CS-61-049

CONVAIR MODEL 30

OPERATION MANUAL

MODEL 30

GENERAL DYNAMICS | CONVAIR

SAN DIEGO, CALIFORNIA

Published By
CUSTOMER SERVICE DEPARTMENT
Commercial Publications
GENERAL DYNAMICS | CONVAIR
San Diego, California

INTRODUCTION

Sections 1 through 4 comprise the Airplane Flight Manual for the Convair 990, Model 30, Turbojet transport.

Sections 5 through 20, the Operational Manual for the Convair 990, Model 30, have been prepared as a reference guide to be used by the flight crew of the airplane. The Operation Manual is not intended to give detailed coverage for each airplane or each system but does contain sufficient information to explain the function and operation of the equipment. More detailed information regarding the airplane components and systems is available from the various repair and maintenance manuals.

In all cases of variation between the first four sections of the Flight Manual and the sections comprising the Operation Manual, the FAA Approved Airplane Flight Manual shall govern.

LIST OF EFFECTIVE PAGES

SECTION	PAGE	DATE	SECTION	PAGE	DATE	SECTION	PAGE	DATE
	T/P	Oct. 23/61	6	6-38	Oct. 23/61	8	i	Oct. 23/61
	ii	Oct. 23/61	6	6-39	Oct. 23/61		Blank	
	I	Oct. 23/61	6	6-40	Oct. 23/61	8	8-1	Oct. 23/61
	II	Oct. 23/61	7	i	Oct. 23/61	8	8-2	Oct. 23/61
	III	Oct. 23/61	7	ii	Oct. 23/61	8	8-3	Oct. 23/61
	IV	Oct. 23/61	7	7-1	Oct. 23/61	8	8-4	Oct. 23/61
	V	Oct. 23/61	7	7-2	Oct. 23/61	8	8-5	Oct. 23/61
	Blank		7	7-3	Oct. 23/61	8	8-6	Oct. 23/61
5	5-1	Oct. 23/61	7	7-4	Oct. 23/61	8	8-7	Oct. 23/61
	Blank		7	7-5	Oct. 23/61	8	8-8	Oct. 23/61
6	i	Oct. 23/61	7	7-6	Oct. 23/61	8	8-9	Oct. 23/61
6	ii	Oct. 23/61	7	7-7	Oct. 23/61	8	8-10	Oct. 23/61
6	6-1	Oct. 23/61	7	7-8	Oct. 23/61	9	i	Oct. 23/61
6	6-2	Oct. 23/61	7	7-9	Oct. 23/61		Blank	
6	6-3	Oct. 23/61	7	7-10	Oct. 23/61	9	9-1	Oct. 23/61
6	6-4	Oct. 23/61	7	7-11	Oct. 23/61	9	9-2	Oct. 23/61
6	6-5	Oct. 23/61	7	7-12	Oct. 23/61	9	9-3	Oct. 23/61
6	6-6	Oct. 23/61	7	7-13	Oct. 23/61	9	9-4	Oct. 23/61
6	6-7	Oct. 23/61	7	7-14	Oct. 23/61	9	9-5	Oct. 23/61
6	6-8	Oct. 23/61	7	7-15	Oct. 23/61	9	9-6	Oct. 23/61
6	6-9	Oct. 23/61	7	7-16	Oct. 23/61	9	9-7	Oct. 23/61
6	6-10	Oct. 23/61	7	7-17	Oct. 23/61	9	9-8	Oct. 23/61
6	6-11	Oct. 23/61		Blank		9	9-9	Oct. 23/61
6	6-12	Oct. 23/61	7	7-19	Oct. 23/61	9	9-10	Oct. 23/61
6	6-13	Oct. 23/61	7	7-20	Oct. 23/61	9	9-11	Oct. 23/61
6	6-14	Oct. 23/61	7	7-21	Oct. 23/61	9	9-12	Oct. 23/61
6	6-15	Oct. 23/61	7	7-22	Oct. 23/61	9	9-13	Oct. 23/61
6	6-16	Oct. 23/61	7	7-23	Oct. 23/61	9	9-14	Oct. 23/61
6	6-17	Oct. 23/61	7	7-24	Oct. 23/61	9	9-15	Oct. 23/61
6	6-18	Oct. 23/61	7	7-25	Oct. 23/61	9	9-16	Oct. 23/61
6	6-19	Oct. 23/61	7	7-26	Oct. 23/61	9	9-17	Oct. 23/61
6	6-20	Oct. 23/61	7	7-27	Oct. 23/61	9	9-18	Oct. 23/61
6	6-21	Oct. 23/61	7	7-28	Oct. 23/61	10	i	Oct. 23/61
6	6-22	Oct. 23/61	7	7-29	Oct. 23/61	10	ii	Oct. 23/61
6	6-23	Oct. 23/61	7	7-30	Oct. 23/61	10	10-1	Oct. 23/61
6	6-24	Oct. 23/61	7	7-31	Oct. 23/61	10	10-2	Oct. 23/61
6	6-25	Oct. 23/61	7	7-32	Oct. 23/61	10	10-3	Oct. 23/61
6	6-26	Oct. 23/61	7	7-33	Oct. 23/61		Blank	
6	6-27	Oct. 23/61	7	7-34	Oct. 23/61	10	10-5	Oct. 23/61
6	6-28	Oct. 23/61	7	7-35	Oct. 23/61	10	10-6	Oct. 23/61
6	6-29	Oct. 23/61	7	7-36	Oct. 23/61	10	10-7	Oct. 23/61
6	6-30	Oct. 23/61	7	7-37	Oct. 23/61	10	10-8	Oct. 23/61
6	6-31	Oct. 23/61	7	7-38	Oct. 23/61	10	10-9	Oct. 23/61
6	6-32	Oct. 23/61	7	7-39	Oct. 23/61	10	10-10	Oct. 23/61
6	6-33	Oct. 23/61	7	7-40	Oct. 23/61	10	10-11	Oct. 23/61
6	6-34	Oct. 23/61	7	7-41	Oct. 23/61	10	10-12	Oct. 23/61
6	6-35	Oct. 23/61	7	7-42	Oct. 23/61	10	10-13	Oct. 23/61
6	6-36	Oct. 23/61	7	7-43	Oct. 23/61	10	10-14	Oct. 23/61
6	6-37	Oct. 23/61		Blank		10	10-15	Oct. 23/61

LIST OF EFFECTIVE PAGES (CONT)

SECTION	PAGE	DATE	SECTION	PAGE	DATE	SECTION	PAGE	DATE
10	10-16	Oct. 23/61	13	13-15	Oct. 23/61	14	14-34	Oct. 23/61
10	10-17	Oct. 23/61	13	13-16	Oct. 23/61	14	14-35	Oct. 23/61
10	10-18	Oct. 23/61	13	13-17	Oct. 23/61	14	14-36	Oct. 23/61
10	10-19	Oct. 23/61	13	13-18	Oct. 23/61	14	14-37	Oct. 23/61
10	10-20	Oct. 23/61	13	13-19	Oct. 23/61	14	14-38	Oct. 23/61
10	10-21	Oct. 23/61	13	13-20	Oct. 23/61	14	14-39	Oct. 23/61
10	10-22	Oct. 23/61	13	13-21	Oct. 23/61	14	14-40	Oct. 23/61
10	10-23	Oct. 23/61	13	13-22	Oct. 23/61	14	14-41	Oct. 23/61
10	10-24	Oct. 23/61	13	13-23	Oct. 23/61	14	14-42	Oct. 23/61
10	10-25	Oct. 23/61	13	13-24	Oct. 23/61	14	14-43	Oct. 23/61
10	10-26	Oct. 23/61	14	i	Oct. 23/61	14	14-44	Oct. 23/61
11	i	Oct. 23/61	14	ii	Oct. 23/61	14	14-45	Oct. 23/61
	Blank		14	iii	Oct. 23/61	14	14-46	Oct. 23/61
11	11-1	Oct. 23/61		Blank		14	14-47	Oct. 23/61
11	11-2	Oct. 23/61	14	14-1	Oct. 23/61	14	14-48	Oct. 23/61
11	11-3	Oct. 23/61	14	14-2	Oct. 23/61	14	14-49	Oct. 23/61
11	11-4	Oct. 23/61	14	14-3	Oct. 23/61	14	14-50	Oct. 23/61
11	11-5	Oct. 23/61	14	14-4	Oct. 23/61	14	14-51	Oct. 23/61
11	11-6	Oct. 23/61	14	14-5	Oct. 23/61	14	14-52	Oct. 23/61
11	11-7	Oct. 23/61	14	14-6	Oct. 23/61	14	14-53	Oct. 23/61
	Blank		14	14-7	Oct. 23/61	14	14-54	Oct. 23/61
12	i	Oct. 23/61	14	14-8	Oct. 23/61	14	14-55	Oct. 23/61
	Blank		14	14-9	Oct. 23/61	14	14-56	Oct. 23/61
12	12-1	Oct. 23/61	14	14-10	Oct. 23/61	14	14-57	Oct. 23/61
12	12-2	Oct. 23/61	14	14-11	Oct. 23/61	14	14-58	Oct. 23/61
12	12-3	Oct. 23/61	14	14-12	Oct. 23/61	14	14-59	Oct. 23/61
12	12-4	Oct. 23/61	14	14-13	Oct. 23/61	14	14-60	Oct. 23/61
12	12-5	Oct. 23/61	14	14-14	Oct. 23/61	14	14-61	Oct. 23/61
12	12-6	Oct. 23/61	14	14-15	Oct. 23/61	14	14-62	Oct. 23/61
12	12-7	Oct. 23/61	14	14-16	Oct. 23/61	15	i	Oct. 23/61
12	12-8	Oct. 23/61	14	14-17	Oct. 23/61	15	ii	Oct. 23/61
13	i	Oct. 23/61	14	14-18	Oct. 23/61	15	15-1	Oct. 23/61
13	ii	Oct. 23/61	14	14-19	Oct. 23/61	15	15-2	Oct. 23/61
13	13-1	Oct. 23/61	14	14-20	Oct. 23/61	15	15-3	Oct. 23/61
13	13-2	Oct. 23/61	14	14-21	Oct. 23/61	15	15-4	Oct. 23/61
13	13-3	Oct. 23/61	14	14-22	Oct. 23/61	15	15-5	Oct. 23/61
13	13-4	Oct. 23/61	14	14-23	Oct. 23/61	15	15-6	Oct. 23/61
13	13-5	Oct. 23/61	14	14-24	Oct. 23/61	15	15-7	Oct. 23/61
13	13-6	Oct. 23/61	14	14-25	Oct. 23/61	15	15-8	Oct. 23/61
13	13-7	Oct. 23/61	14	14-26	Oct. 23/61	15	15-9	Oct. 23/61
13	13-8	Oct. 23/61	14	14-27	Oct. 23/61	15	15-10	Oct. 23/61
13	13-9	Oct. 23/61	14	14-28	Oct. 23/61	15	15-11	Oct. 23/61
13	13-10	Oct. 23/61	14	14-29	Oct. 23/61	15	15-12	Oct. 23/61
13	13-11	Oct. 23/61	14	14-30	Oct. 23/61	15	15-13	Oct. 23/61
13	13-12	Oct. 23/61	14	14-31	Oct. 23/61	15	15-14	Oct. 23/61
13	13-13	Oct. 23/61	14	14-32	Oct. 23/61	15	15-15	Oct. 23/61
13	13-14	Oct. 23/61	14	14-33	Oct. 23/61	15	15-16	Oct. 23/61

LIST OF EFFECTIVE PAGES (CONT)

SECTION	PAGE	DATE
15	15-17	Oct. 23/61
15	15-18	Oct. 23/61
15	15-19	Oct. 23/61
15	15-20	Oct. 23/61
15	15-21	Oct. 23/61
15	15-22	Oct. 23/61
16	i	Oct. 23/61
	Blank	
16	16-1	Oct. 23/61
16	16-2	Oct. 23/61
16	16-3	Oct. 23/61
16	16-4	Oct. 23/61
16	16-5	Oct. 23/61
16	16-6	Oct. 23/61
17	i	Oct. 23/61
	Blank	
17	17-1	Oct. 23/61
17	17-2	Oct. 23/61
17	17-3	Oct. 23/61
17	17-4	Oct. 23/61
17	17-5	Oct. 23/61
17	17-6	Oct. 23/61
17	17-7	Oct. 23/61
17	17-8	Oct. 23/61
18	i	Oct. 23/61
18	ii	Oct. 23/61
18	18-1	Oct. 23/61
18	18-2	Oct. 23/61
18	18-3	Oct. 23/61
18	18-4	Oct. 23/61
18	18-5	Oct. 23/61
18	18-6	Oct. 23/61
18	18-7	Oct. 23/61
18	18-8	Oct. 23/61
18	18-9	Oct. 23/61
18	18-10	Oct. 23/61
18	18-11	Oct. 23/61
18	18-12	Oct. 23/61
18	18-13	Oct. 23/61
18	18-14	Oct. 23/61
18	18-15	Oct. 23/61
18	18-16	Oct. 23/61
18	18-17	Oct. 23/61
18	18-18	Oct. 23/61
18	18-19	Oct. 23/61
18	18-20	Oct. 23/61
18	18-21	Oct. 23/61
18	18-22	Oct. 23/61
18	18-23	Oct. 23/61
18	18-24	Oct. 23/61
18	18-25	Oct. 23/61
18	18-26	Oct. 23/61
18	18-27	Oct. 23/61
18	18-28	Oct. 23/61
18	18-29	Oct. 23/61
18	18-30	Oct. 23/61
18	18-31	Oct. 23/61
18	18-32	Oct. 23/61
18	18-33	Oct. 23/61
18	18-34	Oct. 23/61
18	18-35	Oct. 23/61
18	18-36	Oct. 23/61
18	18-37	Oct. 23/61
18	18-38	Oct. 23/61
18	18-39	Oct. 23/61
18	18-40	Oct. 23/61
19	i	Oct. 23/61
	Blank	
19	19-1	Oct. 23/61
19	19-2	Oct. 23/61
19	19-3	Oct. 23/61
19	19-4	Oct. 23/61
19	19-5	Oct. 23/61
19	19-6	Oct. 23/61
19	19-7	Oct. 23/61
19	19-8	Oct. 23/61
20	i	Oct. 23/61
20	ii	Oct. 23/61
20	20-1	Oct. 23/61
20	20-2	Oct. 23/61
20	20-3	Oct. 23/61
20	20-4	Oct. 23/61
20	20-5	Oct. 23/61
20	20-6	Oct. 23/61
20	20-7	Oct. 23/61
20	20-8	Oct. 23/61
20	20-9	Oct. 23/61
20	20-10	Oct. 23/61
20	20-11	Oct. 23/61
20	20-12	Oct. 23/61
20	20-13	Oct. 23/61
20	20-14	Oct. 23/61
20	20-15	Oct. 23/61
20	20-16	Oct. 23/61
20	20-17	Oct. 23/61
20	20-18	Oct. 23/61

GENERAL CONTENTS

SECTION 5....CRUISE PERFORMANCE DATA

SECTION 6....GENERAL AIRPLANE DESCRIPTION

SECTION 7....FLIGHT CONTROLS

SECTION 8....AUTOPILOT

SECTION 9....POWER PLANT

SECTION 10....FUEL SYSTEM

SECTION 11....OIL SYSTEM

SECTION 12....HYDRAULIC SYSTEM

SECTION 13....ELECTRICAL SYSTEM

SECTION 14....AIR CONDITIONING AND PRESSURIZATION SYSTEMS

SECTION 15....BLEED AIR AND ADVERSE WEATHER SYSTEMS

SECTION 16....OXYGEN SYSTEMS

SECTION 17....FIRE PROTECTION

SECTION 18....FLIGHT AND NAVIGATION INSTRUMENTS

SECTION 19....RADIO AND COMMUNICATION SYSTEMS

SECTION 20....MISCELLANEOUS SYSTEMS

5 CRUISE PERFORMANCE

Section 5

CRUISE PERFORMANCE DATA

THIS SECTION IS PROVIDED FOR CUSTOMER CONVENIENCE. NO DATA TO BE INSERTED IN THIS SECTION BY CONVAIR.

6

GENERAL DESCRIPTION

Section 6

GENERAL AIRPLANE DESCRIPTION

TABLE OF CONTENTS

TITLE	PAGE
GENERAL DESCRIPTION	6-1
Crew Members	6-1
Passengers	6-1
Windows	6-1
Doors	6-1
Upholstery and Trim	6-1
Drainage System	6-7
Nacelles and Pylons	6-7
Air Conditioning and Pressurization	6-7
Automatic Flight Control	6-7
Flight Instruments	6-7
Electrical Power Supply	6-7
Flight Controls	6-8
Fuel System	6-8
Hydraulic Power Supply	6-8
Adverse Weather Protection	6-9
Flight Crew Instruments	6-9
Landing Gear	6-9
Oxygen System	6-9
Pneumatics	6-11
Water and Waste Systems	6-11
DIMENSIONS AND AREAS	6-11
Fuselage Station Lines	6-11
Wing Station Lines	6-11
Anti-Shock Body Station Lines	6-11
Engine Pod Station Lines	6-11
Horizontal Stabilizer Station Lines	6-17
Vertical Stabilizer Station Lines	6-17
Water and Buttock Station Lines	6-17
Dimensions	6-17
GENERAL SERVICING DATA	6-22
Servicing Replenishment Chart	6-22
WALKWAYS	6-30
JET BLAST CLEARANCE AREAS	6-30
AIRPLANE GROUND HANDLING PROCEDURES	6-30
Nose Gear Towing	6-30
Main Gear Towing	6-33

TABLE OF CONTENTS (CONT)

TITLE	PAGE
AIRPLANE JACKING PROCEDURES	6-33
Jacking the Entire Airplane	6-33
Jacking at the Landing Gear	6-33
Lifting with Pneumatic Bags	6-39
AIRPLANE MOORING PROVISIONS	6-39
AIRPLANE PARKING	6-39

GENERAL AIRPLANE DESCRIPTION

GENERAL DESCRIPTION

The Convair 990 turbojet transport airplane is a high-speed, medium range, low swept wing jet transport. The airplane is of all metal construction with full cantilever wing and tail surfaces, a pressurized and air conditioned semi-monocoque soundproofed fuselage, four wing-pylon mounted General Electric CJ 805-23 aft fan turbojet engines and a fully retractable tricycle-type landing gear. The engines are equipped with thrust reversing mechanisms. Dimensions are illustrated in Figure 6-1 and the general arrangement in Figure 6-2.

Crew Members

The flight crew consists of pilot, copilot, flight engineer and per various configurations either radio operator/ navigator or additional crew member and observer. The airplane carries up to four cabin Attendants.

Passengers

The airplane seats as many as 113 passengers, depending on the configuration of the cabin area. The seating arrangement, lounge or club areas and the closet space varies with the standard, coach and mixed configurations according to operator requirements.

Windows

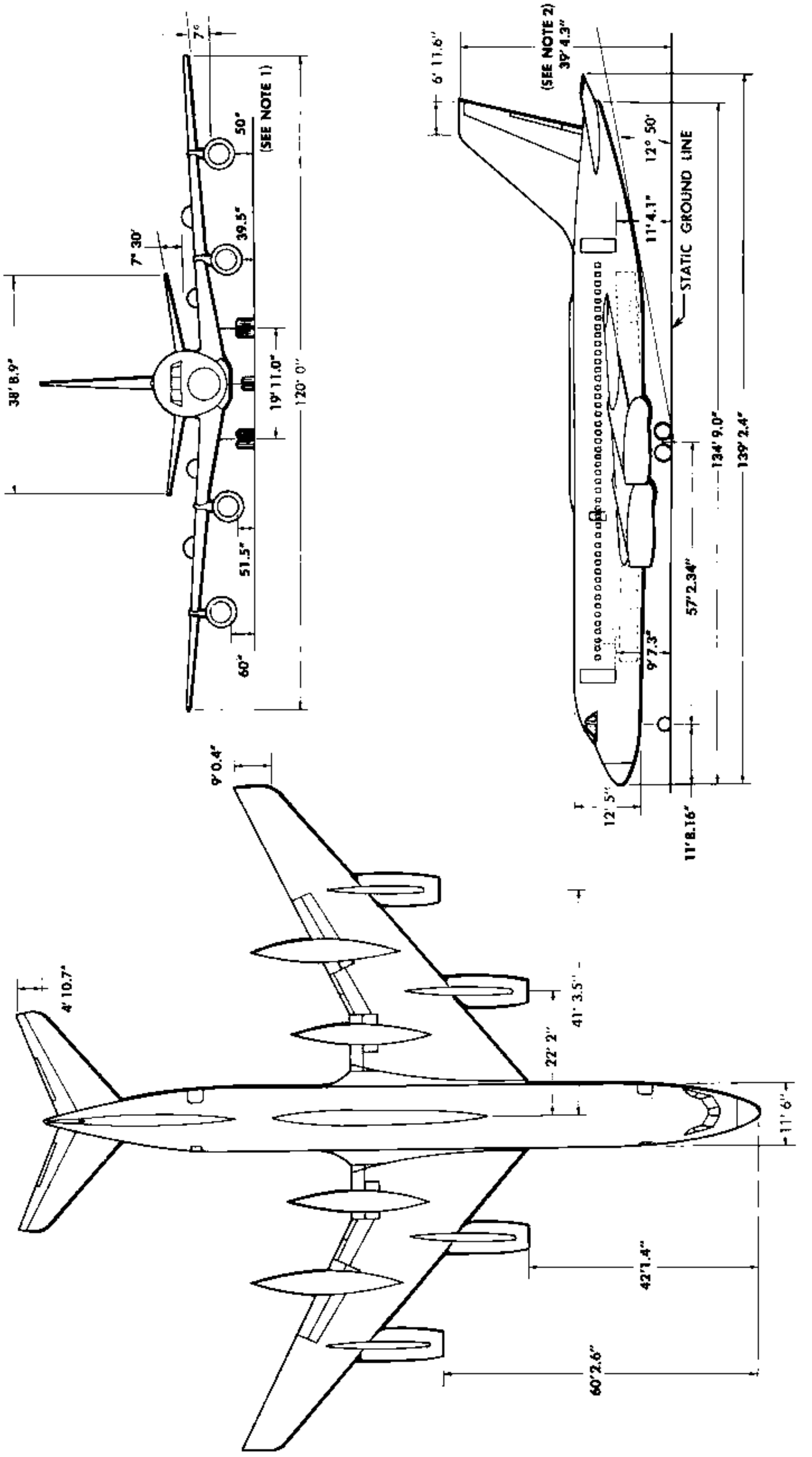
All flight compartment windows are laminated of Polyvinyl Butyral plastic and tempered plate glass. Only the three main windshields are electrically anti-iced; however, all flight compartments are electrically anti-fogged. The anti-fogging function also improves the "bird impact" integrity of the flight compartment windows. Each passenger cabin window incorporates an inner and an outer window for increased blowout protection.

Doors

Two main entrance doors are provided on the left side of the fuselage and two service doors on the right side. The doors are wedge shaped and of the plug type construction. Rotation of either the inside or outside door handle to the open position slides the doors upward and out of the retainer tracks so that they open out and fold against the fuselage for maximum access. Other access doors are located throughout the airplane to provide ease of maintenance of the various systems and components. Door-open warning lights are located on the Flight Engineer's panel. The access doors are shown in Figure 6-3.

Upholstery and Trim

The interior is designed for beauty, comfort and ease of maintenance. The materials and fabrics used in the cabin interior will not readily stain and can be easily cleaned with soap and water. Fire resistant material has been used throughout the airplane.

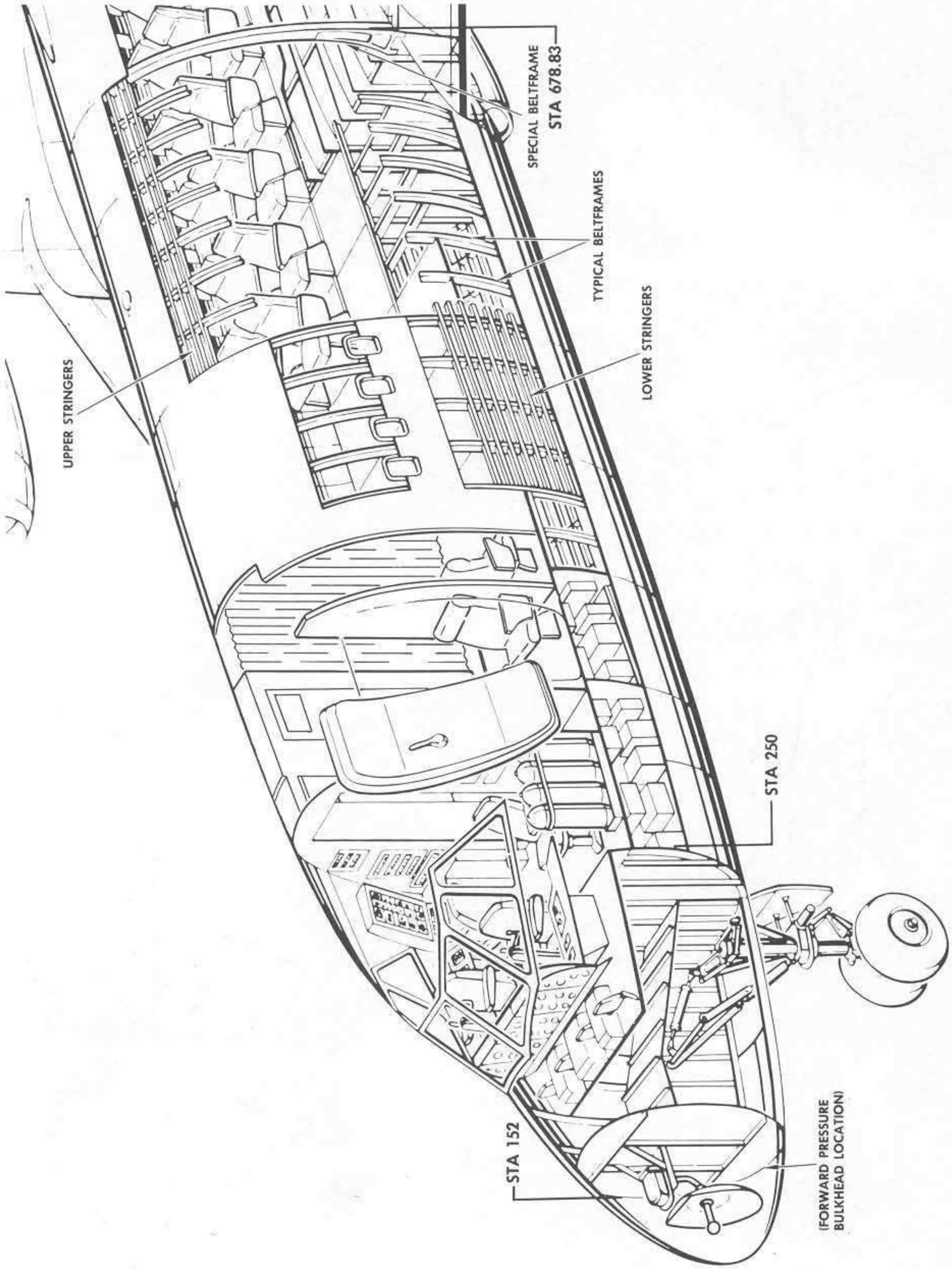


NOTE

1. ENGINE POD CLEARANCE AND AIR INTAKE HEIGHT DIMENSIONS PREVAIL ONLY DURING AIRPLANE MANUFACTURE. FULL FUEL LOADS DEPRESS INBOARD ENGINE PODS 1.5 INCHES AND OUTBOARD ENGINE PODS 8.0 INCHES.
2. VERTICAL STABILIZER HEIGHT DIMENSION SHOWN IS WITH AIRPLANE AT FULL GROSS WEIGHT. THIS HEIGHT WILL INCREASE FROM FOUR TO SIX INCHES WITH AIRPLANE AT EMPTY WEIGHT.

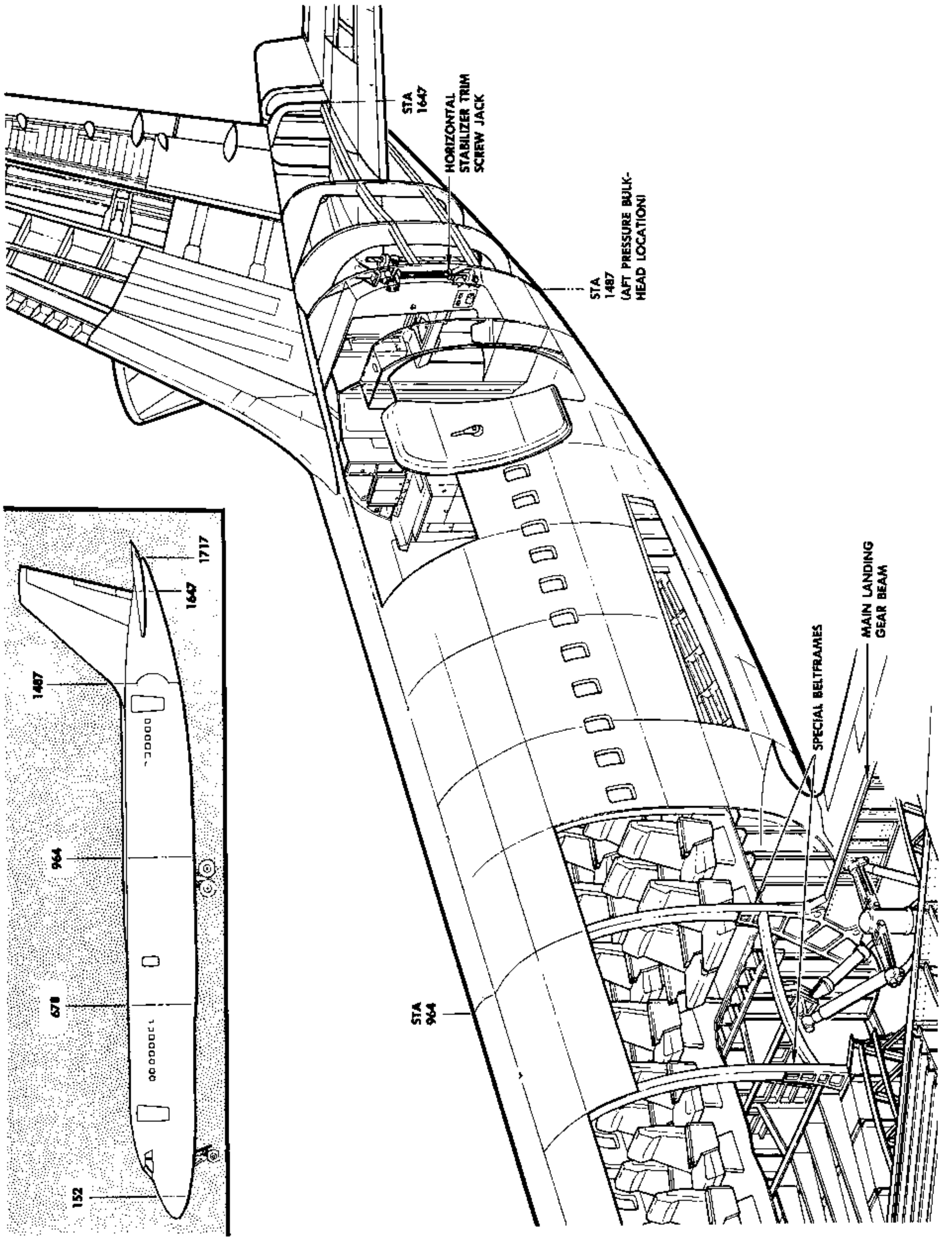
30.30.06.003 B

Airplane Three View
Figure 6-1

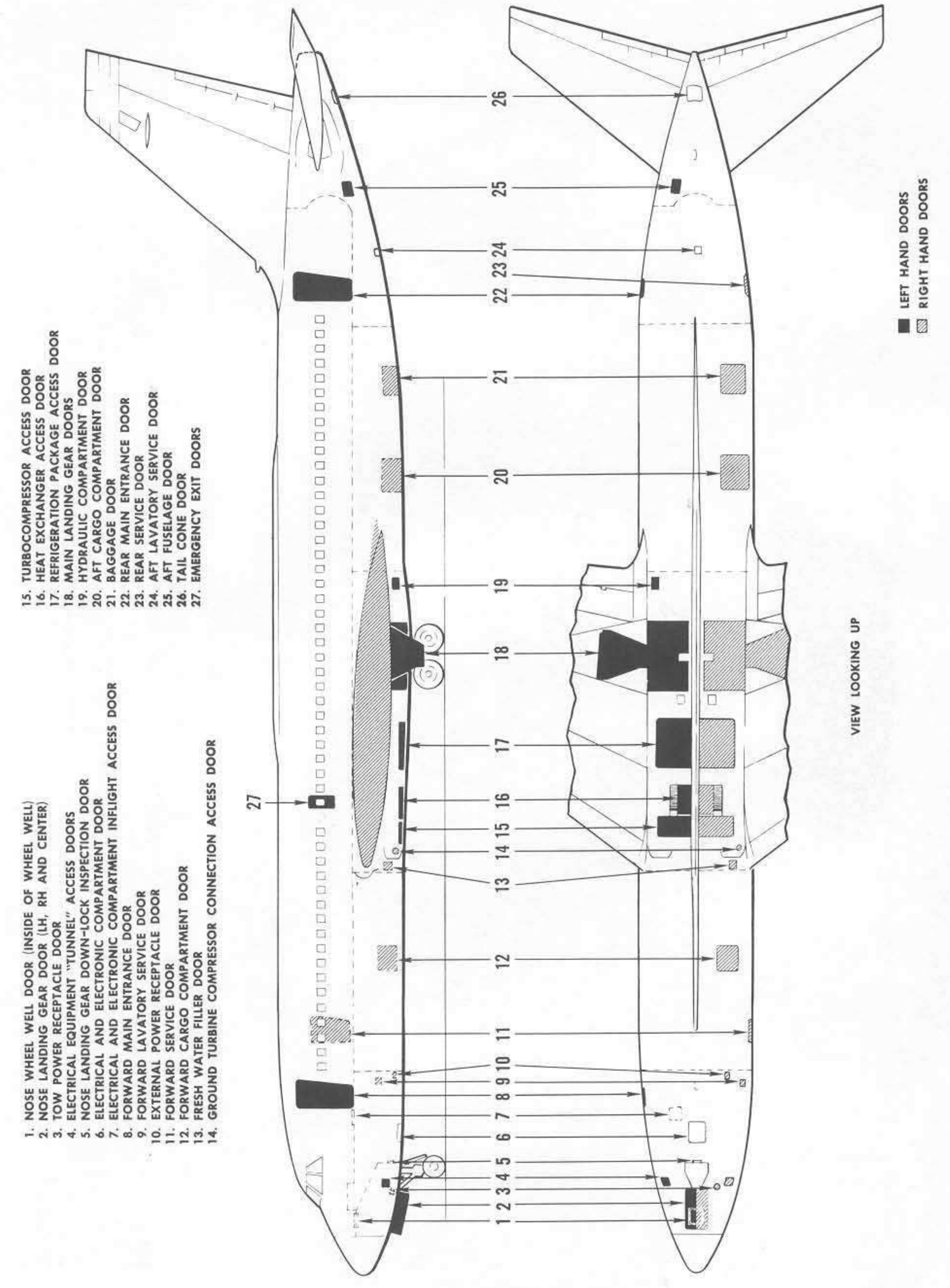


30.30.51.003-1

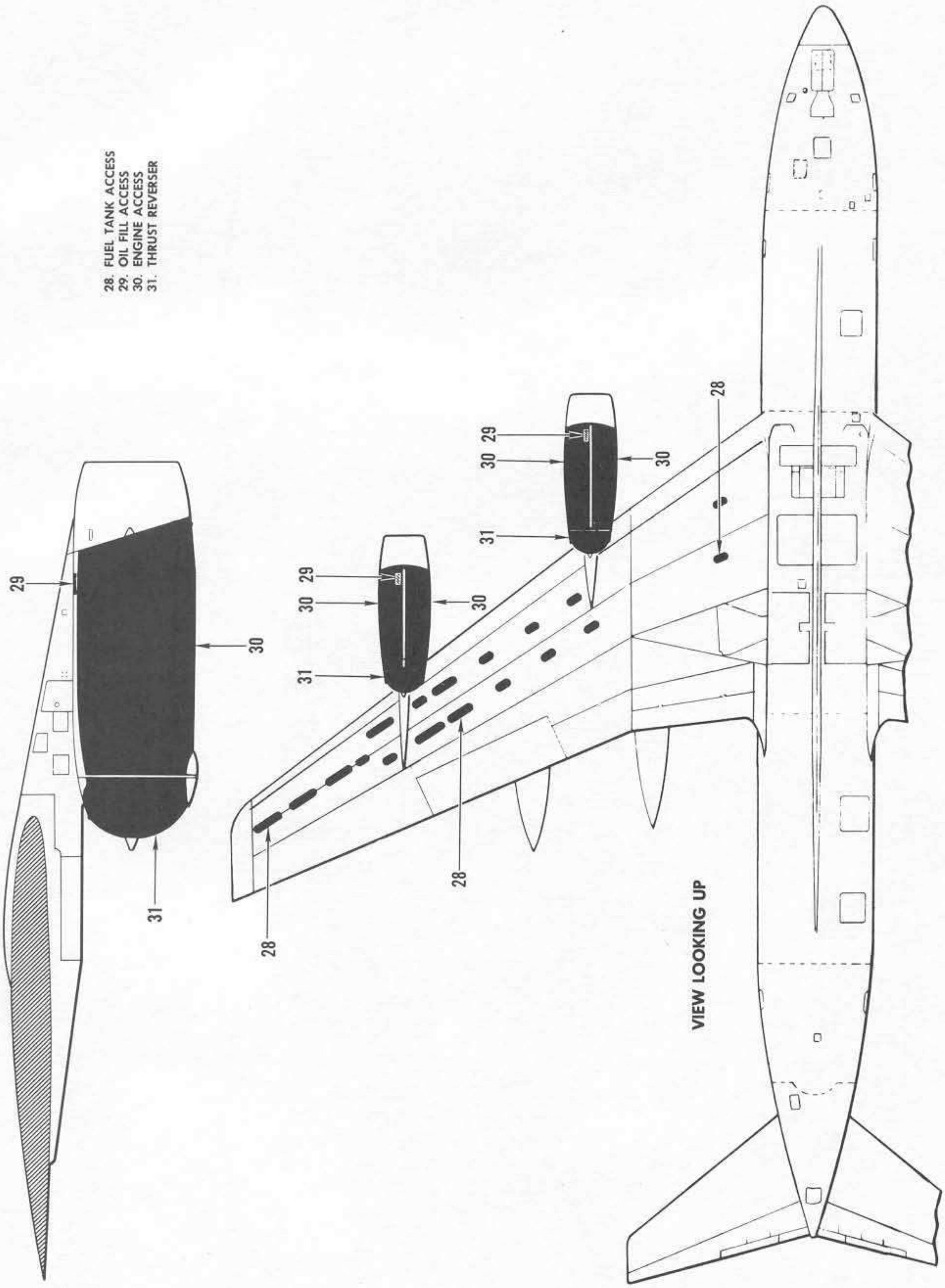
General Airplane Arrangement
Figure 6-2 (Sheet 1 of 2)



General Airplane Arrangement
Figure 6-2 (Sheet 2 of 2)



Airplane Access Doors
Figure 6-3 (Sheet 1 of 2)



Airplane Access Doors
Figure 6-3 (Sheet 2 of 2)

Drainage System

A positive draining system is included to drain all areas where water or moisture might collect and also in lines carrying flammable fluids. Drained fluids cannot re-enter the airplane, tailpipe, heating, anti-icing and ventilating ducts, or any other potential fire sources.

Nacelles and Pylons

The four General Electric aft fan turbojet engines are enclosed in pods attached to the lower surface of the wing by means of pylons. The pylons are of aluminum alloy construction with steel fittings at the wing and pod attaching points. The pods are constructed of aluminum alloy, steel, and titanium, depending on the fire potentials in each pod area. Cowl doors provide full accessibility to the engines during maintenance.

Air Conditioning and Pressurization

The air conditioning and pressurization system is designed to supply all occupied compartments with an airflow of 120 pounds per minute at an airplane altitude of 41,000 feet. Circulating air, either heated or cooled, is supplied as required by the temperature control system. The air conditioning system is composed of two independent systems, either of which can furnish adequate air conditioning and pressurization for both the passenger cabin and flight compartment. Heating and cooling of the baggage compartments and the electrical and electronic equipment is also accomplished by the air conditioning system. The pressurization system can maintain a sea level cabin altitude up to an airplane altitude of 21,300 feet, and an 8300 foot cabin altitude up to an airplane altitude of 41,000 feet. The maximum normal cabin differential operation pressure is 8.3 psi. Integral warning systems are provided to indicate any system malfunctions.

Automatic Flight Control

An automatic flight control system is provided consisting of an automatic pilot and flight instrumentation. The system also provides automatic guidance under localizer, glide path, and omnirange radio control. System instruments, displays and controls are integrated for ease of control and simplicity.

Flight Instruments

A Kollsman integrated flight instrument system (KIFIS) incorporates integral scale error correctors in the instruments and compensates for errors due to static pressure variations. Individual systems are provided for the pilot and copilot.

Electrical Power Supply

The electrical power supply system provides adequate power for the operation of all electrical and electronic circuits with a built-in power reserve. Alternating current is supplied by four engine-mounted ac generators driven by constant speed drives. Four transformer-rectifier units supply normal 28-volt dc power to all dc buses when the ac system is energized. A single aircraft type battery supplies emergency dc power to the emergency bus when required. The battery remains on a floating charge when not in use.

Flight Controls

The flight control systems may be considered, basically, as a primary and a secondary group. The primary group includes the power rudder, ailerons, elevators and spoilers; the secondary group is composed of the trim tabs for the rudder and ailerons, the horizontal stabilizer trim system, the leading edge slats, Kreuger flaps, trailing edge flaps and the speed brakes. The elevators and ailerons are aerodynamically operated by flight tabs that are positioned by the use of the control column and control wheel. The rudder is hydraulically operated.

The spoilers serve a dual function, working in conjunction with the ailerons and also as speed brakes. The inboard and outboard spoilers can be split by the flight crew to provide either a "pitch-up" or a "pitch-down" reaction, as required for emergency longitudinal trim. The horizontal stabilizer is built as a single unit and is adjustable for longitudinal trim. Hydraulic gust dampeners are provided on the ailerons and elevators; a gust lock is provided on the rudder.

Fuel System

Five individual fuel tanks in the Convair 990 hold approximately 15,628 U.S. gallons. Four of these tanks are integrated into the wing structure and consist of a main tank, a replenishment tank and an anti-shock-body (ASB) tank. The fifth tank is composed of five fuel tight compartments in the wing center section. Normal operation is main wing tank to engine, however, by cross-feeding, any main wing tank or center section tank may be used to supply fuel to any or all engines. The fuel from the ASB tanks is transferred by electric pumps into the main tanks, and from the main tanks to the engine. The engine fuel control system regulates the amount of fuel in relation to the desired RPM, compressor inlet air pressure and temperature, and acceleration and deceleration rate. The normal refueling procedure utilizes underwing pressure refueling adapters. Automatic selection of any quantities up to full tanks, exclusive of the minimum allowable expansion space, is available. Fuel service openings installed in the replenishment tanks, and accessible from the wing upper surfaces, allow alternate gravity flow refueling. The auxiliary center section tank and the ASB tanks have similar service openings. Drip sticks are provided for ground checking of the fuel quantity on board. A fuel jettison system is provided. Scavenge systems allow complete removal of all fuel in the inboard and center section tanks.

