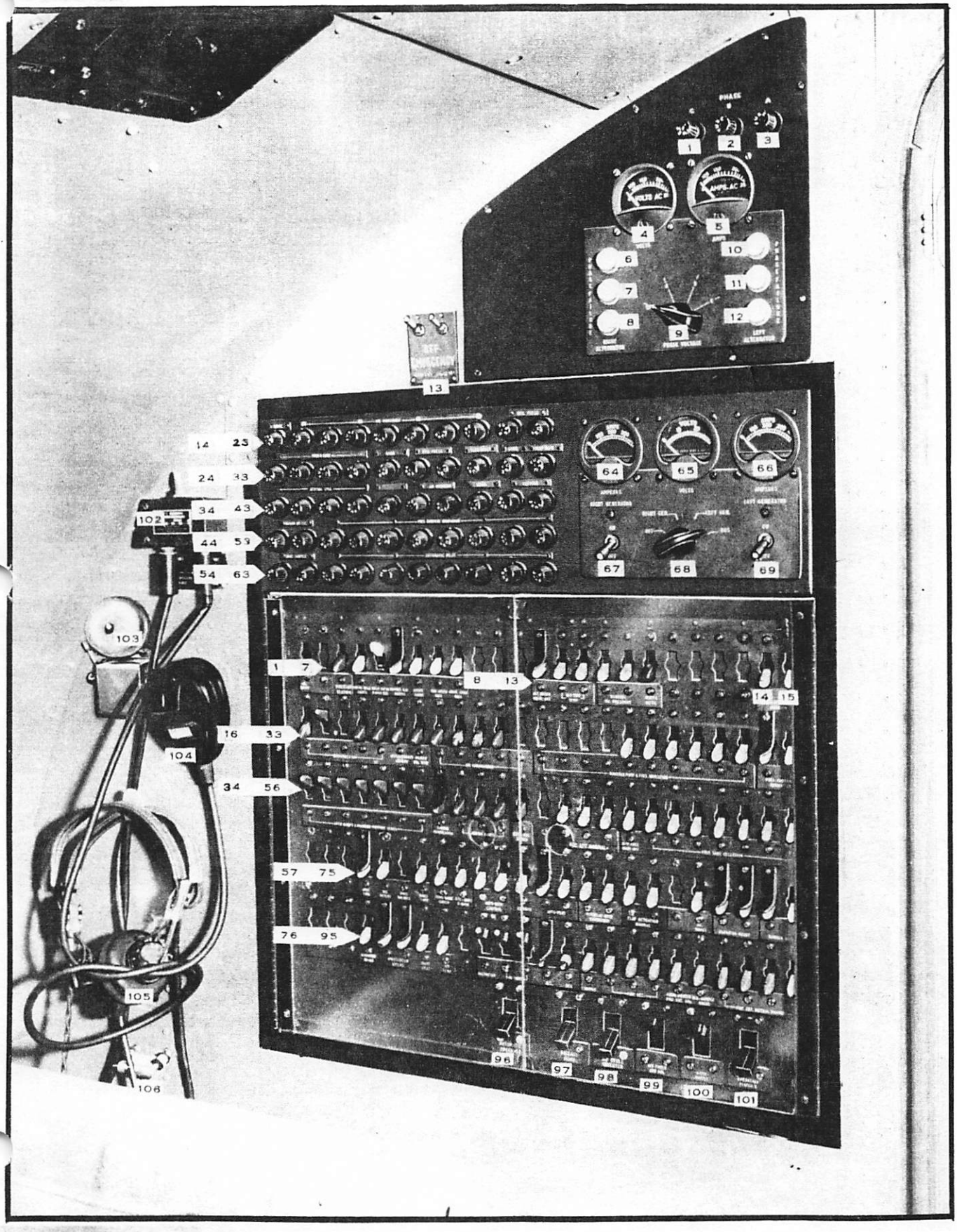
1-97 POWER SUPPLY:

(1) Two engine driven generators, one on each auxiliary gear box, supply 28.5 volts DC current for lighting, relay controls, radio and certain instrumentation. The system is fault and over voltage protected. Both generators are controlled from the power supply panel (Fig. 36) while adjustments relative to voltage and current should only be made in flight at the load centering units located in the accessories compartment (Fig. 38).

(2) The two inverters supply power for the 115 volt, 3 phase 400 cycle equipment. One inverter rated at 250 volt amperes, supplies power for emergency 400 cycle equipment only. The other inverter is rated at 3,500 volt amperes and supplies power to selected flight and engine instruments, and to certain items of radio equipment located in the accessories compartment (Fig. 38). A group of transformers (Fig. 39) supplies 26 volt, 3 phase 400 cycle power. These transformers are connected across the 115 volt, 3 phase power for general instrumentation purposes.

(3) Two engine driven alternators, one on each auxiliary gear box, supply 208 volts, 3 phase 400 to 800 cycle power to the deicing systems. This service will eventually feed the deicing systems comprising engine bullets, wings and empennage; also galley services. Relative alternator controls consisting of

ELECTRICAL POWER
SUPPLY PANEL
FIG. 36



ALTERNATOR AND POWER SUPPLY PANELS

FIG. 36

ALTERNATOR CONTROL PANEL

1. } AC Fuses
2. }
3. }
4. Alternator Voltmeter
5. Alternator Ammeter
6. } Right Alternator Phase Voltage
7. } Failure Warning Lights
8. }
9. Alternator Phase Voltage Selector
10. } Left Alternator Phase Voltage
11. } Warning Lights
12. }
13. Alternator Exciter Control Switches

POWER SUPPLY PANEL

14. Spare Fuse
15. } Oil Pressure Transmitter Fuses
16. }
17. }
18. }
19. }
20. }
21. }
22. } Brake Pressure Indicator Fuses
23. }
24. } Turn and Bank Indicator Fuses
25. }
26. }
27. }
28. Spare Fuse
29. } Hydraulic Pressure Indicator Fuses
30. }
31. Elevator Boost Fuse
32. Spare Fuse
33. Flap Indicator
34. } Attitude Gyro Fuses
35. }
36. }
37. }
38. Spare Fuse
39. } Air Control Fuses
40. }
41. Radio Supply Fuse
42. } Gyrosyn Compass Fuses
43. }
44. } Windscreen De-icer Fuses
45. }
46. } Fuel Contents Indicator Fuses
47. }
48. }
49. }
50. }
51. }
52. } 115 Volt D.C. Supply Fuses
53. }
54. }
55. }
56. } Auto-Pilot Fuses
57. }
58. }
59. }
60. }
61. }
62. }
63. }
64. Right Generator Ammeter
65. Generator Voltmeter
66. Left Generator Ammeter
67. Right Generator Master Switch
68. D.C. Voltage Selector Switch
69. Left Generator Master Switch

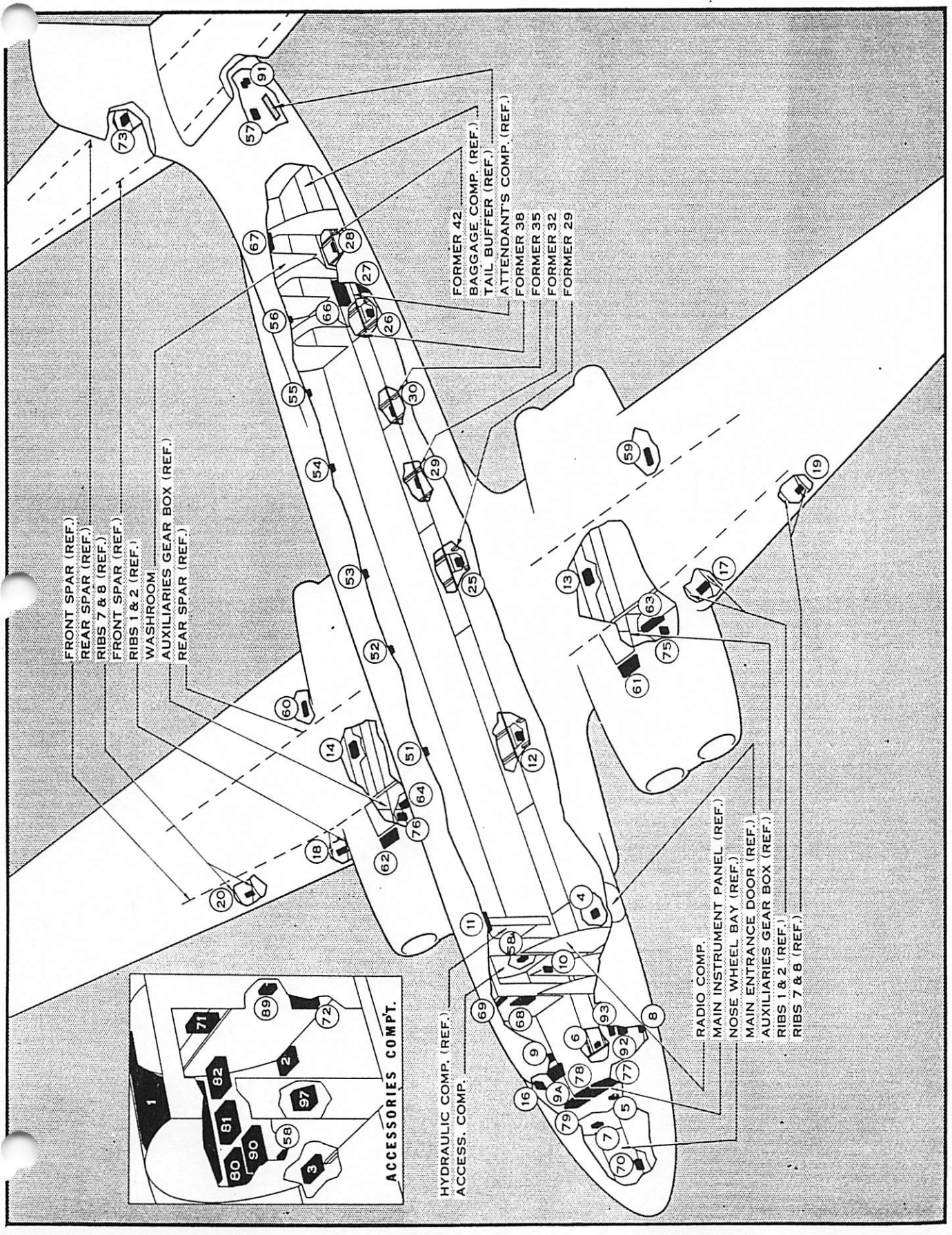
CIRCUIT BREAKERS

1. Pressurization
2. Wheel Well lights
3. Washroom Cargo lights
4. Refuel Bay lights
5. Electrical Signs
6. Cabin Attendant's Call System
7. Ground Crew Call System
8. Jet Bleed Valve Actuating
9. Jet Pipe Temperature
10. Bearing Oil Pressure
11. Bearing Oil Pressure
12. Bearing Oil Pressure
13. Water Methanol
14. Starting & Ignition
15. Starting & Ignition
16. Pressurization
17. Pressurization
18. Pressurization
19. Pressurization
20. Supercharger Airflow
21. Heater Discharge Temperature
22. Air Conditioning
23. Air Conditioning
24. Air Conditioning
25. Air Conditioning
26. Booster Pump & Fuel Indication
27. Booster Pump & Fuel Indication
28. Booster Pump & Fuel Indication
29. Booster Pump & Fuel Indication
30. Booster Pump & Fuel Indication
31. Booster Pump & Fuel Indication
32. Booster Pump & Fuel Indication
33. Fuel Contents Indication
34. Not used
35. Flight Deck & Magnetic Compass lights
36. Flight Deck & Magnetic Compass lights
37. Flight Deck & Magnetic Compass lights
38. Flight Deck & Magnetic Compass lights
39. Flight Deck & Magnetic Compass lights
40. Flight Deck & Magnetic Compass lights
41. Leading Edge Floodlights (not operative)
42. Landing lights
43. Landing lights
44. Landing lights
45. Navigational Position lights
46. Landing Gear - Normal Actuation
47. Landing Gear - Normal Actuation
48. Landing Gear - Normal Actuation
49. H.P. Fuel Shut-off
50. Fuel Tank Selection
51. Fuel Tank Selection
52. Fuel Tank Selection
53. Fuel Tank Selection
54. Fuel Tank Selection
55. Fuel Tank Selection
56. Fuel Tank Selection
57. Gust Locks (not operative)
58. Ice Detection
59. Stall Warning
60. Door Warning lights
61. Landing Gear Indication (lights & horn)
62. Anti-skid Control
63. Anti-skid Control
64. Gyrosyn Compass
65. Auto-Pilot (not operative)
66. Auto-Pilot
67. Landing Gear and Flaps Emergency actuation
68. Landing Gear and Flaps Emergency actuation
69. Normal Flap actuation
70. Normal Flap actuation
71. Aileron Booster Control
72. Elevator Booster Control (not oper.)
73. Elevator Booster Control (not oper.)
74. Rudder Booster Control (not oper.)
75. Rudder Booster Control (not oper.)
76. Windscreen De-icing
77. Windscreen Wipers
78. Windscreen Wipers
79. Pitot Heat
80. Air Temperature
81. Voltmeter
82. Inverter
83. Inverter
84. Alternator
85. Generator
86. Supply
87. Fire Detection
88. Fire Detection
89. Fire Detection

90. Combustion Heater Fire Extinguisher - CO2 System
91. M.S. Sampling Valve
92.
93. Fire Extinguisher - Methyl Bromide System
94.
95.
96. Automatic Observer
97. Ballast Pump
98. 60-cycle Inverter
99. Aft Power Junction Box
100. Radio Supply
101. Emergency Flaps and Landing Gear

EQUIPMENT

102. Observer's Interphone 'Plug In' Socket
103. Fire Warning Bell
104. Observer's Interphone
105. Observer's Earphones
106. Emergency Flaps and Landing Gear Switch


FRONT SPAR (REF.)
REAR SPAR (REF.)
RIBS 7 & 8 (REF.)
FRONT SPAR (REF.)
RIBS 1 & 2 (REF.)
WASHROOM
AUXILIARIES GEAR BOX (REF.)
REAR SPAR (REF.)
FORMER 42
BAGGAGE COMP. (REF.)
TAIL BUFFER (REF.)
ATTENDANT'S COMP. (REF.)
FORMER 38
FORMER 35
FORMER 32
FORMER 29
HYDRAULIC COMP. (REF.)
ACCESS. COMP.
RADIO COMP.
MAIN INSTRUMENT PANEL (REF.)
NOSE WHEEL BAY (REF.)
MAIN ENTRANCE DOOR (REF.)
AUXILIARIES GEAR BOX (REF.)
RIBS 1 & 2 (REF.)
RIBS 7 & 8 (REF.)
ACCESSORIES COMP'T.

G.A. ELECTRICAL JUNCTION BOXES

BOX	POSITION	FUNCTION
1	Outbd. wall Accessories Compt.	Main Junction Box.
2	Accessories Compt. Formers 14/15 at floor level.	Pull Box - services from right wing root to E1 & 5.
3	Accessories Compt. Formers 13/14 at floor level.	Pull Box - services fwd. of Access. Compt. to E5.
4	Left side under floor adjacent to Main Entrance Door.	Pull Box - services from aft and left wing root and E1 and fwd.
5	Centre fuselage fwd. of Throttle Control Pedestal.	Junction Box & Pull Box - fwd. services.
6	Under Flt. Compt. centre floor panel	Junction Box - Pedestal & Control Column services.
7	Nose wheel well - on right side of fuselage fwd. of Pressure Dome.	Nose wheel well services.
8	Left side wall of Flt. Compt. below Pressurization Panel.	Buzzer Panel.
9	Roof of Flt. Compt. - aft of Roof Control Panel.	Junction Box for Roof Control Panel.
9A	Roof of Flight Compartment.	Switch Panel.
10	Inboard wall of Radio Compt.	Auto Pilot Junction Box.
11	Centrally disposed on roof of main cabin and level with after wall of Acc. Compt.	Hall, Roof and Cargo light services.
12	Under floor centre panel, immed. fwd. of front spar.	Junction & Pull Box.
13	Wheel well roof - Left.	Wheel well services and trailing edge services.
14	Wheel well roof - Right.	Wheel well services and trailing edge services.
16	Right Side Panel.	Air Cond. - Pressurization - De-icing.

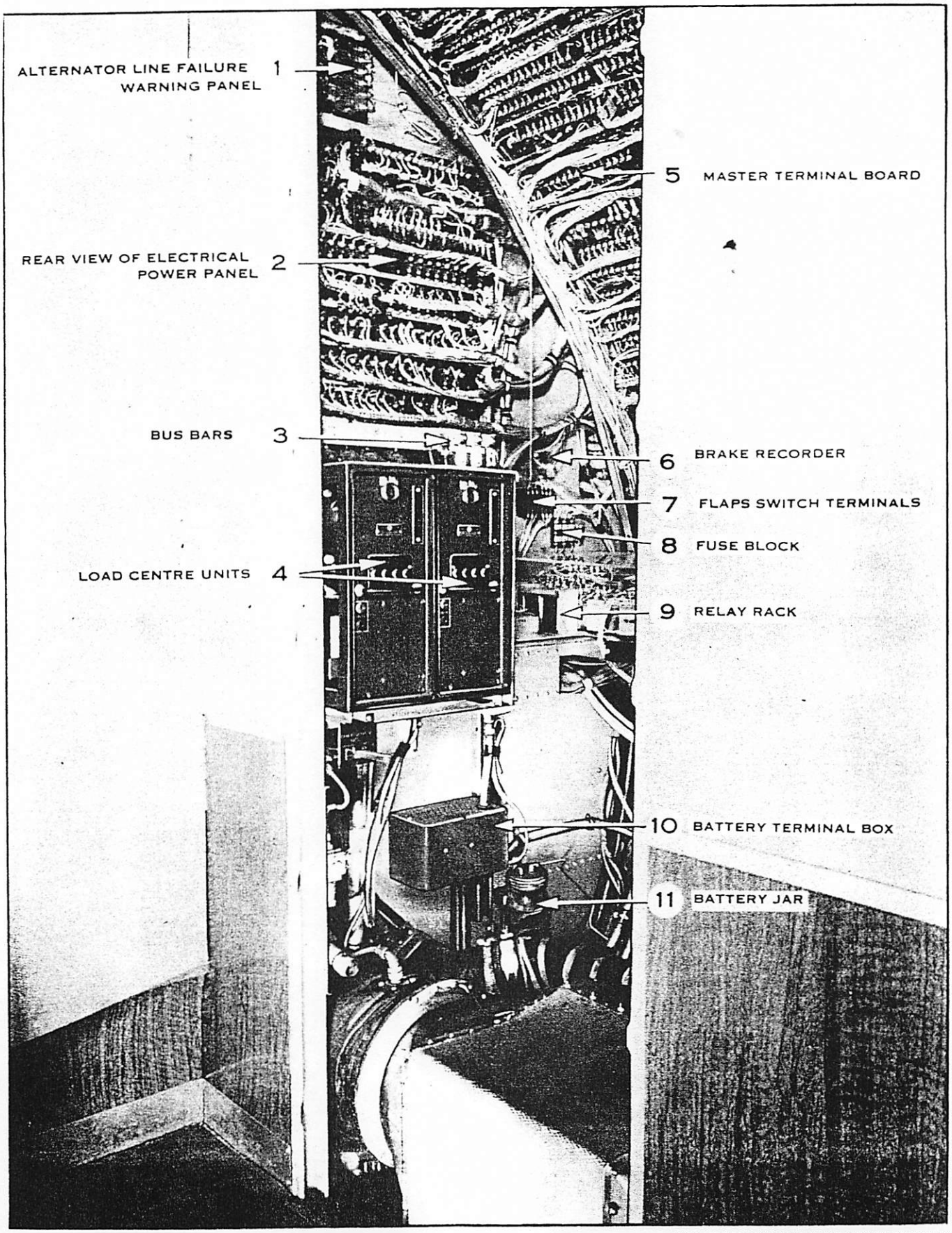
BOX	POSITION	FUNCTION
17	Rib 1-2 Left outer wing leading edge.	Leading edge services and fuel pumps.
18	Rib 1-2 Right outer wing leading edge.	Leading edge services and fuel pumps.
19	Rib 7-8 Left outer wing leading edge.	Leading edge services.
20	Rib 7-8 Right outer wing leading edge.	Leading edge services.
25	Under centre floor panel immed. aft of Main Spar.	Pull & Junction Box - fuselage and tail services.
26	Under centre floor panel in line with Cabin Alt. Panel.	Pull & Junction Box - fuselage and tail services.
27	Left side aft fuselage - under Cabin Attendant's Panel.	Pull & Junction Box - aft Power Box.
28	Under centre floor just aft of aft bulkhead.	Pull & Junction Box - tail & aft fuselage services.
29	Under centre floor in line with 7th window.	Seat services.
30	Under centre floor aft of 8th window.	Seat services.
31 to 50	Not used.	Not used.
51	Centre of main cabin roof in line with 2nd window.	Roof - Cargo - Hall light services.
52	Centre of main cabin roof in line with 4th window.	Roof - Cargo - Hall light services.
53	Centre of main cabin roof between windows 5 & 6.	Roof - Cargo - Hall light services.
54	Centre of main cabin roof immediately fwd. 8th window.	Roof - Cargo - Hall light services.
55	Centre of main cabin roof in line with 9th window.	Roof - Cargo - Hall light services.

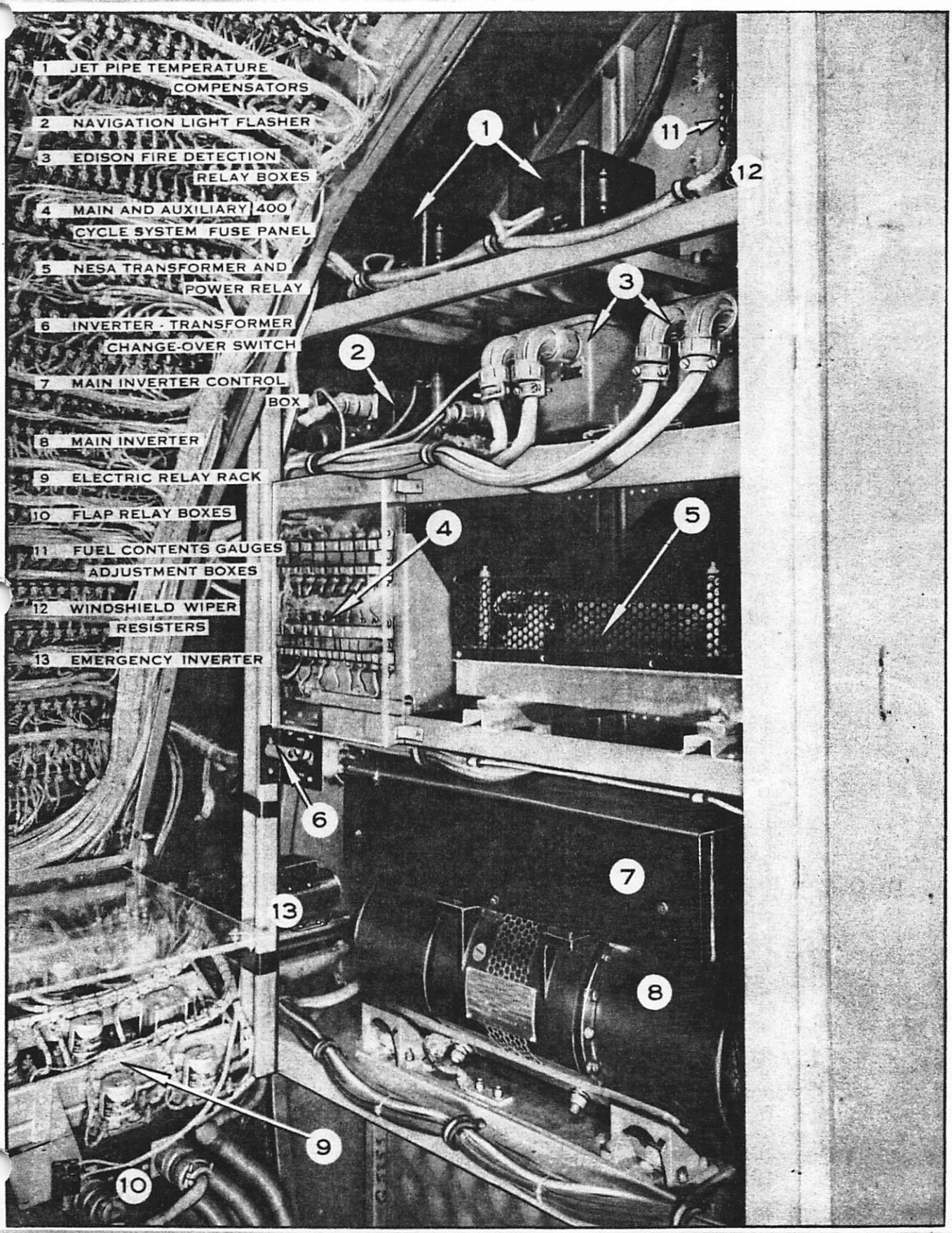
GENERAL ASSEMBLY
OF
JUNCTION BOXES
FIG. 37

BOX	POSITION	FUNCTION
56	Centre of main cabin roof in line with Cabin Attendant's Panel.	Roof - Cargo - Hall light services.
57	Formers 47-48 - Tail cone.	Tail unit services.
58	Access. Compt. fwd. wall - just above floor level.	Battery Relay.
58A	Hydraulic Compt. Aux. Battery Assembly.	Auxiliary Battery Relay Box.
59	Rib 1-2 Left outer wing - trailing edge.	Trailing edge services.
60	Rib 1-2 Right outer wing - trailing edge.	Trailing edge services.
61	Centrally disposed on the upper surface of the left nacelle.	Engine services.
62	Centrally disposed on the upper surface of the left nacelle.	Engine services.
63	Left nacelle Accessories Bay.	AC & Starting - External supply services.
64	Right nacelle Accessories Bay.	AC & Starting services.
66	Left side of fuselage immed. fwd. of aft bulkhead.	Cabin Attendant's Panel.
67	Centre of fuselage roof immed. aft of aft bulkhead.	Aft services.
68	Right side of Flt. Compt. aft wall.	Circuit Breaker Panel.
69	Right side of Flt. Compt. aft wall above E68.	Alternator Control Panel.
70	Nose wheel well Formers 3-4.	Nose wheel services.
71	Access. Compt. centre shelf aft.	Inverter Control Panel.

BOX	POSITION	FUNCTION
72	Access. Compt. inbd. wall and facing outboard.	Air Conditioning Calibration Box.
73	Tailplane Rib 2A (right side).	Ground call.
75	Left Nacelle Access. Bay wall.	AC Fuse Box.
76	Right Nacelle Access. Bay wall.	AC Fuse Box.
77	Pilot's Instrument Panel.	Instrumentation.
78	Engine Instrument Panel.	Instrumentation.
79	Co-Pilot's Instrument Panel.	Instrumentation.
80	Access. Compt. outboard. Relay Rack (forward).	Relays.
81	Access. Compt. outboard. Relay Rack (centre).	Relays.
82	Access. Compt. outboard. Relay Rack (aft).	Relays.
89	Access. Compt. After Wall.	Air Conditioning J.B.
90	Access. Compt. Forward Shelf.	Fire Extinguisher.
91	Fuselage Former 49 - Access through Tail Buffer.	Ground call.
92	Pressurization Panel - Left Side Panel.	Pressurization Control & Instrumentation.
93	Left Side Panel - above E92.	Pressurization Switch Panel.
97	Access. Compt. at floor level.	Heater & De-icer control box for AC supply.

ACCESSORIES COMPT.
ELECTRICS
(FORWARD BULKHEAD)
FIG. 38




1 JET PIPE TEMPERATURE COMPENSATORS
2 NAVIGATION LIGHT FLASHER
3 EDISON FIRE DETECTION RELAY BOXES
4 MAIN AND AUXILIARY 400 CYCLE SYSTEM FUSE PANEL
5 NESA TRANSFORMER AND POWER RELAY
6 INVERTER - TRANSFORMER CHANGE-OVER SWITCH
7 MAIN INVERTER CONTROL BOX
8 MAIN INVERTER
9 ELECTRIC RELAY RACK
10 FLAP RELAY BOXES
11 FUEL CONTENTS GAUGES ADJUSTMENT BOXES
12 WINDSHIELD WIPER RESISTERS
13 EMERGENCY INVERTER

rectifiers, transformers, ballast resistors, carbon pile regulators, field relays and circuit breakers, may be reached from the triangular access panels to the rear of the engine driven gear boxes between the nacelles.

(4) A further transformer located in the accessories compartment, is connected across the 208 volt, 3 phase power supply. This unit supplies a range of voltages between 120 and 400 volts 3 phase power for the windshield Nesa deicing system.

1-98 POWER DISTRIBUTION:

(1) The aircraft is equipped with a multibus system. The main bus is located in the distribution panel of the accessories compartment, while the auxiliary buses are located in the flight deck circuit breaker panel, the aft power junction box and in the cabin attendant's panel. The aft power junction box and the cabin attendant's panel are located adjacent to each other on the left side of the main cabin just forward of the rear cabin bulkhead. Each of the auxiliary bus groups consist of one or more buses to facilitate the distribution of current required to operate the services.

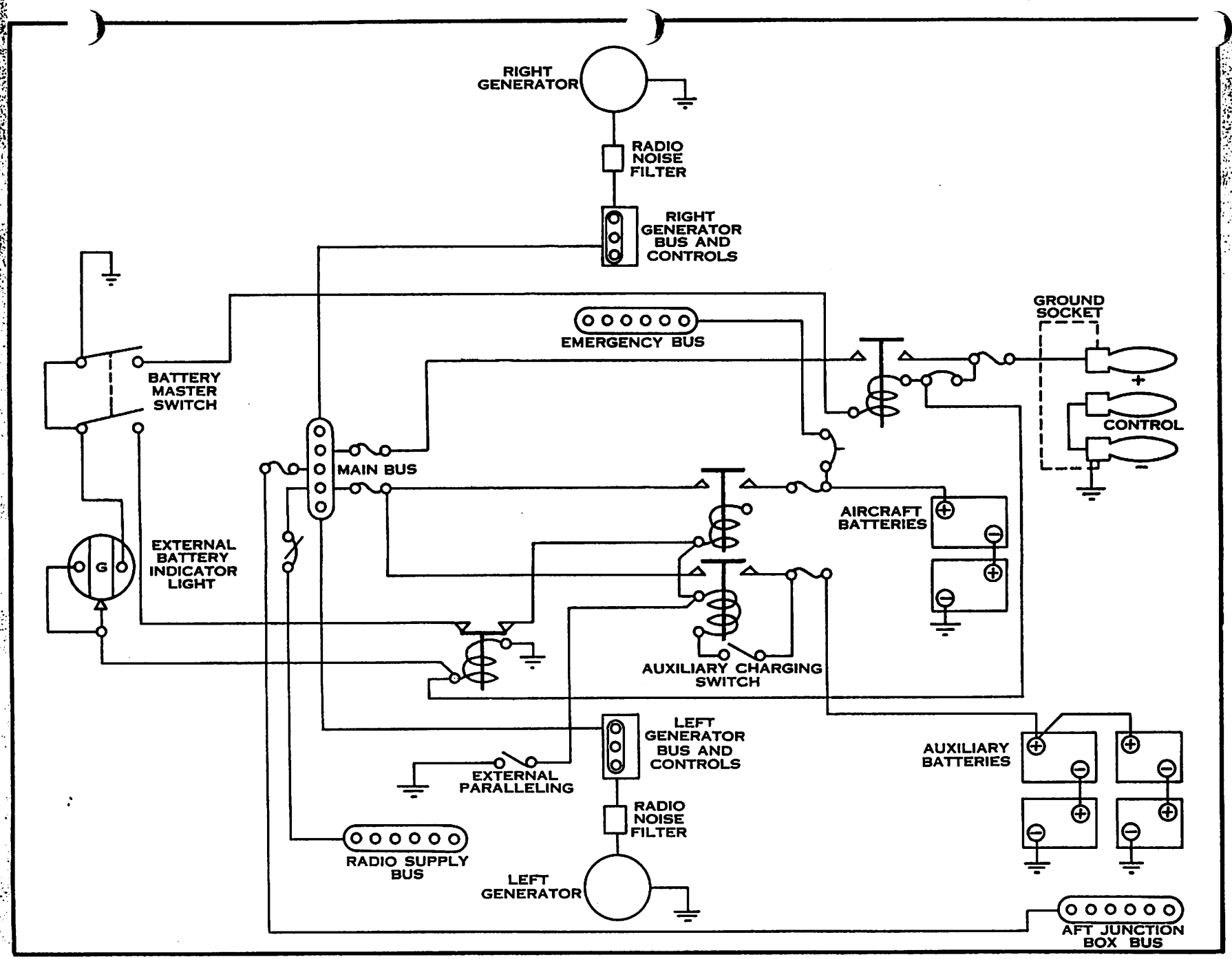
(2) All power buses and their interconnecting leads including the source of power supply, are protected by circuit breakers, current limiters or fuses.

(3) All circuit breakers and circuit breaking switches installed on the circuit breaker panel, cabin attendant's panel and the aft power junction box are of the trip free type and are functionally grouped as far as is practicable. AC circuit breakers and fuses are segregrated wherever possible.

(4) Fuses for light leads are used throughout the 400 cycle fixed frequency and alternator systems.

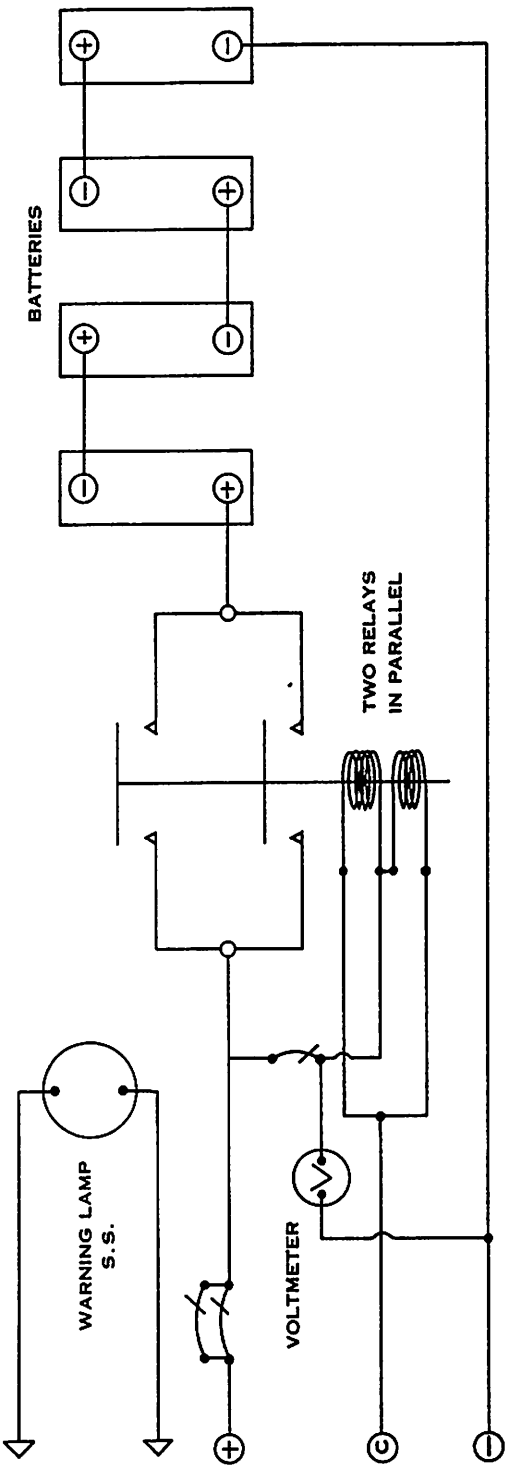
1-99 BATTERIES:

(1) Two 12 volt 89 ampere hour batteries connected in series to supply 24 volts, are used for ground testing and generator stabilization in flight. These batteries are located on the flight deck floor to the rear of the co-pilot's seat. Four additional batteries are installed in the fuselage hydraulic compartment. These batteries are provided for engine starting and can be used in conjunction with an external supply, the latter having a capacity lower than that required for direct starting of the engines. When no external power supply is available, limited starting on the auxiliary batteries is permitted. The auxiliary batteries are connected to the main bus is series parallel to the engine driven generator supply circuit. The auxiliary batteries can be charged in flight when connected to the main bus through the paralleling switch. All batteries are located in acid proof battery boxes. Neutralizing jars containing a solution of sodium bicarbonate are located in the accessories compartment.

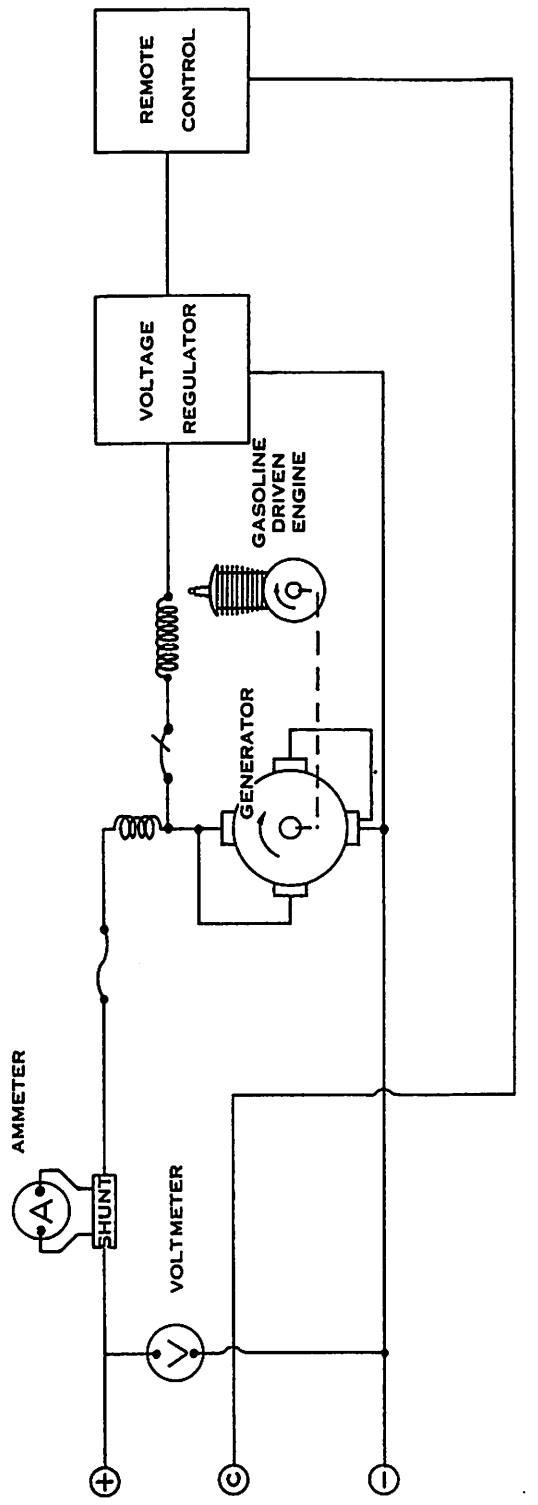


WIRING DIAGRAM
DC GENERATOR SUPPLY
FIG. 40

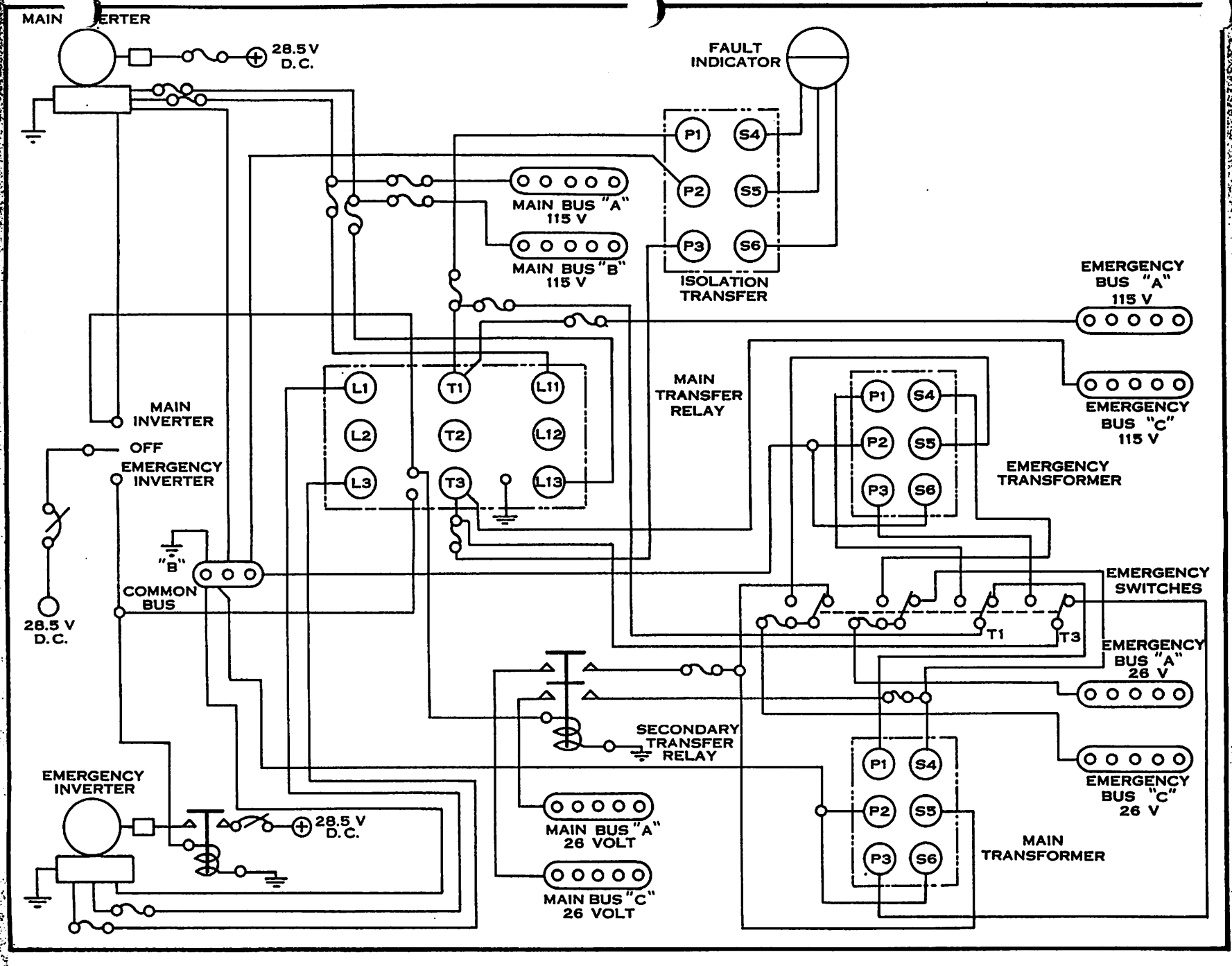
WIRING DIAGRAM
28.5V DC
EXT. POWER SUPPLY
FIG. 41



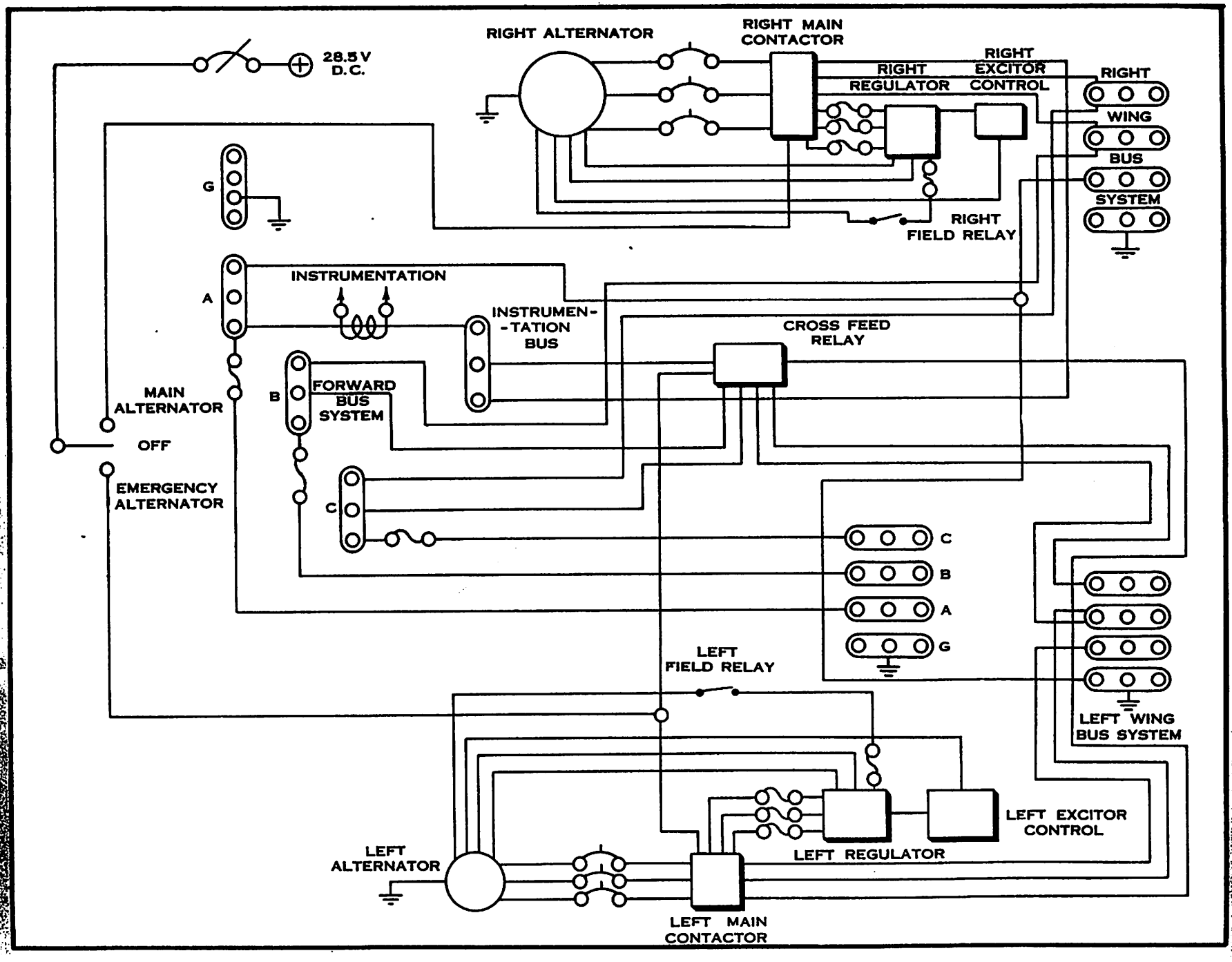
BATTERY ENERGIZER 28.5 V. DC



GENERATOR ENERGIZER (GASOLINE DRIVEN) 28.5V. DC

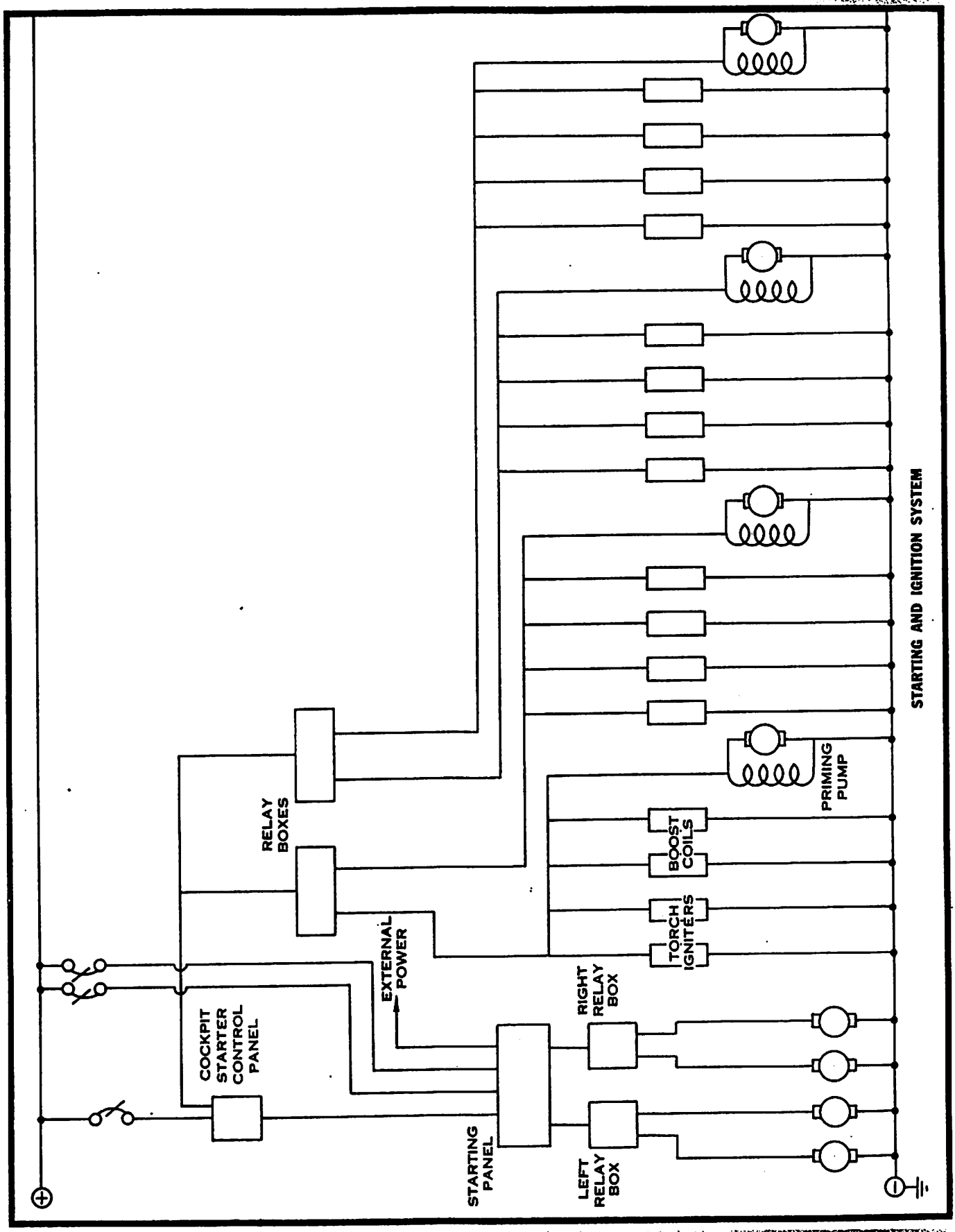


WIRING DIAGRAM
115V/26V AC
FIXED FREQ. SUPPLY
FIG. 42



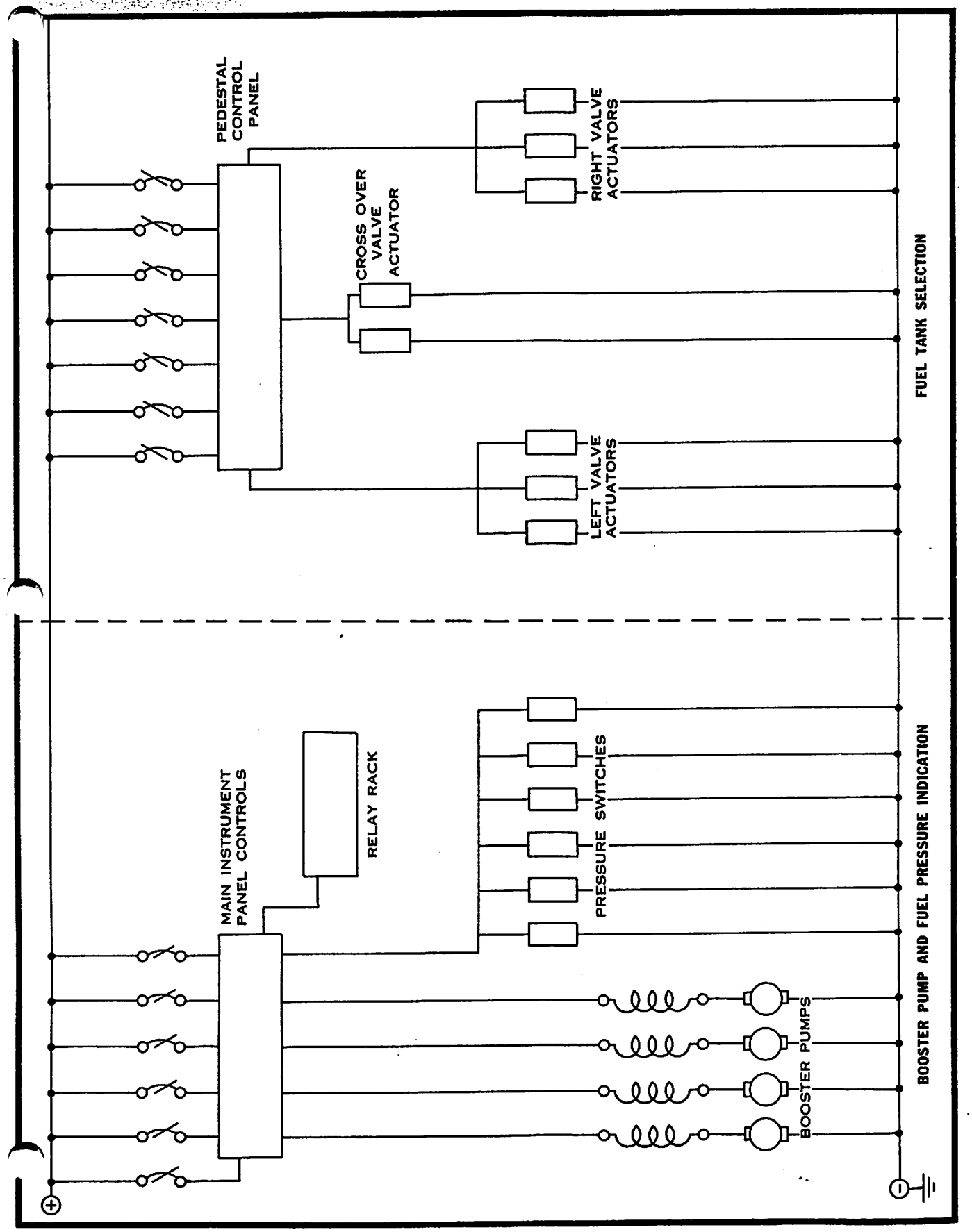
WIRING DIAGRAM
208V. AC
POWER SUPPLY
FIG. 4.3

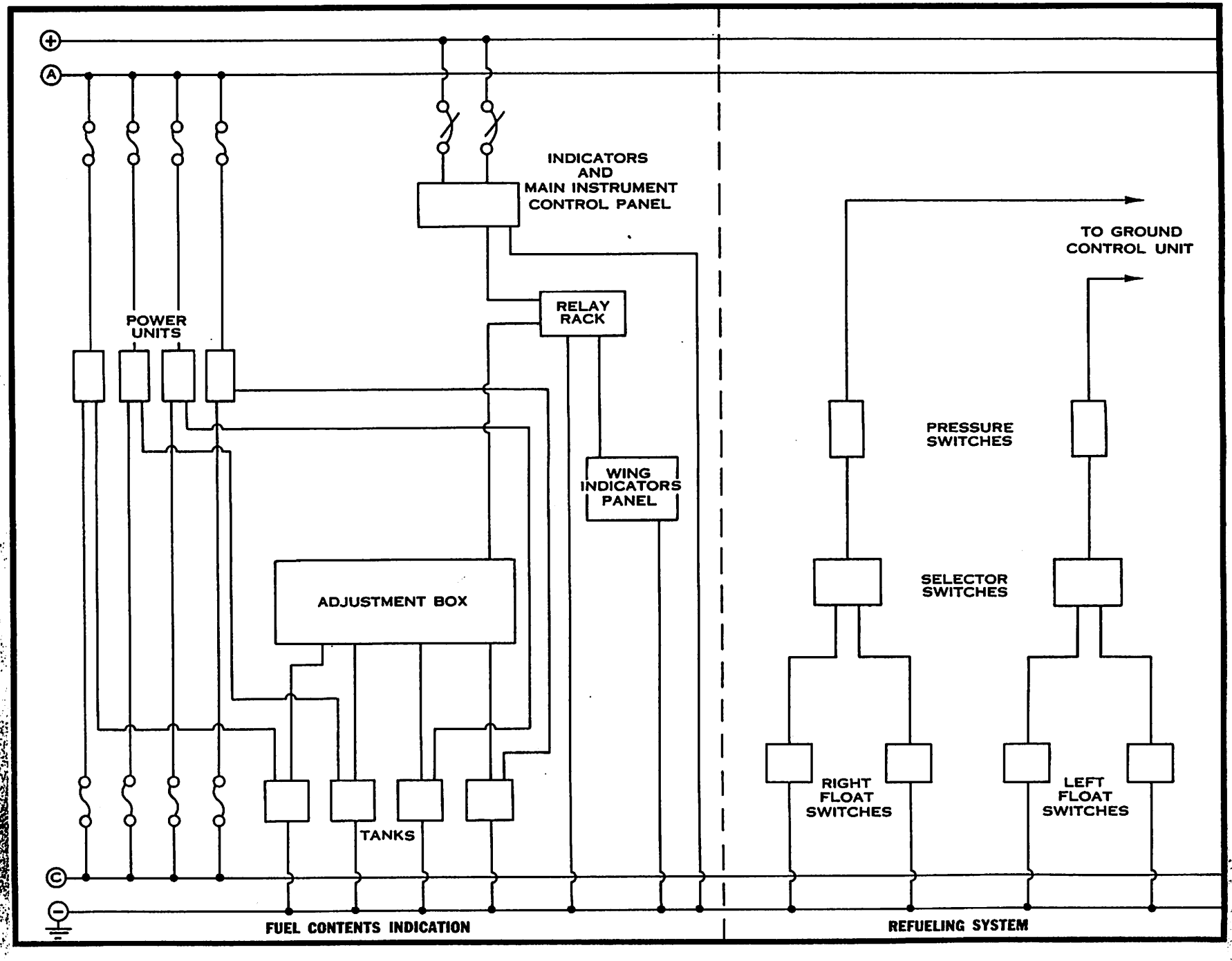
WIRING DIAGRAM
FOR
COMPLETE ELECTRICAL
SERVICES
FIG. 43(a)



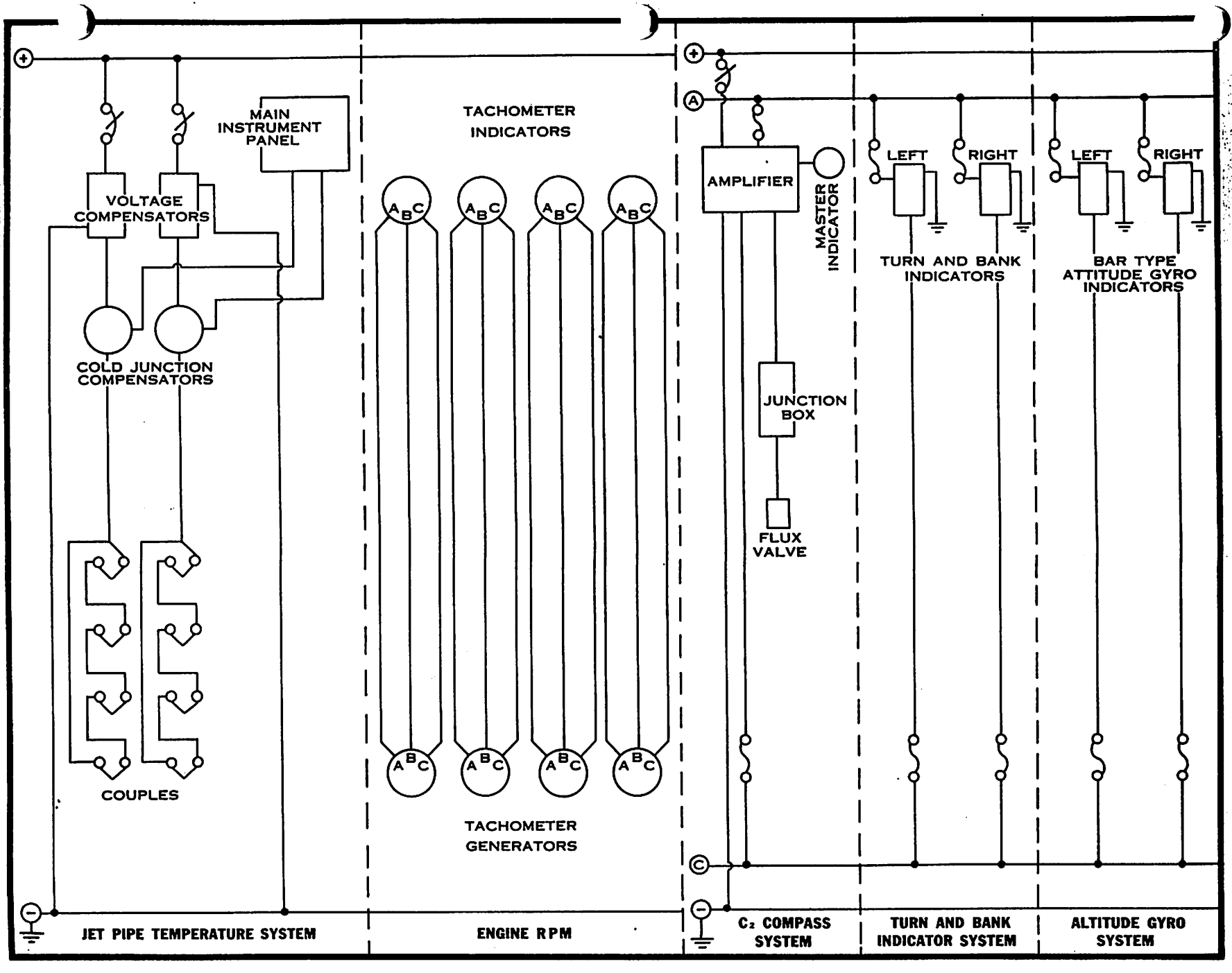
WIRING DIAGRAM
FOR
COMPLETE ELECTRICAL
SERVICES

FIG. 43(b)



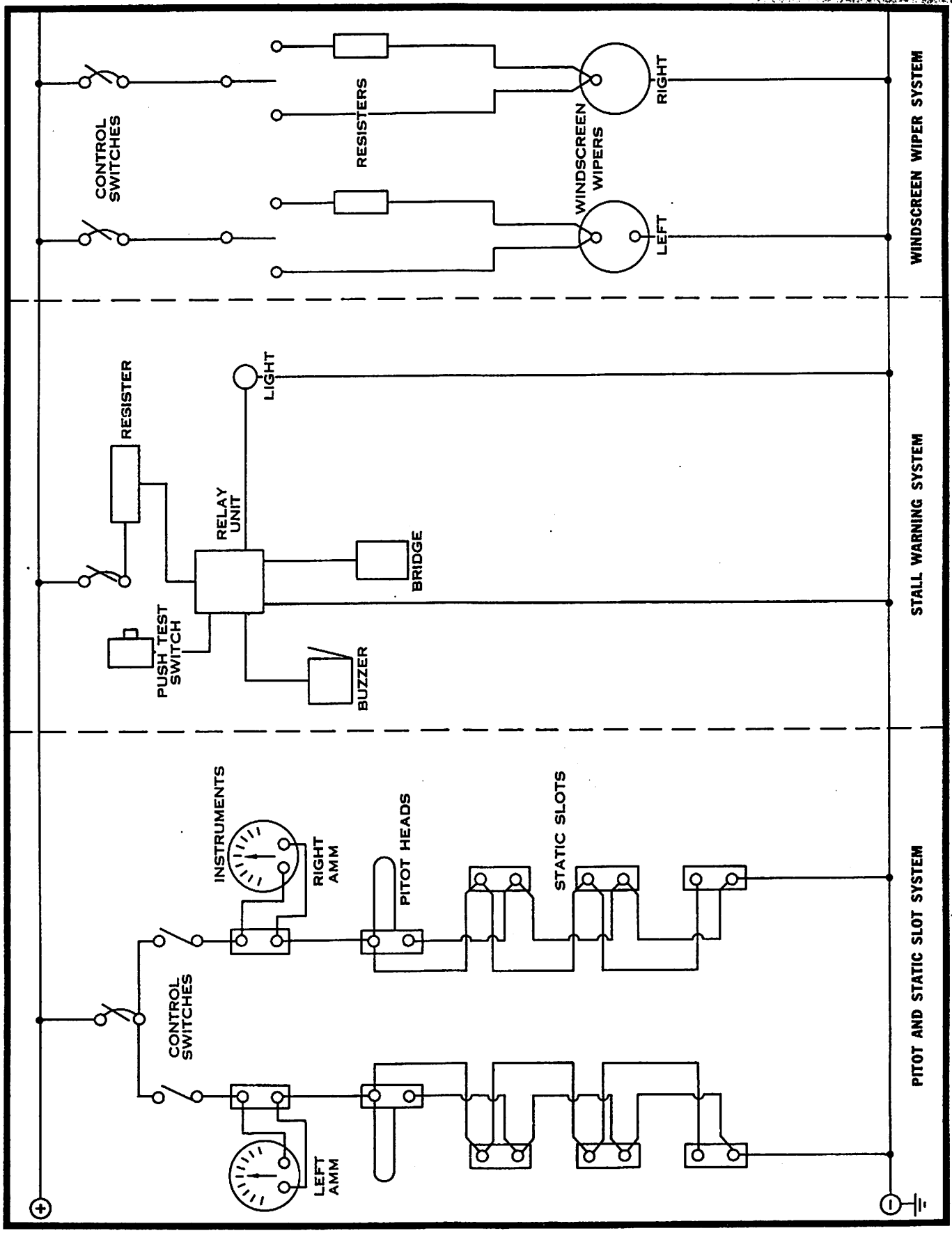


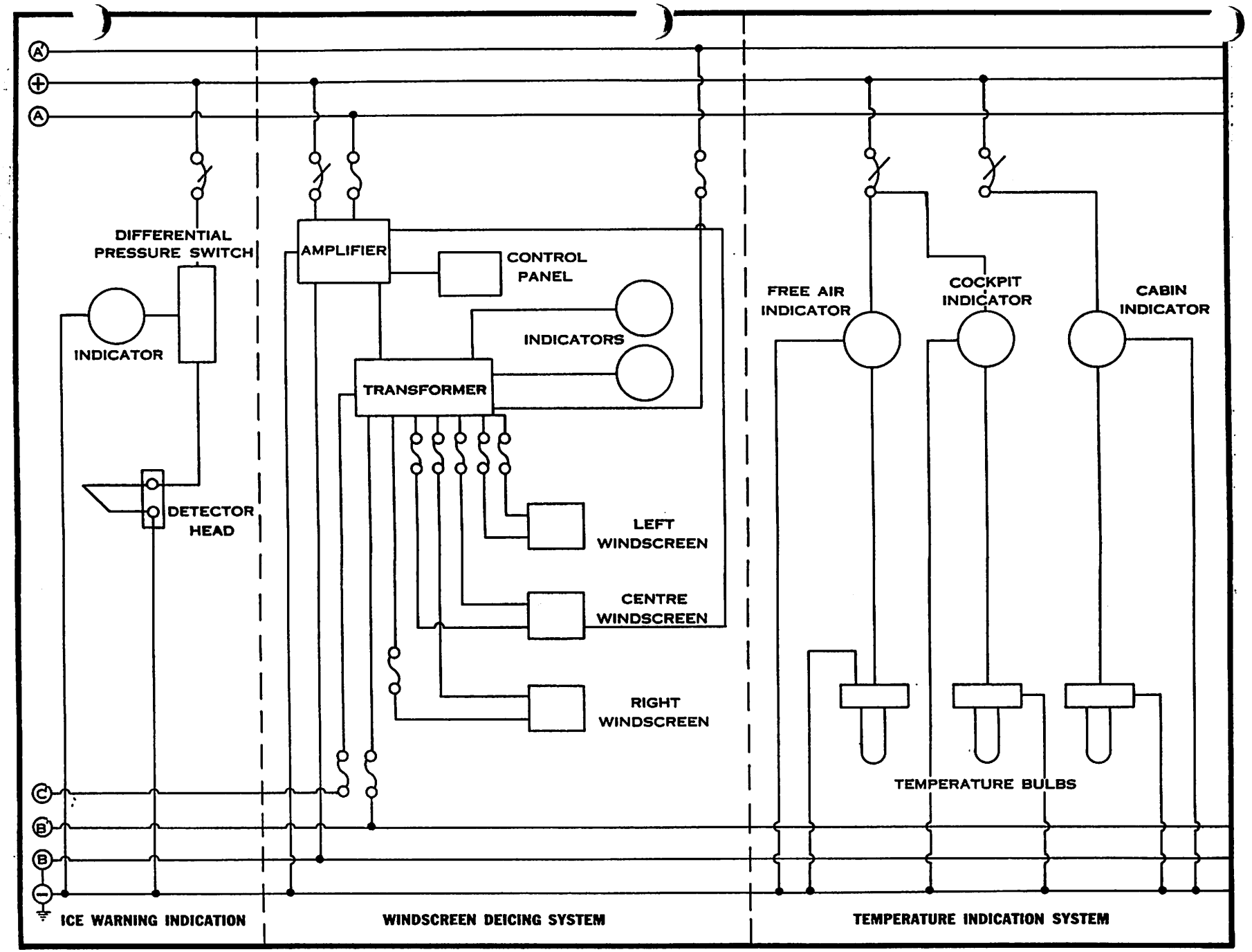
WIRING DIAGRAM
FOR
COMPLETE ELECTRICAL
SERVICES
FIG. 43(c)



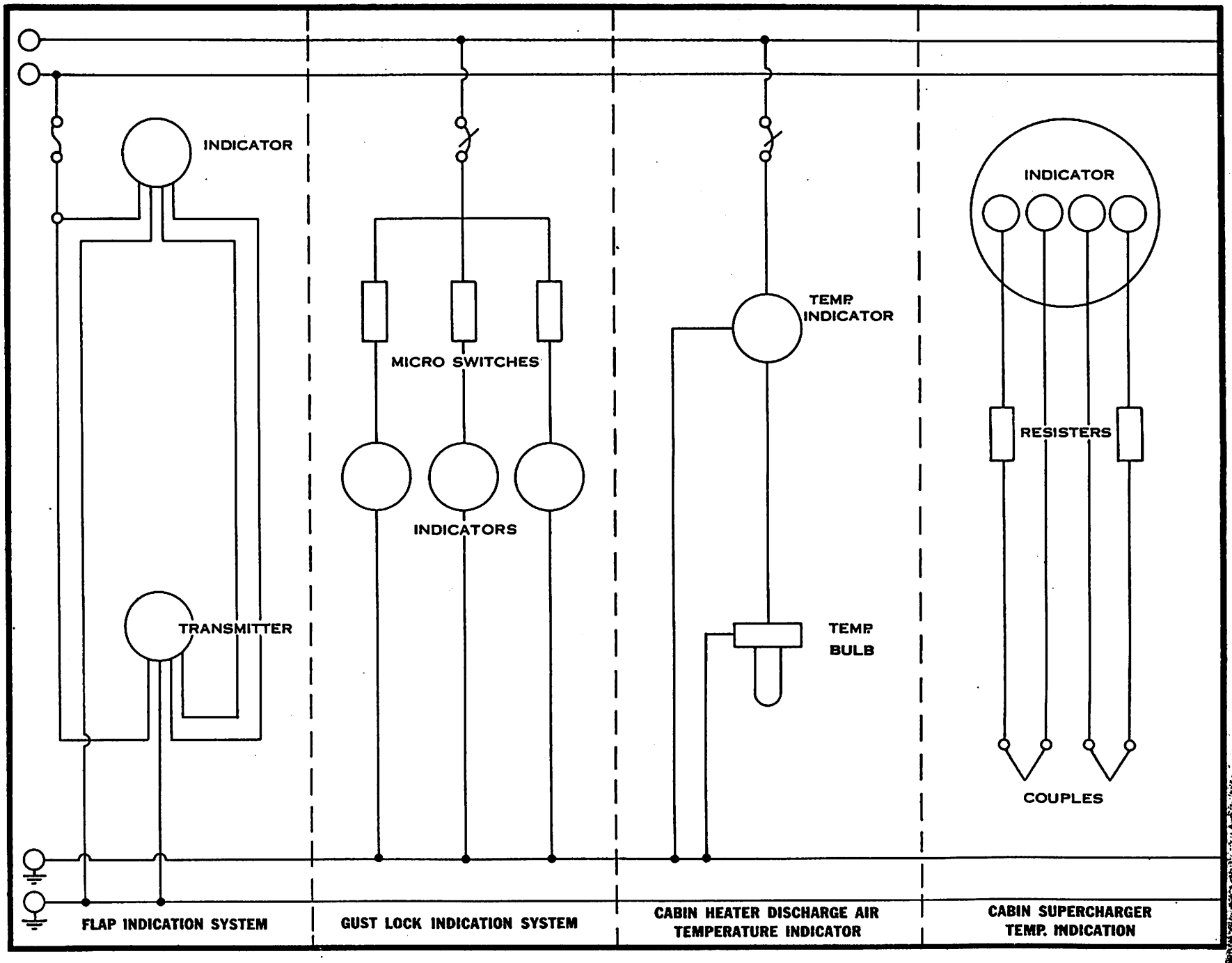
WIRING DIAGRAM FOR COMPLETE ELECTRICAL SERVICES
FIG. 43(d)

WIRING DIAGRAM
FOR
COMPLETE ELECTRICAL
SERVICES
FIG. 43(e)

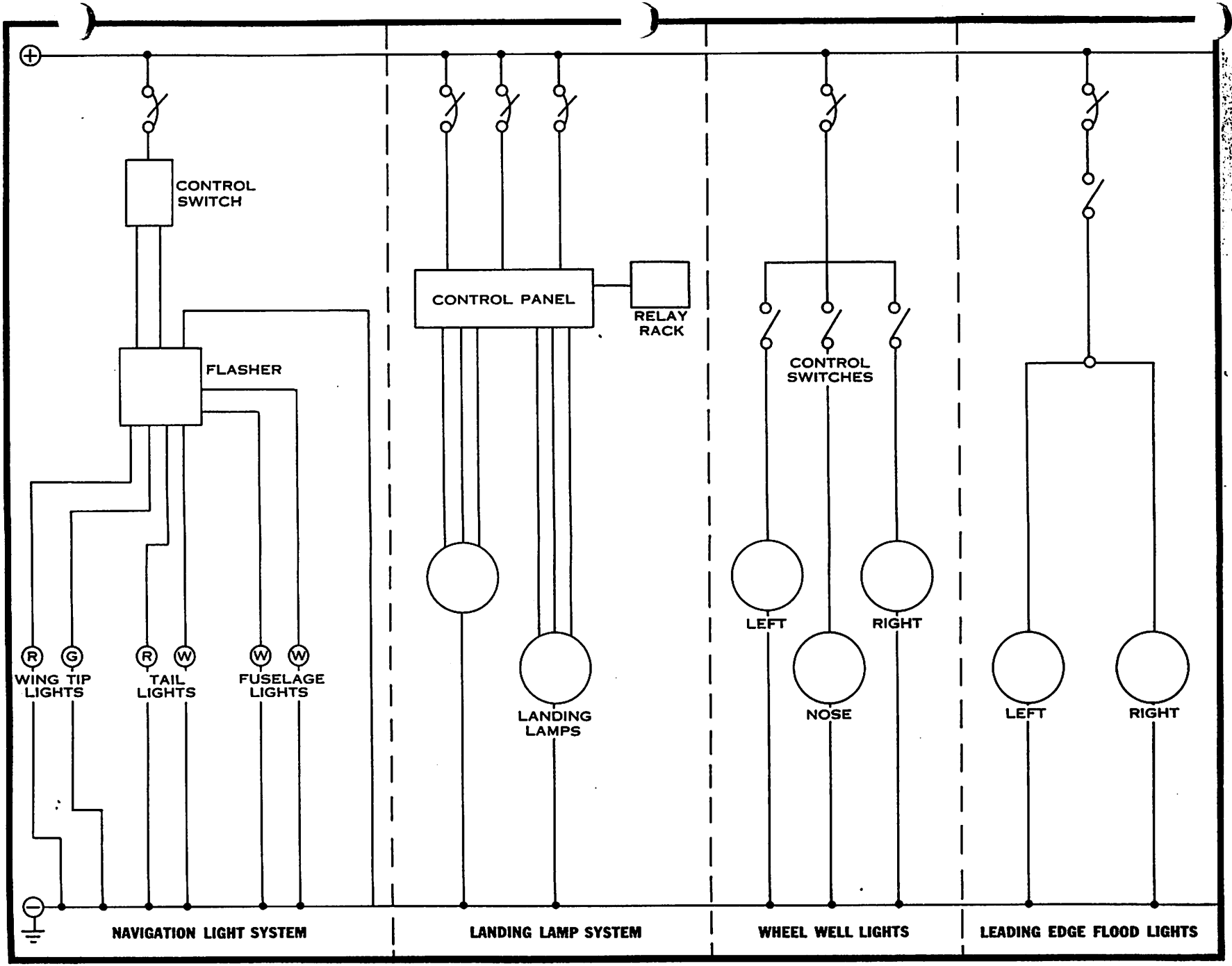




WIRING DIAGRAM FOR COMPLETE ELECTRICAL SERVICES
FIG. 43(1)

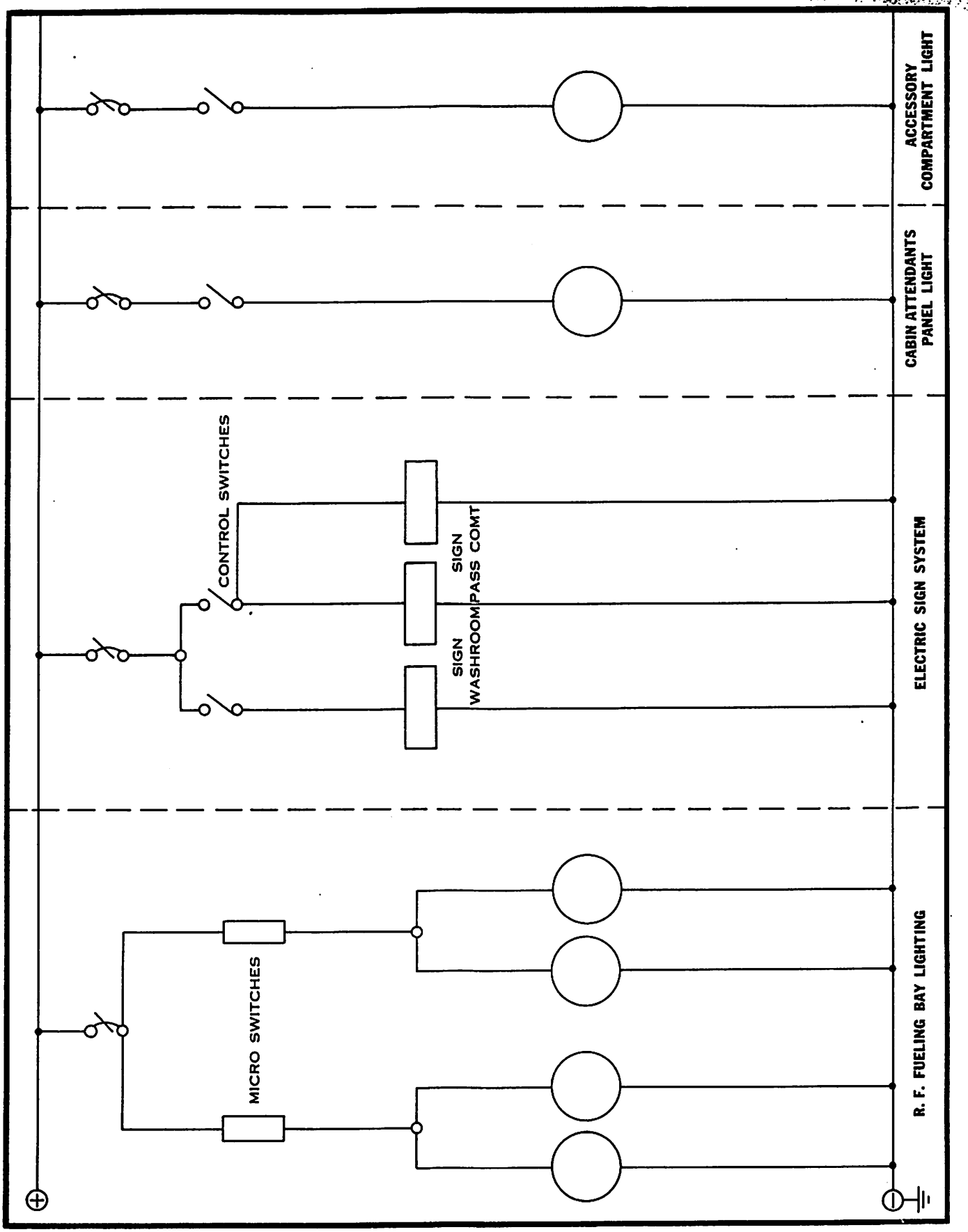


WIRING DIAGRAM FOR COMPLETE ELECTRICAL SERVICES
FIG. 43(8)

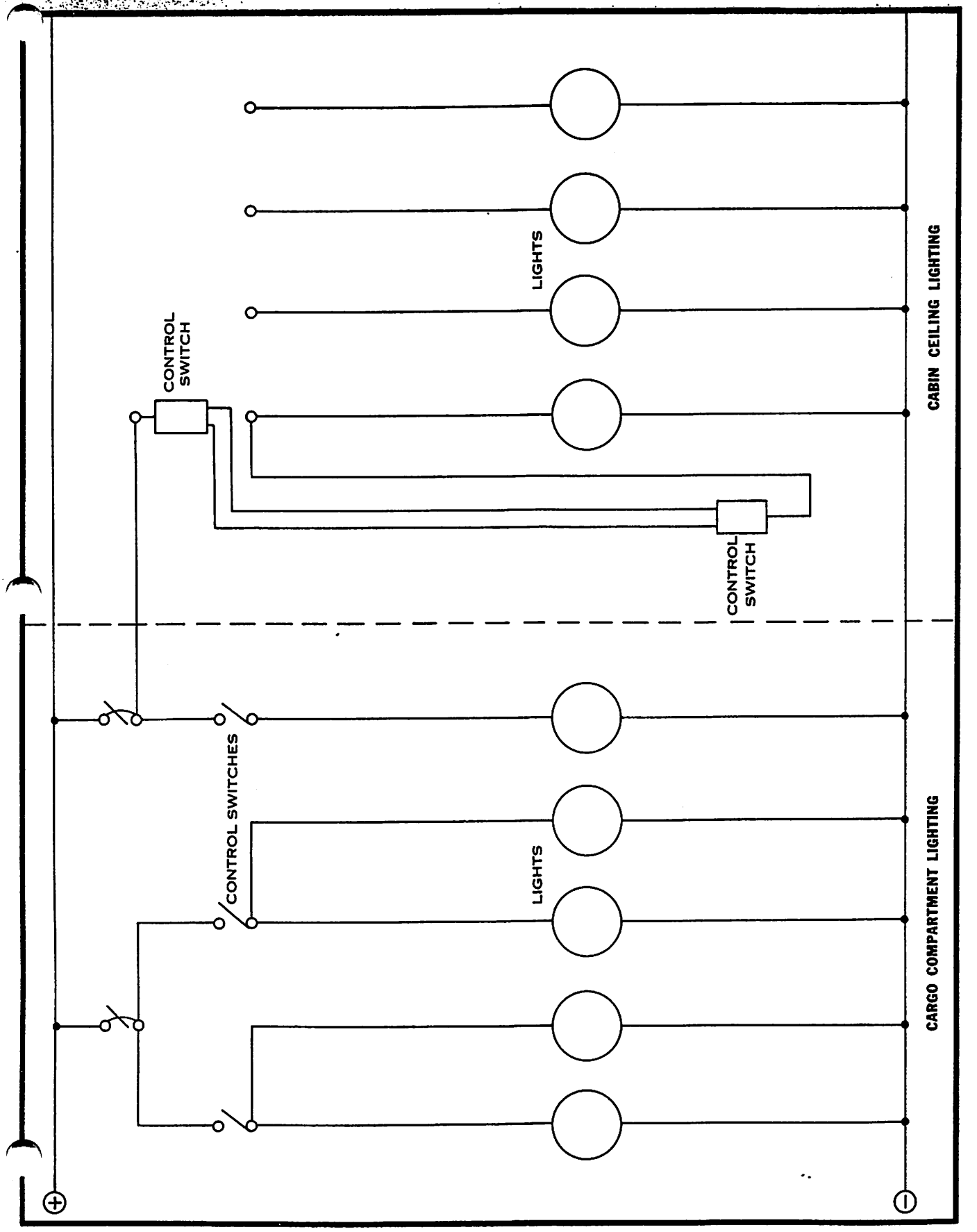


WIRING DIAGRAM
FOR
COMPLETE ELECTRICAL
SERVICES
FIG. 43(h)

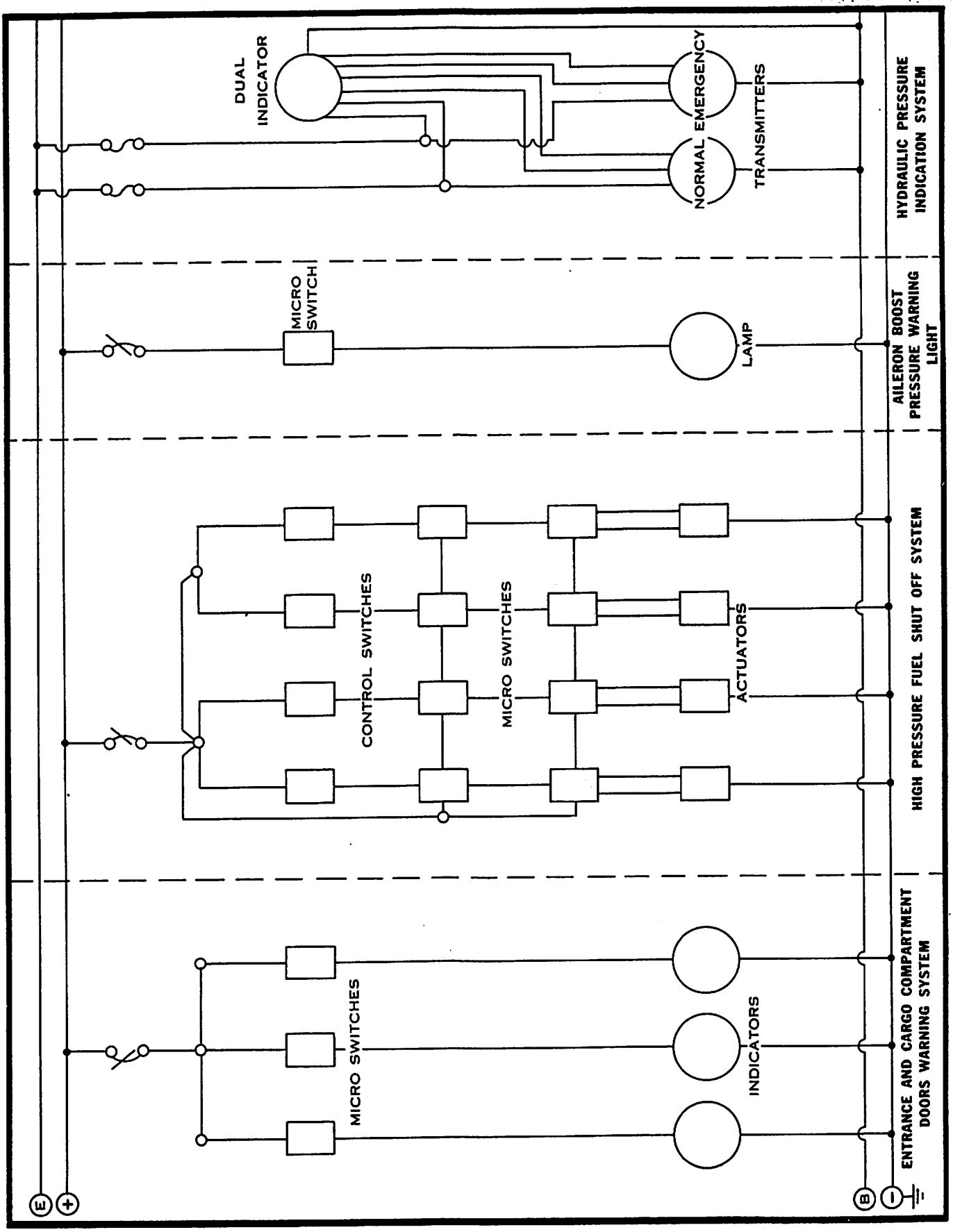
WIRING DIAGRAM
FOR
COMPLETE ELECTRICAL
SERVICES
FIG. 43(j)



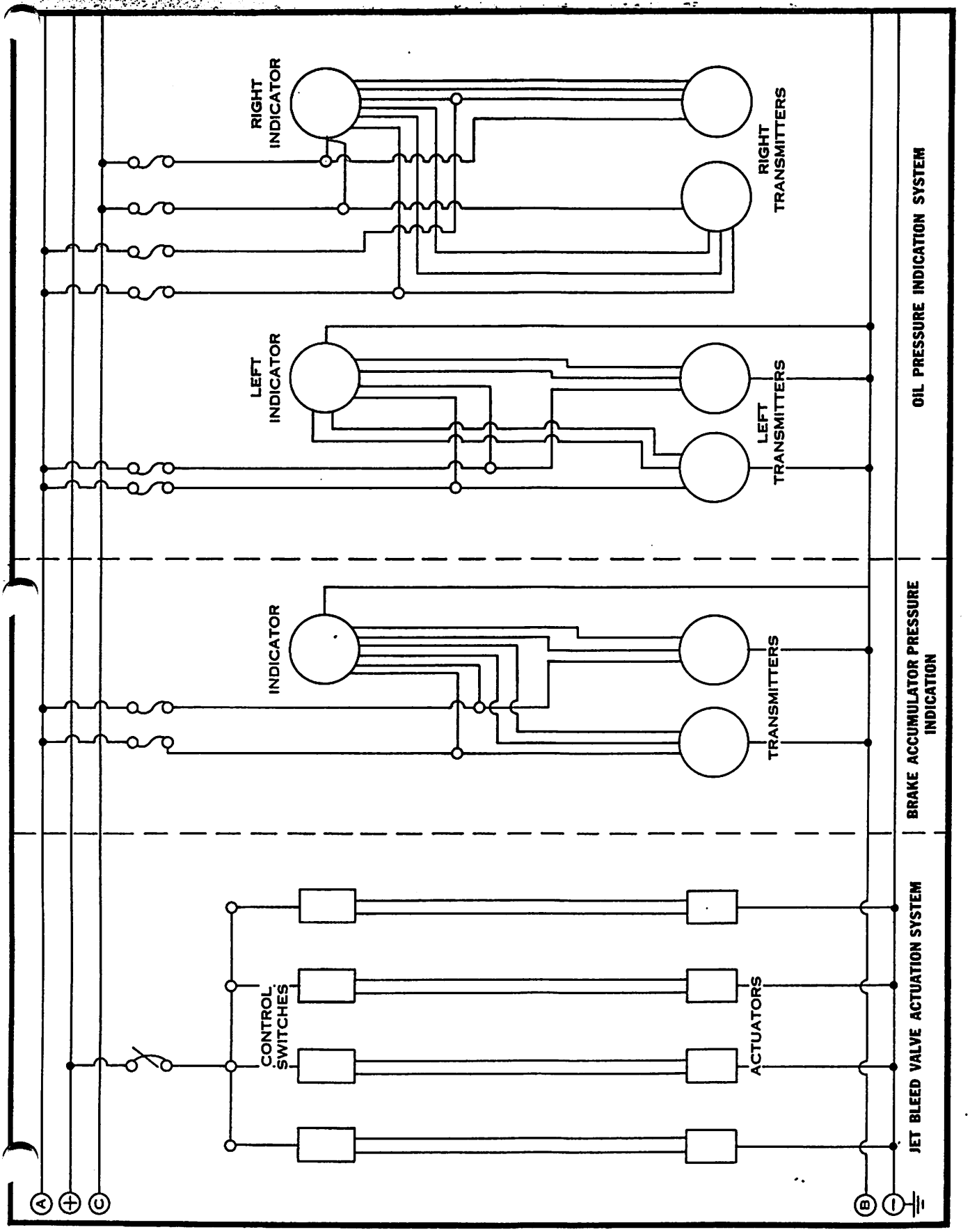
WIRING DIAGRAM
FOR
COMPLETE ELECTRICAL
SERVICES
FIG. 43(k)

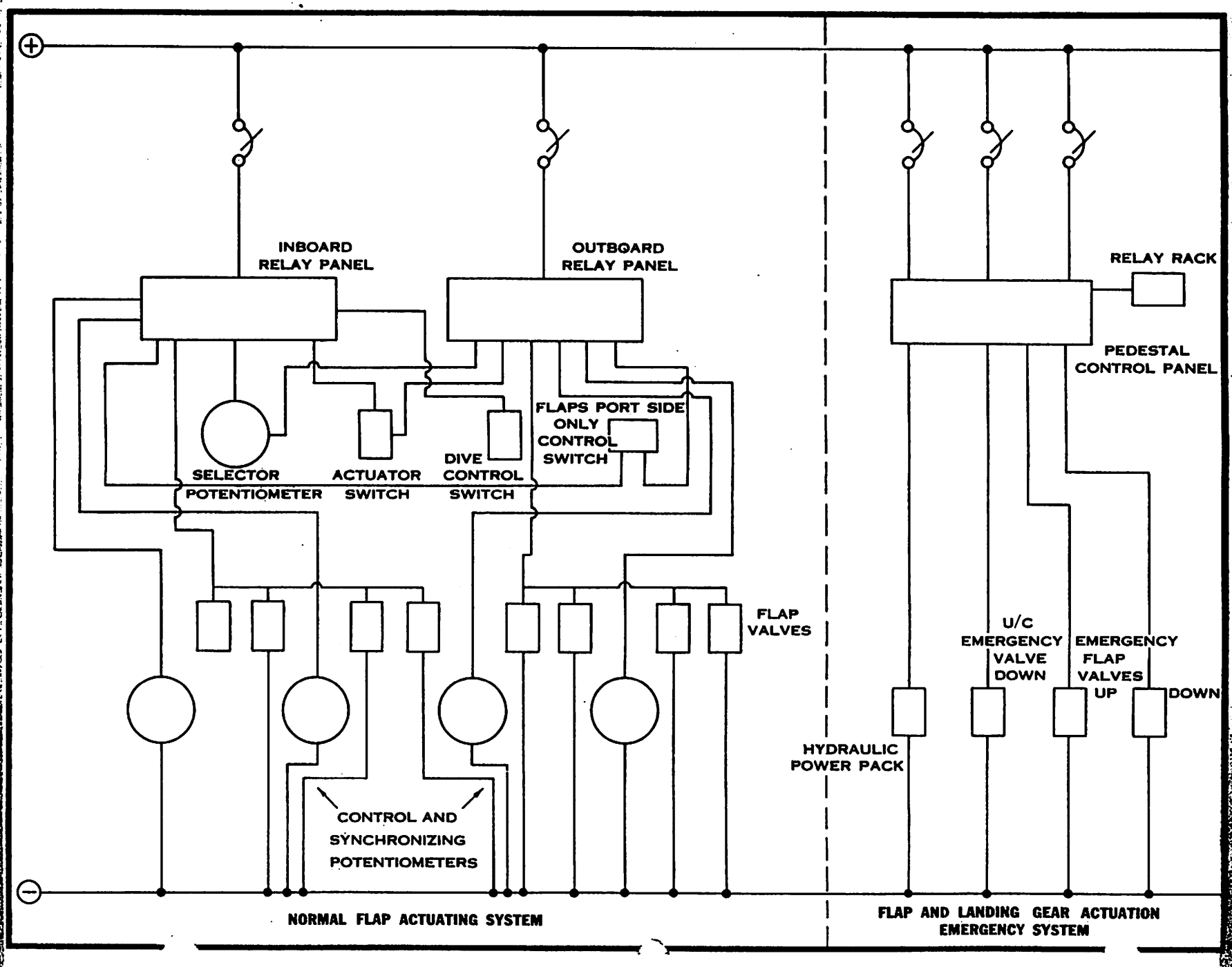


WIRING DIAGRAM
FOR
COMPLETE ELECTRICAL
SERVICES
FIG. 43(L)



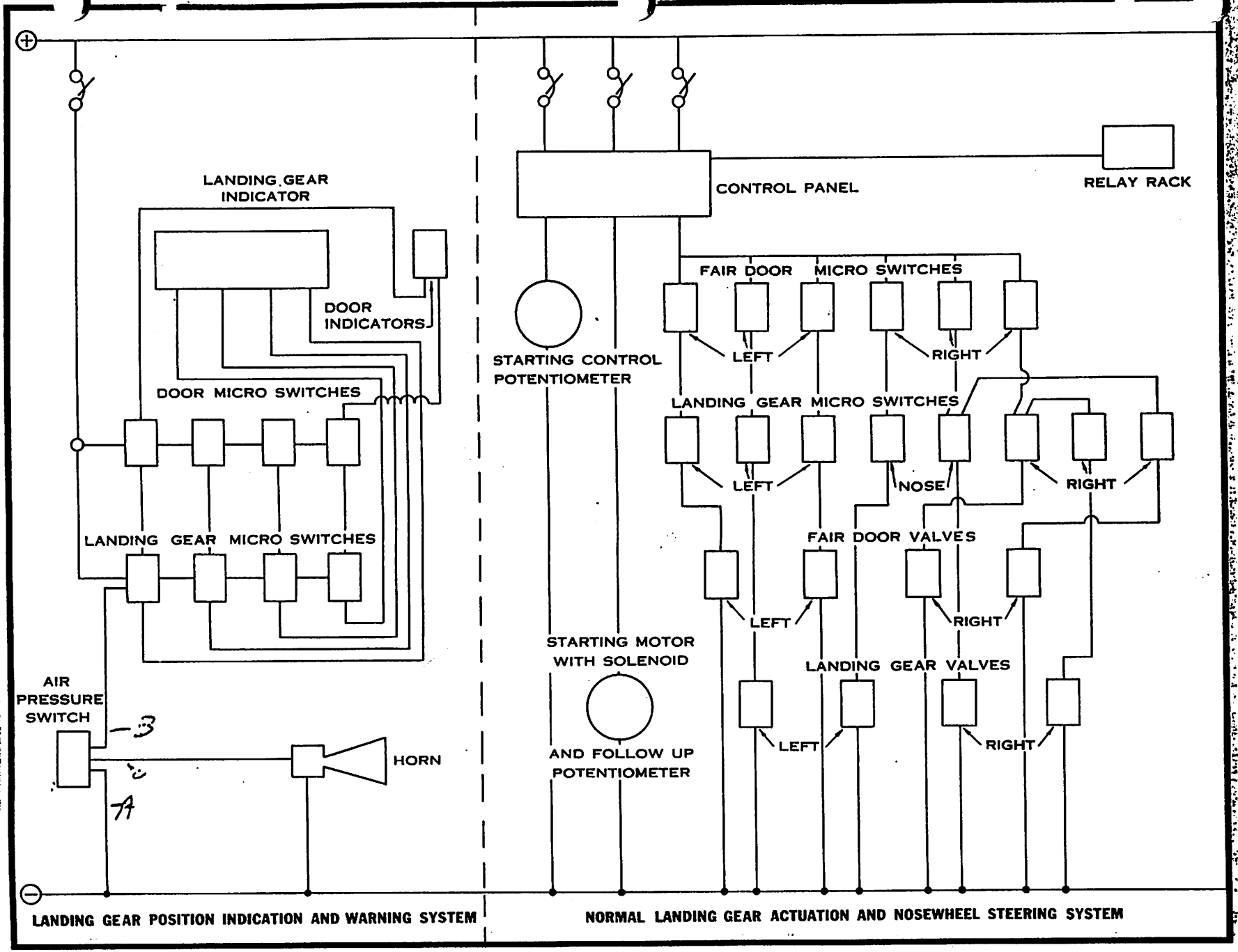
WIRING DIAGRAM
FOR
COMPLETE ELECTRICAL
SERVICES
FIG. 43(m)



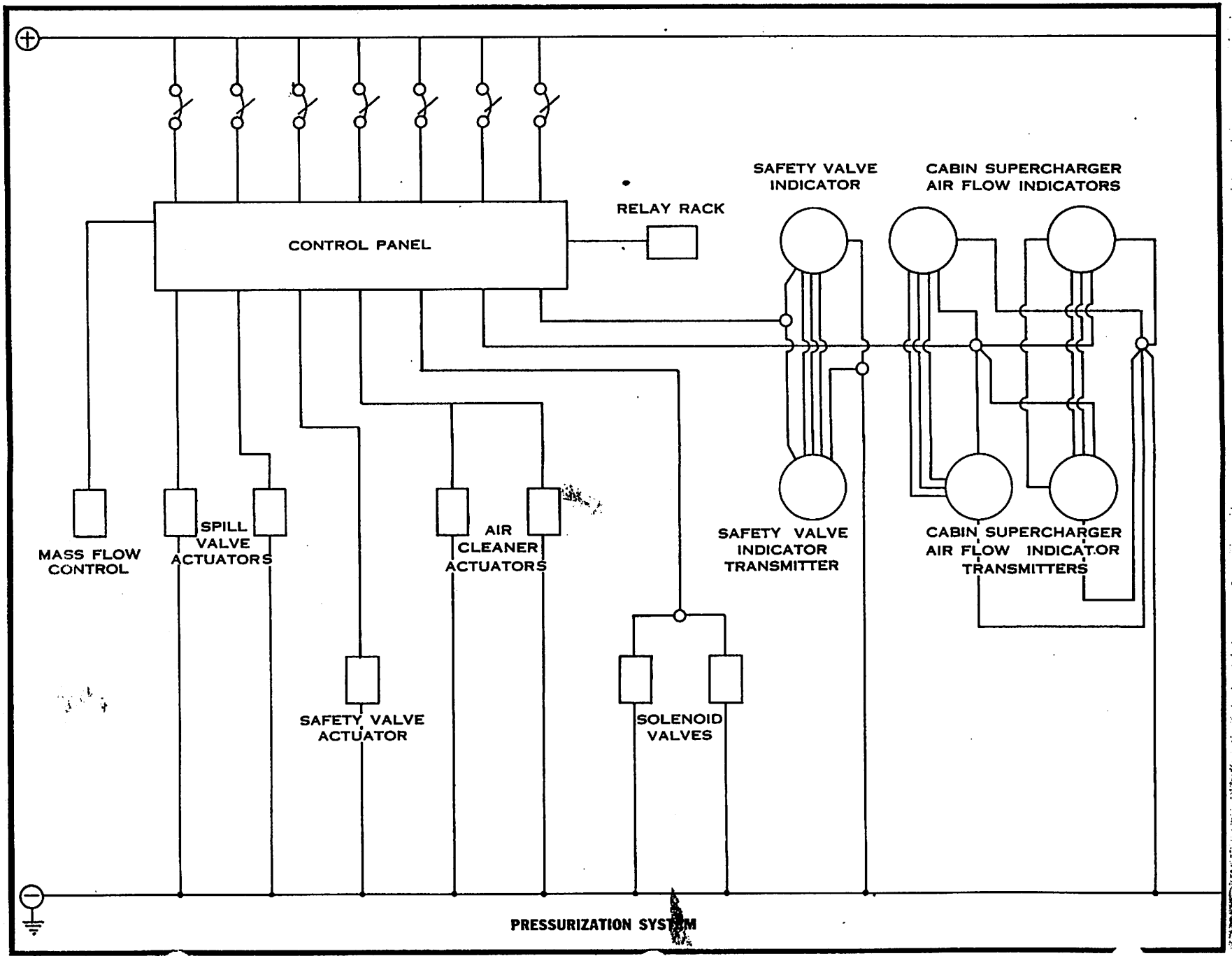


WIRING DIAGRAM FOR COMPLETE ELECTRICAL SERVICES
FIG. 43(n)

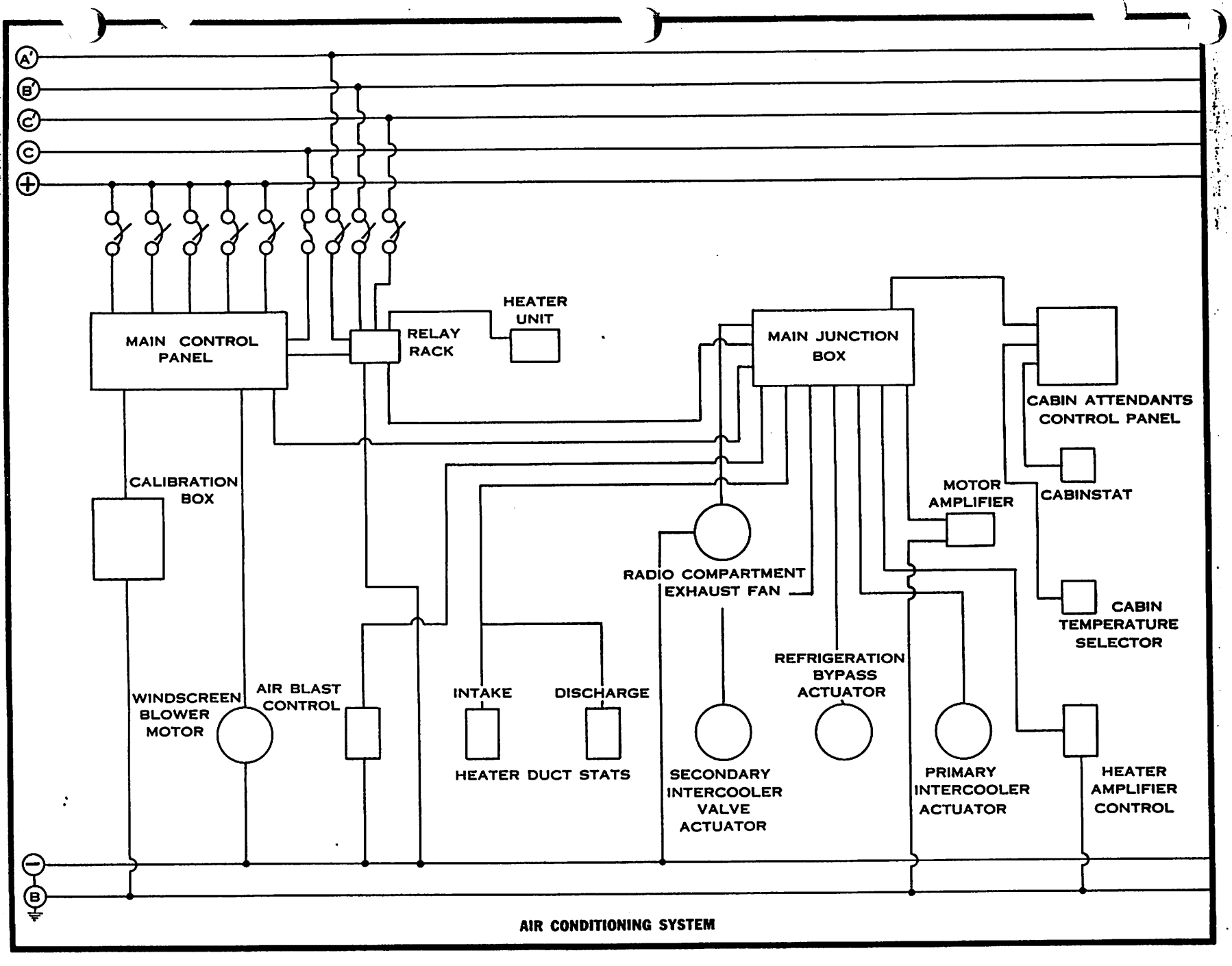
WIRING DIAGRAM
FOR
COMPLETE ELECTRICAL
SERVICES
FIG. 43(o)



JETLINER Flight Manual
TYPE C-102

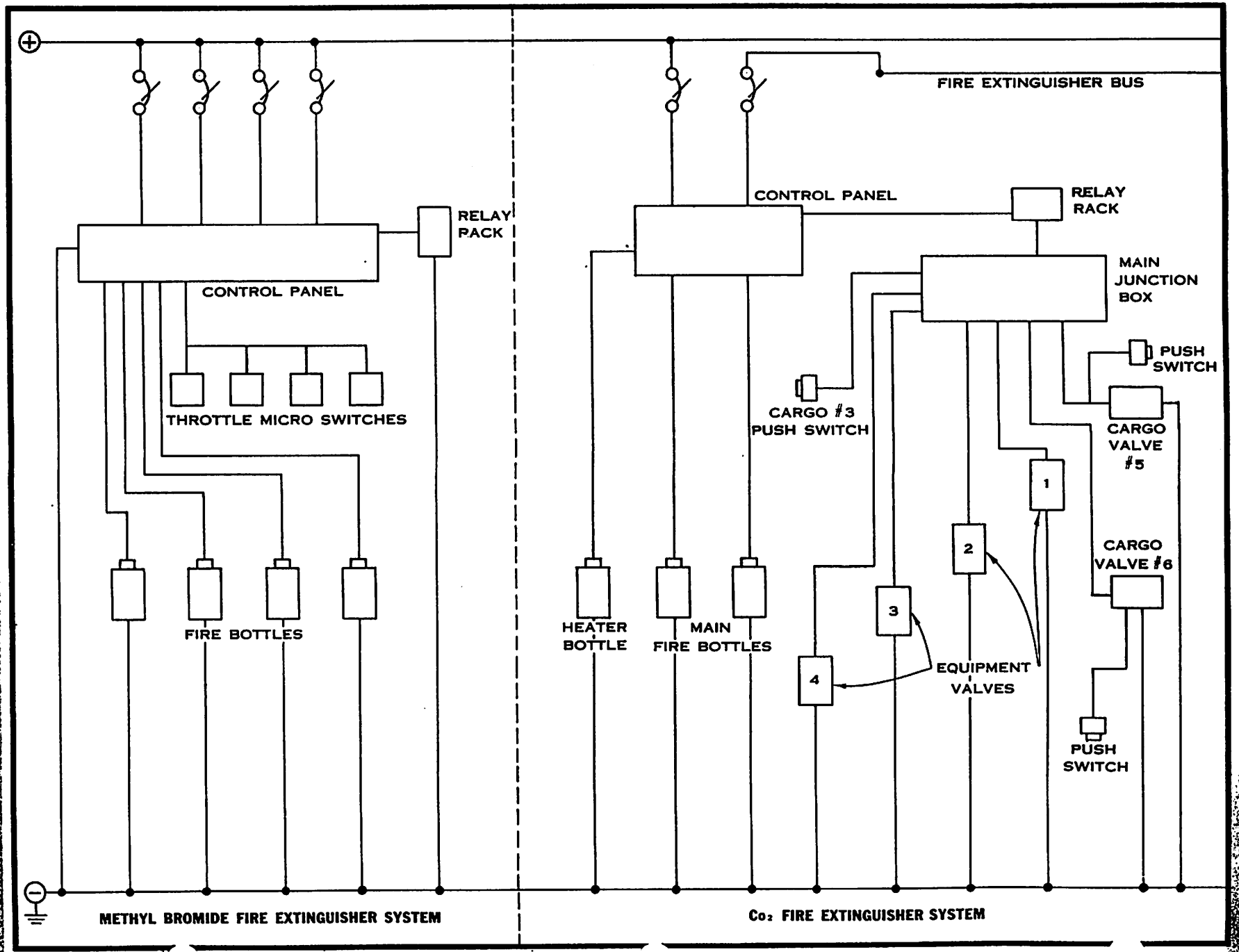


WIRING DIAGRAM
FOR
COMPLETE ELECTRICAL
SERVICES
FIG. 43(p)



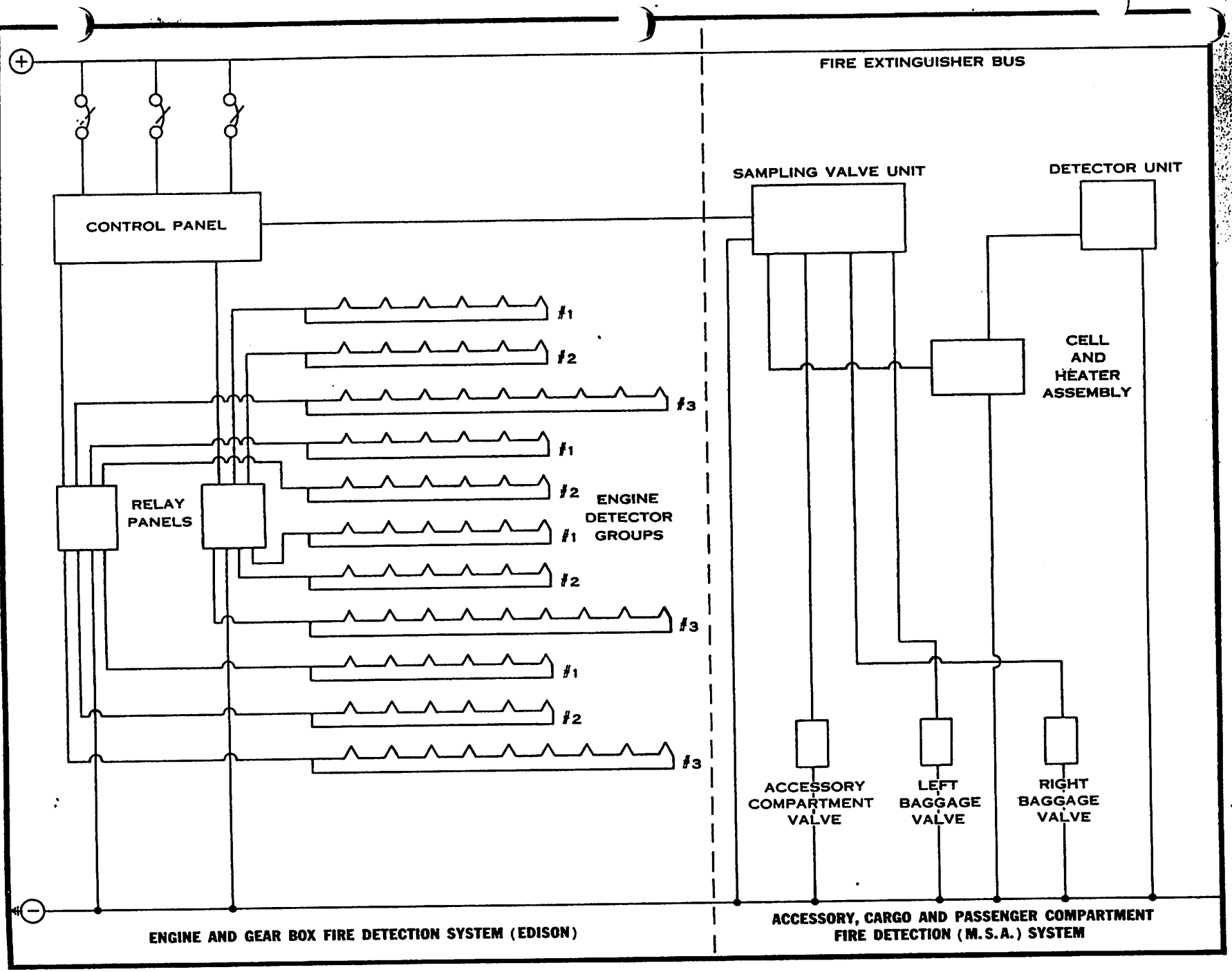
AIR CONDITIONING SYSTEM

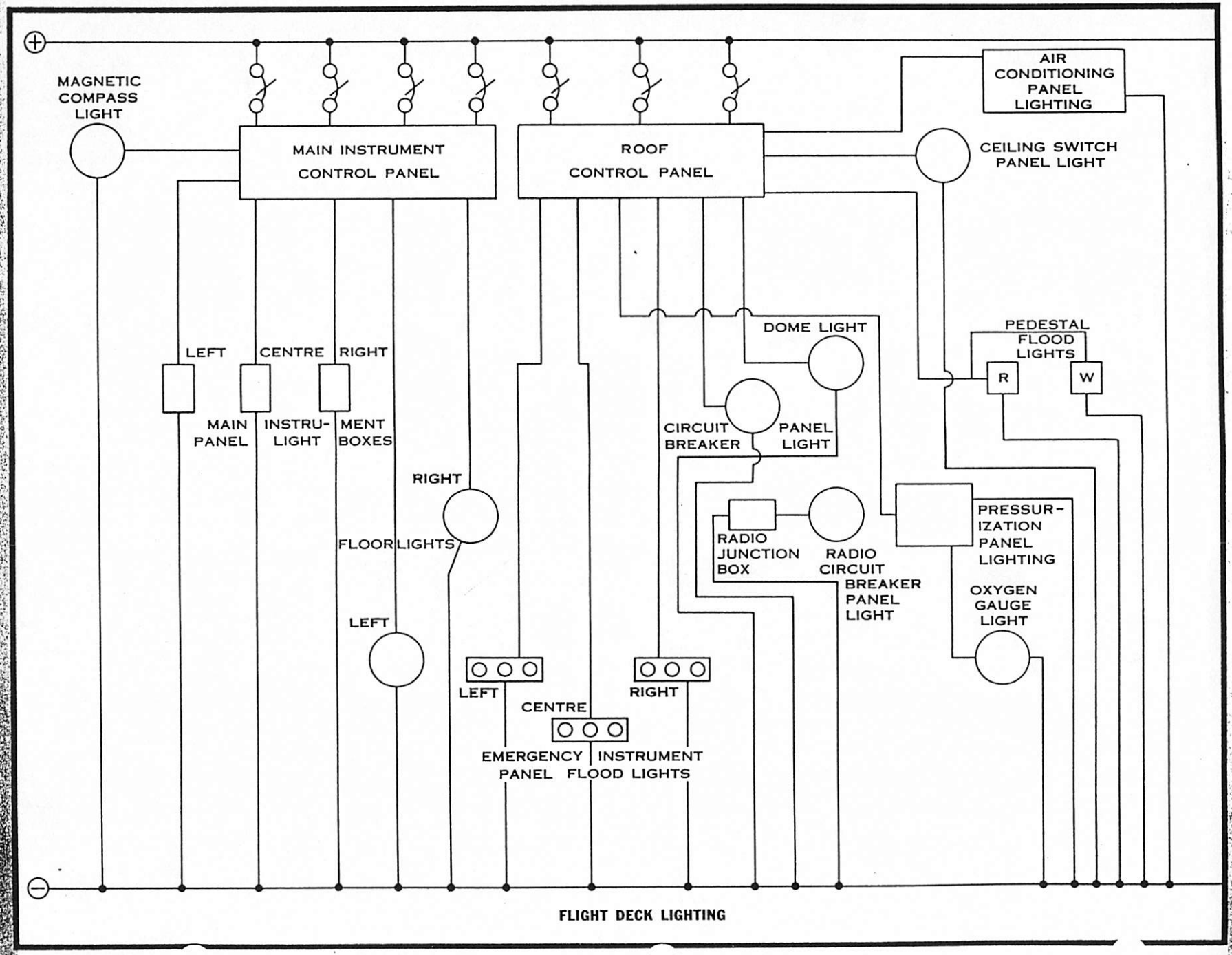
WIRING DIAGRAM FOR COMPLETE ELECTRICAL SERVICES
FIG. 43(9)



WIRING DIAGRAM FOR COMPLETE ELECTRICAL SERVICES
FIG. 43(r)

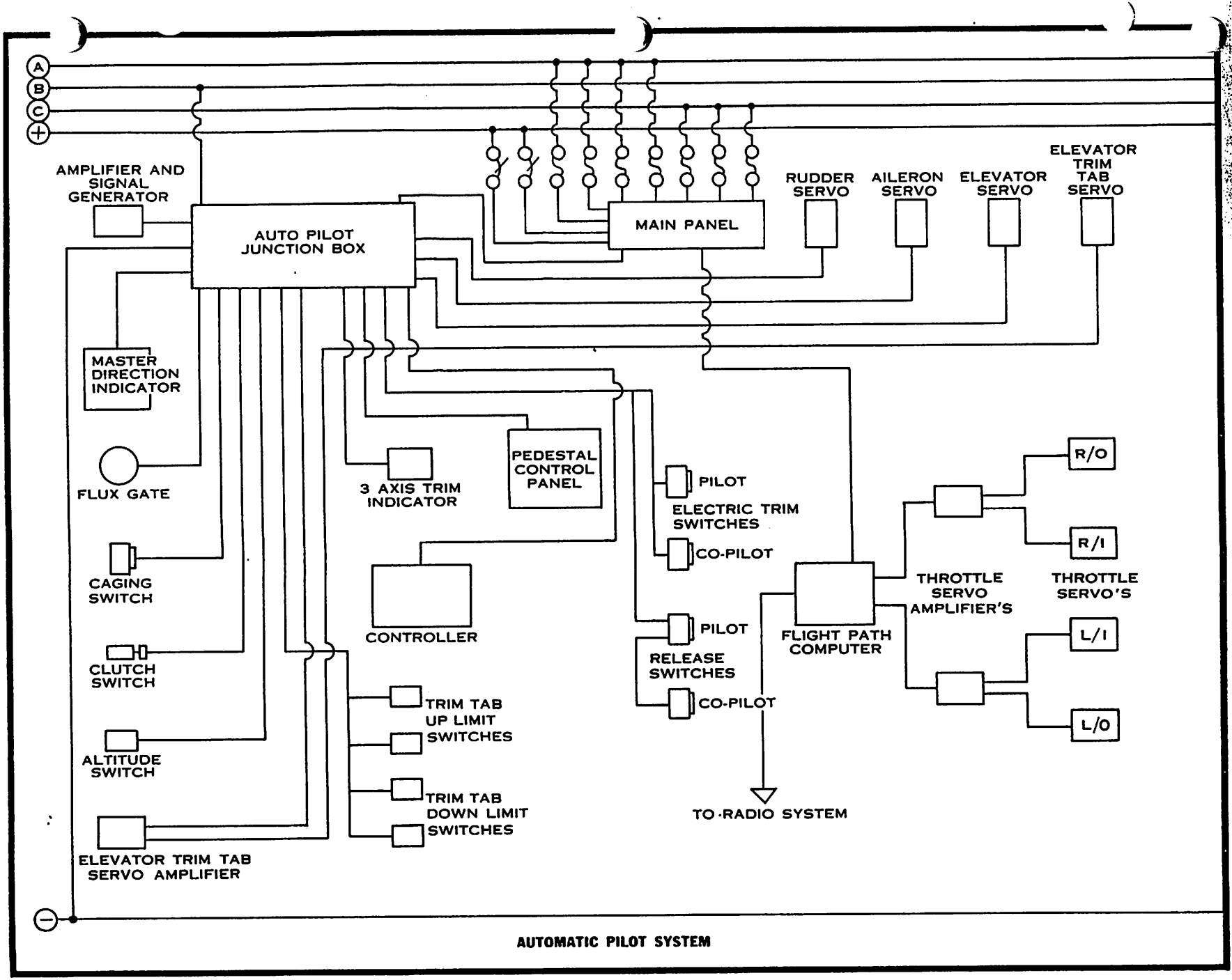
WIRING DIAGRAM
FOR
COMPLETE ELECTRICAL
SERVICES
FIG. 43(8)





FLIGHT DECK LIGHTING

WIRING DIAGRAM FOR COMPLETE ELECTRICAL SERVICES
FIG. 43(t)



AUTOMATIC PILOT SYSTEM

WIRING DIAGRAM FOR COMPLETE ELECTRICAL SERVICES
FIG. 43(u)

400-CYCLE LOAD DISTRIBUTION

No. of Fuse at Load	PHASE "A" VOLTAGE	No. of Fuse at Load	PHASE "C" VOLTAGE	INVERTERS No.1 Normal 115V 400 BUS	INVERTERS No.2 Emergency 115V 400 BUS
		3 ✱	A.P. Electric Trim	✱	
		1 ✱	C2 Gyrosyn Compass	✱	
		1 ✱	Air Conditioning System	✱	
		4 ✱	Fuel Contents Ind.	✱	✱
		1 ✱	Flux Gate Compass	✱	✱
3 ✱	A.P. & Electric Trim			✱	
1 ✱	Radio Supply			✱	✱
1 ✱	C2 Gyrosyn Compass			✱	
1 ✱	Nesaglass Deicing			✱	✱
4 ✱	Fuel Contents			✱	✱
1 ✱	Flux Gate Compass			✱	✱
				26V BUS	26V BUS
		2 ✱	Turn & Bank	✱	✱
		1 ✱	Flap Indicator	✱	✱
		2 ✱	Gyro Horizon	✱	✱
		2 ✱	Hydraulic Press.	✱	✱
		2 ✱	Oil Press. Indicator	✱	✱
1 ✱	Elev. Boost			✱	✱
2 ✱	Turn & Bank			✱	✱
2 ✱	Attitude Gyro			✱	✱
5 ✱	Oil Press. Ind.			✱	✱
2 ✱	Brake Accum. Press.			✱	✱

✱ Denotes alternative sources of power supply.

1-101 WIRING:

(1) All wiring is to AN Specifications and is routed through an aluminum conduit system, the latter being connected to suitably located control panels and junction boxes.

(2) Where the wiring terminates from the various conduits, panels or junction boxes, it is routed through plastic tubing or flexible aluminum metal conduit.

(3) Wiring to AN-J-C-48A specification is used throughout the aircraft, except in high temperature regions where Deltabeston silicone covered wiring is used, and on all high voltage circuits where Flamenol to TRB 1/32 specification is used.

(4) Break points are provided by standard AN connectors at the wing roots and at junction box E6 below the flight deck floor.

(5) An air conditioned accessories compartment houses all generator controls, relay and fire detection controls, and auxiliary actuator relay controls.

(6) In addition to providing an easily accessible and compact installation, provision is made for the testing of almost all circuits travelling between the various panels and junction boxes. Almost all circuits terminate in a master terminal board in the accessories compartment.

(7) Special flexible rough usage wiring is used throughout the landing gear system.

(8) All wiring and conduits are attached to junction boxes and panels with AN fittings.

(9) Circuit protection is provided by circuit breakers and limiters on the 28.5 volts DC and 208 volt AC systems. Fuses protect all 400 cycle and some variable frequency equipment.

1-102 AIR CONDITIONING AND CABIN HEATING: (GENERAL)

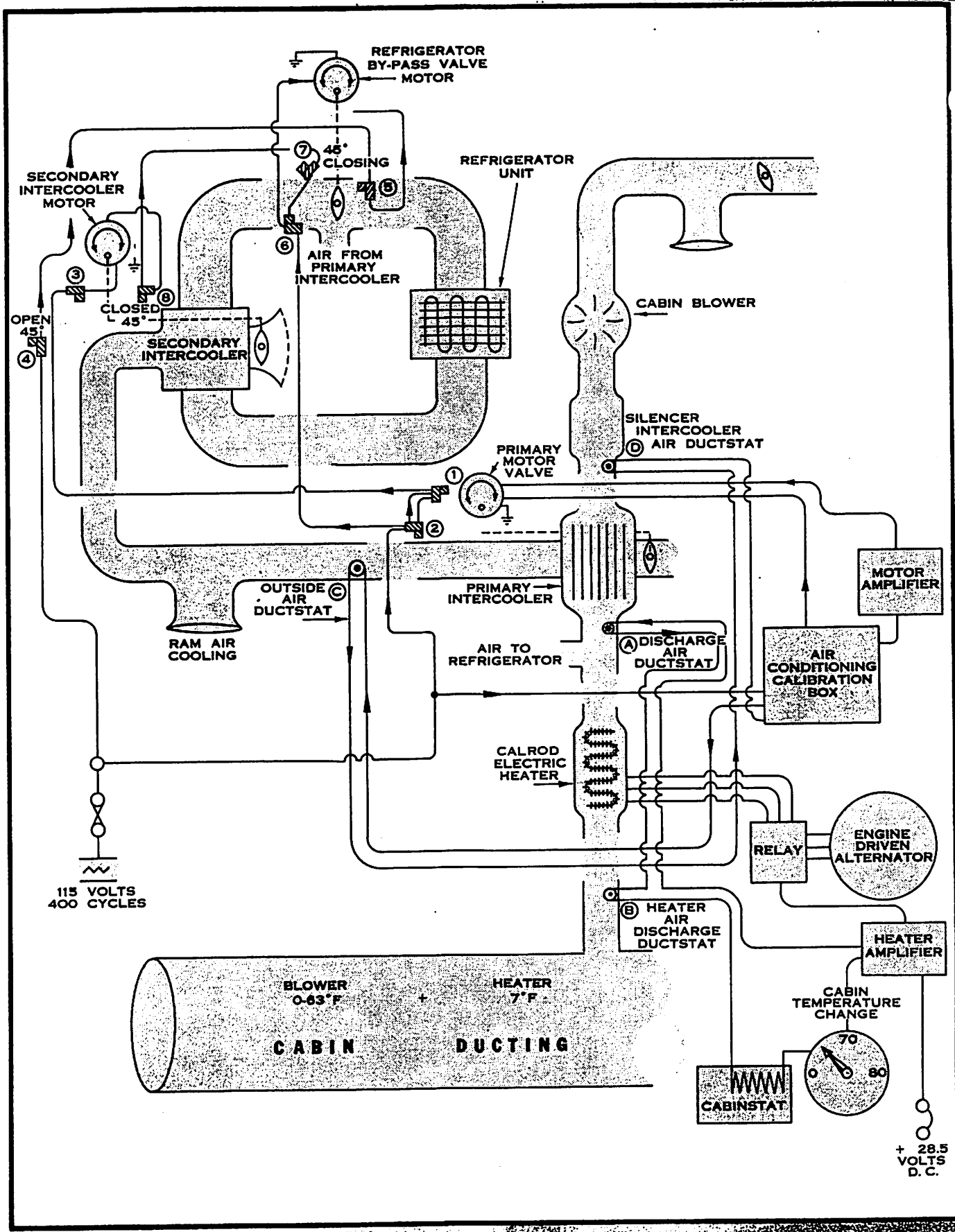
(1) The cabin heating and air conditioning systems are controlled from the master switch located on the air conditioning panel adjacent to the co-pilot's seat.

(2) A supply of AC power at 115 volts from the 400 cycle system and 28.5 volts from the DC supply, serves the air conditioning circuit. The heater operates on a three phase variable frequency of 208 volts.

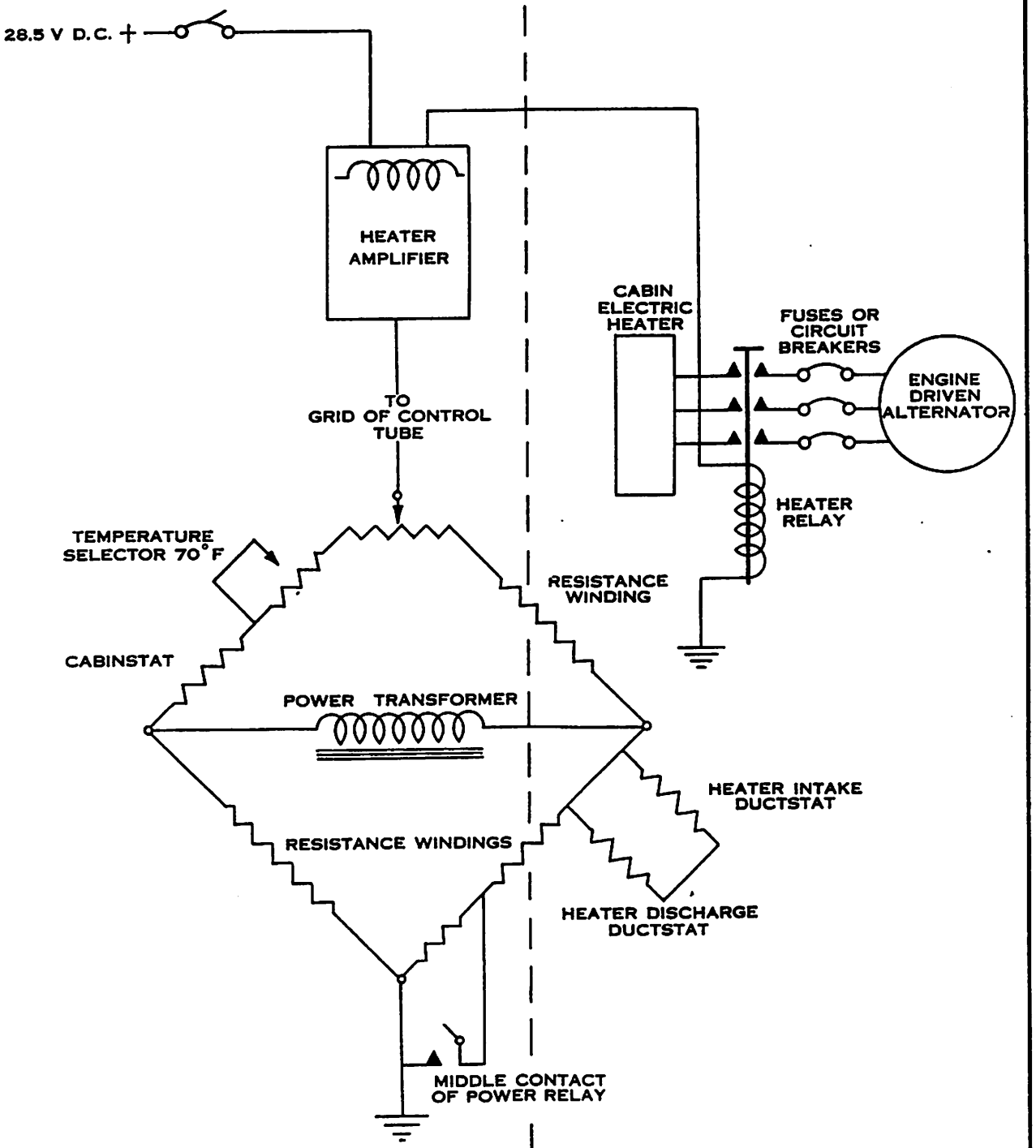
(3) With the control switches set in their correct operating sequence, signals will be transmitted to the control units. These units comprise a motor amplifier, heater amplifier and a calibration junction box.

WIRING DIAGRAM
AIR CONDITIONING
& CABIN HEATING

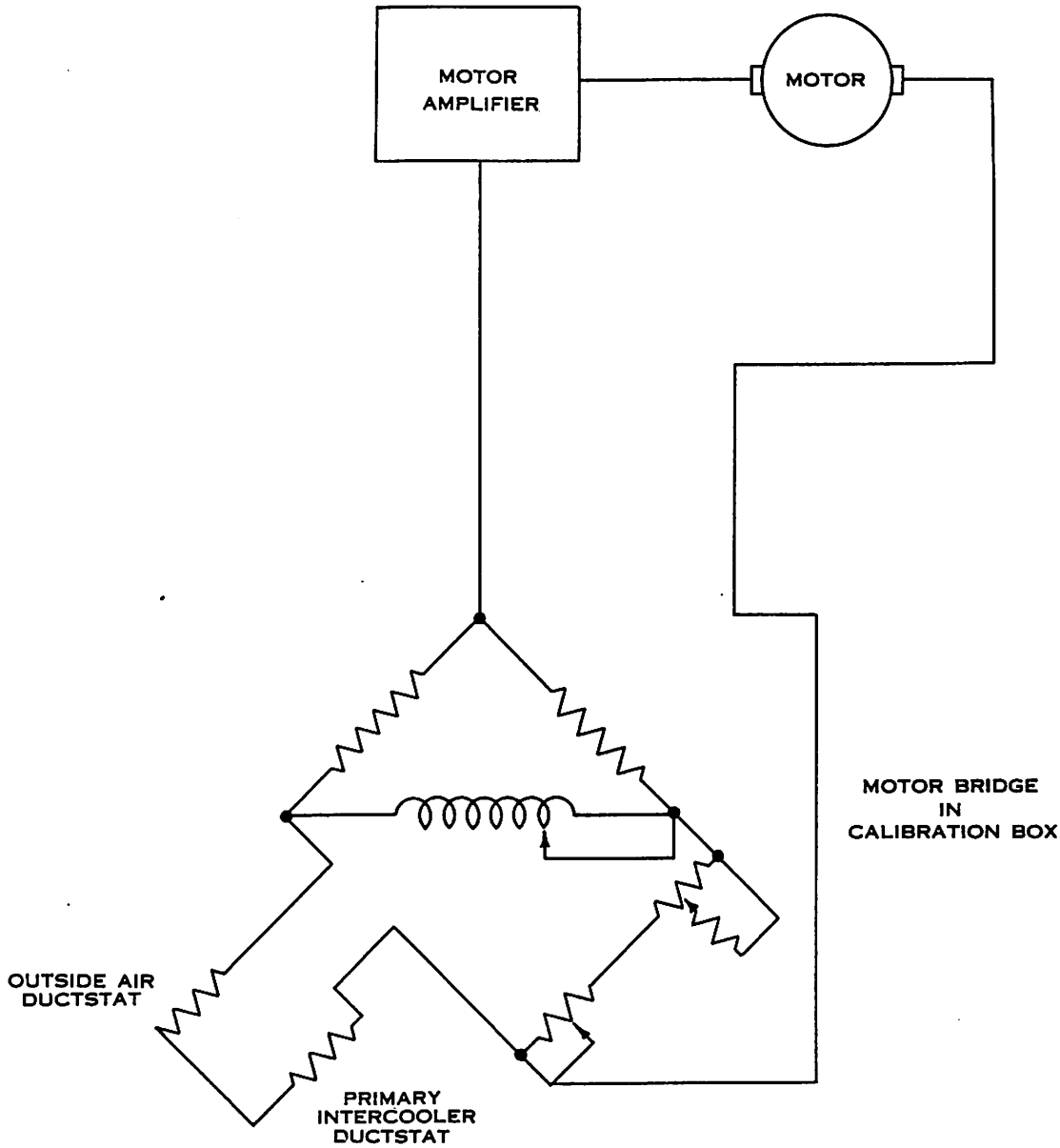
FIG. 44



WIRING DIAGRAM
CABIN HEATING
BRIDGE CIRCUIT
FIG. 45



WIRING DIAGRAM
AIR CONDITIONING
BRIDGE CIRCUIT
FIG. 46



(4) Variable adjustment of these units is through the master switch on the cabin attendant's panel.