Hydraulic Power Supply

The main hydraulic power supply system is divided into two independent power supply subsystems with a minimum of interconnection. Each is a 3000 psi, closed center, continuously operating system. Each system consists of two variable displacement engine-driven pumps, a reservoir, accumulators and various other components. The two systems are interconnected at the reservoirs to permit single point servicing, however, one system cannot deplete the other below the refill level. Ground service connections for both systems are located in the hydraulics compartment.

Adverse Weather Protection

The anti-icing and de-icing systems utilize bleed air and electrical current for their operation. Bleed air flow provides heat for wing anti-icing (including leading edge slats), engine inlet duct lip, nose cone, inlet guide vanes, and aft fan front frame and outlet guide vanes anti-icing. Pressure is also supplied for windshield rain clearing. The horizontal and vertical stabilizer leading edges are de-iced and the windshields anti-iced and anti-fogged by electrical current. The windshields are constructed of laminated glass with conductive layers for electrical current. Operation of the adverse weather systems is manual.

Flight Crew Instruments

Instruments have been arranged on a "need" basis for each of the crew members (see Figure 6-4). The pilot and copilot are provided with duplicate flight instruments and various other instruments to suit their particular requirements. Engine instruments are grouped on a center panel and are readily visible to either pilot. Each pilot has access to the overhead switch panel and individual console panels which contain various instruments and controls. The control panel for the flight engineer is divided into sub panels, each of which includes all the instruments and controls for a particular airframe system. Instruments and controls on all panels are so arranged that they can be removed or replaced without disturbing or disconnecting adjacent units.

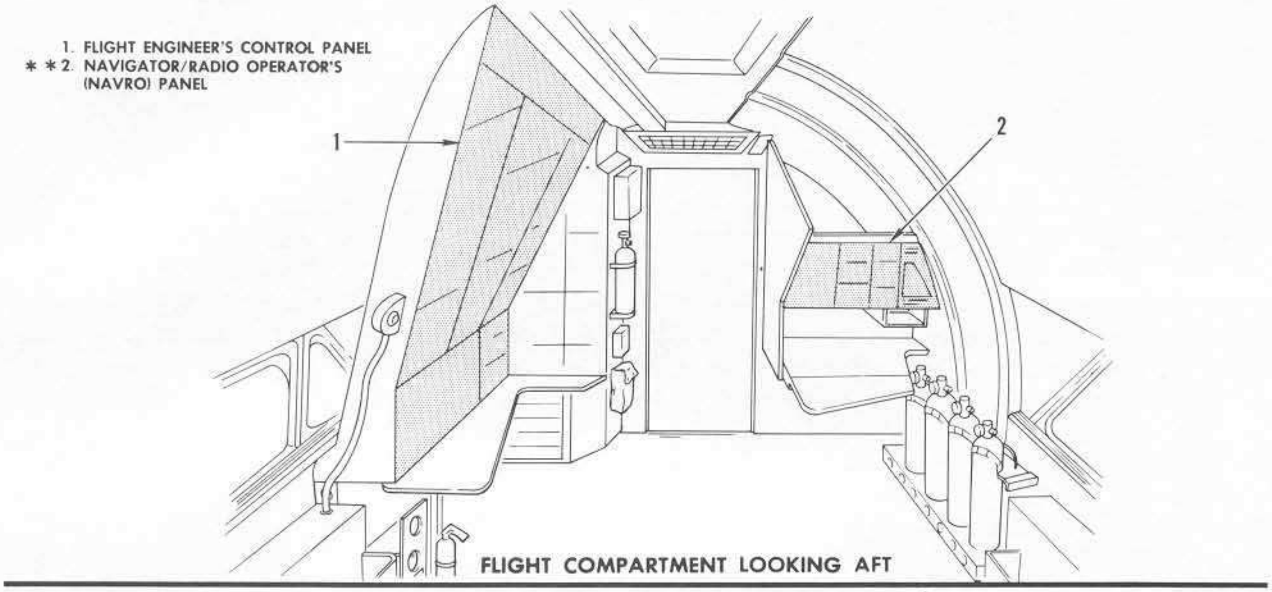
Landing Gear

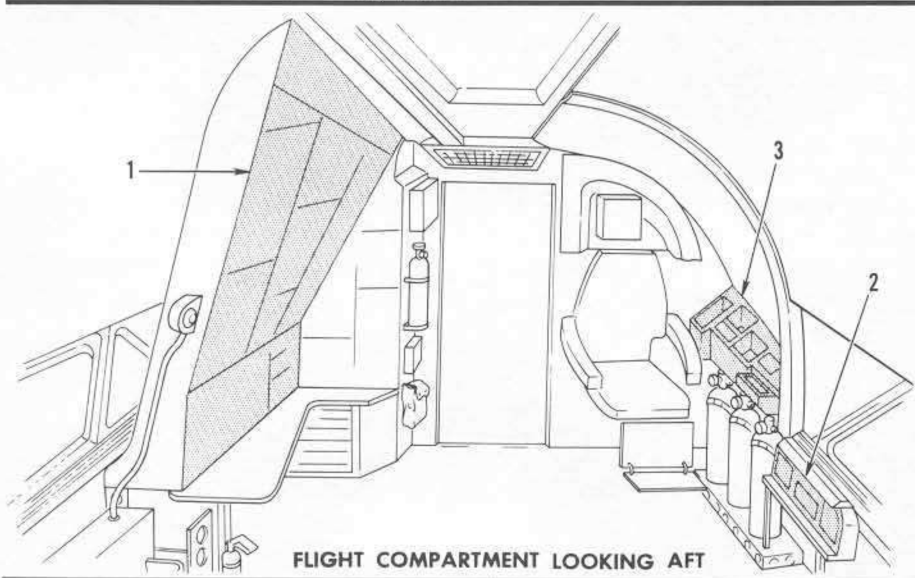
The tricycle type landing gear is fully retractable and consists of a steerable, dual-wheel nose gear and two main gears with tandem-mounted dual wheels. The main gears retract inboard into the fuselage at the wing junction point and the nose gear retracts forward into the nose wheel well. Each gear has an integral self-adjusting brake and anti-skid system. Other features incorporated in the landing gear system include shimmy dampening, attitude positioners, brake equalizers and tubeless tires. Visual inspection of all landing gear locking mechanism in flight is provided.

Oxygen System

The oxygen system is a high pressure, gaseous type system. Oxygen is supplied from light weight steel cylinders, each having a capacity of 107 cubic feet at a pressure of 1800 psig. The airplane is also equipped with portable oxygen cylinders to augment the main system. The flight crew is supplied by a diluter-demand system, while the passengers and cabin attendants are supplied by a continuous-flow system. Release of oxygen masks and oxygen flow to the passenger compartments is automatic with manual over-ride provisions.

1. FLIGHT ENGINEER'S CONTROL PANEL
* * 2. NAVIGATOR/RADIO OPERATOR'S (NAVRO) PANEL

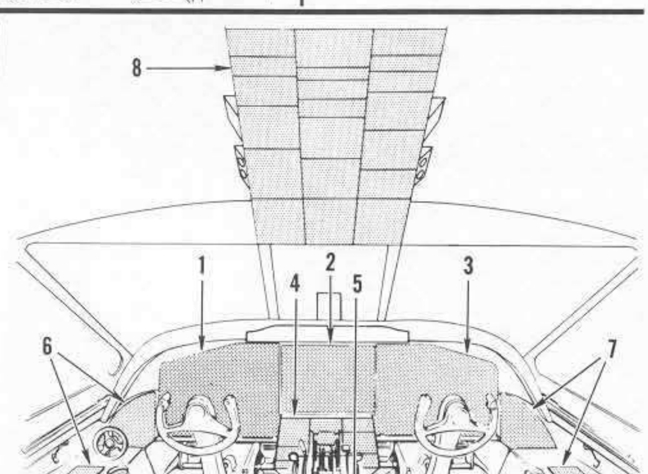




1. FLIGHT ENGINEER'S CONTROL PANEL
* 2. SECOND OFFICER'S PANEL
* 3. OBSERVER'S CONSOLE

EFFECTIVITY	
APPLICABLE TO AIRPLANES	
*	CV1 THROUGH CV25
* *	CV26 AND ON

1. PILOT'S INSTRUMENT PANEL
2. PILOTS' CENTER INSTRUMENT PANEL
3. COPILOT'S INSTRUMENT PANEL
4. PILOTS' FORWARD PEDESTAL PANEL
5. PILOTS' PEDESTAL PANEL
6. PILOT'S AUXILIARY INSTRUMENT AND CONSOLE PANELS
7. COPILOT'S AUXILIARY INSTRUMENT AND CONSOLE PANELS
8. PILOTS' OVERHEAD SWITCH PANEL



30.02.31.057

Flight Compartment Instrument Panels
Figure 6-4

Pneumatics

High pressure air is utilized in the landing gear system to provide an emergency brake system. The No. 2 hydraulic power system accumulator located in the hydraulic compartment serves as an air supply for emergency opening of the main gear doors.

Water and Waste Systems

A fresh water system is provided for use in the buffet and lavatory areas. Water for the system is supplied from a water tank located below the cabin floor. Sump tanks for water waste are located in the lavatory compartments, and draining and filling accomodations are easily accessible for servicing.

DIMENSIONS AND AREAS

Station lines are unit measurements along the longitudinal axis of the fuselage, the lateral axis of the wings and horizontal stabilizer, and the vertical axis of the vertical stabilizer. All station lines are identified by their respective components, such as the fuselage station lines, wing station lines, and are always measured in inches.

Fuselage Station Lines

The foremost station on the fuselage nose is station 100, thus eliminating any negative or minus stations. Fuselage stations progress aft to the tail at station 1717. Figure 6-5 illustrates all the fuselage station lines for the frames, bulkheads, doors, windows, etc., from the tip of the radome to the tail.

Wing Station Lines

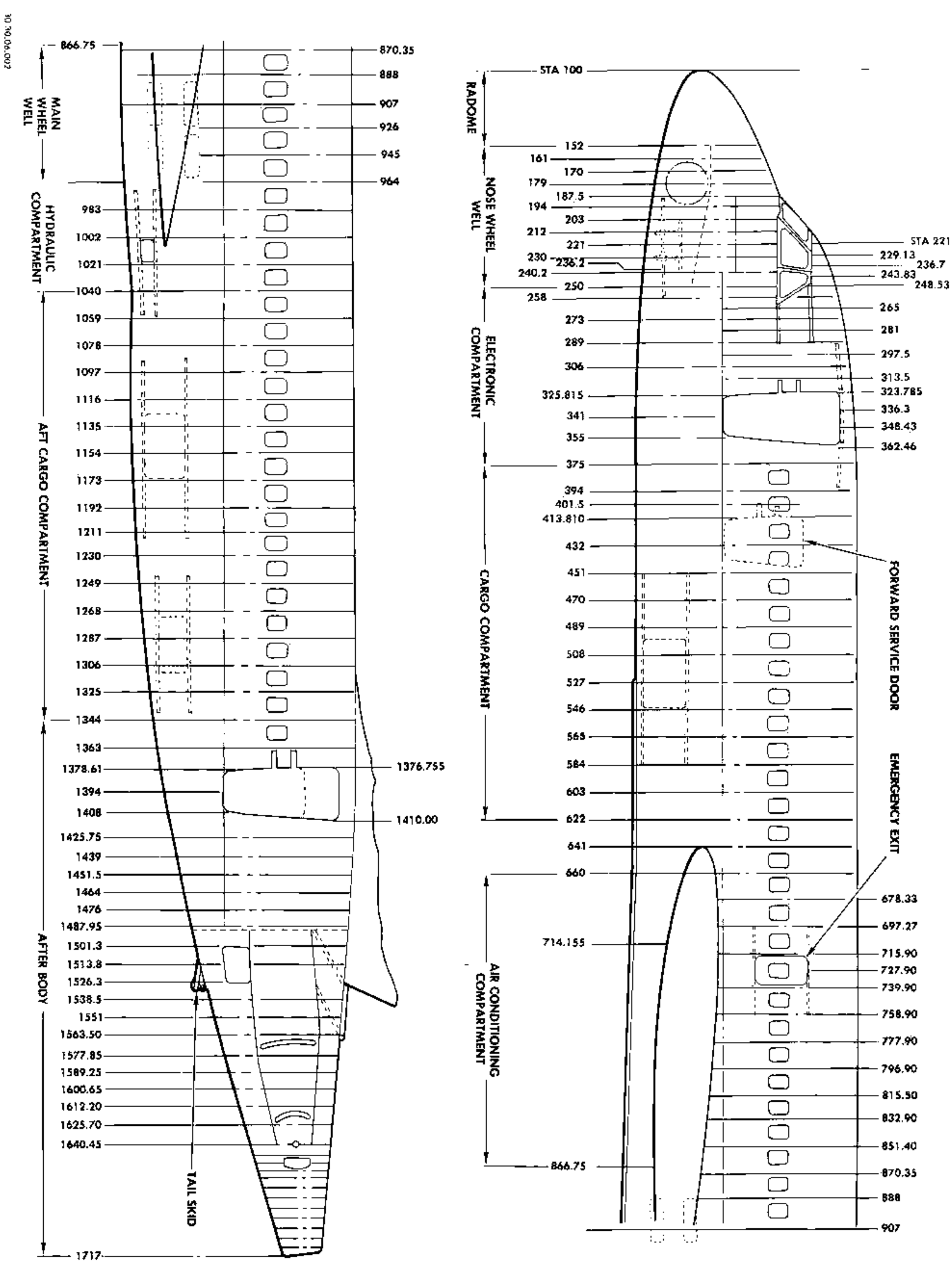
Wing stations are measured in inches from BL 0.0 to provide spanwise, or lateral axis, reference points and are illustrated in Figure 6-6. All wing stations between WS 69 and WS 224, and outboard of WS 688, are parallel to the airplane centerline, while stations 261.413 through 797.540 are perpendicular to the wing center spar. Wing station 69.518 intersects the tariling edge at the break point and is a very useful reference point. In addition to the wing inch-stations, all wing bulkheads and/or ribs are numbered consecutively from 1 through 36 starting at the first wing bulkhead at wing station 69.518 outboard of the fuselage. Any of these numbers can be cross-referenced to a wing station if necessary. If needed, wing stations can be further identified as right-hand or left-hand.

Anti-Shock Body Station Lines

Anti-shock body station lines are measured aft from the theoretical nose tips of the body. The theorectical nose tip is station 0.0 (see Figure 6-7). No individual water line or buttock line system is used for the Anti-Shock Bodies.

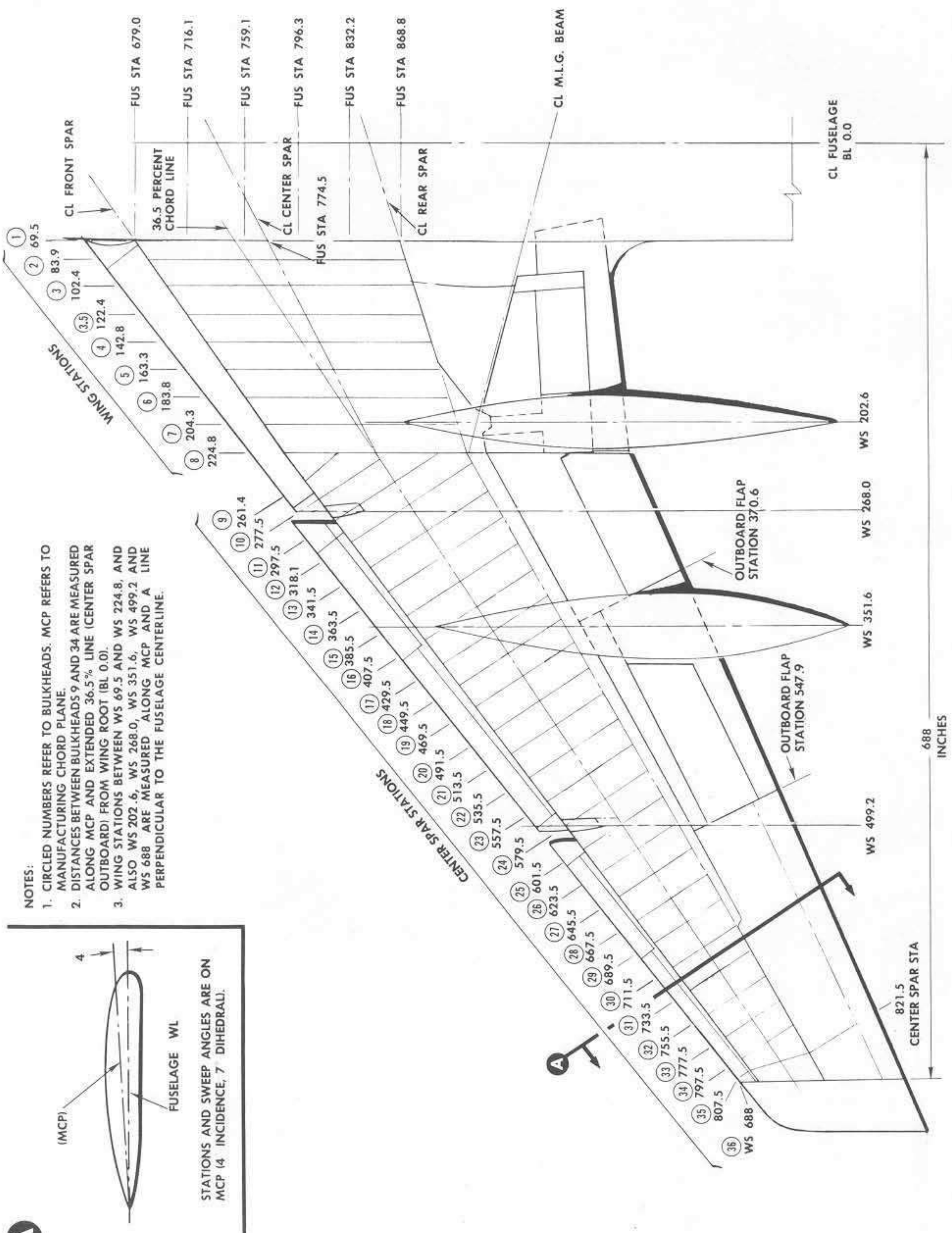
Engine Pod Station Lines

The station numbering system for the engine pods and pylons is independent of all other station numbering systems (see Figure 6-8). A zero reference point,

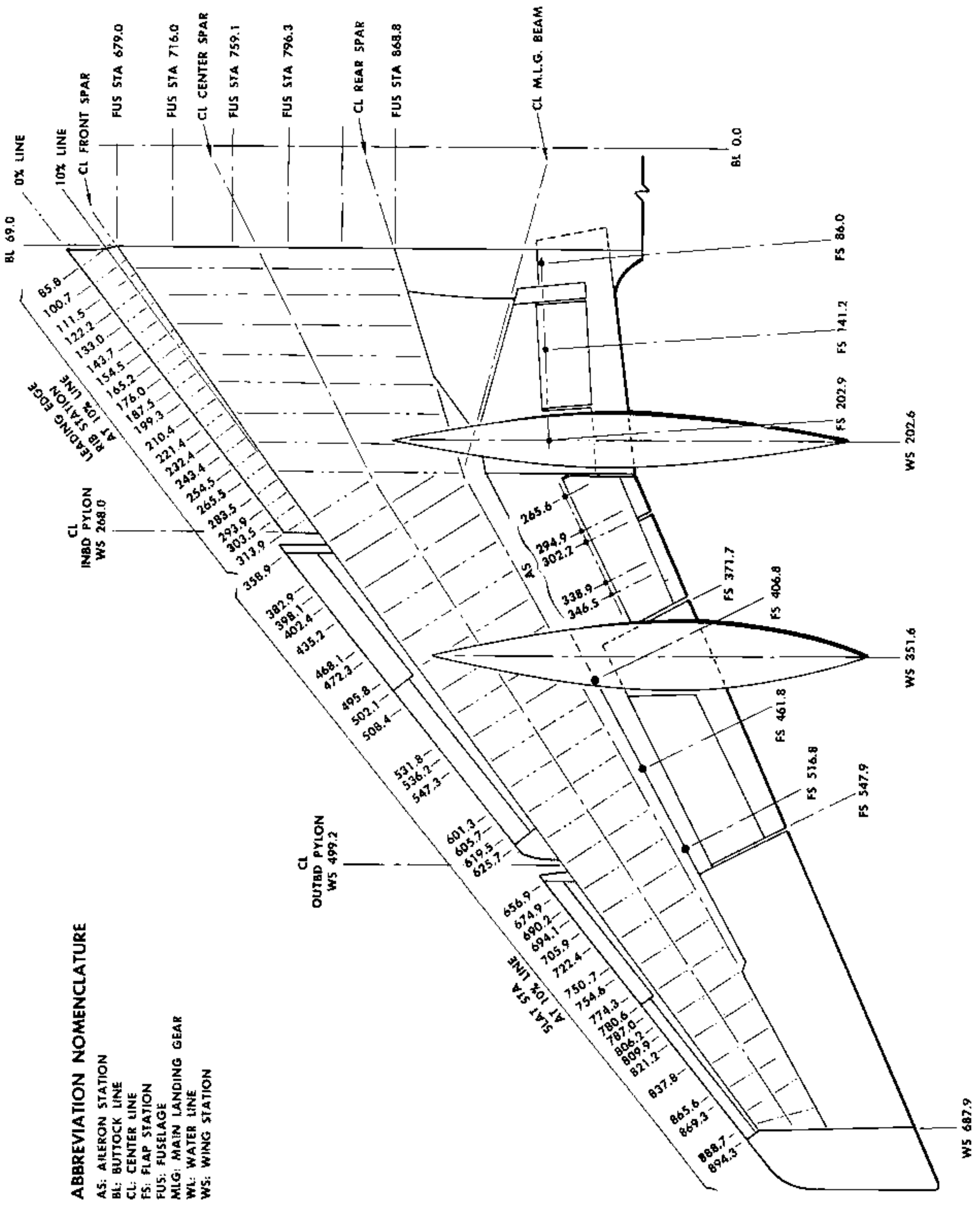


Fuselage Station Lines
Figure 6-5

Oct. 23/61
K

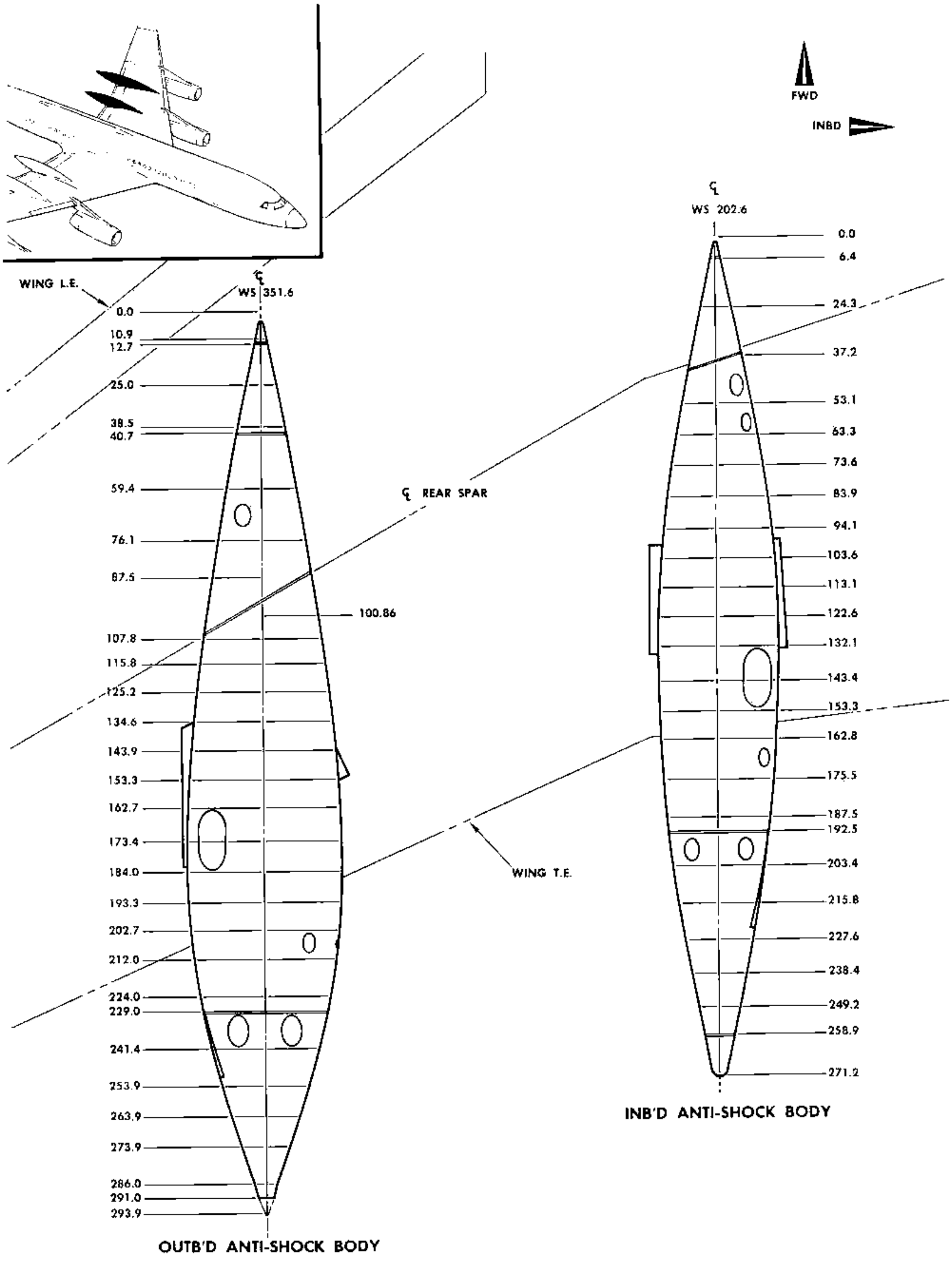


Wing Station Lines
Figure 6-6 (Sheet 1 of 2)

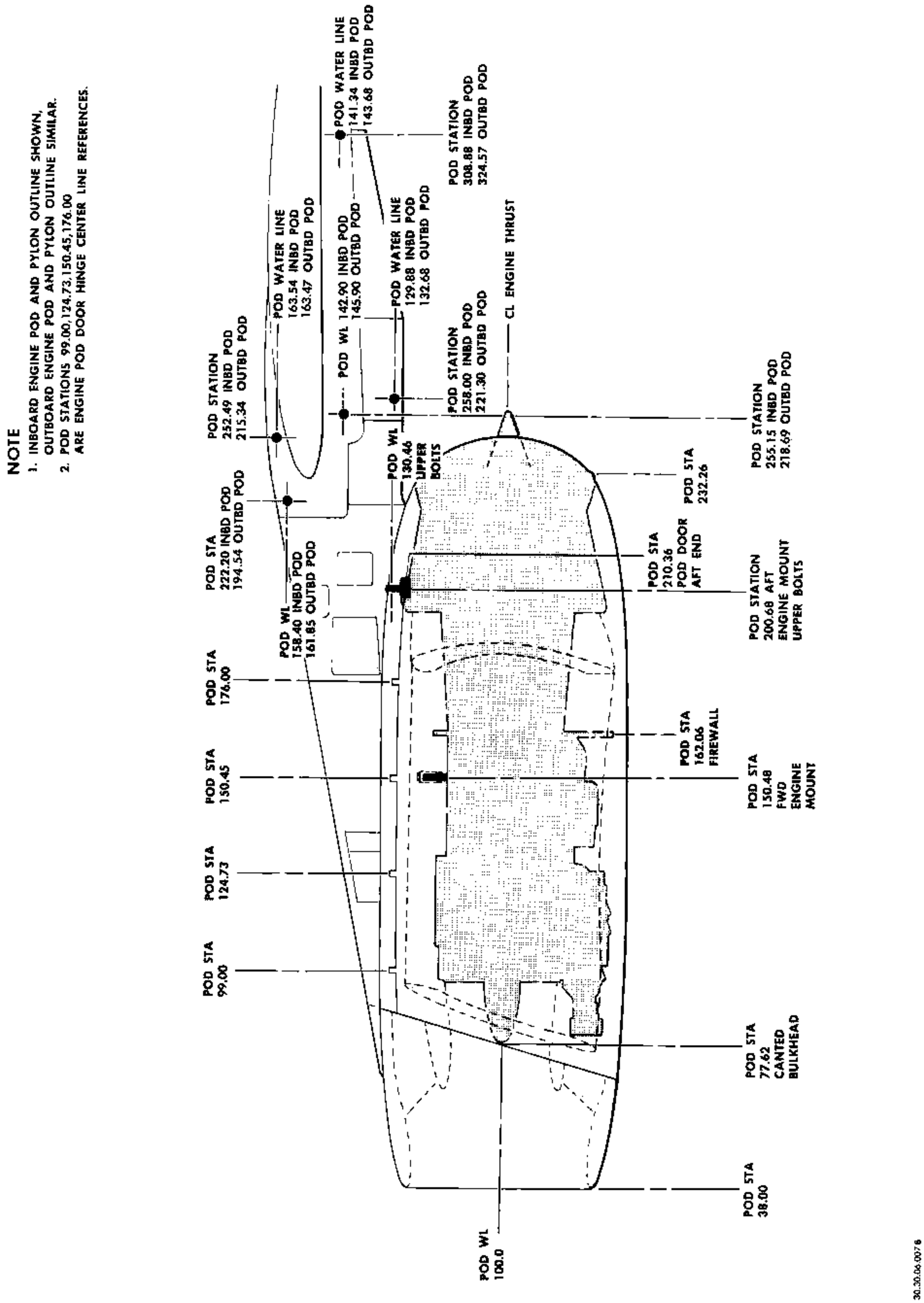


30.30.06.005-2

Wing Station Lines
Figure 6-6 (Sheet 2 of 2)



Anti-Shock Body Station Lines
Figure 6-7



Pod and Pylon Station Lines
Figure 6-8

located 38.00 inches forward of the engine intake cowling on the longitudinal reference axis, has been utilized to determine the station measurements. Stations are measured in inches aft from this point. Fuselage station lines, buttock lines and waterlines are also utilized for the location and installation of pod and pylon equipment items.

Horizontal Stabilizer Station Lines

Horizontal stabilizer inch-stations are measured from a zero reference point and are perpendicular to the stabilizer front spar (see Figure 6-9). Buttock lines and fuselage station lines can also be used for location and installation of equipment in the horizontal stabilizer.

Vertical Stabilizer Station Lines

The vertical stabilizer inch-stations are measured from zero reference point and are perpendicular to the stabilizer front spar as shown in Figure 6-10. In addition to these stations, waterlines and fuselage station lines can be utilized for installation and location of equipment.

Water and Buttock Station Lines

Waterlines are vertical reference points (WL) measured in inches from the inside surface of the lower fuselage skin at the airplane center line. This inside surface of the lower fuselage skin is at WL 0.0; all waterlines below this point are negative, or minus, waterlines. Several easy waterlines to remember are: WL 0.0 at the bottom of the fuselage center line, the top surface of the cabin floor at WL 59.00, and the fuselage leveling rivets on the outside surface of the fuselage (right and left sides) at WL 68.00.

Buttock lines (BL) are reference points along the lateral or Y-axis and are measured in inches to the right and left of the fuselage center line, also illustrated in Figure 6-16. The fuselage centerline is at BL 0.0, while the inside surface of the fuselage skin at the widest point in the fuselage (WL 81.75) is at BL 69.00. Buttock lines are also used as reference points in the wings and horizontal stabilizer, and extend to the right and left of the fuselage centerline. The extreme side of the wing tips are at BL 720.00.

Dimensions (Approximate)

Wing

Span .... 120 feet

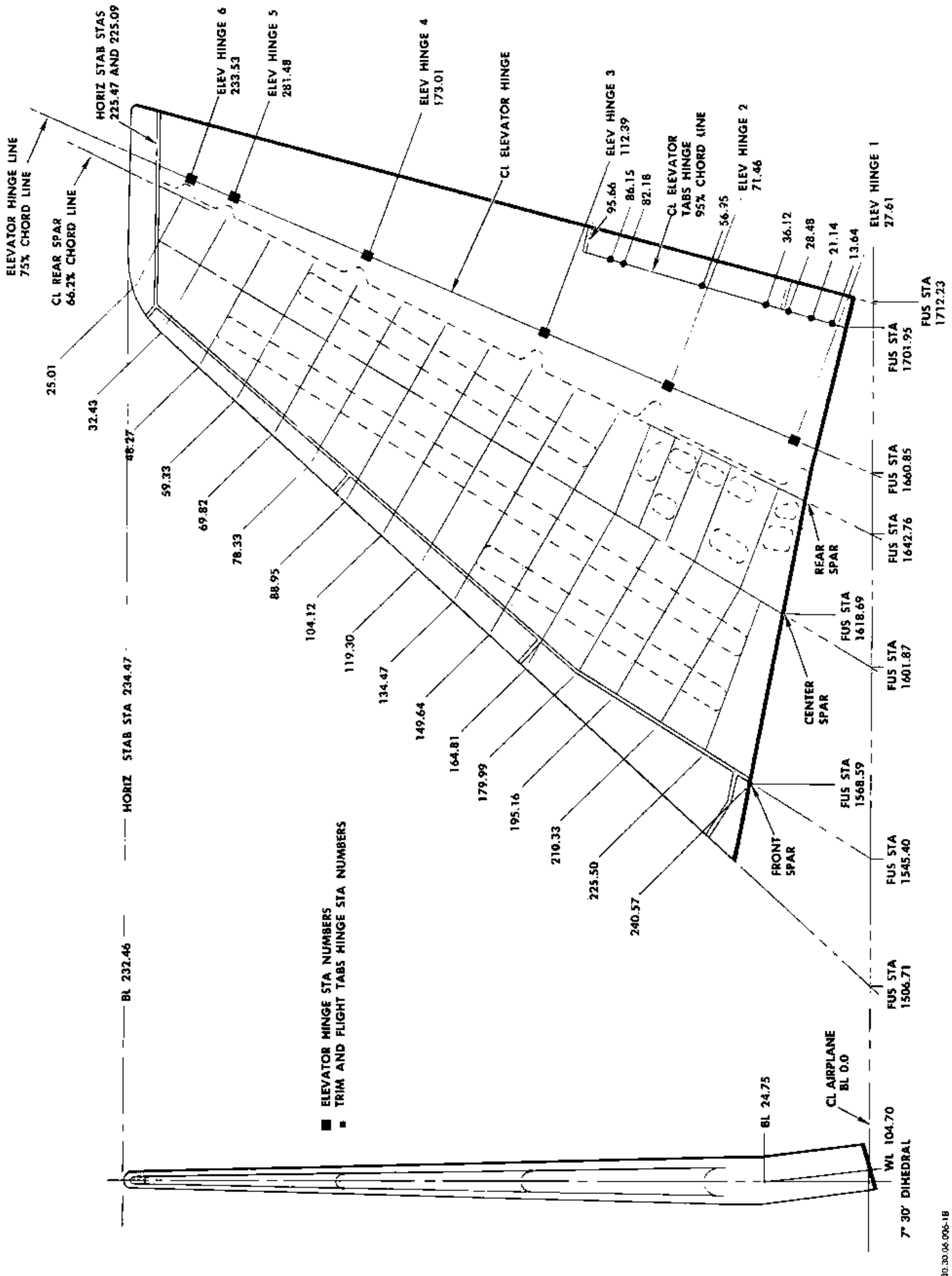
Chord at Root .... 29 feet

Chord at Tip .... 9 feet

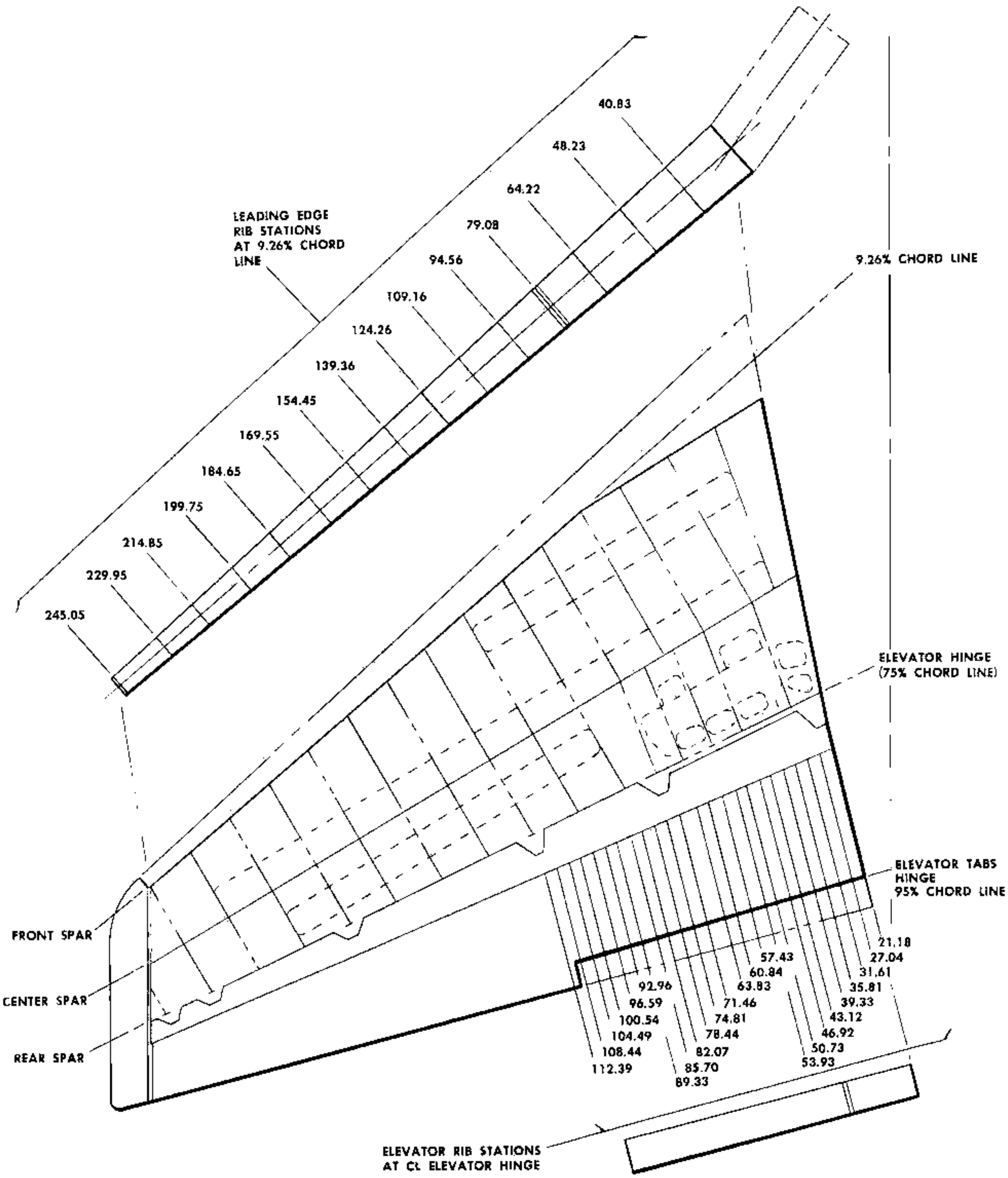
Landing Gear

Wheel Base .... 57 feet

Tread (Main Gear) .... 20 feet

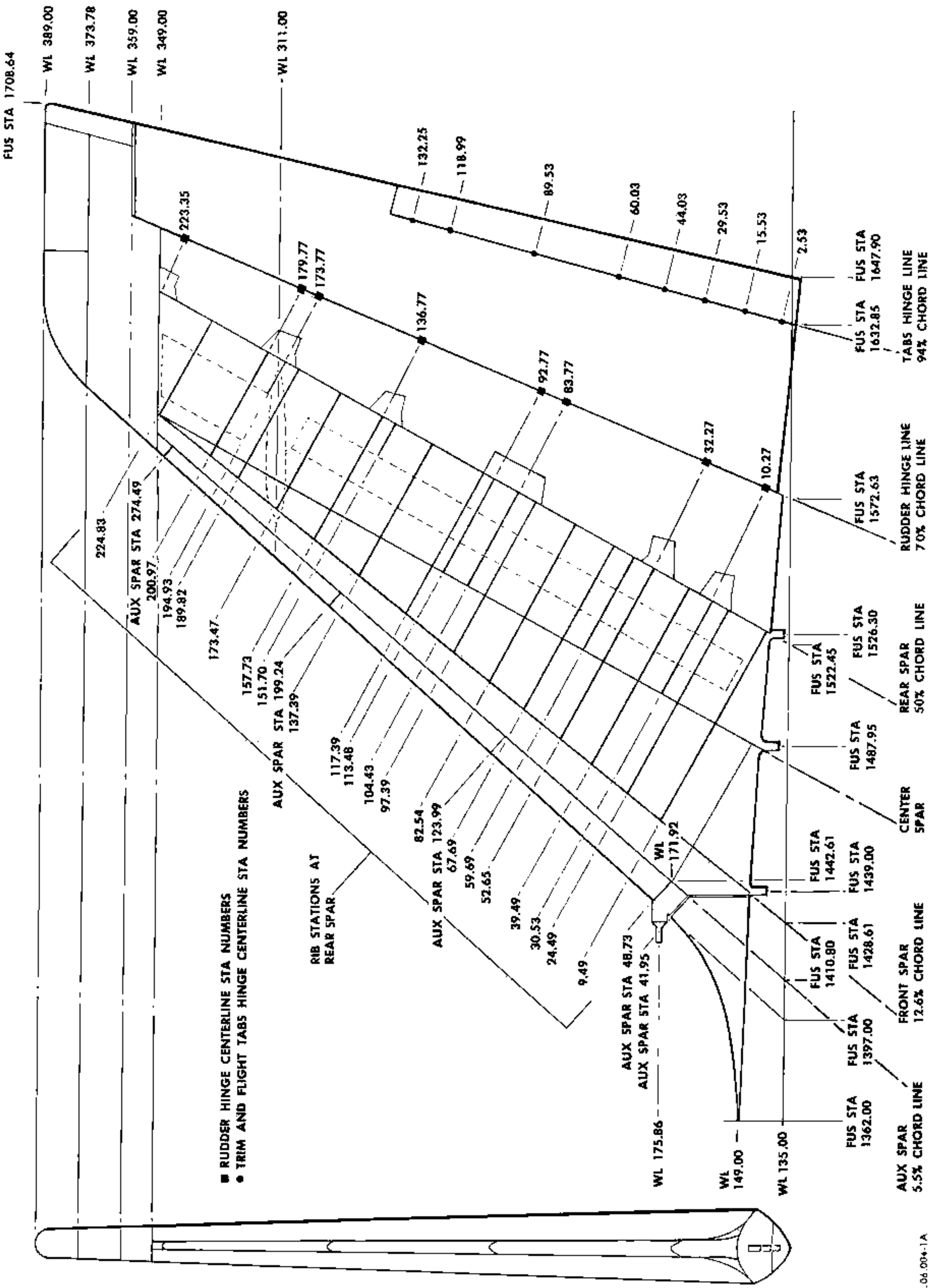


Horizontal Stabilizer Station Lines
Figure 6-9 (Sheet 1 of 2)

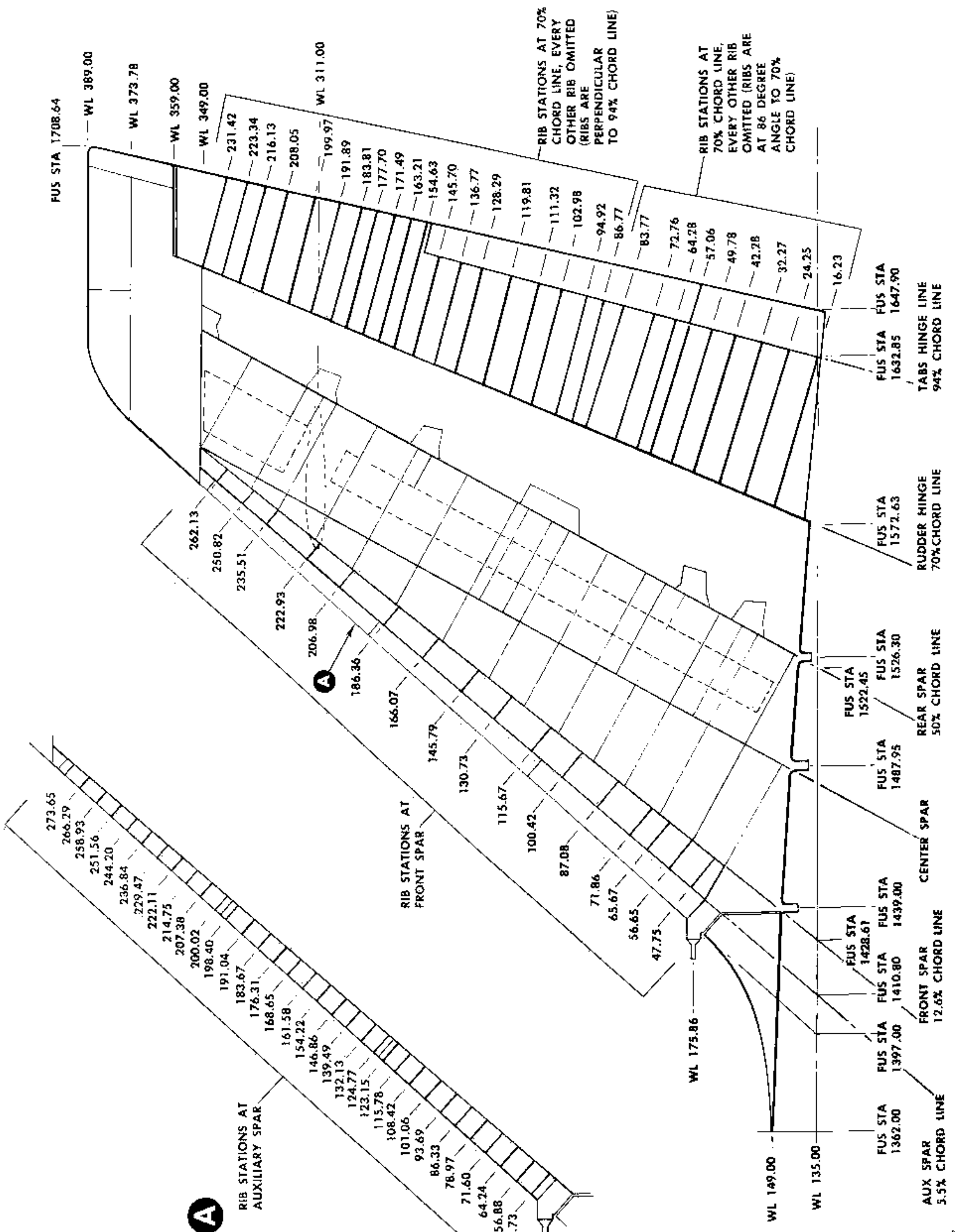


30.30.06.006-2

Horizontal Stabilizer Station Lines
Figure 6-9 (Sheet 2 of 2)



Vertical Stabilizer Station Lines
Figure 6-10 (Sheet 1 of 2)



Vertical Stabilizer Station Lines
Figure 6-10 (Sheet 2 of 2)

Landing Gear (Continued)

Tread (Main Gear Tires)		
Longitudinal Centerline	46	inches
Spanwise Centerline	24	inches
Tread (Nose Gear Tires)	17	inches
Nose Gear Centerline to Station 100	12	feet

Areas

Wing (Basic)	2250	square feet
Vertical Stabilizer (Total)	295	square feet
Rudder (Aft of Hinge Line, Including Tabs)	83	square feet
Rudder Balance	45	square feet
Servo Tab (Aft of Hinge)	8	square feet
Trim Tab (Aft of Hinge)	4	square feet
Horizontal Stabilizer (Total)	427	square feet
Elevator (Aft of Hinge, Including Tabs)	97	square feet
Tab (Aft of Hinge)	8	square feet
Aft Cargo Compartment	440	cubic feet
Forward Cargo Compartment	488	cubic feet

GENERAL SERVICING DATA

The Convair 990 is designed so that it can be easily serviced in a minimum of time with little interference between the crews servicing the airplane systems, galleys, lavatories and loading and unloading the cargo areas. All servicing of the fuselage systems, except for the potable water system, is accomplished from the right side of the airplane; the left side of the airplane, except for the fuel, engine and potable water system servicing, is reserved for passenger and flight crew enplaning and deplaning (see Figure 6-11). Various servicing points and locations are shown in Figure 6-12.