(5) To carry out a complete functional test of the system, air must be flowing through the ducts at a minimum flow of 30 lbs. per minute. It is advisable however, to carry out this test in the air, as true operating conditions cannot be simulated on the ground.

(6) In the event of automatic failure, provision is made for manual control of cabin heat.

1-103 CONTROL SETTING:

(1) Assuming that aircraft power supply is available, and that the master switch previously described is placed in the ON position, a supply of AC and DC current will flow to the control units.

(2) To provide for automatic control, it is necessary to adjust the component switches located on the attendant's control panel.

(3) The positioning and sequencing of the switches are as follows:

(a) An emergency switch to be placed in the OFF or closed position.
(b) A three-way manual control switch to be left in the OFF position.
(c) An automatic switch to be placed in the AUTO position.
(d) An emergency heat switch to be left in the OFF position.
(e) A variable selector switch to be set at 70 degrees.

1-104 HEATING CYCLE:

(1) Assuming cabin temperature is below 70 degrees F., signals from ductstats A and B (Fig. 45) are transmitted to the heater amplifier.

(2) With the signals calling for "more heat" the heater amplifier bridge system becomes unbalanced and effects the grid of the control tube.

(3) The signal voltage from the grid control tube is then amplified through a resistor condenser network, coupling between the output of the control tube and the input of the output tube.

(4) The amplified signal appears in the plate circuit of the output tube and operates a sensitive relay connected in series between the output tube plate and the plate positive supply.

(5) The sensitive relay, when actuated by the vacuum tube, closes its contacts and applies 28 volts DC to the cycling relay for the electric heater (Fig. 45).

(6) The electric heater air output then raises cabin temperature to 70 degrees F., changing the resistance of the cabin temperature selector bridge section and dropping the signal voltage of the amplifier to zero.

(7) Dropping of this voltage, reduces the amplified signal in the plate circuit of the output tube to zero, causing the sensitive relay contacts to open, thereby disconnecting the power supply to the heater through the opening of the heater cycling relay.

1-105 AIR CONDITIONING CYCLE:

(1) Assuming a drop in outside ambient, and a resulting drop in cabin temperature to approximately 68 degrees F., signals will be transmitted to the air conditioning bridge system and cause it to become unbalanced.

(2) Since the outside air ductstat C resistance changes with the change in ambient, a signal is applied to the motor amplifier control grid. This signal is amplified through a resistance condenser network to a phase discriminator tube. The latter is required in order to relay the correct phase voltage to the primary intercooler motor for the cranking of the valve (Fig. 46). In this instance the valve will close, as the signal is calling for more heat.

(3) The air temperature will then rise on the output side of the primary intercooler and alter the resistance of the intercooler air ductstat D. This will cause a bridge balance, nullifying the signal to the grid of the control tube in the motor amplifier, and maintain the desired valve setting.

1-106 COOLING CYCLE:

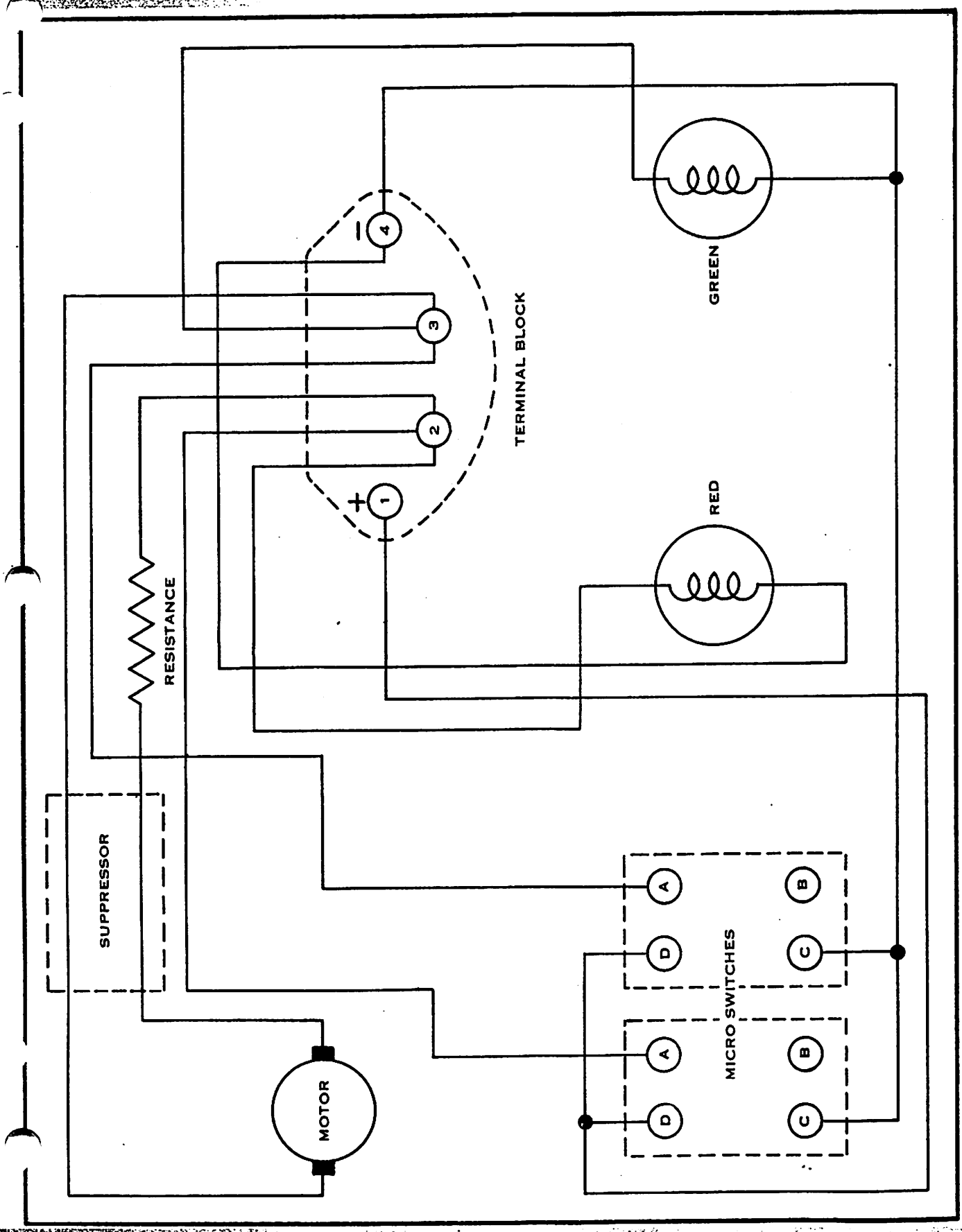
(1) Assuming an outside ambient above 70 degrees F., the resistance of the outside air ductstat C will change. This unbalances the bridge in the calibration junction box, and a signal is transmitted to the motor amplifier calling for more cooling.

(2) The motor amplifier reverses the phase voltage to the primary intercooler motor to crank the valve open.

(3) As soon as the primary intercooler valve reaches the 45 degree open position, the valve trips micro-switch (1) and applies power to the opening phase of the secondary intercooler motor through micro-switch (3).

(4) As the secondary intercooler valve reaches the 45 degree open position, micro-switch (4) is tripped, and power is applied through micro-switch (5) to crank the refrigerator valve to the fully open position.

(5) Conditions described in this operation are maintained as long as the signal calls for more cooling.


GREEN
TERMINAL BLOCK
4
3
2
1
+
RED
RESISTANCE
SUPPRESSOR
A
B
D
C
MICRO SWITCHES
A
B
D
C
MOTOR

(6) When the outside ambient temperature decreases, the outside air ductstat will call for less cooling, and a signal will be passed to the control grid in the motor amplifier, resistance condenser network and phase discriminator tube as previously described, and the voltage relayed to the primary intercooler motor (Fig. 45).

(7) When the primary intercooler motor closes the valve, the adjacent micro-switch (2) is tripped, the latter applying power through micro-switch (6) to the closing phase of the refrigerator by-pass valve motor.

(8) As the refrigerator by-pass valve reaches the 45 degree closing position, the micro-switch (7) is tripped, and voltage flows to the closing phase of the secondary intercooler valve through micro-switch (8).

(9) The final "more heat" cycle is reached, when the refrigerator by-pass valve is fully closed and the bridge system balances with the primary inter-cooler valve.

1-107 PRESSURE CONTROLLER (INTERNAL WIRING):

(1) The details of the Pressure Controller internal wiring (Fig. 47) illustrates that the ends of the leads are fitted with numbered identification sleeves to correspond with those on the numbered terminal block. The unit operates on 24/29 volt direct current, with a maximum consumption of 0.5 amps. Connections from the source of supply are made through a master switch and fuse; positive being connected to terminal No. 1 and negative to No. 4.

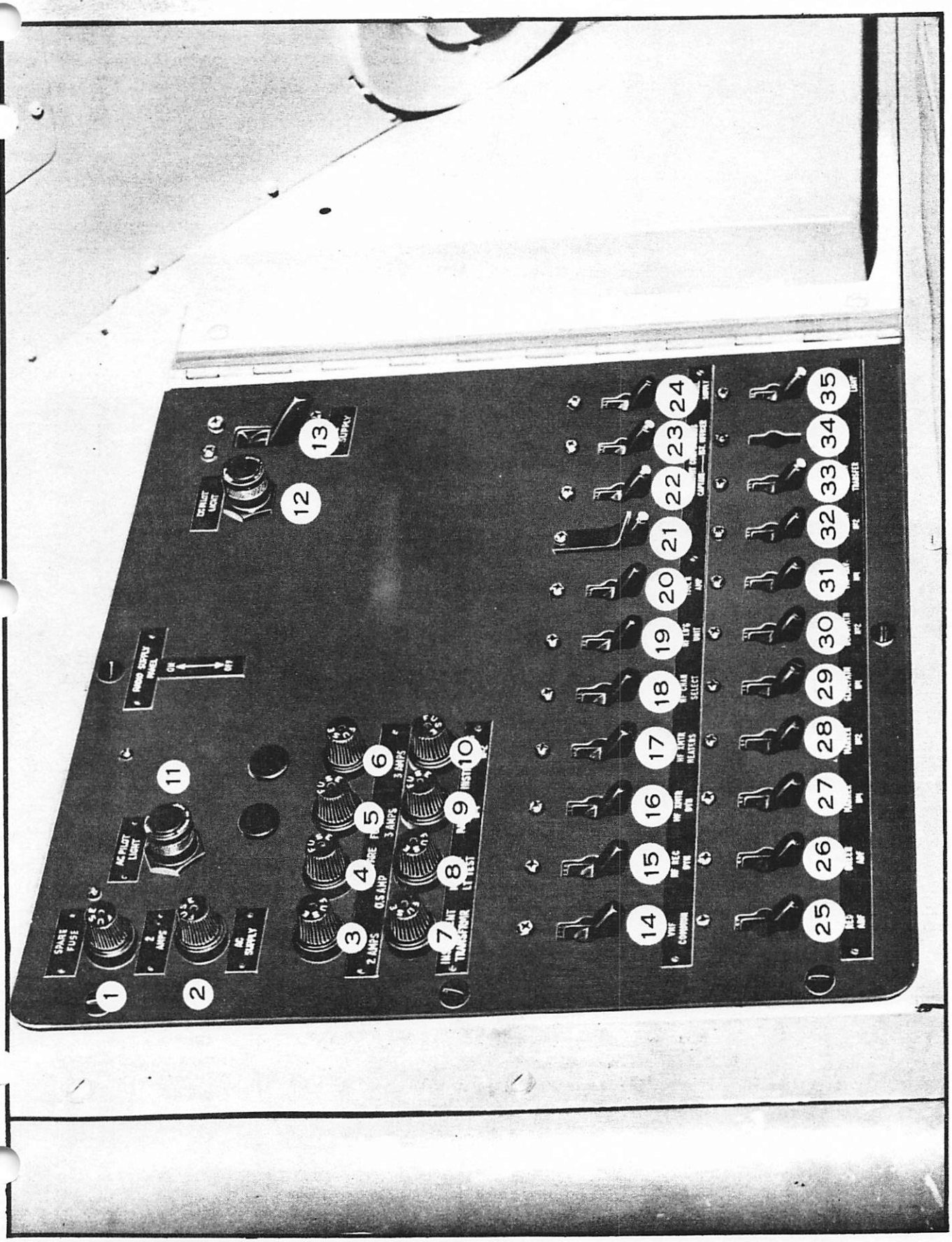
(2) When the selector and camshaft mechanisms are stationary neither micro switch plungers are depressed and the circuit is through A to C in each case, with the result that the positive feed to the electric motor is broken, negative is fed to both sides of the electric motor, and both indicator lamps are out of circuit.

(3) When the cabin altitude selector is set to increase the cabin altitude, the switch mechanism causes the left hand micro-switch circuit to be through A to D so that positive is fed to one side of the electric motor, instead of negative; the other micro switch circuit remaining through A to C. The electric motor rotates in the direction required to cause the camshaft mechanism to increase cabin altitude and the red indicator lamp is placed in circuit.

(4) When the cabin altitude selector is set to decrease the cabin altitude, the switch mechanism causes the right hand micro-switch circuit to be through A to D, feeding positive to the other side of the electric motor, the other micro switch having reverted to a circuit through A to C to feed negative to the electric motor. The electric motor operates the camshaft mechanism in the direction required to decrease cabin altitude and the green indicator lamp is placed in circuit.

(5) When the required change in cabin altitude has been achieved the circuit

RADIO POWER
SUPPLY PANEL
FIG. 48



KEY TO FIG. 48 RADIO POWER SUPPLY PANEL

FUSES:

1. Spare Fuse - Radio AC Supply
2. Main Fuse Radio AC Supply
3. Spare Fuse Instrument Transformer AC Supply
4. Spare Fuse Marker Light Test Transformer AC Supply
5. Spare Fuse No. 1 Radio Navigational Instrument Supply
6. Spare Fuse No. 2 Radio Navigational Instrument Supply
7. Fuse - Radio AC Supply
8. Fuse - Marker Light Test Transformer AC Supply
9. Fuse - No. 1 Radio Navigational Instruments AC Supply
10. Fuse - No. 2 Radio Navigational Instruments AC Supply

PILOT LIGHTS:

11. Pilot Light - Radio AC Supply
12. Pilot Light - Radio DC Supply

CIRCUIT BREAKERS:

13. Main Circuit Breaker Radio DC Supply
14. VHF Communication Transmitter-Receiver
15. HF Communication Receiver Dynamotor
16. HF Communication Transmitter Dynamotor
17. HF Communication Transmitter Heaters
18. HF Communication Channel Selector Motor
19. HF Communication Antenna Loading Unit
20. Isolation Amplifier
21. Not Used
22. Transmit Control Relay - Pilot
23. Transmit Control Relay - Co-pilot
24. Pedestal Panels
25. Red ADF (Automatic Radio Compass) Receiver
26. Green ADF (Automatic Radio Compass) Receiver
27. Marker Beacon Receiver No. 1
28. Marker Beacon Receiver No. 2
29. Glide Path Receiver No. 1
30. Glide Path Receiver No. 2
31. VHF Localizer Omnirange Receiver No. 1
32. VHF Localizer Omnirange Receiver No. 2
33. ILS Indicator Transfer Relays
34. Not Used
35. Radio Power Panel Floodlight

reverts to that indicated in paragraph (2), the motor stops and the camshaft ceases to rotate.

NOTE

(1) Left and right refer to the switches as viewed in Fig. 47. The lettering of the terminals can be seen when the cover is removed. That marked 'B' is not used in this installation, and is filled with an insulating material.

1-108 RADIO EQUIPMENT: (GENERAL)

(1) The radio equipment is grouped in the radio compartment on the left side of the fuselage between the entrance door and the flight deck rear wall (Fig. 49). Access to the units and their relative connectors, is through the quick release panels on the side and rear walls of the compartment.

(2) All units are mounted on individual shock mounts. Most of the equipment is of the rear-connector type to facilitate the rapid changing of units. All units withdraw rearwards from the rack.

(3) The compartment is ventilated by means of a blower which provides a negative pressure relative to cabin atmosphere.

(4) The control panels are of the "plug in" type and are located on the flight deck control pedestal. The entire radio control assembly is a separate unit with disconnect plugs located at floor level. The individual panels may be removed without disturbing the radio control assembly. This arrangement is achieved by means of eject screws located near the centre of each panel. No flexible tuning shafts are used. Microphone and headphone jacks are faired into the side panels. Separate loudspeakers are provided for pilot and co-pilot. Communication with the cabin attendant is through the interphone system. The interphone system is also wired to jacks at the nosewheel well and external servicing points, the external interphone jacks being provided for use by ground personnel during ground maintenance and starting.

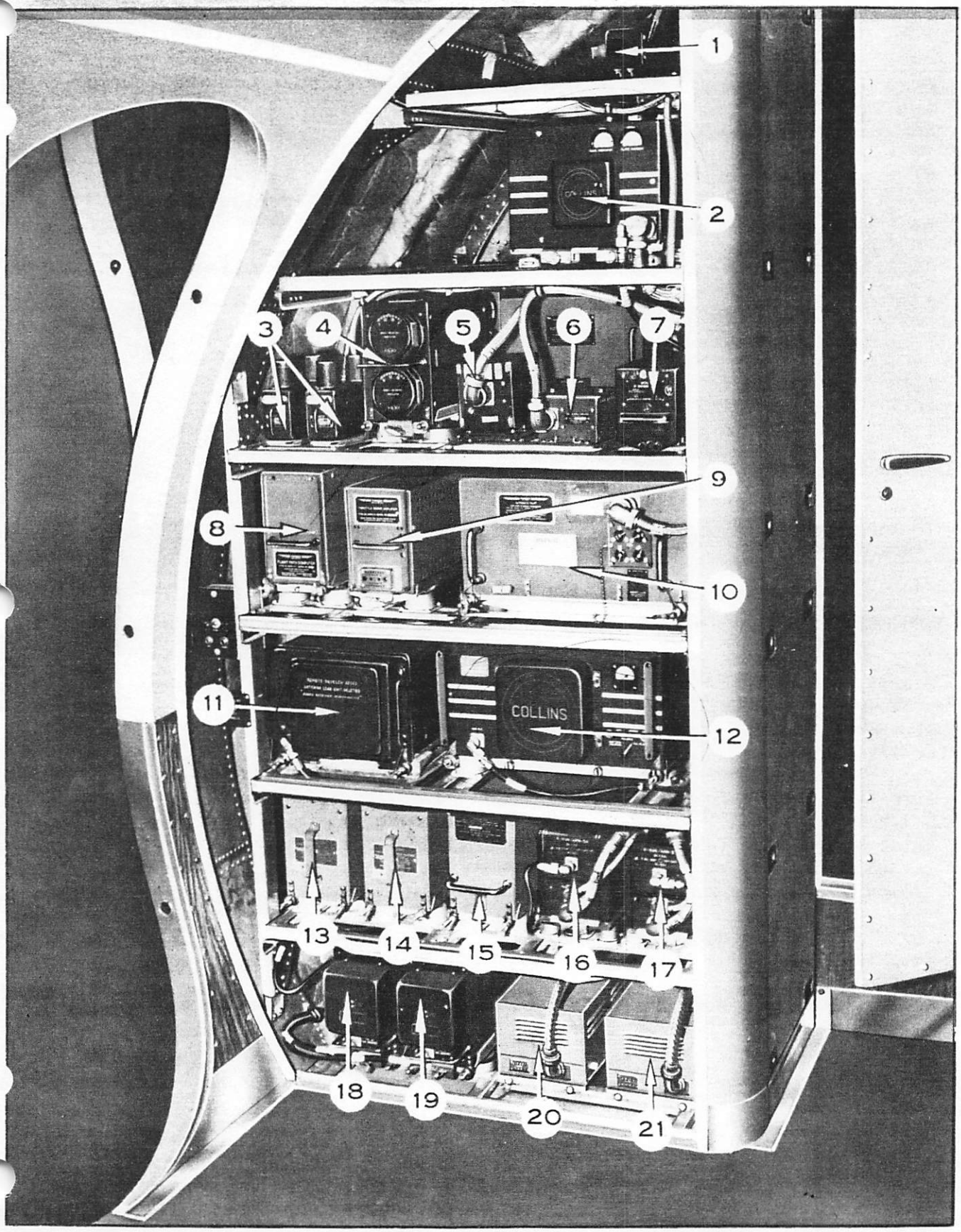
(5) All radio navigational instruments are duplicated on the pilot's and co-pilot's instrument panels.

(6) All wiring is carried in metal conduit and identified by coding. All relays are of the "plug in" type.

1-109 RADIO POWER PANEL AND RADIO JUNCTION BOX:

(1) The radio junction box and power panel forms the left rear wall of the flight compartment (Fig. 48). The junction box contains the terminal strips for interconnection of all components of the radio system, and such miscellaneous items as relays, range-voice filters, interphone amplifier volume control,

GENERAL ARRANGEMENT
OF
RADIO COMPARTMENT
FIG. 49



KEY TO FIG. 49 RADIO COMPARTMENT

1. AC Power Transformer for Radio Instrument Supply
2. HF Communication Antenna Loading Unit
3. Fuel Contents Indicator Power Units
4. VHF Navigation Accessory Unit
5. Auto-Pilot Pitch and Bank Rate Unit
6. Auto-Pilot Trim Tab Servo Amplifier
7. Amplifier-Air Conditioning System
8. Auto-Pilot Flight Path Computer
9. Throttle Servo Amplifier
10. Auto-Pilot Amplifier and Signal Generator
11. VHF Communication Transmitter-Receiver
12. HF Communication Transmitter-Receiver
13. No. 1 Marker Beacon Receiver
14. No. 2 Marker Beacon Receiver
15. Isolation Amplifier-Receivers
16. No. 1 Glide Path Receiver
17. No. 2 Glide Path Receiver
18. No. 1 Localizer Omnirange Receiver
19. No. 2 Localizer Omnirange Receiver
20. Red ADF Amplifier Unit
21 Green ADF Amplifier Unit

resistor panels, etc. A removable cover protects the lower part of the box. The inboard upper cover, which is the radio power panel, is hinged to provide access to the top three terminal strips. The outboard upper cover is removable and acts as a retainer for the plug-in relays.

(2) The radio power panel carries the circuit breakers for all radio DC power. The lower right hand breaker operates the floodlight illuminating the panel. A radio supply DC circuit breaker on the electrical power supply panel (Fig. 36) protects the DC feed to the radio junction box. There is also a radio DC master breaker on the radio power panel. The panel carries active and spare fuses for all radio AC power circuits.

NOTE

(1) The DC feed for the HF communication set is sectionalized through separate circuit breakers for receiver, transmitter and antenna load unit. This arrangement allows for limited operation of the HF transmitter-receiver should a partial failure occur.

1-110 TYPES OF EQUIPMENT: (GENERAL)

(1) RADIO COMMUNICATION EQUIPMENT: Principal radio communication units are as follows: Collins type 18S-1 medium-high-frequency transmitter-receiver; type AN-RT18-ARC-1 very-high-frequency transmitter-receiver, microphones, antennas, control panels and accessories. A lever switch on each of the two audio selector panels on the pedestal permits the pilot and co-pilot to select individually either the HF or VHF transmitter. One crew member may operate the HF transmitter while the other operates the VHF transmitter.

(2) RADIO NAVIGATION EQUIPMENT: Principal radio navigation units are as follows: Dual installation Lear type ADF-12 automatic direction finders (radio compass); dual installation Bendix MN-61A marker beacon receivers, dual installation Collins 51R-1 omnirange and localizer receiver, dual installation type AN-R89/ARN-5A glide path receiver, antennas, control panels, and associated equipment.

(3) INTERPHONE EQUIPMENT: The interphone equipment consists of a Bendix type MI-32A multi-channel audio amplifier. This is used as the interphone amplifier, pilot's and co-pilot's isolation amplifiers, and pilot's and co-pilot's loudspeaker amplifiers.

1-111 MEDIUM HIGH FREQUENCY TRANSMITTER-RECEIVER:

(1) The type 18S-1 transmitter-receiver unit operates in conjunction with a type 180K-3 antenna loading unit. Both units are located on the main radio rack. The transmitter-receiver normally provides radio communication between aircraft, or between aircraft and ground stations within the frequency range of 2.7 to 12.0 mc. The transmitter has an output of 100 to 125 watts on any

frequency within its range. The equipment in this installation has been modified to raise the upper limit of its frequency range to about 13.0 mc.

(2) TRANSMITTER-RECEIVER UNIT: The transmitter-receiver unit consists of a crystal-controlled transmitter and receiver, and a dynamotor, housed together in one cabinet. The transmitter-receiver unit is installed on the radio rack and is remotely controlled from the frequency selector panel on the pilot's pedestal. Twenty channels can be accommodated, consisting of ten pairs of frequencies. The two frequencies in any pair should be within 1% of each other.

(3) ANTENNA LOADING UNIT: The type 180K-3 antenna loading unit is located on the top shelf of the radio rack, and is connected to the transmitter-receiver unit by a 52-ohm coaxial cable. Two meters indicate transmitter final amplifier plate current and standing wave ratio on the coaxial cable (the plate current meter is not connected in this installation).

(4) ANTENNA: The HF communication antenna runs from a short stub mast above the radio rack to the top of the vertical stabilizer to which it is grounded.

1-112 VERY HIGH FREQUENCY TRANSMITTER-RECEIVER:

(1) The type AN-RT18/ARC-1 transmitter-receiver is the set designated on the control panels as the VHF communication set. It is a crystal-controlled installation providing radio communication between aircraft and between aircraft and ground stations in the frequency range of 100 to 156 mc. The transmitter has an output of approximately 8 watts. The set has been modified to operate on ten pairs of frequencies according to a CAA-approved method instead of the original ten frequencies. The two channels in any pair must be within 400 kc. or less of each other. A guard receiver strip is also provided in the set to enable simultaneous monitoring of a guard channel. The guard receiver has been disconnected internally in this installation.

(2) ANTENNA: A whip-type antenna projects downward from the underside of the tail cone.

1-113 RADIO COMPASS RECEIVERS (ADF):

(1) RECEIVERS: The type ADF-12 compass receiver is an airborne navigation aid designed primarily for use as an aircraft automatic direction finder which provides aural and visual signals to the pilot. Two receivers are designated RED and GREEN ADF respectively. Each receiver is independent of the other in performance of its function. Each compass receiver is a multichannel superheterodyne receiver that operates at a signal frequency of 200-1750 kc. covered in three bands. A Bendix Radio MR-16D Range-Voice filter is provided in the radio junction box for each radio compass receiver. Functional selection, band selection and tuning are accomplished from the ADF tuners above the instrument panel. "Voice-Range" selection is accomplished from the audio selector panels provided for the pilot and the co-pilot. The receivers will also operate as

straight receivers. The IF-AF amplifier of the radio compasses are located in the radio rack. For loop and sense antenna location see sub-para. 3.

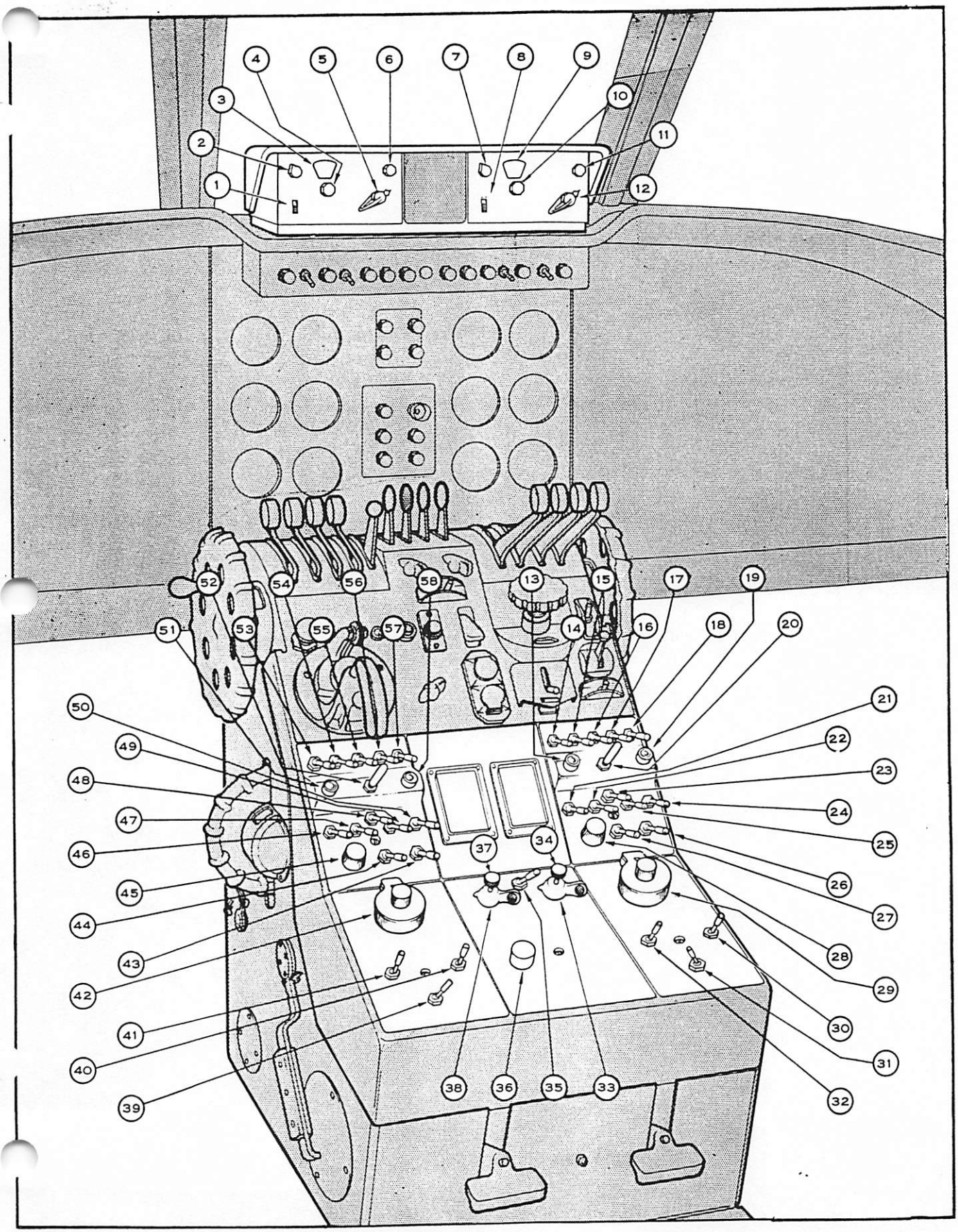
(2) RADIO MAGNETIC INDICATORS: The Bendix Radio MN-72A or Collins 332C-1 Radio Magnetic Indicator is a dual radio navigation azimuth indicator which combines in one instrument directional information from two radio direction finding installations (either two low frequency ADF receivers or two VHF omnirange receivers, or one of each) and the magnetic heading from a remote, stabilized magnetic compass. In this installation the usual arrangement is to operate the scale of the pilot's RMI from the flux gate compass (part of the PB-10 automatic pilot). The narrow pointer of the pilot's instrument gives the directional indication of the No. 1 omnirange equipment while the broad pointer gives the indication of the red ADF. The co-pilot's instrument has its scale slaved to the gyrosyn compass. The narrow pointer of the co-pilot's RMI is connected to the No. 2 omnirange equipment and the broad pointer to the green ADF. The wiring in the junction box can be altered very easily to give dual ADF or dual omnirange on either or both RMI if desired.

(3) ANTENNAS: The sense antennas for the radio compasses are supported at their front ends by masts located on either side of the nosewheel well. The rear ends are attached to the air scoop. The loop for the red ADF is in a plastic fairing at the rear of the air scoop, while the green ADF loop is inside a flush plastic window in the underside of the fuselage just aft of the air scoop.

1-114 MARKER BEACON RECEIVER:

(1) Two receivers are provided. The type MN-61A marker receiver is a continuous operation airborne radio navigation aid that provides aural and visual signals to the pilot by distinctive audio-frequency tones in the headset and by specifically colored indicator lights. These signals indicate passage of the aircraft over outer, inner, or airways markers. The marker receiver is a single channel, fixed-tuned crystal-controlled superheterodyne receiver that operates at a preset signal frequency of 75 mc. This signal, modulated 30% at 400 cycles (corresponding to the instrument landing system outer marker) will produce a 400 cycle tone in the headset and will illuminate the purple indicator light; modulated 30% at 1300 cycles (corresponding to the instrument landing system inner marker) will produce a 1300-cycle tone in the headset, and will illuminate the amber indicator light; modulated 30% at 3000 cycles (corresponding to the airways markers) will produce a 3000-cycle tone in the headset, and will illuminate the white indicator light. A "narrow-broad" switch controls the sensitivity of the receiver. Two sets of indicator lights are used. The pilot's indicator lights are operated by the No. 1 marker receiver and the co-pilot's lights by the No. 2 marker receiver.

(2) ANTENNAS: A separate antenna is provided for each marker receiver. Flush-type antennas (Ken Standard Corp. N600) are used. These are mounted on the underside of the fuselage.


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KEY TO FIG. 50 PILOT'S AND CO-PILOT'S RADIO CONTROLS

ADF (Automatic Radio Compass) Tuning Units:

1. Red ADF Function Switch
2. Red ADF Volume Control
3. Red ADF Dial
4. Red ADF Tuning Control
5. Red ADF Band Switch
6. Red ADF Dim Control
7. Green ADF Volume Control
8. Green ADF Function Switch
9. Green ADF Dial
10. Green ADF Tuning Control
11. Green ADF Dim Control
12. Green ADF Band Switch

Pedestal Controls:

13. Co-Pilot's VHF Transmitter Ready Light
14. Co-Pilot's VHF Communication Audio Switch
15. Co-Pilot's No. 1 Localizer Omnirange Audio Switch
16. Co-Pilot's Marker Beacon Audio Switch
17. Co-Pilot's No. 2 Localizer Omnirange Audio Switch
18. Co-Pilot's HF Communication Audio Switch
19. Co-Pilot's HF Transmitter Ready Light
20. Co-Pilot's HF Transmitter-Interphone-VHF Transmitter Microphone Switch
21. Co-Pilot's Red ADF Voice Audio Switch
22. Co-Pilot's Red ADF Range Audio Switch
23. Co-Pilot's Interphone Audio Switch
24. Co-Pilot's Green ADF Voice Audio Switch
25. Co-Pilot's Green ADF Range Audio Switch
26. Co-Pilot's Normal Emergency Switch
27. Co-Pilot's Red ADF - Green ADF Simultaneous Range and Voice Audio Switch
28. Co-Pilot's Audio Volume Control
29. VHF Localizer Omnirange Frequency Selector - Set 2.
30. Localizer Tone-Phase Switch - Set 2.
31. Marker Sensitivity Switch - Set 2.
32. Not Used
33. HF Communication Channel Selector
34. HF Communication Volume and Sensitivity Control
35. Not Used
36. VHF Communication Volume Control
37. VHF Communication Sensitivity (Squelch) Control
38. VHF Communication Channel Selector
39. Pilot's ILS Indicator and Auto-Pilot Flight Path Control Transfer Switch

40. Marker Sensitivity Switch - Set 1.
41. Localizer Tone-Phase Switch - Set 1.
42. VHF Localizer Omnirange Channel Selector - Set 1.
43. Pilot's Normal Emergency Switch
44. Pilot's Red ADF-Green ADF Simultaneous Range and Voice Audio Switch
45. Pilot's Audio Volume Control
46. Pilot's Red ADF Voice Audio Switch
47. Pilot's Red ADF Range Audio Switch
48. Pilot's Interphone Audio Switch
49. Pilot's Green ADF Voice Audio Switch
50. Pilot's Green ADF Range Audio Switch
51. Pilot's VHF Transmitter Ready Light
52. Pilot's HF Transmitter-Interphone-VHF Transmitter Microphone Switch
53. Pilot's VHF Communication Audio Switch
54. Pilot's No. 1 Localizer Omnirange Audio Switch
55. Pilot's Marker Beacon Audio Switch
56. Pilot's No. 2 Localizer Omnirange Audio Switch
57. Pilot's HF Communication Audio Switch
58. Pilot's Transmitter Ready Light

1-115 RADIO LOCALIZER, OMNIRANGE AND GLIDE PATH RECEIVERS:

(1) GENERAL: Two complete localizer-omnirange and glide path installations are provided. The glide path antenna is a Bendix Radio MN-92A flush-type unit installed in the centre of the nose cap of the airplane and the localizer-omnirange antenna is a specially designed unit which fits under a laminated fiberglas cap on the top of the fin. Both glide path receivers are operated from the one glide path antenna and both localizer-omnirange receivers are operated from the one localizer-omnirange antenna. No. 1 localizer and glide path receivers are controlled from the panel at the lower left part of the pedestal and the No. 2 receivers from the lower right pedestal panel. A switch on the left panel permits connection of the pilot's cross-pointer indicator and the flight path control equipment of the automatic pilot to either the No. 1 or No. 2 receivers. Relays in the junction box accomplish the switching, and at the same time switch in appropriate load resistors on the outputs of the No. 1 and No. 2 receivers to keep their sensitivity constant.

(2) VHF NAVIGATION RECEIVER (LOCALIZER AND OMNIRANGE): The Collins type 51R-1 navigation receiver and accessories are designed to fulfill the navigation and communication receiving requirements of aircraft using the 108 to 136 mc band. Full instrumentation circuits are provided in the receiver and accessories for reception and presentation of all radio services now available in this portion of the VHF spectrum. These facilities include:

(a) Localizer: tone type (including flag alarm).
(b) Localizer: phase type (including flag alarm).
(c) Omnidirectional ranges: automatic indication through selsyn differential, giving combined ADF and magnetic compass presentation on radio magnetic indicator. (See section 3-25, paragraph (2).)
(d) Omnidirectional ranges: indication on cross-pointer meter of the course chosen by manual bearing selector. Operation of a TO-FROM indicator and flag alarm is also provided.
(e) Voice Communication: on localizer and omniranges as well as on all the regular communications channels.

(3) In addition to the above facilities, the channel selection circuits of the equipment are arranged to accommodate the addition of a glide path receiver to the system. Thus choice of appropriate localizer and glide path channel is afforded from a single control unit.

(4) The 51R-1 receiver covers the frequency range 108-136 mc on a 100 kc channel basis. Any one of these 280 channels may be selected from the cockpit over a positive nine-wire control system. The channel selection is made by two coaxial switches which combine to read directly in frequency. The receiver is a double conversion superheterodyne having a tunable first IF amplifier while the second IF amplifier operates at a fixed frequency. Injection frequencies for the two converters are obtained from two groups of high stability crystals. Only 34 crystals are employed to provide complete 280-channel coverage. The receiver may also be set up to cover the above frequency range on a 200 kc basis simply by omitting 10 crystals.

(5) A Collins 351A-1 accessory chassis is provided in the radio rack. This chassis accommodates the omnibearing indicators, RMI servo amplifiers and dynamotors for the two 51R-1 receivers.

(6) GLIDE PATH RECEIVER: The glide path receiver is a three-channel, crystal controlled, superheterodyne receiver. The glide path signal, modulated at 90 cycles, will deflect the horizontal pointer in the indicator DOWN; the signal, modulated at 150 cycles, will deflect the horizontal pointer in the indicator UP. All external electrical connections are made through the connectors located on the front of the receiver. The operating frequency selection is accomplished from the localizer frequency selector panel. The receivers have been modified in accordance with standard airline practice to provide a signal for the horizontal alarm flags of the cross-pointer indicators (sub-para. 7).

(7) INDICATORS: Weston model 888 cross-pointer indicators with flag alarms are provided on the pilot's and co-pilot's instrument panels. This instrument contains four indicator mechanisms consisting of vertical and horizontal pointers operated from the localizer-omnirange and glide path receivers respectively and two red alarm flag pointers operating as warning signals to show when the receiver circuits are functioning properly. When the circuits are inoperative, the flags lie on the dotted centre lines of the instrument scale and show an OFF signal. As the signal current builds up the flags move away from the dotted lines and at full operating current, disappear behind the black shield. The marker lamp normally included in this instrument has been removed since triple marker lights are provided.

1-116 INTERPHONE AND AUDIO SYSTEM:

(1) GENERAL: Headphone and microphone jack panels are provided on the side panels of the cockpit for the pilot and co-pilot. In addition, there is a loudspeaker on each side of the cockpit located in the roof above the windows. Each jack panel has a speaker-phones switch, enabling each crew member to feed the audio signals selected on his audio selector panel on the pedestal, through either his headphones or loudspeaker. A jack box for the flight engineer or observer is provided on the right hand rear wall of the flight compartment. A switch on this jack box enables selection of the audio output of either the pilot's or co-pilot's audio panel. The microphone jack on this jack box connects only to the interphone system. Transmit relays are provided in the junction box for the pilot and co-pilot and are controlled by their respective microphone buttons. Each relay simultaneously operates the transmitter selected by the pilot or co-pilot and also mutes the adjacent loudspeaker when his microphone is in use. For safety, each transmit relay is fed from a separate circuit breaker.

(2) MULTI-CHANNEL AUDIO AMPLIFIER: The type MI-32A multi-channel audio-amplifier permits reception from any combination of radio receivers wired into the system. The unit contains provisions for six amplifier sections. Three of these are type MI-33A amplifiers functioning as pilot's isolation amplifier, interphone amplifier and co-pilot's isolation amplifier respectively. Two

positions carry AI-28B amplifier sections used for pilot's and co-pilot's loudspeaker amplifiers. The last position is plugged with a type MI-35A dummy amplifier to complete the circuit. The system permits reception from any combination of radio receivers at each operating station without interference from other operating stations. Selection of the desired reception is accomplished at the audio selector panels provided for the pilot and co-pilot.

(3) "NORMAL-EMERGENCY" SWITCH: One "Normal-Emergency" switch is located on each audio selector panel. With the switch in the "Emergency" position, the isolation amplifier is bypassed and reception from one channel only is directed from the receiver to the operator. The emergency position is used only in case of failure of the isolation amplifier.

(4) INTERPHONE FACILITIES: The attendant's handset, and also the interphone outlets at the observer's position, radio rack, nosewheel well, and external power supply connection point, all tie into the interphone system. Call facilities are provided from the flight deck to the attendant, and to a call horn in the nosewheel well, and from the attendant's and external interphone points to the flight deck.

CHAPTER 1

OPERATING LIMITATIONS

2-1 FLIGHT RESTRICTIONS:

(1) The aircraft may be operated by a minimum crew of two, consisting of pilot and co-pilot.

(2) The type is restricted to normal flight manoeuvres as applied to passenger and cargo aircraft.

2-2 FLIGHT RESTRICTIONS:

(1) The TAKE-OFF gross weight of the aircraft must not exceed the sum of the permissable gross weight at the intended destination, plus the weight of fuel and oil normally consumed enroute. Permissable operating weights of this aircraft is as follows:-

(a) Maximum permissable take-off weight - 65,000 lbs.

(b) Maximum landing weight - 55,000 lbs.

(c) Maximum zero fuel weight conditions - lbs.

(2) Operational weight empty is defined as the basic weight of the aircraft prior to loading for dispatch. This weight varies with flight requirements as shown in paragraph 2-3. The operational weight empty includes the weight of the aircraft with its equipment, flight crew, oxygen, unusable oil and fuel, hydraulic fluid, crew baggage and certain items of ancillary equipment.

(3) The maximum permissable aircraft payload is the difference between the 'operational weight empty' and the 'maximum zero fuel weight' of the aircraft.

NOTE

(1) For further particulars on fuel loading and consumption, refer to paragraph 3-11.

2-3 C. of G. LIMITATIONS:

(1) This aircraft must be operated and loaded in accordance with the following table and approved charts contained in section 4 of this manual.

C. OF G. MOVEMENT			
Condition	Weight	Landing Gear Position	C of G Travel Limits
Take-Off	Max. Gross	Down	15% - 25% SMC
Climb	" "	Up	15% - 25% SMC
Enroute Cruise	" "	Up	15% - 25% SMC
Enroute Cruise Below	" "	Up	15% - 25% SMC
Approach and Landing	" "	Down	15% - 25% SMC

NOTE

(1) Retraction of the landing gear moves the C of G forward approximately .4% of the S.M.C. at 65,000 lbs. (Maximum permissible weight).

(2) The aircraft is fully controllable and stable throughout the entire C of G range specified. No appreciable variation in speed or range will result for any loading condition within these limites. (For further information, refer to paragraph 2-22).

(3) Location of the DATUM for zero inch reference is 6" forward of the nose.

2-4 FLIGHT LIMITATIONS:

(1) Do not exceed the maximum allowable indicated airspeeds as outlined in Section 4, Charts 101 and 105.

2-5 POWER PLANT LIMITATIONS:

(1) The life of a gas turbine engine is adversely affected by high temperatures and by rapid changes of temperature. It is therefore desirable to eliminate unnecessary ground runs and accelerations.

(2) Jet pipe temperature limits are critical and must not be exceeded.

OPERATING LIMITATIONS

Conditions	Max. r.p.m.	Max. J.P.T.	Time Limit
Ground Idling	3,500 ± 200	500° C	-
Take-Off	14,700 ± 50	680° C	5 Mins.
M.E.T.O.	14,100	630° C	-
Recommended Max. Cruise	13,600	570° C	-

(3) TAKE-OFF: Engine speed is governed on the ground at 14,700 r.p.m. or 660°C J.P.T. At this speed 3,600 lbs. thrust is obtained at take-off. In flight, 14,700 r.p.m. is available for emergency conditions.

(4) IDLING: With the Duple burners fitted, satisfactory idling will be obtained at altitude with the throttles fully closed. Idling speeds will increase with altitude.

OIL PRESSURES	
Idling	5 p.s.i. (min.)
13,600 r.p.m. and above	30 p.s.i. (min.)
CONSUMPTION	
Maximum of 1 pint / hour	

(5) For power rating of engine, refer to Engine Performance Chart (Fig. 56).

2-6 TEMPERATURE LIMITS:

(1) Should the J.P.T. exceed that given in paragraph 2-4, the engine should be stopped and allowed to cool. On no account open up to full throttle.

(2) Maximum permissible J.P.T. - 680° C

2-7 FUEL SPECIFICATIONS:

(1) Grade of Fuel - U.S. Spec. MIL. F-5616
British Spec. D.Eng. RD 2482

(2) Emergency Alternative: mixture of aviation fuel and oil - See paragraph 5-5.

(3) Fuel Pressure (Low Pressure System) - 3-5 p.s.i. (minimum)

(4) Water Methanol injection fluid - 61 parts by volume inhibited distilled water. (60 parts distilled water, 1 part Dromus oil). 40 parts by volume denatured Methanol to British Spec. D.E.R.D. 2470.

(5) Water Methanol tank injection pressure - 12-15 p.s.i. (provisional)

2-8 OIL SPECIFICATIONS:

(1) Engine Oil - British Spec. DTD. 44D
U.S. Spec. AAF. 3580D

(2) Auxiliary Gear Box Oil - British Spec. DED. 2479
U.S. Spec. N/A

(3) Bevel Gear Box Oil - British Spec. DTD. 44D
U.S. Spec. AAF. 3580D

2-9 SUMMARY OF SPEED AND LOAD FACTORS:

CONDITIONS	Flaps & Landing Gear UP	Flaps Down With Flaps Set to			Ailerons Deflected	Landing Gear Extended in Flight	
		30°	45°	60°		Take-off	Landing
Limit Positive Manoeuvre Factor	2.5	2.0	2.0	2.0	1.67	2.5	2.5
E.A.S. in M.P.H.	140-375	250	193	176	140-375	200	200
Weight lbs.	35000 to 65000					65000	55000
Limit Negative Man. Load Factor	-1	-	-	-	-		
E.A.S. in M.P.H.	110-300	-	-	-	-		
Weight lbs.	35000 to 65000	-	-	-	-		
Limit Positive Gust Load Factor	3.35	3.09	2.86	2.81	2.80	1.46	1.52
E.A.S. in M.P.H.	300	250	210	200	110-375	200	200
Weight lbs.	✱ 48000				65000	65000	55000
Limit Negative Gust Load Factor	-1.35	-1.09	-.86	-.81	-.80	-.46	-.52
E.A.S. in M.P.H.	300	250	210	200	110-375	200	200
Weight lbs.	✱ 48000				65000	65000	55000

✱ Figures assume 13,000 lbs. in payload and fuel.

OPERATIONAL WEIGHT EMPTY: lbs.- includes crew of three - zero fuel.

2-10 FLIGHT CREW:

(1) A minimum crew of two, consisting of pilot and co-pilot are required for all operations.

2-11 AIRCRAFT TYPE:

(1) This aircraft is in the transport category. Operation in light icing conditions is permitted, as is instrument night flying, when the required equipment is installed.

2-12 WING FLAPS:

(1) The basic performance information described in Section 4 of this manual is based on the following flap positions:

Take-off	25°
Cruise	0°
Approach	12½°
Landing	60°

(2) Landing distances described in Section 4 are based on flap retraction being executed during the landing roll, after speed is reduced.

2-13 INSTRUMENT COLOUR CODING:

Red radial line	Limit of Operation
Yellow arc	Cautionary Range
Green arc	Normal Operating Range
White arc	Flap Operating Range
White radial line	Index (alignment check point)

2-14 TAXYING LIMITATIONS:

(1) Eliminate sharp turns during high speed taxying, avoid ruts, soft ground etc., or any condition causing rapid deceleration or requiring high power.

2-15 CRITICAL CROSS WIND OPERATION:

(1) Under these conditions this aircraft has been found to compare favourably with other aircraft of its type.

2-16 FUEL SYSTEM:

(1) Fuel is normally fed from the four integral wing tanks (two in each wing) to their respective inner and outer engines through independent piping systems. Provision is made to feed any engine from any tank. A cross-feed system is incorporated, but should not be used for take-off purposes. When operating on cross-feed, the tank(s) not being used, should be turned OFF. The cross-feed should not be used for landing except in emergencies. The booster pumps are normally switched to the ON position, but should be switched OFF as the tanks are emptied.

2-17 WATER METHANOL SYSTEM:

(1) The water methanol system is provided to give increased maximum thrust under take-off conditions at temperatures above 20°C. Selection is made from the centrally disposed engine instrument panel, the system then being automatically operated by the throttles when the latter are at the take-off position only. Sufficient fluid is carried in the system to allow for 1 minute operation.

2-18 HYDRAULIC SYSTEM:

(1) The hydraulic system pressure is provided by two engine driven pumps and a remotely controlled emergency electrical power pack. The engine driven pumps supply fluid to the services through the NORMAL supply lines. The emergency power pack supplies fluid through separate supply lines to specified services (Para. 1-56). The EMERGENCY supply must be switched ON for take-off and landing. Operation of the EMERGENCY supply during flight is optional.

(2) Maximum NORMAL system pressure is 2,700 p.s.i.

(3) Maximum EMERGENCY power system pressure is 2,250 p.s.i.

(4) The relief valves are adjusted to by-pass pressures in excess of 2,700 p.s.i. back to the reservoir.

2-19 AIR CONDITIONING SYSTEM:

(1) Air conditioning and cabin heating normally operate as one system. Temperature control under normal conditions is automatic, but manual controls located on the attendant's panel allow for independent control of the heating system.

2-20 PRESSURIZATION:

(1) In the event of either inboard engine failure, or the failure of an auxiliary engine gear box, one blower is normally capable of maintaining an air flow of 60 lbs. per minute to an altitude of 15,000 ft.

(2) Maximum cabin differential pressure is 4.2 p.s.i.

2-21 SPECIAL NOTE:

(1) The limitations described in paragraphs 2-1 to 2-20 may be supplemented or superseded by instructions included in relative data or publications.

2-22 LOAD ADJUSTER SLIDE RULE: (General)

(1) This rule is devised to facilitate the calculating of the take-off C of G for the first prototype aircraft only.

(2) It comprises two fixed parallel scales, with a sliding scale interposed and a sliding hairline or cursor covering all three scales.

(3) The rule is divided as shown in Figure 51, the lower scale being inscribed with index numbers or MOMENTS. The larger numbers designate millions of pounds inches, and the smaller figures are graduations of the same. The centre sliding scale carries the configuration relative to fuel, crew, passenger and station loads. Each individual configuration has a ZERO reference line.

(4) With reference to station loads, the scales have been designed to meet with all possible load requirements between formers 6 and 45. (The pressurized area of the fuselage). The scales read from 200 to 2,000 lbs. on every second former. When moments of intermediate formers are required, the mean reading of the formers immediately forward and aft of the one desired should be taken.

(5) The upper scale carries a table for the conversion of ARM readings to a percentage of SMC. The rear face of the rule carries a chart incorporating former and station numbers and the "C" and "D" scales of a standard slide rule. The standard "C" and "D" scales are incorporated to facilitate the division of MOMENTS BY WEIGHT to provide the C of G ARM.

(6) The basic principle of operation, is that the rule operates on a summation of a group of smaller moments to the ORIGINAL or TARE moment which is set at the beginning of operations.

2-23 HANDLING THE SLIDE RULE:

(1) To take a reading, the TARE WEIGHT must first be set on the rule.

This is done by placing the cursor hairline over the desired MOMENT on the lower scale.

(2) Assuming the TARE WEIGHT of the aircraft is equal to 35,000 lbs., the relative moment about the rear spar will be equal to 3,500,000 pound inches. When the cursor is set over this MOMENT, the first addition to the TARE WEIGHT may be calculated.

(3) To add the FUEL CONTENTS, move the sliding portion of the rule until the ZERO fuel setting is under the hairline, then move the cursor to the left until the hairline is under TANKS FULL NO. 1. Repeat this action for tanks 2, 3, and 4 as applicable. (Although tanks 2 and 3 are on separate scales, the handling procedures are identical to those used for tanks 1 and 2). Provision will be made at a later date to include a scale for the auxiliary fuselage tanks.

(4) When adding the CREW weight, assuming a minimum requirement of two members, pilot and co-pilot, set the sliding portion of the rule so that ZERO CREW falls under the hairline of the cursor, then move the cursor so that the hairline is set to the extreme left position. When including additional crew members, follow through with the setting of the cursor hairline and take MOMENT readings at relative settings.

(5) To include the weight of PASSENGERS, the cursor hairline is initially set to the ZERO position, then moved to LEFT or RIGHT along the corresponding horizontal line relative to the number of seats being used in each row. A reading can be taken for each of the nine rows.

(6) With reference to STATION LOADS, the cursor line should first be set on the relative ZERO mark on the sliding scale; then moved to either right or left along the desired horizontal (or former) line. When the hairline on the cursor registers with the known weight, take a MOMENT reading.

NOTE

(1) When loading the aircraft, an accurate account of the added weight must be kept, as without this total weight the C of G movement cannot be obtained.

(2) With reference to sub-para (3) and the corresponding reference in Figure 51, it will be seen that provision has been made for four types of loading. This arrangement covers all passenger loading arrangements by making allowance for 1, 2, 3, or 4 passengers being distributed laterally across each row of seats.

(3) MOMENTS are taken about the REAR SPAR and NOT from the nose datum.

(4) Since the REAR SPAR is used as the DATUM, it is necessary to have positive and negative moments positioned respectively aft and forward of the rear spar.

(5) Once the BASIC or TARE WEIGHT has been set, the slide or cursor should not be moved unless a calculation is being made. Failure to observe this precaution, would result in incorrect readings.

2-24 EXAMPLE ON DETERMINING THE C. OF G. POSITION:

(1) The following table provides a typical example on how to use the slide rule. All figures are arbitrary:

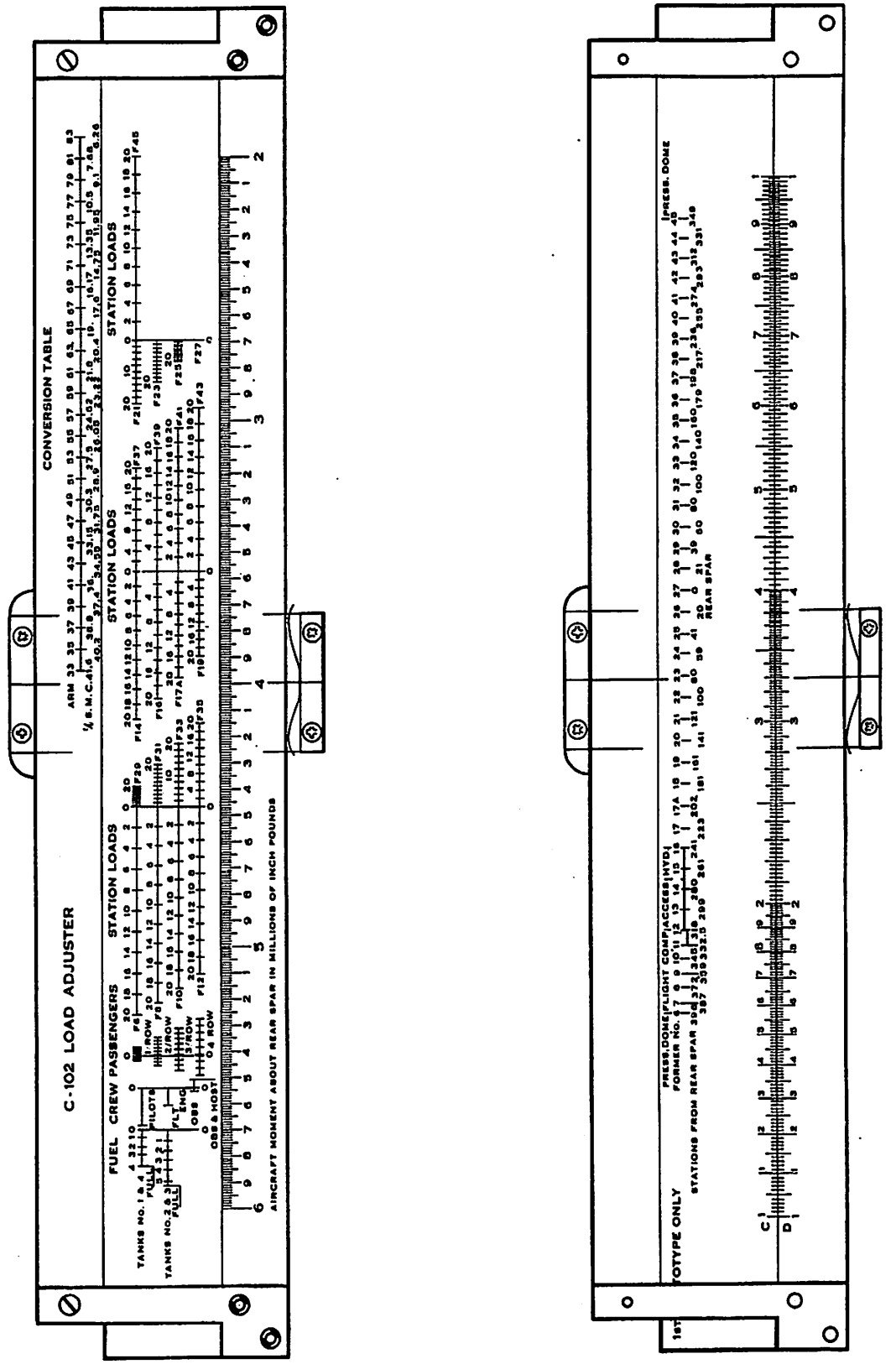
Tare Weight	38,526 lbs.
Fuel Load (Wing Tanks)	19,000 lbs.
Fuel Load (Aux. Fuselage Tanks)	8,447 lbs.
Crew Weight	400 lbs.
Passengers (12)	2,040 lbs.
Station Loads (Luggage, Ballast etc.)	- lbs.
TOTAL ALLOWABLE WEIGHT	65,000 lbs.

(2) By using the rule in the prescribed manner (para. 2-23) an arbitrary TOTAL MOMENT of 3,705,000 lb./ins. may be found. Therefore, the distance from the REAR SPAR is $\frac{3,705,000}{65,000}$ - 57.00" (C of G ARM)

(3) To find the percentage of S.M.C. of the C. of G. ARM, use the conversion table on the front face of the slide rule.

(4) By taking the arbitrary figures supplied in sub-para (2) (57.00") and by checking with the conversion table, it will be found that a reading of 24.62% SMC is obtained. This is a safe flying condition for the aircraft.

LOAD ADJUSTER
SLIDE RULE
FIG. 51



NOTES

CHAPTER 1

OPERATING PROCEDURES

3-1 BEFORE ENTERING THE AIRCRAFT:

(1) The aircraft should be checked for loading in accordance with the Load Adjuster slide rule (Para. 2-22).

(2) Check for removal of landing gear safety ground locks.

(3) Check for removal of right and left wing and nose pitot head covers.

(4) Carry out a visual check on tires.

(5) Check nacelle intakes and jet pipe installations for obstructions. See that all covers are removed from the ram air intakes and that exhaust covers are removed.

3-2 ON ENTERING THE AIRCRAFT:

(1) Check for security and distribution of load.

(2) Check for security and closing of main entrance door, escape hatch and emergency exits.

(3) If installed, check long range fuel system for shut-off.

(4) Check the fluid level in the hydraulic reservoir sight gauge. Level should not be below the RED marker, or above the GREEN marker. On completion of check, switch on EMERGENCY power pack until a hydraulic system pressure of 2,250 p.s.i. is reached.

(5) On entering the flight deck, observe and carry out the standard procedure outlined in the following sub-paragraphs.

(a) Check escape hatch and windows for security in the closed positions.
(b) Check safety belts and seats for adjustment and security.
(c) Check manual flight controls, including trim, for freedom of movement and travel.
(d) Ensure landing gear selector is in the DOWN position.
(e) Determine source of battery power supply. (i.e. internal or external supply).
(f) Place master battery switch in the ON position.

NOTE

(1) Limited starting is permissible on aircraft batteries, providing the four auxiliary batteries are installed.

(2) If ground receptacle supply is connected, a green indicating light will show on panel adjacent to MASTER SWITCH.
(3) A further alternative is for a ground battery supply to be connected in parallel with the aircraft batteries. This will provide additional power for starting. When this procedure is followed, the paralleling switch must be selected ON. This switch is located on the forward wall of the hydraulic compartment, and must be in the OFF position before the external supply is removed. (For further particulars refer to para. 1-99 of this manual).

WARNING: Before carrying out the procedure outlined in notes (1) or (2), it is essential that the MASTER SWITCH be placed in the OFF position when the ground supply is being connected or disconnected.

(5) Place MASTER SWITCH in the ON position.

(6) Place all circuit breakers in the ON position.

(7) Generator switches OFF.

(8) Alternator switches OFF.

(9) Normal inverter ON.

(10) Hydraulic emergency power switch ON.

(11) Parking brake ON.

(12) Cabin pressure spill valves OPEN.

(13) Cabin pressure safety valve OPEN.

(14) Cabin controller - select 1500 feet.

(15) Rate of cabin change adjust to 300 feet per minute.

(16) Select MASTER fire control switch ON.

3-3 ENGINE STARTING PROCEDURE:

(1) Before attempting to start the engines the aircraft should be headed into wind and the ground in the vicinity of the engine air intakes checked for loose debris. Failure to observe the latter precaution may result in severe damage to an engine if small stones etc. are drawn into the intake.

(2) For clarification purposes, the engines are numbered 1 to 4 consecutively from the left side of the aircraft. When STARTING, select No. 4 engine FIRST, then follow through with engines 1, 2, and 3.

(3) The starting procedure for individual engines is as follows:

(1)	Select both tanks	- ON (right and left switches to both tanks)
(2)	Cross-feed switch	- OFF
(3)	Low pressure cocks	- OPEN
(4)	High pressure cocks	- CLOSED
(5)	Throttles	- CLOSED
(6)	Select all booster pumps	- ON
(7)	Select starter safety switch	- ON
(8)	Turn starter selector switch	- To desired engine.
(9)	Select booster coil isolating switch	- To desired engine.
(10)	Engine start button	- PRESS (for two seconds)
(11)	Select high pressure cock (When engine speed reaches 500 r.p.m.)	- OPEN (desired engine)

NOTE

(1) The engine will accelerate to an idling speed of 3,500 r.p.m. at an idling temperature of not more than 500 degrees C. Check for minimum oil pressure of 5 p.s.i. at idling speed. The WARNING LIGHT should be out.

(12) When engine starts, select booster isolating switch - OFF

NOTE

(1) Repeat starting procedure for the remaining engines as noted in sub-paragraph (2).
(2) When ALL, or selected engines have been started, place the STARTER SAFETY SWITCH in the OFF position.
(3) When starting on GROUND SUPPLY leave the engine generator switches OFF until after the ground supply has been disconnected.
(4) If the aircraft auxiliary batteries are being used, the RIGHT generator should be switched ON immediately No. 4 engine has been started. Accelerate No. 4 engine to 4,000 r.p.m. and use generator output to assist in starting the remaining engines.
(5) The engines will run at idling speed without throttle adjustment. Do not open throttles before idling speed is reached.

3-4 TAXYING:

(1) With the EMERGENCY hydraulic system power switched ON, and the pressure up:

(1)	Check that NORMAL system HYDRAULIC pressure and BRAKE pressure register a MINIMUM of	- 2,000 p.s.i.
(2)	Nosewheel STEERING switch	- ON and clutch ENGAGED. (This will be indicated by the illumination of the ORANGE warning light).
(3)	PARKING brake	- OFF

NOTE

(1) Due to high fuel consumption at engine idling speeds, it is advisable to taxi out without delay.
(2) Throttles should be operated smoothly to avoid excessive J.P.T.'s and the possiblility of surge. (Surge is indicated by a fast rumbling sound).
(3) Do not taxi in the jet stream of another aircraft. Failure to oberve this ruling will result in high J.P.T.'s.
(4) When finally positioned on the runway, preparatory to TAKE-OFF, apply and hold the brakes in the FULL ON position.

3-5 CHECKS PRIOR TO TAKE-OFF:

(1) Position all switches.

(2) Seat belt and NO SMOKING sign - ON.

(3) Pressurization air intake switch to - FILTERED AIR.

(4) Hydraulic normal-emergency pressure check 2000 lbs. minimum.

(5) Trim - for take-off.

(6) Manual controls - check full and free travel.

(7) Aileron power booster engaged - warning light out.

(8) Flaps - take-off position (25°).

(9) Fuel - contents and routing.

(10) Cabin pressure controller - set to 2000 ft. above airport altitude.

(11) Cabin pressure safety valve - CLOSED.

(12) Cabin pressure spill valves - OPEN.

(13) Cabin heat - as desired.

(14) Lighting - as desired.

(15) Auto-pilot - OFF.

(16) Doors and hatches secured (Check warning lights out).

(17) All gyros uncaged.

(18) Windscreen anti-icing: Nesa glass switch adjusted to low setting, 90-95°F.

(19) Anti-skid braking switch - ON.

(20) Nosewheel steering - OFF.

(21) Pitot static heat switch - as required.

(22) Wing anti-icing - as required.

NOTE

(1) With brakes held ON open throttles to 13,600 r.p.m. then check:

(a) Oil pressure - 30 lbs. minimum.
(b) J.P.T. normal for r.p.m.
(c) Generators ON and charging.
(d) Alternators ON and charging.
(e) Accelerate to full r.p.m.
(f) Check J.P.T. normal for r.p.m.
(g) Select spill valves to automatic operation.

(23) Release brakes and carry out take-off.

3-6 TAKE-OFF:

(1) With brakes applied, open up the throttles to full power, or until the aircraft starts slipping forward.

(2) Release brakes, and open throttles fully.

NOTE

(1) During take-off, directional control must be maintained by use of nose wheel steering and differential braking until a speed of 60 m.p.h. has been reached. At speeds in excess of 60 m.p.h. the rudder can be used to maintain directional control of the aircraft.
(2) For information on V1 and V2 speeds refer to Fig. 63 and 64.

3-7 CLIMB:

(1) Once safely clear of the runway, select landing gear and flaps UP.

(2) The NORMAL climb is at the MAXIMUM continuous rating (14,100 r.p.m.) with a J.P.T. of not more than 630 degrees C. Care must be taken at all times not to exceed engine limitations, particularly at high altitudes.

(3) Do not allow airspeed to drop below the recommended climbing I.A.S. for the thrust being used. For further particulars refer to the charts on performance details at the rear of this section.

(4) Once the aircraft has reached relatively dust free air change the setting of the AIR INTAKE switches to RAM AIR.

(5) When cabin blower delivery stabilizes, set the cabin altitude selector to 8,000 feet.

(6) Check that cabin altitude increases at the correct rate, and stabilizes at 15,000 feet prior to the aircraft reaching the 30,000 ft. level.

(7) Check the operation of the cabin temperature control system.

3-8 GENERAL FLYING:

(1) COMBUSTION: Above 20,000 feet, any throttle movement must be handled with care as rapid opening up of the engines may cause SURGE, excessive J.P.T.'s or even FLAME EXTINCTION. Should the latter occur, close the H.P. cock IMMEDIATELY. Combustion may also be affected if negative "G" is applied for more than 15 seconds. This condition applies to all altitudes.

(2) CABIN PRESSURE: Above 30,000 feet, the minimum engine speed required to maintain cabin pressure may exceed the idling limitations of the engine. Throttling back above 30,000 feet should be carried out very cautiously, one engine at a time when possible. When this operation is being performed, a close watch should be kept on both the cabin differential pressure gauge and the MASS FLOW indicator. If the pressure differential drops, engine speed must be increased.

(3) LEVELLING OFF: For optimum cruising performance, maintain climb to at least 500 feet above the desired cruising altitude. Once this height has been reached, descend to the desired cruising altitude in a slight dive. As the aircraft is levelled off at the desired altitude, reduce engine speeds to the recommended cruising r.p.m.

(4) TURBULENCE: When encountering severe turbulence, reduce airspeed according to severity of gusts. An indicated air speed of 200 m.p.h. will safeguard the structure. For effects of weight on stalling speeds, refer to Minimum Control Speed Chart (Fig. 65).

(5) To determine limitations on speed as the wing tanks fuel capacity is reduced, refer to the Chart on Maximum Level Flight Speeds (Fig. 84).

3-9 APPROACH AND LANDING:

(1) DESCENT:

(a) Prior to landing, it must be borne in mind, that thrust response is not so rapid as that of a propeller driven aircraft.
(b) Never exceed the permissible indicated gliding speeds given on the Approach Speeds Chart (Fig. 103).
(c) As descent commences, set AIRFIELD height on the cabin altitude selector, and 300 f.p.m. on the rate of change selector.
(d) Check that the cabin descends to AIRFIELD height at the correct rate, and that it stabilizes just before landing.

(2) INITIAL APPROACH:

(a) Switch Automatic Pilot OFF, and centralize controller.
(b) Seat belts and NO SMOKING switches ON.
(c) Fuel system: Ensure that a sufficient supply of fuel is available to each engine. Avoid use of CROSSFEED during normal landing procedures.
(d) Parking brake OFF.
(e) Switch ON hydraulic power pack.
(f) Hydraulic System and Brake Pressures - 2,000 to 2,500 p.s.i. (Normal and Emergency Systems).
(g) Cabin pressure intakes on FILTER.
(h) Cabin pressure spill valves OPEN.
(i) Initial flap setting $12\frac{1}{2}$ degrees at not more than 250 I.A.S.
(j) Landing gear DOWN. I.A.S. not to exceed 200 m.p.h. Check warning lights; leave selector switch in the DOWN position.
(k) Recommended MINIMUM air speed for initial approach 138 m.p.h. Flaps $12\frac{1}{2}^{o}$.

(3) FINAL APPROACH: (Gear Down)

(a) Recommended final approach airspeed 130 m.p.h. (1.3 V_{s_1} , 45° Flaps).
(b) Engine speed 7,500 r.p.m.
(c) Wing flaps 60 degrees. Approach at an I.A.S. of 126 m.p.h. (1.3 V_{s_1})

(4) LANDING:

(a) The landing speed and distance required may be obtained from the Approach Speeds and Landing Distance Charts (Fig. 103 and 104. If necessary, the shorter landing distances, as given on the chart, may be obtained by making contact with the nosewheel, immediately after the main wheels have touched the ground; and by applying maximum braking. In the case of the latter, the automatic Hytrol braking control will eliminate the possibility of skidding.
(b) Make a normal main wheel landing. Do not make contact with the nosewheel first. The nose will be slightly above the horizontal. Do not stall and drop the aircraft on the ground. A tail buffer is incorporated to guard against damage to the rear fuselage.
(c) After ground contact, hold the nosewheel OFF as long as possible by means of the elevators and then ease the nosewheel to the ground.
(d) On completion of the landing, and while taxying clear of the runway, proceed with the following operations:

(1) Adjust throttles to give the desired forward speed.
(2) Use the nosewheel steering for directional guidance.
(3) Flaps UP.
(4) Pitot heat OFF.
(5) Cabin pressure safety valve OPEN.
(6) Check hydraulic pressure - 2,000 to 2,500 p.s.i.
(7) Seat belt switch OFF.

(5) NIGHT LANDING: During night landings, or when visibility is poor, land at a speed faster than that normally used. It is safer to land at higher speeds than to drop the aircraft in. The practice of stalling the aircraft prior to touch down causes the nosewheel to strike the ground first and places abnormal loads on the nose section.

(6) CROSSWIND LANDING: The approach is made longer than normal to allow the pilot sufficient time to establish a heading to give a ground track parallel to the runway. Alter the heading of the aircraft just prior to ground contact so that it is parallel to the runway and, if necessary, apply slight bank to prevent drift. Establish ground contact with all three wheels as soon as possible. Do not raise the nosewheel from the ground once contact has been made.

(7) PARKING:

(a) When taxying is completed apply parking brakes prior to shutting off the engines.

CAUTION

Never leave the brakes in the parked position if they are over-heated, (too hot to touch). Pressure on the brakes, together with excessive temperatures resulting from hard braking, may damage the brakes or cause them to seize. When this condition occurs, allow the brakes to cool to a reasonable temperature before applying and leaving the parking brakes in the ON position.

(b) Stop the engines. See paragraph 3-10 for stopping procedure.
(c) Signal "SWITCH OFF and cabin DEPRESSURIZED" to service personnel. This signal confirms that the engines are stopped, the cabin depressurized and it is safe to open the cabin door.
(d) All lights and electrical switches OFF.
(e) Gyro Horizon CAGED.
(f) Fuel Tank selectors OFF.

3-10 STOPPING THE ENGINES:

(1) H.P. cocks OFF.

(2) Booster pump switches OFF.

NOTE

(1) The L.P. cocks should be left in the open position.

CHAPTER 2

OPERATIONAL MANAGEMENT AND CHECKS

3-11 FUEL SYSTEM MANAGEMENT:

(1) GENERAL:

(a) Fuel is carried in four tanks forming part of the wing structure. Each inboard and outboard tank has a fuel capacity of 658 and 495 Imperial gallons respectively. There are virtually two separate fuel systems, one in each wing, but inter-connected by a cross-feed pipe incorporating shut-off valves.

(b) Each tank normally supplies one engine. A booster pump in the base of each tank delivers fuel under pressure through a group of selector valves and cocks to its respective engine connection. When the valves are in their normal positions, each inboard engine is fed from an inboard tank, and each outboard engine from an outboard tank. (Fig. 19).

(c) By means of the selector valves and cross-feed valves, it is possible to direct fuel from any tank to any number of engines. Each tank booster pump can meet the maximum consumption of all four engines with an adequate supply of fuel at 3 p.s.i. Even complete booster pump failure will not render the system inoperative. Due to the low pressure loss in the fuel lines, the high pressure pump on each engine can draw a sufficient supply of fuel from the tank(s) at pressures as low as -2 p.s.i. For details of possible tank selections, see sub-para. 3, (Fig. 52).

(d) On each side of the aircraft, all fuel lines meet at a selector unit which includes six remotely-controlled slide valves and a drain valve. One slide valve is mounted on the selector unit of the inlet from each tank, these two main passages being interconnected by a third passage in which is interposed another slide valve. These three valves are motor-operated and are inter-connected so that they are controlled by one four-position switch in the flight deck. The two cross-feed cocks are motor-operated and are both controlled by one switch. The low pressure fuel cocks - one for each engine - are manually controlled so that each individual engine can be isolated even if a complete electrical power failure occurs.

(e) Four fuel contents transmitters are installed in each tank. These transmitters are co-ordinated to give an average reading which is registered in pounds of fuel on dual indicators for each tank. The indicators are mounted on the engine instrument panel.

(f) The red warning lights on the booster pump control panel are installed for the purpose of indicating that fuel pressure in the low pressure lines has fallen below 3 p.s.i. Illumination of these lights suggest either booster pump failure, line breakage, vent trouble, or other remote defects in the system.

(g) As each inboard tank contains approximately 163 gallons (of usable fuel) more than each outboard tank, a right and left inboard tank only selection should be made for a period of time during the cruising flight. This selection will compensate for the difference in tank capacities. This condition applies only when a full normal fuel load is carried. When long-range fuselage tanks are used, the procedure given in sub-para. 4 should be followed.

NOTE

(1) Fuel used is Kerosene to British Specification D. Eng. R.D. 2482. Specific weight of the mixture is 0.8 at a temperature of 20°C. Equivalent U.S. Specification is MIL. F-5616.

(2) FUEL DISTRIBUTION: The total volume of fuel should be equally disposed between the corresponding tanks in the right and left wings. The quantities of fuel in tanks 1 and 4, and tanks 2 and 3 should be balanced.

NOTE

(1) When fuselage overload tanks are fitted, it is surmized that the loading will be affected. (Information concerning this factor will be issued as it is made available).

(2) As wing fuel is consumed, the difference in fuel loading alters the C. of G. position. For further information refer to (Fig. 88).

WARNING

Fuel contents of the outboard wing tanks must not be allowed to fall below the requirements specified in the following chart.

(3) FUEL SYSTEM CONTROLS: The system is controlled by two four-position rotary selector switches, a cross-feed switch, four L.P. cocks, four H.P. cocks, and four toggle type individual booster pump switches. All controls, with the exception of the four H.P. cocks and the pump switches, are positioned on the throttle pedestal. The booster pump switches are mounted on an auxiliary panel immediately above the centrally disposed engine instrument panel (Fig. 5). The four H.P. cocks are located one on each engine and are electrically controlled from switches in the roof panel. The following sub-paragraphs describe NORMAL and ALTERNATIVE methods of tank selections.

WING FUEL TANK CAPACITIES

(IMP. GALS. AND LBS.)

TANK	FUEL									AIR SPACE	
	GROSS VOL.	USABLE UNDER FULL PUMP PRESSURE		USABLE AT DECLINING FLOW		UNUSABLE		TOTAL			
	GAL	GAL	LB	GAL	LB	GAL	LB	GAL	LB	GAL	% OF TANK VOL
LEFT OUTER	526	495.75	3966.00	2	16	8.25	66.00	506	4048.00	20	3.8%
LEFT INNER	702	658.75	5270.00	2	16	10.75	86.00	671.5	5372.00	30.5	4.35%
RIGHT INNER	702	658.25	5266.00	3.5	28.00	7.75	62.00	669.5	5356.00	32.5	4.63%
RIGHT OUTER	526	497.6	3980.80	4.7	37.60	4.00	32.00	506.3	4050.40	19.7	3.75%
TOTAL	2456.0	2310.35	18482.80	12.2	97.60	30.75	246.00	2353.3	18826.40		

AUXILIARY FUSELAGE TANK CAPACITIES

(ESTIMATED IMP. GALS. AND LBS.)

TANK	FUEL							AIR SPACE	
	GROSS VOL.	USABLE		UNUSABLE		NET TOTAL FUEL			
	GAL	GAL	LB	GAL	LB	GAL	LB	GAL	% OF TANK VOL
EACH TANK	265	260	2080	1	8	261	2088	4	2%
TOTAL 4 TANKS	1060	1040	8320	4	32	1044	8352	16	2%

(a) NORMAL TANK SELECTION:

(1)	Low Pressure cocks	-	OPEN
(2)	Tank Selector switches to	-	BOTH TANKS ON
(3)	Cross-feed switch	-	OFF
(4)	Booster pump switches	-	ON

NOTE

(1) This selection allows each tank to feed fuel direct to its respective engine installation.

(2) All four booster pumps are switched ON to run continuously throughout the running time of the engines. This ruling applies only to a NORMAL tank to engine selection.

(3) The RED warning lights remain illuminated until the booster pumps are developing a pressure of 3 p.s.i. This pressure is sufficient to start the engines. Should a red light reappear under operating conditions, the respective pump may be defective, the fuel tank may be empty, or the warning device may be defective. If the engine continues to run satisfactorily and the fuel indicators show there is sufficient fuel in the tank, it is probably the fault of the warning device. If it appears that one of the other conditions exist, an alternative tank selection should be made.

(b) OUTBOARD TANK SELECTION:

(1)	Low Pressure cocks	-	OPEN
(2)	Tank Selector switches	-	BOTH AT OUTBOARD TANK
(3)	Cross-feed switch	-	OFF
(4)	Booster pump switches (outer only)	-	ON

NOTE

(1) This selection allows each pair of engines to be fed directly from their respective outboard tanks.

(c) INBOARD TANK SELECTION:

(1)	Low Pressure cocks	-	ALL OPEN
(2)	Tank Selector switches	-	BOTH AT INBOARD TANK
(3)	Cross-feed switch	-	OFF
(4)	Booster pump switches (inner only)	-	ON

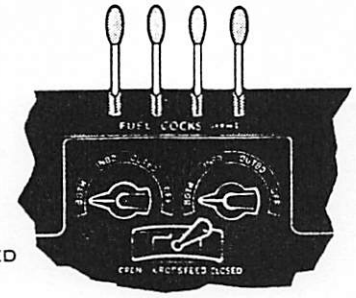
NOTE

(1) This selection allows for each pair of engines to be fed from their respective inboard tanks.

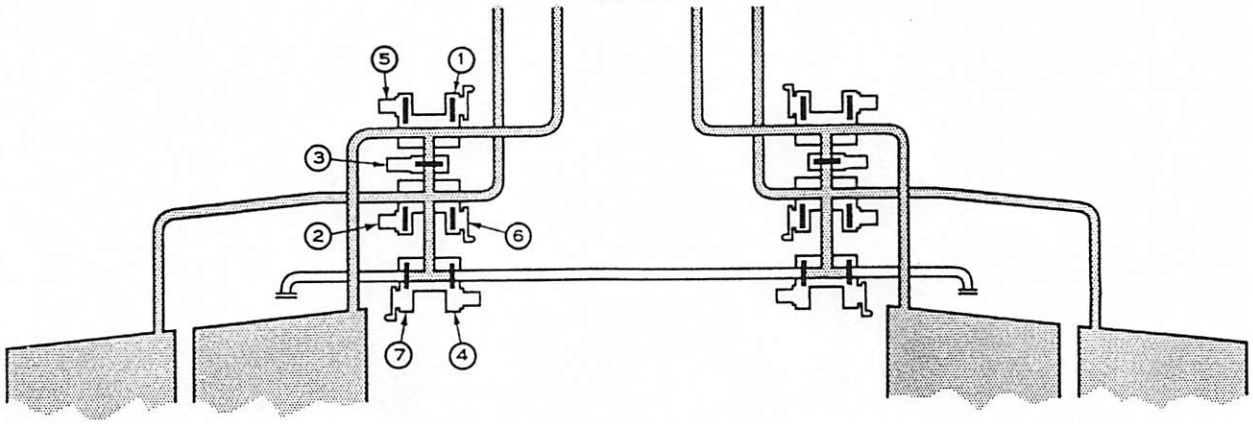
ALTERNATIVE FUEL TANK SELECTIONS

FIG. 52

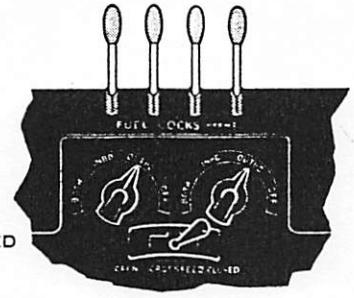
1. LOW PRESSURE FUEL COCK. INBOARD ENGINE
2. OUTBOARD TANK SHUT-OFF COCK
3. FUEL TRANSFER VALVE
4. CROSSFEED COCK
5. INBOARD TANK SHUT-OFF COCK
6. LOW PRESSURE FUEL COCK. OUTBOARD ENGINE
7. DRAIN VALVE. MANUALLY OPERATED



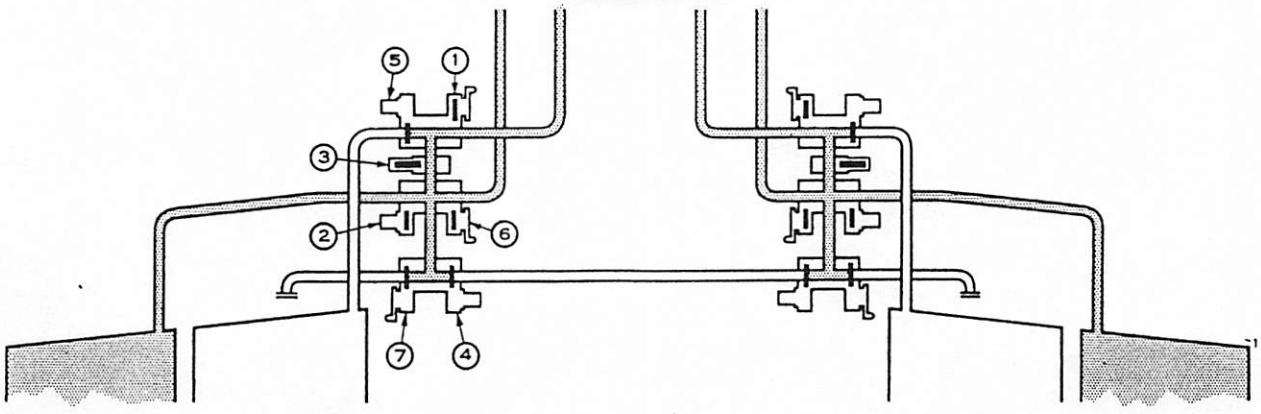
INDIVIDUAL TANKS FEEDING DIRECTLY TO CORRESPONDING ENGINES VALVES 1, 5, 6 AND 2 OPEN ON EACH ASSEMBLY



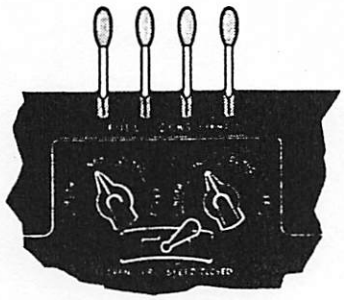
1. LOW PRESSURE FUEL COCK. INBOARD ENGINE
2. OUTBOARD TANK SHUT-OFF COCK
3. FUEL TRANSFER VALVE
4. CROSSFEED COCK
5. INBOARD TANK SHUT-OFF COCK
6. LOW PRESSURE FUEL COCK. OUTBOARD ENGINE
7. DRAIN VALVE. MANUALLY OPERATED



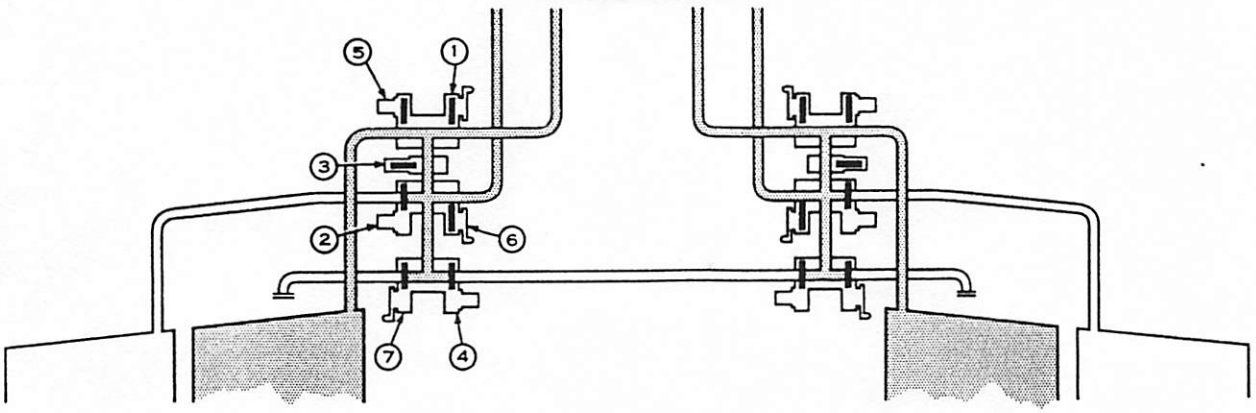
OUTBOARD TANKS FEEDING ALL FOUR ENGINES. VALVES 1, 2, 3 AND 6 OPEN ON EACH ASSEMBLY



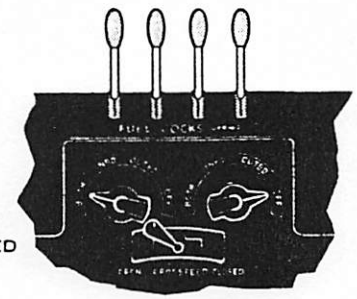
1. LOW PRESSURE FUEL COCK. INBOARD ENGINE
2. OUTBOARD TANK SHUT-OFF COCK
3. FUEL TRANSFER VALVE
4. CROSSFEED COCK
5. INBOARD TANK SHUT-OFF COCK
6. LOW PRESSURE FUEL COCK. OUTBOARD ENGINE
7. DRAIN VALVE. MANUALLY OPERATED



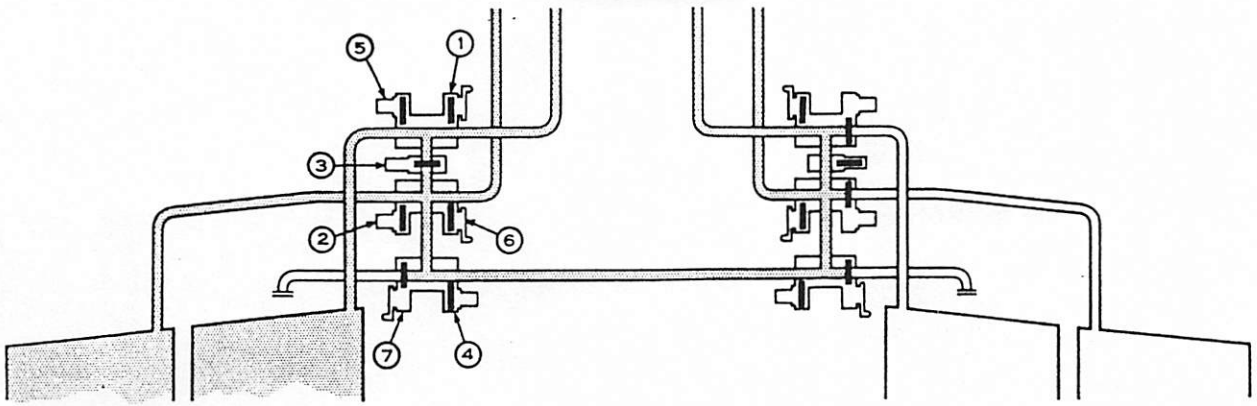
RELATIVE INBOARD TANKS FEEDING BOTH ENGINES. VALVES 1, 5, 3 AND 6 OPEN ON EACH ASSEMBLY



1. LOW PRESSURE FUEL COCK. INBOARD ENGINE
2. OUTBOARD TANK SHUT-OFF COCK
3. FUEL TRANSFER VALVE
4. CROSSFEED COCK
5. INBOARD TANK SHUT-OFF COCK
6. LOW PRESSURE FUEL COCK. OUTBOARD ENGINE
7. DRAIN VALVE. MANUALLY OPERATED

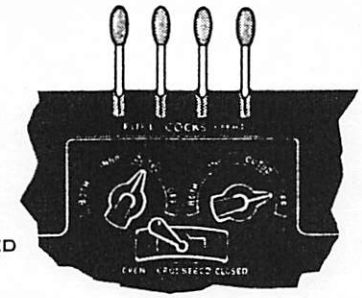


LEFT OUTBOARD TANK SUPPLYING LEFT OUTBOARD ENGINE ONLY. LEFT INBOARD TANK FEEDING REMAINING THREE ENGINES. VALVES 1, 6 AND 4 OPEN ON EACH ASSEMBLY.
VALVES 2 AND 5 OPEN ON LEFT ASSEMBLY ONLY.
VALVE 3 OPEN ON RIGHT ASSEMBLY ONLY.

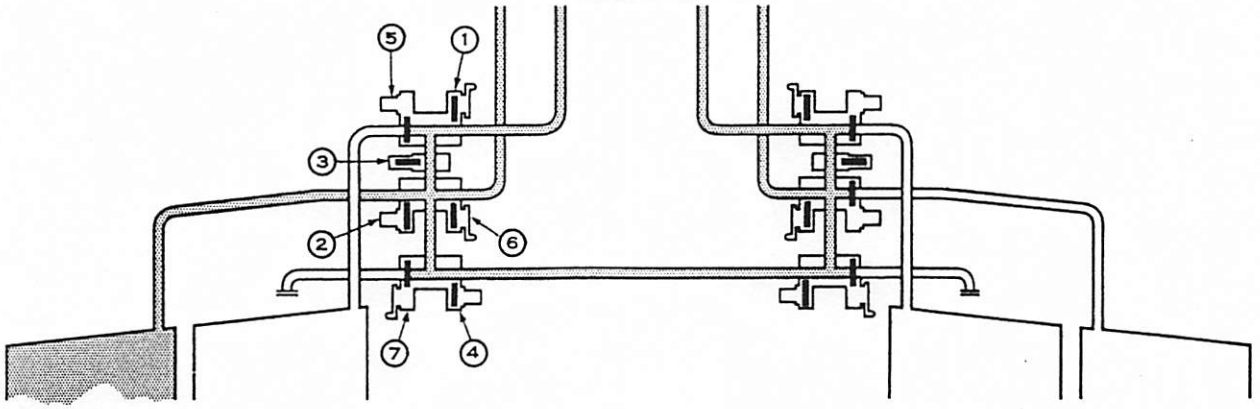


ALTERNATIVE FUEL
TANK SELECTIONS
FIG. 52

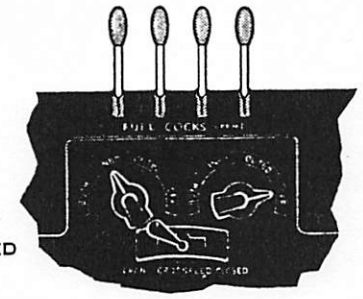
1. LOW PRESSURE FUEL COCK, INBOARD ENGINE
2. OUTBOARD TANK SHUT-OFF COCK
3. FUEL TRANSFER VALVE
4. CROSSFEED COCK
5. INBOARD TANK SHUT-OFF COCK
6. LOW PRESSURE FUEL COCK, OUTBOARD ENGINE
7. DRAIN VALVE, MANUALLY OPERATED



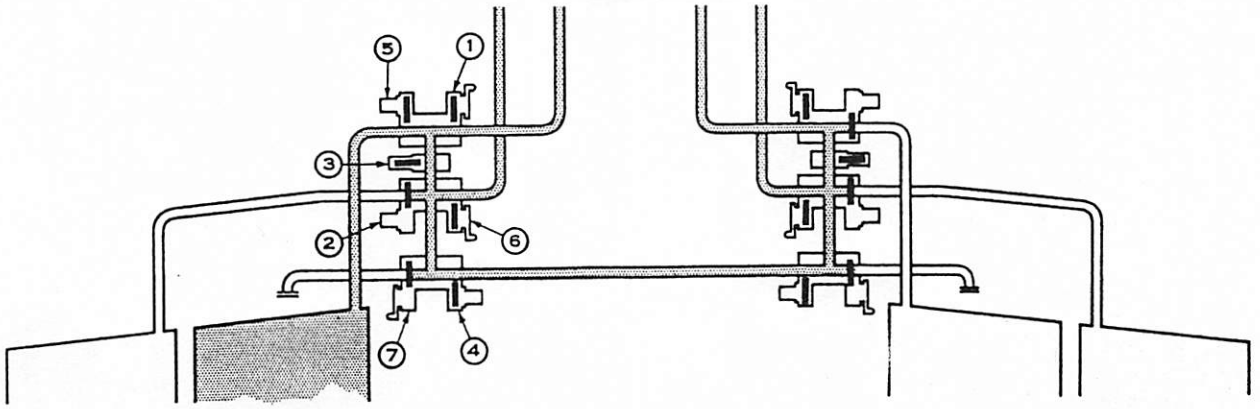
LEFT OUTBOARD TANK SUPPLYING ALL ENGINES. VALVES 1, 3, 6 AND 4 OPEN ON EACH ASSEMBLY
VALVE 2 OPEN ON LEFT ASSEMBLY ONLY



1. LOW PRESSURE FUEL COCK, INBOARD ENGINE
2. OUTBOARD TANK SHUT-OFF COCK
3. FUEL TRANSFER VALVE
4. CROSSFEED COCK
5. INBOARD TANK SHUT-OFF COCK
6. LOW PRESSURE FUEL COCK, OUTBOARD ENGINE
7. DRAIN VALVE, MANUALLY OPERATED



LEFT INBOARD TANK FEEDING ALL FOUR ENGINES VALVES 1, 3, 6 AND 4 ARE OPEN ON EACH ASSEMBLY
VALVE 5 IS OPEN ON LEFT ASSEMBLY ONLY.



(d) TO SUPPLY THREE OR ALL ENGINES FROM ONE TANK:

(1)	Low Pressure cocks	-	ALL OPEN
(2)	Tank Selector switches	-	ONE AT OFF Other at INBOARD TANK, OUTBOARD TANK, or BOTH TANKS ON, according to requirements.
(3)	Cross-feed switch	-	ON
(4)	Booster Pump switches	-	ON for each supply tank used.

NOTE

(1) All four engines can be fed from any one tank when this selection is made.

(2) When operating all engines from one tank, (through the cross-feed system) the four L.P. red warning lights will show when engine speed exceeds 13,600 r.p.m.

(3) It is permissible to operate all engines from one tank under full power conditions.

3-12 AUXILIARY FUEL SYSTEM MANAGEMENT: (PROVISIONAL)

(1) The auxiliary fuel system is intended for use on long range flights. The system should not be used for take-off or landing. Selections should be made only when the aircraft has reached the desired operating altitude.

(2) Fuel pressurized from the cabin through each individual tank filler neck (once the cap is removed) flows to relative inboard wing tanks, once the individual manually operated Whittaker slide valve controls are turned to the ON position.

(3) Due to the effect on aircraft loading the contents of the right and left rear tanks must first be used.

(4) The following procedure must be strictly adhered to when the system is being used. To operate:

(a) Select INBOARD TANKS ONLY on the NORMAL fuel system.
(b) Remove the right and left rear tank filler caps and check that the contents indicators rise above the level of the filler necks and that the external vents are sealed.
(c) Select right and left rear fuselage tanks ON.

NOTE

(1) As the fuel contents of the rear fuselage tanks are depleted, the float valve attached to the contents indicator will fall into the sump of its respective tank and automatically close the tank outlets. When this condition occurs, proceed as follows:

(d) Place rear tank selectors in the OFF position and replace the tank filler caps.
(e) Carry out identical procedures for the two forward tanks as outlined in items (b) to (c).
(f) Once the auxiliary fuselage tanks have been emptied the contents of the wing tank should be checked and their selector switches re-adjusted so that each tank supplies fuel to its respective engine. Providing the fuel consumption of each individual engine is equal, no further selection should be necessary before the final approach and landing. In the event of unequal fuel consumption, the tank contents should be balanced laterally by means of the cross-feed and the selector valves.
(g) In the event of cabin pressure failure, fuel will continue to flow from the fuselage tanks to the inboard wing tanks by means of gravity.

3-13 MANAGEMENT OF WATER METHANOL INJECTION SYSTEM:

(1) This system is normally intended for use only when taking off from short runways in hot weather.

(2) When the electrical services for this system are switched ON, be sure that the two TANKS FULL green lights are illuminated.

(3) Carry out NORMAL engine starting procedure as outlined in para. 3-3.

(4) On completion of the taxy run, and when the aircraft is lined up on the runway ready for take-off, apply FULL brakes, open up throttles to the FULLY OPEN position, then press the PUSH ON switch.

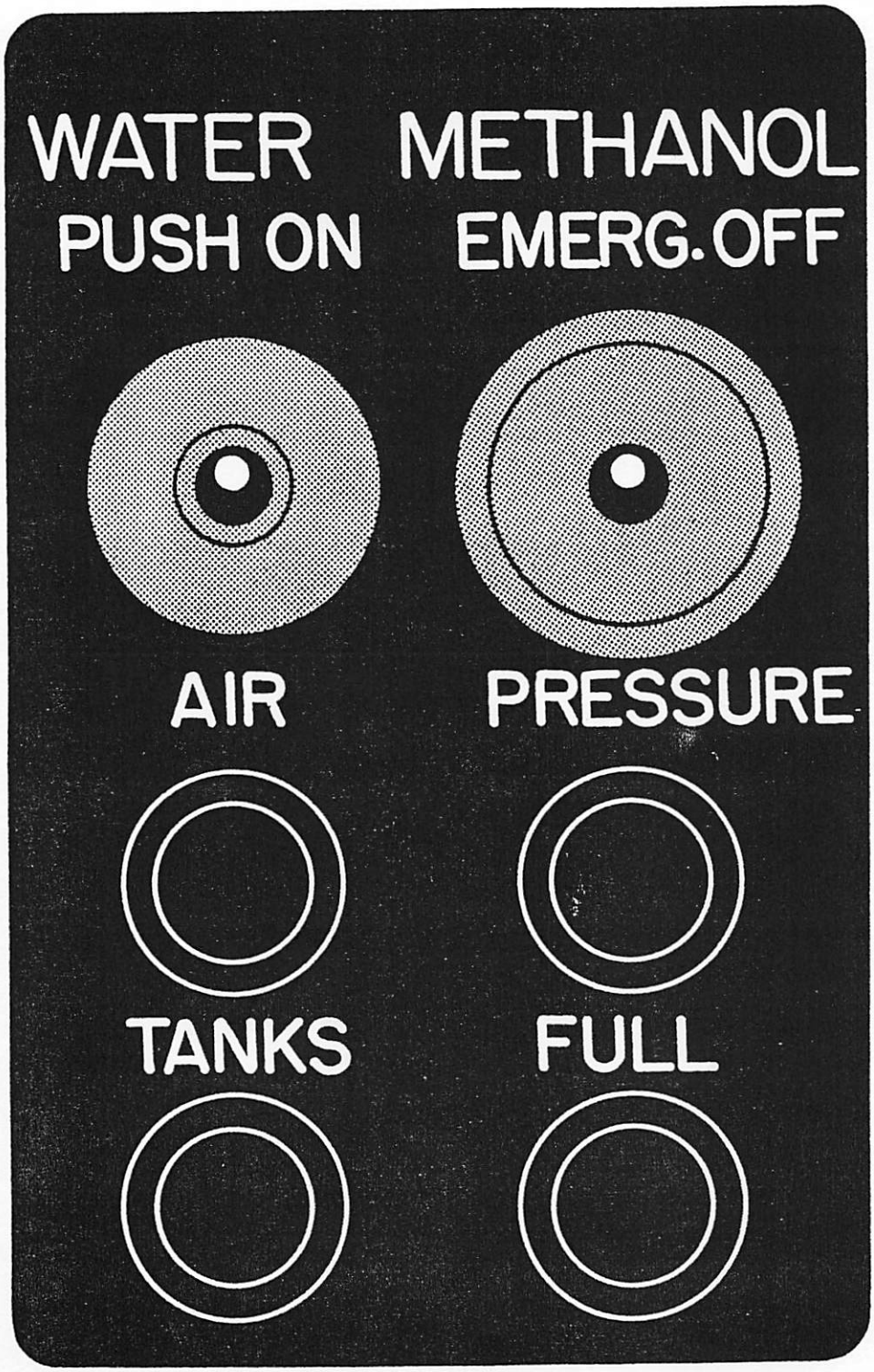
(5) Watch for the illumination of the AIR PRESSURE LIGHTS. These will indicate that the tanks are pressurized and delivery of water methanol has commenced.

(6) Release the brakes and take-off.

NOTE

(1) Injection will cease immediately the throttles are retarded from the fully open position, or when the EMERGENCY OFF button is pressed.

(2) Jet pipe temperatures should not exceed 700 degrees C. during injection.


WATER
METHANOL
PUSH ON
EMERG. OFF
AIR
PRESSURE
TANKS
FULL

3-14 ENGINE GROUND CHECK:

(1) Open up the engines to 4,000 r.p.m. Check generator output, voltmeter and ammeter readings. The voltmeter should register 28.5 volts, while the ammeters should indicate 60 amps., or higher.

NOTE

(1) These readings may vary depending on the amount of DC loads switched ON in the aircraft and on the condition of the batteries. The type of batteries used in this system are designed to permit a high rate of charge (approximately 368 amps.) for a short period of time. This high charge rate is dependent on the amount of discharge the battery has been subjected to during ground testing.

(2) Carry out the necessary control checks, including hydraulic and vacuum pump functional checks, etc.

(3) Open up throttle and pause at 13,600 r.p.m. Note that a minimum oil pressure of 30 p.s.i. is indicated.

(4) At full throttle, check that the governed r.p.m. is 14,550 ± 50 and that the J.P.T. does not exceed 680 degrees C. The J.P.T. should normally read approximately 30 degrees C. below maximum.

(5) Carry out one acceleration check of NOT LESS than 10 seconds duration. The range of this check, is from idling to take-off speeds. During acceleration, the J.P.T. may exceed the maximum of 680 degrees C. This increase in temperature should only be momentary, the drop taking place when engine r.p.m. is stabilized.

NOTE

(1) Be sure that the engines have reached full idling r.p.m. before opening the throttles.

(2) Before opening up, check that a minimum oil pressure of 5 p.s.i. is indicated, and that J.P.T. does not exceed 500 degrees C.

(3) Most troubles appear as increased J.P.T.'s at specified engine speeds.

(4) The J.P.T. for a given engine condition tends to increase with the life of the engine.

3-15 MANAGEMENT OF AUTOMATIC ANTI-SKID CONTROLS:

(1) The control panel for this service comprises four GREEN operating lights, four RED warning lights and a master ON-OFF switch. The panel is

located on the left side of the flight deck adjacent to the pilot's instrument panel (Fig. 8). The system is controlled in the following manner:

(a) Circuit Breakers - ON
(b) Master Control switch - ON AT ALL TIMES

NOTE

(1) Normal braking procedure should be carried out at all times.

(2) When the aircraft is taxying and the system is ON, all lights should be OUT. Any signs of malfunctioning will be indicated on the red warning lights.

(3) When making an approach with the landing gear DOWN and the system ON, check that the four GREEN lights are illuminated. (This condition will indicate that the system is armed).

(4) Approximately four seconds after touch down, the green lights will go OUT. With skids resulting from heavy braking, the GREEN lights will flash ON and OFF with each skid correction.

(5) Electrical failure of automatic control will be indicated at all times by operation of the RED WARNING lights. When an electrical failure takes place, a period of approximately four seconds will elapse before the warning lights will show.

(6) If the automatic anti-skid control fails, place the MASTER switch in the OFF position.

(7) During ground handling, the following points relative to brake operation should be observed:

(a) Landings involving the heavy use of brakes should not be made at intervals of less than ten minutes.
(b) When heavy differential steering is used in conjunction with (a); the time interval should be doubled.
(c) It is permissible to apply and to leave the brakes in the parked position after normal use.

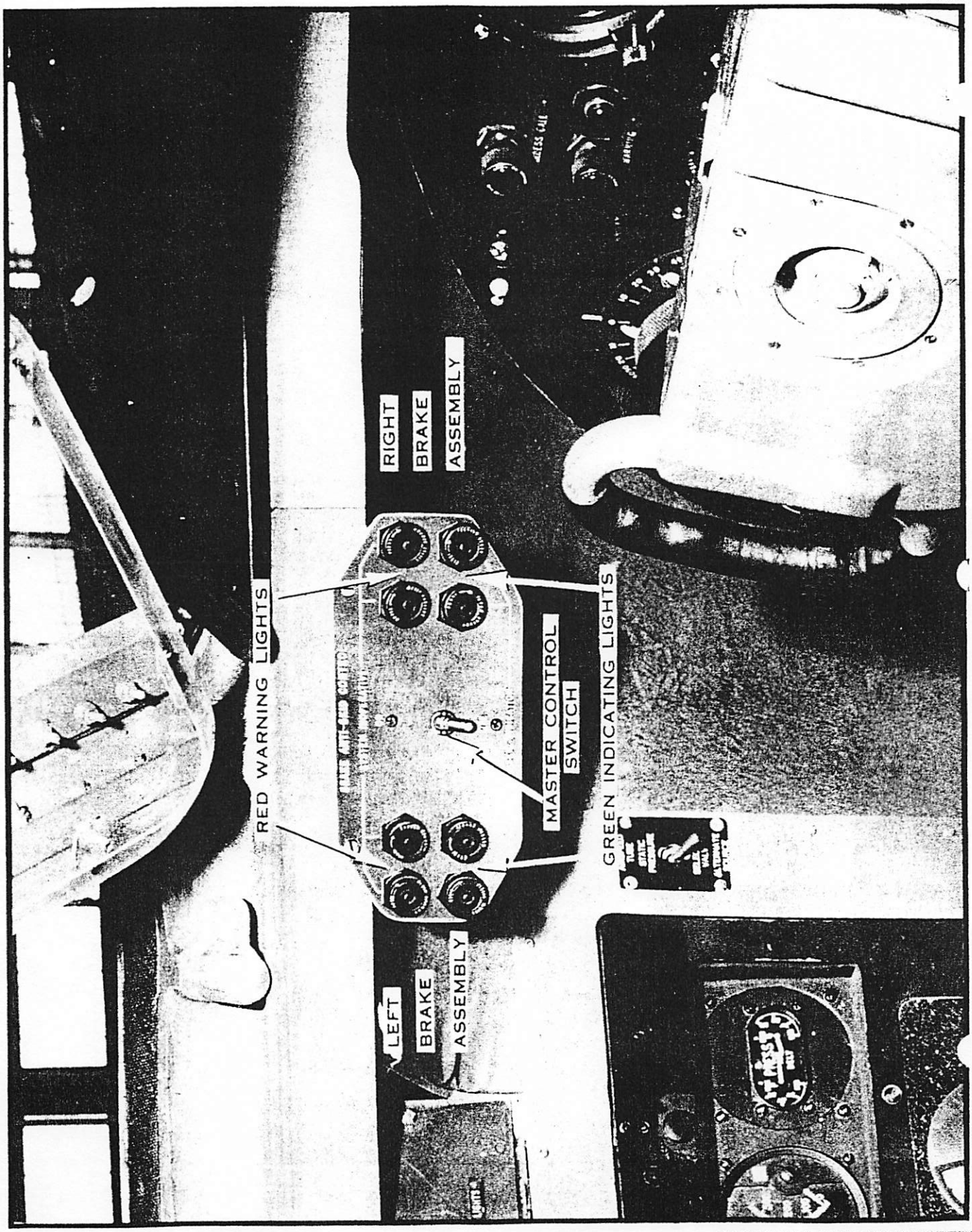
3-16 AUTO PILOT PRE-FLIGHT CHECKS: (PROVISIONAL)

(1) Switch on power master switch AC and DC circuits.

(2) Allow gyros approximately two minutes to reach correct speed - 20,000 r.p.m.

(3) Press the caging button.

ANTI-SKID CONTROL
PANEL
FIG. 54



(4) Check that beacon light flashes once every 1½ seconds.

(5) Align the three controller indices to zero.

(6) Centralize all manual controls.

(7) Push Auto-Pilot engagement switch ON.

(8) Check that all controls are engaged.

(9) Use controller to check each control channel for correct operation.

(a) Bank knob for aileron operation.
(b) Pitch knob for elevator operation.
(c) Pistol-grip handle for rudder operation.

NOTE

(1) Depress trigger switch when moving handle.

(10) Set the altitude control switch to ON.

(11) Engage the inter-lock altitude switch on the controller.

(12) Disengage the inter-lock altitude switch, check that altitude control switch is released by this action.

(13) Press the caging button, check that PUSH AUTO PILOT ON switch automatically disengages. Re-engage the auto-pilot.

(14) Press the release button on the pilot's handwheel, check that the clutch switch (PUSH AUTO PILOT ON) extends upwards to indicate that the auto-pilot is disengaged.

(15) Re-engage auto-pilot. Pull mechanical auto-pilot release lever on the side of the pedestal. Check that auto-pilot is disengaged and only normal manual control is available.

CHAPTER 3

CONTROLLING OF CABIN PRESSURE

3-17 GENERAL:

(1) It is common practice to start controlling cabin pressure at 2,000 feet after taking off and to terminate control, prior to landing, at 2,000 feet above the airfield. This allows time for such essential operations as landing gear and flap drill etc., to be completed without interference. The figure of 2,000 feet is an arbitrary one, and can of course be amended to suit particular conditions and to suit particular operators requirements.

(2) After climbing 2,000 feet from take-off, the cabin height required for cruising is selected and the aircraft is then climbed at a predetermined rate, so that the cabin altitude goes from 2,000 feet to the selected cabin altitude (at a predetermined and acceptable rate of climb), while the aircraft climbs from 2,000 feet to its operating altitude. For descent, a cabin altitude equal to landing ground height plus 2,000 feet is selected and the aircraft descends at a predetermined rate while the cabin descends to the selected height at a given and acceptable rate.

(3) An alternative method of control is to leave the cabin altitude selector at 2,000 feet and continue to climb the aircraft to its cruising height. From 2,000 to an intermediate altitude (i.e. the altitude at which the maximum permitted differential pressure is reached with the cabin at 2,000 feet) the aircraft can climb at any desired speed. Above that intermediate altitude the cabin pressure and atmospheric pressure change at the same rate as the master unit maintains a constant differential pressure. Consequently, the latter stage of the climb must be at a controlled rate (say 300 f.p.m.). This point causes this alternative method of control to have a serious disadvantage in comparison with the method outlined at sub-para. 2. The action of the pressure controller when the various methods are being applied is described in the subsequent paragraphs.

(4) Before take-off the cabin altitude selector must be set to 2,000 feet above the airfield height (assume the airfield is at sea level in this case) and the selector mechanism must have completed its action before take-off commences. This results in the cam being preset by the electric motor and associated mechanisms so that the absolute capsule assembly has been lowered sufficiently to cause the outlet valve to increase the size of the outlet E a given amount, so that it is larger than the fixed inlet orifice D. Pressure cannot build up in the capsule chamber, consequently the discharge valve offers a negligible resistance to the passage of air from the cabin. Between 0 and 2,000 feet therefore, the cabin pressure is substantially the same as the atmospheric pressure.

(5) As the aircraft climbs from 0 to 2,000 feet the absolute capsule pack expands as the atmospheric pressure becomes less, and the movement of the beam causes the outlet E to close gradually. At 2,000 feet the outlet orifice has closed sufficiently to cause a pressure inside the capsule chamber sufficient to make the discharge valve offer a resistance to the air passing from the cabin.

(6) The resistance of the discharge valve is such that it will keep the cabin at a constant pressure, and therefore a constant altitude. Should the aircraft continue to climb with 2,000 feet set on the selector, the cabin will remain at 2,000 feet and thus the differential pressure will increase. This increase in differential pressure will cause the differential capsule pack to be compressed until, when the maximum permitted figure is reached, the pin of the capsule end fitting comes against the beam.

(7) Any further climb and consequently any tendency to increase the differential pressure beyond the maximum permitted causes the differential capsules to compress further and pull down on the beam, causing the outlet to increase in size, reduce the capsule chamber pressure and lessen the resistance of the discharge valves. This results in the cabin altitude increasing slightly so that the absolute capsules extend a little.

(8) If the aircraft continues to climb above this point the action is smoothly progressive as the differential capsules pull down on the left hand end of the beam and the absolute capsules allow the right hand end to lift. The result is that the cabin altitude increases as the aircraft climbs and the maximum permitted differential pressure is not exceeded.

(9) The best method, and one which obviates the disadvantage mentioned in sub-paragraph 3 is to select a cabin altitude in accordance with the projected aircraft cruising height, and the maximum differential pressure of the aircraft, so that the cabin climbs from 2,000 feet to the selected altitude at an acceptable rate while the aircraft climbs from 2,000 feet cruising height, at a predetermined rate of climb. If this form of control is followed, the required cabin altitude (assume 8,000 feet) is set on the cabin altitude selector at 2,000 feet and the pilot is instructed to climb at a rate that will ensure that the aircraft reaches its operating altitude at the time that the cabin reaches 8,000 feet, having climbed from 2,000 feet at an acceptable rate of climb, say 300 feet per minute.

(10) When the pointer is set to 8,000 the electric motor is switched on, the red indicator lamp lights, and the camshaft rotates in the direction required to decrease the cam lift. Decreasing the cam lift causes the absolute pack to be lowered gradually so that the outlet valve is adjusted to decrease the capsule chamber pressure. The discharge valve resistance lessens slightly to cause cabin pressure to drop and the altitude to increase. The consequent lowered pressure in the cabin is transmitted to the capsule chamber, the absolute capsules increase in overall length and thus decrease the outlet valve opening.

(11) In actual fact the action as described is smooth and progressive; as the aircraft climbs, the camshaft action causes the absolute capsules to be lowered, and the falling cabin pressure causes the capsules to extend in overall length. The net result is that the outlet valve is adjusted, and the discharge valve resistance is varied at a very slow rate so that the cabin climbs very slowly to 8,000 feet over a period of 20 minutes (if the rate of change selector is set to control the motor for a speed of climb of 300 feet per minute). During this time the differential pressure has increased to the maximum permitted.

(12) When the cabin is at 8,000 feet the absolute capsule end of the beam comes against a stop, the motor is switched off, and the differential capsules have become compressed by the increasing differential pressure. Exceeding the maximum permitted differential pressure will be prevented by the action of the differential capsules on the beam and outlet valve.

(13) During descent the reverse procedure occurs. The cabin altitude selector is set from 8,000 to 2,000, the motor starts, the green indicator lamp lights, and the camshaft rotates in the direction required to increase the cam lift and raise the absolute capsule assembly very slowly.

(14) This controlling action results in the cabin pressure descending at a controlled rate to 2,000 feet, as the outlet valve is progressively adjusted by the balancing action of the cam raising the pack bodily, and the increasing pressure shortening the overall length of the absolute capsule pack.

(15) If the speed of aircraft descent is correct the aircraft arrives at 2,000 feet at the same time as the cabin reaches that altitude. The motor is then switched off automatically and the green light goes out. During subsequent descent, the outlet valve is open to such an extent so that insufficient pressure is transmitted to the discharge valve to cause it to offer any appreciable resistance to the air from the cabin, and the cabin and aircraft altitudes decrease at a common rate.

(16) If it is required to increase the cabin altitude during flight, while the aircraft maintains a constant height, the cabin altitude selector is turned to indicate the required altitude. The motor starts, the red light comes on and the camshaft rotates to reduce cam lift. The result is that the outlet valve is adjusted and the size of the outlet is increased very gradually, decreasing the pressure in the capsule chamber at a like rate. Consequently, the resistance of the discharge valve is gradually lessened until the resistance is that required to maintain a cabin pressure equal to the new cabin altitude. When this is established the motor stops and the red light goes out.

(17) If it is required to decrease the cabin altitude, while maintaining a constant aircraft height, the altitude selector is set accordingly. The reverse action to that indicated in the preceding paragraph occurs, the camshaft raises the absolute pack bodily and the outlet valve is adjusted to increase the pressure in the capsule chamber and so cause the discharge valve to increase its resistance. When the required cabin altitude is achieved, the motor is

automatically switched off and the light goes out. In both these cases the absolute capsule assembly keeps the new cabin altitude constant after the action of the selector has ceased.

3-18 GROUND TESTING OF PRESSURE CONTROLLER:

(1) To set the unit so that the differential setting of the pressure controller can be checked, press one of the clutch buttons protruding through the control panel, and set the associated indicator wheel so that "Ground Test" appears against the datum line on the panel (Fig. 29).

(2) Depression of the clutch button (16) causes the operating lever assembly (15) to rotate about its axis. Double levers at the bottom of the assembly cause the sliding collar to withdraw the pin locating the clutch plate (23) with the pin carrier (22) thus permitting the hollow shaft bearing the indicator wheel (18) and the operating cam (24), to be rotated independent of the remainder of the camshaft.

(3) When the hand indicator wheel is at Ground Test, the cam is applying maximum lift to the absolute capsule pack, and the outlet valve is closed. The right hand end of the beam is against its stop.

(4) The foregoing operation causes the absolute pack of the associated capsule chamber to be non-effective, and when air is introduced to the cabin by means of blowers, or by a Ground Test truck, the discharge valve resistance is sufficient to allow pressure to build up in the cabin. Should the cabin pressure tend to exceed the maximum permitted differential pressure, the outlet valve is opened by the action of the differential capsule pack, and the cabin pressure is kept constant at the maximum permitted value. It is thus possible to check the differential setting of each individual control mechanism of the master unit.

CHAPTER 4

OPERATING INSTRUCTIONS FOR RADIO CONTROLS AND INSTRUMENTS

3-19 RADIO POWER PANEL:

(1) The radio power panel (Fig. 48) is located behind the pilot's seat. Circuit breakers are provided for all DC supplies, and fuses for all AC supplies. Normally all circuits should be switched on when the crew enters the flight deck.

(2) The lower right-hand breaker operates the floodlight illuminating this panel. All radio equipment is turned on at the power panel. There are no power on-off switches on the radio control panels.

3-20 PEDESTAL PANELS:

(1) AUDIO PANELS are duplicated on the right and left sides of the upper row of the control pedestal for the convenience of the pilot and co-pilot (Fig. 50). They carry the controls and switches described in the following sub-paragraphs (2 to 7).

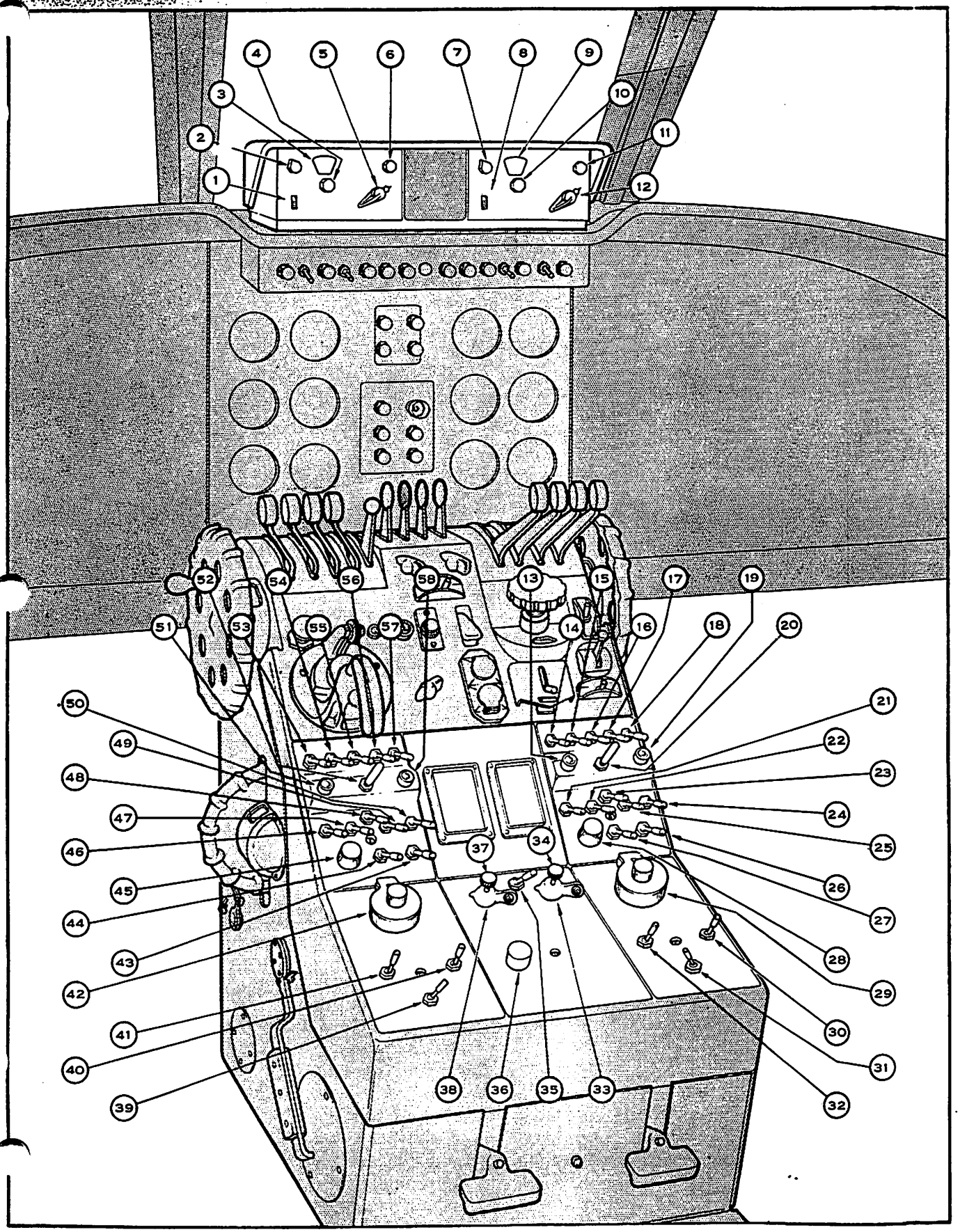
(2) AUDIO SELECTOR SWITCHES are provided for selecting the signals from any one or more receivers and from the interphone. Any two or more receivers, as well as interphone, may be heard simultaneously if desired.

(3) ADF R & V SWITCH enables selection of either red ADF or green ADF signals (simultaneous range and voice) giving improved quality compared with that obtained when the range and voice switches in the row above are turned on simultaneously. Red ADF is heard in the top switch position, green ADF in the bottom position. If range or voice is desired individually, the R & V switch should be left in the centre ('NIL') position and the individual range and voice switches used instead.

(4) HF COMMUNICATION-INTERPHONE-VHF COMMUNICATION MICROPHONE SWITCH has a large black lever and is located slightly above the centre of the panel between two pilot lights. In the centre position, marked 'TRANS OFF', it connects the microphone to the inter-phone amplifier. In the left hand position it connects the microphone to the VHF communication transmitter and simultaneously lights the amber VHF 'READY' light indicating that the VHF communication transmitter will operate when the microphone button in pressed. In the right hand position the microphone is connected to the HF communication transmitter and the blue HF 'READY' light goes on indicating that pressing the microphone button will put the HF transmitter on the air. It is possible for the HF transmitter to be operated by one crew member while the VHF transmitter is being operated by the other crew member. In order to hear the respective communication receivers, it is necessary to turn on the appropriate audio switch as described above. When communicating with towers or range stations which receive on HF and transmit

RADIO CONTROLS
THROTTLE PEDESTAL

FIG. 55



on LF, the automatic radio compasses (ADF) are used for receiving and the appropriate ADF audio switch is turned on.

(5) EMERGENCY-NORMAL SWITCH is usually wire-locked in the normal (DOWN) position. 'EMERGENCY' should be used only in case of failure of the isolation amplifier feeding the headphones or speaker. In the emergency position only one receiver signal or interphone can be selected at a time and the ADF R & V switch is inoperative.

(6) VOLUME CONTROLS are of the conventional type and control the volume of signals to headphones or speaker.

(7) MARKER AUDIO. Audio from the No. 1 marker receiver is fed to the pilot's audio panel only, and audio from the No. 2 marker receiver is fed to the co-pilot's audio panel only.

3-21 FREQUENCY CHART PANEL:

(1) The frequency chart panel is centrally disposed on the upper row of the control pedestal (Fig. 50). Its purpose is to hold the charts listing HF and VHF communication channel frequencies, and ILS localizer frequencies. ADF loop correction data may also be listed if desired.

3-22 VHF NAVIGATION PANELS:

(1) The VHF navigation panels are located in the lower row and are duplicated on the right and left sides of the control pedestal (Fig. 50).

(2) The left hand panel operates No. 1 localizer-omnirange receiver, No. 1 marker receiver and No. 1 glide path receiver. The right hand panel operates No. 2 localizer-omnirange receiver, No. 2 marker receiver and No. 2 glide path receiver.

(3) CHANNEL SELECTOR DIAL is used to dial directly the channel frequency of the omnirange and localizer receiver. For example to dial 113.7 megacycles:

(a) Spin outer rim of dial to 113.
(b) Turn inner knob of dial to .7.

The glide path channel is selected automatically by dialing the proper localizer channel. ILS channels are listed on the frequency chart panel. The localizer-omnirange receivers may also be used as auxiliary VHF communication receivers.

(4) TONE-PHASE SWITCH is used to set the localizer-omnirange receiver for either tone-modulated or phase-comparison localizer. Most localizers are still the tone type, although it is expected that eventually they will be changed over to the phase type. Two-course VHF ranges may also be received in the TONE position.

(5) MARKER SENSITIVITY SWITCH is of the conventional type; high sensitivity in the BROAD position and low sensitivity in the SHARP position.

(6) INDICATOR (REC. NO. 1 AND REC. NO. 2) SWITCH is at the bottom of the VHF navigation panel. The indicator switch on the left panel connects the pilot's crosspointer indicator and also the autopilot flight path computer to either No. 1 or No. 2 localizer and glide path receivers. On the right-hand panel this switch is inoperative.

3-23 COMMUNICATION PANEL:

(1) This panel is centrally disposed on the lower row of the flight deck pedestal and carries the following controls.

(2) VHF COMMUNICATION CHANNEL SELECTOR - lever knob in upper left corner of communication panel. This switch is marked 1 to 18 and GC for selecting VHF transmit-receive frequencies listed on the frequency chart panel.

(3) VHF COMMUNICATION SQUELCH CONTROL - round knob concentric with channel selector lever knob. This knob acts as a sensitivity control for the VHF communication receiver. The squelch suppresses all background noise unless a signal is present. Weak signals require clockwise adjustment of the knob giving lower squelch threshold level or, in other words, greater sensitivity.

(4) VHF COMMUNICATION VOLUME CONTROL - lower left section of panel. Conventional.

(5) GUARD RECEIVER SWITCH. Top centre of panel. The guard receiver switch is not used in this installation.

(6) HF COMMUNICATION CHANNEL SELECTOR SWITCH - lever knob is in the upper left corner of the panel. This switch is marked 1 to 20 for selecting HF transmit-receive frequencies listed on the frequency chart panel.

(7) HF COMMUNICATION SENSITIVITY CONTROL - is a round knob concentric with channel selector lever knob. This is a simultaneous sensitivity and volume control, and operates conventionally.

3-24 AUTOMATIC RADIO COMPASS (ADF) TUNING UNITS:

(1) These units are located above the centre of the centrally disposed engine instrument panel. Coloured knobs indicate red and green receivers.

(2) FREQUENCY RANGE - Carries the range band and standard broadcast band.

(3) BAND SELECTOR SWITCH - selects any one of the three frequency ranges covering the above bands. This switch is at the right side of the tuning unit.

(4) TUNING KNOB is at the centre of the tuning unit and is conventionally operated.

(5) DIMMER CONTROL is located at upper right corner of the tuning unit. It controls the brightness of tuning dial and tuning meter illumination.

(6) VOLUME CONTROL is positioned on the upper left corner of the tuning unit. It is conventionally operated.

(7) FUNCTION SWITCH is located to the left of the tuning unit. When placed at the 'REC' position the set functions as a straight receiver. When at the 'ADF' position the set functions as an automatic radio compass. Because automatic volume control is used on both 'REC' and 'ADF' functions in the standard model of this radio compass, range station cone of silence is not as pronounced as is the case with a receiver having manual volume control. However, if the volume is reduced to a very low level the cone of silence can be identified if necessary as a check on the accuracy of homing on a range station.

(8) DIRECTIONAL INDICATION. The usual arrangement is to have the RED ADF operating the narrow pointer of the pilot's radio magnetic indicator (RMI) and the GREEN ADF the narrow pointer of the co-pilot's RMI, the broad pointers being used for No. 1 and No. 2 omnirange respectively. If desired the connections in the radio junction box may be altered to give dual ADF presentation on both RMI's, the labelling of the instruments being changed accordingly.

3-25 INSTRUMENT PANEL CONTROLS:

(1) MARKER BEACON LIGHTS are conventional, except that dimmer and press-to-test features are provided.

(2) RADIO-MAGNETIC INDICATOR (RMI) is a dual pointer directional indicator. As mentioned under ADF the pointers may be operated by either ADF or omnirange receivers. The rotating scale of this instrument acts as a repeater of the flux gate or gyrosyn compass so that the instrument gives magnetic bearings of stations from the aircraft.

(3) CROSS POINTER INDICATOR. Both pointers are used for instrument landing system (ILS). The vertical pointer only is used for omnirange and two-course VHF range. Red alarm flags marked 'OFF' are provided opposite the ends of the two pointers. When a flag shows the signal is either absent or too weak for dependable operation. For use of the vertical pointer with omnirange equipment, refer to Collins Radio Company publication "Omni-directional Radio Range Equipment and Utilization".

(4) OMNI-BEARING SELECTOR (OBS). This instrument is used for course selection with omnirange equipment. The pilot's selector operates No. 1 omnirange receiver and the co-pilot's selector operates No. 2 receiver. Refer to Collins Radio Company publication "Omni-directional Radio Range Equipment and Utilization".

3-26 JACK PANELS: (PILOT & CO-PILOT)

(1) These Jack Panels are duplicated for use by pilot and co-pilot and are respectively located on the left and right flight deck side panels. (Fig. 8).

(2) Jacks are provided for headphones and microphones. A toggle switch on each jack panel sends signals through the headphones in the lower position and through the loudspeaker in the upper position. When the loudspeaker is used the volume control on the audio mixer panel operates exactly the same as when using headphones.

3-27 OBSERVER'S JACK BOX:

(1) This box is located to the right of observer's jump seat to the rear of the co-pilot.

(2) Jacks provided for observer's microphone and headphones. A switch enables selection of the signals from either the pilot's or co-pilot's audio panel. The microphone operates on interphone only, no provision being made for radio transmission from the observer's jack box.

3-28 AUTOMATIC FLIGHT CONTROL:

(1) Signals from the radio equipment in this aircraft, can be fed into the autopilot flight path computer for automatic flight control. The computer is connected in parallel with the pilot's cross-pointer indicator and is, therefore, actuated by the same signals as this indicator.

(2) AUTOMATIC RANGE FLYING may be done with omnirange or two-course VHF ranges.

(a) The omnirange receiver to be used (either No. 1 or No. 2) is tuned to the channel frequency of the omnirange station desired.

(b) The proper course is set on the omni-bearing selector (pilot's selector for No. 1 receiver, or co-pilot's selector for No. 2 receiver).

(c) The indicator (Rec. No. 1 - Rec. No. 2) switch on the pilot's VHF navigation panel is switched to the proper receiver.

(d) The autopilot sequencing switch is turned to the proper function (either blue left or blue right range).

NOTE

(1) For a clear explanation of this procedure, reference should be made to the Collins Radio Company publication "Omni-directional Radio Range Equipment and Utilization".

(e) Two-course VHF ranges may be used similarly with the exception that the omni-bearing selector is not operative. The TONE-PHASE switch must be in the TONE position. Operation is similar to localizer "blue right" procedure (see below).

(f) It is possible to fly an uninterrupted course from one VHF range station to another by starting off on No. 1 receiver, meanwhile tuning No. 2 receiver for the next station along the airway. As soon as the service range of the next station is reached, the indicator (Rec. No. 1 - Rec. No. 2) switch on the left VHF navigation panel is switched to the No. 2 position and the aircraft will then be flying a course navigated from the next station.

(3) AUTOMATIC INSTRUMENT LANDINGS can be proceded with in a similar manner to that used for omnirange flying, except that the localizer channel of the desired runway is tuned on the VHF navigation channel selector. The autopilot sequencing switch is first set at "localizer blue right". When the aircraft is aligned with the runway and the horizontal pointer of the cross-pointer indicator indicates that the glide path has been intersected, the autopilot sequencing switch is then switched to "localizer and glide path". The aircraft will then be controlled in both vertical and horizontal axes by the autopilot.