Servicing Replenishment Tables

NOTE: The values shown in the following servicing-replenishment information are only approximate. For actual airplane servicing, use the proper Maintenance Manual information and procedures or, when applicable, the FAA Approved Airplane Flight Manual Values.

TABLE I - FUEL TANK DATA (POUNDS AND KILOGRAMS)

	POUNDS (KEROSENE)	KILOGRAMS (KEROSENE)	POUNDS (JP4)	KILOGRAMS (JP4)
Total Fuel Weight	101,344 *	46,020 *	98,535 *	44,780 *
	103,026 **	47,184 **	100,167 **	45,502 **
	103,327 ***	47,321 ***	100,460 ***	45,635 ***
Total Fuel Weight - Each Outboard Tank (No. 1 and No. 4)				
Main	11,880	5,384	11,544	5,247
Replenishment	4,436	2,016	4,303	1,955
Anti-shock Body	1,447	658	1,404	638
(outboard)	3,129 ** & ***	1,822 ** & ***	3,035 ** & ***	1,360 ** & ***
Unusable	60	27	59	26
Total (Tank No. 1 or 4)	17,832	8,085	17,310	7,867
	19,514 ** & ***	9,249 ** & ***	18,942 ** & ***	8,589 ** & ***
Total Fuel Weight - Each Inboard Tank (No. 2 and No. 3)				
Main	10,606	4,820	10,289	4,677
Replenishment	9,460	4,300	9,178	4,171
Anti-shock Body (inboard)	2,265	1,029	2,295	1,042
Unusable	40	18	39	17
Total (Tank No. 2 or No. 3)	22,371	10,167	21,801	9,907

* Applicable to airplanes CV1 through CV25
** Applicable to airplanes CV26 through CV35
***Applicable to airplanes CV35 and on

TABLE I - FUEL TANK DATA (POUNDS AND KILOGRAMS) (CONT)

	POUNDS (KEROSENE)	KILOGRAMS (KEROSENE)	POUNDS (JP4)	KILOGRAMS (JP4)
Total Fuel Weight - Center Section				
Total (usable)	20,804	9,456	20,182	9,174
	21,105 ***	9,593	20,475 ***	9,306 ***
(unusable)	134	60	130	59
Total - Center Section	20,938	9,516	20,313	9,232
	21,239 ***	9,653 ***	20,605 ***	9,365 ***

NOTE: Weight of Jp-4 is based on a factor of 6.5 pounds per US gallon at 32 degrees F (0 degrees C). Weight of kerosene is based on a factor of 6.7 pounds per US gallon at 32 degrees F (0 degrees C). The tanks have minimum expansion space equal to two percent of the total gross volume.

CAUTION: SPECIFIC GRAVITY (FUEL WEIGHT) VARIES WITH TEMPERATURE. IN DETERMINING FUEL LOAD, CONSIDERATION SHALL BE GIVEN TO POSSIBLE CHANGES IN AMBIENT TEMPERATURE WHILE THE AIRPLANES ARE GROUNDBORNE.

***Applicable to airplanes CV35 and on.

TABLE II - FUEL TANK DATA (GALLONS AND LITERS)

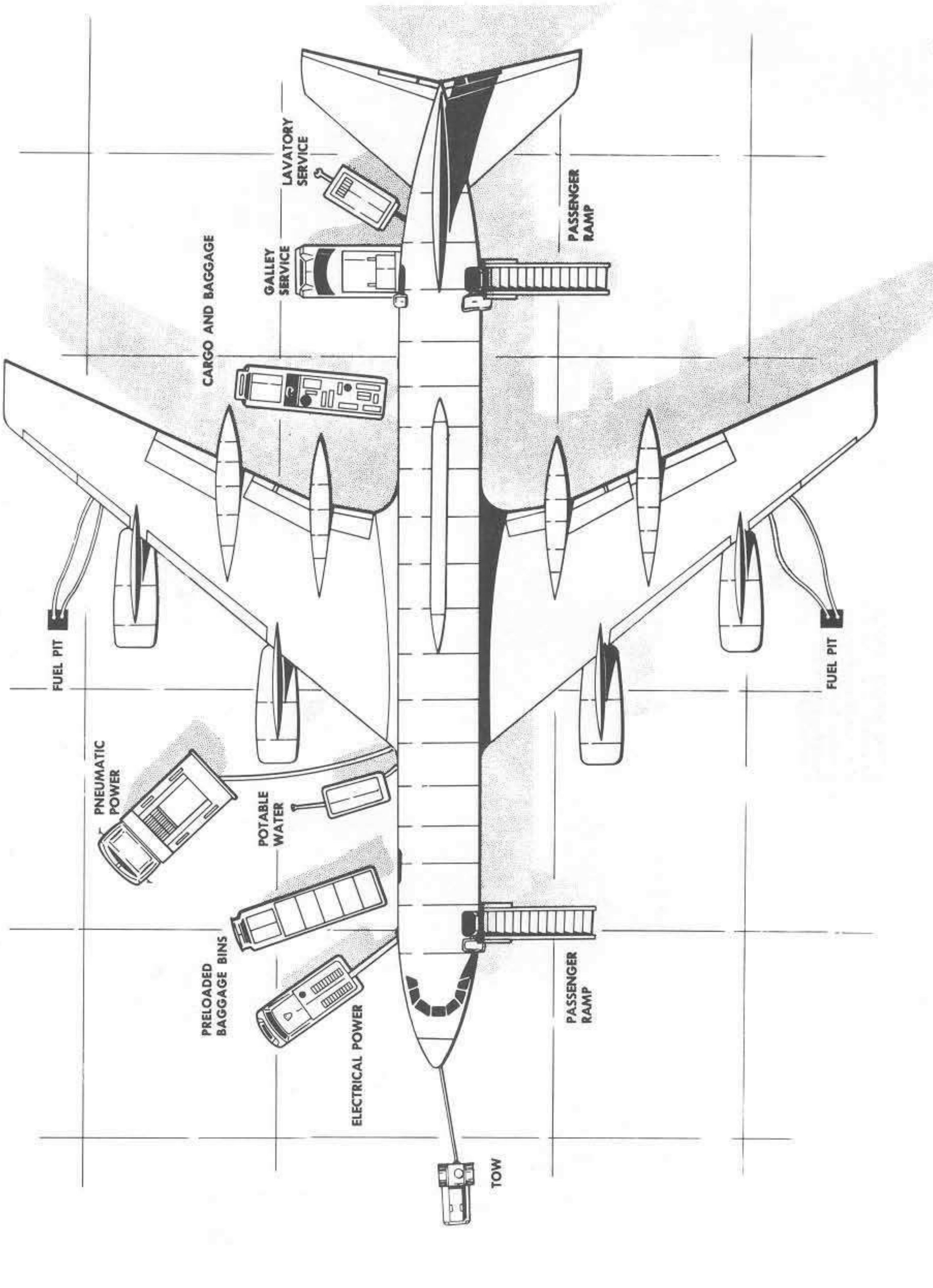
	US GALLONS	IMPERIAL GALLONS	LITERS
Total Fuel Quantity	15,159 *	12,582 *	57,604 *
	15,661 **	12,999 **	59,519 **
	15,706 ***	13,036 **	59,683 ***
Total Quantity - Each Outboard Tank (No. 1 and No. 4)			
Main	1,776	1,474	6,749
Replenishment	662	549	2,515
Anti-shock Body	216	179	821
	467 ** & ***	388 ** & ***	1,775 ** & ***
Unusable	9	7	34
Total (Tank No. 1 or No. 4)	2,663	2,210	10,119
	2,914 ** & ***	2,419 ** & ***	11,073 ** & ***
Total Quantity - Each Inboard Tank (No. 2 and No. 3)			
Main	1,583	1,314	6,215
Replenishment	1,412	1,172	5,366
Anti-shock Body	353	293	1,341
Unusable	6	5	23
Total (Tank No. 2 or No. 3)	3,354	2,774	12,745
Total Quantity - Center Section			
Total (usable)	3,105	2,577	11,799
	3,150 ***	2,695 ***	11,970 ** & ***
(unusable)	20	17	76
Total - (Center Section)	3,125	2,594	11,875
	3,170 ***	2,712 ***	12,046 ** & ***

* Applicable to airplanes CV1 through CV25
** Applicable to airplanes CV26 through CV34
***Applicable to airplanes CV35 and on

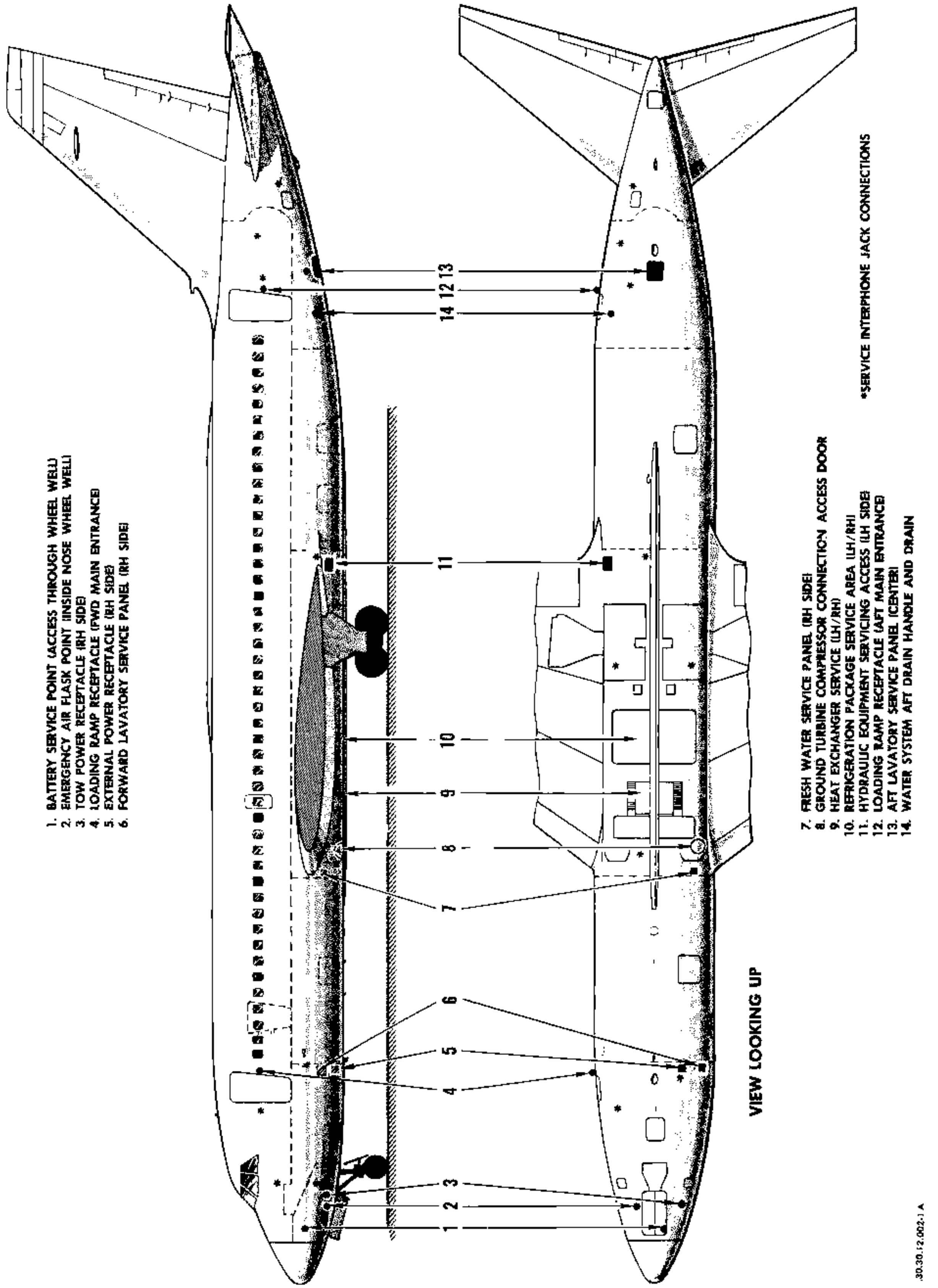
TABLE III - POWER PACKAGE AND AIRFRAME SYSTEMS

	US GALLONS	IMPERIAL GALLONS	LITERS
CSD and Engine Oil Tanks (Oil Specification MIL-L-7808)			
No. 1 Engine (CSD 1.72 Eng 4.15)	5.87	4.88	22.21
No. 2 Engine (Same)	5.87	4.88	22.21
No. 3 Engine (Same)	5.87	4.88	22.21
No. 4 Engine (Same)	5.87	4.88	22.21
Total all four engines	23.48	19.52	88.84
Potable Water Tank	50 *	41.5 *	190.0 *
	75 **	62.3 **	285.0 **
Main Hydraulic System (Skydral-500)			
No. 1 Reservoir	2.8	2.3	10.6
No. 1 System less reservoir	19.0	15.4	72.2
No. 2 Reservoir	5.8	4.8	22.0
No. 2 System less reservoir	28.5	23.7	111.0
Total hydraulic fluid	56.1	46.2	215.9

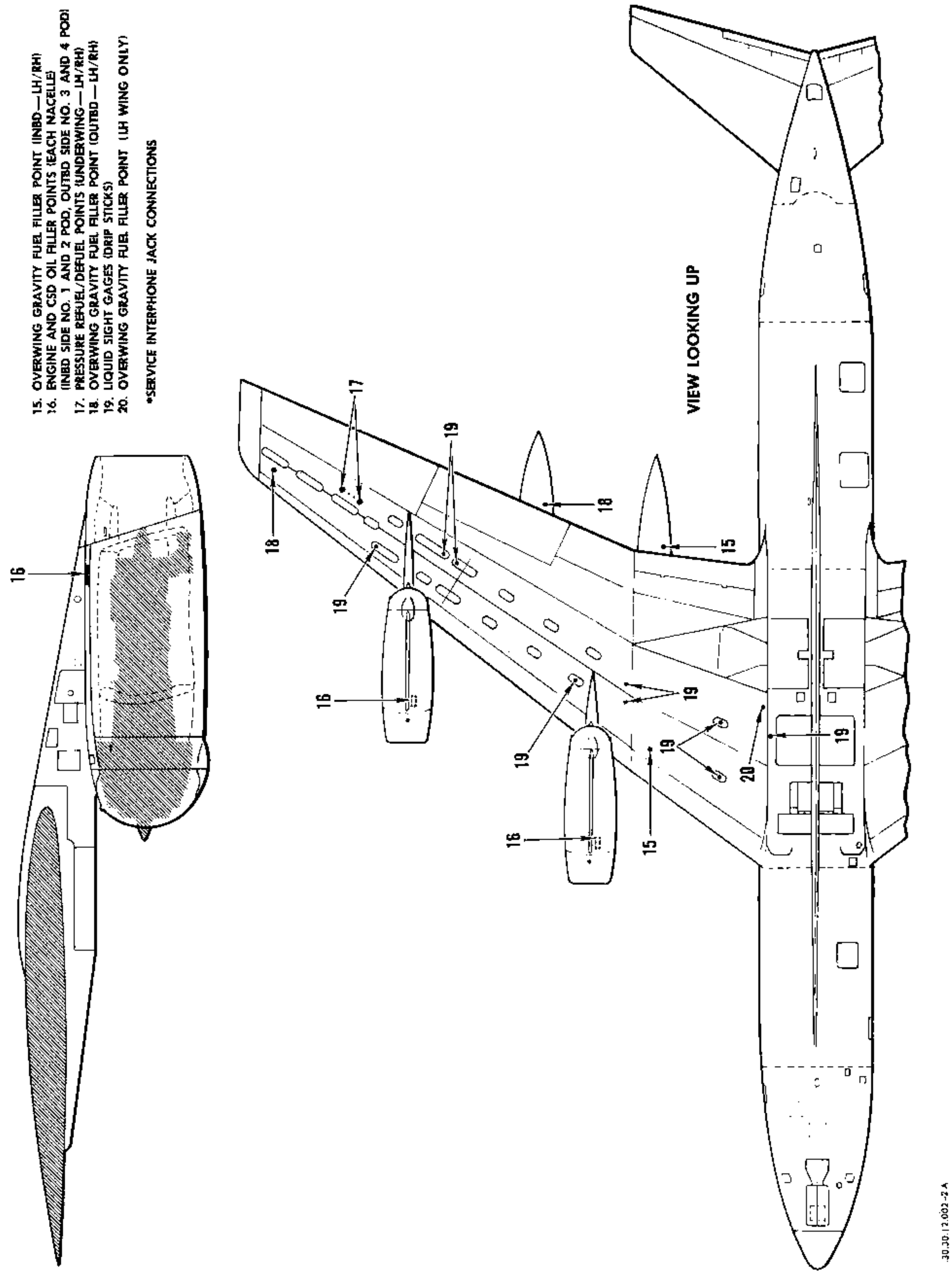
* Applicable to airplanes CV1 through CV25
**Applicable to airplanes CV26 and on



Airplane Ground Servicing - Typical
Figure 6-11



Airplane Servicing Points
Figure 6-12 (Sheet 1 of 2)



Airplane Servicing Points
Figure 6-12 (Sheet 2 of 2)

WALKWAYS

Walkways, as such, are not designated since this airplane has been designed to permit as much servicing as possible to be accomplished either from work stands or from the ground. Figure 6-13 illustrates that the area between the front and rear spars on the wings and the horizontal stabilizer are the only places where walking is permitted. Maintenance personnel should remove their shoes or wear protective slip-on shoe coverings any time it is necessary to walk on the wings or horizontal stabilizer so as to preserve the required aerodynamic smoothness of these surfaces. Scratches due to shoes, or dragging hoses across the surfaces, will be paid for by loss of speed and higher fuel consumption. This condition is not appreciable on piston-powered airplanes but is an item of considerable magnitude in jet transport operations. The areas on which walking is prohibited are marked "NO STEP".

JET BLAST CLEARANCE AREAS

Figure 6-14 illustrates the danger areas that exist at various power settings due to engine operation (zero wind conditions). The areas shown should be considered as minimums since unpleasant, though possibly not dangerous, effects may be noticeable at greater distances. High crosswinds, or downwinds, may increase the hazardous conditions, as may also ice or snow.

AIRPLANE GROUND HANDLING PROCEDURES

NOTE: The information contained under this heading is for familiarization only. Before actual accomplishment of any ground handling, consult the proper sections of the Maintenance Manual.

Towing is normally accomplished from the nose gear when the airplane is on a hard, dry surface. When mud, snow or rough terrain may create excessive towing loads, towing must be accomplished from the main gears. Main gear towing can be either backward or forward as required by circumstances.

1. Use the proper size tug (or tugs) with appropriate electrical capability.
2. Place down-lock safety pins in the main and nose gear braces.
3. Make an operational check of the hydraulic brake system.
4. Check emergency air brake pressure.
5. Disconnect the nose wheel torque arms.

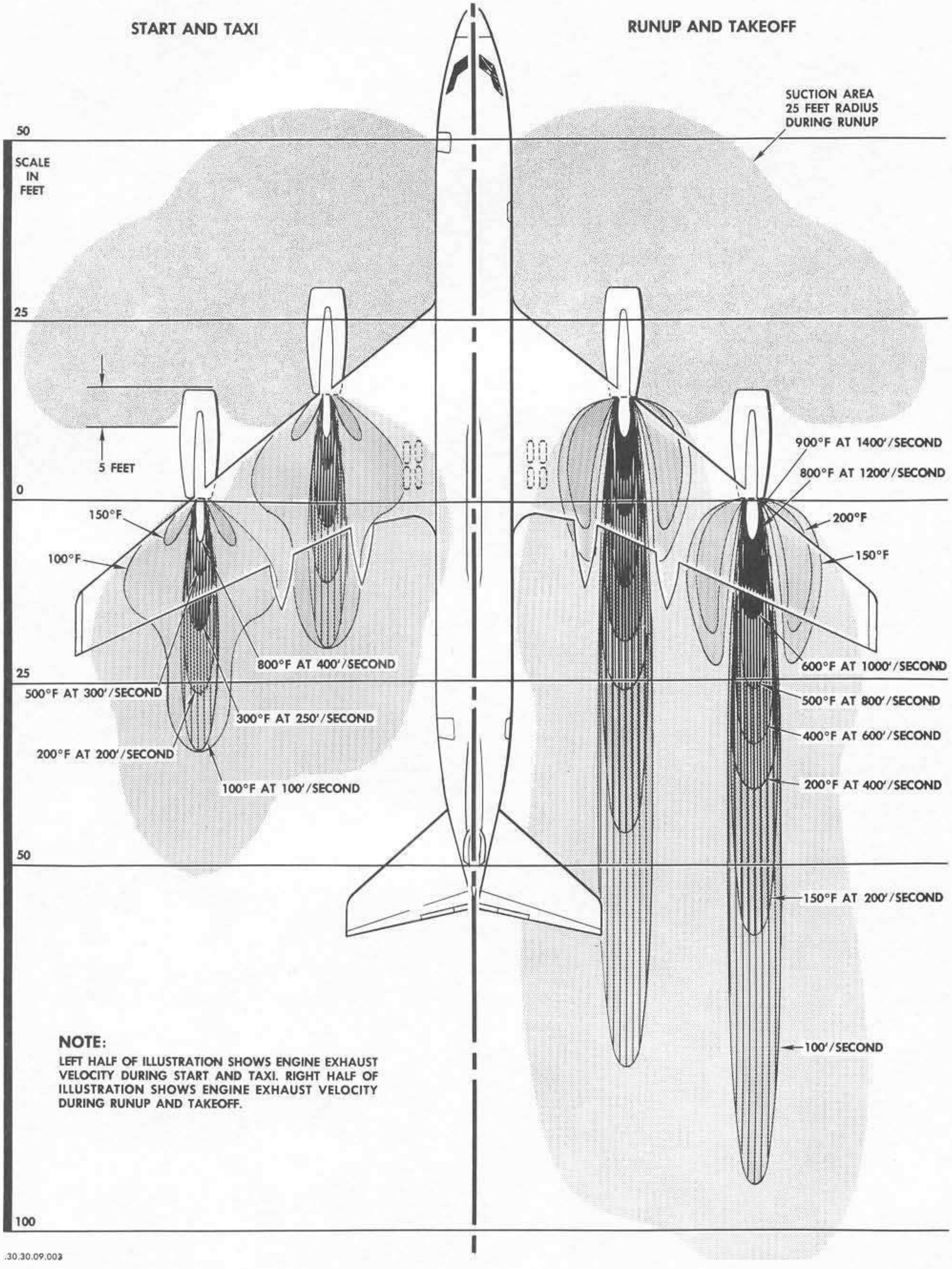
CAUTION: THE AIRPLANE SHOULD NEVER BE TOWED WITH A CG AFT OF STATION 912. TAIL TIP-DOWN MAY OCCUR UNDER SUCH A CONDITION.

Nose Gear Towing

Equipment required for nose gear towing consists of a tow bar and a tow truck equipped with an electrical power source suitable for use with the airplane. Four men should be used during towing activities; one for the operation of

TO BE FURNISHED

Airplane Walkways Diagram
Figure 6-13



Engine Inlet and Exhaust Danger Areas
Figure 6-14

wheel brakes and equipment in the cockpit, one on the tug, and one at each wing tip to check wing and tail clearance. Towing speed should not exceed a fast walking gait (approximately 5 to 6 mph). The nose wheel steering disconnect permits a swivel of 360 degrees; however, towing at angles greater than 60 degrees can result in scuffing of the main tires (see Figures 6-15 and 6-16).

CAUTION: DO NOT DISCONNECT EQUIPMENT OR LINES AT THE NOSE GEAR. IMPROPER RECONNECTION CAN CAUSE MALFUNCTION OF ONE OR MORE SYSTEMS.

Main Gear Towing

Integral towing lugs are installed on the front and rear of each main gear truck (see Figure 6-17). By utilizing a manilla rope or aircraft cable, towing can be accomplished either forward or backward in mud, snow or over rough terrain. Personnel should be utilized in the same manner as for nose gear towing. Brake and electrical requirements are also the same.

AIRPLANE JACKING PROCEDURES

Lifting the airplane consists of such operations as jacking the entire airplane, jacking at each landing gear, hoisting the airplane, and raising the airplane with pneumatic lifting bags. Precautions such as removing all unnecessary equipment in the area and the checking of overhead clearances should be observed.

WARNING: DURING JACKING AND LIFTING PROCEDURES, ALL PERSONNEL NOT ACTIVELY ENGAGED IN THE OPERATION SHOULD REMAIN CLEAR OF THE AIRPLANE. ALL EQUIPMENT SHOULD BE REMOVED FROM THE AREAS DIRECTLY BENEATH FUSELAGE, WINGS AND TAIL.

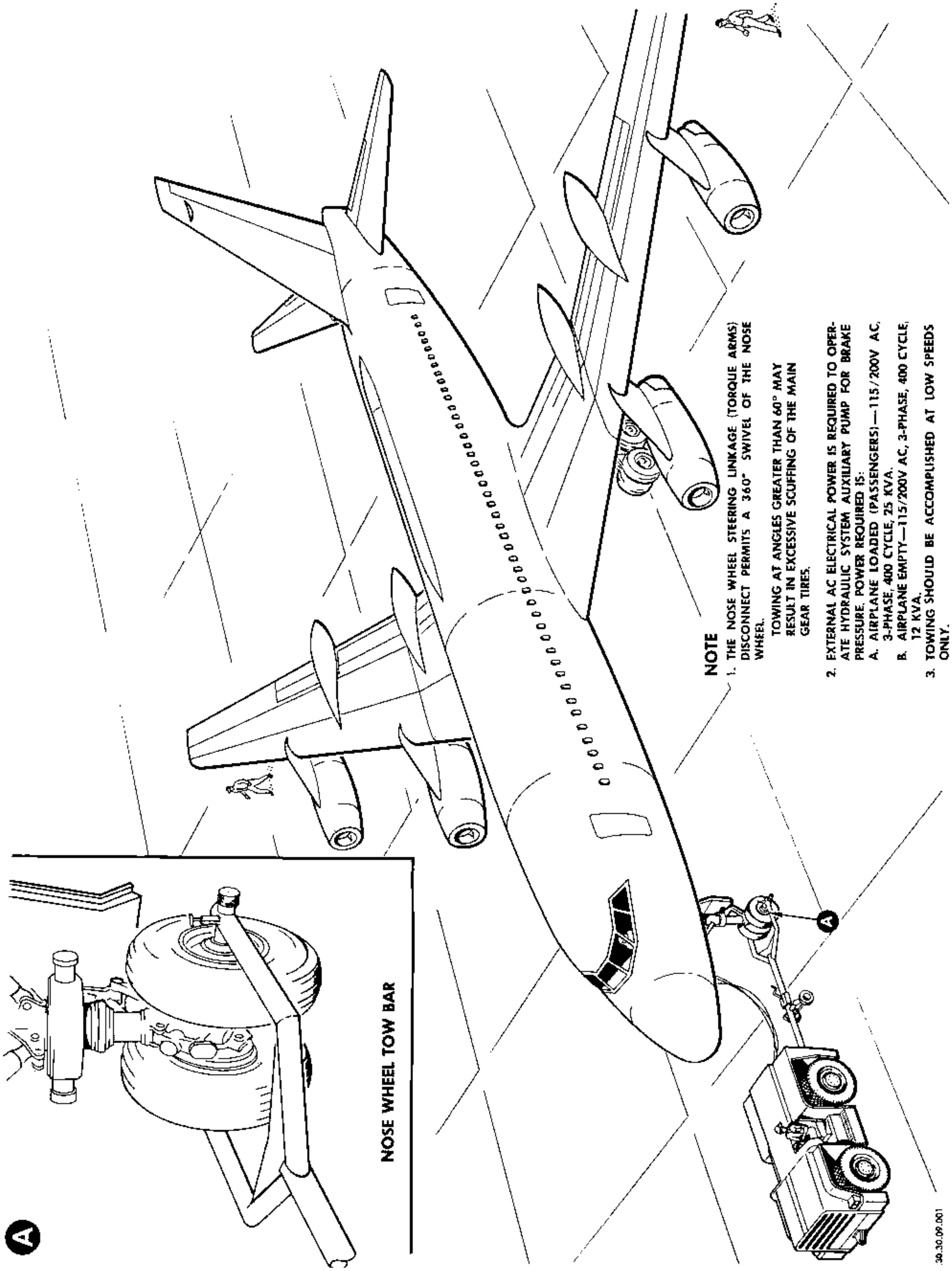
Jacking the Entire Airplane

The entire airplane can be raised using the three airplane jack points (see Figure 6-18). The fuselage jack point is located at station 153.6 just forward of the nose landing gear well. The two wing jack points are located at wing station 224.7 four inches aft of the rear spar. When not in use, the jack fitting openings are covered with flush covers to retain aerodynamic smoothness.

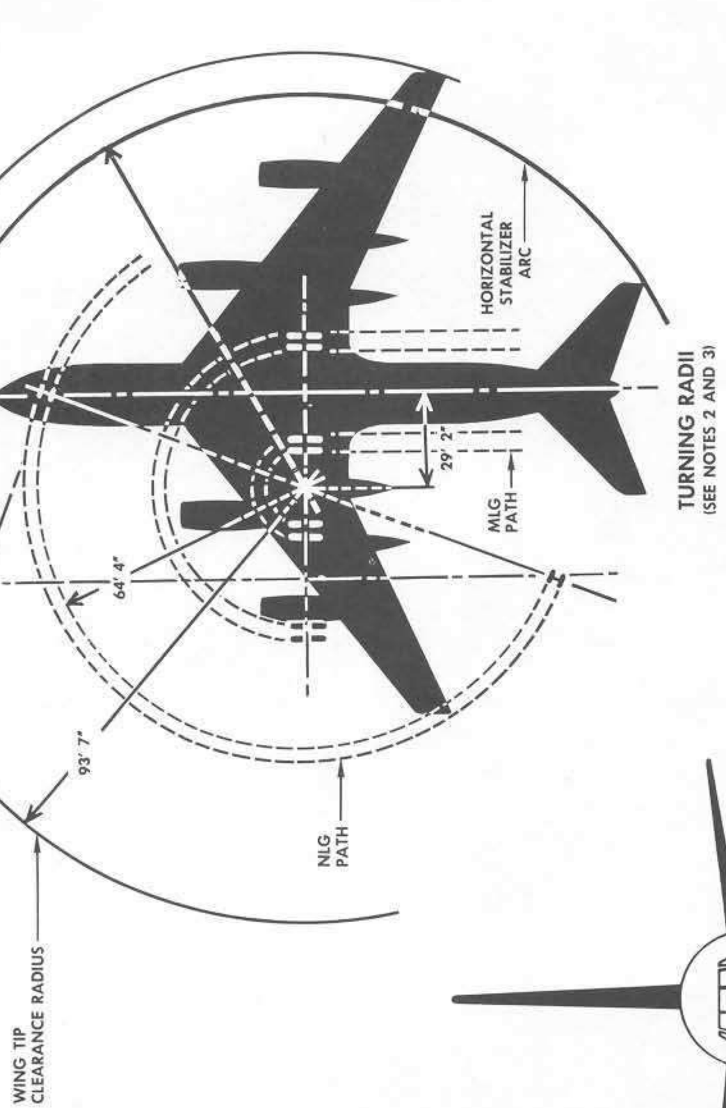
CAUTION:
1. JACKING OPERATIONS MUST BE SIMULTANEOUS, WITH THE AIRPLANE KEPT IN A LEVEL ATTITUDE.
2. PLACE A RESTRAINING CLAMP ON EACH GEAR STRUT TO PREVENT STRUT EXTENSION WHEN LIFTING AIRPLANE.

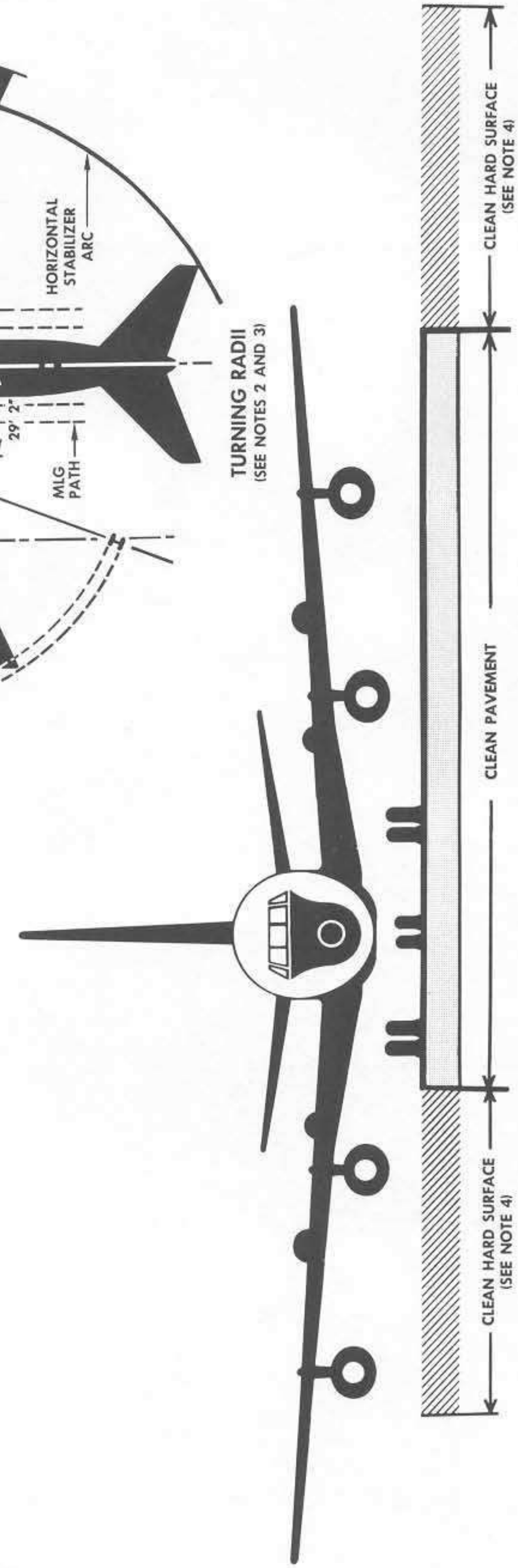
Jacking at the Landing Gear

Three integral jack pads are installed on the main gear trucks and one on the nose gear strut (see Figure 6-19). Individual wheel or brake servicing is accomplished by raising the front or rear of the main gear trucks by means of the integral pads. An added safety precaution when jacking under a strut is to place a jack under the respective wing or fuselage jack point. When jacking under the strut, only lift the gear high enough so that the tires clear the ground.



Towing From the Nose Gear
Figure 6-15





NOTES

1. MAXIMUM NOSE WHEEL STEERING ANGLE IS 63°

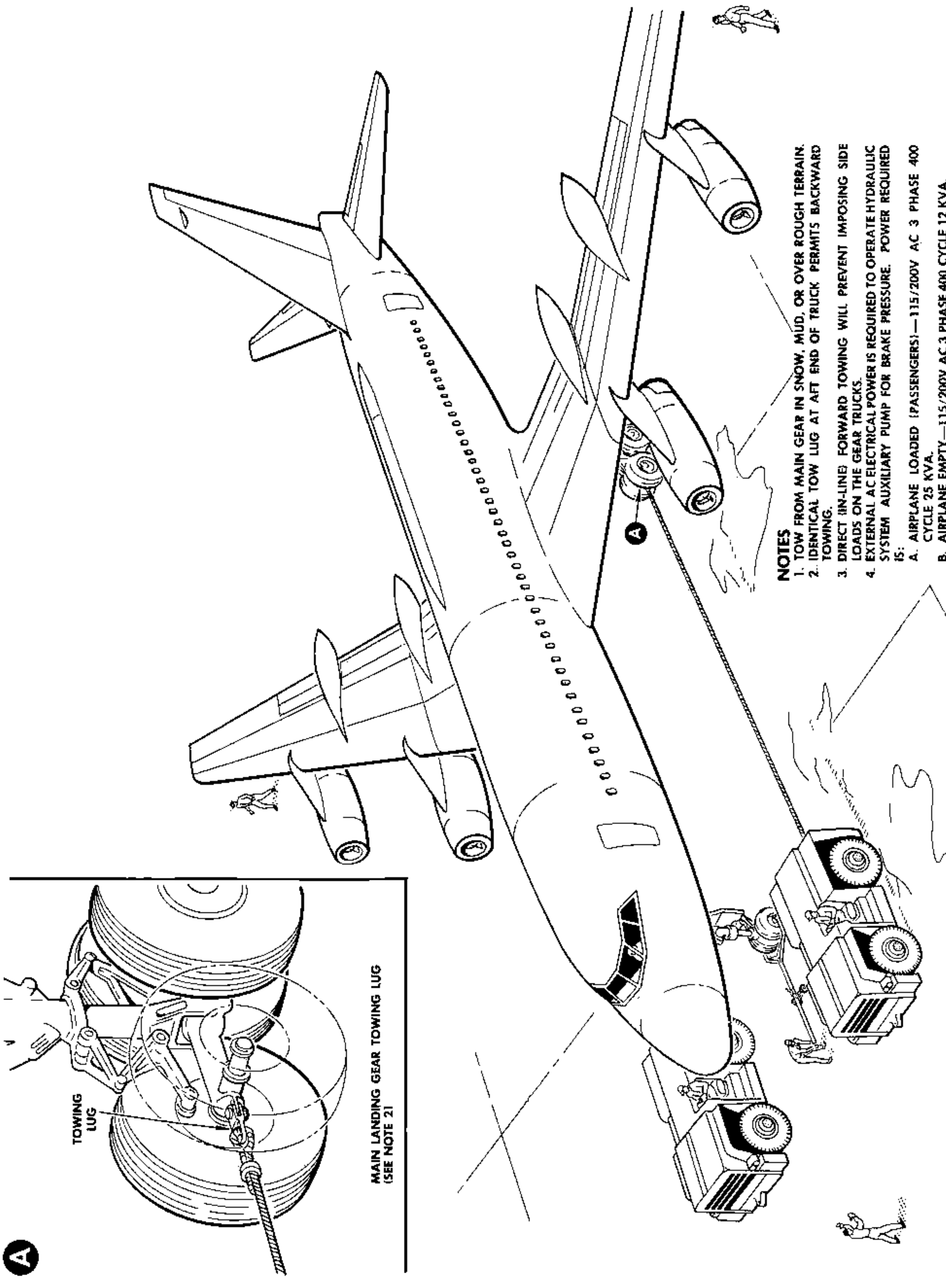
CAUTION

TURNING ANGLES GREATER THAN 50° CAN RESULT IN SCUFFING OF THE MAIN GEAR TIRES.