(4) It is also possible to fly a localizer outbound automatically by setting the autopilot sequencing switch to "localizer blue left". In this case the glide path signal is not utilized by the autopilot flight path computer. The autopilot will then maintain the flight path angle with respect to the ground that was in effect at the instant that the autopilot was engaged.

3-29 INTERPHONE CALL SYSTEM:

(1) An interphone handset is located at the cabin attendant's station. Interphone jacks are located at a number of external points enabling the ground crew to plug in handsets for communicating with the flight deck or with other external points.

(2) FLIGHT COMPARTMENT CALL LIGHT is located on the main instrument panel and indicates that the flight crew is wanted on the interphone. When the cabin attendant's handset is lifted from its hanger, the flight compartment call light glows continuously.

(3) CABIN ATTENDANT CALL BUTTON is located on the ceiling switch panel and operates a call buzzer at the cabin attendant's station.

(4) GROUND CREW CALL BUTTON is located on the roof switch panel and operates a buzzer in the nosewheel well to call ground crew.

(5) FLIGHT COMPARTMENT CALL BUZZER is operated by push buttons at the external interphone position.

CHAPTER 1.

FLIGHT OPERATIONAL DATA

4-1 GENERAL:

(1) The performance data supplied in this section is based on the following wing flap and landing gear position:

TAKE-OFF CONFIGURATION	FLAPS	LANDING GEAR
1st segment climb	25 deg.	Extended
2nd and 3rd segment climb	0 deg.	Retracted
4th segment climb	0 deg.	Retracted
En route configuration	0 deg.	Retracted
Approach configuration	12½ deg.	Retracted
Landing configuration	60 deg.	Extended

4-2 TEMPERATURE ACCOUNTABILITY:

(1) For temperature other than standard, the values of temperature accountability are shown in the relative performance charts.

4-3 PERFORMANCE DATA FOR DERWENT 5/17 ENGINE:

(1) GENERAL: The static performance of the R.D. 7 rating Derwent 5 engine at sea level is 3,600 lb. thrust at 14,700 R.P.M. The maximum permissable J.P.T. at this condition is 680°C.

(2) The non-dimensional performance graphs contained in this manual have been plotted on an R.P.M. basis; as a proportion of N_{Max} where N_{Max} is defined as the R.P.M. at which static thrust of exactly 3,600 lb. is obtained at sea level under standard conditions when no power is being absorbed by the engine accessories.

4-4 TABLE OF PERFORMANCE:

(1) The following table and relevant chart (Fig. 56) gives the performance at sea level under International Standard (I.C.A.N.) atmospheric conditions and with an intake ram efficiency of 90%.

(2) ENGINE RATING:

UNTIL CERTIFIED				
	ENGINE SPEED R.P.M.	TRUE AIR SPEED M.P.H.	THRUST LB.	FUEL CONSUMPTION LB./HR.
Maximum Take-off (5 minutes)	14,700	0 100 200 300 400 500	3,600 3,330 3,170 3,080 3,050 3,050	3,770 3,780 3,830 3,940 4,090 4,270
Maximum Continuous Power	14,100	0 100 200 300 400 500	3,090 2,820 2,650 2,530 2,460 2,430	3,170 3,180 3,230 3,300 3,400 3,520
Idling (Unrestricted duration)	3,500	0	120 max.	

(3) By virtue of very extensive test experience carried out at take-off or maximum power, the Derwent 5/17 engine has been approved for 5 minutes continuous operation at maximum conditions.

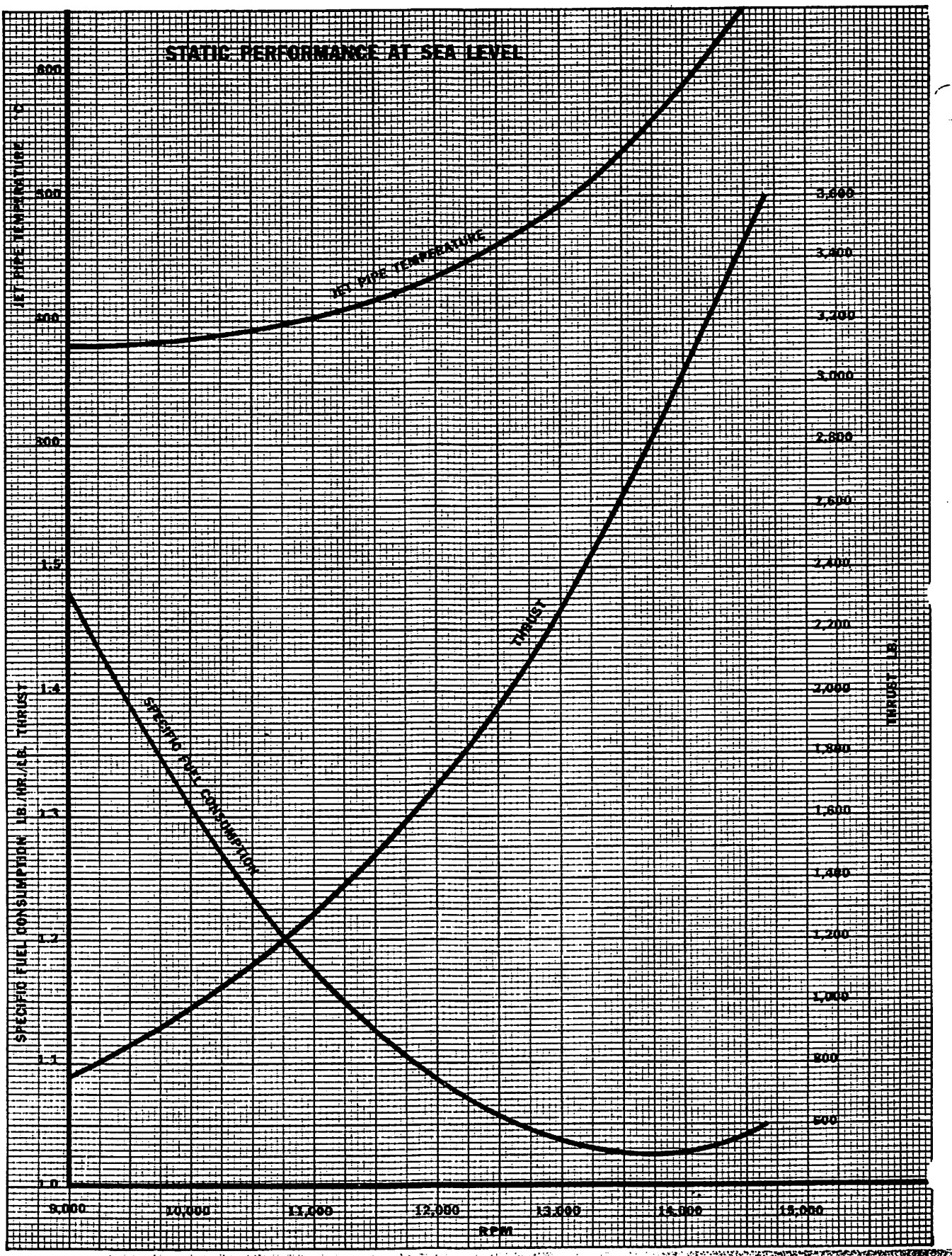
(4) The static performance at sea level is given in Fig. 56.

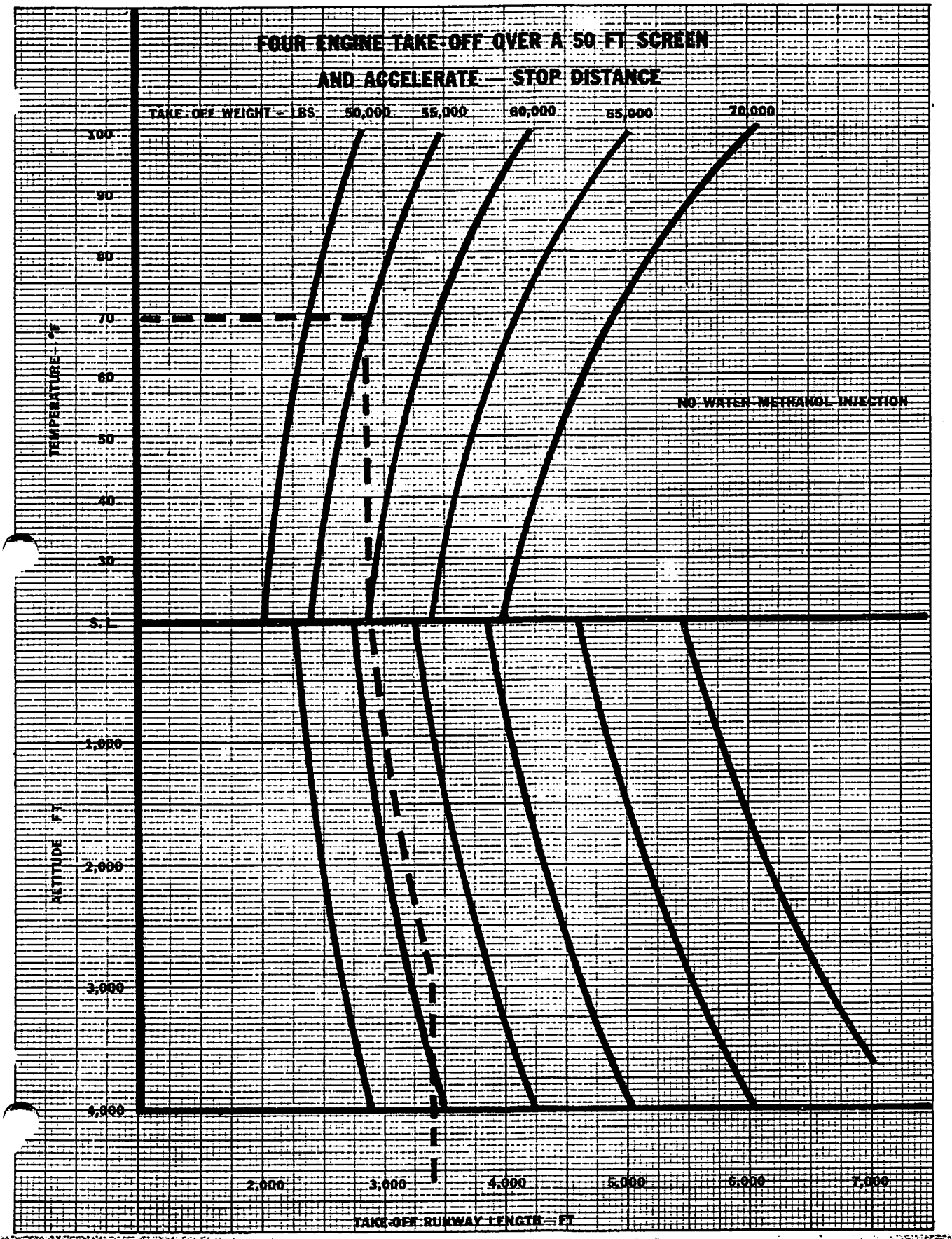
(5) In calculating flight performance, an air intake efficiency of 90% has been assumed. This efficiency is defined as the ratio of the isentropic temperature rise, associated with the increase in air intake pressure due to forward speed, to the actual temperature rise.

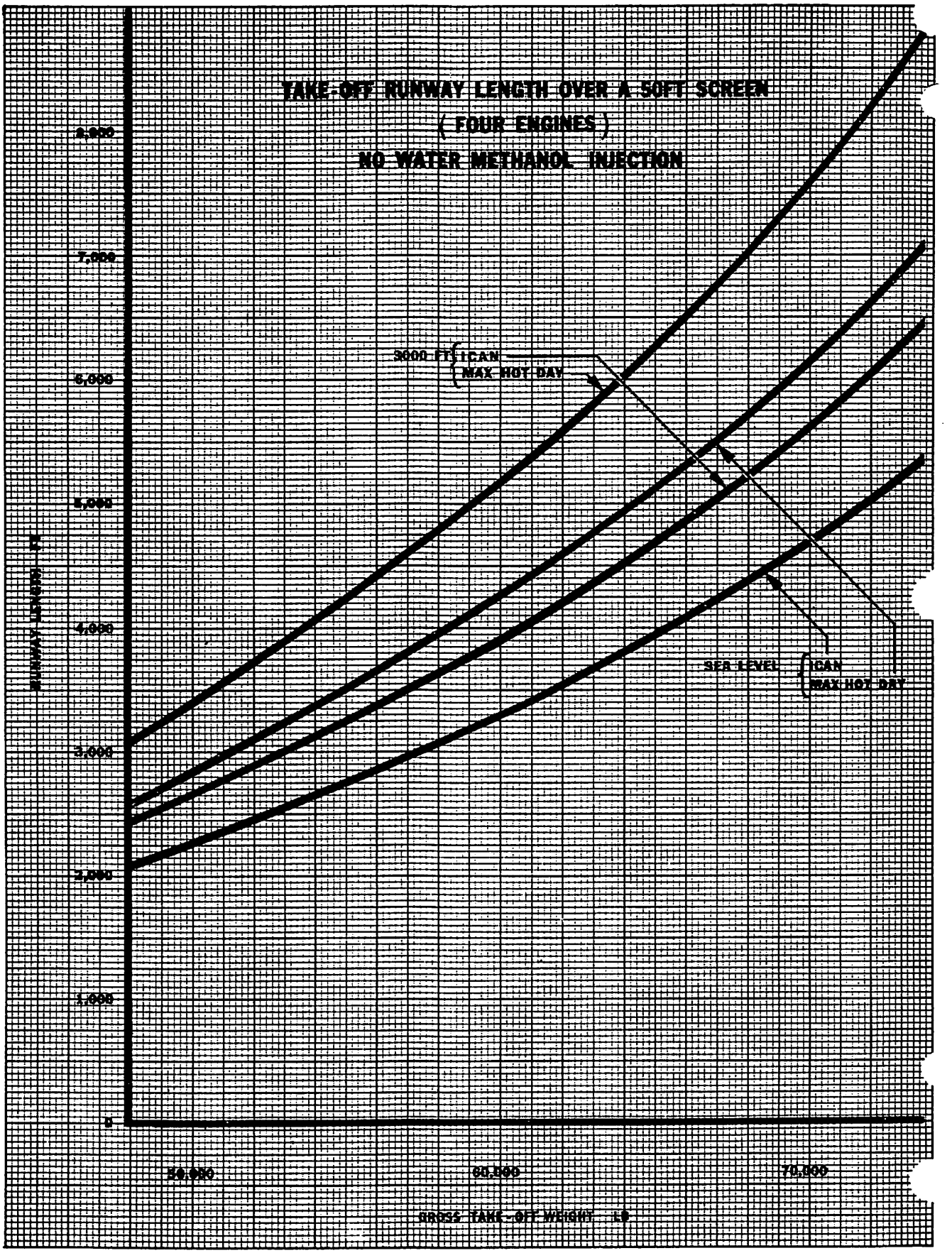
(6) The thrust at engine idling speed will not exceed 120 lb. at any aircraft speed.

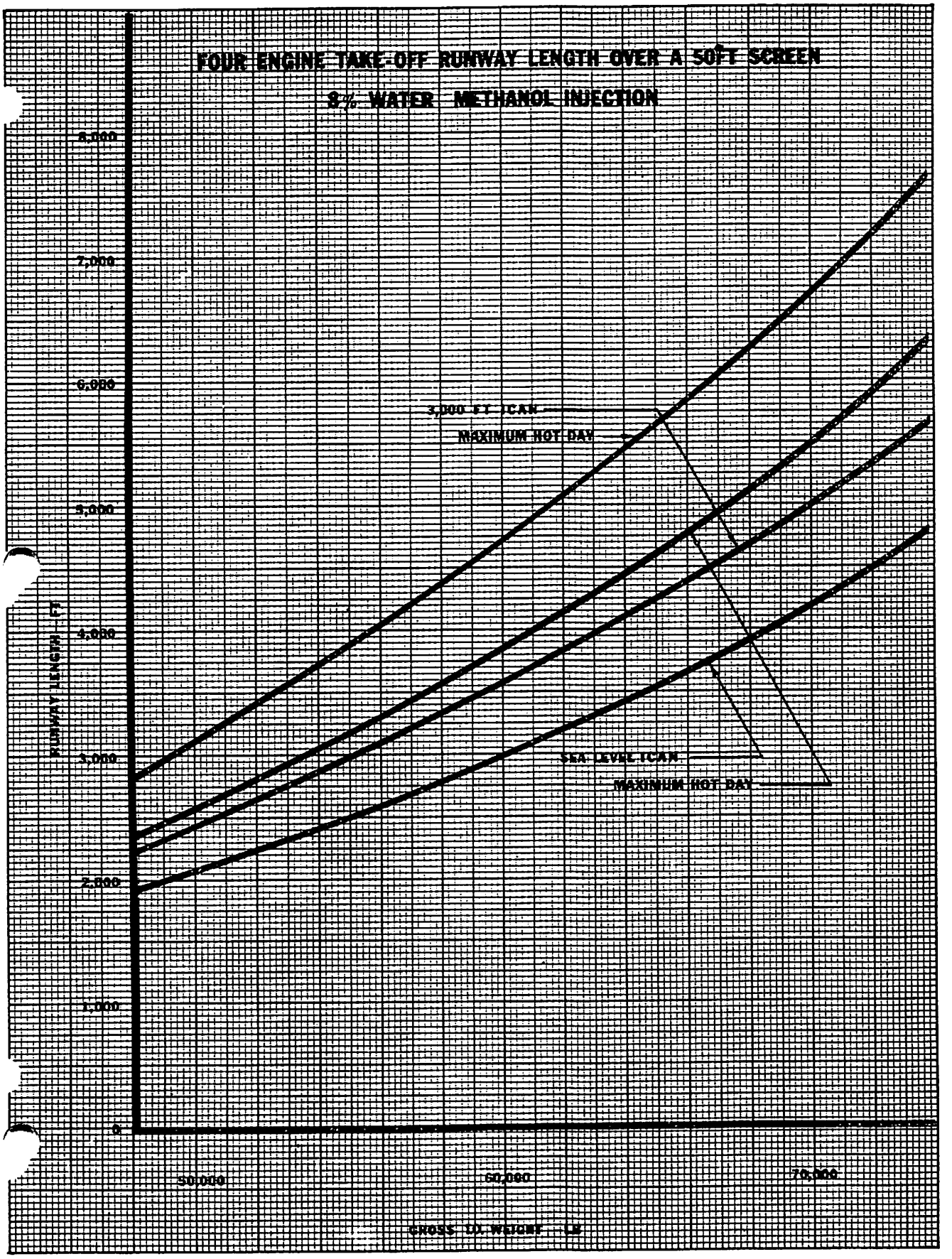
DERWENT 5/17 ENGINE
PERFORMANCE CHART

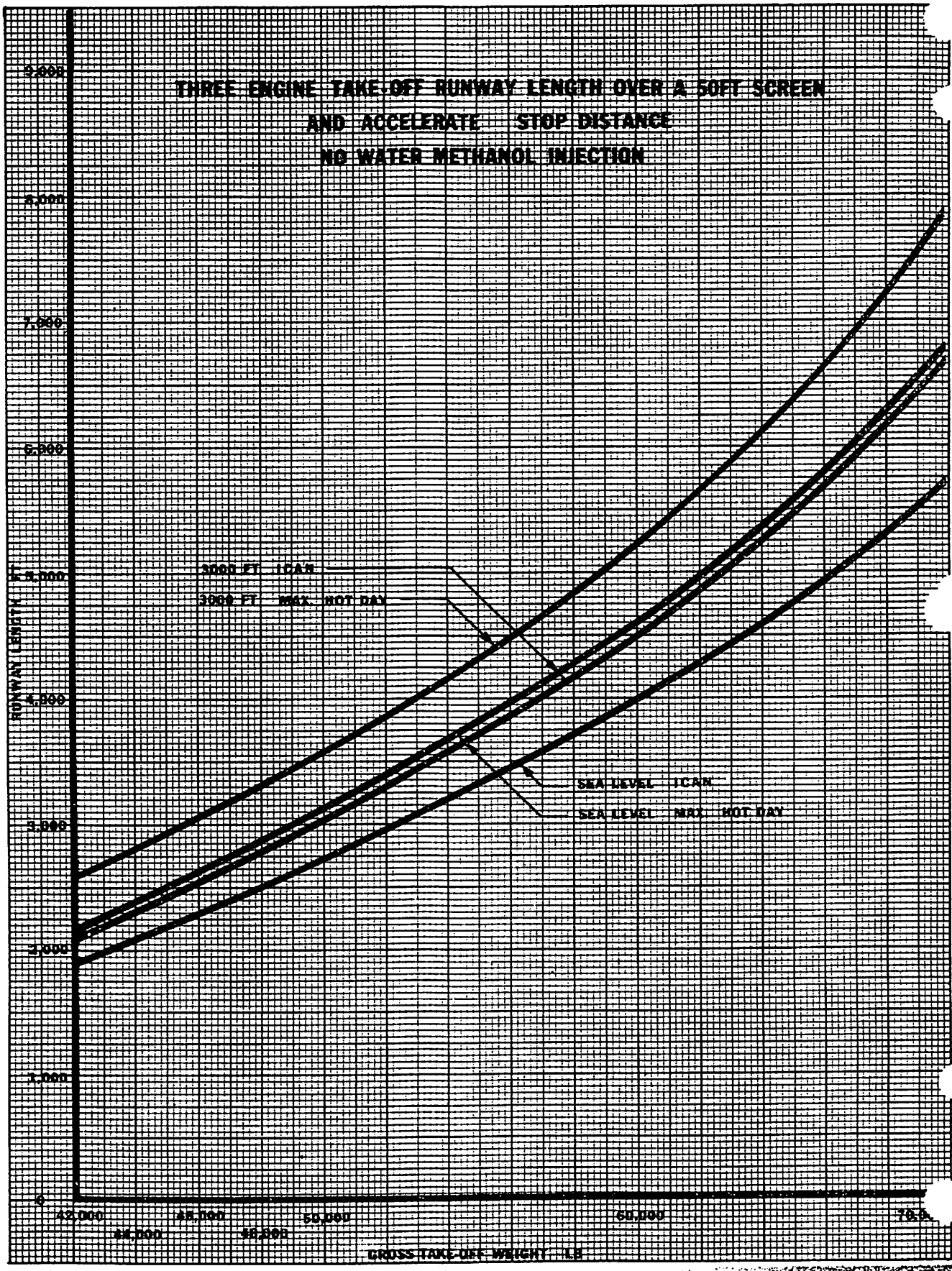
FIG. 56

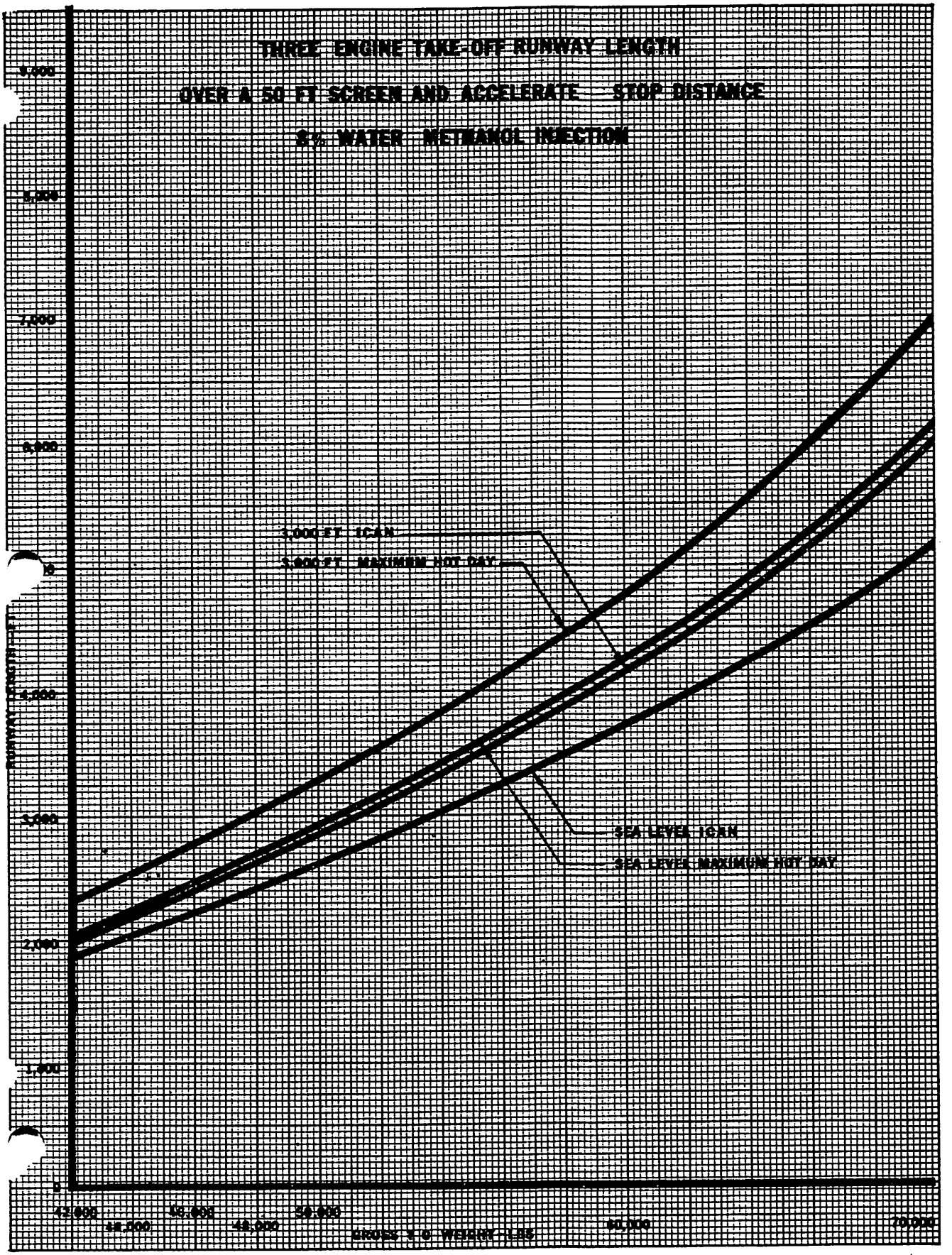



FOUR ENGINE TAKE-OFF OVER A 50 FT SCREEN
AND ACCELERATE — STOP DISTANCE
TAKE-OFF WEIGHT — LBS
50,000
55,000
60,000
65,000
70,000
NO WATER-METHANOL INJECTION
TEMPERATURE — °F
100
90
80
70
60
50
40
30
S.L.
1,000
2,000
3,000
4,000
ALTITUDE — FT
2,000
3,000
4,000
5,000
6,000
7,000
TAKE-OFF RUNWAY LENGTH — FT

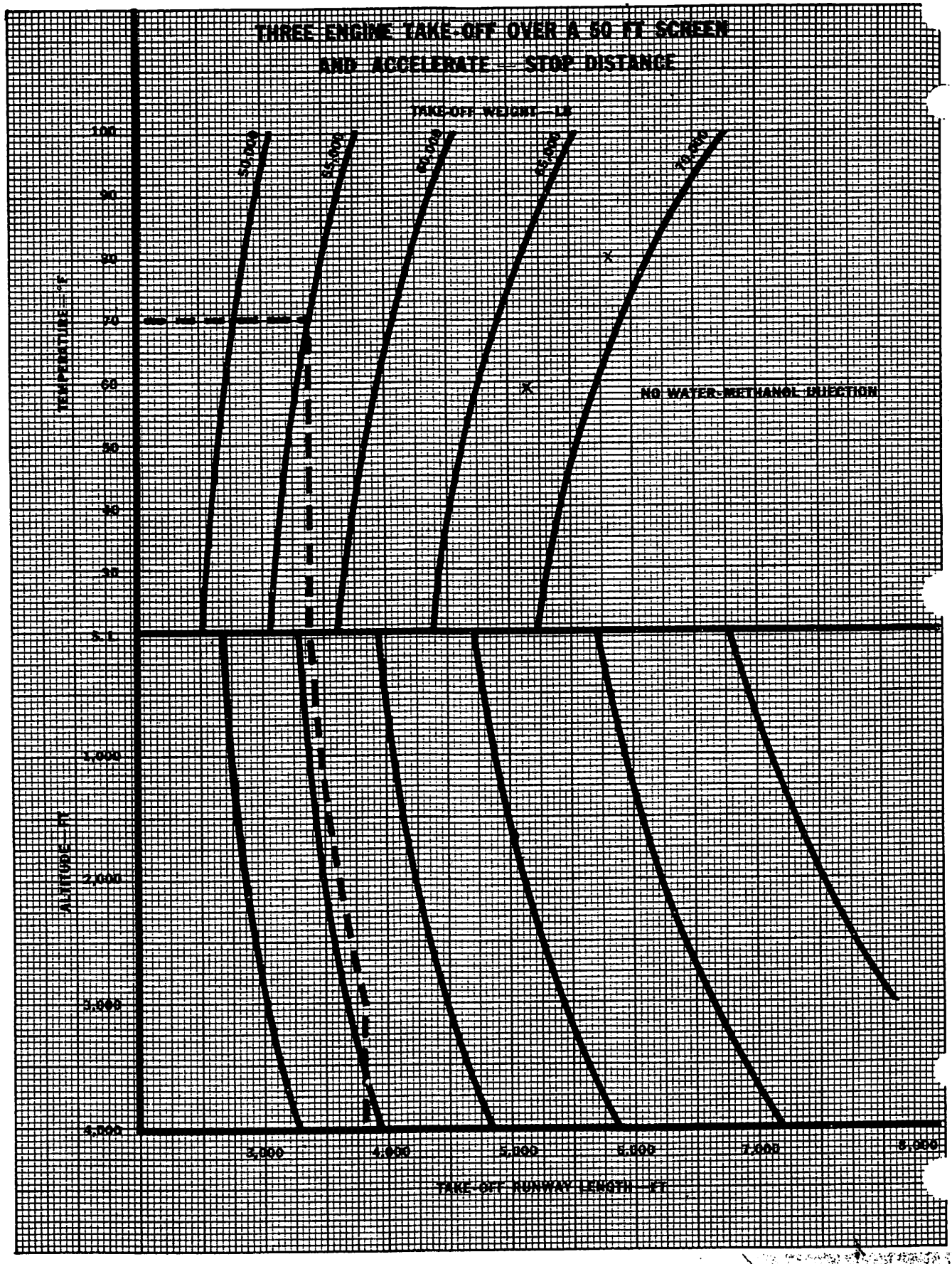

TAKE-OFF RUNWAY LENGTH OVER A 50FT SCREEN
(FOUR ENGINES)
NO WATER METHANOL INJECTION
8,000
7,000
6,000
5,000
4,000
3,000
2,000
1,000
0
RUNWAY LENGTH IN FEET
3000 FT { ICAN
MAX HOT DAY
SEA LEVEL { ICAN
MAX HOT DAY
50,000
60,000
70,000
GROSS TAKE-OFF WEIGHT LB

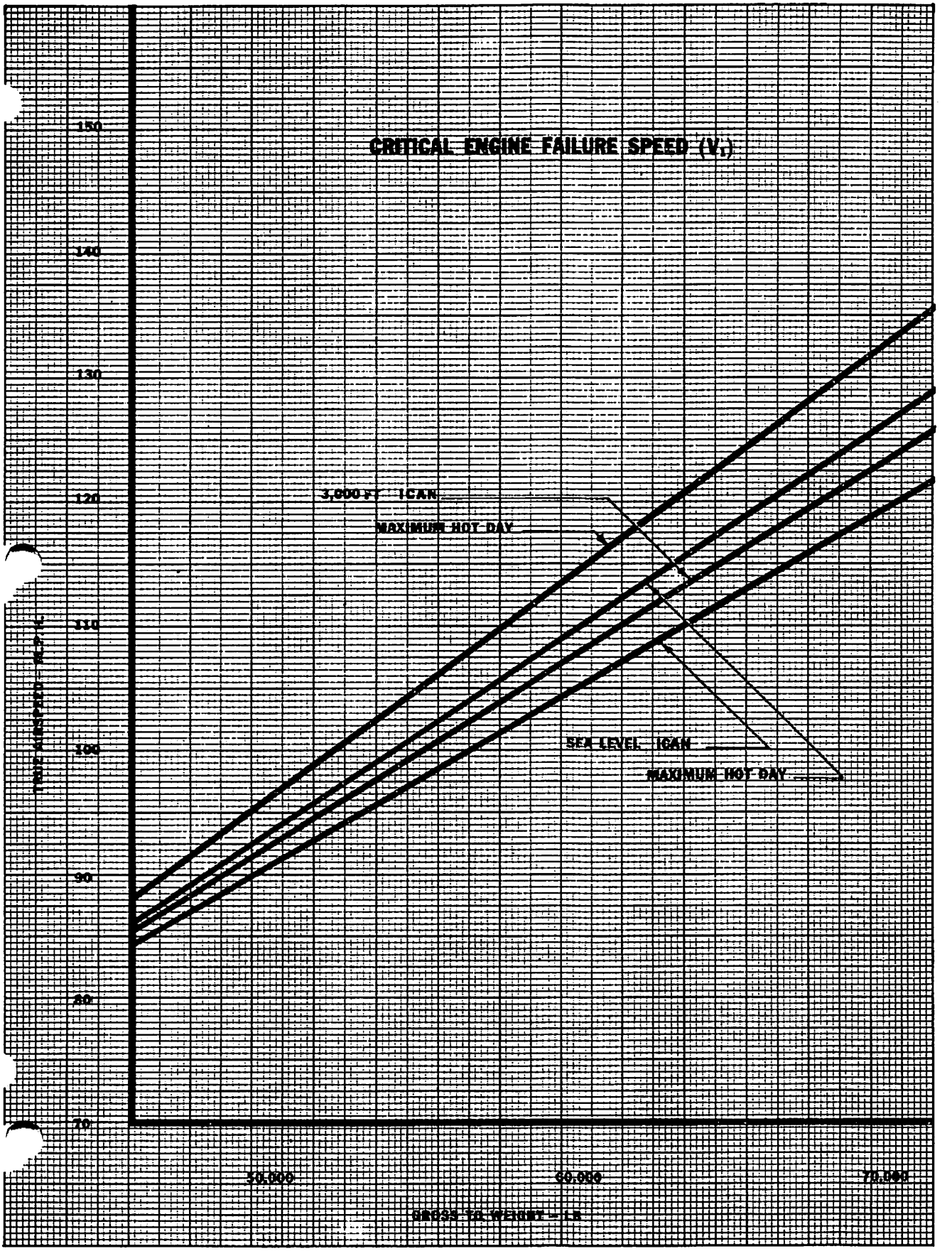

FOUR ENGINE TAKE-OFF RUNWAY LENGTH OVER A 50FT SCREEN
8% WATER METHANOL INJECTION
8,000
7,000
6,000
5,000
4,000
3,000
2,000
1,000
0
RUNWAY LENGTH - FT
3,000 FT ICAN
MAXIMUM HOT DAY
SEA LEVEL ICAN
MAXIMUM HOT DAY
50,000
60,000
70,000
GROSS TO WEIGHT - LB


THREE ENGINE TAKE-OFF RUNWAY LENGTH OVER A 50FT SCREEN
AND ACCELERATE STOP DISTANCE
NO WATER METHANOL INJECTION
8,000
7,000
6,000
5,000
4,000
3,000
2,000
1,000
0
RUNWAY LENGTH
3000 FT ICAN
3000 FT MAX HOT DAY
SEA LEVEL ICAN
SEA LEVEL MAX HOT DAY
42,000
44,000
46,000
48,000
50,000
60,000
GROSS TAKE-OFF WEIGHT LB

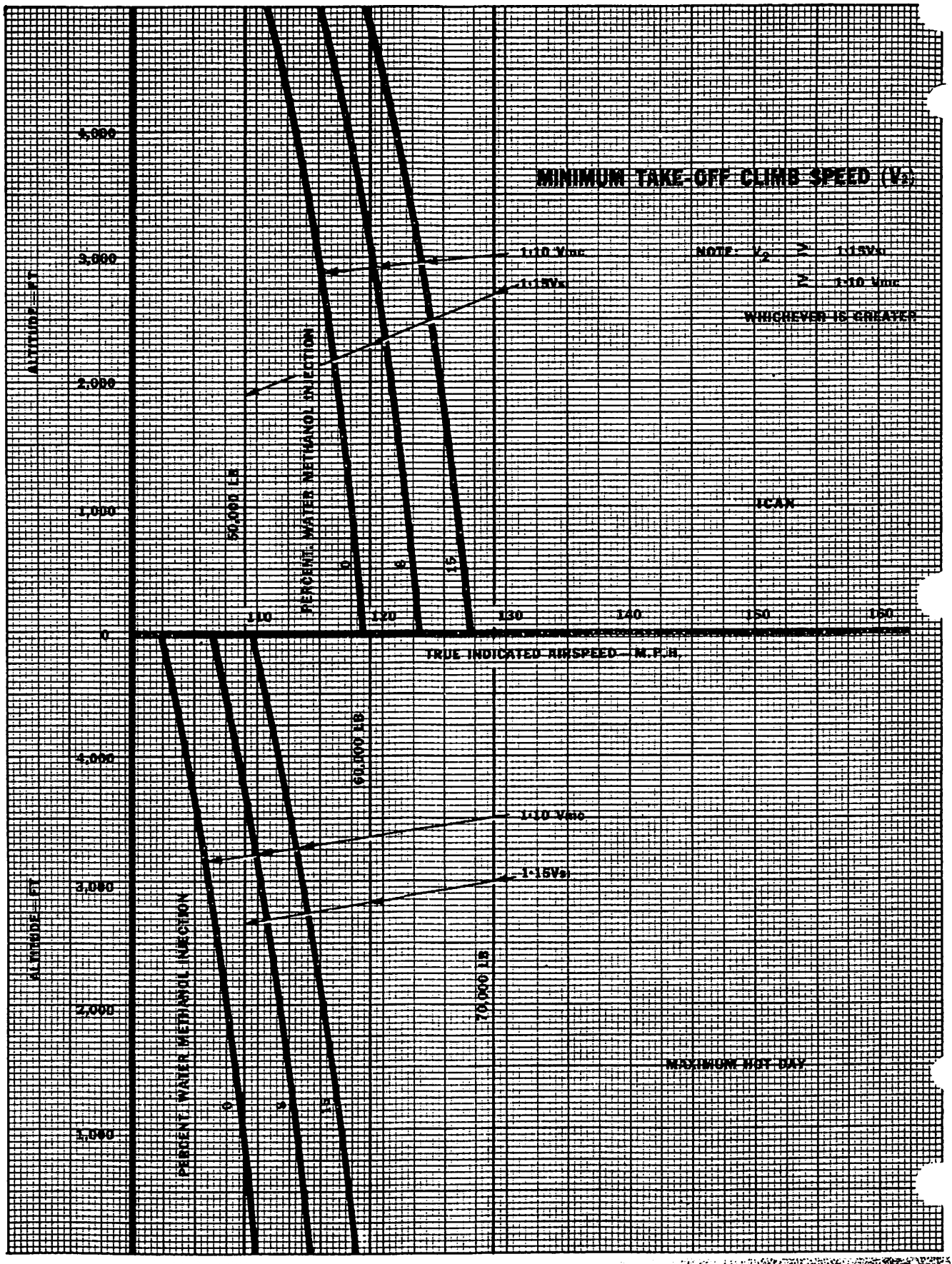

THREE ENGINE TAKE-OFF RUNWAY LENGTH
OVER A 50 FT SCREEN AND ACCELERATE STOP DISTANCE
8% WATER METHANOL INJECTION
5,000 FT 1CAN
5,000 FT MAXIMUM HOT DAY
SEA LEVEL 1CAN
SEA LEVEL MAXIMUM HOT DAY
RUNWAY LENGTH IN FEET
5,000
4,000
3,000
2,000
1,000
0
40,000
45,000
50,000
60,000
70,000
GROSS T O WEIGHT LBS

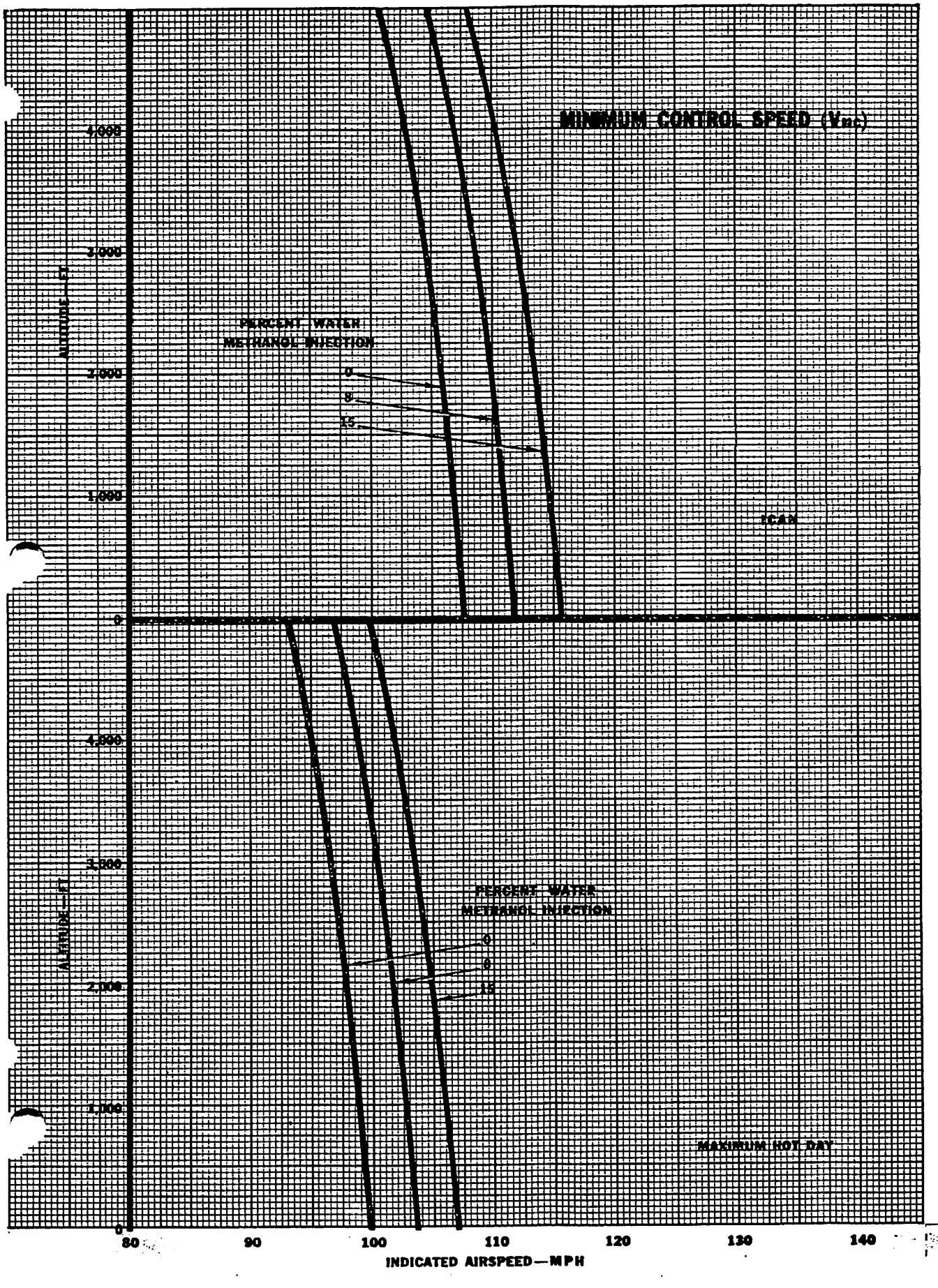
PERFORMANCE
CHART
FIG. 62



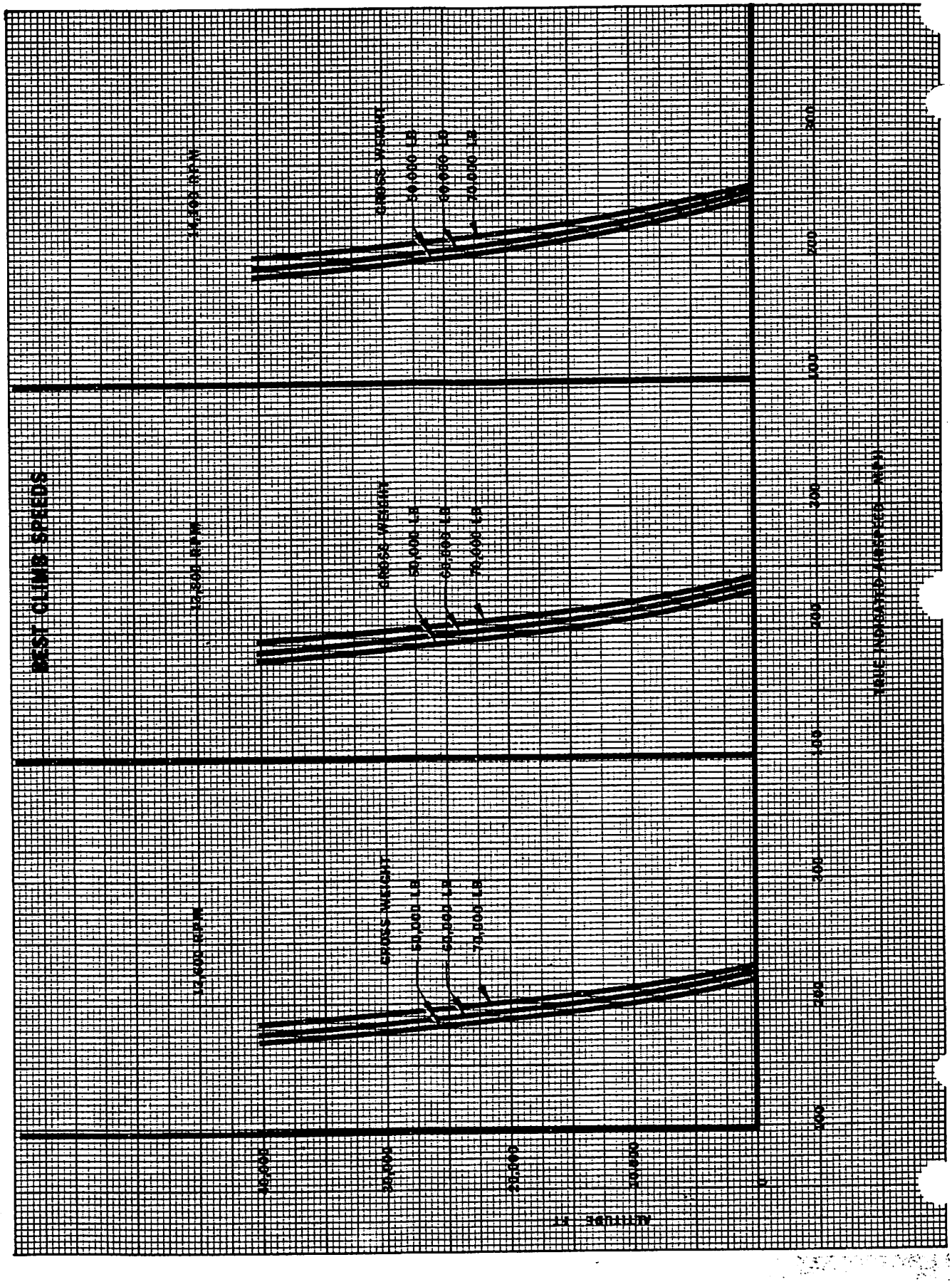

CRITICAL ENGINE FAILURE SPEED (V_1)
3,000 FT ICAN
MAXIMUM HOT DAY
SEA LEVEL ICAN
MAXIMUM HOT DAY
150
140
130
120
110
100
90
80
70
30,000
50,000
70,000
GROSS TK. WEIGHT — LB

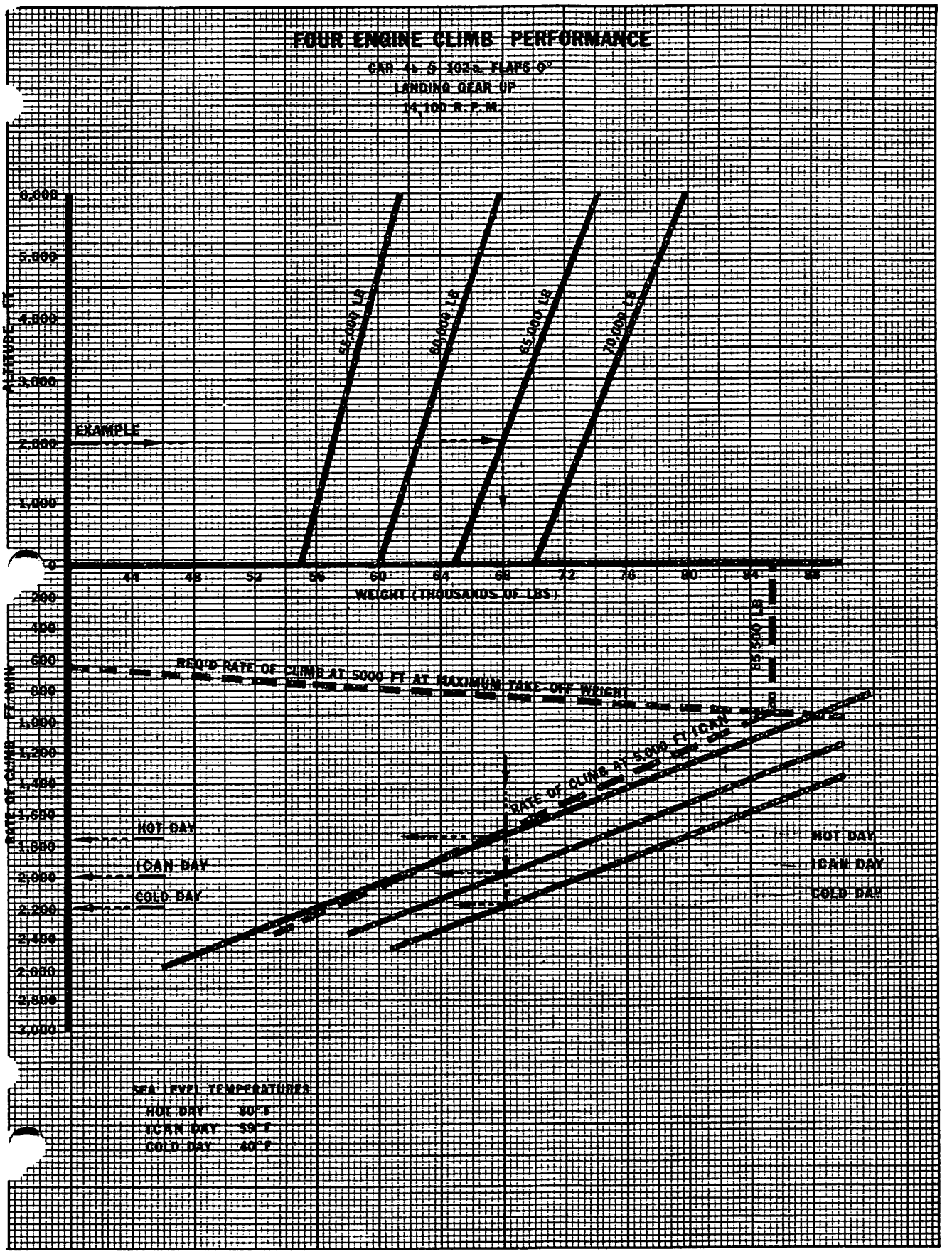
PERFORMANCE
CHART
FIG. 64

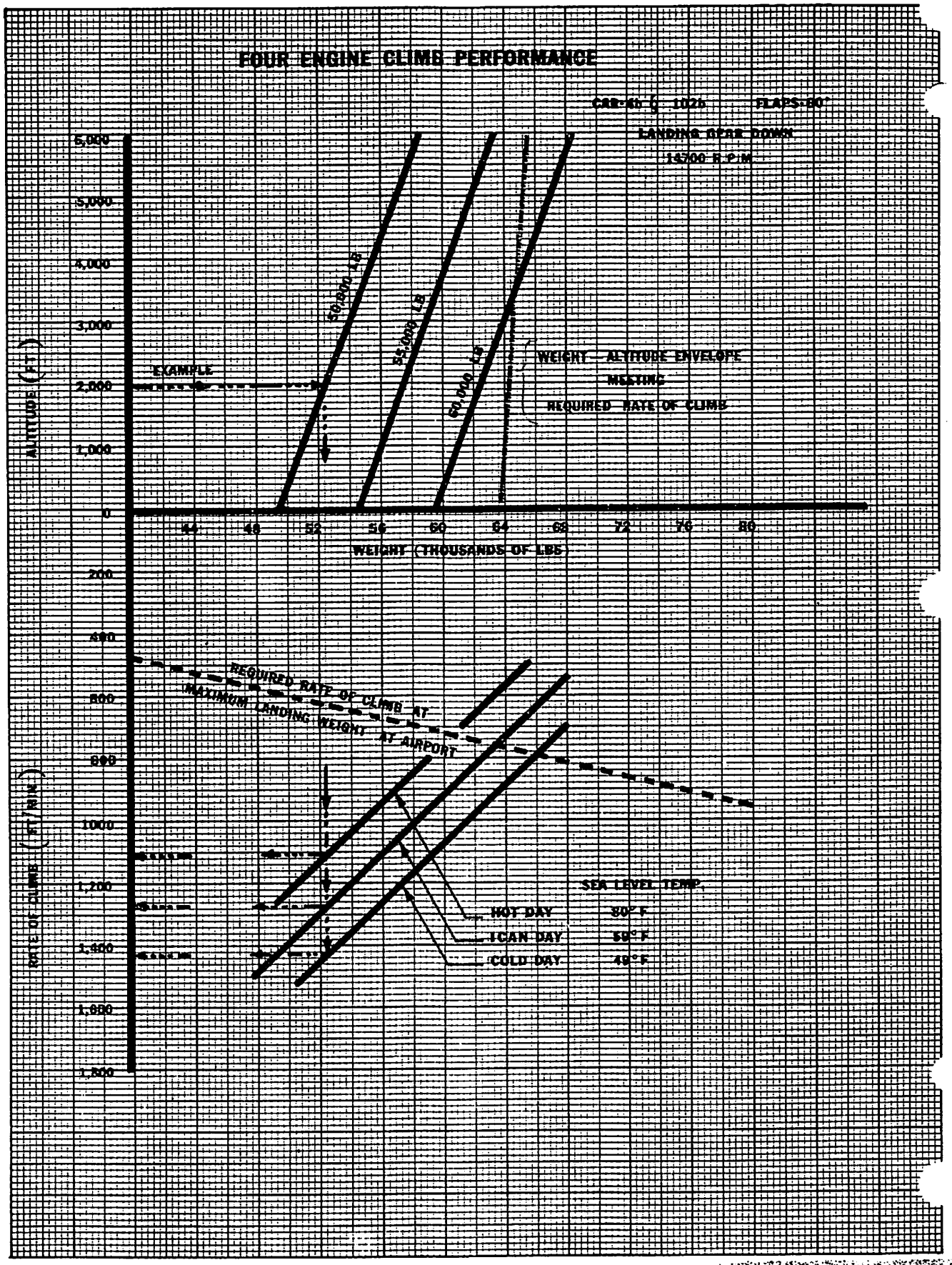


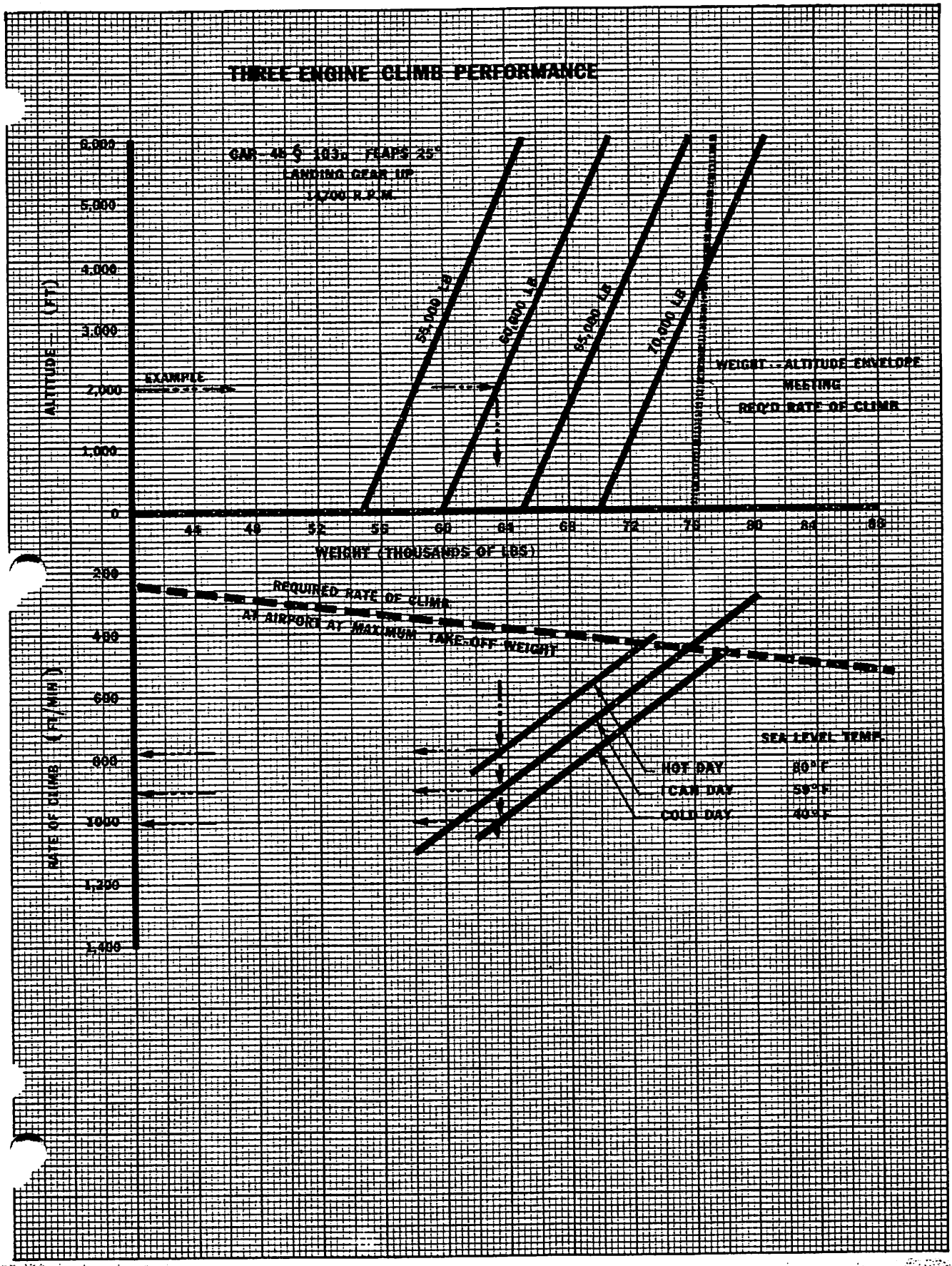

MINIMUM CONTROL SPEED (Vmc)
PERCENT WATER METHANOL INJECTION
0
8
15
ICAN
PERCENT WATER METHANOL INJECTION
0
8
15
MAXIMUM HOT DAY
ALTITUDE—FT
4,000
3,000
2,000
1,000
0
4,000
3,000
2,000
1,000
0
80
90
100
110
120
130
140
INDICATED AIRSPEED—MPH

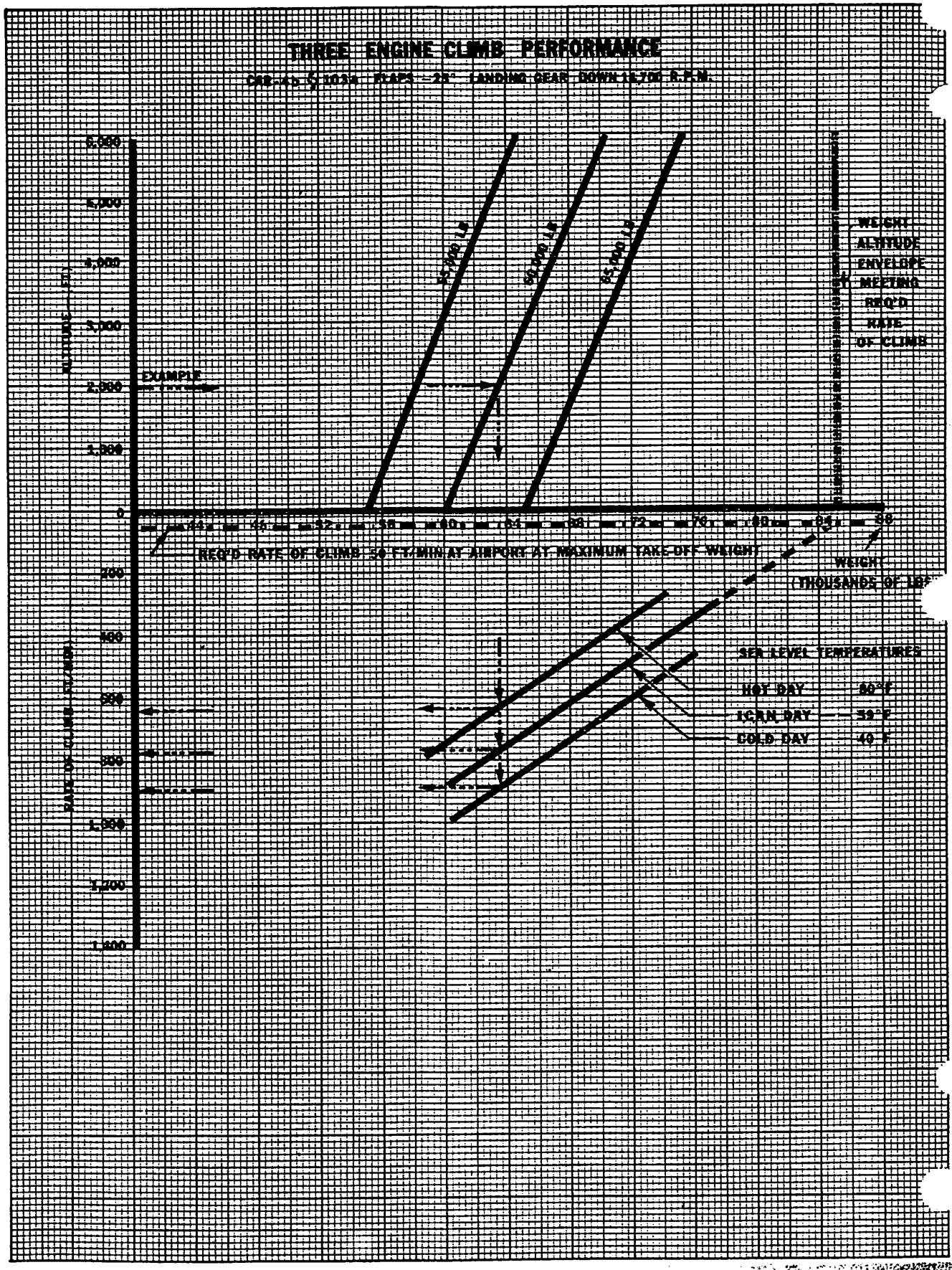
PERFORMANCE
CHART
FIG. 66

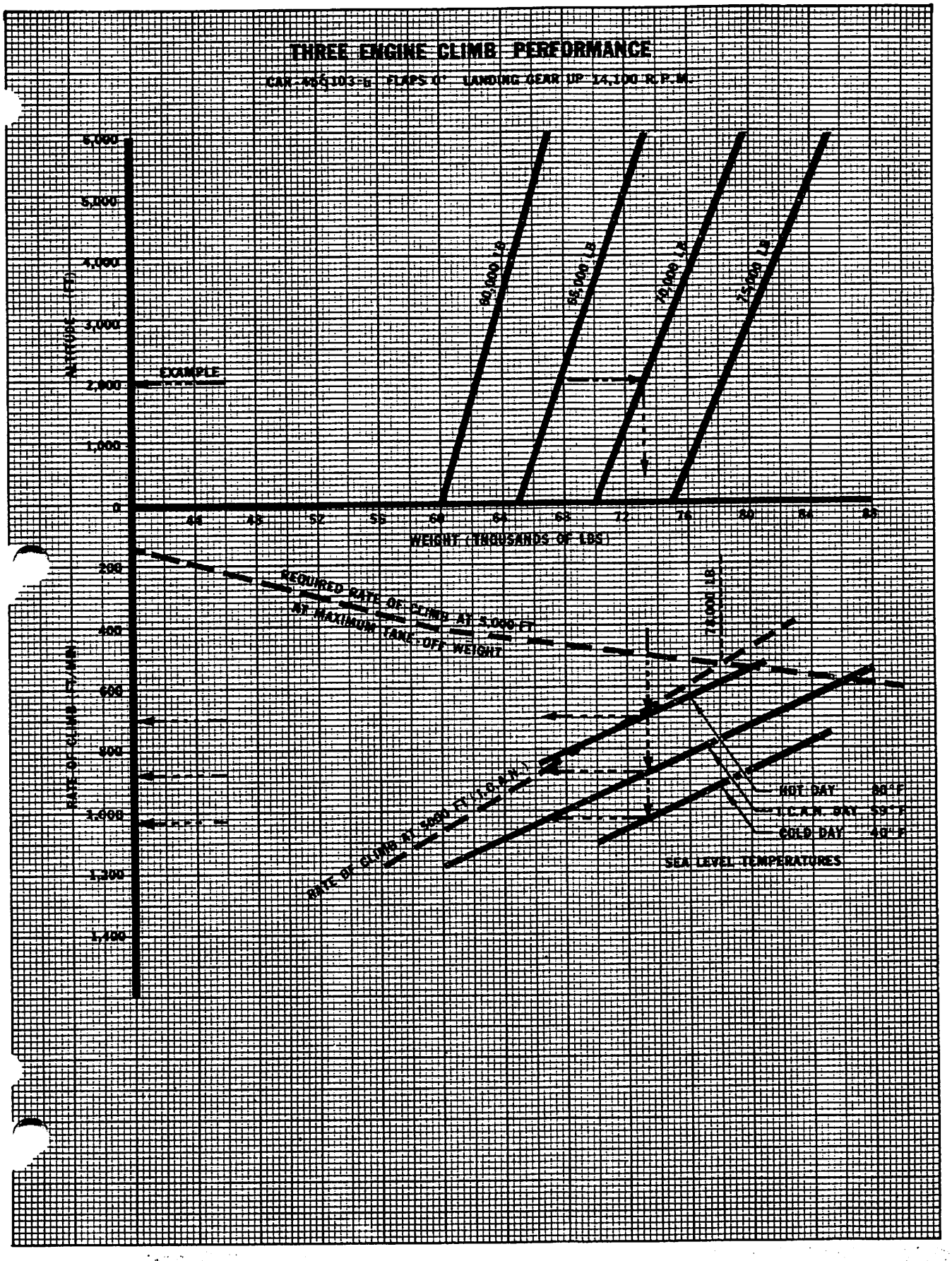


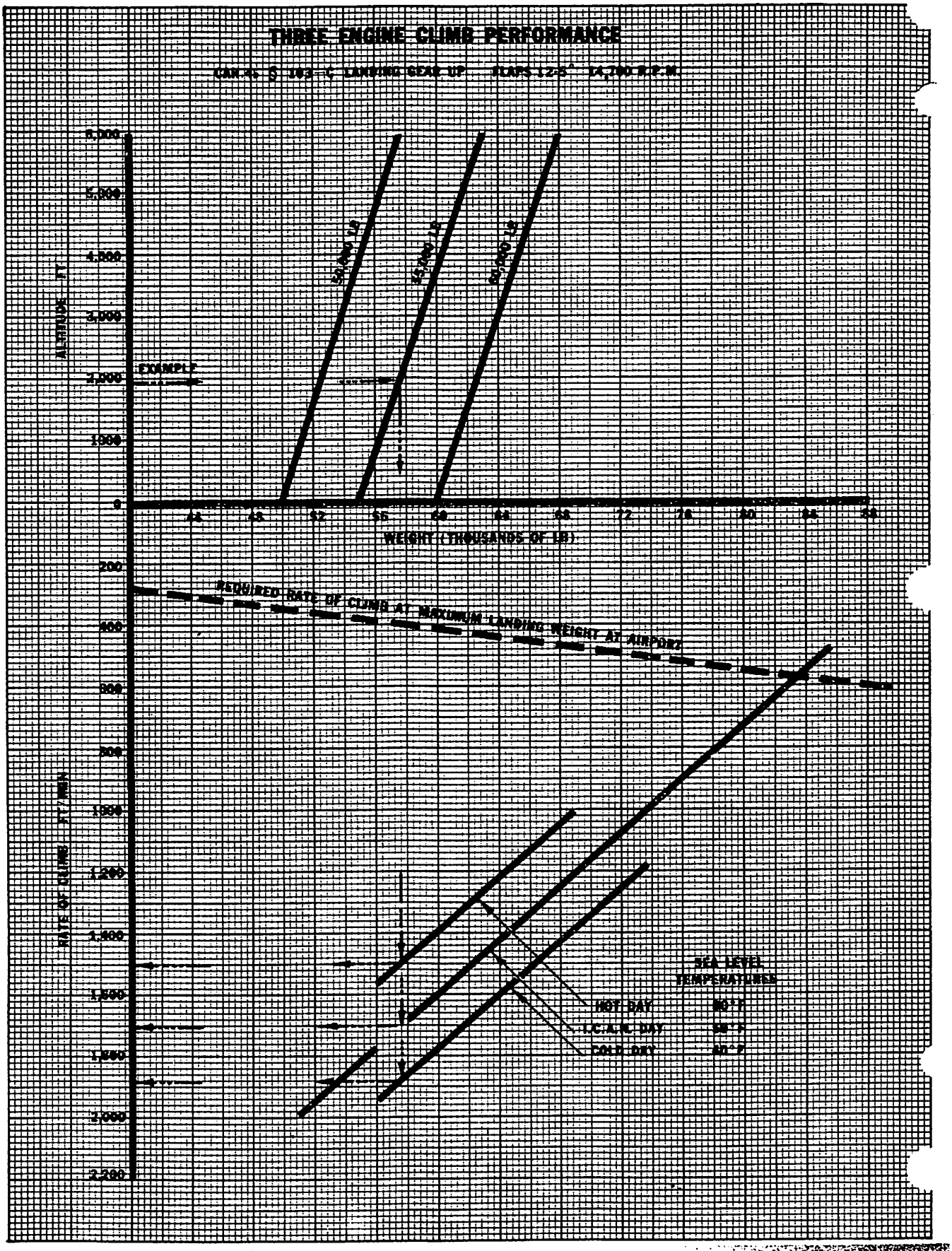

FOUR ENGINE CLIMB PERFORMANCE
CAB 45 S 102c FLAPS 0°
LANDING GEAR UP
14,100 R.P.M.
ALTITUDE FT
RATE OF CLIMB FT/MIN
WEIGHT (THOUSANDS OF LBS.)
35,000 LB
50,000 LB
65,000 LB
70,000 LB
EXAMPLE
REQ'D RATE OF CLIMB AT 5000 FT AT MAXIMUM TAKE-OFF WEIGHT
RATE OF CLIMB AT 5000 FT WITH ONE ENGINE INOPERATIVE
HOT DAY
ICAN DAY
COLD DAY
SEA LEVEL TEMPERATURES
HOT DAY 90° F
ICAN DAY 59° F
COLD DAY 40° F

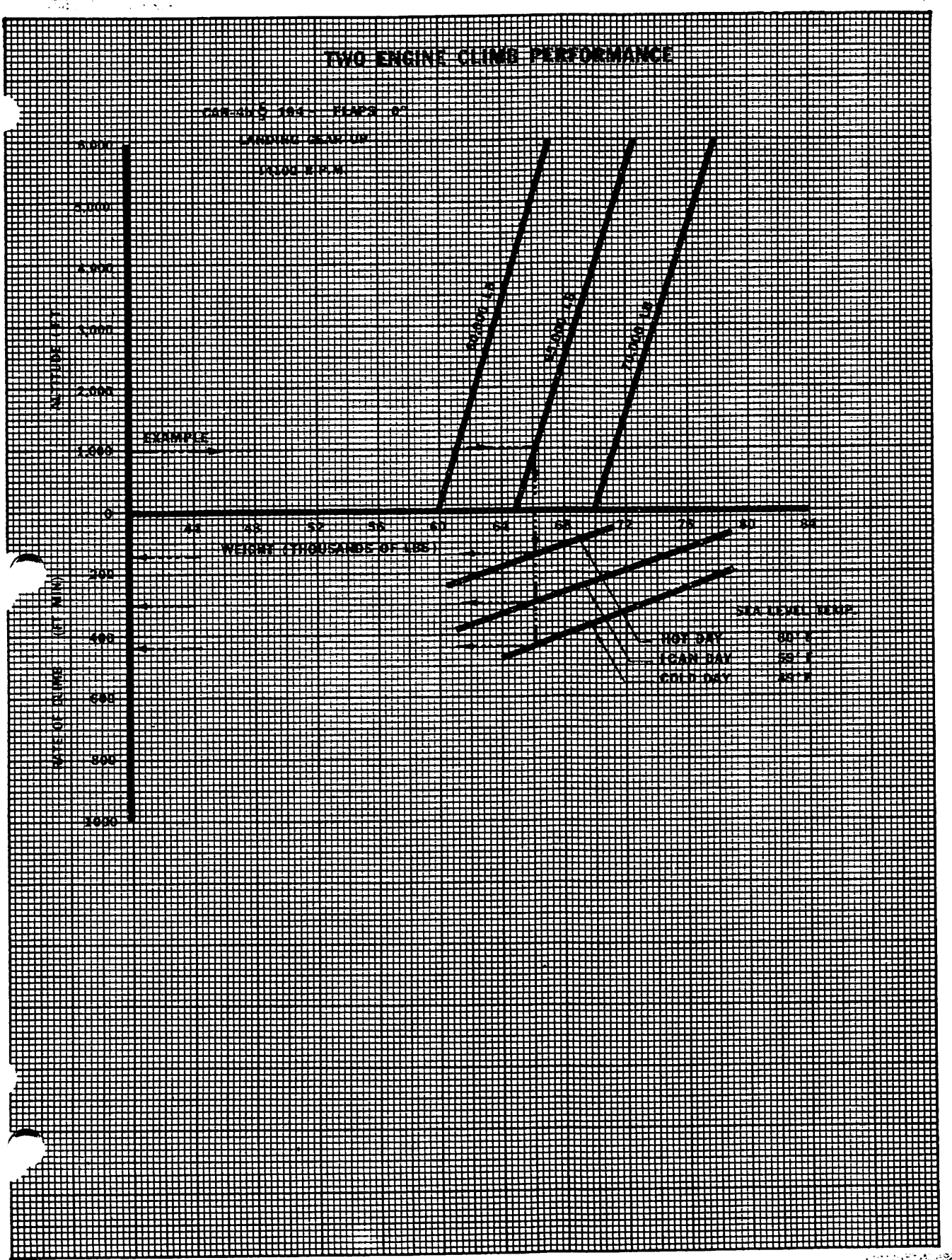

FOUR ENGINE CLIMB PERFORMANCE
CAR-4b & 1026
FLAPS-40°
LANDING GEAR DOWN
14700 R.P.M.
ALTITUDE (FT.)
5,000
4,000
3,000
2,000
1,000
0
60,000 LB
56,000 LB
52,000 LB
EXAMPLE
WEIGHT - ALTITUDE ENVELOPE
MEETING
REQUIRED RATE OF CLIMB
44
48
52
56
60
64
68
72
76
80
WEIGHT (THOUSANDS OF LBS)
RATE OF CLIMB (FT./MIN.)
200
400
600
800
1000
1,200
1,400
1,600
1,800
REQUIRED RATE OF CLIMB AT
MAXIMUM LANDING WEIGHT AT AIRPORT
SEA LEVEL TEMP.
HOT DAY
80° F
ICAN DAY
59° F
COLD DAY
49° F


THREE ENGINE CLIMB PERFORMANCE
CAR 4A § 103a FLAPS 25°
LANDING GEAR UP
14,700 R.P.M.
ALTITUDE (FT.)
6,000
5,000
4,000
3,000
2,000
1,000
0
EXAMPLE
60,000 LB
64,000 LB
68,000 LB
70,000 LB
WEIGHT--ALTITUDE ENVELOPE MEETING REQ'D RATE OF CLIMB
44
48
52
56
60
64
68
72
76
80
84
88
WEIGHT (THOUSANDS OF LBS)
REQUIRED RATE OF CLIMB AT AIRPORT AT MAXIMUM TAKE-OFF WEIGHT
RATE OF CLIMB (FT./MIN.)
200
400
600
800
1000
1,200
1,400
SEA LEVEL TEMP.
HOT DAY 80°F
WARM DAY 59°F
COLD DAY 40°F


THREE ENGINE CLIMB PERFORMANCE
CAR. Alt. S. 1013.2 FLAPS -25° LANDING GEAR DOWN 16,700 R.P.M.
6,000
5,000
4,000
3,000
2,000
1,000
0
EXAMPLE
200
400
600
800
1,000
1,200
1,400
RATE OF CLIMB FEET PER MINUTE
PRESSURE ALTITUDE IN FEET
WEIGHT ALTITUDE ENVELOPE BY MEETING REQ'D RATE OF CLIMB
REQ'D RATE OF CLIMB 50 FT/MIN AT AIRPORT AT MAXIMUM TAKE-OFF WEIGHT
WEIGHT (THOUSANDS OF LBS)
SEA LEVEL TEMPERATURES
HOT DAY 90°F
ICAN DAY 59°F
COLD DAY 40°F

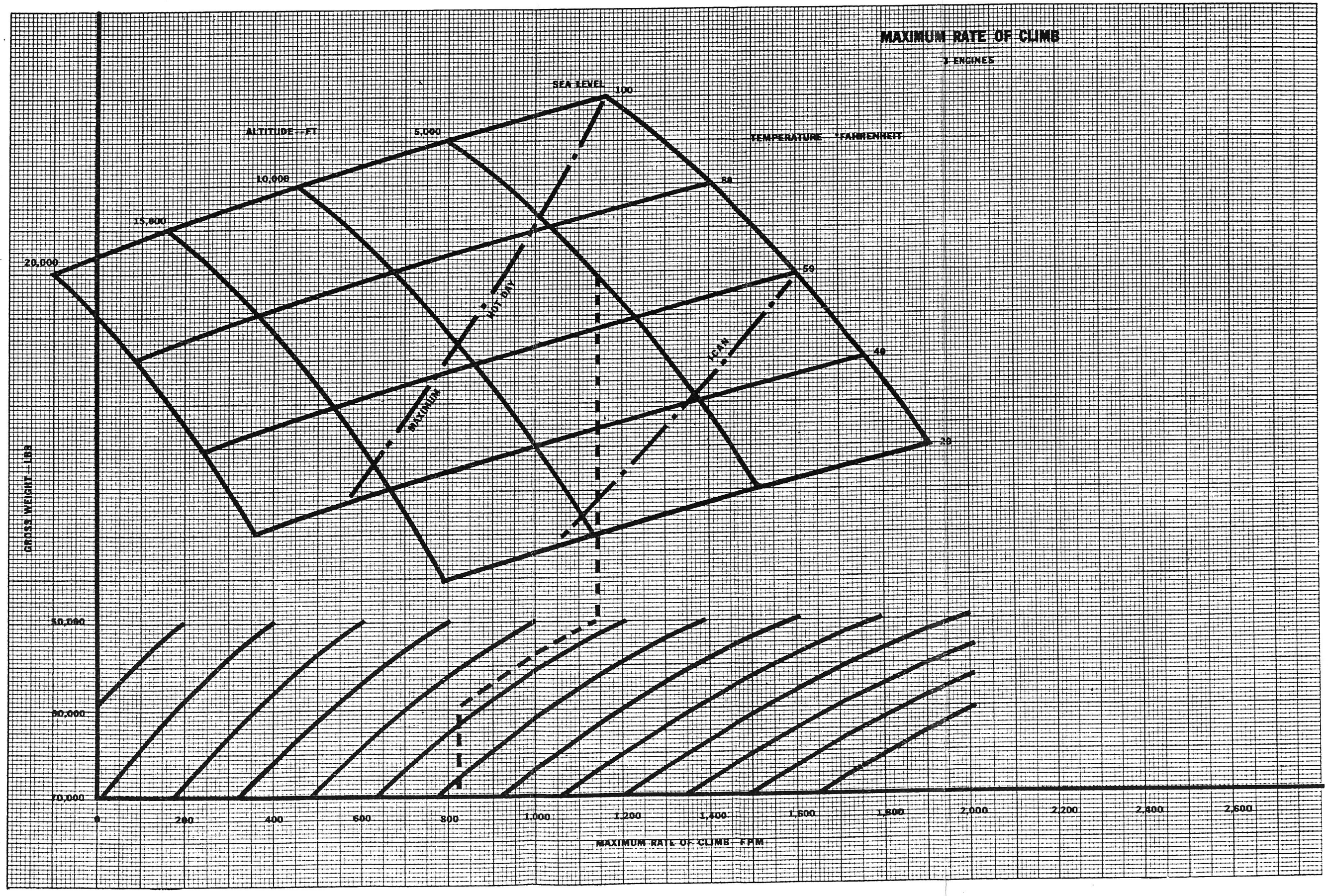

THREE ENGINE CLIMB PERFORMANCE
CAA 465-103-5 FLAPS 0° LANDING GEAR UP 14,100 R.P.M.
ALTITUDE (FT)
5,000
4,000
3,000
2,000
1,000
0
EXAMPLE
90,000 LB
85,000 LB
80,000 LB
75,000 LB
WEIGHT (THOUSANDS OF LBS)
REQUIRED RATE OF CLIMB AT 5,000 FT.
AT MAXIMUM TAKE-OFF WEIGHT
RATE OF CLIMB (FT/MIN)
200
400
600
800
1,000
1,200
1,400
RATE OF CLIMB AT 1000 FT (T.O.P.W.)
HOT DAY 86°F
I.C.A.N. DAY 59°F
COLD DAY 40°F
SEA LEVEL TEMPERATURES

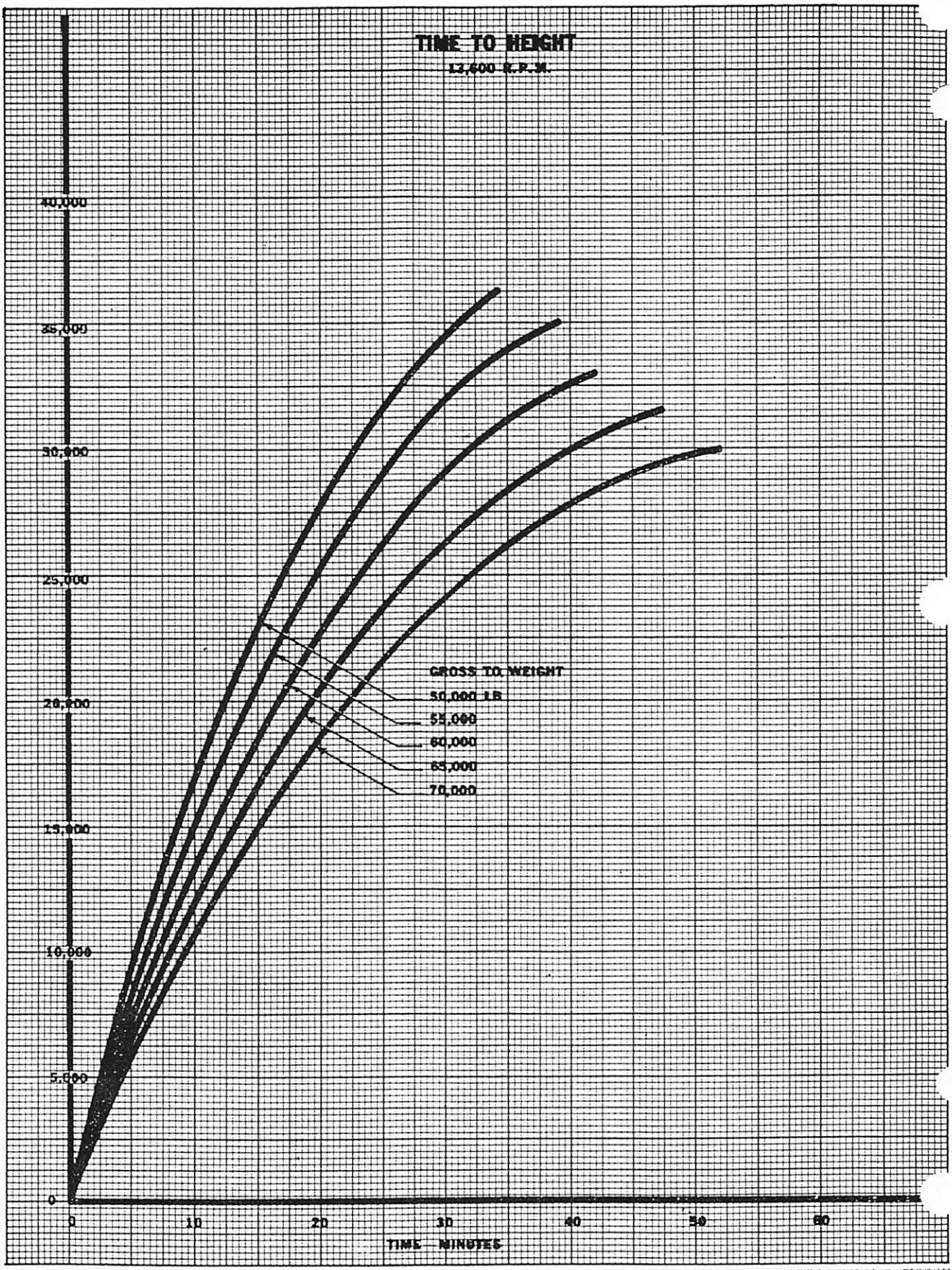

THREE ENGINE CLIMB PERFORMANCE
CAR 4b § 103 - C LANDING GEAR UP FLAPS 12-5° 14,200 R.P.M.
ALTITUDE FT.
5,000
4,000
3,000
2,000
1,000
0
EXAMPLE
50,000 LB.
47,000 LB.
45,000 LB.
42
45
52
55
60
64
68
72
76
80
84
88
WEIGHT (THOUSANDS OF LB)
REQUIRED RATE OF CLIMB AT MAXIMUM LANDING WEIGHT AT AIRPORT
RATE OF CLIMB FT. PER MIN.
200
400
600
800
1000
1200
1400
1600
1800
2000
2200
SEA LEVEL TEMPERATURES
HOT DAY 90° F
I.C.A.N. DAY 59° F
COLD DAY 40° F

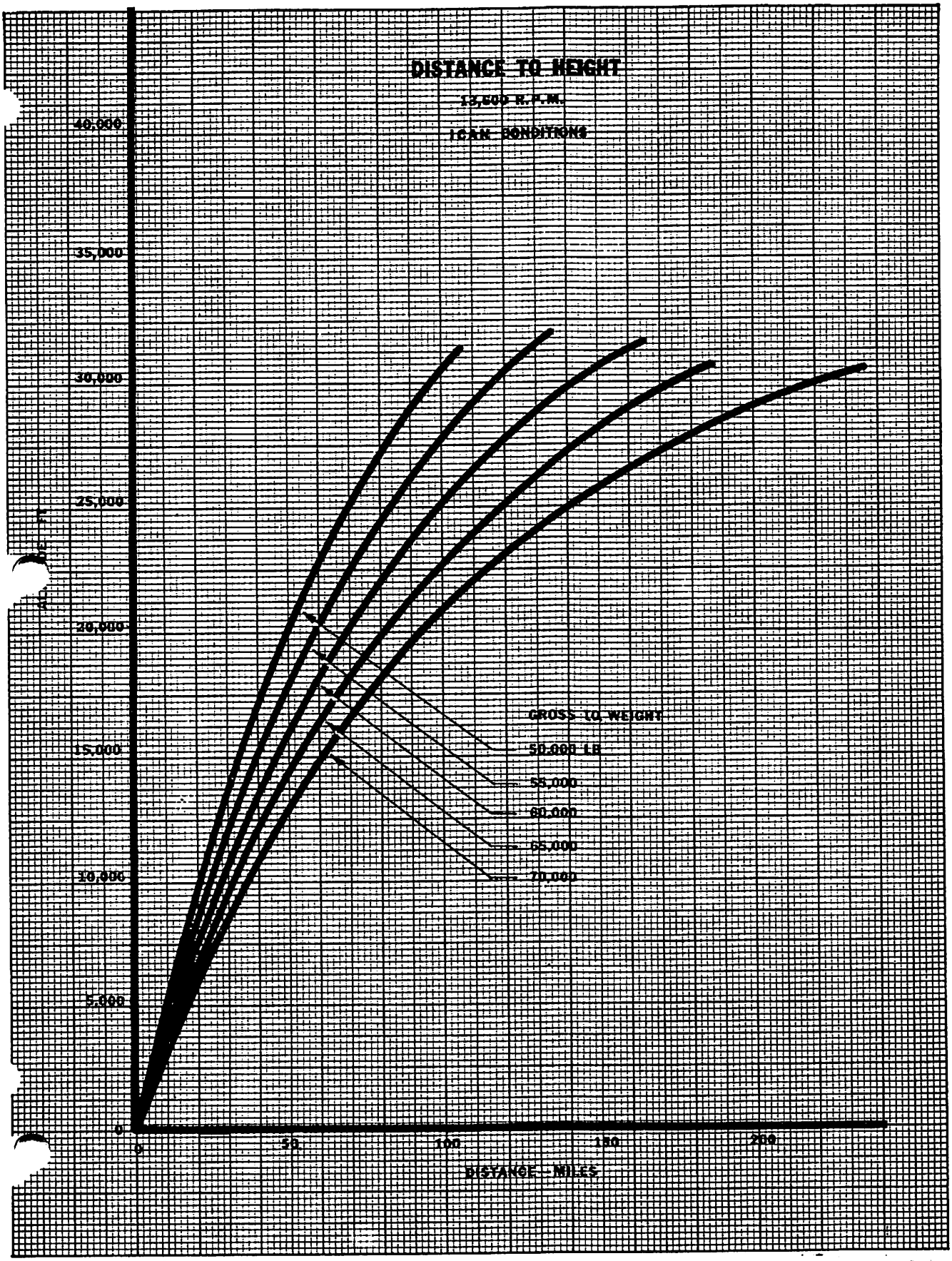

TWO ENGINE CLIMB PERFORMANCE
CAN-AIR S 104 - FLAPS 0°
LANDING GEAR UP
14500 R.P.M.
RATE OF CLIMB (FEET PER MINUTE)
5,000
4,000
3,000
2,000
1,000
0
200
400
600
800
1000
EXAMPLE
46
48
50
52
54
56
58
60
62
64
66
68
70
72
74
76
78
80
82
84
WEIGHT (THOUSANDS OF LBS.)
SEA LEVEL
SEA LEVEL TEMP.
HOT DAY
CAN DAY
COLD DAY

317-G1

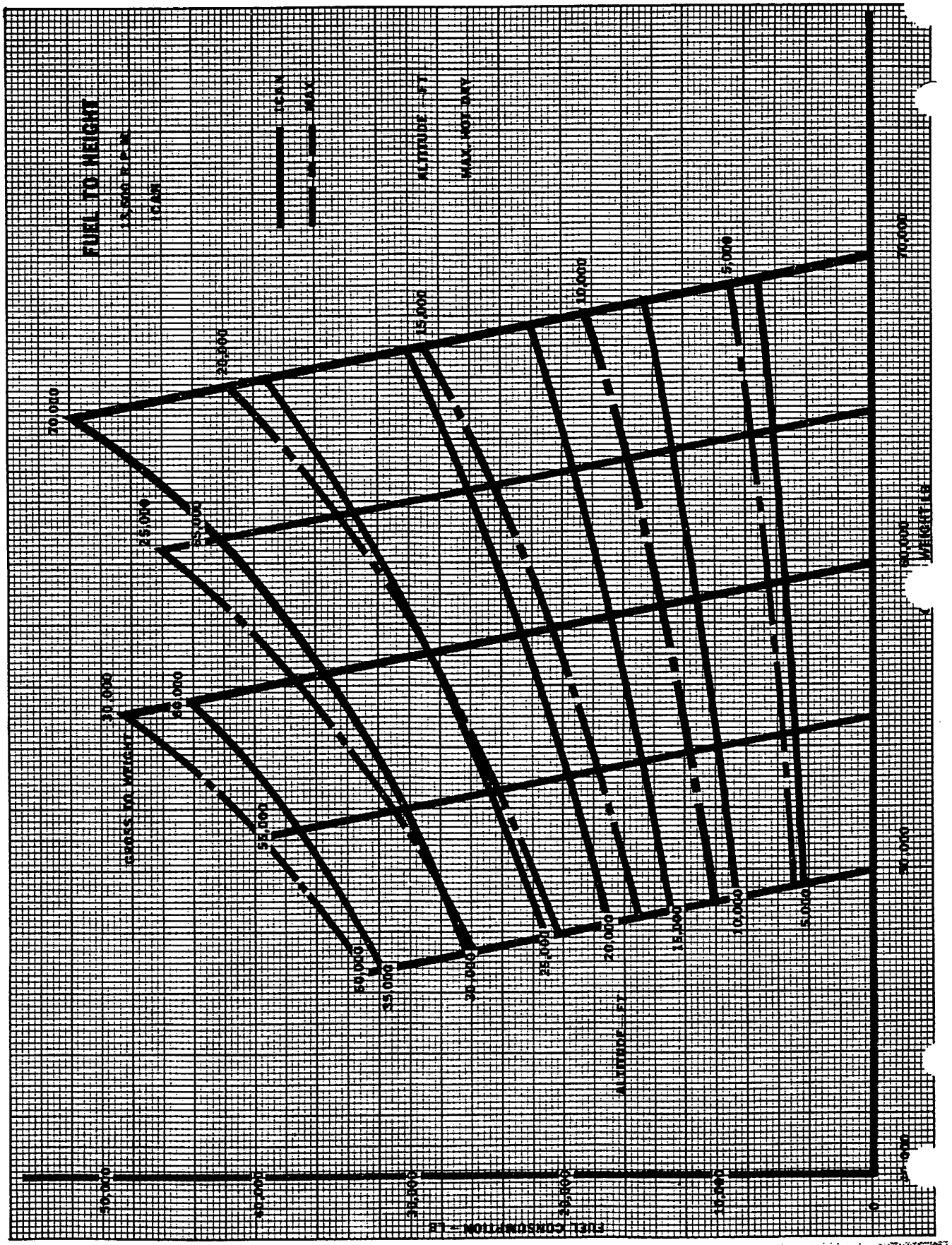
PERFORMANCE
CHART
FIG. 74

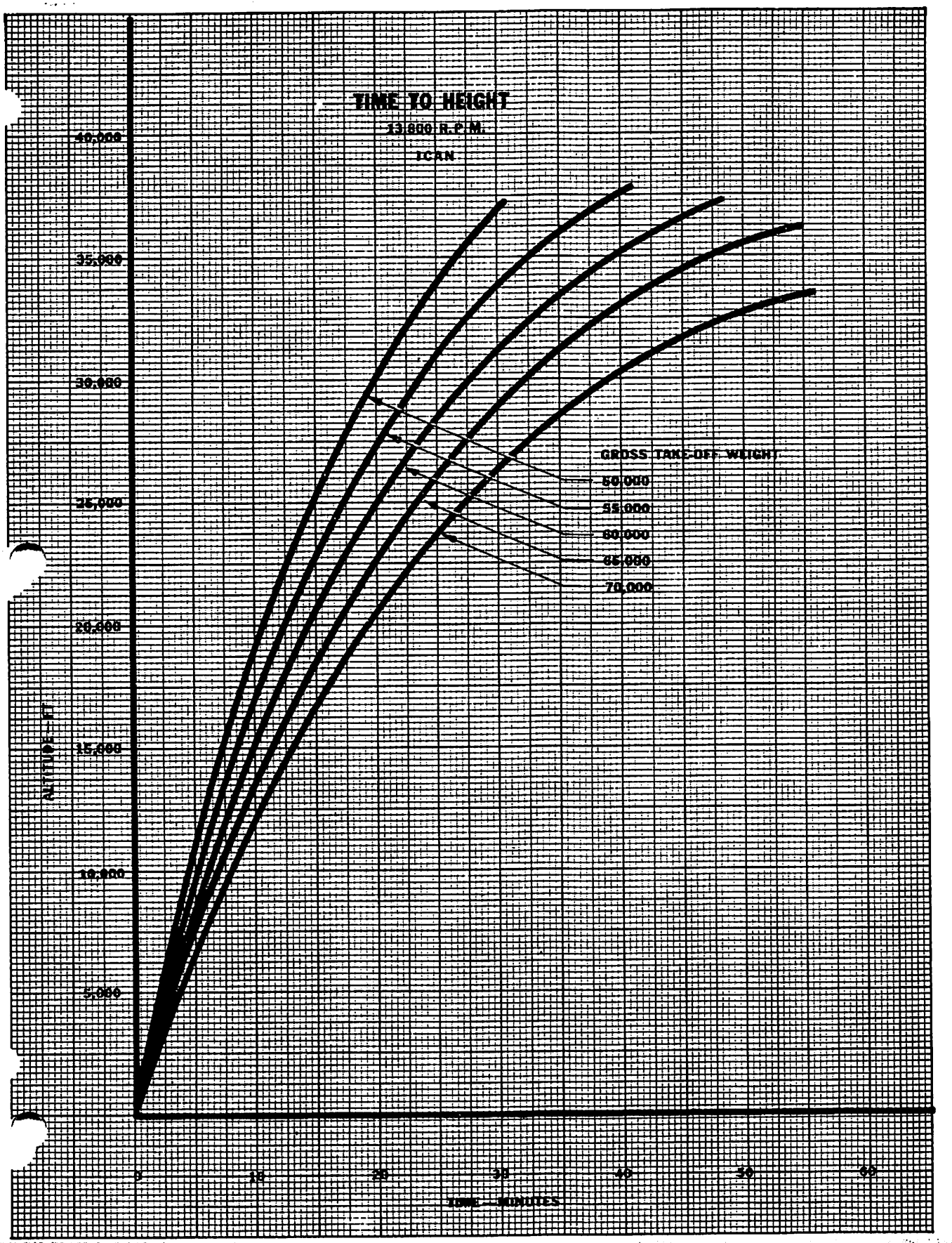



TIME TO HEIGHT
13,600 R.P.M.
40,000
35,000
30,000
25,000
20,000
15,000
10,000
5,000
0
GROSS TO WEIGHT
50,000 LB
55,000
60,000
65,000
70,000
0
10
20
30
40
50
60
TIME - MINUTES

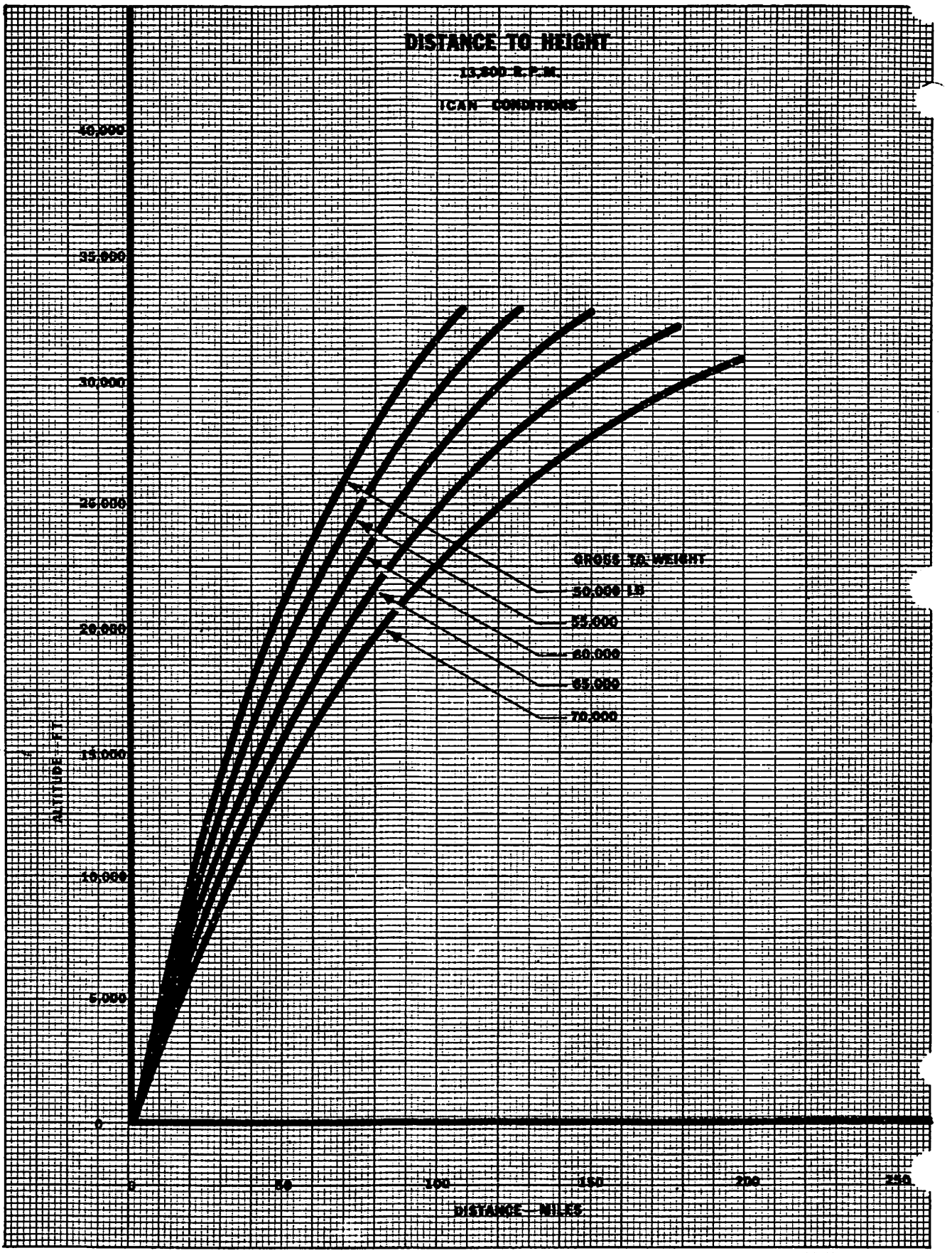

DISTANCE TO HEIGHT
13,500 R.P.M.
ICAN CONDITIONS
40,000
35,000
30,000
25,000
20,000
15,000
10,000
5,000
0
ALTITUDE - FEET
0
50
100
150
200
DISTANCE - MILES
GROSS TO WEIGHT
50,000 LB
55,000
60,000
65,000
70,000

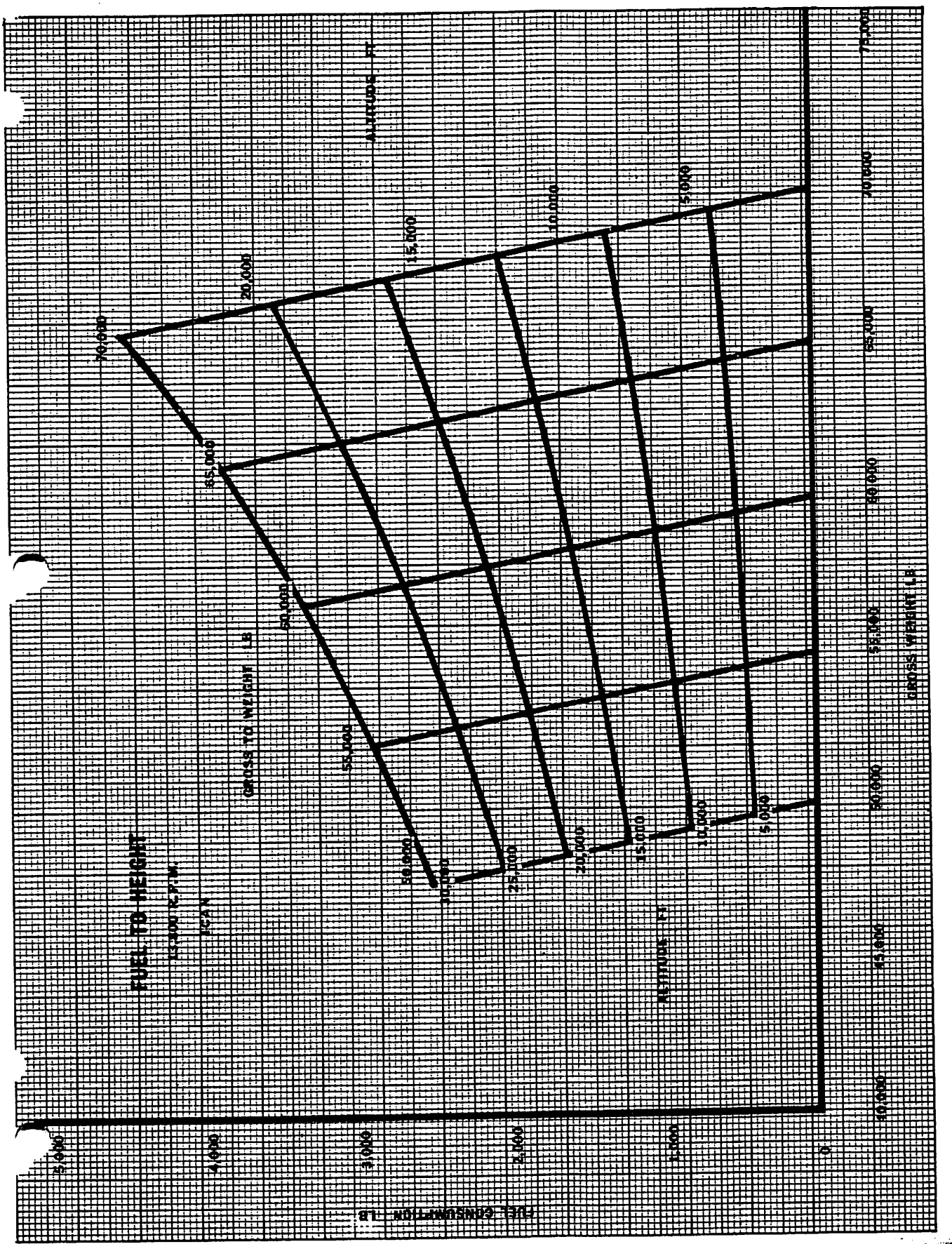
PERFORMANCE
CHART
FIG. 77



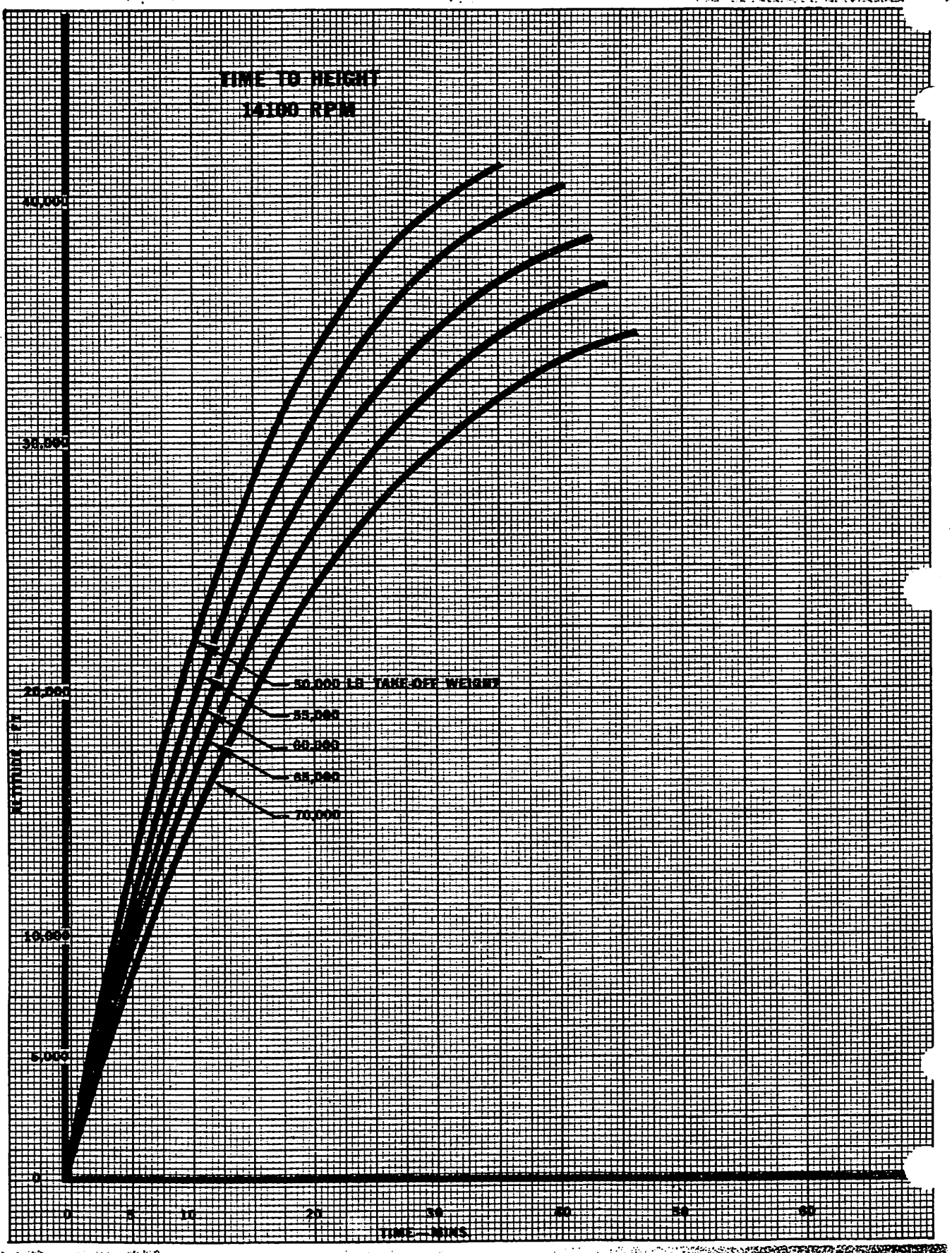

TIME TO HEIGHT
13,800 R.P.M.
ICAN
40,000
35,000
30,000
25,000
20,000
15,000
10,000
5,000
ALTITUDE FT.
GROSS TAKE-OFF WEIGHT
50,000
55,000
60,000
65,000
70,000
0
10
20
30
40
50
60
TIME — MINUTES

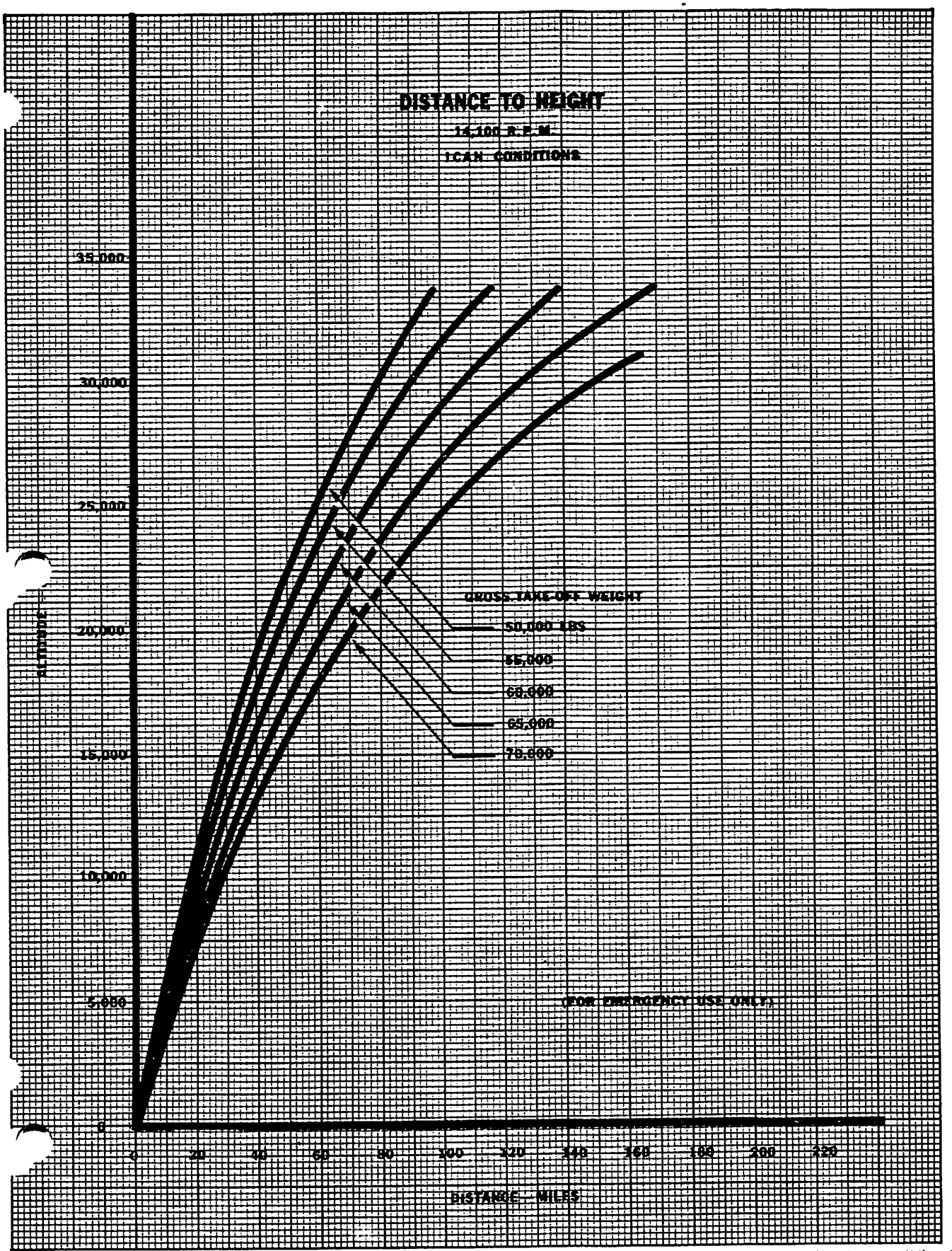
PERFORMANCE
CHART
FIG. 79



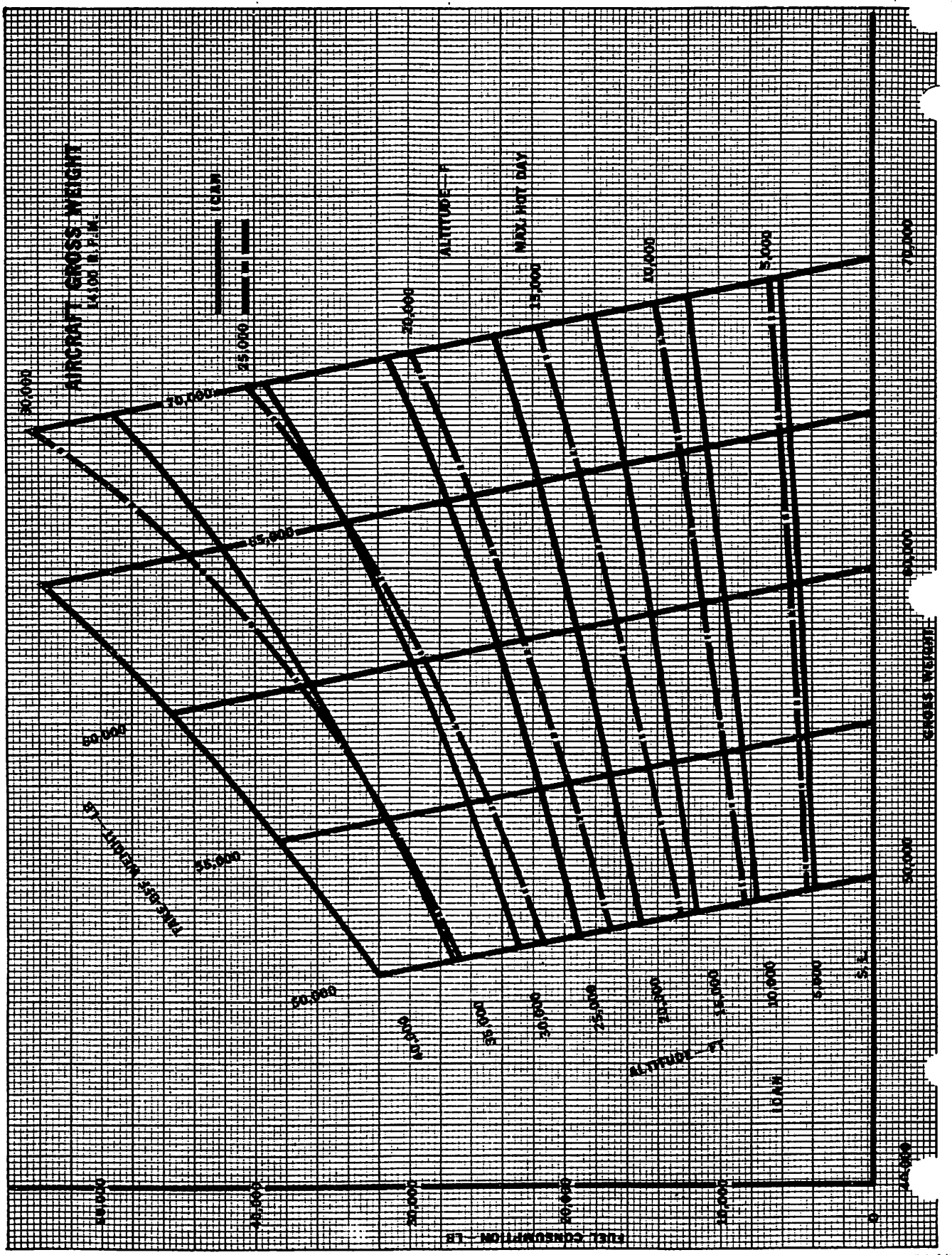

FUEL TO HEIGHT
C-102 TURBOFAN
ICAN
GROSS TO WEIGHT LB
70,000
65,000
60,000
55,000
50,000
45,000
ALTITUDE FT
20,000
15,000
10,000
5,000
75,000
70,000
65,000
60,000
55,000
50,000
45,000
40,000
GROSS WEIGHT LB
30,000
25,000
20,000
15,000
10,000
5,000
ALTITUDE FT
5,000
4,000
3,000
2,000
1,000
0
FUEL CONSUMPTION LB

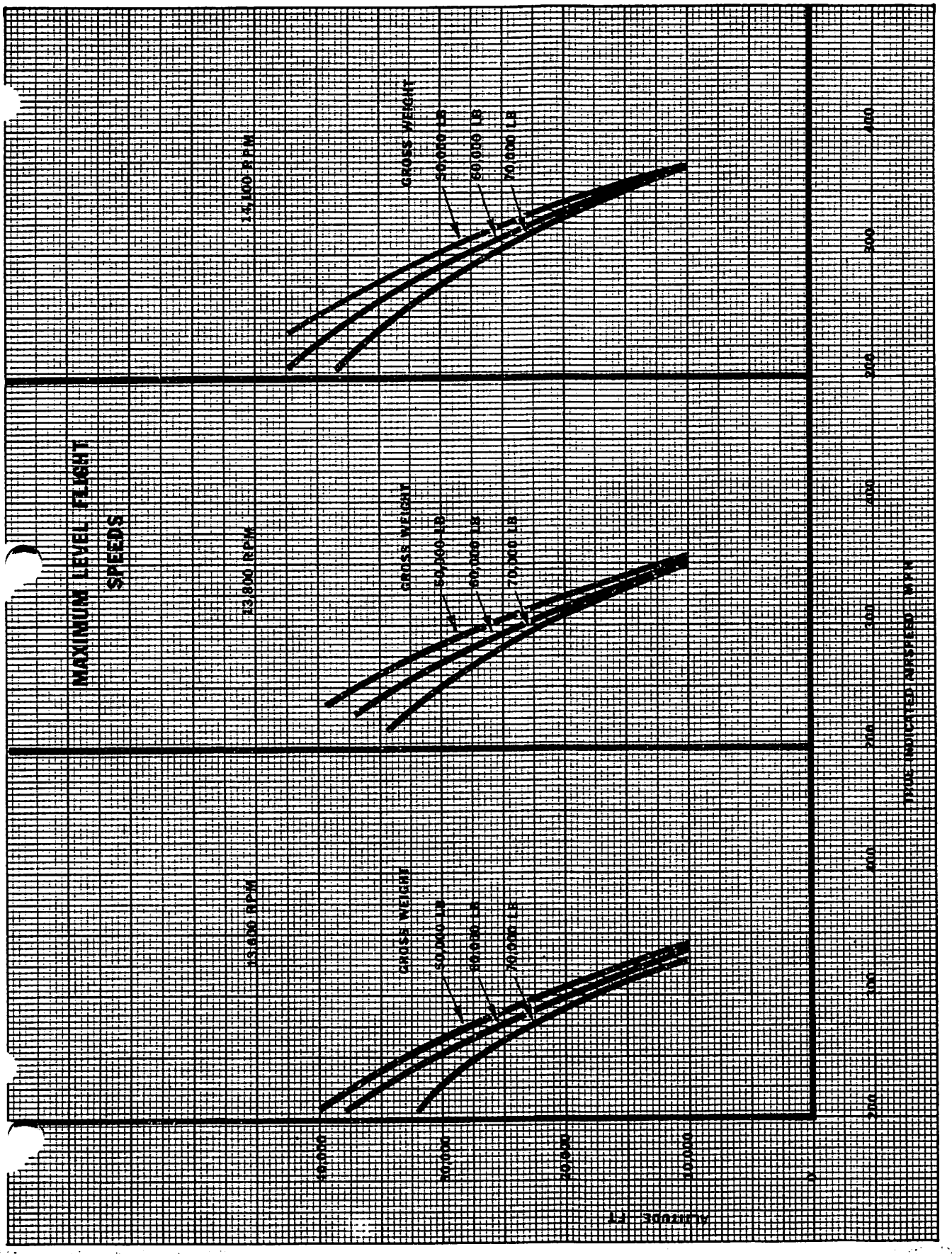
PERFORMANCE
CHART
FIG. 81



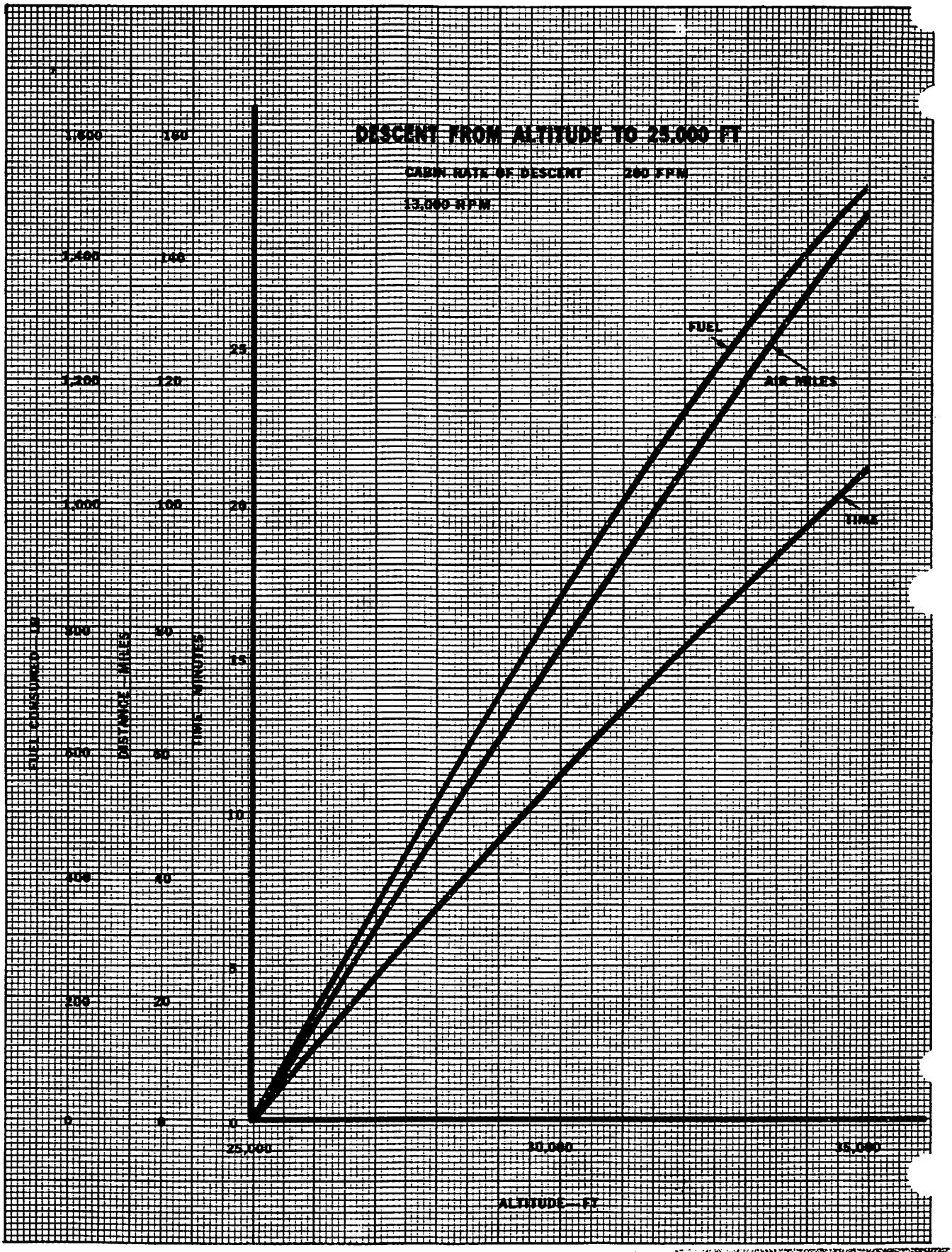

DISTANCE TO HEIGHT
14,100 R.P.M.
I.CAN. CONDITIONS
35,000
30,000
25,000
20,000
15,000
10,000
5,000
0
ALTITUDE
GROSS TAKE-OFF WEIGHT
50,000 LBS
55,000
60,000
65,000
70,000
(FOR EMERGENCY USE ONLY)
0
20
40
60
80
100
120
140
160
180
200
220
DISTANCE - MILES

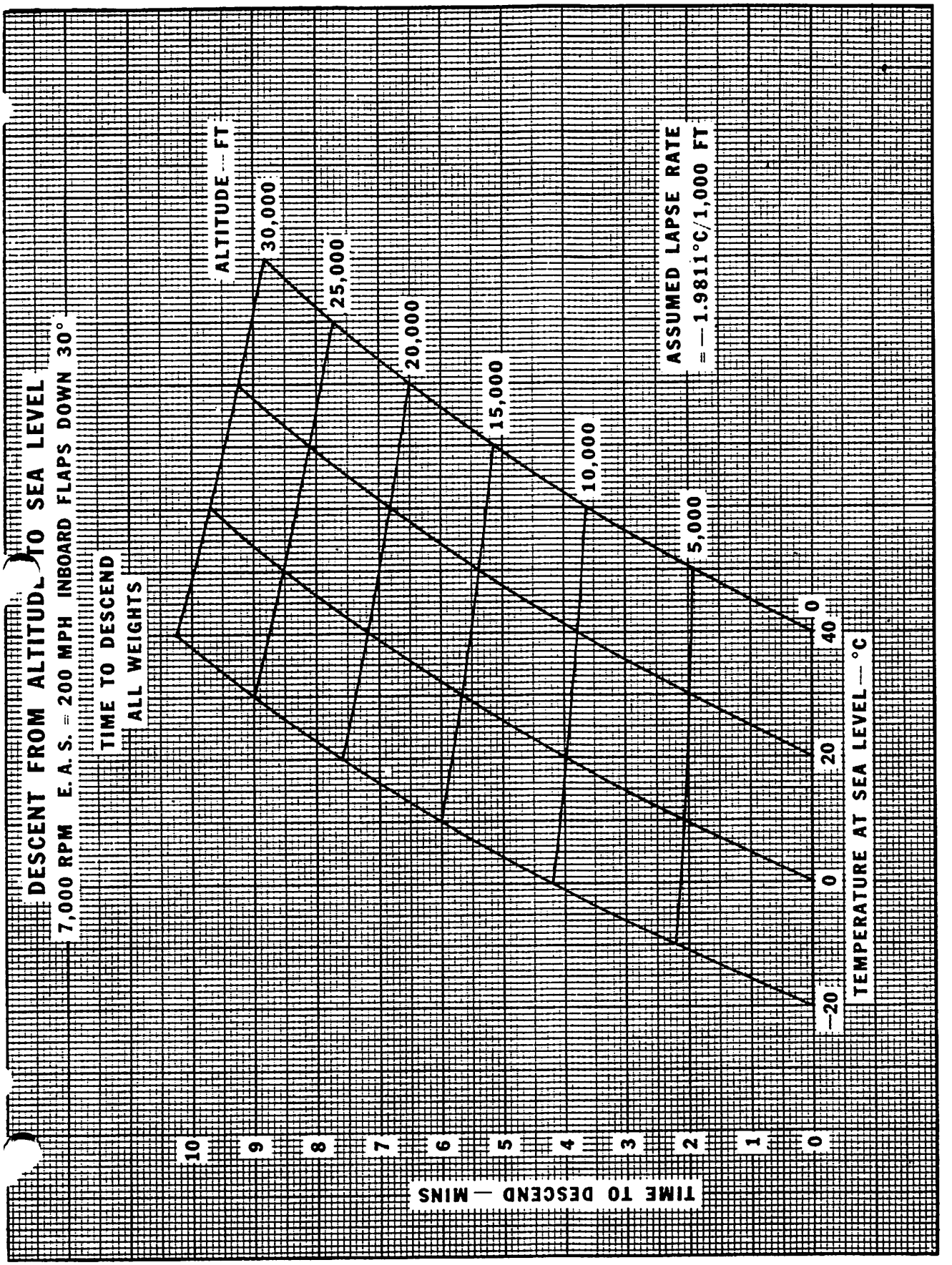
PERFORMANCE
CHART
FIG. 83



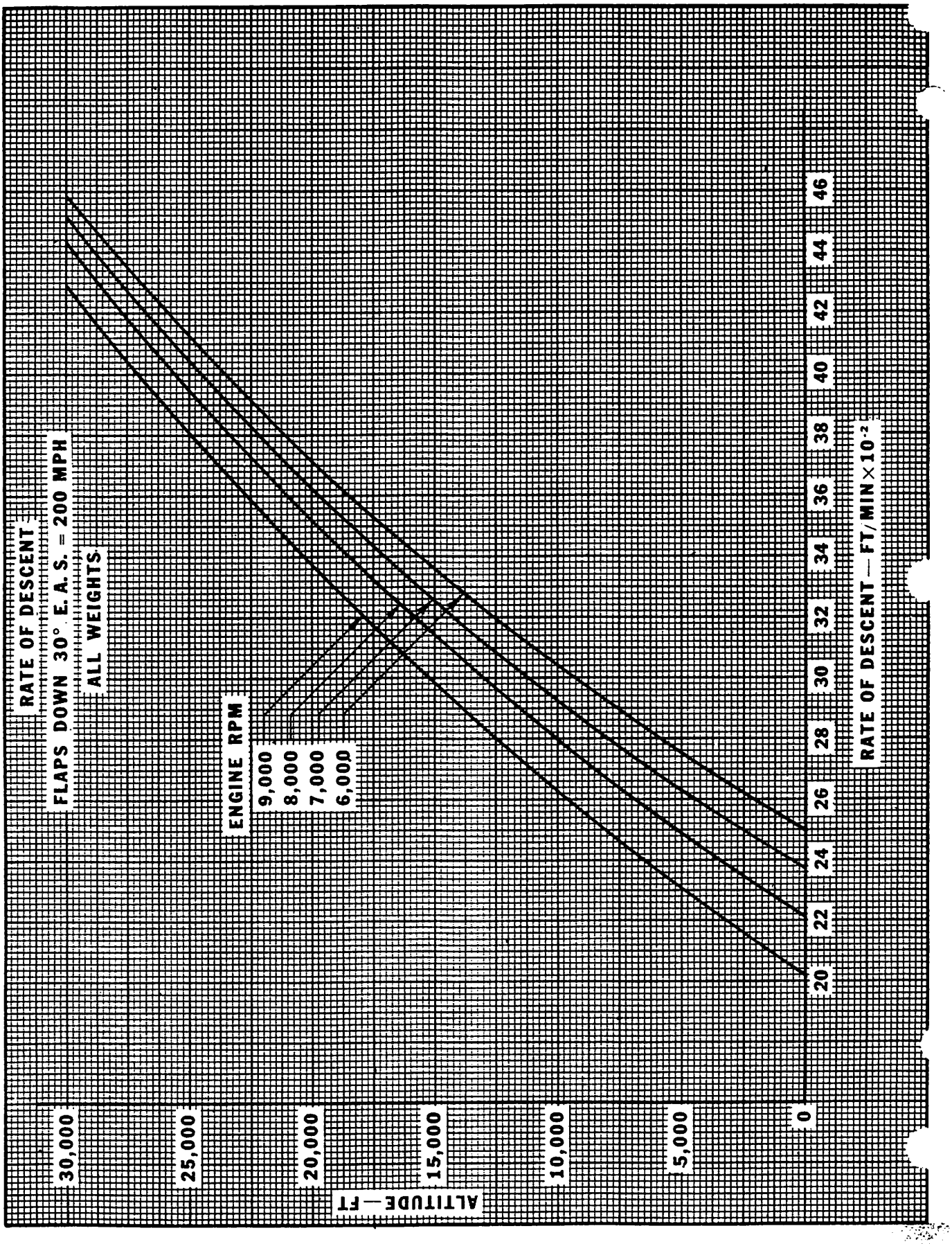

MAXIMUM LEVEL FLIGHT
SPEEDS
14,100 RPM
13,800 RPM
13,600 RPM
GROSS WEIGHT
80,000 LB
70,000 LB
TRUE INDICATED MACH NO.
.7
.6
.5
.4
.3
.2
.1
0
40,000
30,000
20,000
10,000
ALTITUDE - FT