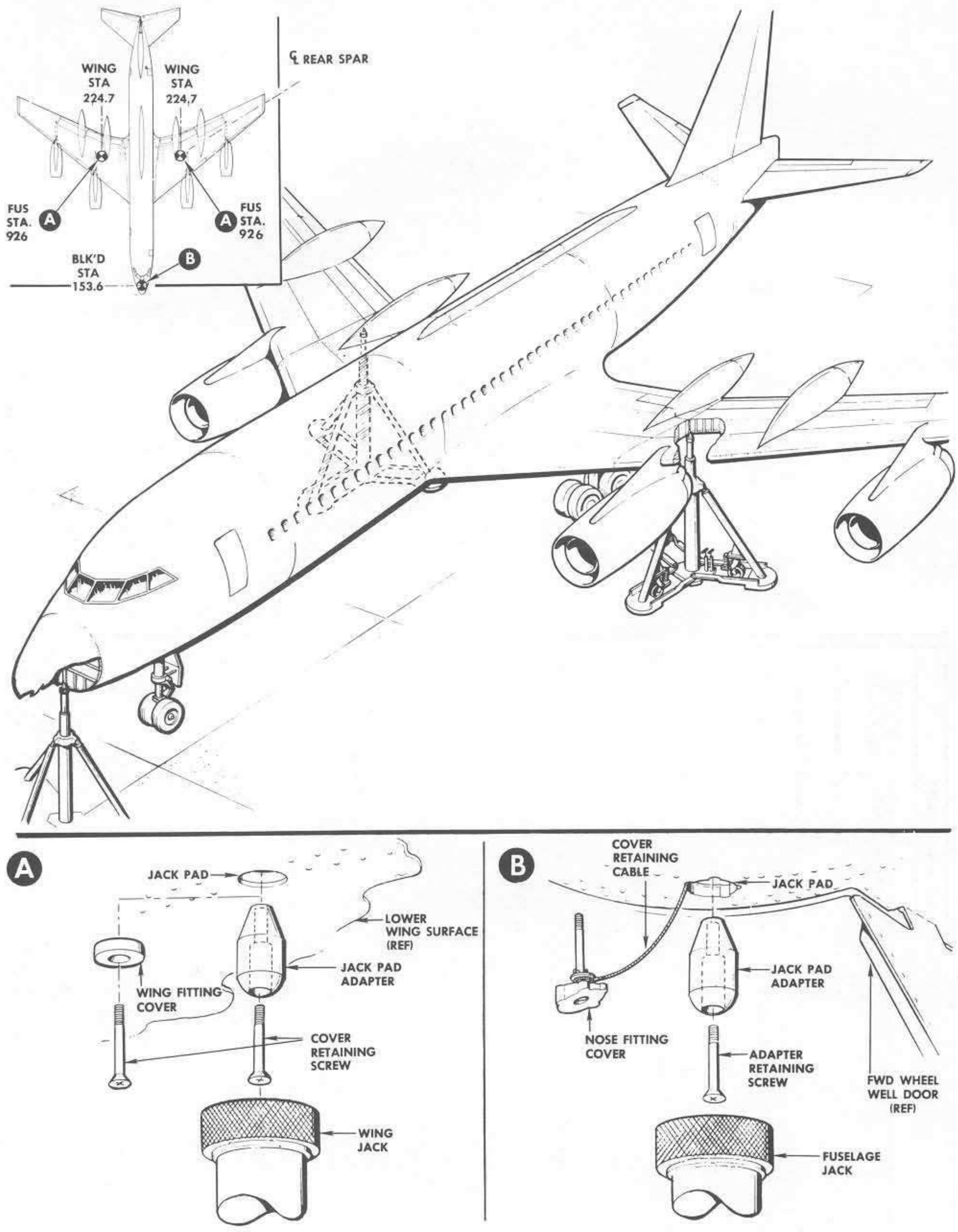
2. TURNING DETAIL ILLUSTRATES MINIMUM WING AND TAIL CLEARANCE RADII FOR AIRPLANE EXECUTING A 180° TURN AT MAXIMUM NOSE WHEEL STEERING ANGLE OF 63°.
3. AREA REQUIRED FOR TURNING AIRPLANE VARIES WITH DEGREE OF NOSE WHEEL STEERING ANGLE.
4. CLEAN HARD SURFACE DESIRABLE TO MINIMIZE POSSIBILITY OF LOOSE DEBRIS ENTERING ENGINES.

30.02.09.004

Airplane Turning Radius
Figure 6-16



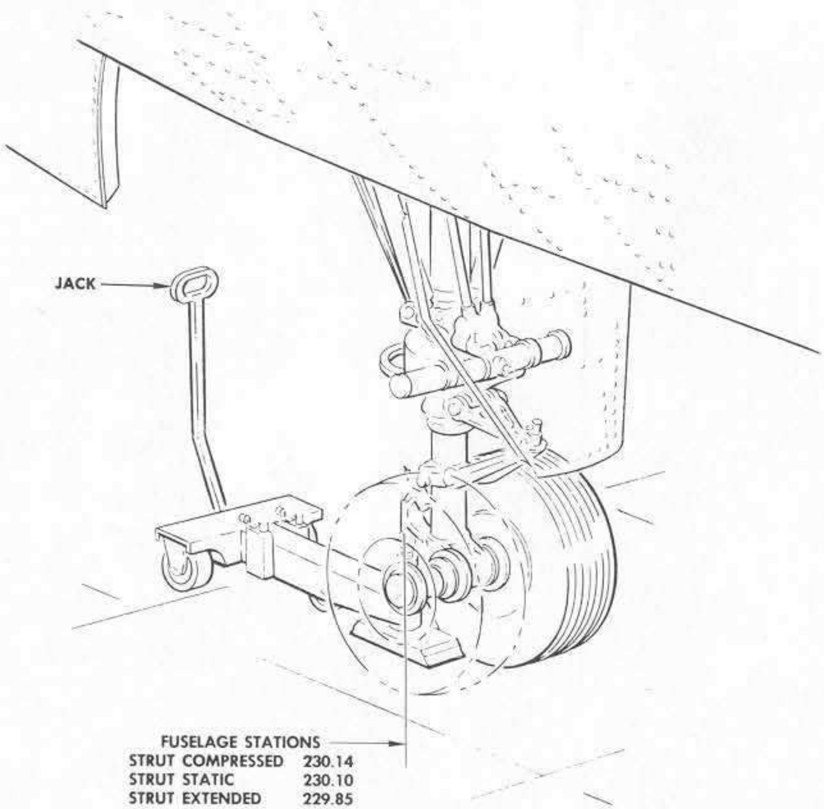
Towing From the Main Gear
Figure 6-17



30,30.07.001

Airplane Jacking Points
Figure 6-18

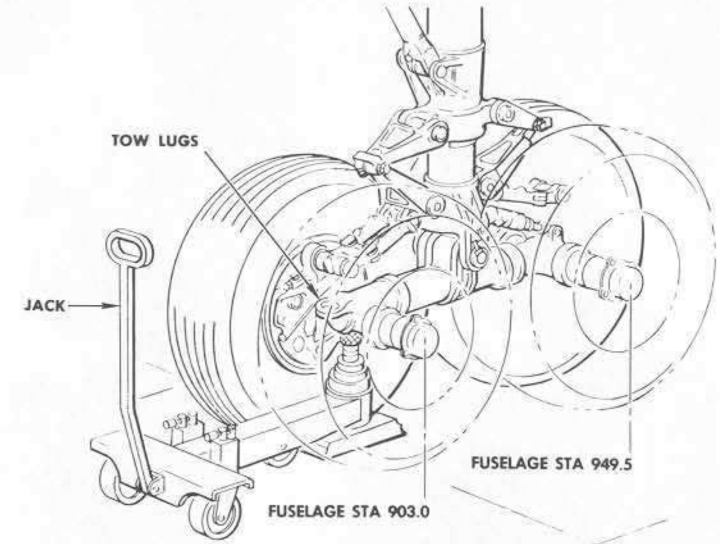
JACK CLEARANCES, CAPACITIES, AND LOADS								
LANDING GEAR JACK POINT LOCATION		CLEARANCE (INCHES)					CAPACITY (TONS)	MAXIMUM ALLOWABLE LOAD (POUNDS)
		TIRES INFLATED	2 FLAT TIRES	4 FLAT TIRES	2 WHEELS ON RIMS	4 WHEELS ON RIMS		
MAIN LANDING GEAR STRUT		13.50	11.56	9.70	8.88	4.50	60 (EACH)	116,500 (EACH)
MAIN LANDING GEAR TRUCK	FOR-WARD	13.62	9.82	9.82	4.62	4.62	30 (EACH)	57,500 (EACH)
	AFT	13.62	9.82	9.82	4.62	4.62	30 (EACH)	59,000 (EACH)
NOSE LANDING GEAR STRUT		14.05	11.65	—	10.35	—	15	20,700



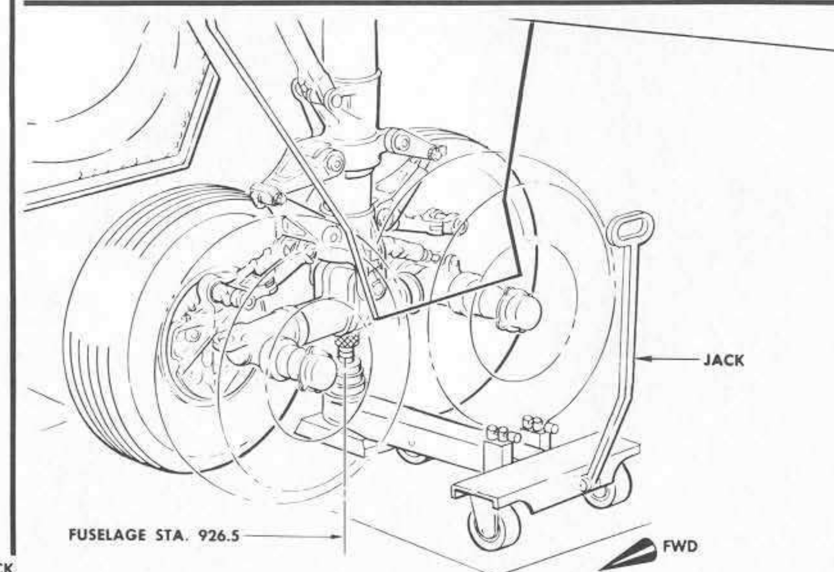
JACKING AT
NOSE LANDING GEAR

30.30.07.002B

CAUTION
MAKE CERTAIN JACK IS CENTERED ON JACK PAD BEFORE LIFTING LANDING GEAR.



JACKING AT
MAIN LANDING GEAR TRUCK



JACKING AT
MAIN LANDING GEAR STRUT

Airplane Landing Gear Jack Points
Figure 6-19

Lifting with Pneumatic Bags

In case of soft terrain or a wheels-up landing, the airplane can be lifted by using pneumatic bags. Place the bags under structural points of the wings and fuselage and inflate evenly, holding the airplane as level as possible. Sharp objects on the airplane surfaces or on the ground must be removed to prevent puncturing the bags.

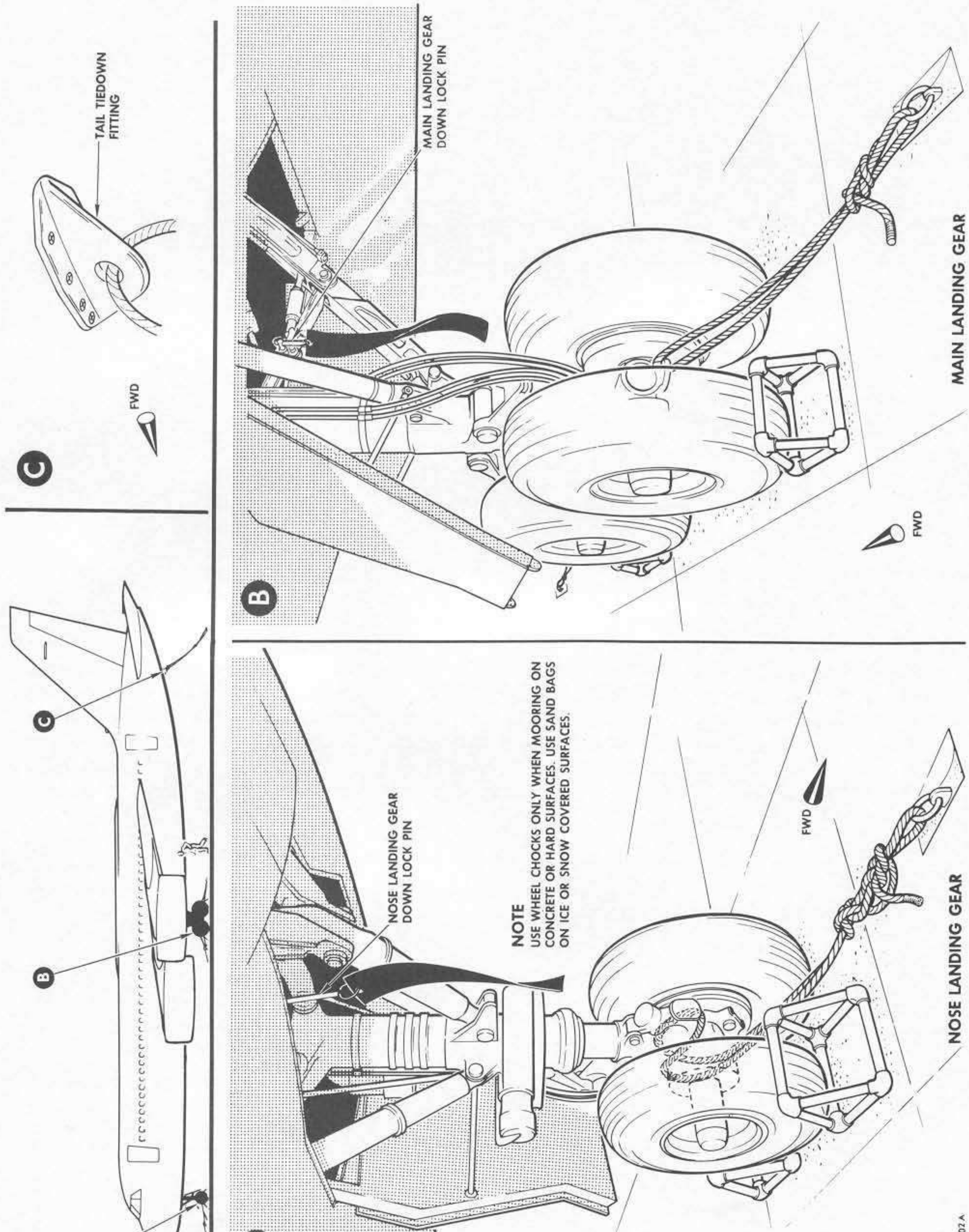
CAUTION: MOOR THE AIRPLANE LATERALLY AND LONGITUDINALLY BEFORE INFLATING BAGS. ALL PERSONNEL AND EQUIPMENT NOT ACTIVELY ENGAGED IN THE OPERATION SHOULD BE REMOVED OR RESTRAINED TO A SAFE DISTANCE.

AIRPLANE MOORING PROVISIONS

Mooring provisions are provided on the forward and aft main gear trucks and on the tail skid (see Figure 6-20). Tie-down is accomplished by utilizing manilla rope or aircraft cable between the mooring lugs and the tie-down points on the ramp. Integral control surface gust dampers are provided in the flight control system to protect the airplane control surfaces from wind damage. The rudder is provided with a gust lock.

AIRPLANE PARKING

When parking the airplane for a short time, the parking brake should be set by depressing the brake pedals and pulling the parking brake handle ON. Always install landing gear down-locks, chocks and static ground cables. When unfavorable weather prevails, care should be taken to shield the airplane as much as possible and head it into the wind. During windy and dusty weather, window and door openings should be closed and safety covers installed where necessary. Extreme winter weather conditions will require ice and snow shields over the windshields, wings and tail surfaces. Precautions should also be taken to prevent the airplane tires from freezing to the ramp. Parking brakes should NOT be set in cold weather as an accumulation of moisture may cause them to freeze.



Airplane Mooring Procedure
Figure 6-20

7 FLIGHT CONTROL SYSTEMS

Section 7

FLIGHT CONTROLS

TABLE OF CONTENTS

TITLE	PAGE
GENERAL FLIGHT CONTROL SYSTEMS	7-1
Primary Flight Controls	7-1
Secondary Flight Controls	7-1
FLIGHT CONTROL HYDRAULIC SYSTEMS	7-3
Flap Hydraulic System	7-3
Spoiler Hydraulic System	7-3
Horizontal Stabilizer Trim Hydraulic System	7-3
Rudder Hydraulic System	7-6
AILERON CONTROL SYSTEM	7-6
Aileron and Aileron Flight Tab Travel	7-6
Aileron Ground Check System	7-8
Aileron Balance Panels	7-8
Aileron Trim System	7-8
SPOILER CONTROL SYSTEM	7-8
Separate Spoiler Control	7-10
Spoiler Travel	7-10
ELEVATOR CONTROL SYSTEM	7-10
Elevator, Elevator Tab, and Auxiliary Tab Travel	7-11
Elevator Pressure Balance Boards	7-11
Elevator Flutter Dampers	7-11
HORIZONTAL STABILIZER TRIM SYSTEM	7-11
Stabilizer Travel and Trim Rates	7-14
Normal Manual Trim System	7-14
Normal Electric Trim System	7-14
Emergency Electric Trim System	7-16
Emergency Manual Trim System	7-16
Horizontal Stabilizer Trim Indicators	7-16
Autopilot Trim Control	7-16
Mach Trim Augmentation System	7-19
Horizontal Stabilizer Position Warning Horn	7-19
RUDDER CONTROL SYSTEM	7-19
Rudder and Rudder Flight Tab Travel	7-23
Rudder Pedals	7-23
Rudder Pedal Gust Lock	7-25
Rudder Feel System	7-25
Rudder Control Package	7-27
Rudder Hydraulic Actuating Cylinders	7-29
Rudder	7-30
Flight Tab	7-30

TABLE OF CONTENTS (CONT)

TITLE	PAGE
RUDDER CONTROL SYSTEM (CONT)	
Rudder Hydraulic Control System	7-30
Rudder Warning System	7-32
Autopilot	7-33
Rudder Trim System	7-33
Rudder Trim and Trim Tab Travel	7-33
FLAP CONTROL SYSTEM	7-34
Flap Test Switch	7-34
Flap Warning Horn and Lights	7-34
Flap Asymmetrical Shutoff Switch	7-34
Flap Position Indicators	7-37
WING LEADING EDGE SLATS AND KREUGER FLAPS	7-37
Slat and Kreuger Position Lights	7-39
CONTROL SURFACE GUST DAMPERS	7-39
Gust Damper Locations	7-39
MACH/AIRSPEED NEVER EXCEED WARNING SYSTEM	7-42

GENERAL FLIGHT CONTROL SYSTEMS

The Convair 990, flight controls consists of four primary and seven secondary flight control systems (see Figure 7-1). The primary systems include the rudder, elevator, aileron and spoiler controls. The secondary systems include all the primary control surface trim systems, the wing flaps, Kreuger flaps, leading edge slats, the stabilizer, and the speed brake function of the spoilers.

Primary Flight Controls

All primary flight control systems except the spoilers, and the rudder in the normal mode of operation, utilize a flight tab arrangement such that pilot action at the controls moves a flight tab, causing the main control surface to move in the desired direction. Due to the large mechanical advantage of the flight tab over the main control surface, plus the mechanical advantage of the pilot over the tab, the large forces required to move the main control surfaces are overcome by relatively small pilot input forces. Balance panels are used to reduce the main control surface hinge moment to minimize pilot input to the aileron and elevator control systems. The rudder system, in the normal mode of operation, is power-operated by the two hydraulic systems. The spoiler system performs functions in both the primary and secondary systems. The primary function is to provide the principal means of lateral control of the airplane. The spoilers are augmented by the aileron system.

Secondary Flight Controls

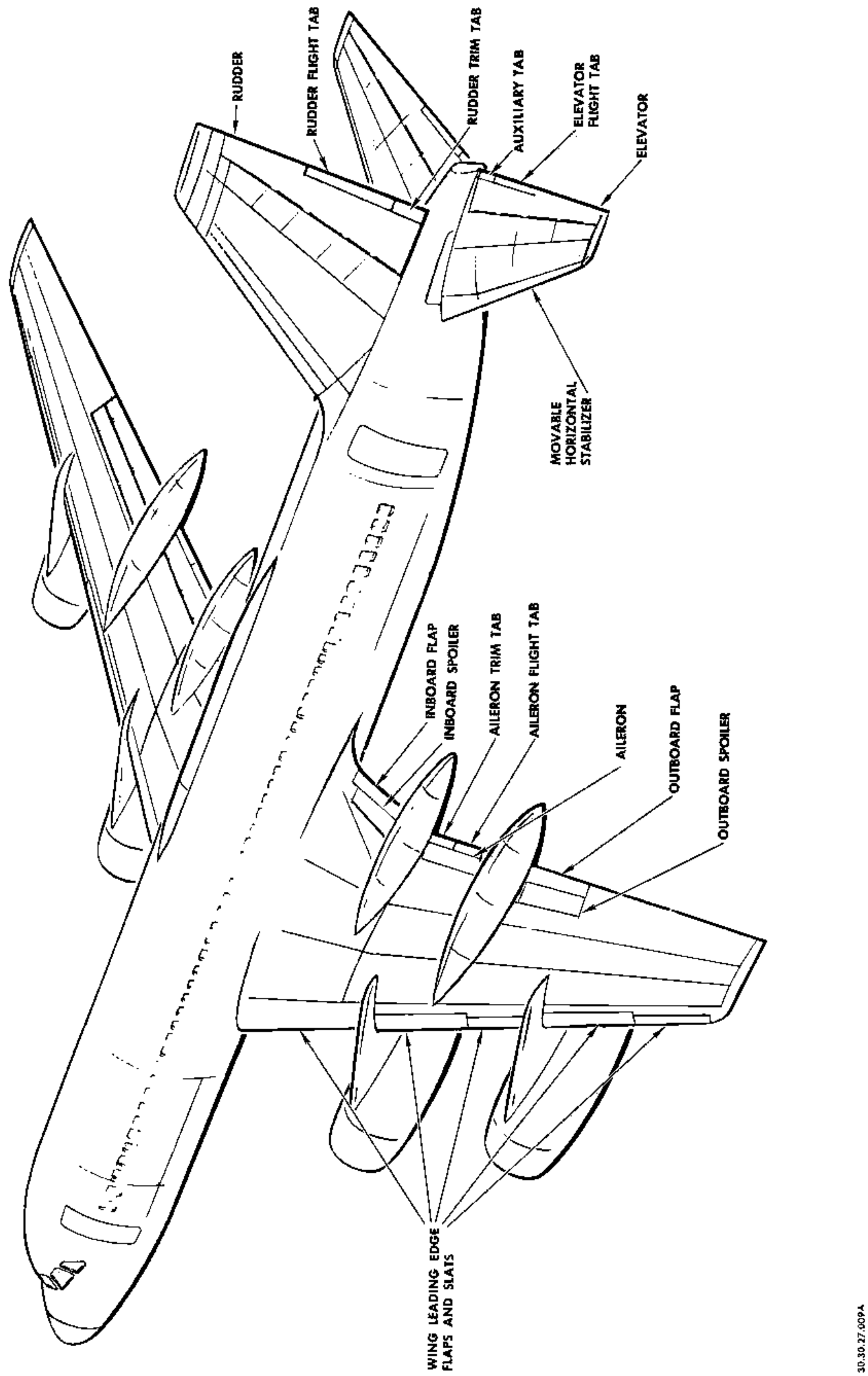
The aileron and rudder trim systems use conventional trim tabs positioned by moving control knobs in the flight compartment. Instead of elevator trim tabs, the horizontal stabilizer moves as a unit for airplane longitudinal trim. The stabilizer is mechanically controlled and hydraulically operated during normal operation. An emergency trim system can be operated either electrically from the flight compartment or mechanically from the aft lavatory. The hydraulically operated stabilizer trim system is also electrically controlled by a sliding button on each control wheel.

In the secondary control system, spoiler function is to act as an airplane speed brake. Moving a speed brake control handle in the flight compartment to the desired setting actuates the spoilers for symmetrical speed brake use. The inboard and outboard spoilers can be raised independently by means of an electric actuator mechanism to provide emergency longitudinal trim.

Double-slotted Fowler type flaps are provided on the wing trailing edge. Each wing incorporates two hydraulically operated flaps that are controlled from the flight compartment.

Four track-mounted, extensible leading edge slats and three Kreuger flaps are installed on the leading edge of each wing, outboard of the inboard pylons. Extension and retraction is controlled by the flap control lever.

A ground gust protection system is included as part of the secondary flight control system to prevent wind damage to primary control surfaces. The system is a permanent installation requiring no manual control or selection, and does not interfere with normal control surface movement.



Flight Control System
Figure 7-1

FLIGHT CONTROL HYDRAULIC SYSTEMS

Hydraulic pressure is used in the flight control system to operate the flaps, the rudder, the leading edge flaps and slats, spoilers, and the horizontal stabilizer trim system. Figures 7-2 and 7-3 show schematic diagrams of the flight control hydraulic systems. The wing flaps can be operated by either the number 1 or number 2 hydraulic system, as selected by the pilot.

Flap Hydraulic System

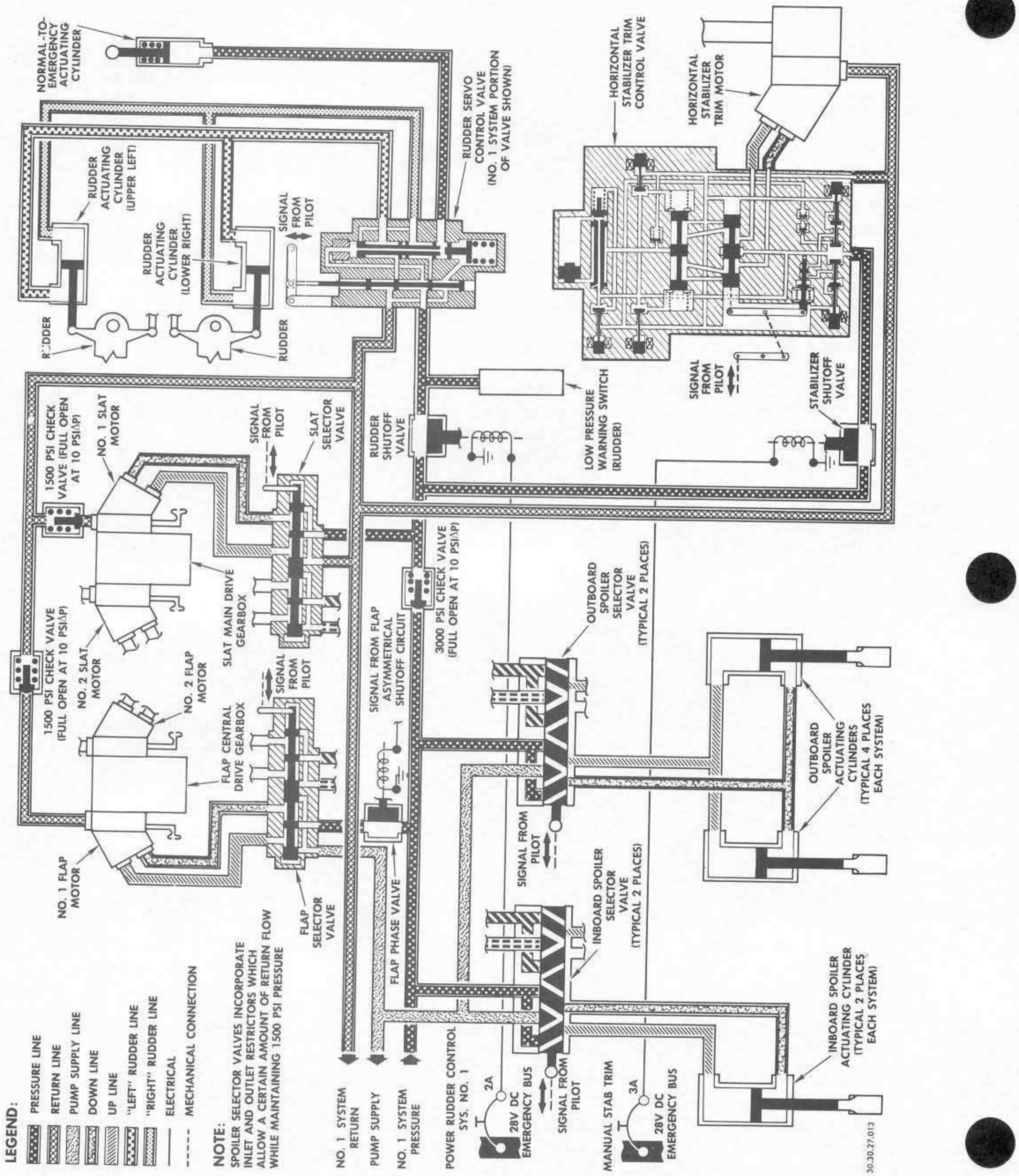
The flaps are moved to the desired position when the pilot moves the flap control lever on the pilots' pedestal to a selected position corresponding to the degree of flap extension desired. When the flap control lever is moved, the flap selector valve is positioned by means of cables, pulleys, quadrants and push-pull rods to allow the selected hydraulic system to supply pressure to its associated motor. The motor drives the main flap drive gearbox. This force is transmitted to torque tubes that rotate screw jacks, mechanically raising or lowering the flaps. The flaps will assume the position selected automatically, without further attendance, by reason of a followup system connected to the selector valve. The selector valve returns to a neutral position when the selected degree of extension or retraction has been reached. Should an asymmetric flap condition occur, a flap phase valve acts as a shutoff valve to shut off system hydraulic pressure electrically by action of asymmetric sensing units. The sensing units are installed on the outboard end of each torque tube assembly. The flap hydraulic system selector switch, placarded NORMAL # 1-ALTERNATE # 2, is located on the pilots' overhead switch panel.

Spoiler Hydraulic System

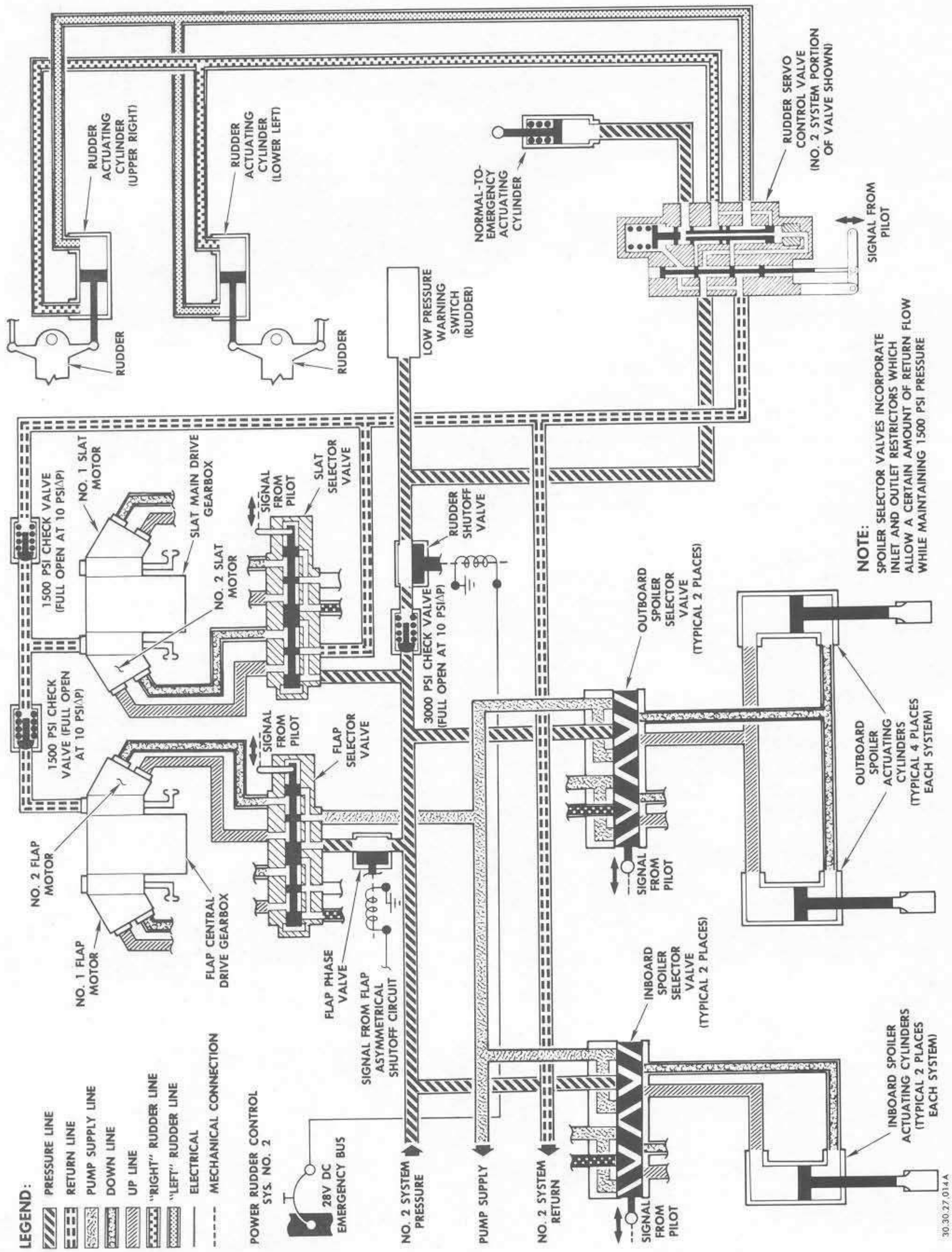
Each inboard spoiler has two hydraulic actuating cylinders and each outboard spoiler has four cylinders. The cylinders are in pairs and are located beneath the spoilers. Each main hydraulic system supplies pressure to one cylinder in each set. The actuating cylinders move the spoilers up or down as required. There is a servo valve assembly for each of the spoilers which is used to control hydraulic pressure at the actuating cylinders. Movement of a control lever on the pilots' pedestal operates a cable and pulley system in the aileron-spoiler mixer assembly. The rotational movement of the pulleys is converted to linear motion by push-pull rods and bell cranks. If the air loads reach a critical amount, the spoilers will partially retract, preventing damage to the spoilers surfaces. This spoiler "blow-down" is permitted by the designed internal leakage of the spoiler selector valves. The spoilers are normally operated by both No. 1 and No. 2 hydraulic systems. Should one of the main systems become inoperative, the spoilers may be operated by use of the remaining system. The spoiler operating rate will be approximately the same as normal; however, spoiler "blow-down" airspeed will be reduced slightly.

Horizontal Stabilizer Trim Hydraulic System

Only the No. 1 hydraulic system is used to operate the horizontal stabilizer trim system. There is no association with the No. 2 hydraulic system. In the normal manual mode of operation, movement of either of the trim levers on the pilots' pedestal opens a valve to port hydraulic pressure to a hydraulic motor. The motor rotates a jack shaft in the horizontal stabilizer screw jack. As the



Flight Control Hydraulic System, No. 1
Hydraulic System Pressure
Figure 7-2



Flight Control Hydraulic System No. 2
Hydraulic System Pressure
Figure 7-3

jack shaft rotates, it travels through a nut attached to the stabilizer leading edge, causing the stabilizer to move up or down. When the stabilizer reaches the desired setting, releasing the trim lever returns the hydraulic valve to the neutral position and stops the flow of hydraulic fluid to the hydraulic motor.

In the normal electrical mode of operation, trim is accomplished by actuating either stabilizer trim switch located on the outboard horns of the pilots' control wheels. The stabilizer trim switches actuate a hydraulic boost stage in the same hydraulic valve used in the normal manual trim system, and rotation of the jack shaft through the stationary nut on the stabilizer leading edge is identical. A standby electric trim motor is available to obtain the desired trim setting in the event of a normal system malfunction.

An electrically operated shutoff valve, controlled by a switch in the flight compartment, is incorporated in the hydraulic pressure supply line to the trim motor. Should the trim system become erratic, this valve may be actuated to shut off hydraulic pressure at the trim motor. The hydraulic motor of this system has a seal drain to avoid the collection of hydraulic fluid in its case. This eliminates the possibility of the hydraulic fluid, which might seep by the seal, damaging or causing a malfunction of the motor.

Rudder Hydraulic System

During normal flight operations, the flight tab control linkage is locked in neutral and the rudder is actuated by two pairs of hydraulic actuators. Each of the airplane's hydraulic systems supplies power to one pair of actuators and either system will provide fully effective rudder control.

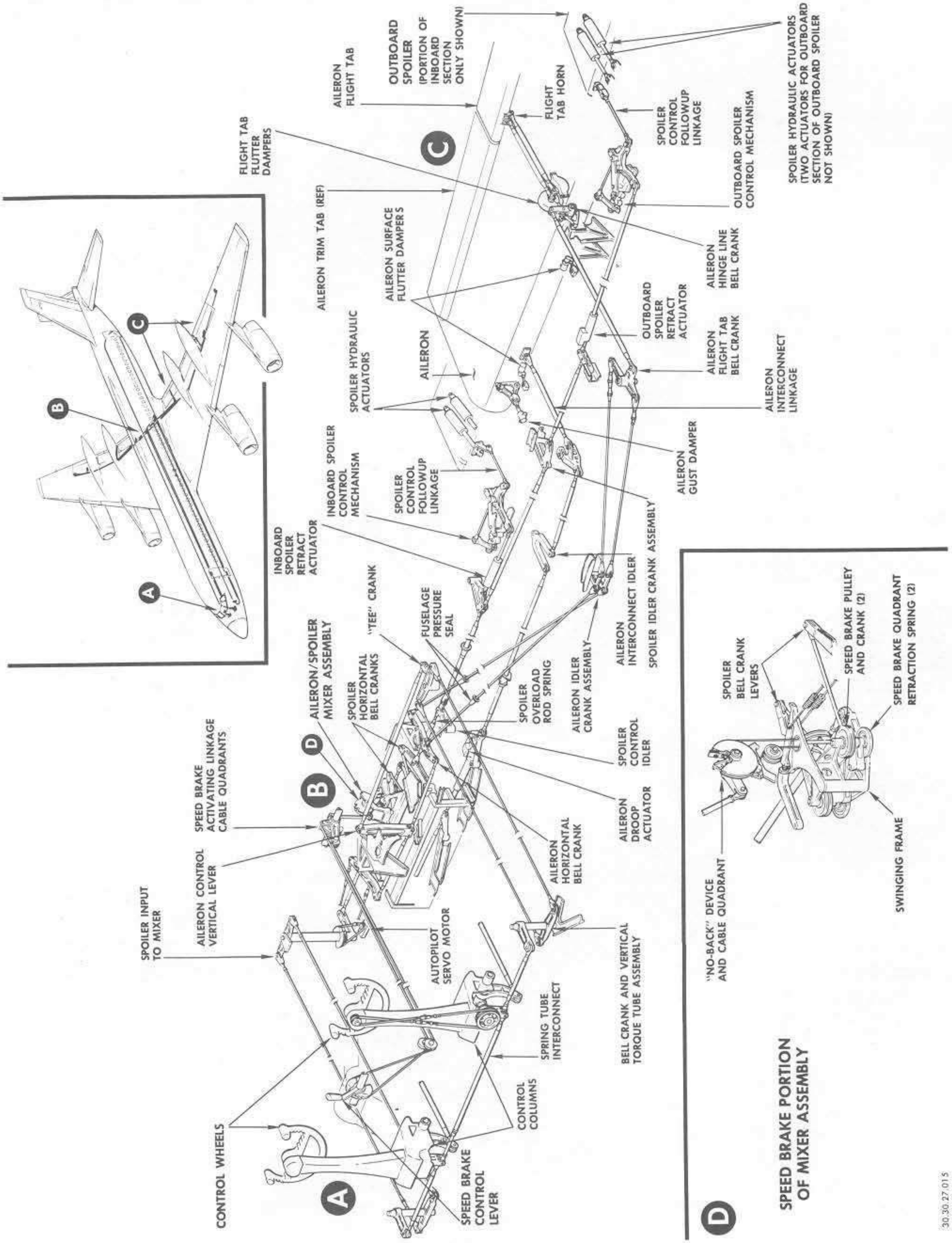
A servo valve in the hydraulic lines leading to the actuators is connected to the rudder pedals through a mechanical linkage of cables, control rods, and bell cranks. The position of the rudder at any particular time is coordinated with that of the rudder pedals by a follow up arm which shuts off the flow of hydraulic fluid to the actuators when the rudder reaches the position selected by the rudder pedals. Rudder deflection is restricted to safe limits by force limiting relief valves in the hydraulic lines leading to the actuators.

AILERON CONTROL SYSTEM

The ailerons are equipped with flight tabs which are connected by a system of bell cranks, control cables, and control rods with the pilot's control wheel. The spoiler system, which is connected to the copilot's control wheel, is interconnected with the aileron system by a spring-loaded crossover tube between the bell cranks at the base of the control columns. A spring-loaded cam and roller unit in the speed brake/spoiler mixer provides artificial feel for the aileron and spoiler control systems. A diagram of the aileron and spoiler control systems is shown in Figure 7-4.

Aileron and Aileron Flight Tab Travel

The ailerons are rigged from a position streamlined with the wings and then drooped one degree to compensate for the flexing action of the air load on the wings during flight. Aileron travel from the normal one-degree drooped position



Aileron and Spoiler Control System
Figure 7-4

is 14 (+1.0, -0.5) degrees up and down. Flight tab travel is limited by stops on the flight tab hinge line to 20 (±0.5) degrees.

Aileron Ground Check System

Since the ailerons are free-floating, with all of the movement of the aileron control wheel being utilized to move the flight tab, an iced or jammed aileron ordinarily could not be detected by the pilot when the airplane is on the ground. A ground check method has been provided by the addition of stops to the front spar of the aileron. The stops are located in such a manner that after approximately 65 degrees of aileron wheel movement, the bell crank located on the aileron hinge line, when moving for down aileron, contacts a checkout stop and as a result, moves the aileron to its full travel. Since the ailerons are interconnected, the opposite aileron will also move. However, in flight the flight tabs will cause the aileron to move and the checkout stops on the aileron will move away from the bell cranks on the aileron hinge line.

Aileron Balance Panels

Balance panels and balance weights are attached to the leading edge of the ailerons to give the ailerons aerodynamic and dynamic balance. Curtains are installed between the balance panels and the wing trailing edge structure to provide seals for the balance chambers.