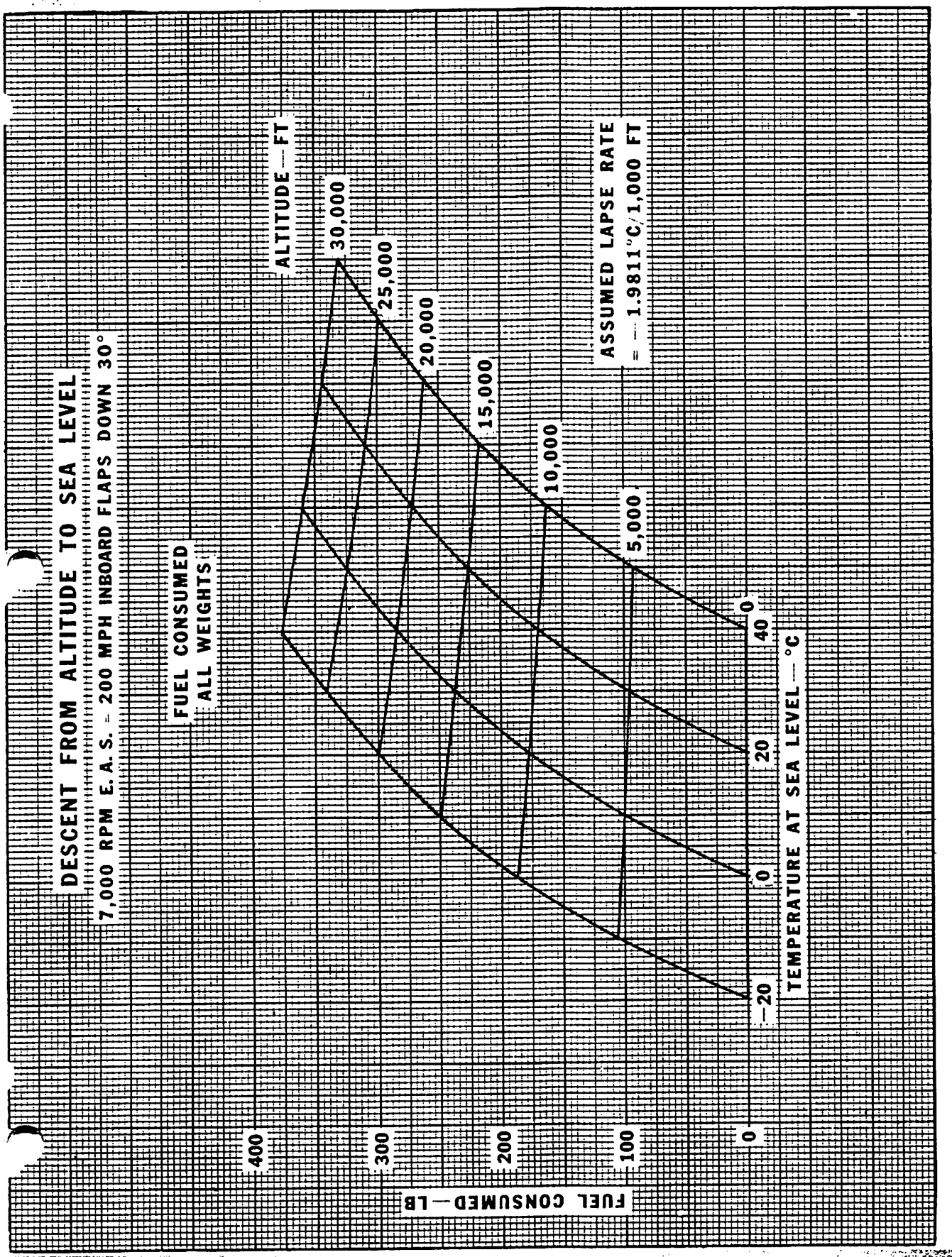
PERFORMANCE CHART
FIG. 85

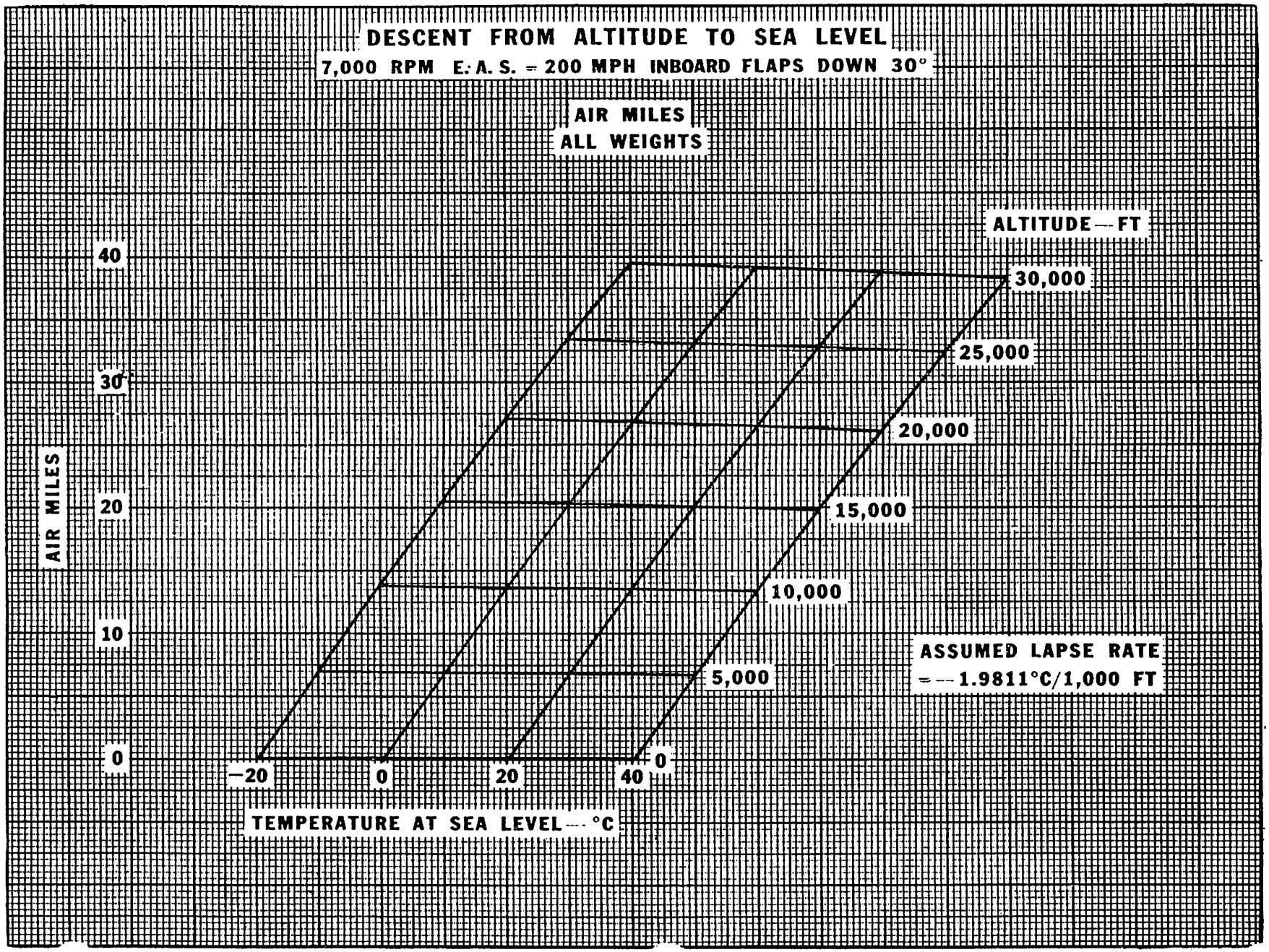



DESCENT FROM ALTITUDE TO SEA LEVEL
7,000 RPM E. A. S. = 200 MPH INBOARD FLAPS DOWN 30°
TIME TO DESCEND
ALL WEIGHTS
ALTITUDE — FT
30,000
25,000
20,000
15,000
10,000
5,000
ASSUMED LAPSE RATE
= — 1.9811°C/1,000 FT
TEMPERATURE AT SEA LEVEL — °C
40
20
0
—20
TIME TO DESCEND — MINS
0
1
2
3
4
5
6
7
8
9
10

PERFORMANCE
CHART
FIG. 87

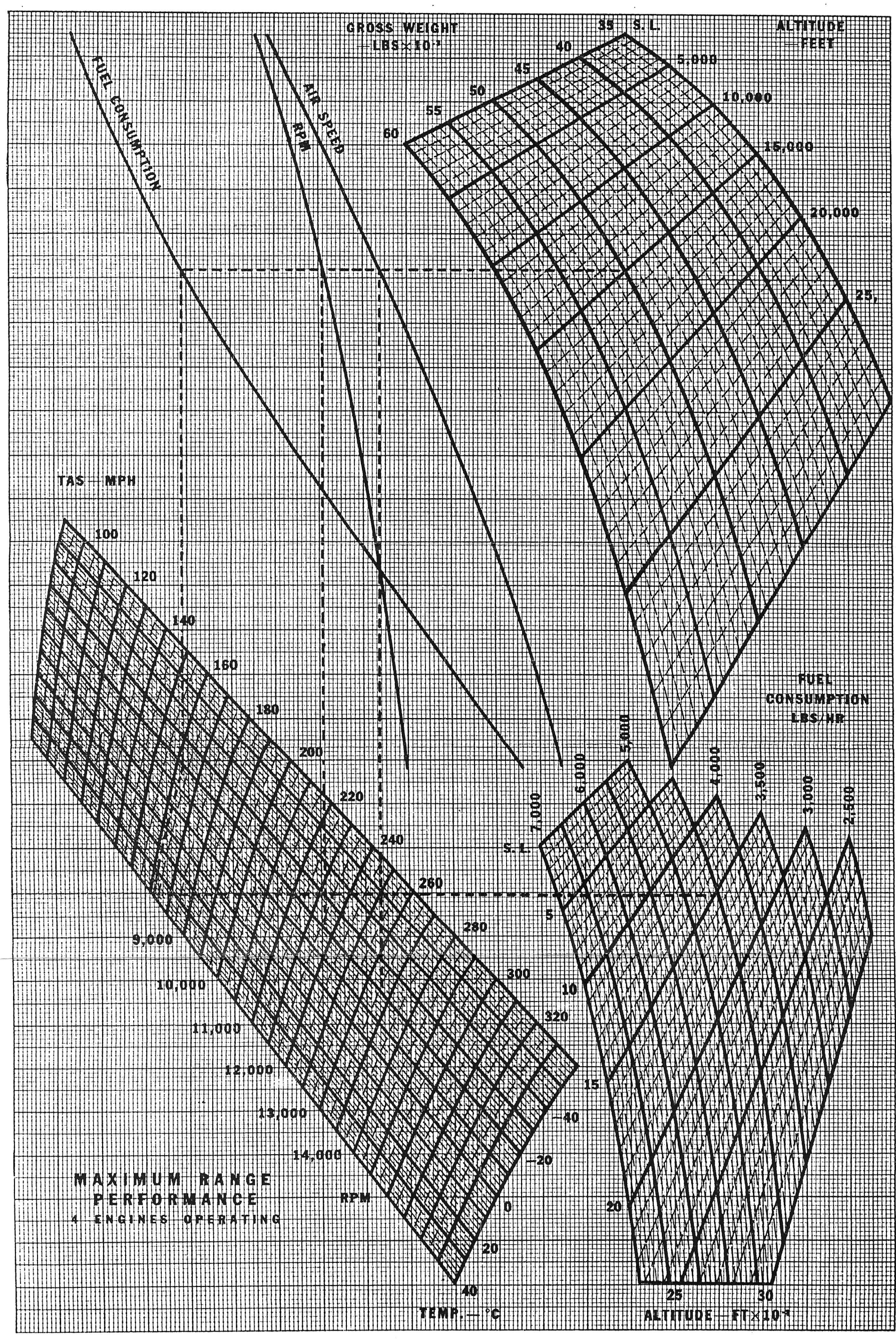



DESCENT FROM ALTITUDE TO SEA LEVEL
7,000 RPM E.A.S. = 200 MPH INBOARD FLAPS DOWN 30°
FUEL CONSUMED
ALL WEIGHTS
ALTITUDE—FT
30,000
25,000
20,000
15,000
10,000
5,000
ASSUMED LAPSE RATE
= —1.9811°C/1,000 FT
TEMPERATURE AT SEA LEVEL — °C
40
20
0
—20
FUEL CONSUMED—LB
400
300
200
100
0



PERFORMANCE
CHART
FIG. 89

317-64



PERFORMANCE CHART
FIG. 90

MAXIMUM RANGE PERFORMANCE

3 ENGINES OPERATING

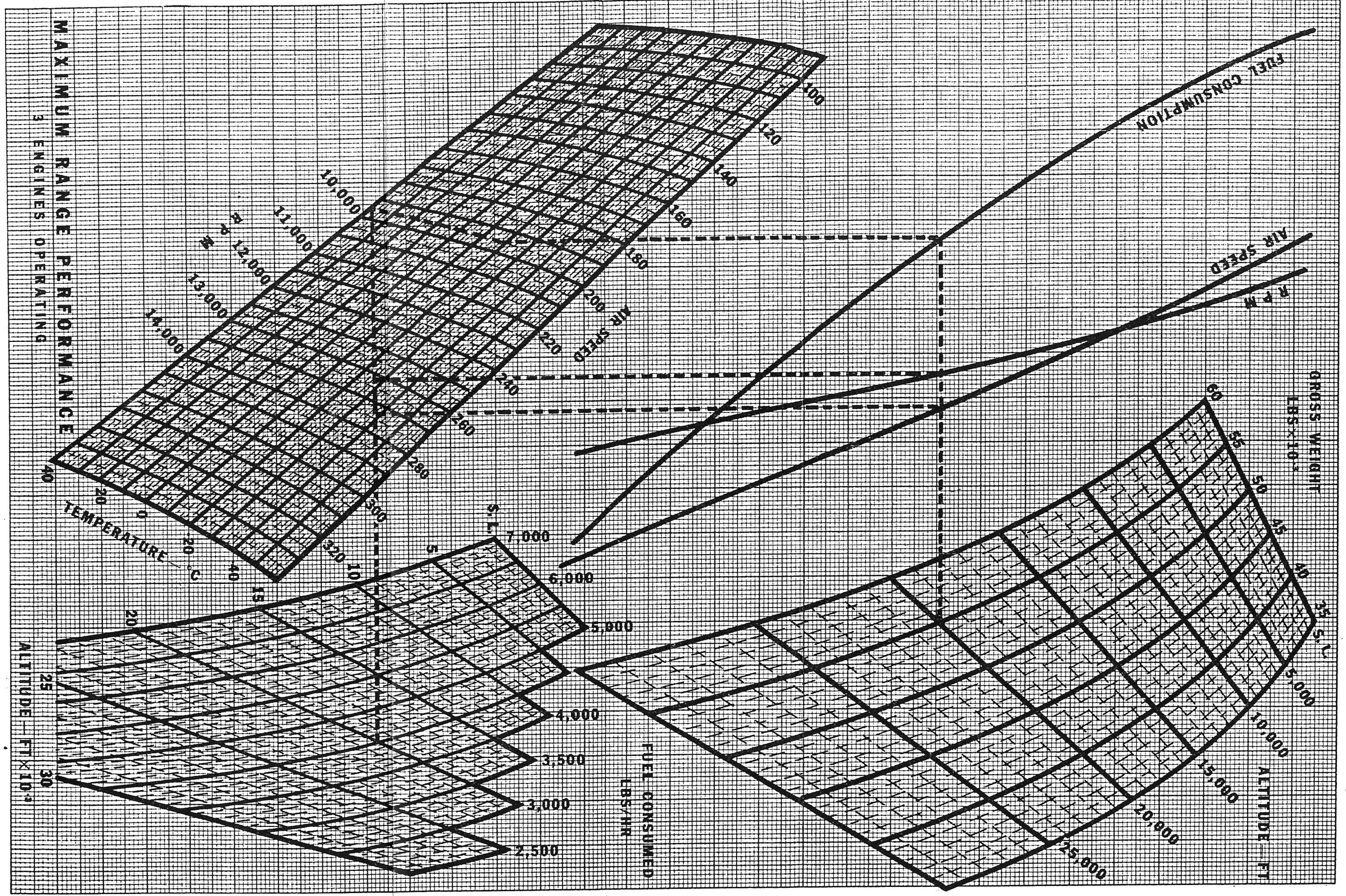
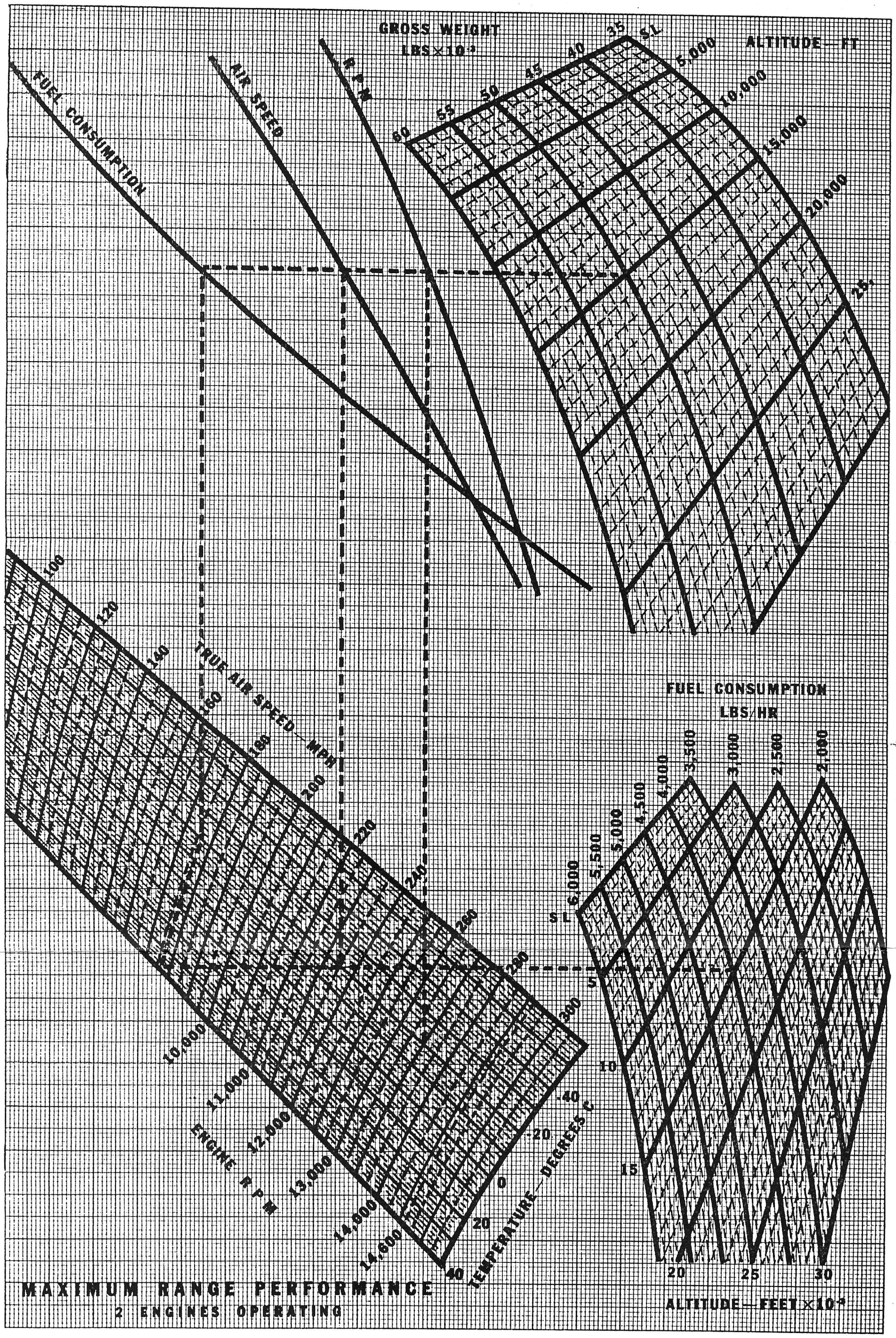


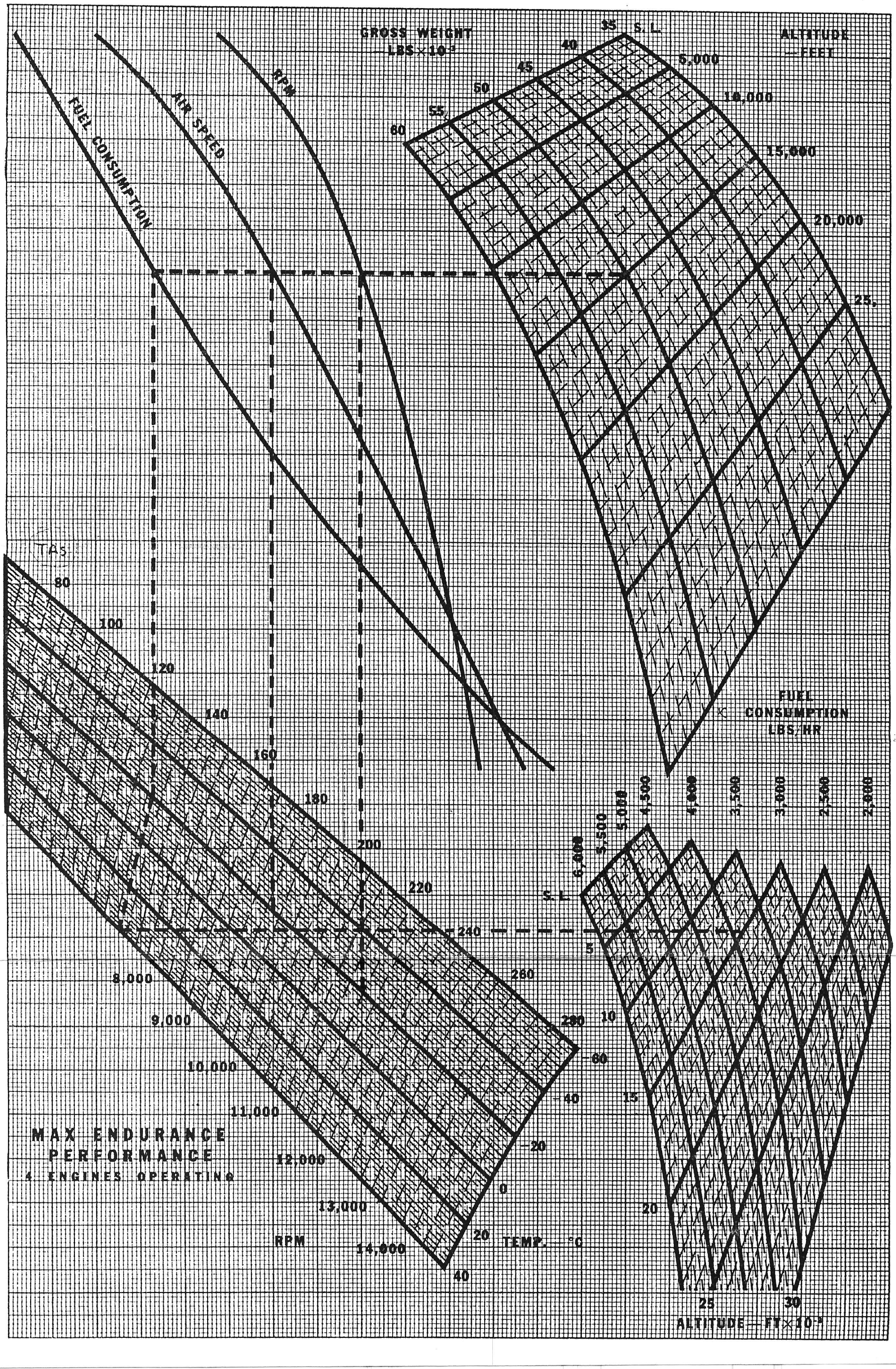
FIG. 91

PERFORMANCE CHART

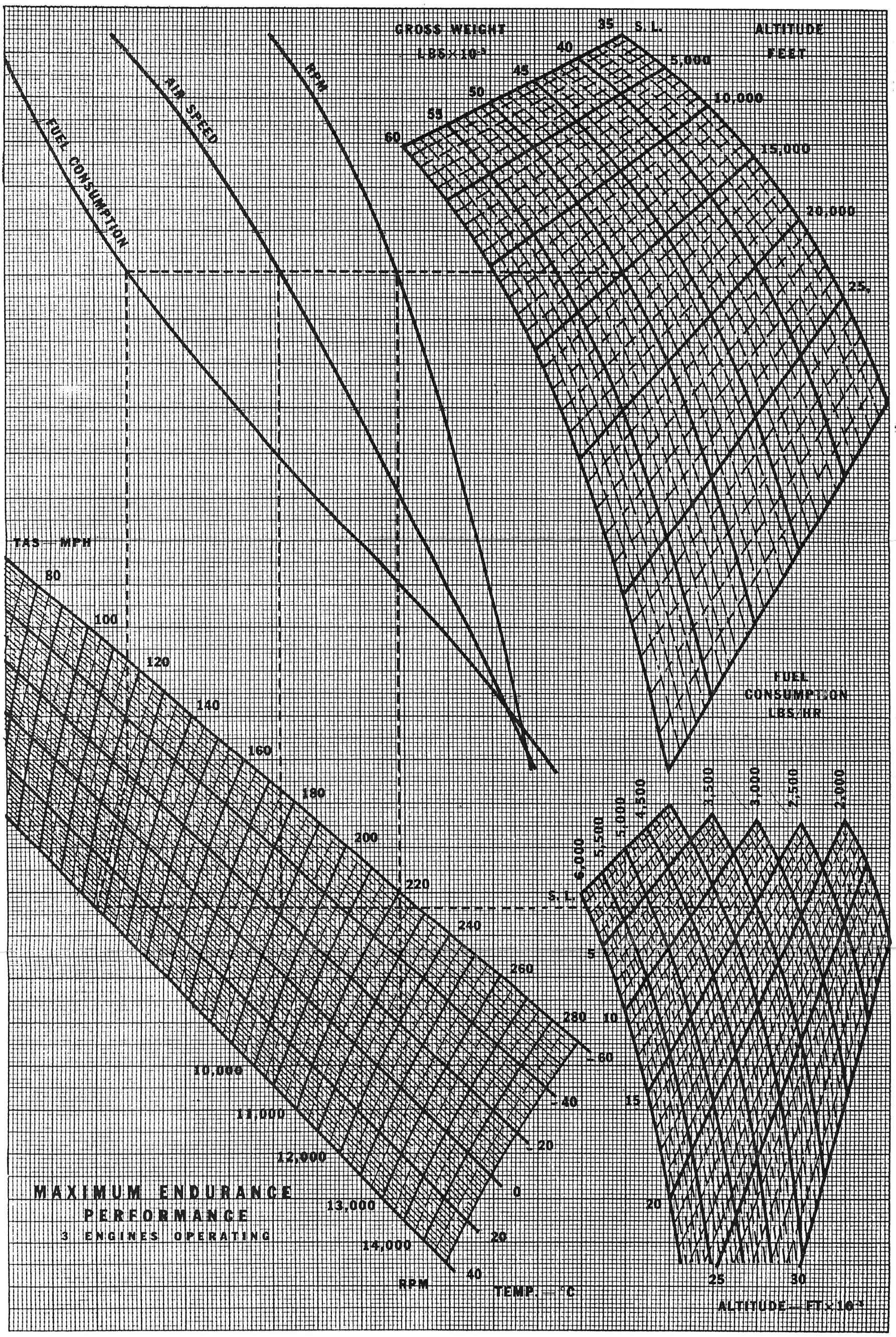


PERFORMANCE CHART

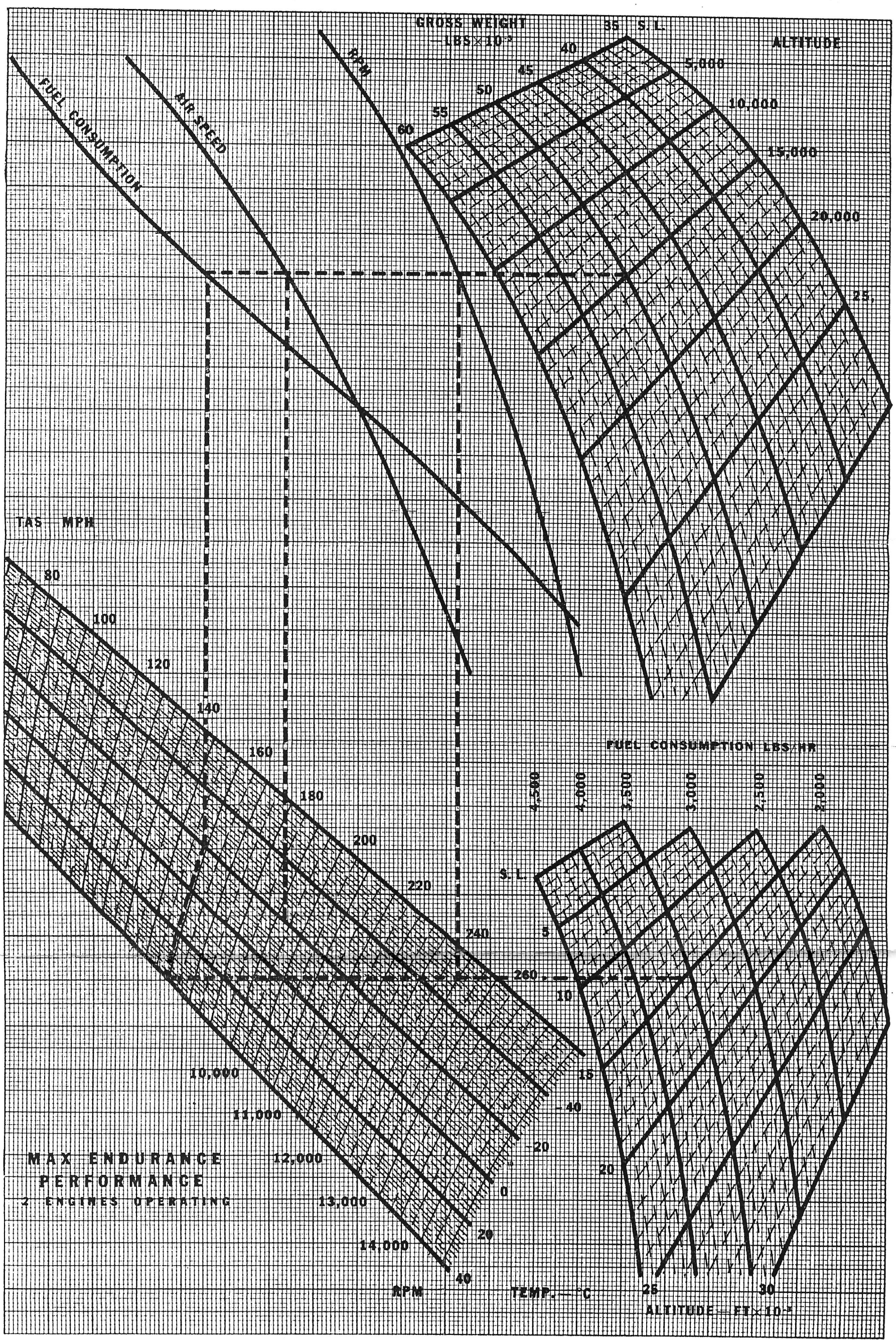
FIG. 92



PERFORMANCE
CHART
FIG. 93

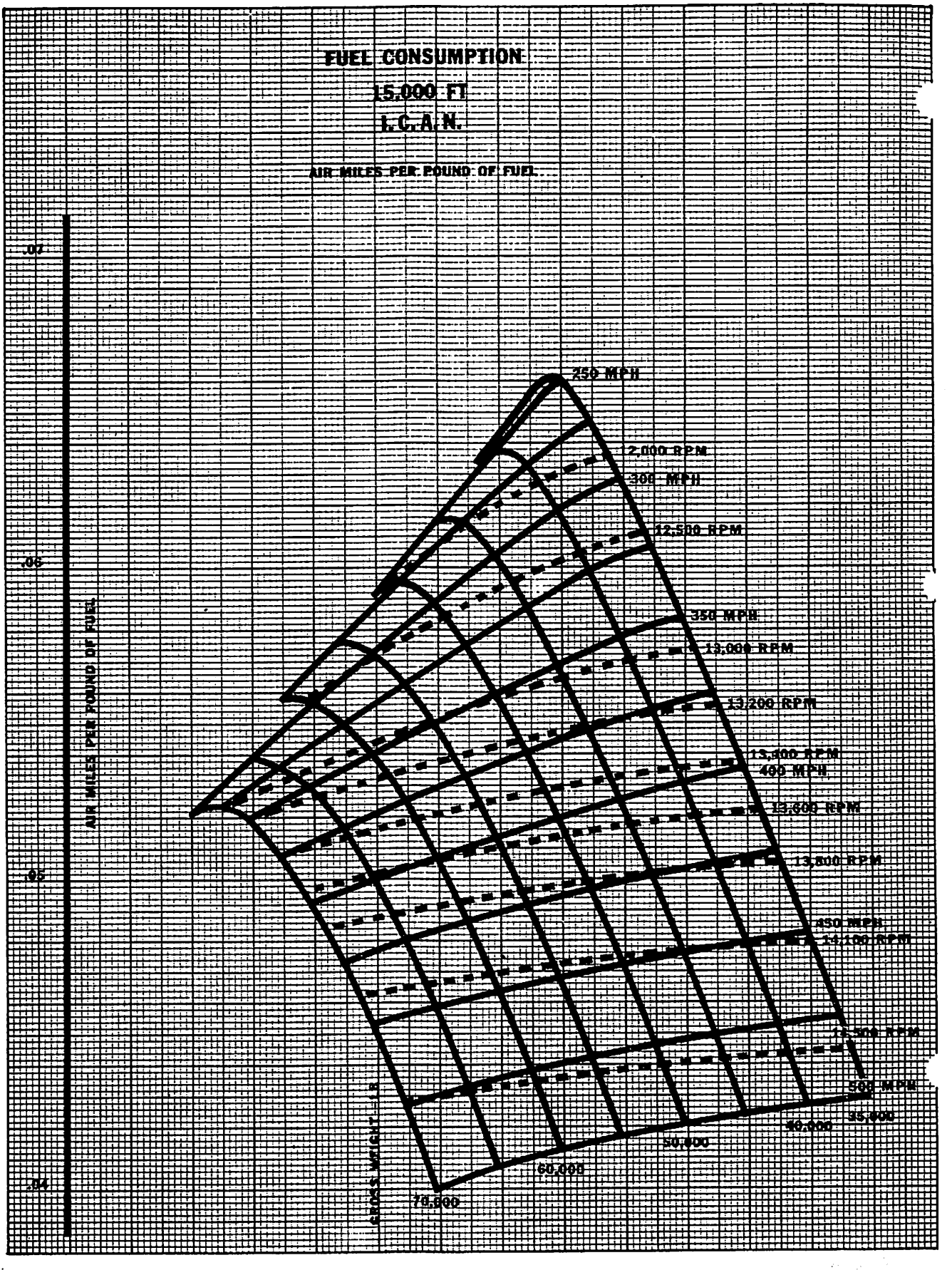


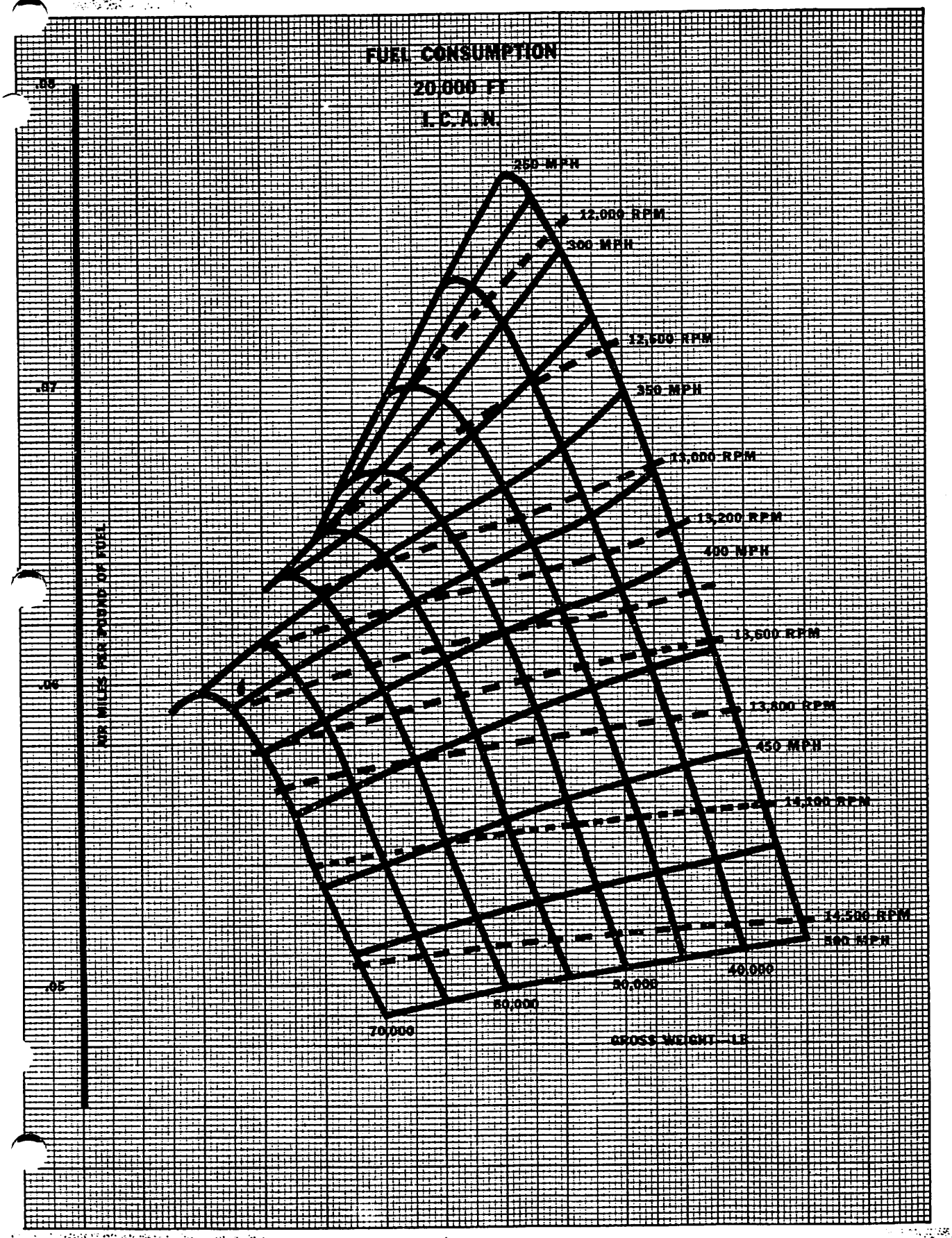
PERFORMANCE
CHART
FIG. 94



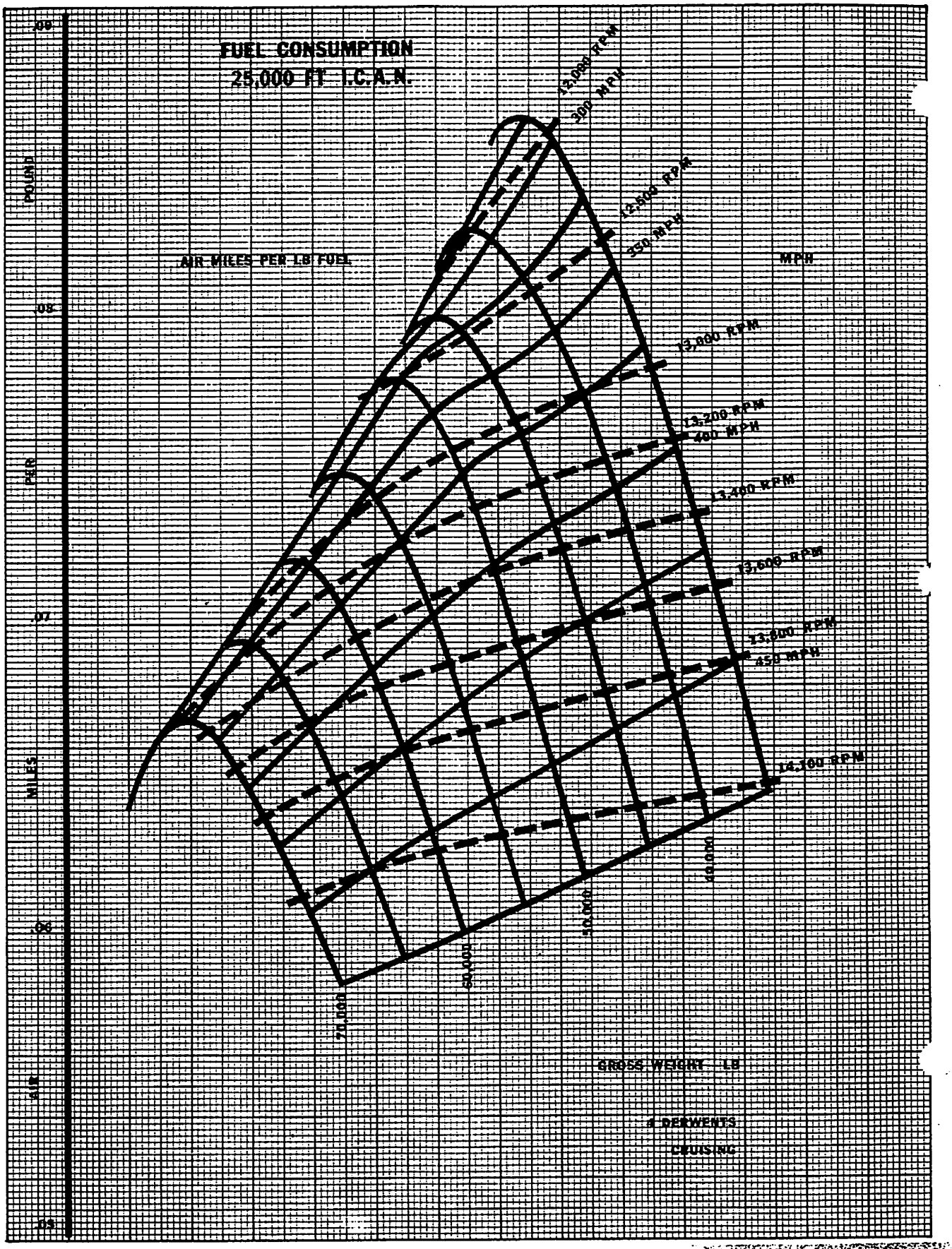
PERFORMANCE
CHART
FIG. 95

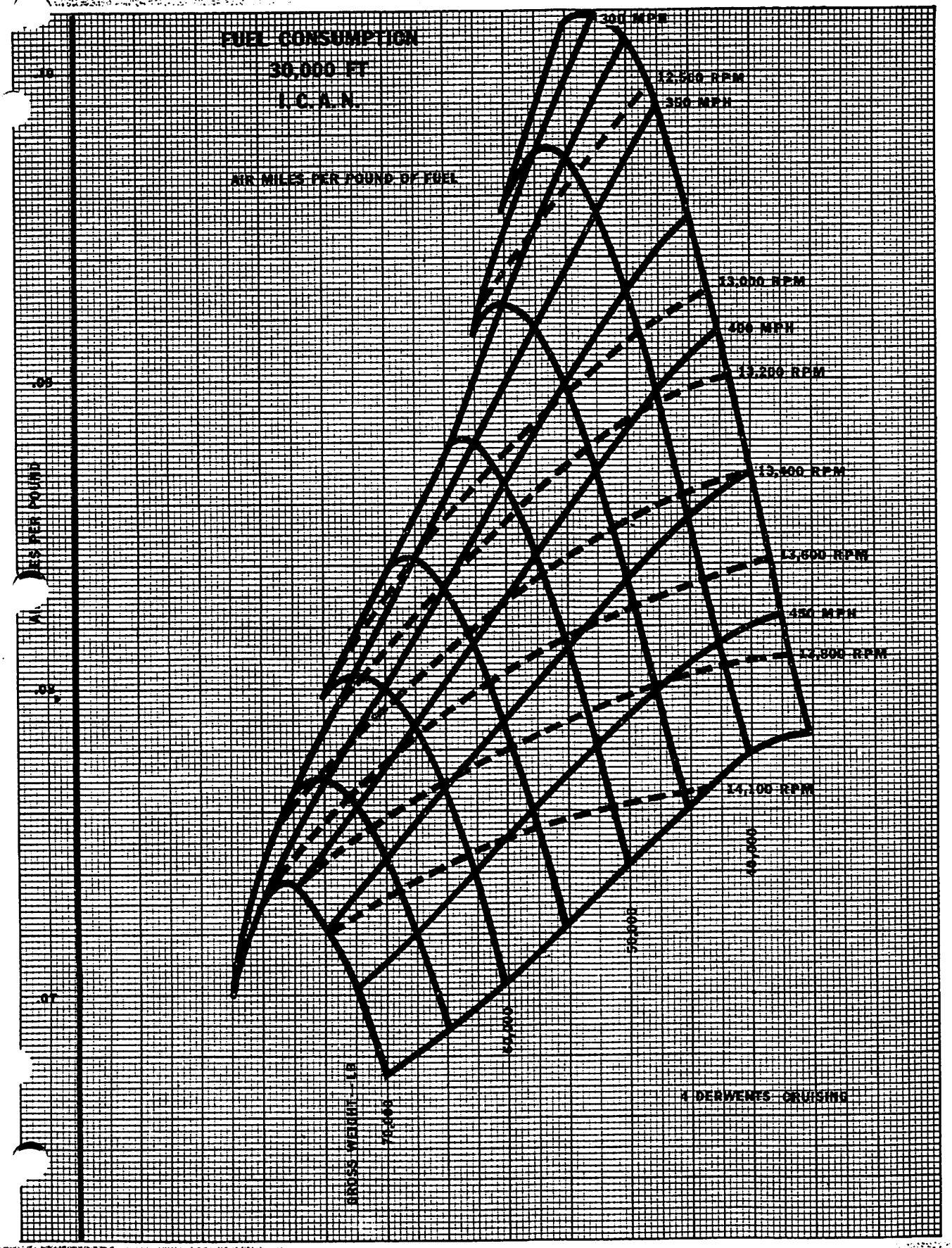
PERFORMANCE
CHART
FIG. 96



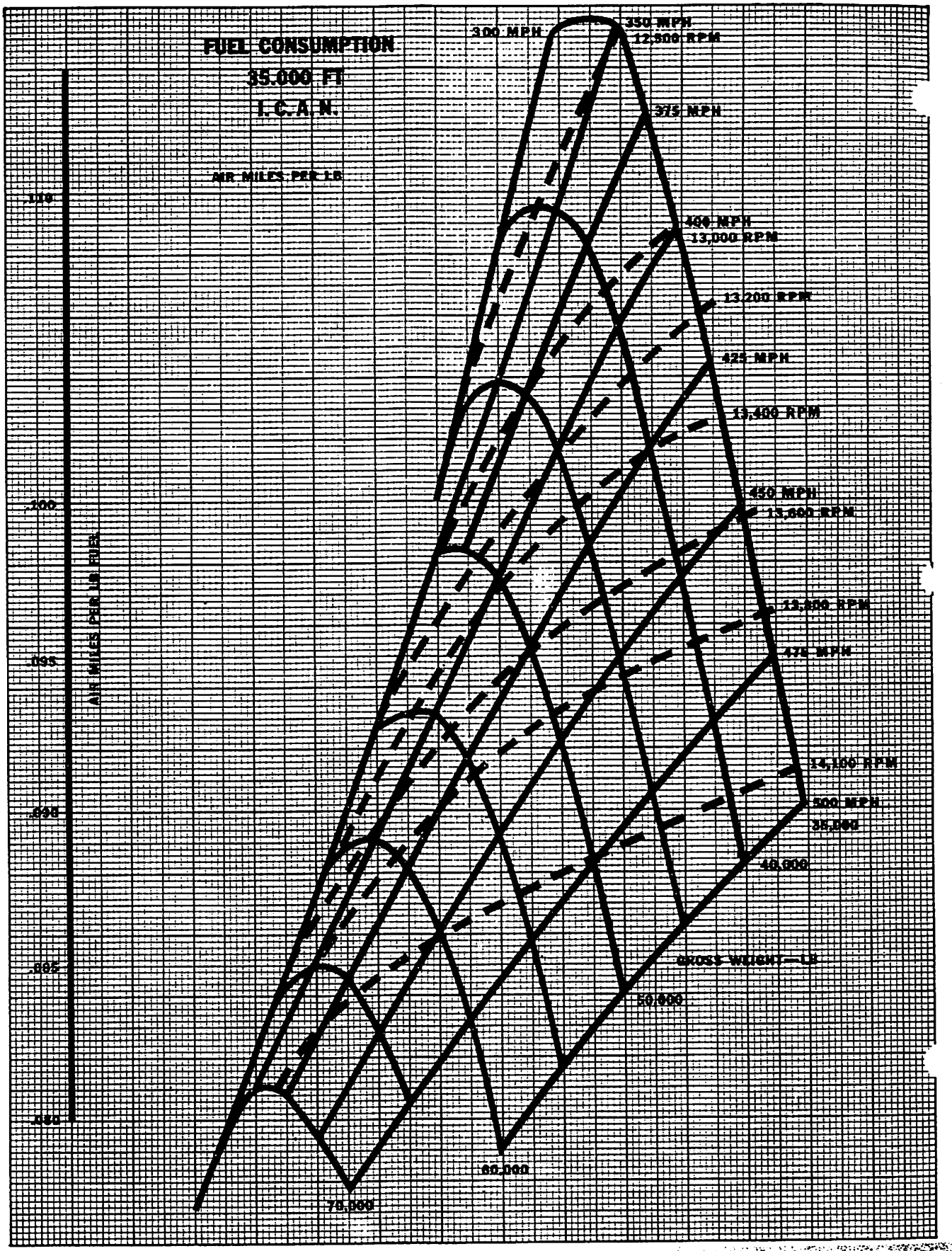

FUEL CONSUMPTION
20,000 FT
I.C.A.N.
250 MPH
12,000 RPM
300 MPH
12,500 RPM
350 MPH
13,000 RPM
13,200 RPM
400 MPH
13,600 RPM
13,800 RPM
450 MPH
14,000 RPM
14,500 RPM
500 MPH
.08
.07
.06
.05
NM MILES PER POUND OF FUEL
40,000
50,000
60,000
70,000
GROSS WEIGHT—LB

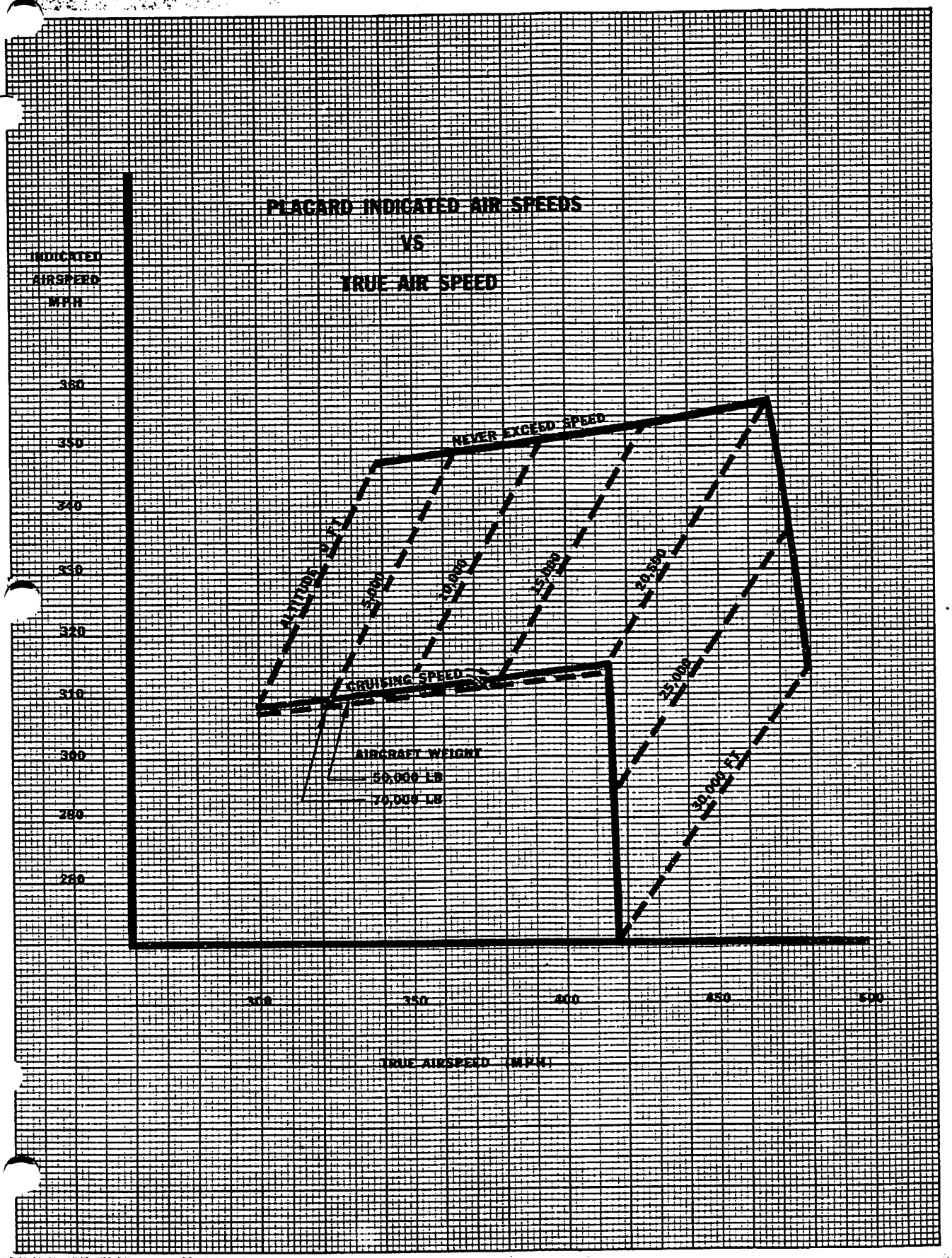
PERFORMANCE
CHART
FIG. 98



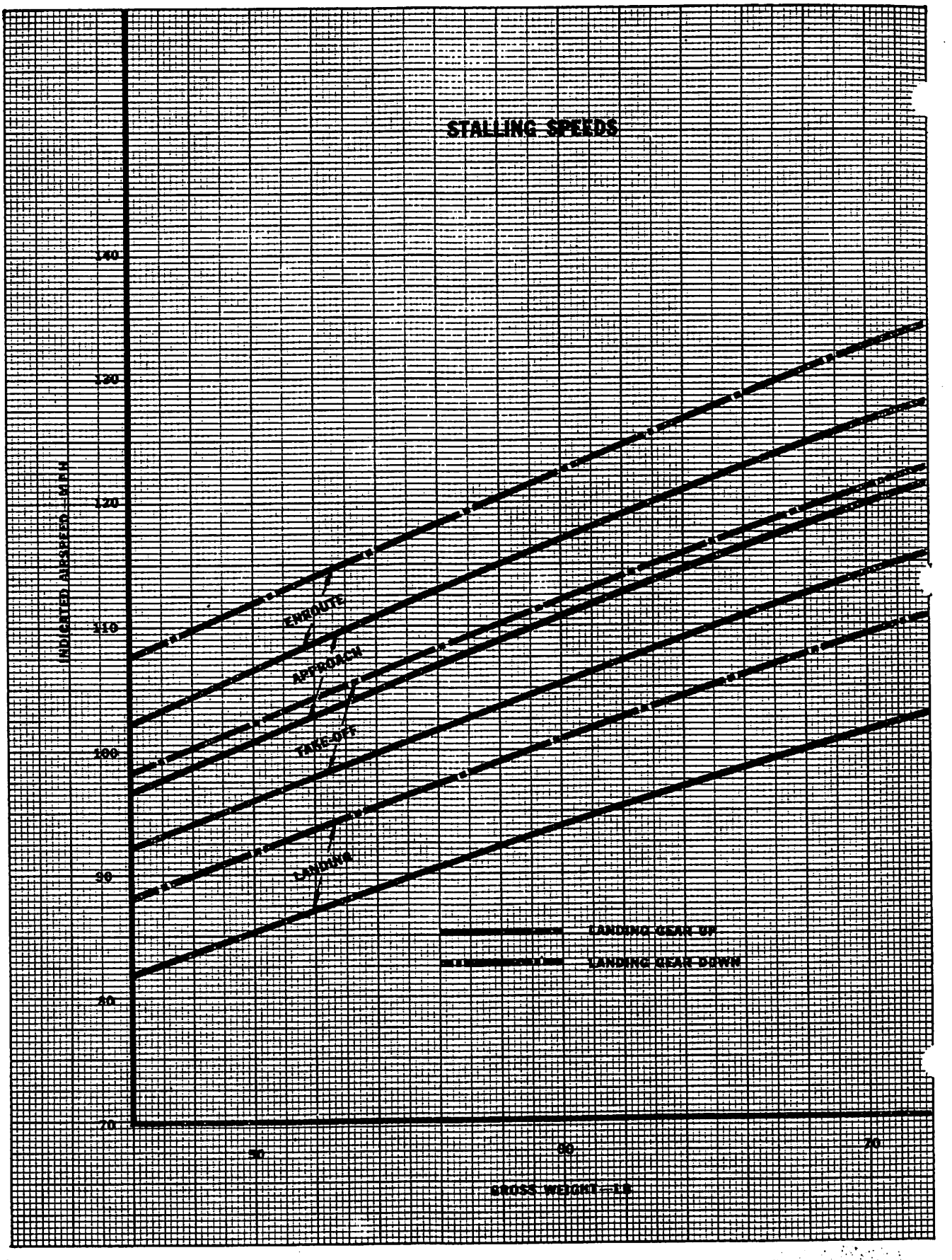

FUEL CONSUMPTION
30,000 FT
I.C.A.N.
AIR MILES PER POUND OF FUEL
.10
.09
.08
.07
AIR MILES PER POUND
300 MPH
12,600 RPM
350 MPH
13,000 RPM
400 MPH
13,200 RPM
13,400 RPM
13,600 RPM
450 MPH
13,800 RPM
14,100 RPM
GROSS WEIGHT - LB
4 DERWENTS CRUISING

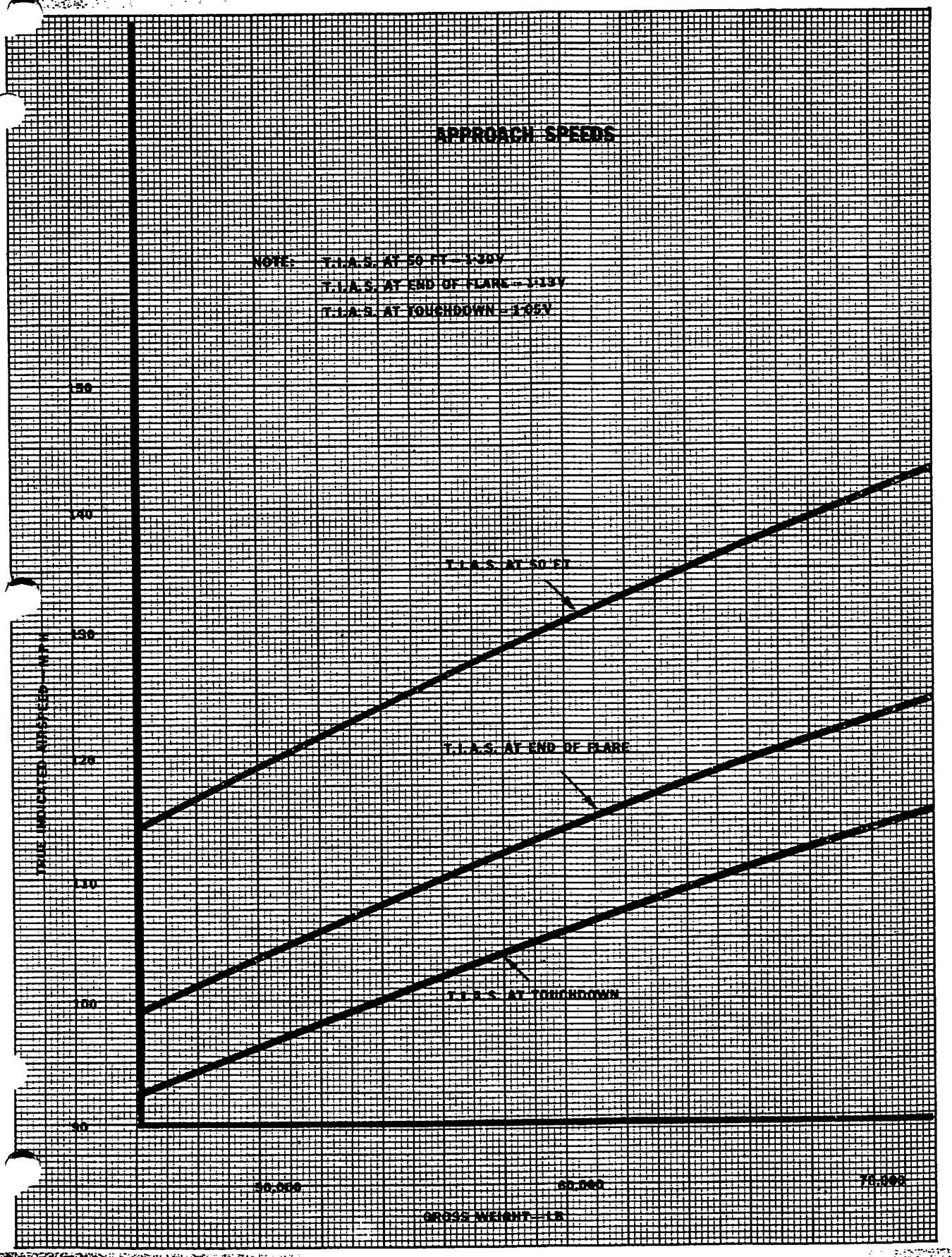
PERFORMANCE
CHART
FIG. 100

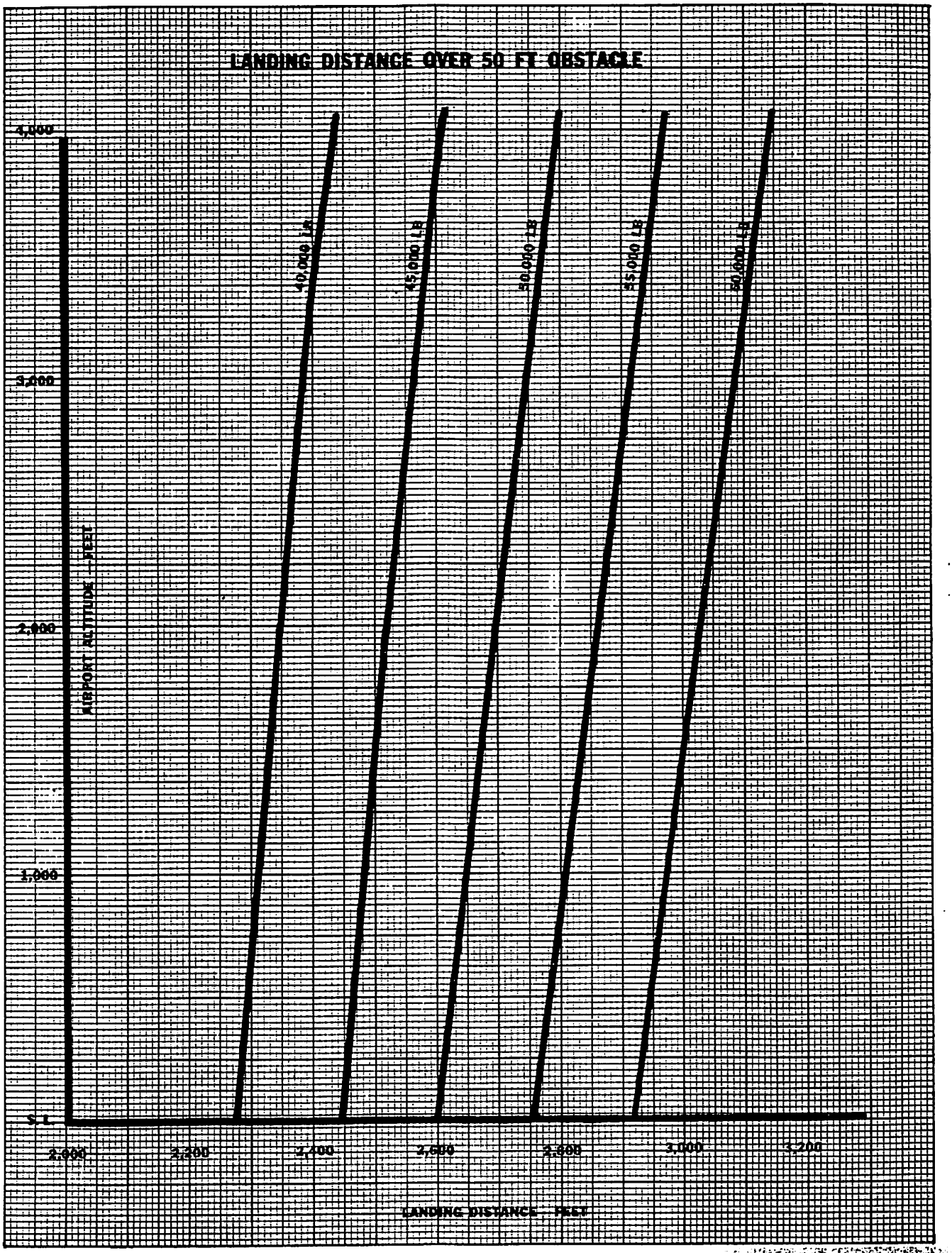



PLACARD INDICATED AIR SPEEDS
VS
TRUE AIR SPEED
INDICATED AIRSPEED MPH
380
360
340
320
300
280
NEVER EXCEED SPEED
CRUISING SPEED
SEA LEVEL
5,000'
10,000'
15,000'
20,000'
25,000'
30,000'
35,000'
AIRCRAFT WEIGHT
50,000 LB
70,000 LB
300
350
400
450
500
TRUE AIRSPEED (MPH)

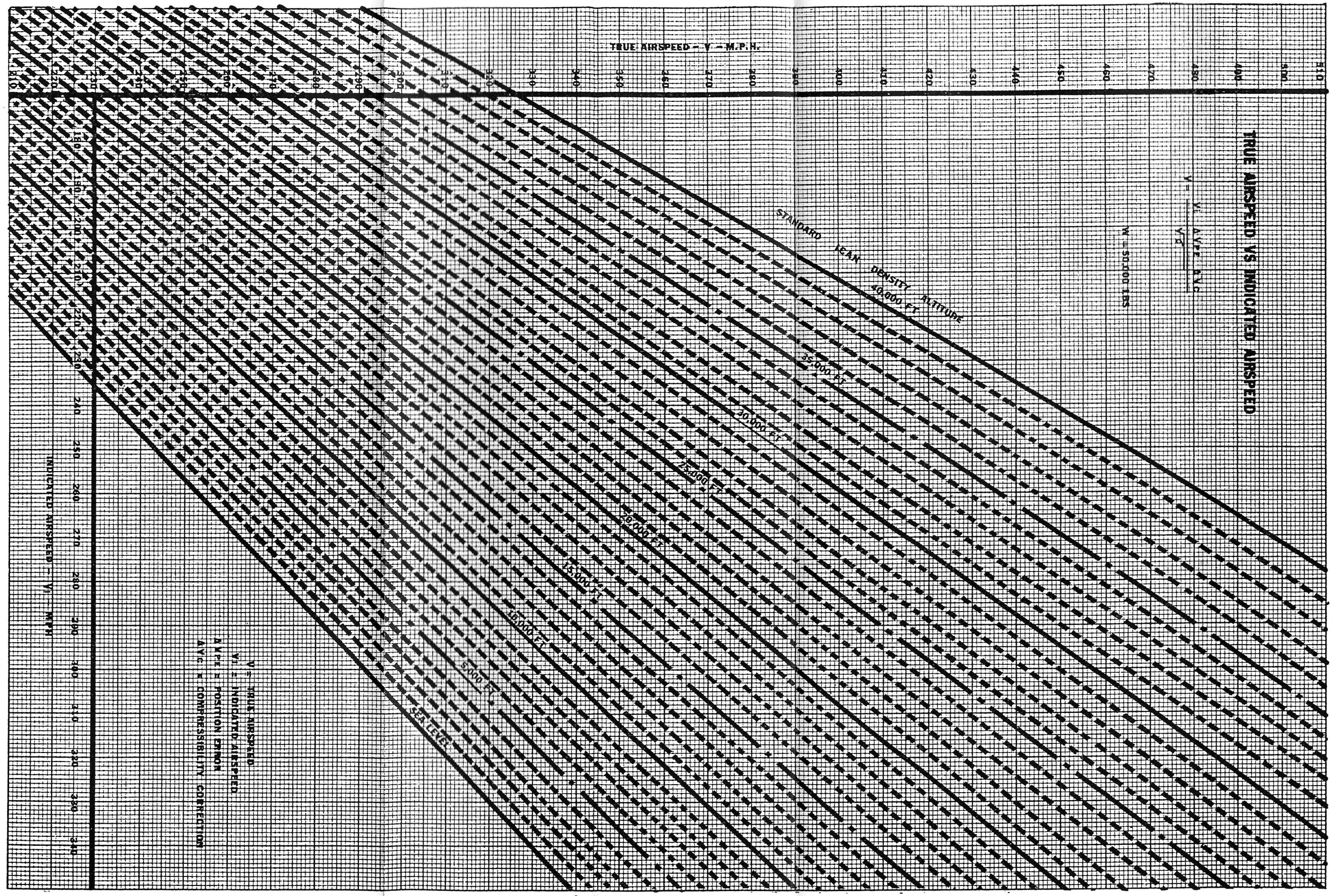
PERFORMANCE
CHART
FIG. 102




APPROACH SPEEDS
NOTE: T.I.A.S. AT 50 FT — 1.30V
T.I.A.S. AT END OF FLARE — 1.13V
T.I.A.S. AT TOUCHDOWN — 1.05V
T.I.A.S. AT 50 FT
T.I.A.S. AT END OF FLARE
T.I.A.S. AT TOUCHDOWN
TRUE INDICATED AIRSPEED—KNOTS
150
140
130
120
110
100
90
50,000
60,000
70,000
GROSS WEIGHT—LB.


LANDING DISTANCE OVER 50 FT OBSTACLE
4,000
3,000
2,000
1,000
S.L.
AIRPORT ALTITUDE — FEET
40,000 LB
45,000 LB
50,000 LB
55,000 LB
60,000 LB
2,000
2,200
2,400
2,600
2,800
3,000
3,200
LANDING DISTANCE - FEET

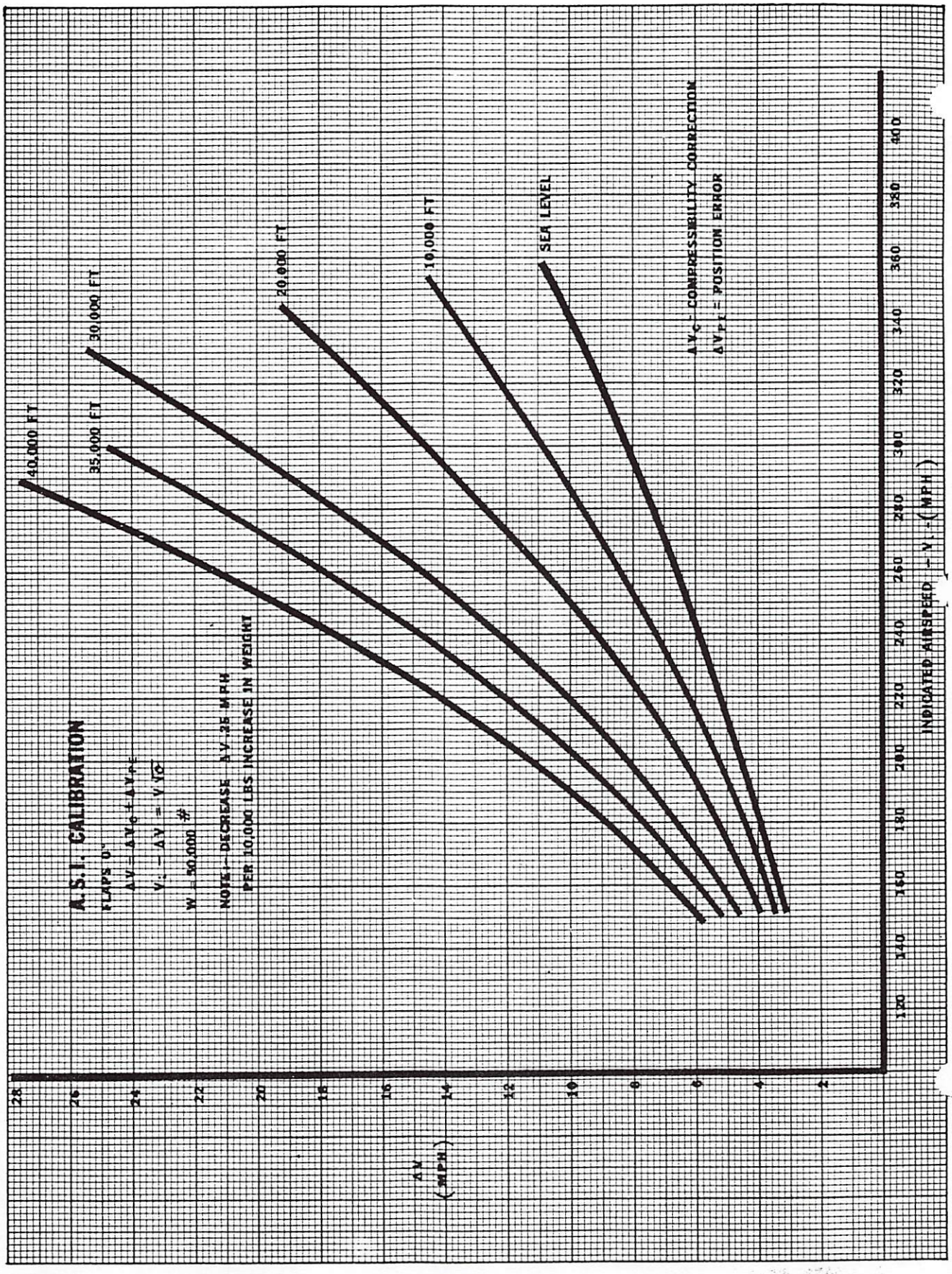
NOTES

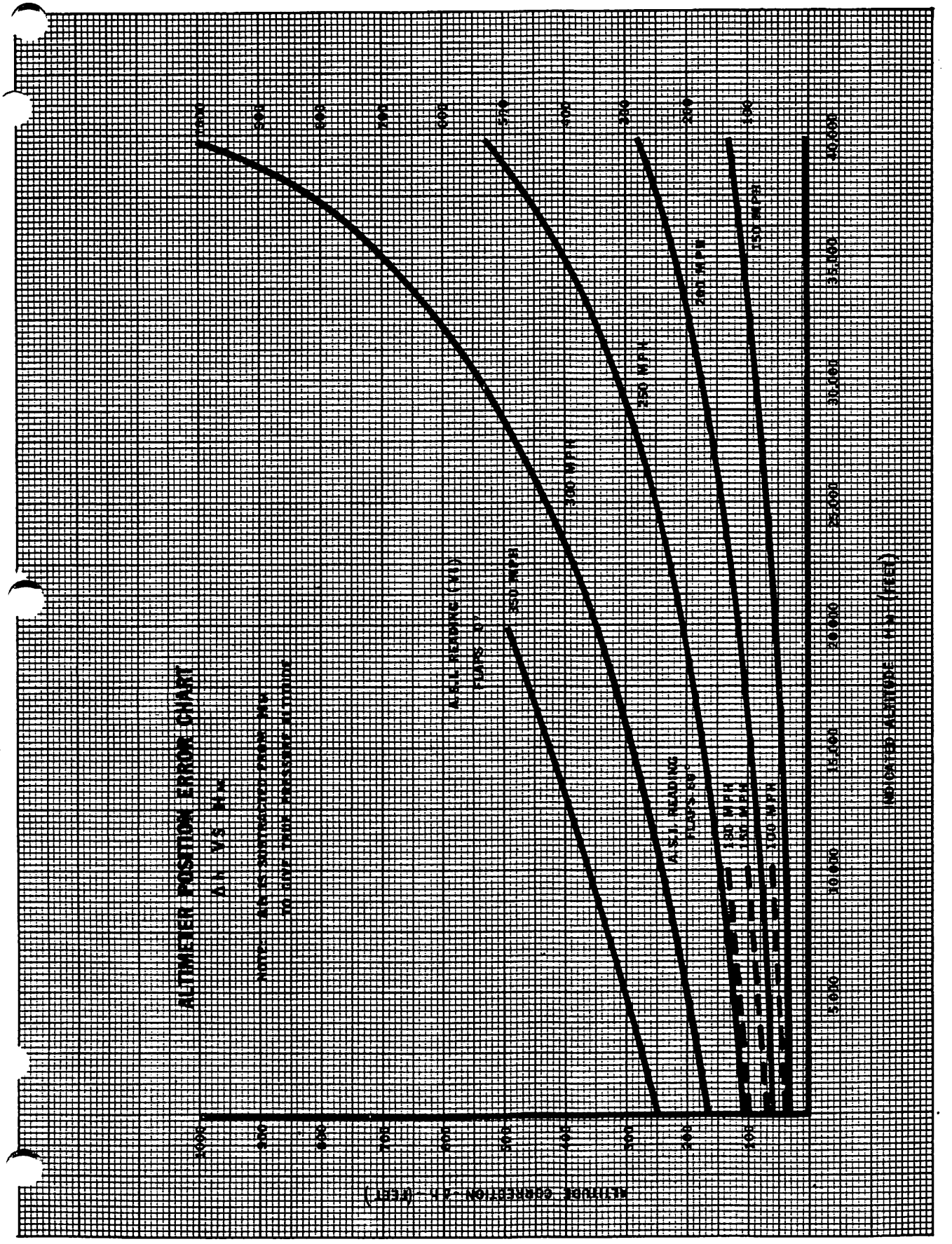


PERFORMANCE
CHART
FIG. 105

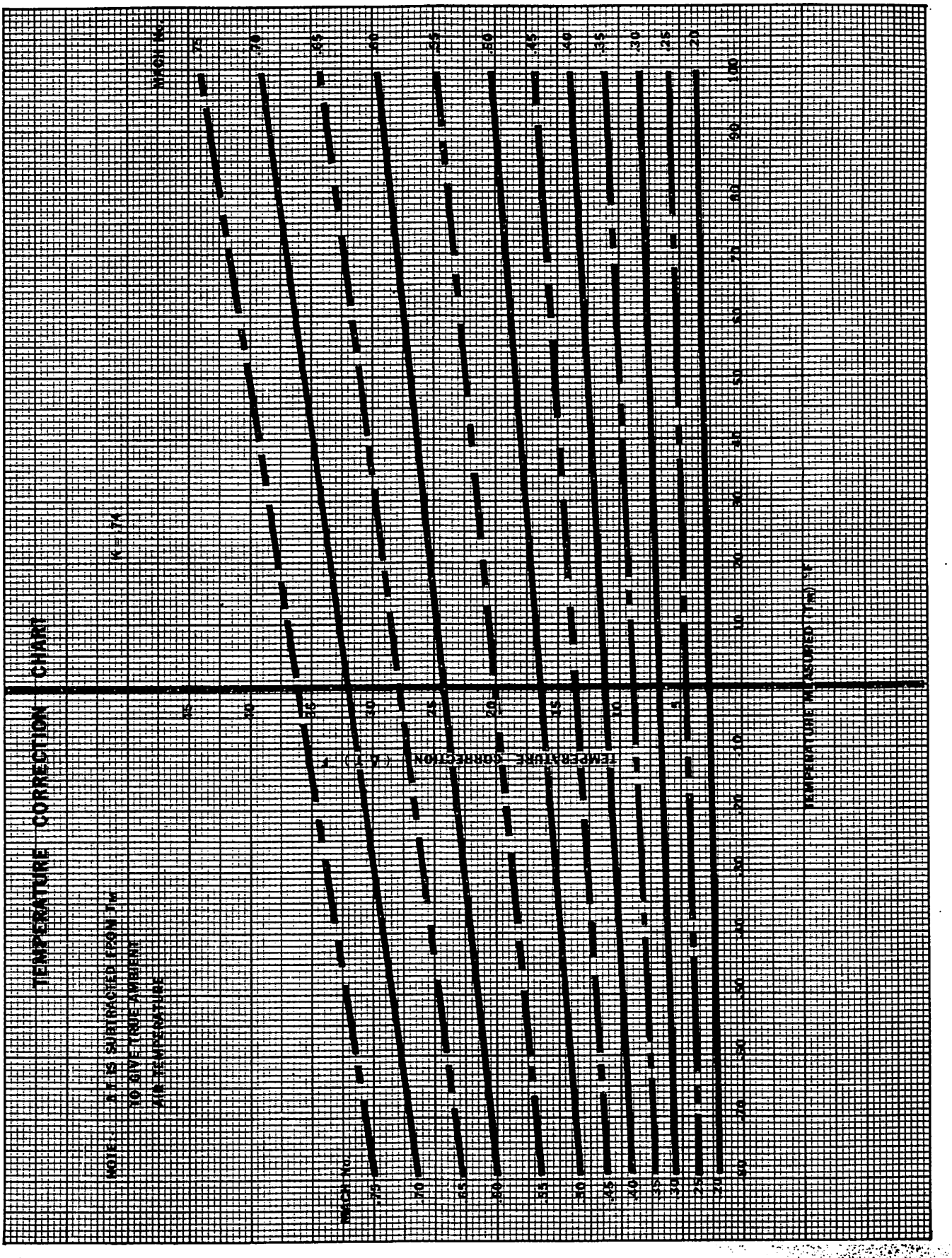
317-66

PERFORMANCE
CHART
FIG. 106




ALTIMETER POSITION ERROR CHART
ΔH VS H
NOTE: ΔH IS SUBTRACTED FROM H
TO GIVE TRUE PRESSURE ALTITUDE
A.S.I. READING (KTS)
FLAPS UP
250 KTS
230 KTS
210 KTS
190 KTS
170 KTS
150 KTS
130 KTS
110 KTS
90 KTS
FLAPS 40°
250 KTS
230 KTS
210 KTS
190 KTS
170 KTS
150 KTS
130 KTS
110 KTS
90 KTS
ALTIMETER CORRECTION - ΔH - (FEET)
INDICATED ALTITUDE - H - (FEET)
-1000
0
1000
2000
3000
4000
5000
10,000
15,000
20,000
25,000
30,000
35,000
40,000

PERFORMANCE
CHART
FIG. 108



CHAPTER 1

EMERGENCY OPERATING INSTRUCTIONS

5-1 ENGINE FAILURE ON TAKE-OFF:

(1) Due to the configuration of this type of aircraft, an engine failure on take-off is less critical than that on the conventional propeller driven aircraft.

(2) Should engine failure occur before an I.A.S. of approximately V_1 speed is reached the aircraft should be stopped. For further information regarding relative weight and temperature conditions refer to Section 4, Figs. 63 and 64. The recommended stopping procedure is as follows:

(a) Close the throttles.

(b) Hold the control column in the fully forward position.

(c) Simultaneous application of brakes.

(3) If an engine fails at take-off in excess of the V_1 speed (Fig. 63), allow the aircraft to reach the necessary V_2 speed, relative to load, before attempting to leave the ground. For further information on V_2 speeds refer to Section 4, Fig. 64.

NOTE

(1) Failure of one engine does not materially affect the handling characteristics of the aircraft to any great extent. (For further information refer to relevant charts in Section 4 of this manual.)

5-2 ENGINE FAILURE IN FLIGHT:

(1) FAILURE OF ONE ENGINE: The flying characteristics of the aircraft will not be greatly affected. The following procedures should be closely followed and an effort made to ascertain the probable cause of failure before any attempt is made to relight the engine.

(a) H.P. cock - OFF

(b) Booster Pump (according to tank selection) - OFF

(c) Adjust rudder trim

(d) Check for probable cause of failure.

NOTE

(1) Certain engine failures are not always apparent to the pilot. The first indication of trouble will be a drop in engine r.p.m. Should further trouble develop, stop the engine.

(2) The L.P. cock should be left in the OPEN position.

(3) The following paragraphs 5-3 and 5-4 outline emergency proceedings under the various conditions.

5-3 RELIGHTING DURING FLIGHT:

(a) Throttle	- OPEN (one third)
(b) Windmilling r.p.m.	- 1,000 to 1,200 (Air speed 200-235 m.p.h.)
(c) Relight button (after 5 seconds)	- PRESS and HOLD
(d) H.P. cock	- ON
(e) Relight button	- RELEASE (when 2,000 r.p.m. has been reached)
(f) Throttle	- REDUCE opening without delay

NOTE

(1) If engine fails to relight after period of 30 seconds, turn H.P. cock OFF. Pause for one minute and make a fresh attempt at a lower altitude with a wider throttle opening. Keep the r.p.m. low.

(2) Relighting of engines must NOT BE ATTEMPTED AT ALTITUDES OVER 15,000 ft.

(3) Relighting should only be carried out at altitudes of 15,000 ft. or less.

(4) The given throttle position is a compromise between the positions giving best relighting (OPEN) and least overheating after relight (CLOSED). The greater the height, the more the throttle may have to be opened. In the case of the latter, the throttle must be closed immediately a relight is obtained.

(5) The range of the given windmilling r.p.m., is a compromise between those giving the best relight (low r.p.m.) and safe I.A.S. when flying on three or less engines at all altitudes.

(6) The possibility of a successful relight is increased with a decrease in altitude.

(7) Before opening the throttle, wait until the engine r.p.m. is steady and the rev. counter indicates the proper idling speed for the specified altitude.

(8) An engine relights more easily when warm.

(9) Care must be taken when opening the throttles. If the throttle opening is too rapid, flame extinction may occur. Do not exceed engine limitations at high altitudes.

5-4 COMBUSTION FAILURE DUE TO MIS-HANDLING:

(1) Proceed as outlined in paragraph 5-2.

5-5 USE OF AVIATION GASOLINE:

(1) Aviation gasoline thoroughly mixed with a 3% volume of suitable aero engine lubricating oil may be used in an emergency. This mixture can be added to any Aviation Turbine Fuel already in the tanks.

(2) It is permissible to mix any of the following fuels and oils, or their recommended equivalents:

AVIATION GASOLINE

73 Octane
83 Octane
91/98 Grade
100/130 Grade

with

AVIATION OILS

DED. 2472	B/O (OM-270)	or	AN-O-8 (Grade 1100)
DED. 2472	A/O (OM-170)	or	AN-O-8 (Grade 1080)
DED. 2472	C/O (OM-370)	or	AN-O-8 (Grade 1120)

MIXING: The gasoline and oil should be poured simultaneously through a funnel approximately 15 inch diameter with a 1 inch drain exit duct; the correct proportion is 10 of gasoline to 1 of oil, the oil being poured at about 1 gal./minute into the surface of the funnel. Good mixing in this concentration allows easy dilution by the addition of further gasoline in order to obtain the required 3% oil mixture. The utmost care to exclude dirt is necessary.

EFFECTS ON PERFORMANCE:

The pilot should remember:

(1) The range will be down by as much as 10%.

(2) Flight above 15,000 ft. is prohibited.

(3) The engines may under - or over - speed by 300 r.p.m. and in the latter case must be controlled by the throttles to give not more than 14,700 r.p.m.

(4) At equal r.p.m. the power is the same on all fuels.

5-6 FIRE CONTROL:

(1) The Methyl Bromide and CO2 fire warning and extinguisher system indicators and controls are located on the lower half of the flight deck roof panel. A description of the system is given in paragraphs 1-78 to 1-86 inclusive.

(2) ENGINE FIRES: In the case of an engine fire, the relative indicator will show the location. The following procedure should then be followed:

(a)	Throttle	-	CLOSED
(b)	L.P. and H.P. cocks	-	OFF
(c)	Fire extinguishing switch to position	-	1 or 2 (bottle to be discharged)
(d)	Aircraft speed	-	REDUCED

NOTE

(1) When the throttle is closed, a time delay switch is energized which will automatically discharge the selected bottle 12 to 15 seconds after the relative throttle is closed. This delay allows time for engine r.p.m. to drop to idling speed prior to discharge of the Methyl Bromide bottle. If idling r.p.m. is reached before the time delay cycle is completed, the bottle may be manually discharged by pressing the relative discharge button.

(2) If the fire persists, select the remaining fire bottle and discharge by means of the manual switch.

(3) AUXILIARY GEAR BOX FIRES: If a fire occurs in either of the auxiliary engine gear boxes, generator, or wing fillet areas, the appropriate warning light and the alarm bell, will signal the danger area. Proceed as follows:

(a) Fire extinguishing switch - Select bottle to be discharged

(b) Discharge button - PRESS

NOTE

(1) If fire persists, reset the selector switch for the alternative bottle and PRESS the discharge button.

(4) ACCESSORIES COMPARTMENT FIRE: If a fire occurs in the accessories compartment, the alarm bell will ring, and the relative panel indicator will be illuminated. To operate the system proceed as follows:

(a) Investigate compartment

(b) Fire extinguishing switch - Select bottle to be discharged

(c) Discharge button - PRESS

NOTE

(1) If fire persists, reselect the switch controlling the alternate bottle and PRESS the discharge button.

(2) Oxygen masks must be used when a fire occurs within cabin areas. This action is necessary in order to eliminate the possibility of crew members being overcome by contaminated air. Under these conditions the flow of oxygen must be regulated to EMERGENCY FLOW REQUIREMENTS.

(5) FUSELAGE FIRES: Should fire occur in the flight deck, main cabin, baggage compartments etc., use the portable extinguishers. For location of these items, see Fig. 2.

NOTE

(1) The procedures described in this paragraph cover action to be taken in the event of electrical fires. CO2 extinguisher fluid is capable of handling electrical fires.

5-7 EMERGENCY EXITS:

(1) There are five emergency exits in the aircraft (Fig. 110).

(2) The flight deck and cabin emergency escape hatches are released by turning the appropriate operating handles to the RELEASE position. All hatches open inwards.

WARNING

Cabin pressure must be released before attempting to open the escape hatches.

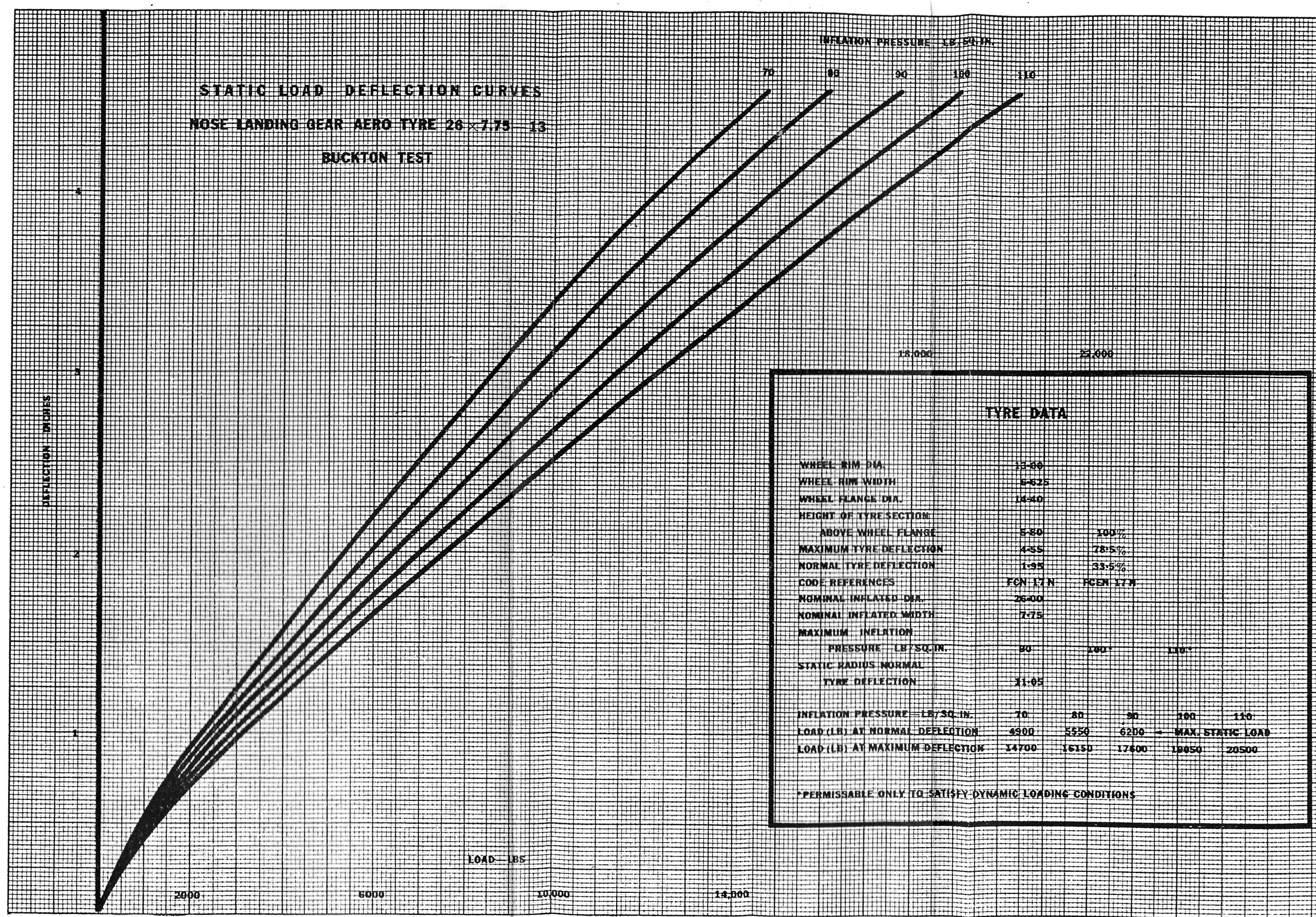
317-68

PERFORMANCE
CHART
FIG. 109

STATIC LOAD DEFLECTION CURVES

NOSE LANDING GEAR AERO TYRE 26×7.75 13

BUCKTON TEST



TYRE DATA

WHEEL RIM DIA.	13·00		
WHEEL RIM WIDTH	5·625		
WHEEL FLANGE DIA.	14·40		
HEIGHT OF TYRE SECTION ABOVE WHEEL FLANGE	5·80	100%	
MAXIMUM TYRE DEFLECTION	4·55	78·5%	
NORMAL TYRE DEFLECTION	1·95	33·5%	
CODE REFERENCES	FCN 17 N	FCEN 17 M	
NOMINAL INFLATED DIA.	26·00		
NOMINAL INFLATED WIDTH	7·75		
MAXIMUM INFLATION PRESSURE LB/SQ.IN.	80	100*	110*
STATIC RADIUS NORMAL TYRE DEFLECTION	11·05		

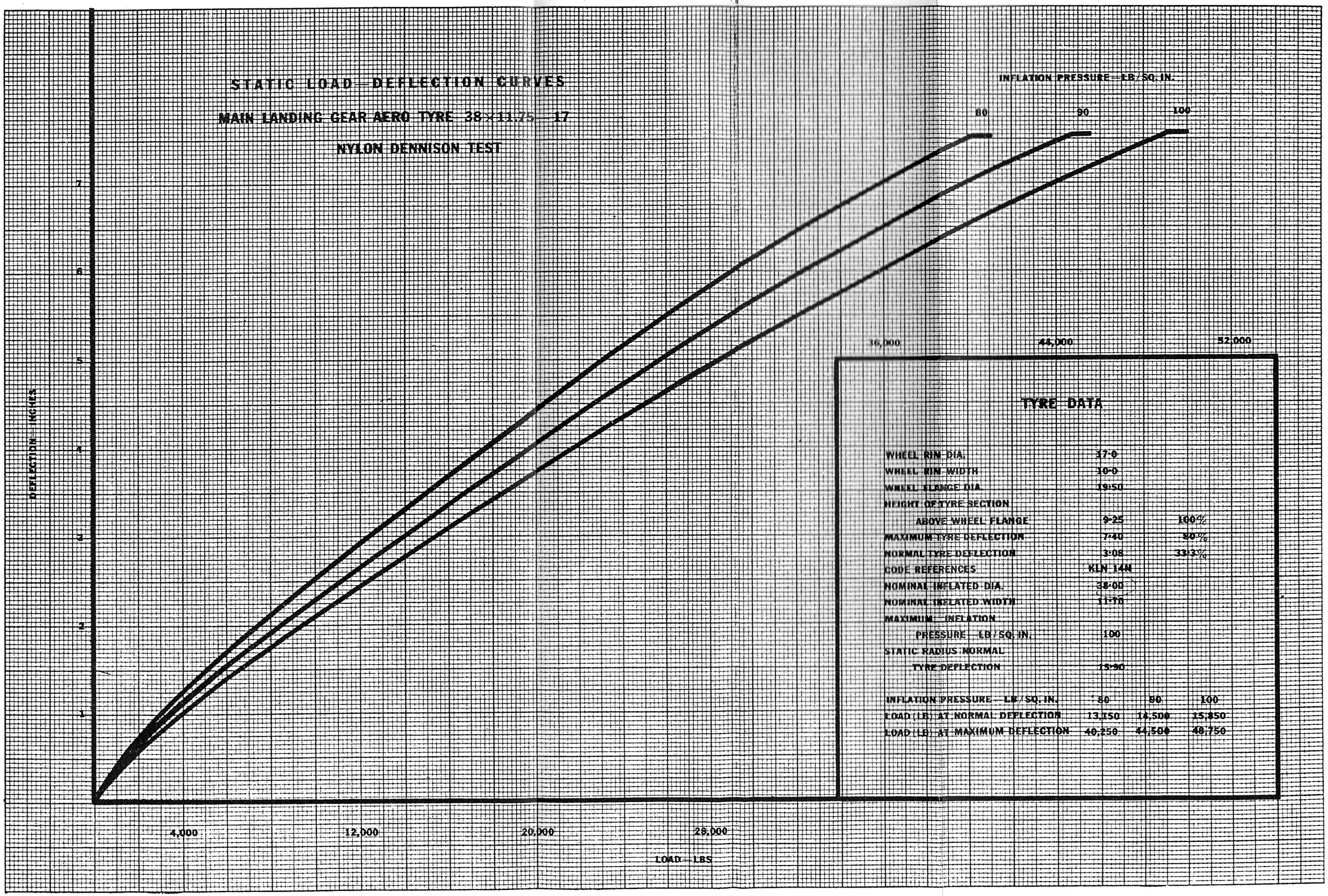
INFLATION PRESSURE — LB/SQ.IN.	70	80	90	100	110
LOAD (LB) AT NORMAL DEFLECTION	4900	5550	6200 →	MAX. STATIC LOAD	
LOAD (LB) AT MAXIMUM DEFLECTION	14700	16150	17600	19050	20500

*PERMISSABLE ONLY TO SATISFY DYNAMIC LOADING CONDITIONS

STATIC LOAD—DEFLECTION CURVES

MAIN LANDING GEAR AERO TYRE 38×11.75—17

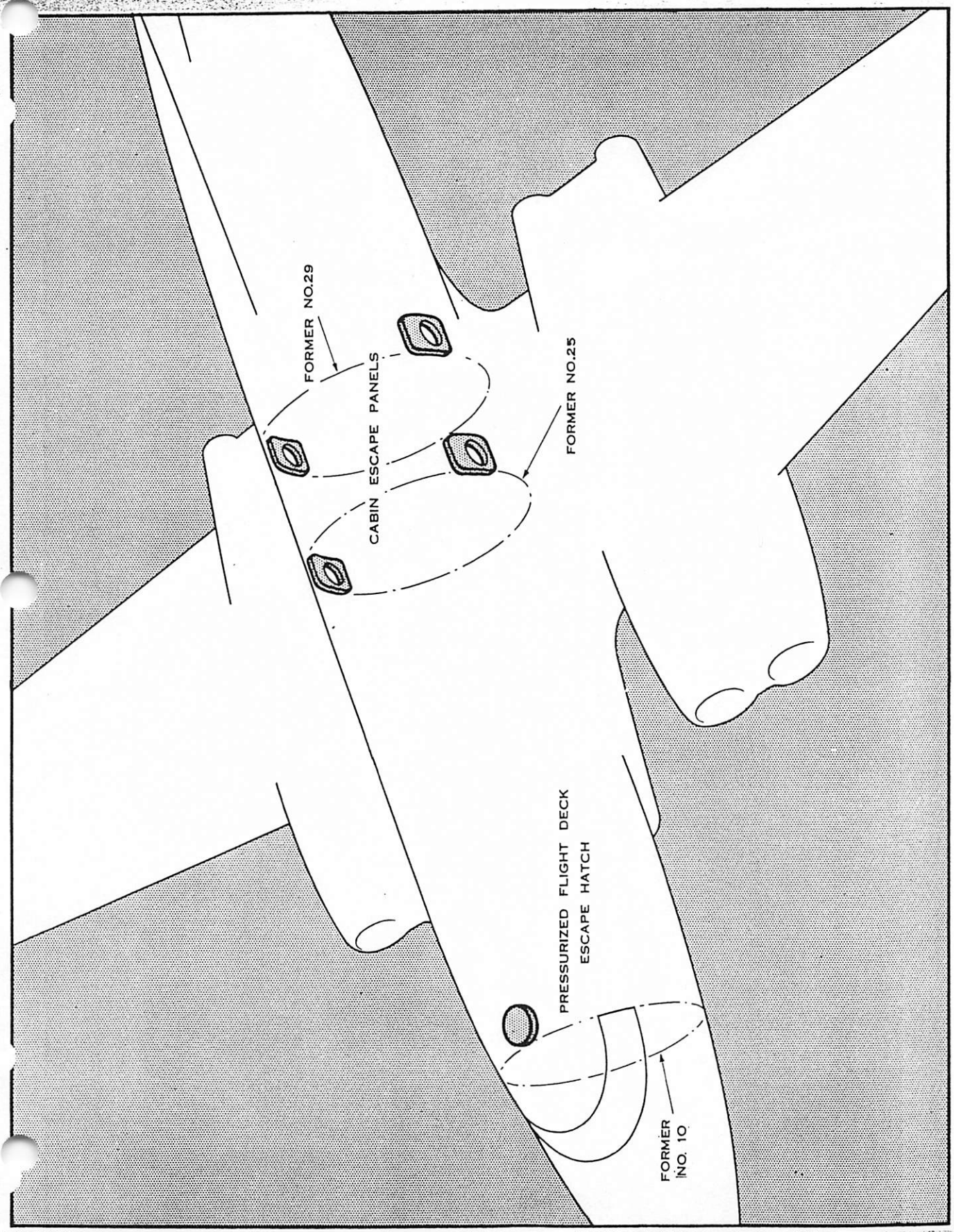
NYLON DENNISON TEST



TYRE DATA

WHEEL RIM DIA.	17·0		
WHEEL RIM WIDTH	10·0		
WHEEL FLANGE DIA.	19·50		
HEIGHT OF TYRE SECTION ABOVE WHEEL FLANGE	9·25	100%	
MAXIMUM TYRE DEFLECTION	7·40	80%	
NORMAL TYRE DEFLECTION	3·08	33·3%	
CODE REFERENCES	KLN 14M		
NOMINAL INFLATED DIA.	38·00		
NOMINAL INFLATED WIDTH	11·70		
MAXIMUM INFLATION PRESSURE LB/SQ.IN.	100		
STATIC RADIUS NORMAL TYRE DEFLECTION	15·90		

INFLATION PRESSURE—LB/SQ.IN.	80	90	100
LOAD (LB) AT NORMAL DEFLECTION	13,150	14,500	15,850
LOAD (LB) AT MAXIMUM DEFLECTION	40,250	44,500	48,750


FORMER NO.29
CABIN ESCAPE PANELS
FORMER NO.25
PRESSURIZED FLIGHT DECK
ESCAPE HATCH
FORMER NO. 10

317-G9

5-8 EMERGENCY OPERATION OF HYDRAULIC SERVICES:

(1) A separate hydraulic emergency system (Fig. 22) operated by an electrically driven power pack feeds through a separate piping system to actuate the various services in the following manner:

(a) Landing gear extension

(b) Flap extension and retraction

(c) Operation of aileron booster

(d) Operation of brakes

NOTE

(1) The following sub-paragraphs outline probable causes of normal system failure and the methods adopted to maintain the various services in operation.

(2) LINE FAILURE: Would become apparent by loss of indicated pressure and a resultant drop in reservoir fluid level. When this condition occurs, carry out the following procedure:

(a) Select EMERGENCY POWER PACK switch - ON

NOTE

(1) Operation of the landing gear and flaps is controlled by the relative switches housed in the emergency power pack control box. The switches are protected by means of a flap hinged to the outer face of the box.

(2) Automatic operation of brake and aileron boost services are maintained without further selection.

(3) PUMP FAILURE: Failure of one pump will give no indication of pressure loss. The only indication will be a greater time lag in the operation of services. Should BOTH hydraulic pumps fail under normal flying conditions, a slow drop in indicated pressure (due to continual operation of the aileron booster) will become apparent. A rapid decrease in pressure will take place immediately any major selection is made. Should either of these conditions become apparent, adopt the following procedure:

(a) Select EMERGENCY POWER PACK switch - ON

NOTE

(1) During the change-over period between NORMAL and EMERGENCY selection, pressure to the brake and aileron boost services will be maintained by the pressure accumulators. This pressure is sufficient for a given number of operations only.

(4) HAND PUMP OPERATION: In the unlikely event of both NORMAL and EMERGENCY pressure pump failure, pressure can be built up by operating the hand pump located in the accessories compartment. The hand pump is capable of supplying fluid pressure through both the NORMAL or EMERGENCY circuits. When this condition exists, carry out the following procedure:

(a) To pressurize the NORMAL system, place the rotary control valve selector in the NORMAL position and operate pump to maintain pressure.

(b) To pressurize the EMERGENCY system, place the rotary control valve in the EMERGENCY position and operate pump to maintain pressure.

NOTE

(1) The rotary control valve is located adjacent to the hand pump in the accessories compartment. The valve is normally wire locked in the EMERGENCY position. The hand pump handle is stowed on the forward wall of the hydraulic compartment.

5-9 EMERGENCY BRAKING:

(1) Place the EMERGENCY POWER PACK switch in the ON position. This action supplies hydraulic fluid to the emergency brake lines and valves. This switch should be 'ON' at all times during taxying, take-off and landing.

NOTE

(1) In the event of NORMAL hydraulic system failure, the EMERGENCY system will automatically supply the necessary pressure for efficient brake operation.

5-10 EMERGENCY LOWERING OF LANDING GEAR: (Electro-Hydraulic)

(1) The emergency landing gear DOWN selector switch (Fig. 111) is located on the flight deck pedestal and provides control for the lowering of the landing gear through the separate emergency hydraulic system pipe lines.

(2) The service is operated by switching ON the adjacent power pack switch and by placing the landing gear selector switch in the DOWN position.

5-11 EMERGENCY LOWERING OF LANDING GEAR: (Manual Release)

(1) In the event of complete hydraulic and electrical failure, the landing gear may be lowered through the combination of manual release and gravity extension. The system is controlled by two operating handles (Fig. 2).

(2) The NOSE GEAR release handle is located on the flight deck floor, outboard of the co-pilot's seat.

(3) The MAIN LANDING GEAR release lever is located aft of the rear spar in the fuselage centre section below floor level. Access to this lever is gained through the centre floor panel.

(4) Both the nose and main landing gears are operated by pulling the relative operating handles to the full extent of their travel.

5-12 EMERGENCY LANDING WITH LANDING GEAR RETRACTED:

(1) Every effort should be made to lower the nose landing gear. This will reduce damage to a minimum, especially in the regions of the engines, nacelle, nose and fuselage centre section.

(2) Make a NORMAL approach.

(3) Use full flap DOWN setting.

(4) When committed to landing, shut off the L.P. fuel cocks, then the electrical MASTER SWITCH.

(5) Land in a nose up attitude. Do not allow the aircraft to stall and drop to the ground.

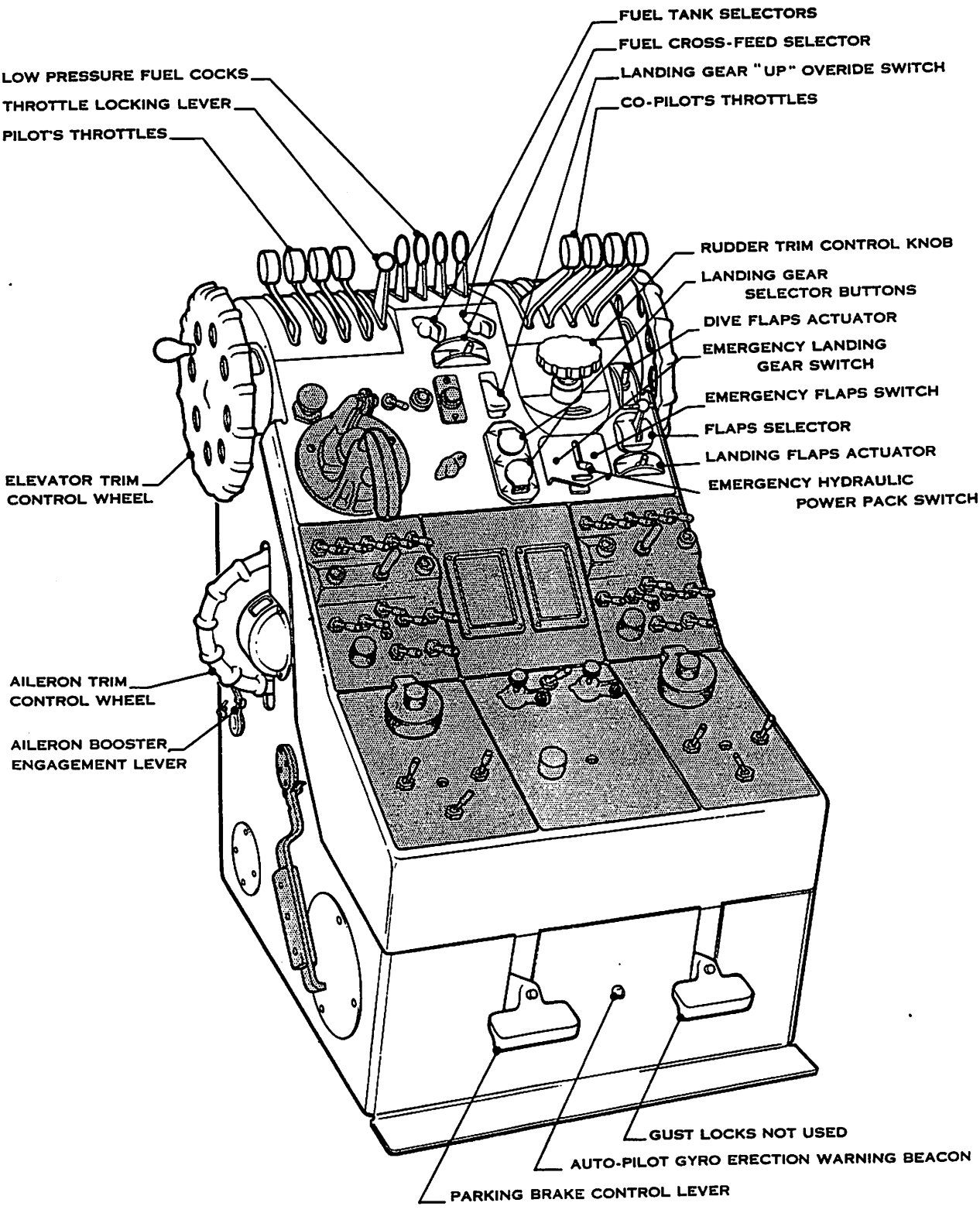
(6) Select CRASH switch ON when sliding to a stop. This provides complete protection against fires in protected areas.

5-13 EMERGENCY FLAP OPERATION:

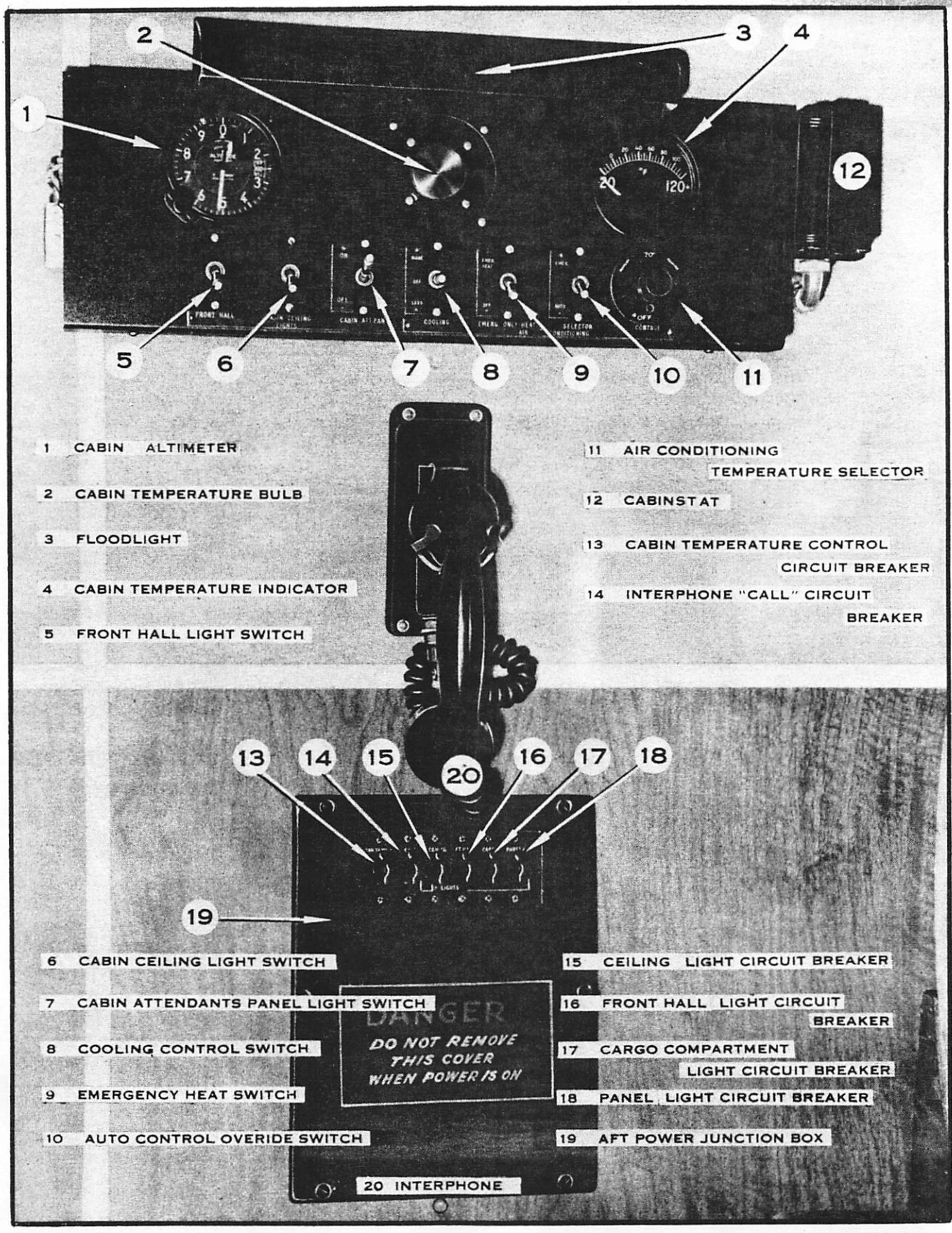
(1) An emergency spring-loaded momentary contact switch having an UP and DOWN selection and an OFF position, provides for direct flap control. This switch is located on the flight deck control pedestal (Fig. 111).

(2) To operate the system, assuming the emergency electrical power pack is switched ON, hold the switch in the UP or DOWN position until the desired flap setting is reached. Flaps will then remain at the selected setting until such time as a further selection is made.

FLIGHT CONTROLS ON THROTTLE CONTROL PEDESTAL
FIG. 111



EM. CABIN HEAT CONTROLS
FIG. 112



5-14 EMERGENCY MANUAL CONTROL OF CABIN HEATING:

(1) If it becomes necessary to override the automatic heating controls, the following procedure should be carried out:

(a) Place the automatic switch in the EMERGENCY position. This will eliminate the automatic control of the system and allow for manual setting.

(b) To provide heat in excess of 68 degrees F., the emergency heat switch should be CLOSED until cabin temperature reaches 70 degrees F.

(c) Should the outside ambient change, and 'more' or 'less' cooling is required, the manual selector switch should be placed in the MORE or LESS cooling position.

(2) The above operation is controlled from the attendant's panel in the rear fuselage section (Fig. 10).

5-15 CABIN PRESSURE FAILURE:

(1) In the event of a cabin blow out, descend to a safe altitude as rapidly as possible.

5-16 EMERGENCY DEPRESSURIZATION:

(1) Operate DUMP VALVE switch on the co-pilot's air conditioning panel.

NOTE

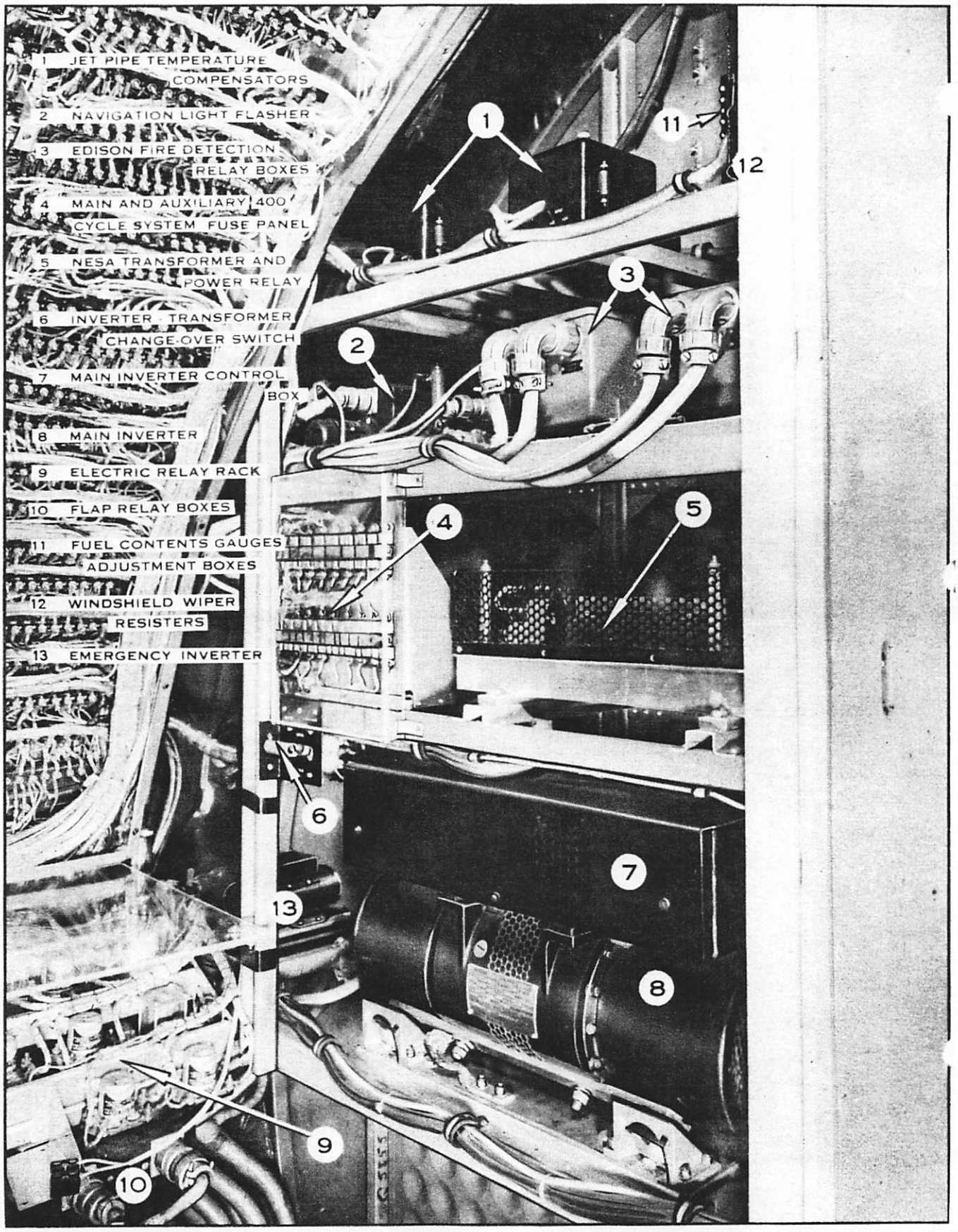
(1) This switch opens the spill valves, safety and relief valves of the pressurization system.

(2) Approximately 30 to 40 seconds is required to relieve the cabin of 4.5 p.s.i. pressure differential.

5-17 CIRCUIT BREAKERS:

(1) All circuit breakers are of the non-trip-free (overriding) type. They may be held in the SET position in an EMERGENCY. This should not become common practice, as it may result in damage to the equipment involved, or in danger to the aircraft.

EM. ELECTRICAL EQUIPMENT (ACCESS. COMPT.) FIG. 113



5-18 GENERATOR FAILURE:

(1) If there is no indication on one ammeter and the other reads 'normal' carry out the following checks:

(a) Check generator voltage, one against the other. Both should read approximately 28 volts.

(b) Check the generator switch. It should be ON. Turn switch OFF and check the reading of the other ammeter. If the reading increases, the trouble is in the ammeter. Turn generator switch ON, and leave on.

(c) Should generator voltage read zero, check the field circuit breakers on the generator panel. If a circuit breaker has tripped, RESET. Should the circuit breaker trip a second time, leave it tripped and turn the generator switch to OFF.

(d) If a generator supply exceeds 30 volts, switch generator OFF and allow to cool. When the generator is sufficiently cooled, switch ON and reduce voltage output by adjusting the relevant load centre control unit in the accessory compartment (Fig. 38).

NOTE

(1) If off-scale high voltage is indicated, do not switch generator back to the ON position.

5-19 SINGLE GENERATOR OPERATION:

(1) If single generator operation is necessary, and indicated amperage output is excessive, the DC power in use must be reduced. This is accomplished by isolating non-essential services.

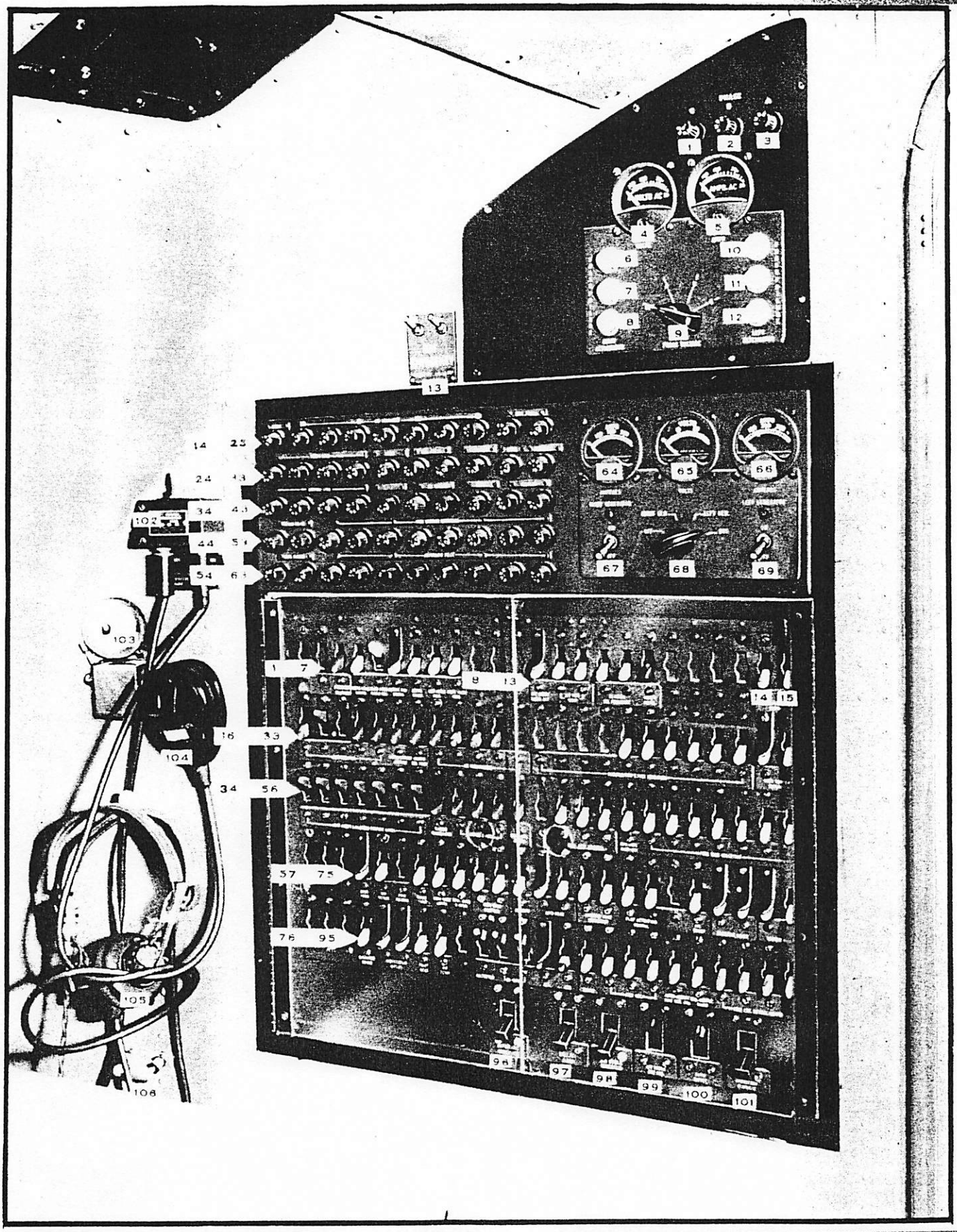
(2) Should a generator be allowed to operate under over-load conditions for a period of time, it will eventually overheat. This condition will be indicated by the illumination of the 'over temperature warning light' located on the flight deck roof control panel.

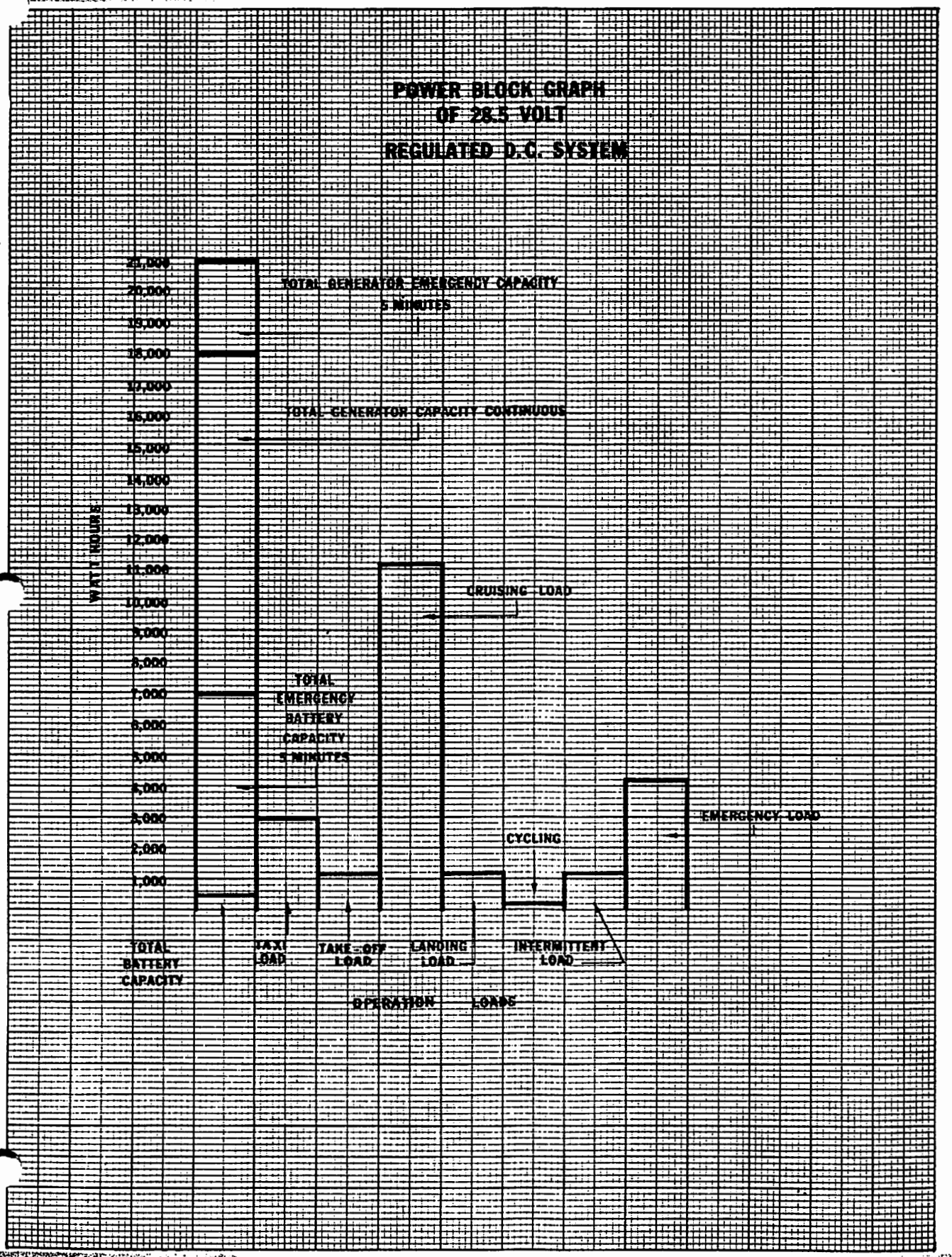
5-20 50 KW ALTERNATORS:

(1) Six warning lamps on the alternator control panel (Fig. 114) indicate power failure on the load side of the alternator limiters.

(2) In the event of right or left alternator failure, the ALT. SEL. switch should be re-selected to allow the serviceable alternator to supply power to the circuits. The system is so arranged, that only one alternator supplies power to the circuits at any time.

GENERATOR & ALTERNATOR
CONTROL PANELS
FIG. 114




POWER BLOCK GRAPH
OF 28.5 VOLT
REGULATED D.C. SYSTEM
WATT-HOURS
20,000
19,000
18,000
17,000
16,000
15,000
14,000
13,000
12,000
11,000
10,000
9,000
8,000
7,000
6,000
5,000
4,000
3,000
2,000
1,000
TOTAL GENERATOR EMERGENCY CAPACITY
5 MINUTES
TOTAL GENERATOR CAPACITY CONTINUOUS
CRUISING LOAD
TOTAL
EMERGENCY
BATTERY
CAPACITY
5 MINUTES
EMERGENCY LOAD
CYCLING
TOTAL
BATTERY
CAPACITY
TAXI
LOAD
TAKE-OFF
LOAD
LANDING
LOAD
INTERMITTENT
LOAD
OPERATION
LOADS

5-21 EMERGENCY A.C. POWER:

(1) An emergency stand by inverter is provided to supply AC power to the electrically operated flight instruments (see 400 cycle load distribution chart, paragraph 1-100) if the main inverter fails. The No. 2 inverter control switch may be placed in the ON position after the main inverter fails. Power to the emergency inverter is supplied direct from the main bus, thus making AC power available, even though the normal generator supply has failed. If a decision is made to extend the period of flight while operating on the emergency inverter only, all DC loads throughout the aircraft must be reduced to a minimum. This requirement is necessary because of the limited capacity of the batteries. Failure to observe this ruling could result in faulty operation of the flight instruments. For further information on battery capacity refer to the load capacity chart (Fig. 115).

5-22 PREPARATIONS FOR DITCHING:

(1) Where possible, use up most of the fuel supply. This reduces the weight of the aircraft and also the stalling speed. Empty fuel tanks are also a considerable contribution to buoyancy. Be sure, however, that a sufficient quantity of fuel is left available for control of the engines during landing.

(2) Open the panel on the floor of the flight deck and PULL the ditching control ring (Fig. 28). This operation seals the pressurized area of the fuselage by closing the valves in the air conditioning and pressurization systems.

NOTE

(1) This system is normally disconnected to prevent inadvertent operation, but can be made operative if required for over water flights.

(3) Depressurize the cabin.

(4) Jettison loose equipment and as much cargo as possible through the emergency exits.

(5) Close all compartment doors.

(6) Prepare life rafts and portable emergency equipment for use.

5-23 DITCHING PROCEDURES: (General)

(1) The decision to ditch rests solely with the pilot of the aircraft and must be made as soon as it is realized that land cannot be reached, but before all fuel is exhausted. It is very important that some power be available on ditching. The radio procedure in particular must be initiated by sending out an SOS by the spoken expression "MAYDAY" by radiotelephony. Surface contact

with a ship should be obtained immediately, if possible. If the danger passes, the distress signals may be cancelled.

5-24 WINDSPEED, DIRECTION & SURFACE CONDITIONS IN RELATION TO DITCHING:

(1) An understanding of sea conditions is essential in order that the situation may be correctly assessed.

(a) Condition of the Sea: From a height the sea always looks calmer than it actually is. In the absence of any fixed landmark the surface wind speed and direction can only be judged from the waves or the wind lanes on the surface which may be observed from a height. With a calm sea there is little or no wind and it is essential to ditch with the lowest possible airspeed. A calm sea is deceptive in judging height, particularly if the surface is "glossy".

(b) Waves: Waves are irregular, short crested, steep and frequently breaking formations of the water surface. They are created by and always move with the wind except close inshore.

(c) Swell: The swell on an open sea is a regular and undulating movement of the surface with long rounded, widely separated crests caused by past or distant disturbances of the wind. A swell does not necessarily move with the wind and it has no breaking crests. Swells are rarely absent from the open sea. If the wind is blowing across the swell a cross-sea is created with the waves which are moving downwind running on the swell. Under these conditions the pilot should choose an approach along the swell as near as possible into wind.

(2) Wind Speed and Direction:

(a) The roughness of the sea is an indication of the strength of the wind if it has been blowing at the same strength, in the same direction for some time.

(b) The wind will be stronger than the appearance of the sea suggests if it is freshening, blowing off a nearby shore, running with the tide or swell and during heavy rain.

(c) Breaking waves may be due to shallow water and in such circumstances must not be used as a means of assessing wind speed and direction.

(d) Sometimes the effect of wind on the surface of the sea is to produce a series of lines, known as "wind lanes" which appear as alternate strips of light and shade. This is a very reliable indication of surface wind direction.

(e) Wind Indication and Assessment:

A few white crests	10 - 20 mph
Many white crests	20 - 30 mph
Streaks of foam along the water	30 - 40 mph
Spray from the crests	40 - 50 mph

When flying over the sea the pilot should always observe and keep in mind the surface wind speed and direction. If time and circumstances permit, the aircraft should be flown at low altitude to assess conditions prior to ditching.

(f) If there is sufficient wind the waves will break downwind. This can be readily observed from a low height. If there is enough wind to blow the spray off the wave crests, the direction in which the spray blows is a reliable indication of the wind direction.

(g) The smoke from a ship is a useful guide in obtaining wind direction but is only entirely reliable when the wind direction is the same or 180° to the path of the ship's motion.

5-25 DITCHING CHARACTERISTICS:

(1) Short, Moderate or Calm Sea: If the aircraft bounces the control column is to be held hard back (in particular, this occurs if the aircraft has been landed at a very high speed). In the average short sea the tail should touch the crest of a wave, and, as soon as it does, the nose is to be kept as high as possible. This will normally cause the fuselage forebody to touch down approximately under the C.G. on the next wave crest.

(2) Contact with Water: If the aircraft alights tail down in the three point attitude (as it should) there will be, initially, a slight impact as the tail strikes. This will be followed by a severe impact with violent deceleration in most cases. If the alighting has been carried out correctly, nose burying will be minimized and the structure will not tend to collapse.

(3) Use of Flaps: The flaps are to be lowered to reduce the speed at which the aircraft can approach and touch down. It is preferable to use only a medium setting, e.g., not over 30°, to avoid too steep an approach. A steep nose-down descent is dangerous if the sea is met too soon, and more height is required for levelling off from such an attitude.

(4) Landing Gear: The landing gear is to be in the fully "UP" position.

(5) Approach Speed: If symetrical power is not available, the normal

glide approach speed is to be used. This will insure control and permit some margin of speed after levelling off; assisting the pilot to choose the best point for ditching on the swell.

(6) Touch Down: In keeping with circumstances and selection of touch down point on the swell, the pilot will hold off until he reaches stalling speed before striking the sea in a three point landing attitude. The approach is to be made along the swell as near into wind as possible unless there is a very strong cross wind. The best point for ditching is on the upslope of an oncoming swell top. In a very high wind condition, regardless of the distance between the wave crests or their size, an into wind ditching is preferable because the impact speed is greatly reduced by reason of the high wind.

(7) Across Wind Approach Along Swell: Approach the sea parallel to the troughs. Eliminate drift by side-slipping.

(8) Use of Engines: If power is available from all four engines it is to be used to secure the flattest possible approach and the slowest possible touch down speed. With one engine inoperative use symmetrical power. If two engines are available on one side of the aircraft use one engine only. Use of both engines on one side generally makes control of the aircraft too difficult for successful ditching, but this is not a hard and fast rule and the pilot should use his own discretion.

(9) Cargo Jettisoning: If possible, all loose equipment and cargo is to be jettisoned unless there is danger that the aircraft will have to be ditched before the hatches can be closed. In the case of the latter, it is more advisable to retain the cargo.

5-26 POINTS ON NIGHT DITCHING:

(1) All bright lights are to be extinguished as soon as possible. This will help to accustom the eyes of crew and passengers to the outside darkness.

(2) The aneroid altimeter is quite unreliable as an indicator of the aircraft's proximity to the sea and great care should be used in judging height.

(3) When attempting a night landing, use the landing lights. When landing lights are used the surface will not be visible very much before the actual impact. It is advisable, under night conditions, as the aircraft nears the sea, to bring the aircraft into a three point attitude if power is available and maintain a very low rate of descent. In this case, if the surface is not seen at all, or until just before the impact, there will be no danger of flying the aircraft into the water in a nose-down attitude.

(4) When ditching by moonlight, if the direction of approach is not fixed by surface conditions, it is advantageous to ditch in a direction that places the moon 30° on the port bow.

(5) After ditching all available lights should be switched on to facilitate escape and rescue.

CHAPTER 2

MISCELLANEOUS EMERGENCY EQUIPMENT

5-27 FIRE AXE:

(1) Is located in a suitable 'quick release' holder mounted on the outer face of the oxygen crate. The latter is mounted on the cabin floor to the right of the main entrance door.

5-28 FIRST AID KIT:

(1) The first aid kit is located in a box secured to the outboard side of the oxygen crate. The kit is contained in a satchel secured in position by means of a snap fastener.

NOTES

RECOMMENDED FUELS, OILS AND LUBRICANTS FOR AVRO C-102 JETLINER

DESCRIPTION & SPECIFICATIONS	SHELL PRODUCTS	ESSO PRODUCTS
FUEL: U.S. - MIL 5616 BRIT. - D Eng. R.D. 2482	AVIATION TURBO FUEL 1	GRADE J.P. 1 FUEL
WATER-METHANOL: 61 parts by volume inhibited distilled water (60 parts distilled water, 1 part Dromus Oil) 40 parts by volume, de-natured water-methanol to British Spec. D Eng. 2470.	N/A	N/A
ENGINE OIL: U.S. - A.A.F. 3580D BRIT. - D.T.D. 44D	AEROSHELL TURBINE 3	UTILITY OIL D.T.D. 44D
AUXILIARY GEAR BOX OIL: U.S. - not available BRIT. - D.E.D. 2479	AEROSHELL TURBINE 9	AVIATION OIL 57
BEVEL GEAR BOX OIL: U.S. - A.A.F. 3580D BRIT. - D.T.D. 44D	AEROSHELL TURBINE 3	UTILITY OIL D.T.D. 44D