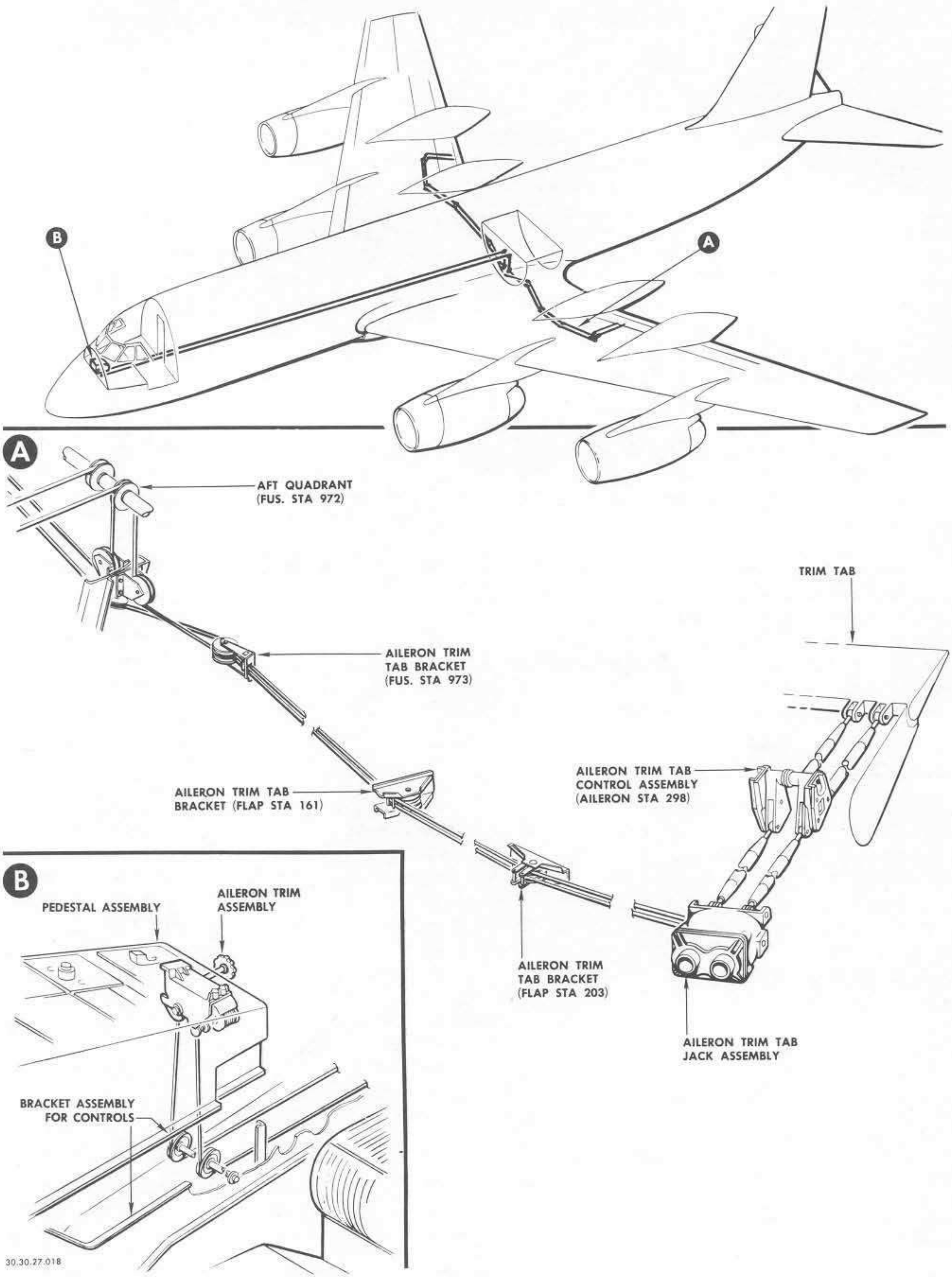
Aileron Trim System

The aileron trim tabs are controlled by a trim knob located on the aft side of the pilots' pedestal (see Figure 7-5). Movement of the aileron trim knob also positions the aileron trim indicator scale on the pilots' pedestal to provide an indication of the amount of the trim introduced. Aileron trim tab travel is 16 degrees "up" and "down" from the streamline position to 7.5 degrees with ailerons fully extended. Stops limit trim tab travel.

SPOILER CONTROL SYSTEM

Two spoilers, mounted on the upper surface of each wing, operate in conjunction with the ailerons and provide the principle means of lateral control. The spoilers are also extended symmetrically to serve a secondary function as speed brakes. Each inboard spoiler is actuated by two hydraulic actuators and is controlled by a dual hydraulic selector valve. Each outboard spoiler, composed of three sections, utilizes four hydraulic actuators. The outboard spoiler actuators are also controlled by a dual hydraulic selector valve. Selector valve operation is provided through a push-pull tube system connected to each side of the aileron-spoiler mixer assembly. An additional push-pull rod is attached between each spoiler, and a followup linkage completes the loop for closing the selector valves when the spoilers reach the selected degree of travel.

As a safety feature, power for operating the spoilers is obtained from both main hydraulic power systems. The spoilers can be extended at any flying speed; however, when the speed of the airplane exceeds approximately 220 knots IAS, "blow down" is permitted by the spoiler selector valves, thereby preventing



Aileron Trim System
Figure 7-5

structural damage due to excessive airloads. Should one of the hydraulic systems lose pressure, hinge moment will be reduced and "blow down" will occur at lower speeds; but full deflection will still be possible at approximately 150 knots IAS or below.

The spoiler and aileron systems are not interconnected at the mixer unit. The only interconnection is through the spring-loaded tube assembly connecting the bell cranks at the base of the control columns. Force applied at the left-hand control column wheel is transmitted to the aileron tabs through the linkage on the left-hand side of the airplane. The force is also carried through the spring tube assembly into the linkage on the right-hand side of the airplane to the mixer unit and finally to the spoiler control valves. The force at either aileron control wheel causes the spoiler surfaces to move.

Separate linkages for the spoiler and aileron flight tabs provide a safety feature. During normal operating conditions, either control wheel operates both systems. Should a malfunction occur causing one system to jam, the system on the opposite side can be operated by overriding the jam through the spring-loaded tube assembly. A jam in the spoiler system on the right-hand side of the airplane can be overridden by the pilot to operate the aileron system. A jam in the aileron system can be overridden by the copilot to operate the spoiler system.

Aileron-spoiler relationship is as follows: With the spoilers flush against the upper wing surface (down position), only the spoilers on the up-aileron side will move. With the spoilers full open (up position), only the spoilers on the down-aileron side will move. With the spoilers selected to speedbrakes to any position other than full up or full down, the spoiler direction of movement will be the same as with both ailerons; that is, one set will move up farther and the other set will move down farther.

Separate Spoiler Control

Either the inboard or the outboard spoilers can be raised to provide additional pitch control if the horizontal stabilizer mechanism should jam. Actuating the emergency pitch control switch on the pilots' pedestal to the UP position lowers both inboard spoilers. The drag from the raised outboard spoilers tends to raise the nose of the airplane. Placing the emergency pitch control switch in the DOWN position lowers the outboard spoilers and the drag from the raised inboard spoilers tends to lower the nose.

Spoiler Travel

The maximum deflection of the outboard spoilers is 72 degrees. The inboard spoilers can be deflected 58 degrees. Up and down spoiler travel is limited by the actuator pistons bottoming out in the actuator cylinders.

ELEVATOR CONTROL SYSTEM

The elevators are used for longitudinal control of the airplane while in flight and are controlled by a mechanical system of push-pull rods, bell cranks, and cables that connect the control columns in the flight compartment to the flight tabs on the elevators (see Figure 7-6). The right and left-hand elevator control systems are interconnected by two spring tube assemblies. One spring tube assembly is located between the first two bell cranks aft of the control columns; the other spring tube assembly is located between bell cranks in the aft section of the elevator controls. In the event of a jam in one side of the elevator control system, the use of spring tube interconnects allows the pilot on the opposite side of the jam condition to maintain partial elevator control.

An auxiliary tab mounted inboard of the flight tab on each elevator is connected to a lever on the stabilizer pivot shaft. When the stabilizer leading edge is lowered, the auxiliary tabs are deflected downward to provide additional "nose up" trim.

An elevator downspring assembly connected to the elevator flight tab "up" control cable provides a forward force on the control columns when the flaps are in the raised position. As the flaps are being lowered, a flexible drive shaft connected to the flap central drive gearbox decreases the tension on the downspring to zero by the time the flaps are fully lowered.

Elevator, Elevator Tab, and Auxiliary Tab Travel

Elevator travel is 25 degrees "up" and 12 degrees "down" from the streamline position. The elevator gust damper also acts as a limit stop for the elevator. Flight tab travel is 12 degrees "up" and 25 degrees "down" from the streamline position. Travel is limited by flight tab stops. Auxiliary tab travel is from 4 degrees to 25 degrees trailing edge down.

Elevator Pressure Balance Boards

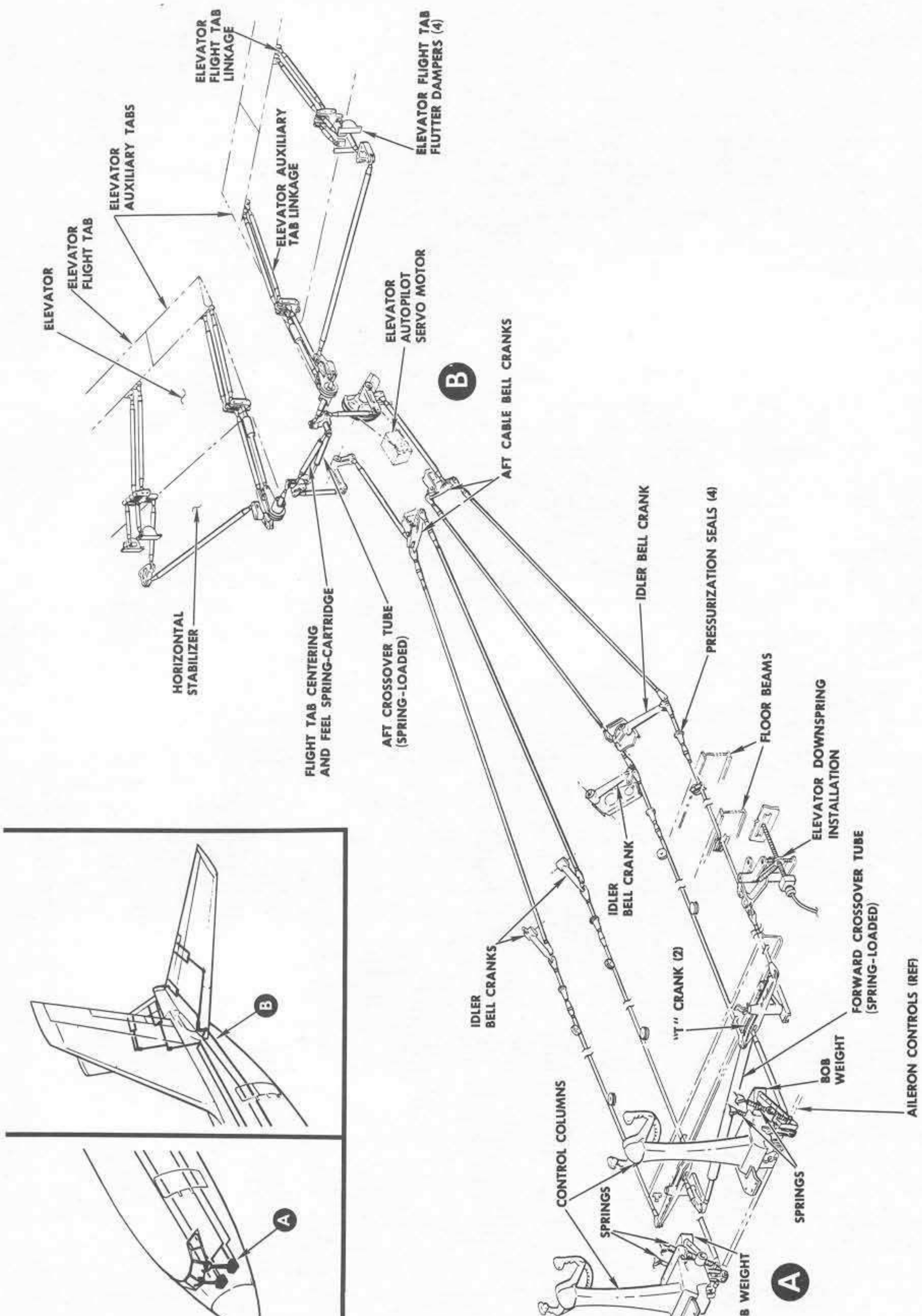
Two pressure balance boards are located forward of each elevator in pressure boxes in the trailing edge of the stabilizer (see Figure 7-7). The aft edge of each balance board is connected to the leading edge of the elevator. A linkage attached to an "A" frame on the front spar of the elevator and a lever on the pressure board moves the pressure board upward when the elevator is deflected. Air pressure from openings in the leading edge of the stabilizer exerts an upward force on the pressure boards and a deflecting force on the elevators when they are moved up or down from streamline.

Elevator Flutter Dampers

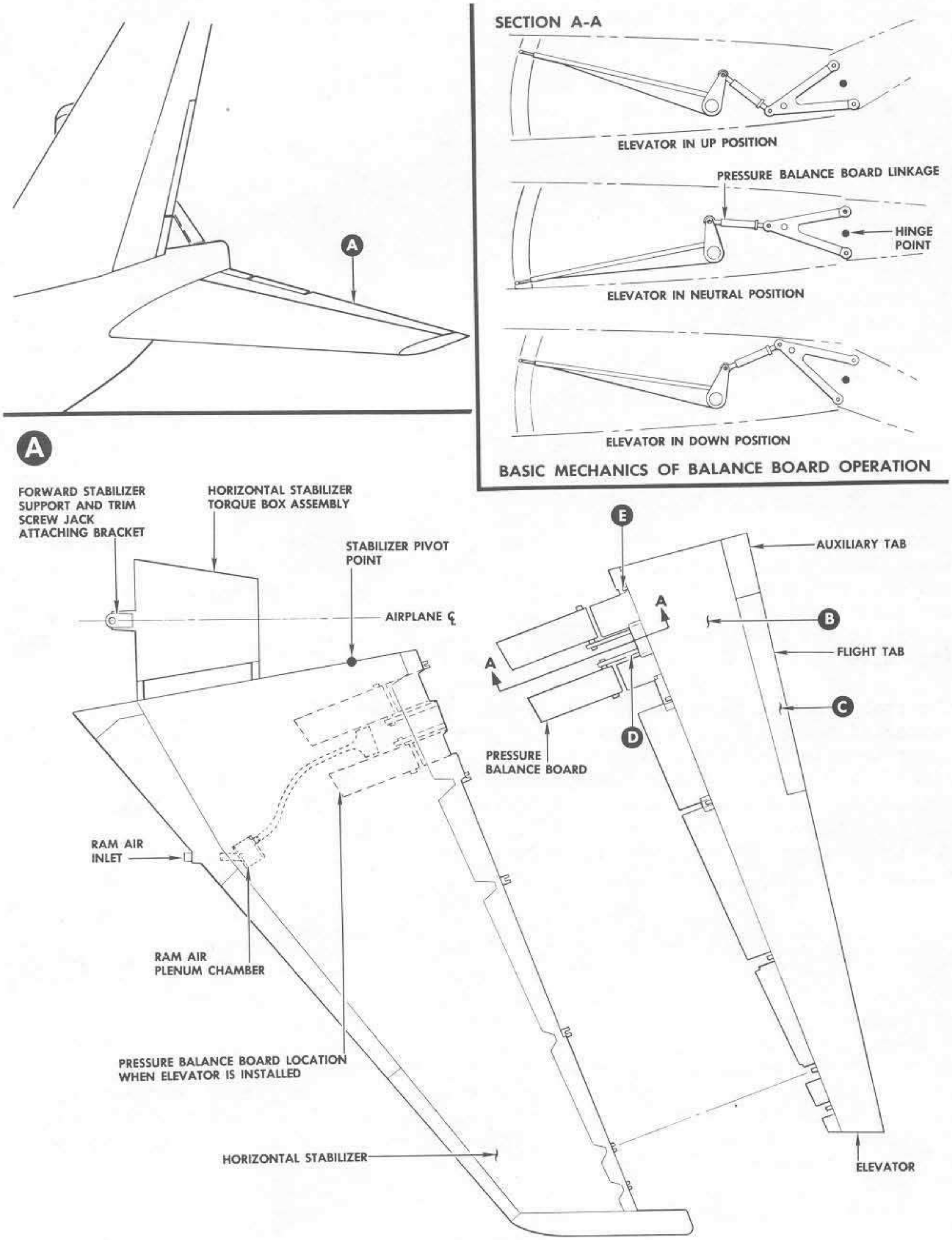
The flight tabs have hydraulic type flutter dampers attached to the linkage near each tab. The dampers eliminate any tab flutter which might occur in flight. The elevator surfaces have hydraulic type flutter dampers.

HORIZONTAL STABILIZER TRIM SYSTEM

In order to avoid the problems encountered in using an elevator trim tab at high airplane speeds, a completely movable horizontal stabilizer is used. Moving the entire stabilizer also provides the large trim moments necessary at takeoff and landing, and the design is aerodynamically clean at cruise speed. The normal mode of horizontal stabilizer operation consists of three subsystems: Electric trim control, manual trim control and autopilot trim control. The emergency mode of operation is divided into two subsystems: Electric trim



Elevator Control System
Figure 7-6



Elevator Installation
Figure 7-7

control and manual trim control. An irreversible Acme type screw jack assembly regulates the positioning of the stabilizer during all conditions of normal, emergency, and Mach trim system operations.

A Mach trim augmentation system adjusts the stabilizer automatically to counteract the tendency of the airplane to become "nose heavy" at high airspeeds. The trim setting of the stabilizer is indicated by a stabilizer trim indicator on the pilots' instrument panel. A stabilizer motion indicator on the forward LH side of the pilots' pedestal gives a visible and audible indication of any change in stabilizer setting.

Stabilizer Travel and Trim Rates

The stabilizer trim range extends from 2.50 degrees leading edge up to 15 degrees leading edge down. Stabilizer trim rate varies with the trim mode and the position of the flaps. In the normal manual or electric mode, the trim rate is 0.53 degree per second when the flaps are extended more than 15 degrees. With the flaps extended less than 15 degrees the trim rate is 0.25 degree per second. The emergency electric trim rate is 0.12 degree per second and on autopilot control the trim rate is 0.06 degree per second.

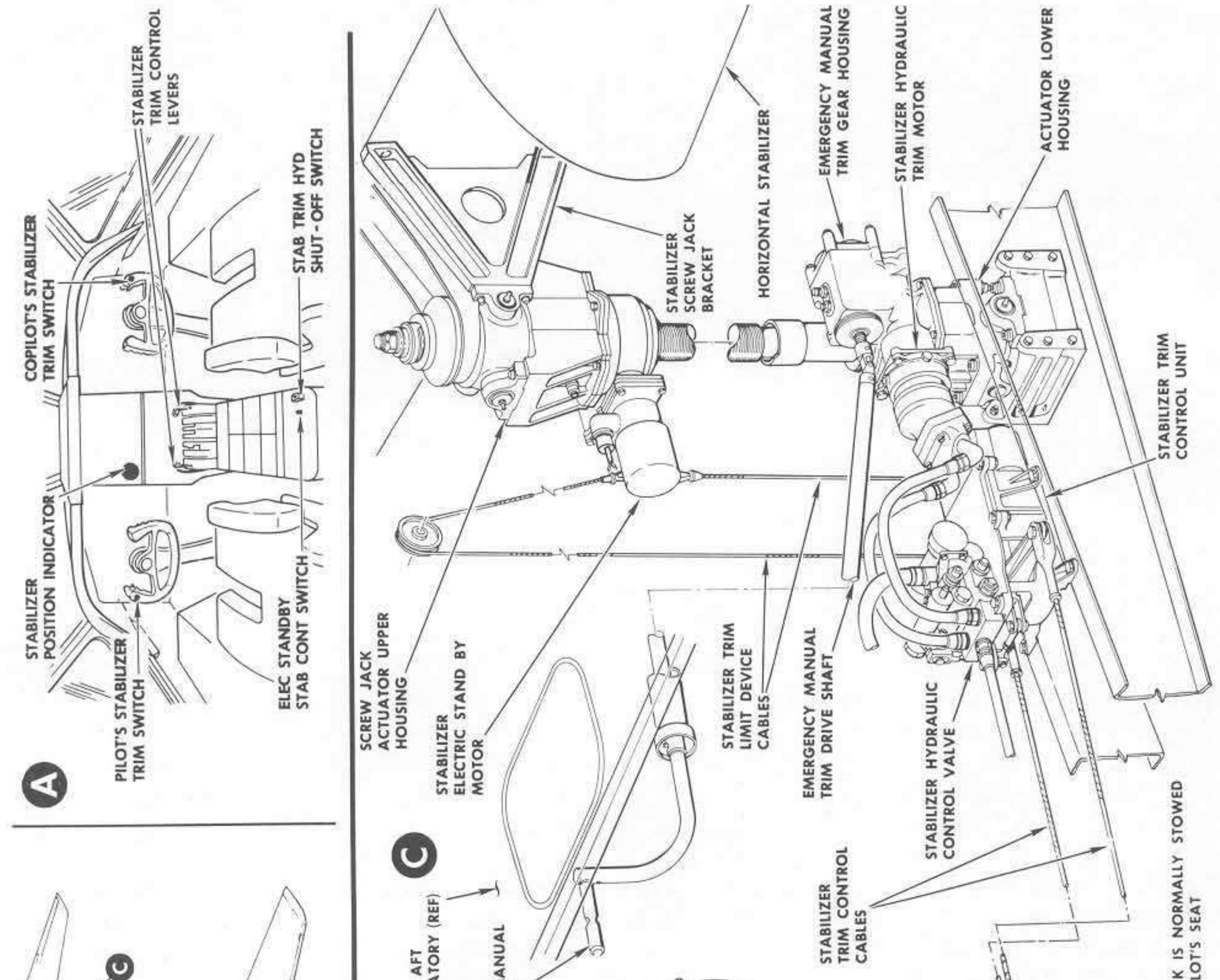
Normal Manual Trim System

During normal operation, the stabilizer is trimmed by moving the trim levers on the pilots' control pedestal or by actuating the sliding switches on the outboard horns of the control wheels. (See Figure 7-8.) The trim levers are connected by a cable system to a bell crank on the stabilizer control unit adjacent to the stabilizer actuator. This bell crank is connected to the hydraulic valve that regulates the flow of hydraulic fluid to the actuator motor.

The stabilizer actuator motor is powered by the No. 1 hydraulic system and is connected mechanically with the screw jack which rotates in the fixed nut in the upper actuator housing attached to a bracket at the leading edge of the stabilizer. Moving the cockpit trim levers forward positions the hydraulic valve to rotate the hydraulic motor and the screw jack in the direction that will raise the leading edge of the stabilizer and lower the nose of the airplane. Moving the trim levers aft reverses the flow of hydraulic fluid through the motor and raises the nose of the airplane.

Normal Electric Trim System

Trim switches located on the outboard horns of the control wheels can be used to change the stabilizer trim setting (see Figure 7-9). These switches are connected electrically to the boost stage of the hydraulic control valve. Sliding either switch upward lowers the nose of the airplane; sliding either switch downward raises the nose of the airplane. The boost stage of the hydraulic control valve actuates the entire normal manual trim linkage and the manual trim levers respond when the electric trim switches are actuated. The electric trim switches should not be actuated when the stabilizer is being trimmed by the trim levers.



Horizontal Stabilizer Trim Control System
Figure 7-8

Emergency Electric Trim System

In case of hydraulic failure, the stabilizer can be adjusted by utilizing an electric motor mounted on the upper housing of the stabilizer actuator. The operation and direction of rotation of the electric motor is controlled by a momentary-contact ELEC STDBY STAB CONT switch mounted on the aft RH side of the pilots' pedestal. Holding the switch in the forward position, which is placarded NOSE DOWN, raises the leading edge of the stabilizer and lowers the nose of the airplane. Holding the switch aft in the NOSE UP position adjusts the stabilizer to raise the nose of the airplane.

Before the emergency electric system can be utilized, the STAB TRIM SEL switch on the pilots' pedestal must be actuated to the OFF position to arm the ELEC STDBY STAB CONT switch and close the hydraulic shutoff valve stabilizer trim hydraulic line. Limit switches control the travel limits of the stabilizer in the emergency electric mode.

Emergency Manual Trim System

In case of both electric and hydraulic power failure, the stabilizer can be adjusted manually with a hand crank which is stowed under the copilot's seat. The first steps in emergency manual trim operation are to position the STAB TRIM SEL switch on ELEC STBY and to insert the lock pin from the hand crank in the pilot's manual trim lever. The manual drive shaft is located behind an escutcheon plate on the wash basin counter apron in the LH aft lavatory. The drive is engaged by removing the plate, inserting the hand crank, and pushing the shaft into engagement with the stabilizer actuator drive.

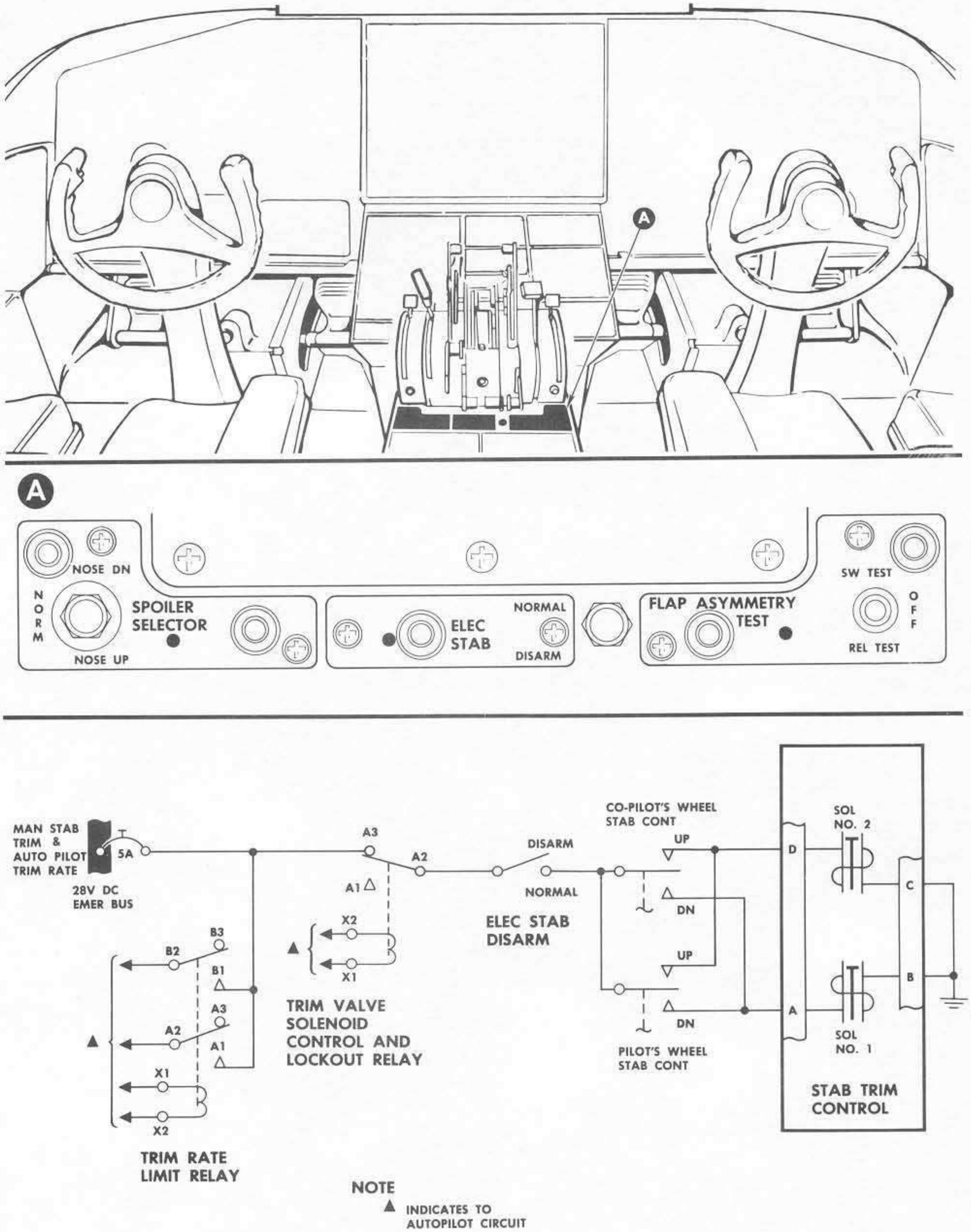
The person operating the hand crank must establish and maintain interphone communications with the pilot to receive instructions. CW rotation of the hand crank raises the leading edge of the stabilizer and lowers the nose of the airplane; CCW rotation has the opposite effect. Approximately 600 turns of the hand crank are required to trim the stabilizer through its full range of travel. Precautions must be taken to ensure that hydraulic power is not applied while the hand crank is installed and in use.

Horizontal Stabilizer Trim Indicator

A stabilizer position indicator dial is located on the pilot's instrument panel. This indicator is a synchro motor operated by a synchro transmitter located on the right drag longeron in the tailcone of the airplane. An arm on the input shaft of the transmitter is connected to one end of a push rod; the other end of the push rod is connected to the rear spar of the horizontal stabilizer torque box in the fuselage.

Autopilot Trim Control

Engaging the autopilot switches autopilot trim into the normal electric trim circuit and actuates the boost stage of the hydraulic control valve. Autopilot longitudinal trim action is noticable in the cockpit by the small movements of the manual trim levers and the visible and audible indications of the stabilizer



30.02.27.110

Normal Electrical Horizontal Stabilizer Trim Control System
Figure 7-9

motion indicator. Autopilot stabilizer trim control can be overridden by normal manual trim at the pilots' discretion. Normal electrical trim is "locked out" during autopilot operation.

Mach Trim Augmentation System

The Mach trim augmentation system consists of the Mach computer, Mach trim augmentation amplifier, the Mach trim power supply, and the Mach trim portion of the stabilizer trim control valve. The power supply; amplifier, and computer are mounted on the autopilot equipment rack at station 341. The Mach trim augmentation system adjusts the stabilizer automatically to counteract the tendency of the airplane to become nose heavy at speeds between 0.80 and 0.95 Mach.

Pitot and static air pressures are routed to the Mach computer. The information is computed and then transformed from its natural analog function into digital signals (step or pulse voltage signals). These signals are used by the amplifier to control the Mach trim portion of the horizontal stabilizer trim control valve. On command from the amplifier the control valve releases a measured quantity of hydraulic fluid to the stabilizer hydraulic motor. The motor then drives the stabilizer leading edge a specified number of degrees (0.072 degrees per signal) in the direction commanded by the Mach computer. The leading edge of the stabilizer is lowered for increased Mach and raised for decreasing Mach. The system is interlocked to a "standby" condition during autopilot controlled flight.

A test button is provided for simulating increasing or decreasing Mach number for pre-flight check of the Mach trim augmentation system.

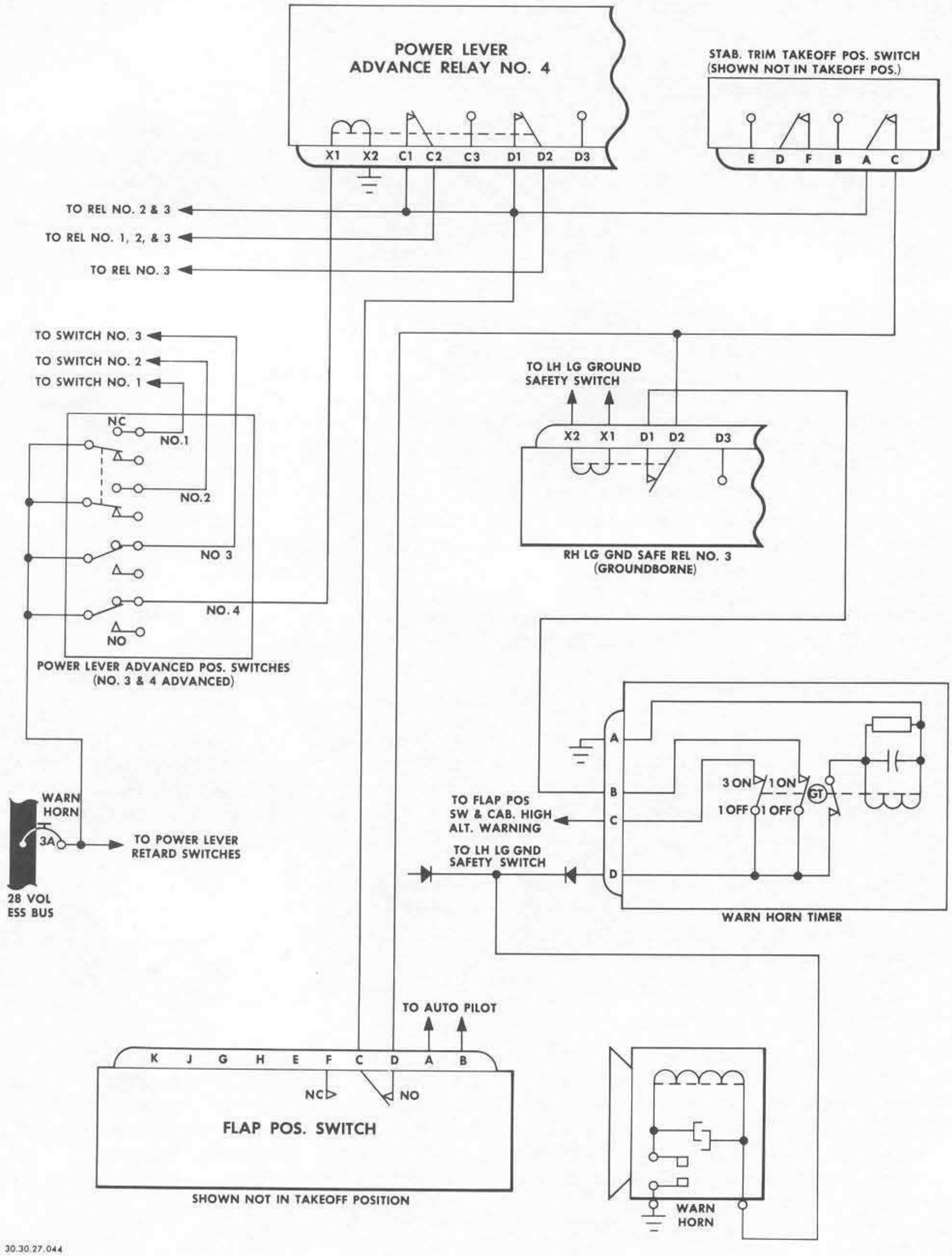
Horizontal Stabilizer Position Warning Horn

A cam located on the arm of the synchro transmitter of the stabilizer position indicator closes a switch in the warning horn circuit whenever the stabilizer trim setting is between 2-1/2 degrees leading edge up and 2 (±1.0) degrees leading edge down or between 6 (±1.0) and 15 degrees leading edge down (see Figure 7-10).

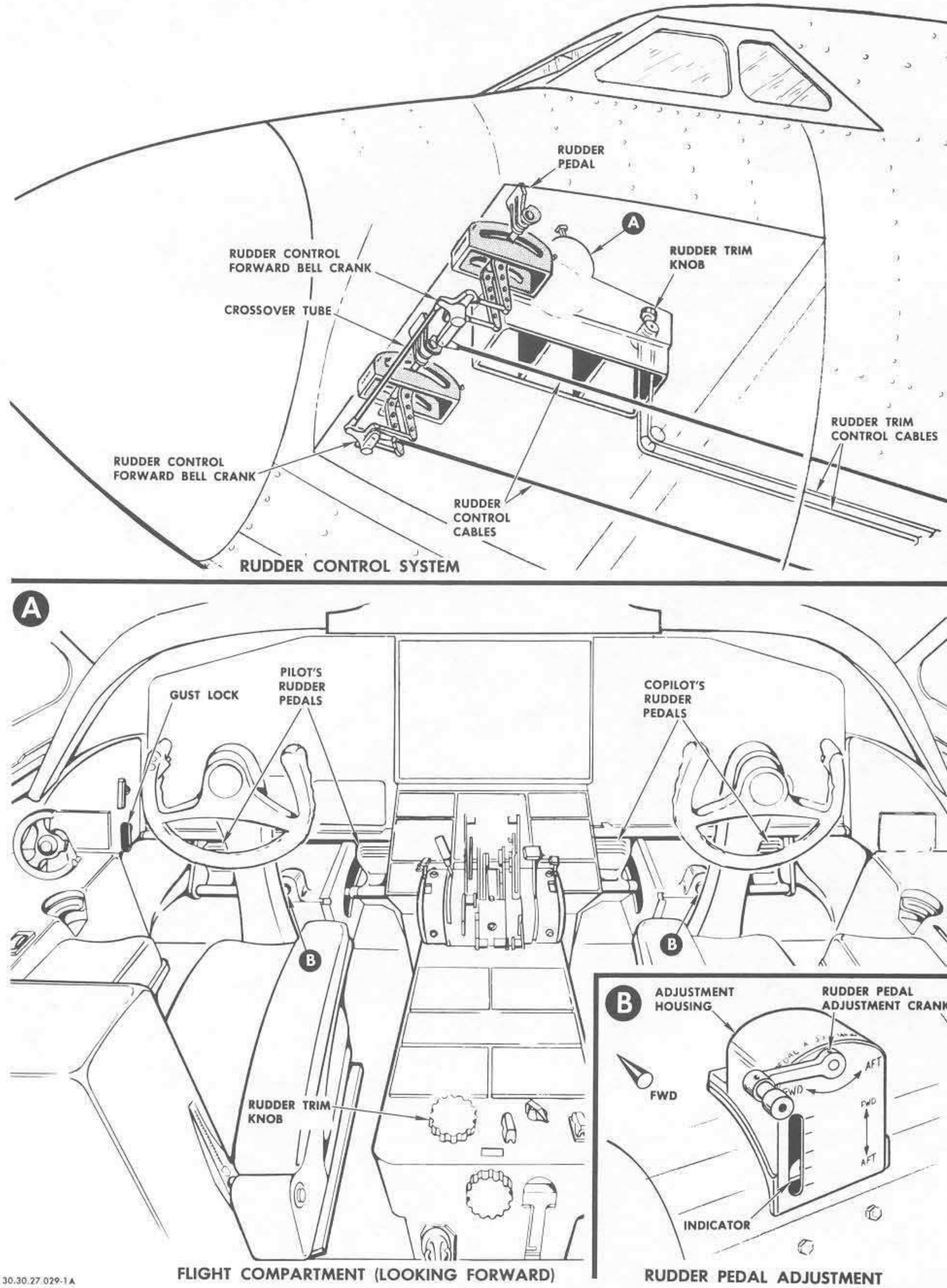
A warning horn will sound "one second on - one second off" if two or more throttle levers are advanced to TAKE OFF position when the airplane is on the ground if the stabilizer is not positioned between 2 (±1.0) and 6 (±1.0) degrees airplane NOSE UP.

RUDDER CONTROL SYSTEM

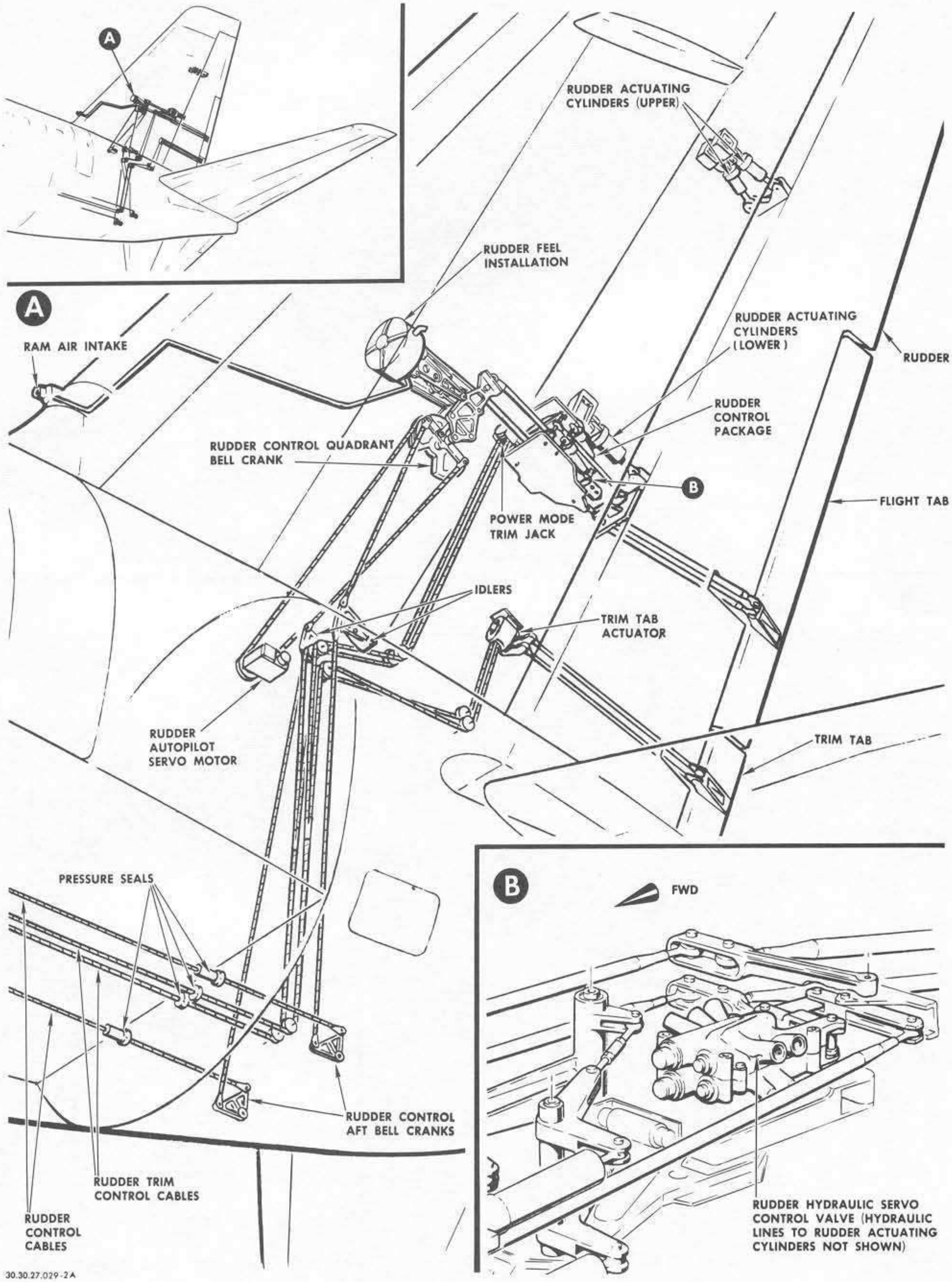
The rudder control system consists primarily of adjustable rudder pedals, "feel" system, control package, hydraulic control system, rudder warning system, flight tab, and rudder. There are two modes of rudder operation; the normal or power mode and the manual or stand-by mode. Rudder pedal movement is transmitted to a servo valve on normal mode, or to the rudder flight tab in manual mode, through a mechanical linkage consisting of cables, bell cranks, and control rods (see Figure 7-11).



Stabilizer Warning Horn Schematic
Figure 7-10



Rudder Control System
Figure 7-11 (Sheet 1 of 2)



Rudder Control System
Figure 7-11 (Sheet 2 of 2)

A single 1/8-inch non-flex cable extends aft from each rudder control forward bell crank beneath the cabin floor and through seals in the pressure bulkhead to a rudder control aft bell crank in the tail cone. Cables extend upward from the aft bell cranks to two cable idlers in the vertical stabilizer. From the idlers, the cables are routed up to the rudder control quadrant bell crank to which the autopilot rudder servo cables are also attached. Rudder pedal action is transmitted from the quadrant bell crank to the upper bell crank and from there to mixer No. 2 in the rudder control package by push-pull rods. In the normal mode of operation, the flight tab control linkage is locked on the neutral position by the manual reversion mechanism in the control package and the rudder is actuated by two pairs of hydraulic actuators. Each hydraulic system supplies power to one pair of actuators and either system will provide fully effective rudder control (see Figure 7-12). The dual servo valve in the control package provides control of the hydraulic actuators. A followup linkage, to coordinate rudder deflection with rudder pedal displacement, is formed by pivoting the servo valve actuating bell crank on the forward spar of the rudder. When the rudder is deflected hydraulically, anti-servo action causes the flight tab to deflect an equal amount in the same direction.

Rudder deflection at high airspeeds is confined to safe limits by force limiting relief valves in the hydraulic control system. Rudder feel is simulated by a "Q" cylinder which varies resistance to rudder pedal movement to correspond to variations in airspeed.

The rudder hydraulic actuators act as mechanical stops for the rudder and on the ground the actuators function as gust dampers. A gust lock on the pilot's auxiliary instrument panel can be set to lock the rudder pedals in neutral and provide additional protection for the rudder.

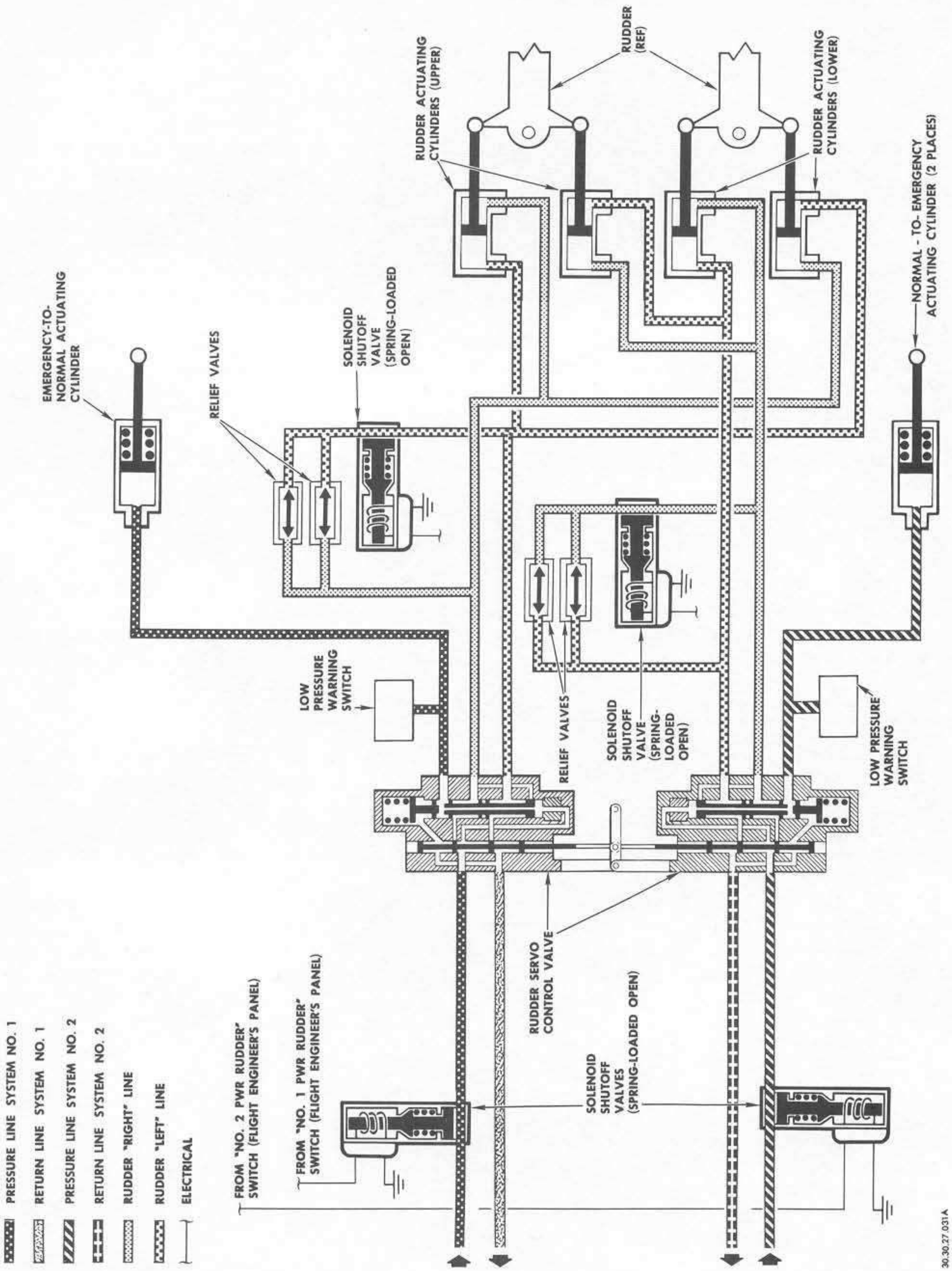
Rudder and Rudder Flight Tab Travel

Full rudder travel in normal operation is 25 degrees to the right and left of streamline. In the manual mode, rudder travel is limited aerodynamically to approximately 14 degrees left and right of neutral.

Full rudder flight tab travel is limited to 26.25 degrees in relation to the rudder surface. In normal operation, the flight tab control is locked in neutral and the flight tab is deflected the same amount and in the same direction as the rudder by the anti-servo action. In the manual mode, the flight tab is controlled through the mechanical linkage to the rudder pedals and the rudder surface is deflected by aerodynamic forces acting on the deflected flight tab.

Rudder Pedals

The pilot's and copilot's rudder pedals are pivoted on the upper ends of swinging arms hinged below the flight compartment floor. Push-pull tubes connect the lower ends of the rudder pedal arms to forward bell cranks interconnected by a crossover tube. Each pair of rudder pedals can be adjusted fore and aft to suit the pilot's individual preference by rotating a hand crank on the aft face of the rudder pedal cover. Rudder pedal travel is limited to 3.87 inches forward from neutral by rudder pedal cockpit stops.



Rudder Control Hydraulic System Schematic
Figure 7-12

Rudder Pedal Gust Lock

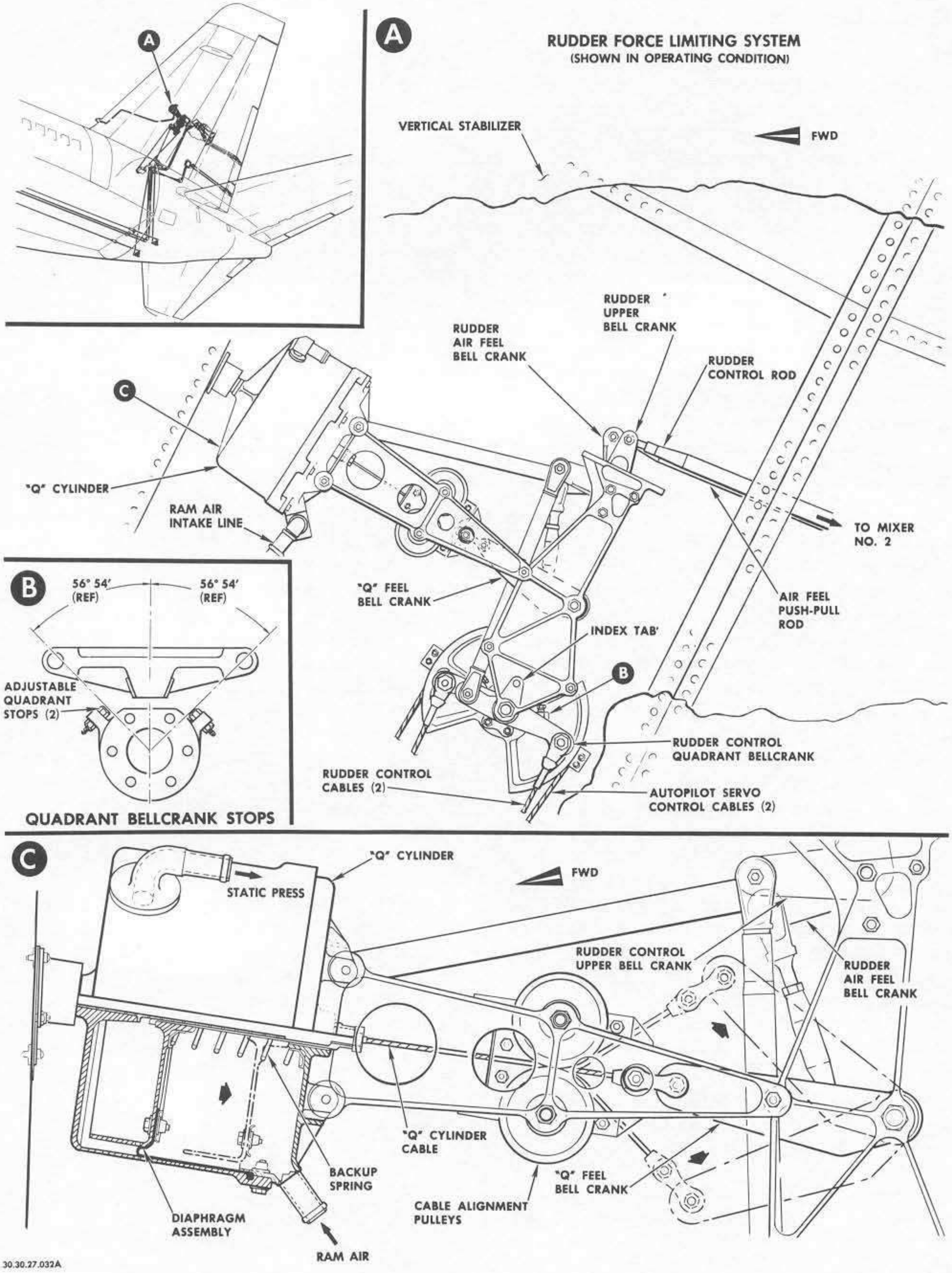
The rudder pedal gust lock consists of a control handle, a spring cartridge, a shaft assembly, and locking pawls. The control handle is mounted in the pilot's auxiliary instrument panel and is connected by cable through a spring-loaded cartridge to a quadrant on the shaft assembly. Bell cranks on the ends of the shaft assembly are connected through links to the locking pawls. Interlocking springs extend forward from the shaft assembly bell cranks and are connected to structural members.

Pulling the gust lock handle rotates the shaft assembly bell cranks and drives the locking pawls against the bottom ends of the pilot's rudder pedal arms. The spring cartridge allows the handle to be pulled out and locked in a position in which it interferes with the operation of the steering wheel. When the gust lock control handle is released, the unlock springs rotate the shaft assembly to the unlock position. The gust lock restricts the movement of the rudder pedals and the flight tab linkage only. The rudder surface is protected against gusts by the damping effects of the hydraulic actuators.

Rudder Feel System

The rudder feel system consists primarily of a "Q" cylinder installed in the vertical stabilizer, a "Q" feel bell crank installed between the rudder control quadrant and the rudder upper bell crank, and an air feel bell crank installed adjacent to the rudder upper bell crank. The "Q" cylinder is divided into a forward and an aft chamber by a piston which is sealed to the cylinder wall by a rolling curtain seal. The forward chamber is vented to ambient air pressure in the vertical stabilizer; the aft chamber is connected to a heated ram air inlet in the leading edge of the vertical stabilizer. A cable, connected to the "Q" cylinder piston, passes between two tangentially mounted pulleys and is connected to the "Q" feel bell crank. The other arm of the "Q" feel crank is connected, by a push-pull rod, to the air feel bell crank. The other arm of the air feel crank is connected, by a push-pull rod, to mixer No. 2 in the rudder control package. A spring in the aft chamber of the "Q" cylinder, behind the piston, maintains tension on the cable when the airplane is on the ground (see Figure 7-13).

In flight, ram air pressure builds up in the aft chamber of the "Q" cylinder and increases the tension of the cable attached to the piston. In the normal mode of rudder operation, movement of the control linkage displaces the "Q" feel bell crank from its rest position in line with the "Q" cylinder cable, the cable is bent around one of the tangentially mounted pulleys, and the piston is pulled back against ram air pressure. The ram air pressure in the "Q" cylinder exerts a restoring force on the "Q" feel bell crank proportional to linkage displacement. This provides an artificial feel force for the pilot. In the manual mode of operation, the artificial feel system is bypassed and rudder feel is transmitted from the flight tab to the rudder pedals.



Rudder "Q" Feel System
Figure 7-13

Rudder Control Package

The rudder control package consists of the power trim jack, centering spring mechanism, mixer No. 1, mixer No. 2, normal-to-manual (power-to-standby) revision mechanism, hydraulic servo valve, and the flight tab control bell crank (see Figure 7-14). The control package is accessible from either side of the vertical stabilizer.

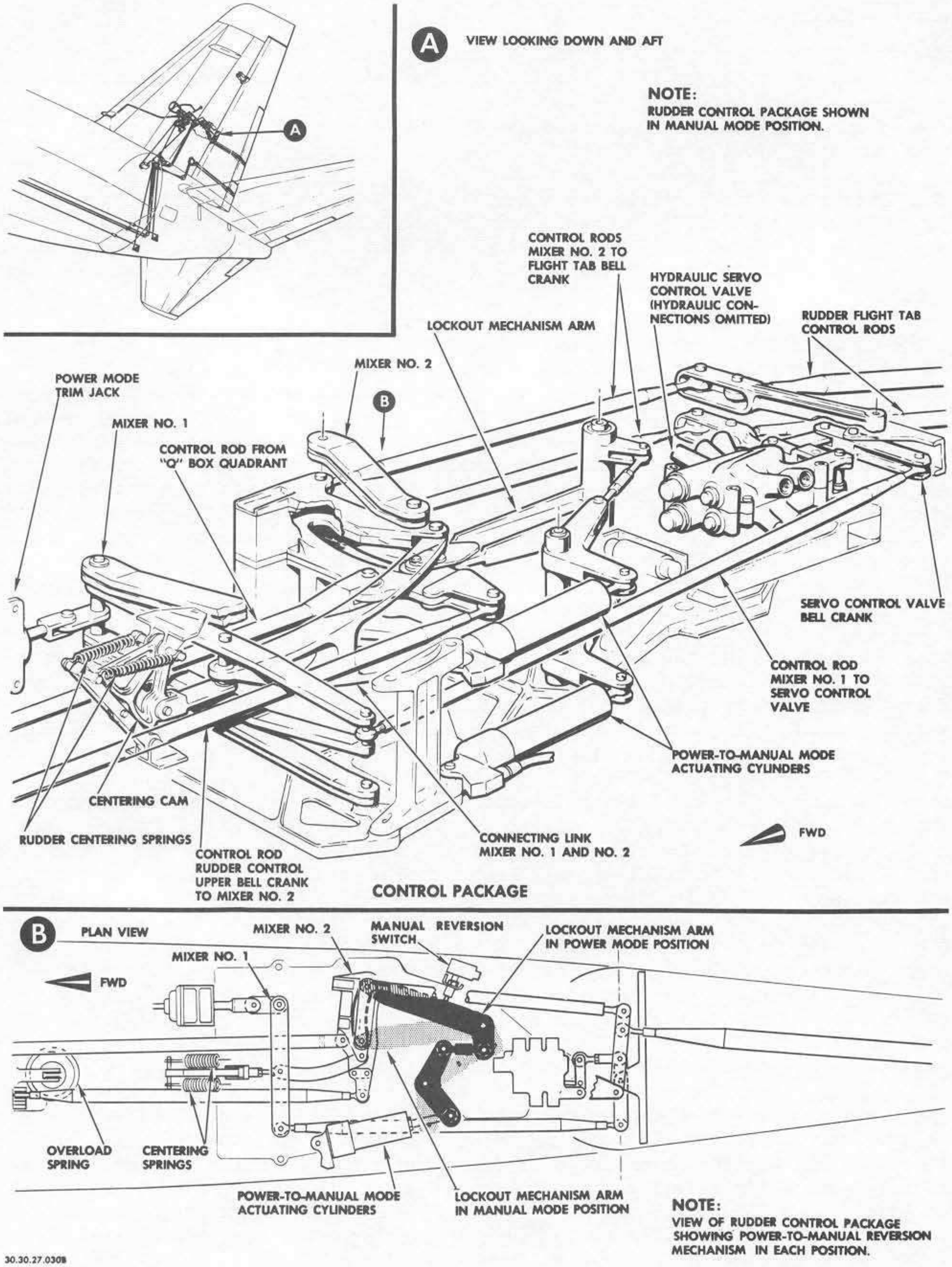
The power trim jack is cable operated through the manual rudder trim system and is connected to mixer No. 1. Actuation of the trim jack transmits motion to the hydraulic servo valve bell crank through the double bell crank in mixer No. 1. The rudder may be trimmed 6-1/2 (±1/4) degrees right or left of the streamlined position.

The centering spring mechanism consists of a roller in the end of a lever which is spring-loaded against a cam with two lobes in tandem. In normal mode of operation, movement of mixer No. 1 bell crank rotates the centering cam away from center and the roller rides up one of the cam lobes. Since spring loading on the roller causes the roller to seek the lowest point between the cam lobes, the cam is forced to rotate in the direction required to center the roller between the cam lobes as soon as input pedal force is removed from the system. When the cam returns to center, mixer No. 1 bell crank returns to center, the servo control valve is opened to port pressure to the actuators to return the rudder to neutral, and as the rudder returns to neutral the servo valve bell crank, pivoted off center on the rudder forward spar, closes the servo valve. The centering spring has no effect during manual mode operation.

Mixer No. 1 performs the power trim and centering functions. It consists of a double bell crank moving between two bell cranks pivoted on the upper and lower plates of the control package. Mixer No. 1 controls the hydraulic servo valve through a push-pull rod connected to the servo valve bell crank and is, in turn, controlled by the output of mixer No. 2.

Mixer No. 2 consists of three bell cranks. The center bell crank is slotted to form a roller track and moves between two bell cranks pivoted on the upper and lower plates of the control package. The feel system connecting rod is attached at the midpoint of the center bell crank. The pilot's input connecting rod is attached at the LH end of the center bell crank. The effective pivot of the center bell crank is determined by the position of the manual reversion (lock-out) arm, the roller of which rides in the slotted track. The center bell crank pivots on its right end during normal mode operation and pivots at mid-point during manual mode operation. In the normal mode of operation, the output of mixer No. 2 is transmitted to mixer No. 1 through a connecting link. In the manual mode of operation, the output of mixer No. 2 is transmitted through push-pull rods to the flight tab bell crank.

The manual reversion mechanism controls the effective pivot point of the center bell crank of mixer No. 2 according to whether or not adequate hydraulic pressure is available for normal mode of operation. The mechanism consists of two spring-loaded single-action hydraulic actuators (lock-out cylinders) and a lock-out arm. A roller in the end of the arm rides in the slot in the center bell crank of mixer No. 2. When available hydraulic pressure is not adequate for rudder control, the control system is held in the manual mode by the return



Rudder Power Control Unit
Figure 7-14

springs in the lock-out cylinders and an over-center lock in the reversion mechanism. A switch, actuated by the lock-out arm, causes a MANUAL MODE WARNING light to illuminate on the flight engineer's panel when the rudder control system is in manual mode. When normal hydraulic pressure is supplied to the rudder control system, the lock-out cylinders are pressurized and extend to move the lock-out arm over to the normal mode position. The system will remain in normal mode position as long as hydraulic pressure in either system remains above 1000 (±100) psi. If pressure in both hydraulic systems drops to below 1000 (±100) psi, the spring-loaded lock-out cylinders will retract and the rudder control system will revert to manual mode of operation. When pressure in either hydraulic system increases to more than 1700 (±200) psi, the lock-out cylinders will extend and the rudder control system will return to the normal mode.

The rudder hydraulic servo control valve is installed in the aft end of the control package, just forward of the rudder hinge line. The valve controls hydraulic pressures up to 3000 psi from each of the airplane's two hydraulic systems. Hydraulic pressure from the servo valve is routed to the four rudder actuating cylinders, the lockout cylinders, and to low pressure warning switches. The valve is actuated by the output from mixer No. 1 through a push-pull rod, the valve control bell crank, and a link connecting the bell crank to the valve. The valve control bell crank is mounted off-center on the rudder forward spar, so that movement of the rudder moves the bell crank pivot point forward or aft to actuate the servo valve toward the closed position.

In the normal mode of operation the servo valve directs hydraulic pressure to the pressure switches, the lock-out cylinders, and the rudder actuaters. If hydraulic pressure falls to less than 1000 (±100) psi in both hydraulic systems, a "fuse" valve in the servo valve assembly opens and dumps pressure from switches, lock-out cylinders, and rudder actuators. The hydraulic pressure lines are closed by the pressure switches and solenoid-operated shutoff valves. The "fuse" valve remains in the open, or bypass, position until pressure in one hydraulic system rises to more than 1700 (±200) psi. When this pressure is reached, the "fuse" valve closes and the system changes over to the normal mode of operation.

The flight tab bell crank is pivoted in the aft end of the control package. The bell crank is connected, by two push-pull rods, to the RH end of the center bell crank of mixer No. 2. In normal operation, the position of the lock-out arm prevents any fore and aft movement of the RH end of the mixer No. 2 bell crank and the flight tab bell crank is effectively locked in neutral position. In manual mode operation, the lock-out arm roller is in the LH end of the slotted track in the bell crank, the bell crank pivots on the roller, and the output of mixer No. 2 is transmitted to the flight tab bell crank.

Rudder Hydraulic Actuating Cylinders

The rudder is actuated, in normal mode of operation, by four identical hydraulic actuating cylinders. Two of the actuating cylinders are installed near rudder hinge number five and two are installed just above rudder hinge number three. The hydraulic servo control valve directs hydraulic system No. 1 pressure to the upper right and lower left actuating cylinders. Hydraulic system

No. 2 pressure operates the upper left and lower right actuating cylinders. When the airplane is on the ground with no pressure in either hydraulic system, the actuating cylinders act as gust dampers and mechanical stops.

Rudder

The rudder includes a flight tab and a trim tab and is attached to the vertical stabilizer by shear type bolts at six points (see Figure 7-15). Balance curtains are installed between the rudder leading edge and the vertical stabilizer. These curtains reduce the rudder hinge moment and thereby assist rudder movement in the manual mode of operation.

Flight Tab

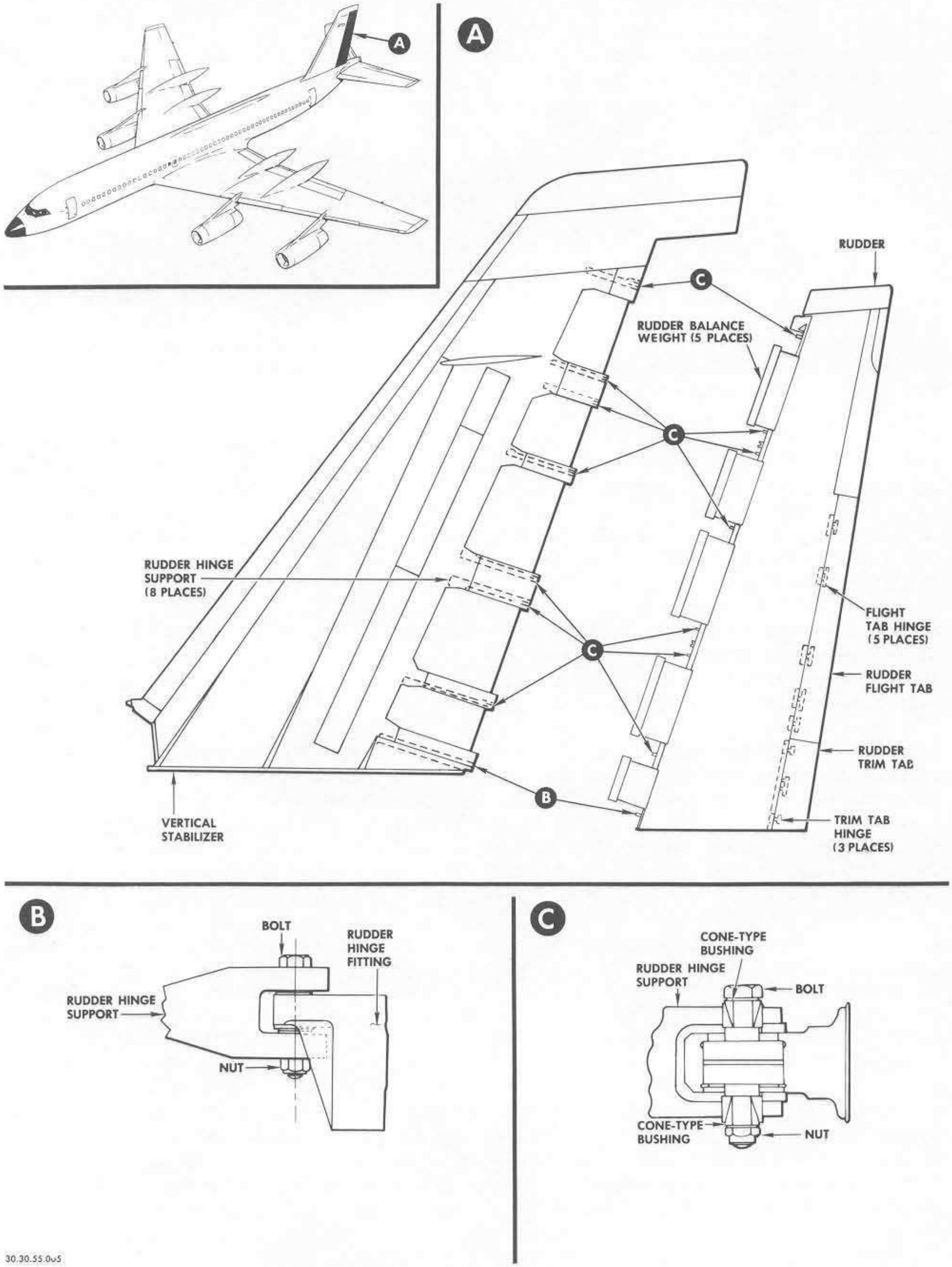
The flight tab extends nearly half the length of the rudder and is of all-aluminum, sandwich-type construction. Four hinges are provided to attach the flight tab to the rudder.

Rudder Hydraulic Control System

In the normal mode of operation, the rudder control system is powered by hydraulic systems No. 1 and No. 2. Hydraulic pressure is routed through a shutoff valve in each system and then to the rudder servo control valve (see Figure 7-12). The servo control valve directs hydraulic pressure from each hydraulic system to one manual reversion mechanism lock-out cylinder, one low pressure warning switch, and two rudder hydraulic actuators. Two high-pressure relief valves and a normally-open solenoid operated bypass shutoff valve are installed in bypass position between the pressure and return lines connecting the servo valve and each pair of actuators. When pressure in the actuating cylinders builds up to more than 1500 (±100) psi, the relief valves open to prevent further build-up of pressure in the cylinders. If operated at normal operating hydraulic pressure of 3000 psi, the four rudder actuating cylinders would be capable of inducing rudder deflections, at high airspeeds, beyond the structural limits of the airplane.

On the ground, sufficient force is developed to fully deflect the rudder and pressure builds up to more than 1500 psi only when the pistons bottom out in the actuating cylinders. In the air, aerodynamic forces tend to drive the deflected rudder to streamline position. These forces, acting against the pistons in the actuating cylinders, cause pressure in the cylinders to build up to more than 1500 psi before the rudder is fully deflected, the relief valves open to limit pressure, and the rudder can not be further deflected. As airspeed increases, aerodynamic forces increase and less rudder deflection is obtained before the relief valves open. This force limiting system restricts the output of each pair of rudder actuators to one-half the output obtainable at full hydraulic system pressure.

If one hydraulic system fails, the remaining system reverts to normal pressure operation and full control effectiveness is retained. The maximum actuating force applied at the rudder in either condition is within the structural limits of the airplane.



Rudder Installation
Figure 7-15

Rudder Warning System

The low pressure warning switch installed in each hydraulic system is normally closed and is actuated to the open position when hydraulic system pressure reaches or exceeds 800 (±75) psi. The switch installed in hydraulic system No. 1 is connected electrically to the bypass shut-off valve in system No. 2; the switch installed in system No. 2 is connected to the valve in system No. 1 (see Figure 7-10). If hydraulic pressure in either hydraulic system falls to less than 800 (±75) psi, the low pressure warning switch in that system reverts to the closed position. When the low pressure warning switch closes, it closes the circuit to illuminate the applicable SYSTEM OFF light on the flight engineer's panel and to energize the bypass shut-off valve in the other hydraulic system. The energized bypass valve is closed and allows no flow through the bypass relief valves in the operating system so that full system pressure is available to power the two rudder actuating cylinders in the operating hydraulic system. Thus, full power is available for rudder operation even though two rudder actuating cylinders are developing no power.

If a pressure switch closes due to loss of pressure in one hydraulic system and the bypass valve does not close in the other system, a circuit is completed through the switch and the valve to illuminate the LOW POWER light on the flight engineer's panel. This light indicates that the rudder control system is operating on half power since the bypass valve in the operating system did not close and the relief valves are still limiting the working pressure in the operating actuating cylinders.

When normal operating pressure in both hydraulic systems is regained, the closed pressure switch opens, breaking the circuit energizing the bypass valve in the other system, and the valve is allowed to open. The bypass relief valves then limit the pressure available at all rudder actuating cylinders.

If a bypass valve does not open when its controlling pressure switch opens, a circuit is completed through a switch in the bypass valve and through the pressure switch which illuminates the EXCESS POWER light on the flight engineer's panel. This light indicates that two rudder actuating cylinders are operating at full power (3000 psi) and two actuators are operating at half power (1500 psi). Operation of the rudder at high airspeeds under this condition could result in damage to airplane structure.

If pressure drops to less than 1000 (±100) psi in both hydraulic systems, the lock-out cylinders, in the control package, retract and the rudder system is in manual mode of operation. When the control package reverts to manual mode, the lock-out arm actuates the manual mode warning switch on the control package. This completes a circuit which illuminates the MANUAL RUDDER MODE WARNING light on the flight engineer's panel.

If hydraulic pressure drops below 1000 psi in both hydraulic systems, spring-loaded lock-out cylinders in the manual reversion mechanism are de-actuated and rudder control is automatically switched to the manual mode. In the manual mode, the lock-out arm actuates a switch which illuminates the MANUAL MODE WARNING light on the flight engineer's panel. Rudder pedal movement is then transmitted mechanically to the flight tab. Aerodynamic forces acting on the flight tab move the rudder in the opposite direction to that in which the flight tab has been deflected. As the rudder moves away from neutral, anti-servo action of the flight tab control linkage decreases the deflection of the flight tab until the deflecting force of the flight tab is balanced by the aerodynamic forces tending to streamline the rudder.

Autopilot

The autopilot rudder servo cables are attached to the rudder control quadrant in the vertical stabilizer.

Rudder Trim System

Directional trim is adjusted by a rudder trim knob on the pilots' pedestal (see Figure 1). The trim knob is conneced by gears, cables, and push-pull rods to the rudder trim tab and the hydraulic servo control valve in the rudder control power package. A gear on the trim knob shaft drives the rudder trim indicator while a bevel gear at the bottom of the shaft drives the forward trim cable drum. Two trim cables extend aft from the forward cable drum beneath the cabin floor, through pressure seals, around pulleys, and up into the vertical stabilizer where each cable joins a pair of cables to form two parallel cable circuits. One pair of cables extend upward to the manual mode trim jack which is connected by push-pull rods to the rudder trim tab. The other pair of cables extend upward to the power mode trim jack which is connected by levers and a push-pull rod to the servo control valve in the power package.

In the power mode of operation, rotation of the trim knob to the right causes a limited right rudder deflection by introducing a bias into the rudder controls through a whiffletree linkage in the rudder control package. This causes a shift from neutral in the rudder servo valve, resulting in a corresponding rudder displacement for right rudder trim. The manual trim jack operates simultaneously with the power trim jack and the rudder trim tab is deflected to the left (RH rotation of trim knob). Aerodynamic forces acting on the trim tab assist in deflecting the rudder to the right.

In the manual mode of operation, the operation of the power trim jack is ineffective and rudder trim is effected solely by the action of the rudder trim tab.

Rudder Trim and Trim Tab Travel

During power mode rudder operation, full travel of the rudder trim control knob on the pilots' pedestal deflects the rudder 6-1/2 degrees left or right from the streamline position and also moves the rudder trim tab 23 degrees from streamline in the opposite direction. During manual operation, with no hydraulic pressure, full travel of the rudder trim control knob will deflect the rudder trim tab 23 degrees left or right from streamline.

FLAP CONTROL SYSTEM

Two double-slotted, Fowler-type flaps are installed in the trailing edge of each wing, one inboard and one outboard of the aileron. The flaps are so designed as to provide low-drag/high-lift when partially extended, and high-lift/high-drag when fully extended. Consult the Airplane Flight Manual for maximum airspeeds at different flap settings.

The flaps are powered by two hydraulic flap motors, each driven by a separate hydraulic system. Both flap motors are mounted on a central drive gearbox and are connected by a gear train with the torque tubes which extend outboard into the wings aft of the rear spars. Screw jack actuators, each containing a set of bevel gears and an irreversible clutch, connect the torque tubes to the individual jack screws. Flow of hydraulic fluid to the flap motors is regulated by a dual three-way selector valve (see Figure 7-16).

A FLAP POWER SELECT switch at the front of the pilots' overhead switch panel can be positioned to operate the flaps on either the No. 1 or No. 2 hydraulic system. When the switch is in NORM position, the flap phase valve in the No. 2 hydraulic system is energized continuously to the closed position and the flaps operate on the No. 1 hydraulic system only. If the No. 1 system should malfunction, the switch is placed in EMER position, and the No. 2 hydraulic power system will operate the flaps.

Flap Test Switch

Actuating a FLAP ASYMMETRY TEST switch on the pilots' pedestal to the TEST position illuminates a FLAP ASYM light on the pilots' instrument panel if the asymmetry switches are functioning properly. Placing the switch to the REL TEST position illuminates the FLAP ASYM light if the flap asymmetry control relay solenoid circuit is operating satisfactorily.

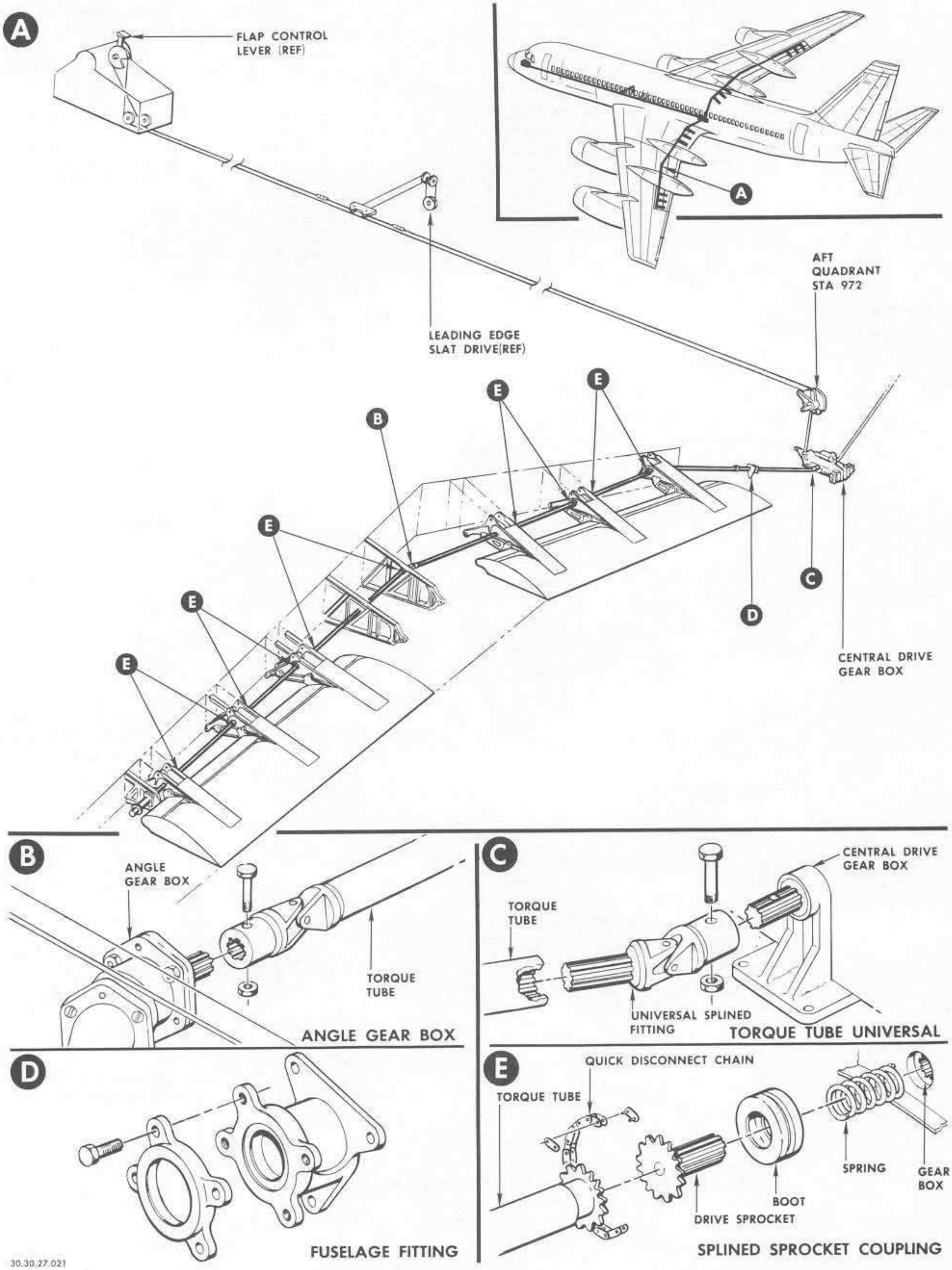
Flap Warning Horn and Lights

A warning horn will sound if takeoff is attempted and the flaps are not in one of the proper takeoff position detents. The horn will also sound if the flaps are lowered past approximately 44 degrees and the landing gears are not down and locked.

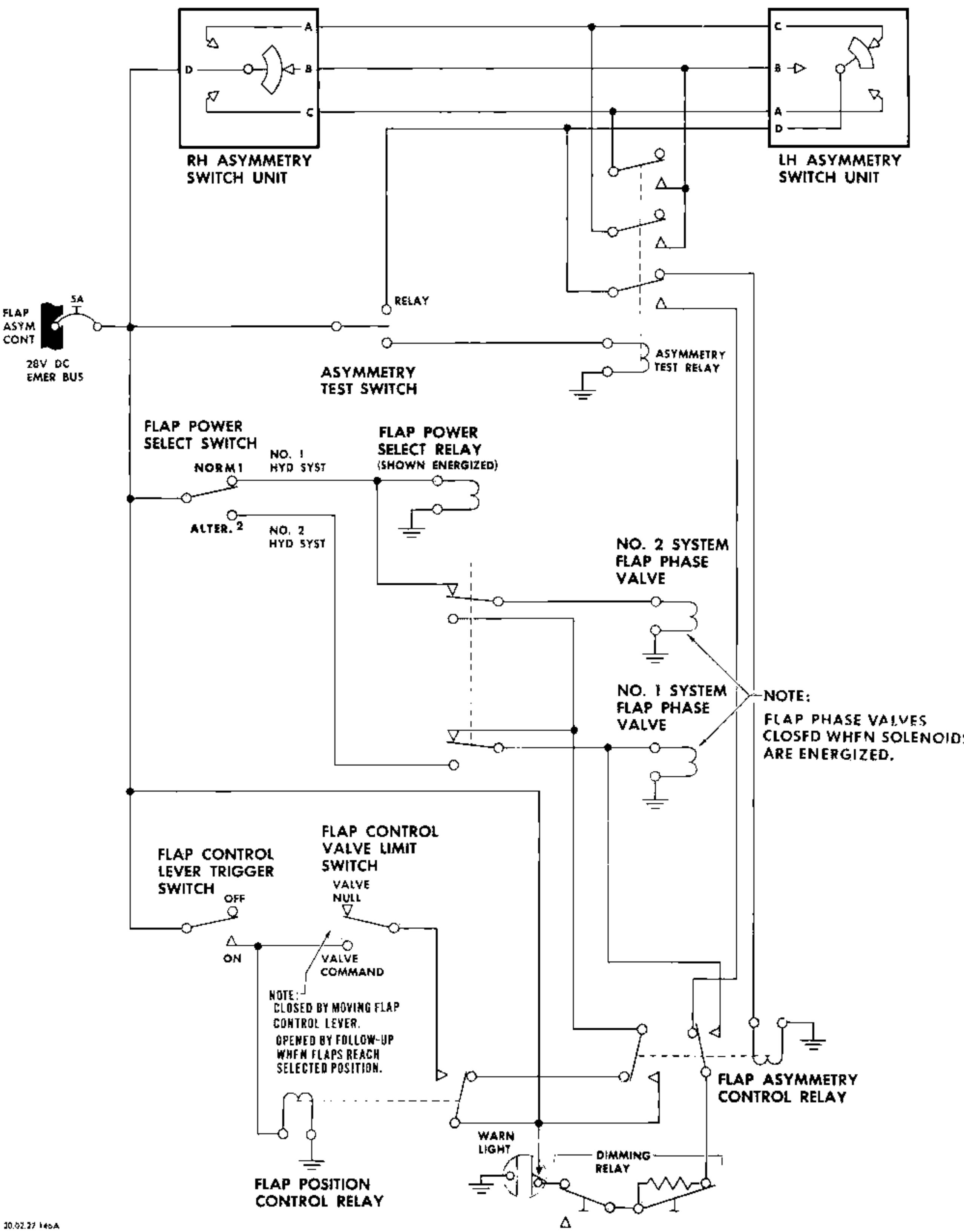
Four flap position vs airspeed switches are provided in the flap circuitry. These switches combine to illuminate a light on the pilots' instrument panel when the flap setting exceeds the extension permitting for the existing airspeed.

Flap Asymmetrical Shutoff Switch

An asymmetric unit is installed at the outboard support of the outboard screwjack gear box in each wing. An asymmetrical shutoff switch is a part of each of these units. The units are electrically interconnected, and connected to, the flap phase valves through the flap asymmetrical control relay (see Figure 7-17). Each asymmetrical switch has three cams which are geared to the flap high-speed torque tube drive. Each cam has three lobes, the high point of



Wing Flap System
Figure 7-16



30.02.27 146A

Flap Asymmetry Shutoff Switch
and Phase Valve Circuit
Figure 7-17

each lobe is 124 degrees apart. Each cam moves a roller actuator to make contact with a stationary contact point. Under normal conditions, the flaps are in symmetry and there is no electric power at the operating flap phase valve. Any condition that causes the flaps to be out of symmetry by 2 degrees will cause one cam of each asymmetrical switch to move its roller actuator. When contacts are made simultaneously at the proper points, a circuit is completed from the 28-volt dc emergency bus through the two asymmetrical switches and the control relay to the solenoid of the operating flap phase valve. The solenoid then closes the valve, shutting off hydraulic pressure to the flap motor on the main drive gearbox. Flap movement is thus stopped immediately.

Flap Position Indicators

A dual-reading flap position indicator is installed on the center instrument panel in the flight compartment. Two position transmitters are provided, one on each outboard end of the left and right outboard flap. Two indicating needles are used on the indicator: a broad red needle for the right-hand flap and a narrow white needle for the left-hand flap. The red needle is directly behind the while needle. When the flaps are moving in unison, a single white pointer with a red outline is shown. However, if a split-flap (asymmetrical) condition develops, the two needles separate and two white pointers are shown. The dial is calibrated in ten degree increments.

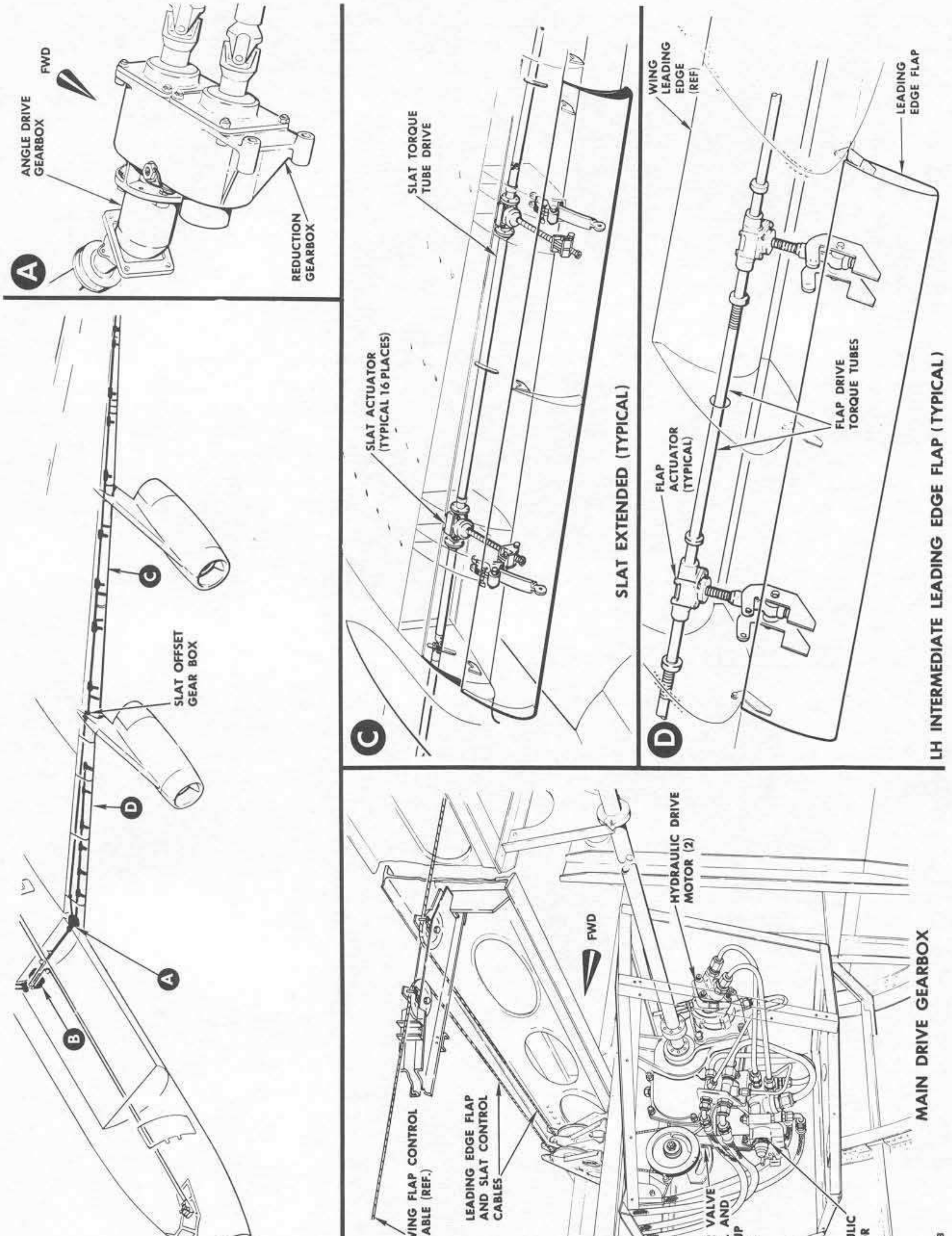
A flap position scale marked off in ten degree increments is located adjacent to the flap control lever on the pilots' pedestal.

WING LEADING EDGE SLATS AND KREUGER FLAPS

Four leading edge slats are mounted on each wing of the airplane, two located between the two engine pylons and two outboard of the outboard engine pylon. Each slat is supported by two curved tracks attached to the slat structure. The tracks slide fore and aft on roller-equipped supports mounted on the wing leading edge structure (see Figure 7-18).

Three Kreuger type flaps are mounted on the lower side of each wing leading edge between the fuselage and the inboard engine. The flaps are hinged at their leading edges and are designed to hinge down and forward. The inboard flap (No. 1) is supported by three double hinges; the intermediate flap (No. 2) by two double and three single hinges; the outboard flap (No. 3) by one double and two single hinges.

The slats and Kreugers are extended and retracted by screw jacks actuated by a torque tube in each wing connected to a common main drive gearbox driven by two hydraulic motors. Each motor is powered by an independent hydraulic system. Each slat and Kreugers is actuated by two screw jacks through recirculating ball bearing ball nuts which are attached to the slat and Kreugers structure and extend or retract the units when the jack screws are rotated. Each screw jack gearbox contains an overload clutch to protect the structure and a "no back" device to prevent air loads from extending or retracting the slats and Kreugers.



Wing Leading Edge Flap and
Slat Control System
Figure 7-18

Flow of hydraulic fluid to the hydraulic slat motors is regulated by a dual three-way valve connected mechanically to the flap control lever on the pilots' pedestal through a followup linkage and a loop of the same cable system that actuates the flaps. The valve is closed by the followup linkage when the slats reach the fully extended or retracted position. Eighty turns of the torque tubes correspond to full slat travel of 15 degrees.

The position of the slats and Kreugers is indicated by two amber SLATS IN-TRANSIT and two green SLATS EXTEND lights on the center instrument panel. Each light indicates the position of the slats and Kreugers in one of the wings.

The slats and Kreugers cannot be set in an intermediate position; when actuated, they move to either the fully extended or fully retracted position. When the flap control lever is moved aft from the 0-degree detent, the slats and Kreugers begin to extend when the flap lever passes the position corresponding to a flap extension of approximately 8 degrees. They begin to retract when the flap control lever is advanced through the 20-degree flap position.

In the retracted position, the slats and Kreugers complete the contour of the wing leading edge; when extended, they deflect the airflow over the upper surface of the wing permitting the airplane to fly at a higher angle of attack and lower airspeed without stalling. The leading edge of each slat is protected against icing by heated air from a telescoping tube connected to the main hot air duct in the leading edge of the wing.

Slat and Kreuger Position Lights

The slats and Kreugers have only two positions - fully extended or fully retracted. Four lights on the center instrument panel, with an amber and a green light for the slats and Kreugers on each wing, indicate position. An amber SLATS INTRANSIT light illuminates when any of the slats or Kreugers on the corresponding wing move away from the retracted position. A green SLATS EXTEND light illuminates when all the slats and Kreugers on the corresponding wing are fully extended (see Figure 7-19).

CONTROL SURFACE GUST DAMPERS

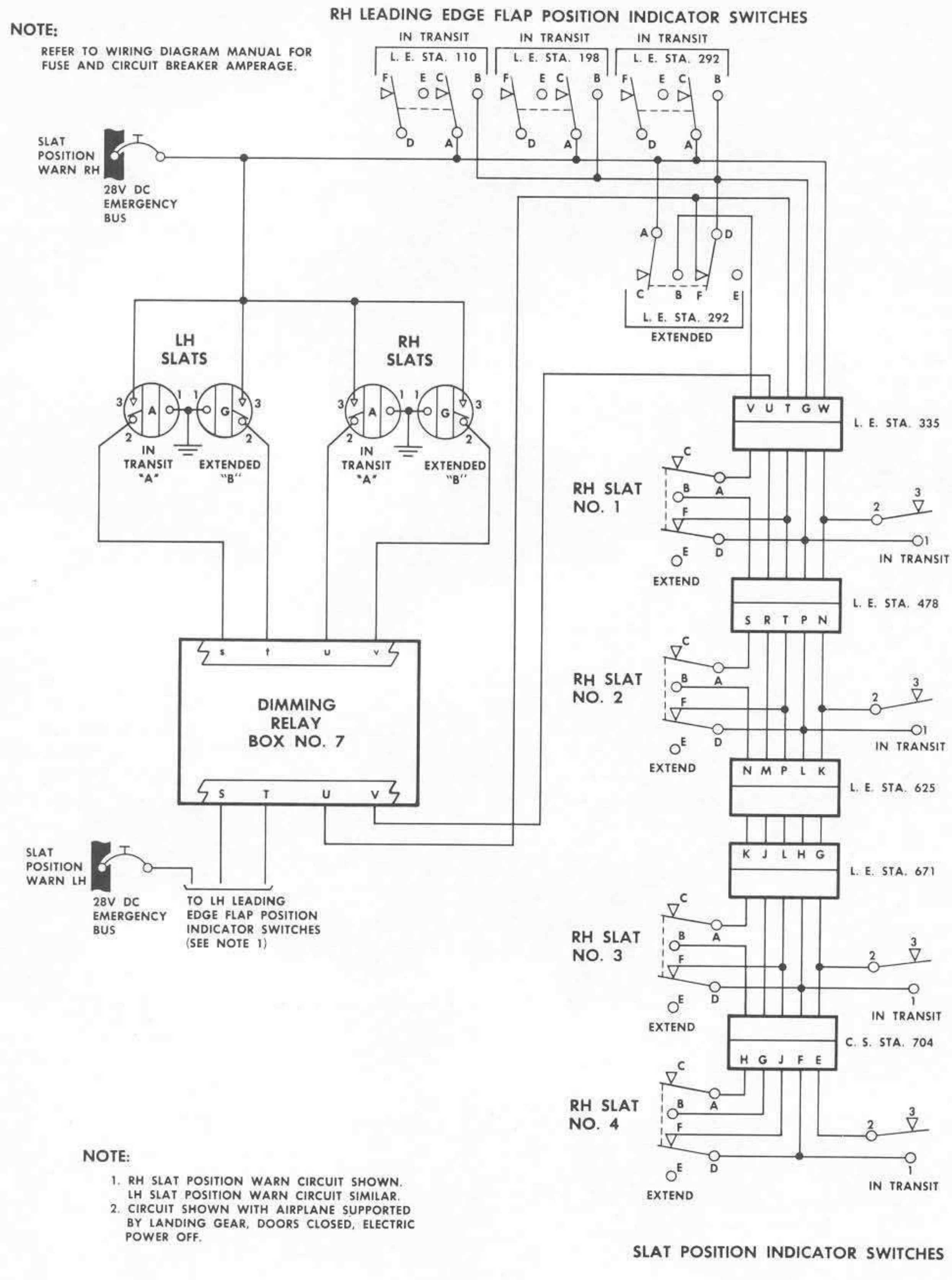
Gust dampers provide control surface gust protection for the ailerons and elevators when the airplane is on the ground. The dampers are self contained units and are not connected to the airplane hydraulic system (see Figure 7-20). The dampers are permanently installed and are designed so that damping rates do not effect normal flight movements of the control surfaces.

Gust Damper Locations

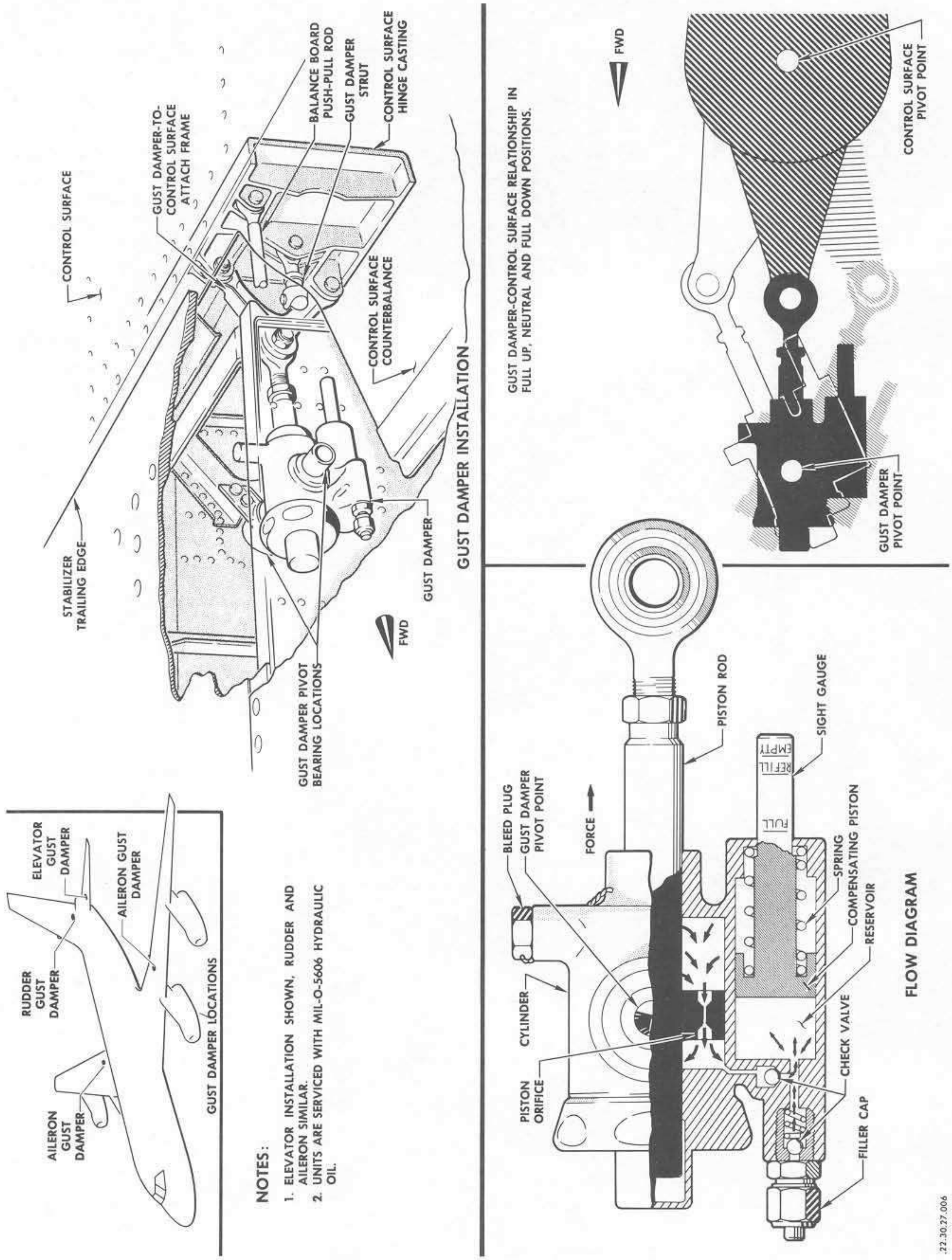
There are four gust dampers installed on the airplane as follows:

1. Two elevator gust dampers are located at the inboard trailing edge of the horizontal stabilizer at BL 39.37.

2. Two aileron gust dampers are located in the wing trailing edge just inboard of the aileron hinge at station 259.3.



Leading Flap and Slat Position
Lights Schematic
Figure 7-19

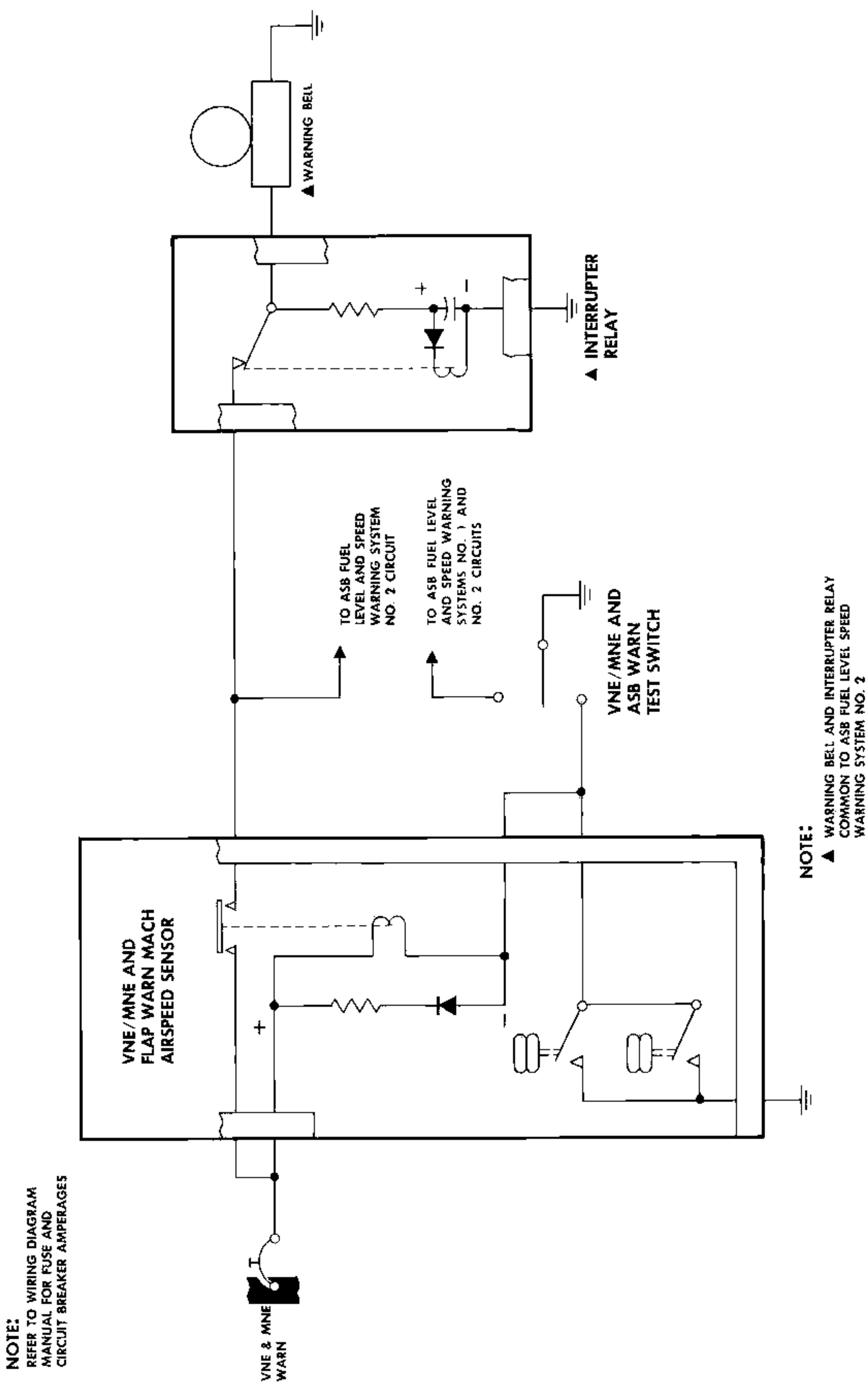


Control Surface Gust Damper
Figure 7-20

MACH/AIRSPEED NEVER EXCEED WARNING SYSTEM

When the airplane is approaching the maximum allowable Mach or maximum allowable indicated airspeed, a warning bell (to the left and forward of the pilot's seat) rings intermittently. This system is commonly referred to as V_{NE}-M_{NE} for velocity never exceed and Mach never exceed. The system consists of an aneroid Mach/airspeed switch, an interrupter relay, a test switch, and the warning bell (see Figure 7-21). When near maximum Mach or near maximum indicated airspeed is reached, a pressure switch in the lower right-hand electrical compartment is actuated to complete a circuit from the 28-volt dc emergency bus to the relay coil in the pressure switch. The energized relay closes double contacts to route the current through the interrupter relay and on to the bell. The system is interconnected with the ASB fuel level and speed warning systems.

The system can be tested by momentarily depressing the V_{NE}-M_{NE} WARN TEST pushbutton located on the pilot's instrument panel. The bell will ring if the circuit is operating properly.



VNE/MNE Warning System
Figure 7-21

8 AUTOPILOT SYSTEM

Section 8

SPERRY AUTOPILOT

TABLE OF CONTENTS

TITLE	PAGE
AUTOPILOT SYSTEM (SPERRY SP-30)	8-1
AUTOPILOT CONTROL SYSTEM	8-1
Altitude Hold	8-5
Pitch Knob	8-5
Turn Knob	8-5
Servos Engage Lever	8-5
Turn Selector Switch	8-7
Manual Take-Over Procedure	8-7
YAW DAMPER SYSTEM OPERATION	8-7
THREE AXIS TRIM INDICATOR	8-7
AIRSPEED TORQUE LIMITING SYSTEM	8-8
AUTOPILOT ELECTRICAL POWER SOURCES	8-10

SPERRY SP-30 AUTOPILOT

AUTOPILOT SYSTEM

The Sperry SP-30 autopilot system controls the airplane in flight in two modes of operation: autopilot and yaw damper. The autopilot automatically controls the airplane in all three axes when the servos engage lever of the control panel is engaged in AUTOPILOT position. The autopilot controls only the rudder when the servos engage lever is engaged in the YAW DAMPER position.

In autopilot mode the SP-30 maintains the airplane in level flight on magnetic headings; turns the airplane to preset magnetic headings, to preselected omnirange radials, or to localizer courses; maintains the airplane at constant pressure altitudes; and makes automatic ILS approaches. Turn and pitch signals can be inserted by the pilots to control the airplane automatically to bank angles up to 35 degrees and to pitch attitudes up to ±15 degrees. The autopilot system can be engaged in any attitude within these limits.

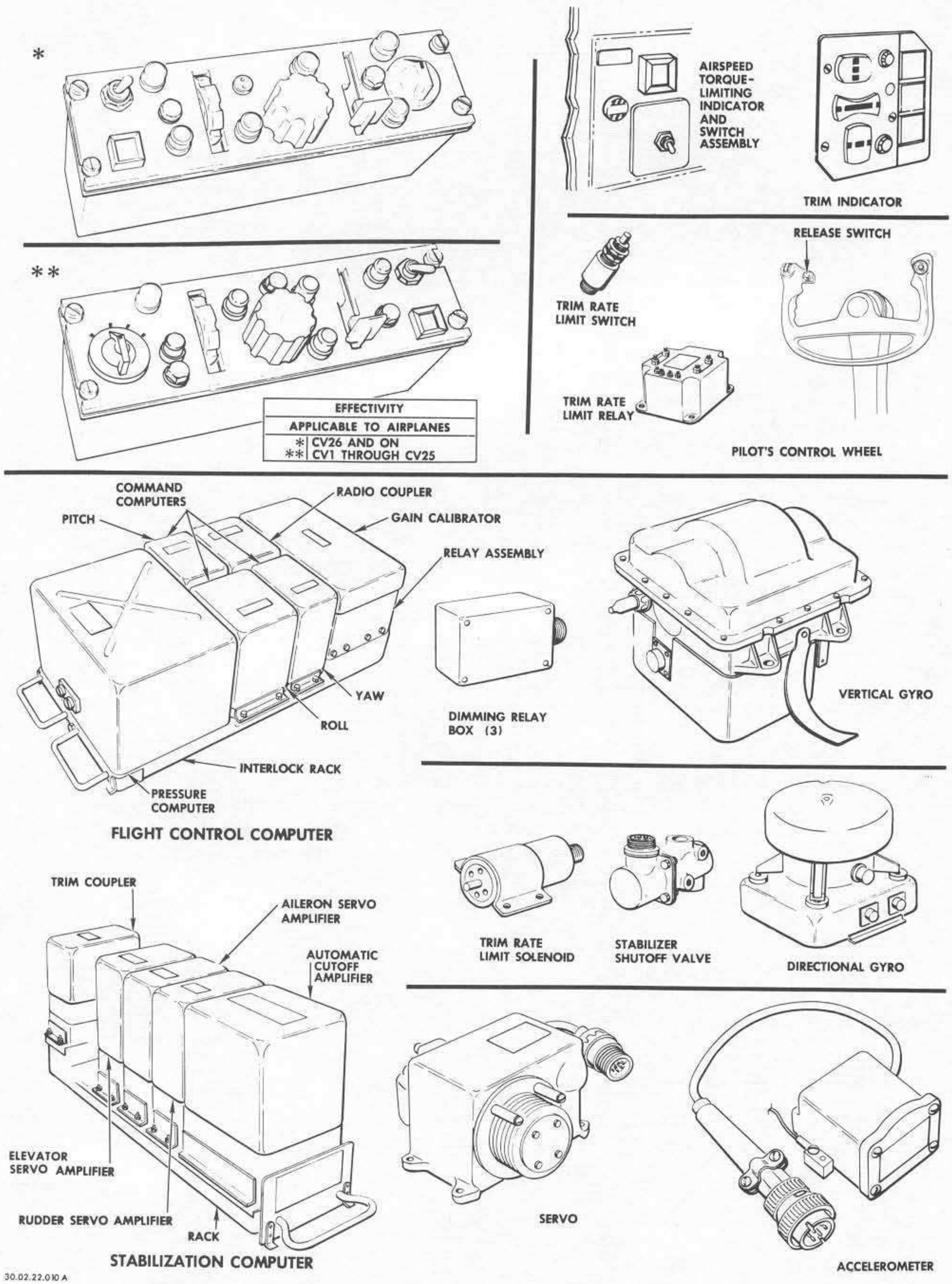
Automatic pitch trim compensates for loading changes caused by fuel consumption, passenger movements, and air gusts. Six accelerometers, two in each of the three axes, sense rate of change of direction; the directional gyro of the remote compass system provides stabilized reference heading signals; and the vertical gyro provides reference attitude signals. Signals from these sources, turn and pitch signals inserted by the pilots at the control panel, and signals from the radio navigation systems are processed in the flight control computer and the stabilization computer. Amplifiers in the stabilization computer raise the signals to power levels adequate to drive the rudder, aileron, and elevator servos. The servos control flight by positioning the flight tabs on the control surfaces.

Switches and controls on the control panel engage the autopilot system, insert turn and pitch signals, and select the modes of operation. A light on the control panel and a light on the trim indicator warn of autopilot disengagement or faulty operation. Figure 8-1 illustrates all the autopilot components and Figure 8-2 shows the location of autopilot components. An airspeed torque limiting system reduces elevator gain in the higher speed ranges since less elevator control surface movement is needed at the higher airspeeds.

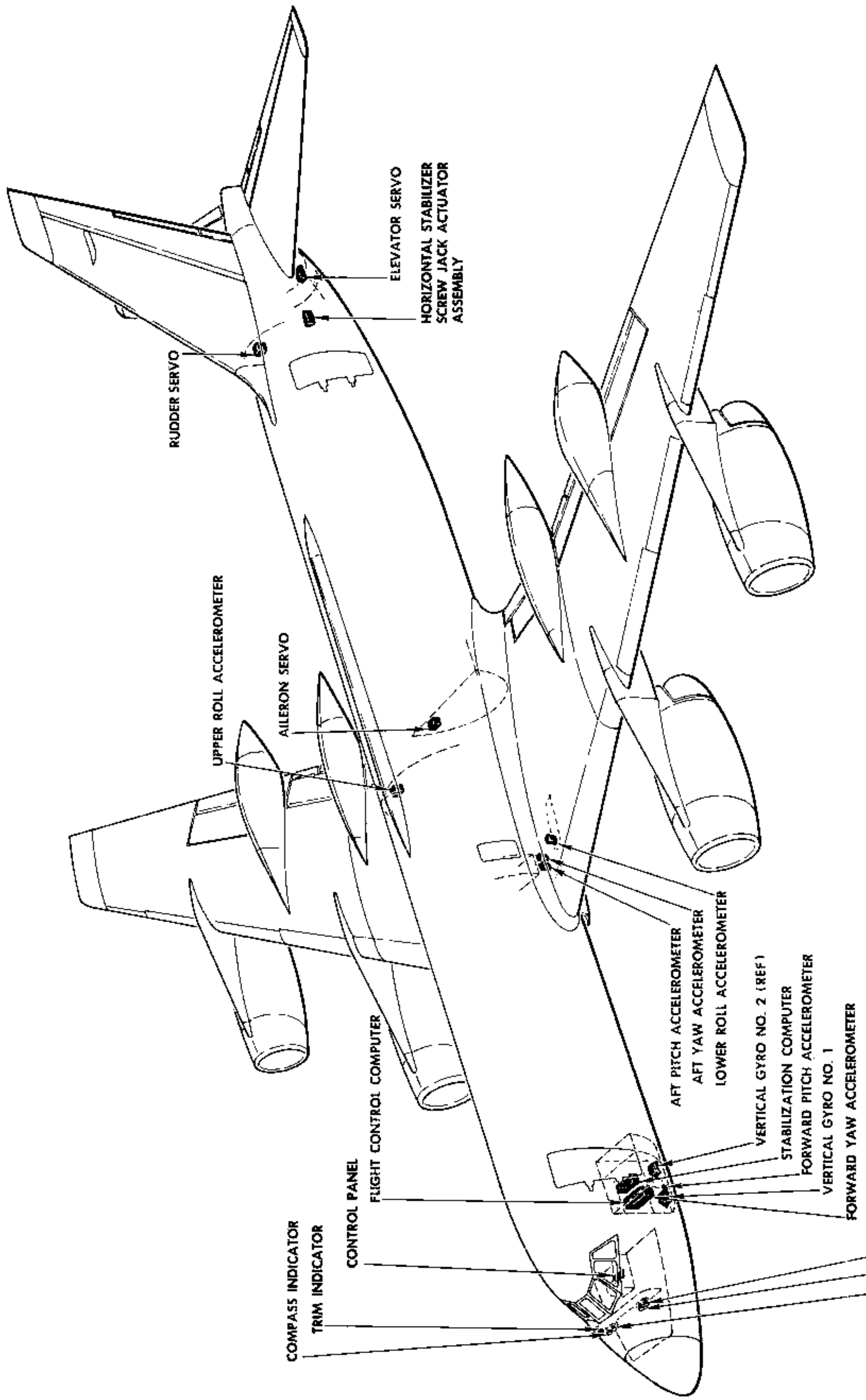
Interlock protection of all essential functions required for safe and reliable system operation is provided. All interlocks must be satisfied prior to engagement or the system will not engage. The opening of an interlock while engaged will disengage the autopilot system.

AUTOPILOT CONTROL SYSTEM

Control of the autopilot system is accomplished by the pilot or copilot from the AUTOMATIC PILOT control panel mounted on the pilots' pedestal (see Figure 8-3). Prior to in-flight engagement, the airplane should be in trim position. When making system checks on the ground, the controls must be placed in a neutral position. The display bars on the three-axis trim indicator should be in the center alignment positions before the autopilot is engaged. The controls for mode switching, turn and pitch maneuvering and engage switching are mounted



Autopilot Components
Figure 8-1



Autopilot Component Locations
Figure 8-2

TO BE FURNISHED

Autopilot Control Panel - Typical
Figure 8-3

on the face of the panel. The three-axis trim indicator, with its three warning lights (AUTO PILOT OFF, GLIDE PATH ARMED and AUTO TRIM OFF) is located on the pilot's instrument panel. The airspeed torque warning light is located adjacent to the three axis indicator (see Figure 8-4).

Altitude Hold Switch

This switch is held to the ALT HOLD (altitude hold) position by a solenoid while the autopilot system is engaged. When this switch is actuated to ALT HOLD position, the airplane holds, or returns to and maintains, the pressure altitude it had when the switch was engaged. When the autopilot disengages or when the glide slope path beam is captured, the altitude hold switch automatically returns to the off position.

Pitch Knob

The PITCH knob controls the rate of change of pitch, either UP or DOWN. Actuating the pitch knob from detent position changes the position of the wiper of a pitch signal potentiometer and closes a pitch knob detent switch through whose contacts the new pitch signal is introduced into the autopilot system. The pitch knob must be held until the desired new pitch attitude is attained. When the pitch knob is released, spring loading returns the pitch knob to detent position and the potentiometer wiper to neutral position; and the detent switch opens to cut off pitch signal input. When the autopilot is initially engaged, therefore, or if the autopilot is disengaged and then reengaged, the autopilot maintains the airplane in the pitch attitude it has upon engagement. In flight, the pitch knob can control pitch attitude to ±15 degrees.

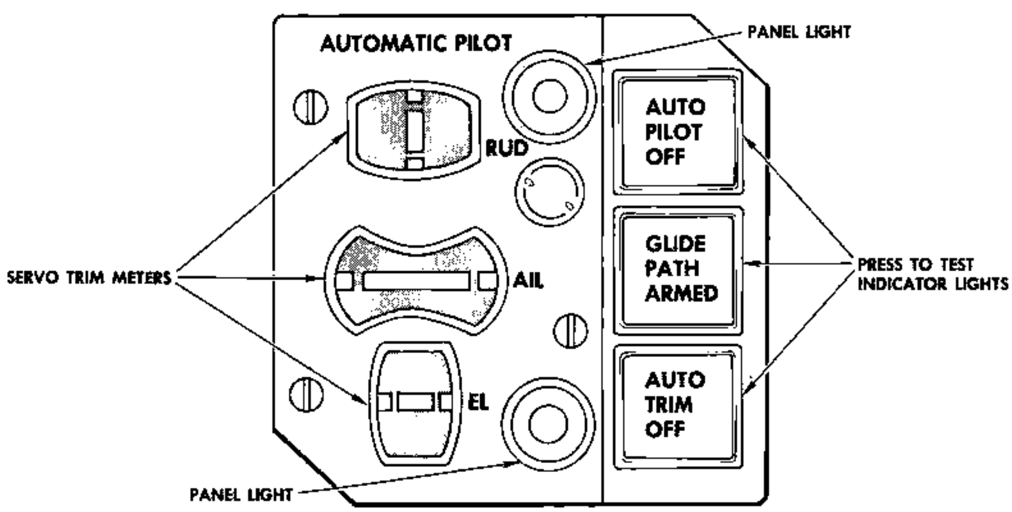
Electrical interlocks give the pitch knob priority over altitude hold and glide path mode engagement. If the pitch knob is actuated, these selections are cancelled; the ALT HOLD switch returns to OFF position and the TURN SELECTOR switch returns to the TURN KNOB position.

Turn Knob

The TURN knob is directly connected to the wiper of the bank signal potentiometer, but this control is not spring loaded to detent position and must be returned to detent position manually. The turn knob inserts a bank signal through a detent switch that closes when the turn knob is rotated from detent position. Large heading changes are made with the turn knob. Small heading changes are usually made by changes in the heading preselected on the radio magnetic direction indicator. After the airplane is turned to a new heading by the turn knob, the knob must be returned manually to detent position.

Servos Engage Lever

This control, called the SERVOS engage lever, engages the autopilot in yaw damper or autopilot mode of operation. The engage lever is solenoid held to both engage positions. The autopilot can be disengaged by moving the servos engage lever to OFF position from either AUTOPILOT or YAW DAMPER position. The lever automatically returns to YAW DAMPER or OFF position if the autopilot system automatically disengages, and to OFF position if a control wheel autopilot release switch is pressed.



AUTOPILOT INDICATOR

.30.30.22.005 B

Autopilot Trim Indicator
Figure 8-4

AUTOPILOT mode gives full autopilot control of the airplane in all three axes. YAW DAMPER mode augments stability in the yaw axis. Either pilot can overpower the autopilot in either mode of operation.

Turn Selector Switch

This four-position rotary switch selects the mode of operation after autopilot engagement. The switch is spring-loaded to TURN KNOB position and automatically returns to this position if the autopilot disengages automatically or is disengaged intentionally. TURN KNOB mode is linked to the magnetic heading of the airplane: directional gyro input to the autopilot maintains the airplane on a constant heading. HDG SEL mode causes the airplane to turn to a heading preset on the radio magnetic direction indicator. LOC-VOR mode selection causes the airplane to turn to a desired VOR radial, preselected on the pictorial deviation indicator, or to a localizer beam, when the VHF navigation system receiver is tuned to an omnirange station or to an ILS frequency channel. GLIDE PATH mode enables the airplane to capture and track a glide slope path beam, when the glide slope system receiver is tuned to an ILS frequency channel. The GLIDE SLOPE ARMED warning light on the autopilot trim indicator lights when the TURN SELECTOR knob is actuated to GLIDE PATH position. The warning light extinguishes when the airplane intercepts the glide slope beam and begins automatic descent along the glide slope path.

Manual Take-Over Procedure

At any time that manual control of the airplane is desired, the autopilot disengage switch on either control wheel may be pressed. This switch returns the airplane to manual pilot control, in a trim condition, by disengaging the autopilot system.

CAUTION: DO NOT DISENGAGE THE AUTOPILOT UNLESS THE AIRPLANE IS IN TRIM. FAILURE TO HAVE THE AIRPLANE IN A TRIM CONDITION CAN RESULT IN UNDUE MOVEMENT OF THE CONTROL SURFACES.

YAW DAMPER SYSTEM OPERATION

The yaw damper system augments airplane stability in yaw during manual control. Placing the autopilot servos lever in yaw damper position engages the yaw damper system. Only the rudder is placed under automatic control; elevators and aileron controls are not affected. Pilots can override the automatic rudder function by applying sufficient force on the applicable rudder pedal. The yaw damper system is disengaged by pressing the autopilot disengage switch on either control wheel. This will automatically return the servos lever to the OFF position. Manually moving the lever to the OFF position will also, of course, disengage the yaw damper.

THREE AXIS TRIM INDICATOR

The autopilot three-axis trim indicator is located on the pilot's instrument panel (see Figure 8-4). The rudder, aileron, and elevator trim indicators are actuated by meter movements connected in parallel with the field windings of the servo motors. When the servo amplifiers supply voltage to drive the servo motors, the trim meters deflect in the direction of the trim needed, and by amounts proportional to servo amplifier output. The trim meters show servo trim condition whether the autopilot is engaged or not, if the autopilot circuit breakers are closed. Trim meter signals are attenuated as a function of airspeed. The trim meters do not show airplane attitude or control surface position. Out-of-trim indications mean that servo amplifiers are supplying power, and that autopilot engagement may cause bumps or transients.

The red AUTOPILOT OFF light illuminates when the autopilot disengages because of malfunction or electrical power loss. The AUTOPILOT RESET light on the control panel is paralleled with the AUTOPILOT OFF light, and both light together. They can both be extinguished by pressing the reset light on the control panel. The amber GLIDE PATH ARMED light illuminates when the GLIDE PATH mode of operation is selected on the control panel. This light extinguishes when the airplane intercepts and captures the glide slope path beam of an ILS station. The amber AUTO TRIM OFF light illuminates when the elevators remain out of trim over a period longer than five seconds. This light warns the pilot of possible malfunction that may require manual control of the airplane.

The GLIDE PATH ARMED and the AUTO TRIM OFF warning lights are energized through a dimming relay box of the master warning light system. The AUTOPILOT OFF warning light and the AUTOPILOT RESET light are energized through another dimming relay box. Thus the autopilot system warning lights are dimmed at the same time as other warning lights, to adjust warning light brilliance to the brightness of the flight compartment lighting.

<u>AIRSPEED TORQUE LIMITING SYSTEM</u>

The airspeed torque-limiting switch and indicator assembly comprises two pressure switches, two relays, a safe airspeed mode indicator, a test switch, and a warning light (see Figure 8-5). This system has the following functions:

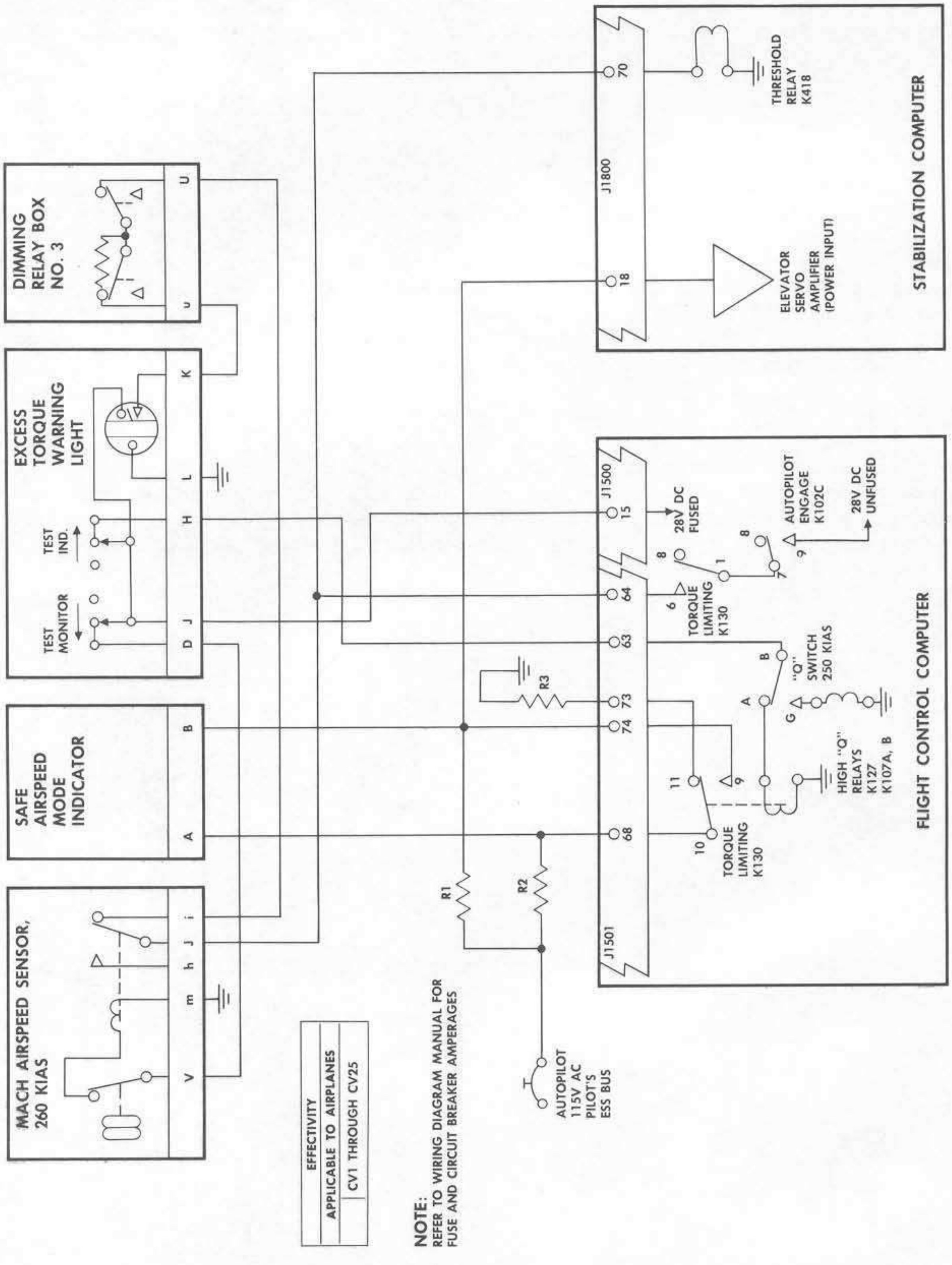
1. It limits autopilot system control of the airplane in the pitch axis so that the autopilot cannot insert pitch rate-of-change signals that make the airplane exceed "G" limits.

2. It lights the autopilot excess torque warning light if the torque limiting system malfunctions. The light warns that high gain of the autopilot is applying high torque to the elevators in the high airspeed range, where only low torque should be applied. The light thus essentially warns of the danger of overcontrol by the autopilot.

3. The safe airspeed mode indicator shows whether high or low torque attenuation is applied to the elevator servo amplifier.

4. The test switch makes a ground test to ensure that the assembly is functioning correctly.

Two pressure switches sense a differential between pitot and static pressures and control the 250 KIAS and the 260 KIAS relays. The 250 KIAS relay is located in the electronic compartment. The 260 KIAS relay assembly, part of the stabilizer trim rate system, is located in the electrical compartment. The pressure switches close their switch contacts below 250 and 260 knots indicated



Airspeed Torque-Limiting System
Schematic - Typical
Figure 8-5

airspeed. On the ground, therefore, with electrical power on the airplane and the autopilot system and stabilizer trim system circuit breakers closed, both relays energize. As a result, the airspeed torque limiting system is switched to the low-airspeed high-torque range. Since the 260 KIAS relay opens the 28-volt dc circuit to the autopilot excess torque warning light in the low-airspeed range, the warning light cannot light.

The 250 KIAS switch opens at 250 ±5 knots IAS. As long as the switch is closed, the autopilot gain attenuation is set for the low-airspeed high-torque range. Above 250 KIAS the pressure switch opens and switches autopilot gain in the elevator channel to the low-torque range. Decreased elevator surface movement is needed at high airspeeds; and decreased autopilot gain decreases elevator movement.

Electrical power from the 28-volt dc power supply of the autopilot system energizes the 250 KIAS relay through a set of its own contacts. If this electrical power is lost, the relay deenergizes and automatically switches autopilot gain to the low-torque range, a fail-safe feature.

AUTOPILOT ELECTRICAL POWER SOURCES

The autopilot system converts airplane electrical power to various ac and dc voltages for autopilot use. Primary power is from the ac essential bus system. Warning lights are supplied from the 28-volt dc system. For detailed circuit information and circuit protection means, consult the WIRING DIAGRAM MANUAL and Chapter 22 of the MAINTENANCE MANUAL.

9 POWER PLANT

Section 9

POWER PLANT

TABLE OF CONTENTS

TITLE	PAGE
ENGINES	9-1
ENGINE DATA	9-1
ENGINE OPERATING CONTROLS	9-1
Power Control Levers	9-1
Fuel Shutoff Levers	9-7
Thrust Reverse Levers	9-7
ENGINE AIR SYSTEMS	9-7
Basic Engine Air	9-7
Aft Fan	9-9
Thrust Reverser	9-9
Turbine Cooling Air	9-9
Sump Pressurizing Air	9-11
Fuel-Heating Air	9-11
Engine Anti-Icing Air	9-11
Engine Cooling Air System	9-11
Engine Air Inlet Vortex Destroyer	9-14
IGNITION AND STARTING SYSTEM	9-14
Ignition and Starting Controls	9-14
Ignition Selector Switch	9-14
Ground-Flight Starter Switches	9-16
ENGINE INSTRUMENTS	9-16
Basic Engine Speed Indicating System	9-17
Aft Fan Speed Indicating System	9-17
Exhaust Gas Temperature Indicating System (EGT)	9-17
Engine Pressure Ratio Indicating System	9-17
Thrust Reverser Position Indicating System	9-17
ENGINE CONTROL AND INDICATION CIRCUIT ELECTRICAL SOURCES	9-17

POWER PLANT

ENGINES

Propulsion power is provided by four pod-mounted General Electric CJ805-23 axial-flow aft fan turbojet engines. Each complete engine weighs approximately 4335 pounds (including thrust reverser) and is rated in the 16,000 pound thrust class. Special features of the engine include: One major rotating compressor-turbine element, a canular type combustion system, a three stage turbine rotor, variable compressor starter blades and inlet guide vanes, and one aft fan free floating compressor turbine air bypass element (see Figures 9-1, 9-2 and 9-3). A thrust reverser mounted on the rear frame of the aft fan is utilized as an additional braking device during the landing roll of the airplane. Performance characteristics of the aft fan and engine exhaust area eliminate the requirement for a sound suppressor.

ENGINE DATA

Model Number	CJ805-23
Stages of Compression	17
Compression Ratio (basic engine)	13 : 1
Compression Ratio (aft fan)	1.6 : 1
Oil System	Recirculation Type
Ignition System (dual)	Capacitor-Discharge Type
Anti-Icing System	Bleed Air
Engine Speed Indicating	Tach-Generator & Indicator
Aft Fan Speed Indicating	Tach-Generator & Indicator
Exhaust Gas Temperature Indicating (EGT)	Thermocouples-to-Indicator

ENGINE OPERATING CONTROLS

The engine control system consists of a power control lever, a thrust reverse lever, and a fuel shutoff lever for each engine. All controls are mounted on the pilot's pedestal. Individual thrust reverse levers, in the form of an "L", are attached to and forward of the power control lever assemblies between the power control lever handles and the pedestal cover. The thrust reverse levers cannot be used until the power control levers are in the IDLE position (see Figure 9-4).

Power Control Levers

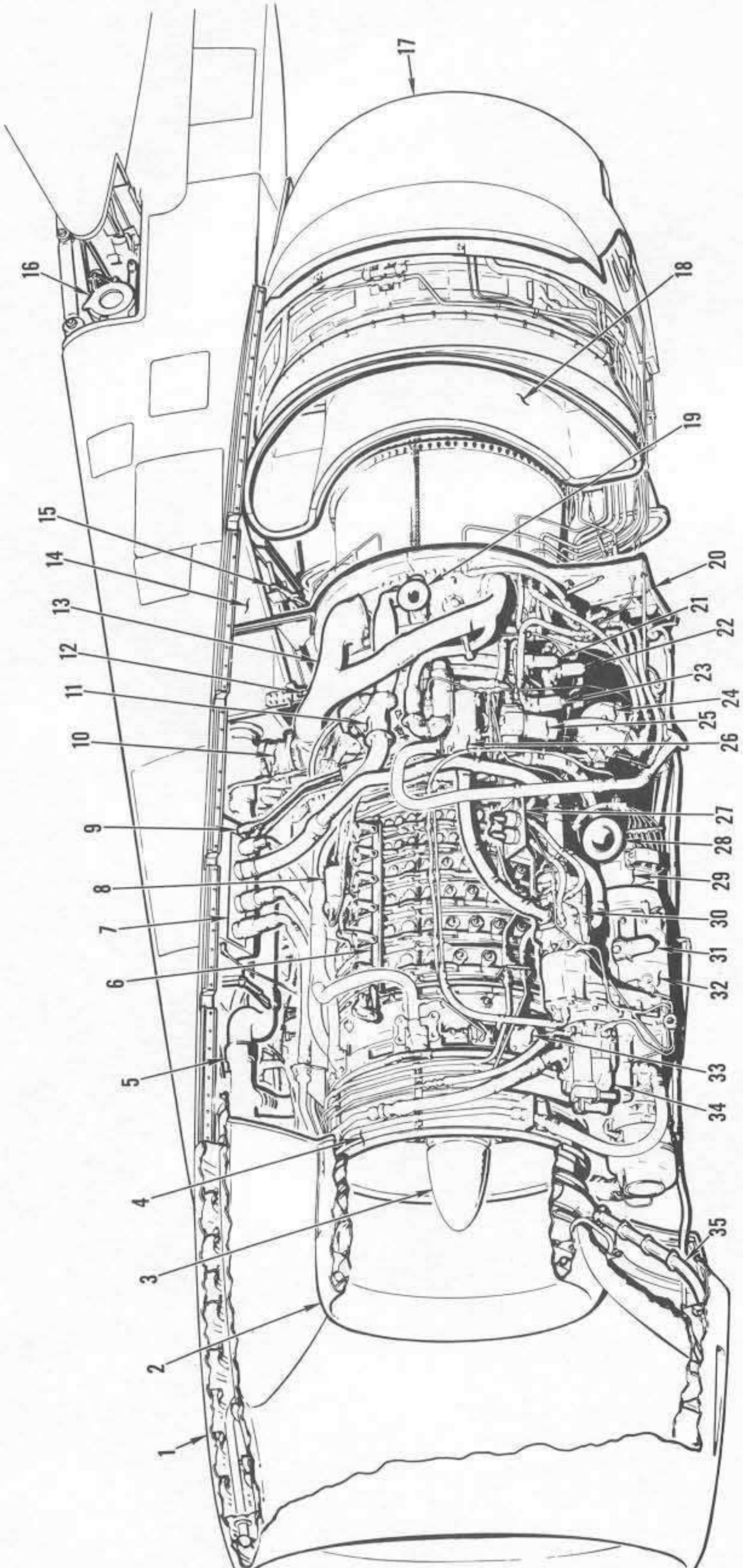
The position of the power control levers during takeoff, cruise and landing, is directly related to the engine fuel scheduling and affects the percent rpm. The power control lever quadrants are marked IDLE POWER and TAKEOFF POWER. Cams in the power control lever mechanisms actuate microswitches to give warning by horn if the flaps, stabilizer trim and landing gear are not in their correct positions at critical times during takeoff and landing.

To Be Furnished

Engine Sections
Figure 9-1

To Be Furnished

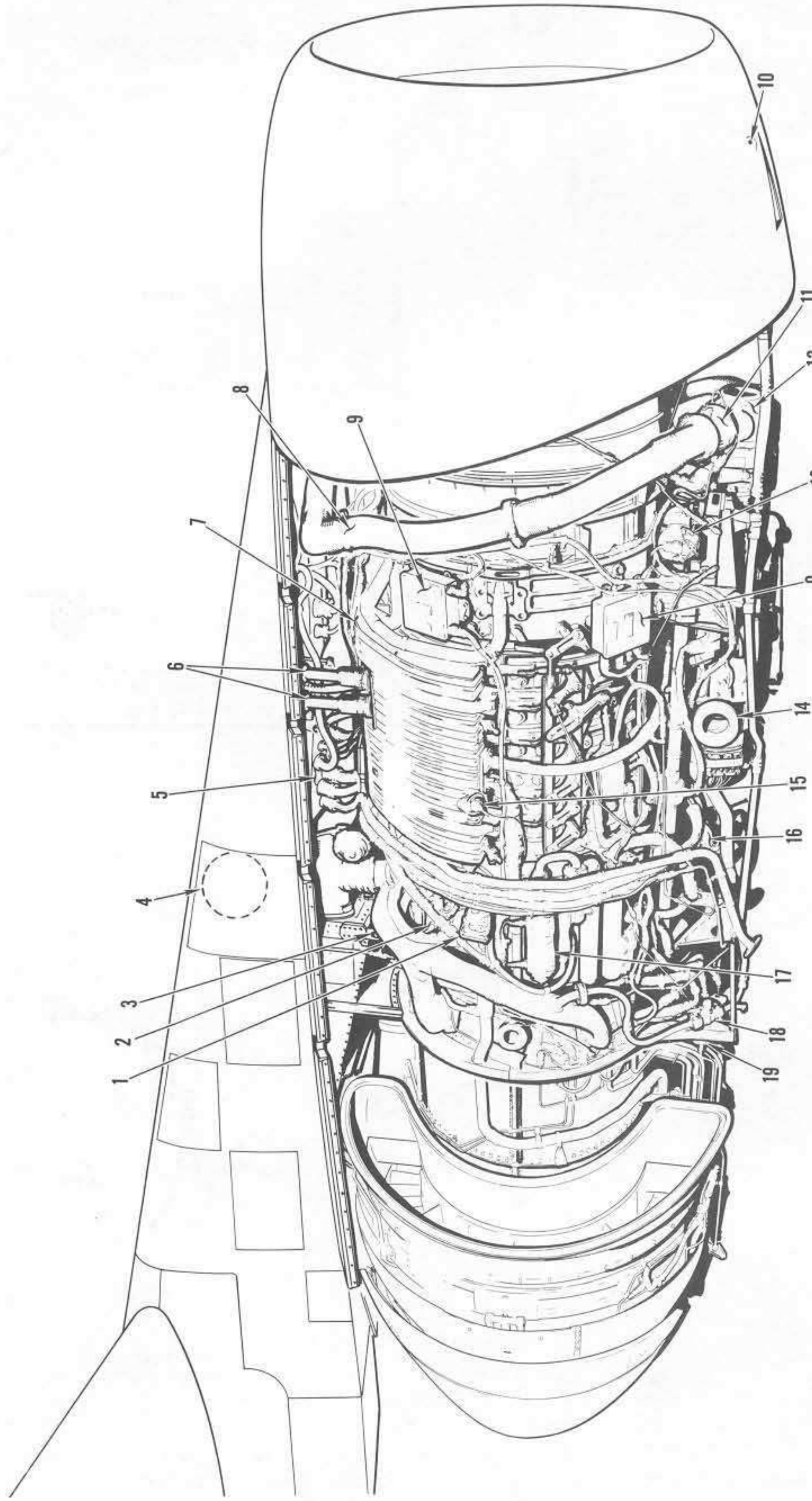
Engine - Exploded View
Figure 9-2



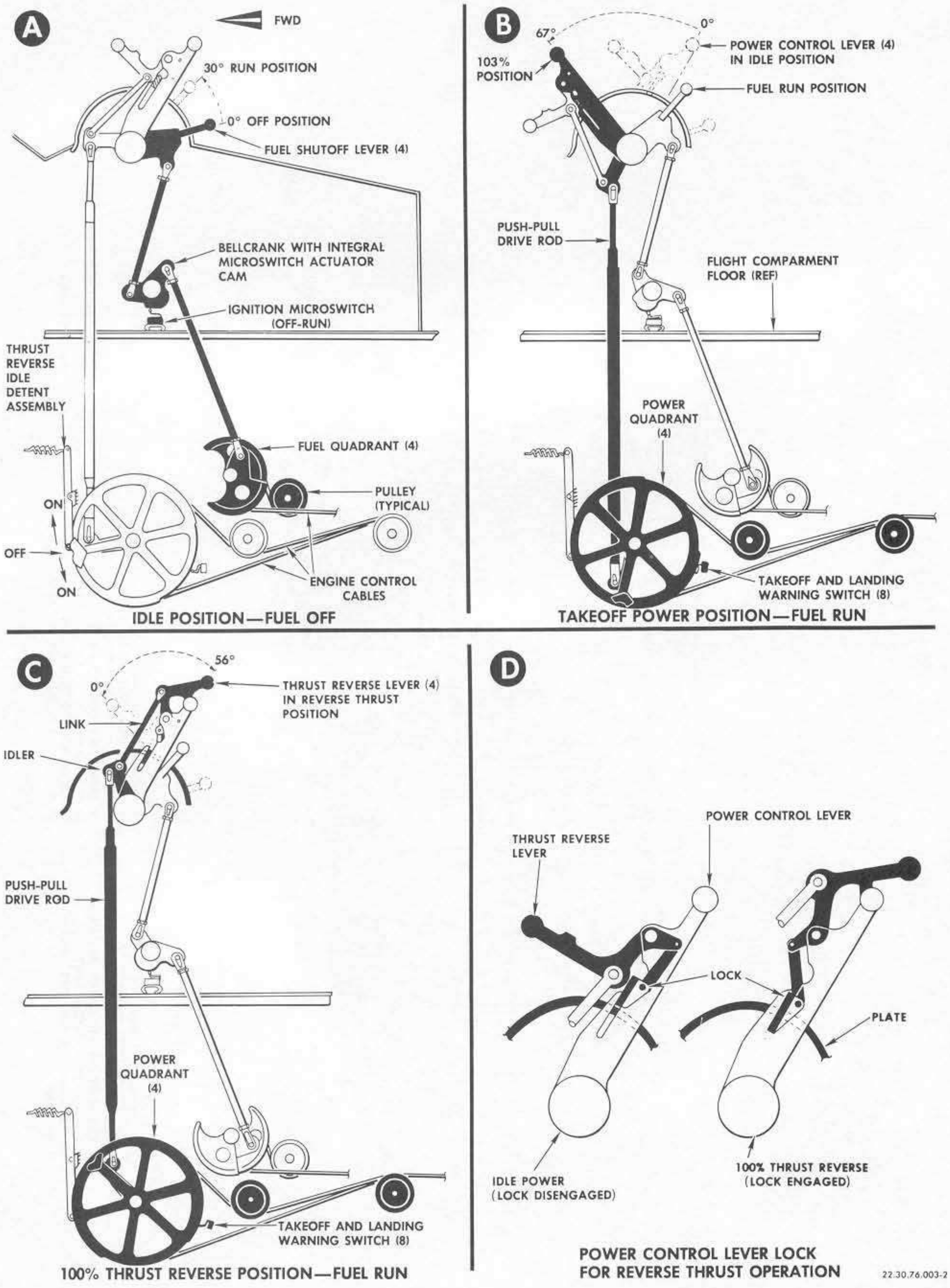
1. NOSE COWL OUTER DUCT
2. NOSE COWL INNER DUCT
3. HUB BULLET NOSE FAIRING
4. ENGINE AIR INLET STUB DUCT
5. STARTER DUCT (POD)
6. VARIABLE STATOR CONTROLS
7. FLUID DISCONNECT PANEL
8. VARIABLE STATOR ACTUATOR
9. POWER AND FUEL SHUTOFF CONTROL QUICK-DISCONNECTS
10. BLEED AIR SHUTOFF VALVE
11. ENGINE ANTI-ICING VALVE
12. ENGINE FORWARD (LH) MOUNT
13. BLEED AIR MANIFOLD
14. PYLON STRUCTURAL TORQUE BOX
15. FIRE EXTINGUISHING AGENT OUTLETS
16. PYLON TORQUE BOX
17. THRUST REVERSER DOOR
18. AFT FAN AIR INLET STUB DUCT
19. DISCHARGE AIR DUCT
20. VERTICAL FIREWALL
21. OIL SCAVENGE FILTER
22. LUBE DISCHARGE FILTER
23. THRUST REVERSER OIL FILTER
24. ENGINE TORQUE BOX
25. FUEL FILTER
26. FUEL HEATER
27. FUEL LOW-PRESSURE WARNING SWITCHES
28. GENERATOR COOLING AIR INTAKE DUCT
29. GENERATOR
30. FUEL PUMP
31. CSD OIL FILTER
32. CSD UNIT
33. ENGINE ANTI-ICING AIR PRESSURE SWITCH
34. HYDRAULIC PUMP
35. ENGINE COMPARTMENT ACCESSORY COOLING DUCT

30.02.71.002 1

Engine Pod and Pylon Relationships
Figure 9-3 (Sheet 1 of 2)



Engine Pod and Pylon Relationships
Figure 9-3 (Sheet 2 of 2)



Engine Control Levers
Figure 9-4

Fuel Shutoff Levers

The fuel shutoff levers open the fuel control valves in the fuel control unit and also actuate the engine ignition switches to the "on" position. When the engine speed reaches 47% (3500 rpm), the ignition switch circuit is automatically opened and the starter is disengaged. The fuel shutoff lever quadrants are marked OFF and RUN. The fuel shutoff lever must be pulled outward slightly to release from the OFF position. A similar index is used at RUN position.

Thrust Reverse Levers

The short side of the "L" shaped thrust reverse lever is attached to, and has its pivot point on, the upper section of each power control lever. The thrust reverse levers are locked against operation until the power control levers are in idle position. With the power control lever in idle position, the thrust reverse levers can be moved up and aft. This movement initiates the thrust reverser "clam shell" action and applies increasing engine thrust as lever movement is continued. A "lock-out" safety feature prevents thrust application in the reverse mode on any engine whose thrust reverser system has failed.

ENGINE AIR SYSTEMS

For purposes of operational information, the air streams can be separated as follows: basic engine air, turbine cooling air, sump pressurizing air system, fuel heater air, anti-icing air system, engine cooling air system and the engine air inlet vortex destroyer (see Figure 9-5).

Basic Engine Air

Air enters the engine at the front frame and passes through seventeen stages of compressor rotor blades and compressor stator vanes. Each stage of blades compresses the air, but not in equal proportions. The exit passages from the compressor section are designed to control the velocity of the air as it enters the combustion section.

The inlet guide vanes and the first six stages of stator vanes are automatically varied. These variable vanes enable the engine to be accelerated and decelerated rapidly without stalling, and to maintain high performance characteristics throughout its operating range.

Air from the compressor section flows into the combustion section. A portion of this air is mixed with fuel and burned. The remaining air blankets the inner and outer casings to provide necessary cooling. The burning of the fuel-air mixture in the liners produces exhaust gases of high velocity. These exhaust gases pass through the transition liners where they are transformed from ten individual streams into a single annular flow. The first stage turbine nozzle then directs the exhaust onto the first stage turbine buckets in the turbine section at the proper angle. The turbine section extracts power from the gas stream to drive the compressor rotor and the various engine mounted accessories. This section consists of the turbine rotor, connected to the compressor rotor, the turbine stators and the turbine frame which is bolted to the outer combustion casing. As the gas stream flows into the turbine section,

To Be Furnished

Engine Air Flow
Figure 9-5

it strikes the first stage turbine rotor blades. It then continues on through the three states of rotor blades and turbine stator nozzle guide vanes. Power imparted to the turbine rotor turns the compressor rotor and accessory drives. The gas stream then flows into the aft fan turbine section. A small portion of the basic engine exhaust gas flows past the fan rotor seals and into the inner area between the fan front frame and fan rear frame. The air flows from this area out through the struts of the fan rear frame and into a hat section (manifold) around the circumference of the frame. This air serves a heating function in order to decrease thermal stresses in the frame.

Aft Fan

The aft fan assembly is bolted to the engine turbine stator casing rear flange. It consists of a single stage free-floating rotor, which is part compressor and part turbine; and a front and rear frame incorporating concentric annular-flow passages. The fan is aerodynamically linked with the engine turbine compressor element and has a speed directly related to that of the engine. Gases from the engine turbine flow into the inner passage of the fan front frame and through the partitions of the fan turbine nozzle. The nozzle diverts the gases at the correct angle of attack towards the fan turbine blades. The gases impart energy to the turbine and pass into the exhaust section. Secondary air ducted through the nacelle passes into the outer section of the fan front frame and is compressed by the fan compressor blades. The compressed air then passes through outlet guide vanes to the exhaust section of the fan.

Thrust Reverser

The thrust reverser provides a means of changing the direction of exhaust gas thrust, to create a reverse thrust braking effect, thereby reducing the length of the airplane landing roll after "touchdown". With the reverser in the stowed, or forward thrust position, the exhaust gases flow aft through the reverser section. With the reverser in the reverse thrust position, the gas flow is diverted and discharged from the engine in a forward direction. (See Figure 9-6.) The thrust reverser actuating system is hydraulic, utilizing oil from the CSD section of the oil reservoir. For operational details, consult Section 11, OIL SYSTEM.

Turbine Cooling Air

The air used to cool the turbine section components is cool only in comparison with the temperature in the turbine section. The paths which this cooling air takes in the process of cooling the turbine section are as follows: Seventh and ninth stage air, bled internally from the compressor rotor, travels through the inside of the turbine shaft, is directed across each internal face of the wheels by baffles and is discharged into the turbine frame through the stub shaft. Cooling air from the combustion section takes three paths into the turbine section. The outer layer of the combustion cooling air is divided. Part is directed through the first stage turbine nozzle partition and part is directed through the second stage partitions, cooling the rear face of the first stage wheel. The inner layer of combustion cooling air is bled into the area between the inner combustion casing and the turbine shaft through holes in the inner casing. This air is allowed to discharge at the first stage

To Be Furnished

Thrust Reverser
Figure 9-6

turbine wheel and acts as a coolant for the forward face of the first stage wheel. The No. 3 bearing air seal leakage passes by a baffle in the turbine frame and is then discharged into the exhaust gases. In taking this path it cools the aft face of the third stage turbine wheel and buckets.

Sump Pressurizing Air

The sump pressurizing air system (described in detail in Section 11, OIL SYSTEM) serves a two-fold purpose; it prevents oil leakage past the oil seals and keeps a positive pressure in the sumps. The air system is divided into two parts: high-pressure supply to the No. 1, No. 2, No. 3, No. 4 and No. 5 bearing areas enclosing the sump; and low-pressure control within the No. 1, No. 2, No. 3, No. 4 and No. 5 bearing sumps, the accessory gearboxes and the oil tank.

Fuel-Heating Air

The fuel-heating air system is a means of preventing ice forming in the fuel filter. Seventeenth stage compressor bleed air is taken from the compressor rear frame and routed through a manifold to the fuel heater, an air-to-liquid type heat exchanger. The heater includes an automatic air shutoff valve which controls the flow of air as a function of heater discharge fuel temperature.

Engine Anti-Icing Air

The engine anti-icing system prevents accumulation of ice in the compressor inlet. Anti-icing is accomplished by passing seventeenth stage compressor discharge air from the compressor rear frame through a valve to the front frame struts and inlet guide vanes by means of a manifold. Specifically, the system anti-ices the compressor front frame hub, bullet, and struts, the inlet guide vanes and the nose cowl inlet air lip and the fan front frame struts (see Figure 9-7). The system can be controlled manually by the crew or can be connected to the airplane ice-sensing system for automatic operation.

Engine Cooling Air System

The engine and accessory cooling system consists of the engine accessory compartment cooling air inlet and distribution duct, generator cooling duct, generator cooling air entrance and exit duct, engine accessory compartment, cooling air exit duct, engine turbine compartment and aft fan cooling inlet and distribution ducts and exit, leakage air manifold and exits and the main engine mount heat shield. The cooling system and figuration eliminates the reverse flow of air from the engine compartment areas back into the engine inlet duct, and provides adequate cooling with minimum drag losses for all airplane operational conditions. Cooling air airscoops are so located that a fire cannot enter from any other pod compartment or from any part of the airplane (see Figure 9-8).

To Be Furnished

Compressor Inlet Anti-Icing Air Flow
Figure 9-7

To Be Furnished

Engine Pod Cooling Air System
Figure 9-8

Engine Air Inlet Vortex Destroyer

A jet air blast nozzle is installed in the lower leading edge of the nose cowl. The jet blast airstream is directed downward from the nose cowl to the ground at the vortex base of the suction pressure that is generated between the ground and the engine air inlet during engine ground operation. The function of the jet blast is to "break-up" forces that occur when a vortex forms, thereby minimizing ingestion of foreign material into the engine during ground engine operation. The jet blast system uses engine bleed air for its pressure source and is controlled by an automatic shutoff valve located in the engine accessory section (5 o'clock position just forward of the firewall). The shutoff valve is connected to a landing gear switch to prevent vortex destroyer operation while airborne.

IGNITION AND STARTING SYSTEM

The ignition circuit provides the spark necessary to ignite the fuel-air mixture in the combustion section of each engine. Once ignited, the fuel-air mixture continues to burn; therefore, it is necessary to power the ignition circuit only during the engine starting period. The ignition circuit consists of two high energy ignition systems, each system having its own lead, voltage booster and ignitor plug. Ground engine starting is accomplished through the use of individual air turbine starting units on each engine (see Figure 9-9). The starter assembly for each engine is attached to a mounting pad on the transfer gearbox; the gear ratio between the engine rotor and the starter spline is one-to-one. Each starter incorporates an engaging mechanism to permit automatic engagement and disengagement with the engine, a turbine chamber which houses the turbine blades, and a radial air inlet nozzle. Equipment built into the starter control valves prevents overspeeding of their turbine wheels when the starters disengage from the engine at starter cutoff speed.

If, through incorrect fuel management, all four engines simultaneously flame out, they may be restarted by using the emergency inverter. This inverter receives its power from the emergency battery. Two engines can be started simultaneously if desired.

Ignition and Starting Controls

The ignition and starting controls consists of one IGN SEL switch and four GROUND/FLIGHT starter switches. All of these switches are located on the pilot's overhead switch panel.

Ignition Selector Switch

The IGN SEL switch is located on the ENGINE STARTER switch panel and enables the pilot to select SYS NO. 1, SYS NO. 2, or BOTH. The BOTH position is operative only on the ground. In flight, one system only will be operating, even though selection to BOTH is made.

To Be Furnished

Engine Starting Air Supply Schematic
Figure 9-9

Ground-Flight Starter Switches

A ground air connection is located on the right underside of the fuselage near the wing front spar. This connection is used to supply air from a ground source through the airplane crossover bleed system manifold to start any engine or engines.

The four ground-flight starter switches are three position, GROUND-(off)-FLIGHT toggle switches. In GROUND position, the switch energizes an ac operated relay that completes its circuit through a centrifugal cutout switch operated by the air-turbine starter. This ac operated relay actuates the air starter control valve, supplying air to the air-turbine starter. At the same time, this switch makes 115-volt ac power available to the ignition switch operated by the fuel shutoff lever. The 115-volt ac power continues through the fuel shutoff lever controlled switch, when in RUN position, to the NO. 1, NO. 2, or BOTH ignition systems, depending on the position of the ignition system selector switch. In FLIGHT position, the air-turbine starter and the BOTH ignition selection are inoperative.

ENGINE INSTRUMENTS

The various engine operation indicating systems are installed on the pilots' engine instrument panel and the flight engineer's panel. Engine anti-icing and bleed air controls and indicators are located in the pilots' overhead switch panel. Engine indicating systems are:

1. Basic engine speed indicating system (RPM).
2. Aft fan speed indicating system (RPM).
3. Engine pressure ratio indicating system (EPR).
4. Exhaust gas temperature indicating system (EGT).
5. Oil system indicating systems: (see Section 11, OIL SYSTEM).
   A. Oil quantity indicating system.
   B. Oil temperature indicating system.
   C. Oil pressure indicating system.
   D. Oil low-pressure warning light system.
6. Fuel indicating systems: (see Section 10, FUEL SYSTEM).
   A. Fuel flow indicating system.
   B. Fuel temperature indicating system.
   C. Fuel pump low-pressure warning light system.

   NOTE: Airplanes CV26 and ON are equipped with Fuel Flow/Consumed totalizers.

7. Thrust reverser position indicating system.

Basic Engine Speed Indicating System

The engine speed indicating system consists of the tachometer generator and an indicator for each engine. The generator is driven through a gearbox and generates its own voltage for operation of the system. The generated voltage is transmitted to the indicator which displays the signal as an indication of engine rpm in percentage figures.

Aft Fan Speed Indicating System

The aft fan speed indicating circuit consists of a tachometer-generator and an indicator (one for each engine). The tachometer-generator is coupled by a spline to the fan rotor. Operation of the fan speed indicating circuit is identical to that of the basic engine.

Exhaust Gas Temperature Indicating System (EGT)

The exhaust gas temperature indicating system incorporates transistorized, null-balancing indicators and dual-loop thermocouple harnesses. One loop of each dual loop thermocouple harness is attached to each side of an engine. The millivolt output of each thermocouple is calibrated to the temperature indicating system's heat range curve, and the resultant voltage is compared to a reference voltage existing in the indicator. The comparison result is displayed in degrees Centigrade on the exhaust gas temperature indicator (EGT).

Engine Pressure Ratio Indicating System

The engine pressure ratio indicating system consists of a transmitter and an indicator for each engine. The transmitter senses pressure from a total pressure pickup in the exhaust area. This pressure is compared to ram pressure supplied by the engine inlet pitot head. The two values are integrated and the results are converted to an electrical signal. The electrical signal is transmitted to the indicator where it is amplified and displayed in pressure ratio readings. A power off warning flag is provided.

Thrust Reverser Position Indicating System

Thrust reverser position indication is accomplished through microswitches which are interlocked with the engine power control levers. The thrust reversers cannot be activated until the power control levers are in IDLE position. Actuation of the thrust levers closes the microswitches, illuminating the blue IN-TRANSIT lights. When the reversers are in full functioning position, amber REV-THRUST lights illuminate. The IN-TRANSIT lights also illuminate during the return of the thrust reversers to normal position, or if the thrust reversers are stuck in-transit.

ENGINE CONTROL AND INDICATION CIRCUIT ELECTRICAL SOURCES

Engine anti-icing air system, pilot's ac essential bus.
Ignition, pilot's ac essential bus.
Starter switch, 28-volt dc emergency bus.
Basic engine speed indicating system, self powered.
Aft Fan speed indicating system, self powered.

Exhaust gas temperature indicating system, pilot's ac essential bus.
Thrust reverser position indicating system, 28-volt dc emergency bus.
Engine pressure ratio indicating system, pilot's ac essential bus.

10

FUEL SYSTEM

Section 10

FUEL SYSTEM

TABLE OF CONTENTS

TITLE	PAGE
GENERAL FUEL SYSTEM	10-1
FUEL TANKS	10-2
Replenishment Tanks	10-2
Main Tank Wells	10-2
Anti-Shock Body Tank	10-2
Center Section Tank	10-7
PRESSURE REFUELING SYSTEM	10-7
GRAVITY REFUELING SYSTEM	10-7
DEFUELING SYSTEM	10-7
ENGINE FUEL SUPPLY SYSTEM	10-9
Fuel Transfer Pumps	10-9
Replenishment Tank Fuel Level Indication	10-9
Fuel Boost Pumps	10-9
Emergency Fuel Shutoff Valves	10-9
Engine Fuel and Control System Block Diagram	10-12
Engine Fuel Pump	10-12
Engine Fuel Heater	10-12
Engine Fuel Filter	10-12
Engine Fuel Control	10-14
Fuel Shutoff Levers	10-14
Engine Fuel-Oil Cooler	10-14
Fuel Flow Transmitter	10-14
Constant Speed Drive Fuel-Oil Cooler	10-15
Fuel Pressurizing and Drain Valve	10-15
Engine Fuel Nozzles	10-15
FUEL JETTISON SYSTEM	10-15
Scavenge System	10-15
Scavenge/Jettison Systems Hydraulic Power Source	10-17
Fuel Jettison Nozzles	10-17
FUEL TANK VENT SYSTEM	10-17
Ram-Vent Scoop	10-17
FUEL CONTROL PANEL	10-17
Fuel Quantity Gaging System	10-19
Pump Low Pressure Warning Lights	10-19
Engine Fuel Pump Warning Lights	10-19
Emergency Fuel Shutoff Indicator Lights	10-20

TABLE OF CONTENTS (CONT)

TITLE | PAGE

FUEL CONTROL PANEL (CONT)
Valve In Transit Indicator Lights.................................. 10-20
De-Fuel Valve Indicator Lights..................................... 10-20
Outboard Anti-Shock Body Warning Light.............................. 10-20
De-Fuel Valve Switches.. 10-20
Line Valve Switches... 10-20
Crossfeed Valve Switches.. 10-20
Boost Pump Control Switches... 10-21
Tank No. 1 and Tank No. 4 Transfer Pump Switches.................... 10-21
Tank No. 2 and Tank No. 3 Transfer Pump Switches.................... 10-21
Jettison Switches... 10-21
Scavenge Switches... 10-21
Center Section Jettison Valve Selector Switches..................... 10-21

FUEL TEMPERATURE INDICATOR.. 10-21

REAR SPAR UNDER-WING REFUELING CONTROL PANEL......................... 10-22
Fuel Quantity Indicators.. 10-22
Precheck Switches... 10-22

LIQUID SIGHT GAGES.. 10-22

FUEL SYSTEM ELECTRICAL SOURCES...................................... 10-24
Boost and Transfer Pump Operating Power Sources..................... 10-25
Defueling Valve Power Source.. 10-25
Fuel Tank Shutoff Valve Power Source................................ 10-25
Crossfeed Valve Power Source.. 10-25
Emergency Fuel Shutoff Valve Power Source........................... 10-25
Fuel Jettison System Power Source................................... 10-25
Fuel Quantity Gaging System Power Source............................ 10-25
Fuel Pump Low Pressure Warning Light Power Source.................... 10-25
Fuel Temperature Indicator Power Source............................. 10-25
Rear Spar Under-Wing Refueling System Power Source................... 10-26
Fuel Flow Indicator Power Source.................................... 10-26

FUEL SYSTEM

GENERAL FUEL SYSTEM

The fuel system includes subsystems designed to supply and control the flow of fuel to the four engines without interruption under all flight conditions. (Figure 10-1). In the supply systems, two main tanks in each wing are divided into four fuel-tight compartments. Two compartments serve as replenishment supply to the two main tanks (Figure 10-2). Two Anti Shock Body tanks, located on the upper surface of each wing, supply fuel directly into their respective main tanks by means of transfer pumps. Each wing fuel system operates independently of the other. Each main tank contains two electric fuel boost pumps, located in a well, each operated from a different power source, and one hydraulically operated jettison pump. In addition, each inboard main tank well contains a hydraulically operated fuel scavenge pump.

One five-bay auxiliary fuel tank is provided in the wing center section. The five bays in the center section each contain a fuel bag. Hydraulically driven boost/jettison pumps in the aft cell of the center section are located in sumps which are similar in function to the wells in the main tanks. These pumps are supplied with fuel by gravity flow, through check valves, from the forward cells. The boost/jettison pumps supply fuel from the center section tank to any engine via the cross-feed system, and also jettison fuel when one or both of the two center section tank jettison valves is opened.

Drain valves for external draining, drain check valves for internal draining, and check valves for fuel control are installed throughout the tank systems.

The cross-feed part of the fuel supply system routes fuel from any main tank or from the center section tank to any or all engines. Selection is controlled by electrically-driven fuel shutoff valves installed on the rear spars. Low pressure switches mounted on the rear spars are connected by tubing to each boost and transfer pump, and to the boost/jettison pumps.

The engine fuel control system regulates the amount of fuel delivered to the engine in relation to the desired RPM, compressor inlet air pressure and temperature, and acceleration and deceleration rate.

The normal refueling procedure uses underwing pressure refueling adapters. Automatic selection of any quantity up to full tank, exclusive of the minimum allowable expansion space, is possible. Fuel tanks can be filled simultaneously by using all four fuel adapters, or individually (Figure 10-3). Overwing gravity refueling is possible into the wing replenishment tanks, through four 3 inch openings. The auxiliary center section tank and the ASB tanks have similar service openings. The center section tank service opening is located on the upper surface of the left wing, over the No. 2 main tank; the ASB openings are on their upper surfaces.

A refuel control panel on the aft side of the rear spar of each wing, between bulkheads 28 and 29, incorporates fuel quantity repeater gages, preset quantity selectors, precheck shutoff controls, and a refueling flow indicator light for each system.

FUEL TANKS

All fuel tanks except the center section fuel cells and the anti-shock body compartments are built as part of the basic airplane structure. The wings and ASB tanks are sealed by the "Scotchweld" process during assembly. A trap compartment in each ASB is located aft of the fuel compartment. The wing center section tank is made up of fuel-tight cells, each a two-ply nylon fabric bag impregnated with buna-N rubber. The airplane structure supports the bags, with the bags held in place by nylon cords attached alternately to the structure at approximately 15 inch spacing. Plumbing fittings in the center section are all flat faced and are supported by the structure through sandwiching of fittings, seal, bag, and structure. Interconnect fittings allow free flow of fuel from one cell to another. An open drain line and a drain valve drain moisture from the center section area outside the fuel cells. Also, a screw in the skin of each of the four front bays may be removed to drain moisture.

The outboard replenishment tanks are located outboard of their respective main tanks; the inboard replenishment tanks are forward of their respective main tanks, between the wing front spar and center spar as shown in Figure 10-2. Fuel is free to flow between the inboard replenishment and inboard main tanks. There are no check valves between these two tanks and fuel flows between them as dictated by the airplane's attitude. The ASB tanks, which house auxiliary fuel for the main tanks, are mounted on the upper aft surfaces of the wings, two on each wing, at wing bulkheads 7 and 16.

Refer to Chapter 12, SERVICING, of the MAINTENANCE MANUAL for complete capacity tabulations and servicing information. The Approved Airplane Flight Manual lists the maximum refuel quantity and usable fuel quantity as required by F.A.A. regulations.

Replenishment Tanks

Each wing fuel tank is composed of two separate sections, called the replenishment tank and the main tank. The replenishment tanks for fuel tank No. 1 and fuel tank No. 4 are located outboard of the No. 1 and No. 4 main tanks. Fuel flows from the two outboard replenishment tanks to the main tanks by gravity flow through check valves in the tank bulkheads.

The replenishment tanks for fuel tank No. 2 and fuel tank No. 3 are located forward of the No. 2 and No. 3 main tanks. Fuel flows freely between the inboard replenishment tanks and the main tanks. Transfer pumps are provided to accomodate low-fuel, high-deck angle transfer conditions.

Main Tank Wells

A "well" section, with an approximate capacity of 135 gallons, is located at the inboard aft corner low point of each main tank. Transfer pumps and check valves maintain the highest possible fuel level in the wells. Each well contains two fuel boost pumps which supply fuel to the engine fuel pumps.

Anti-Shock Body Tanks

Two anti-shock body (ASB) fuel tanks are located on the trailing edge of each wing. Each anti-shock body is located aft of a main tank. Fuel is transferred

NOTES:

1. LH WING SHOWN — VIEW LOOKING DOWN
2. ✻ SCREWS — REMOVABLE FOR DRAINING BAYS

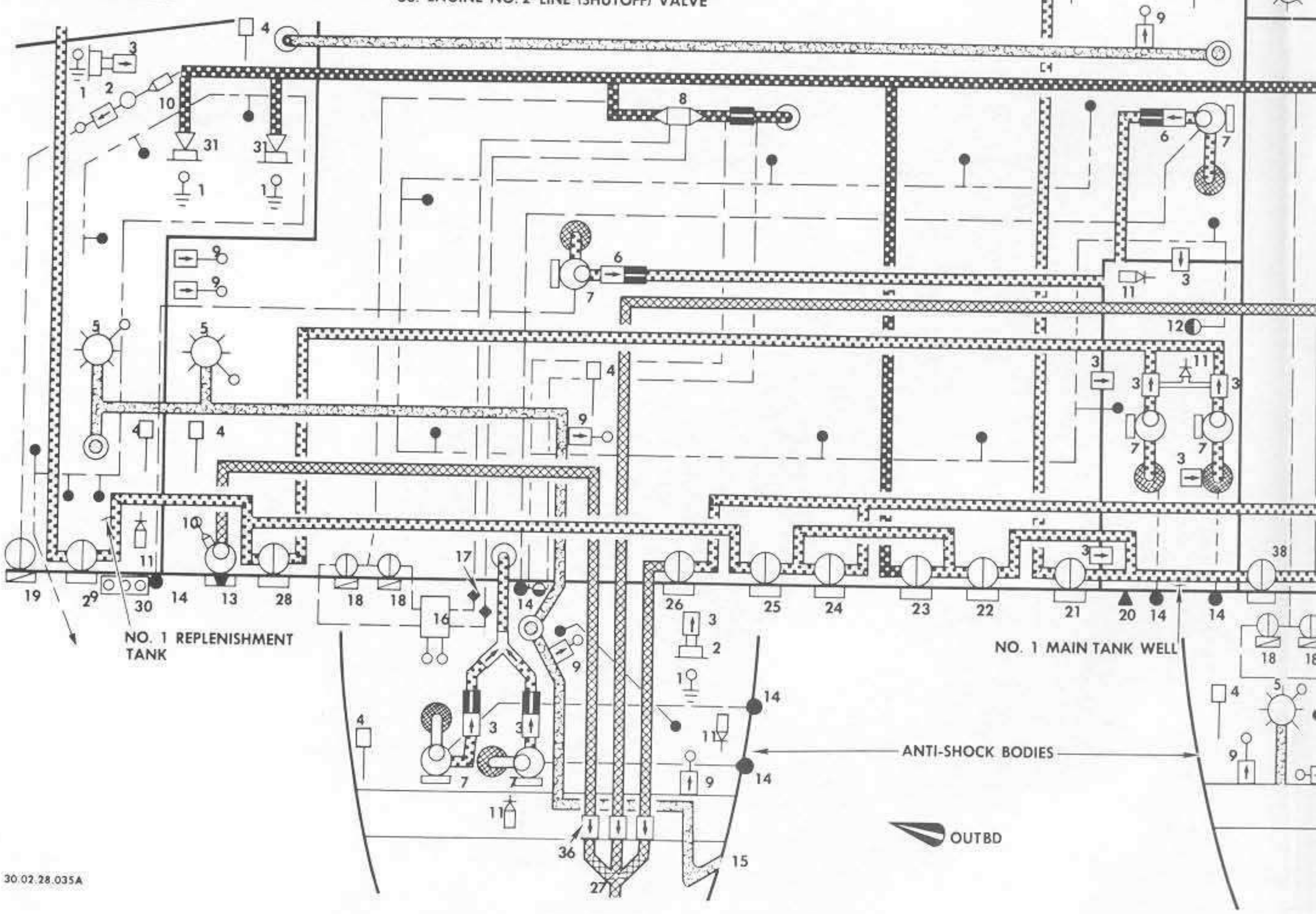
LEGEND:

- FUEL
- REFUEL-DEFUEL
- JETTISON
- VENT
- FUEL
- FUEL SENSING
- ELECTRICAL
- GROUND JACK
- GRAVITY REFUEL ADAPTER
- CHECK VALVE
- DRIPSTICK
- VENT VALVE — FLOAT OPERATED
- CHECK VALVE WITH RESTRICTOR
- ✻✻ DIFFERENTIAL PRESSURE SWITCH

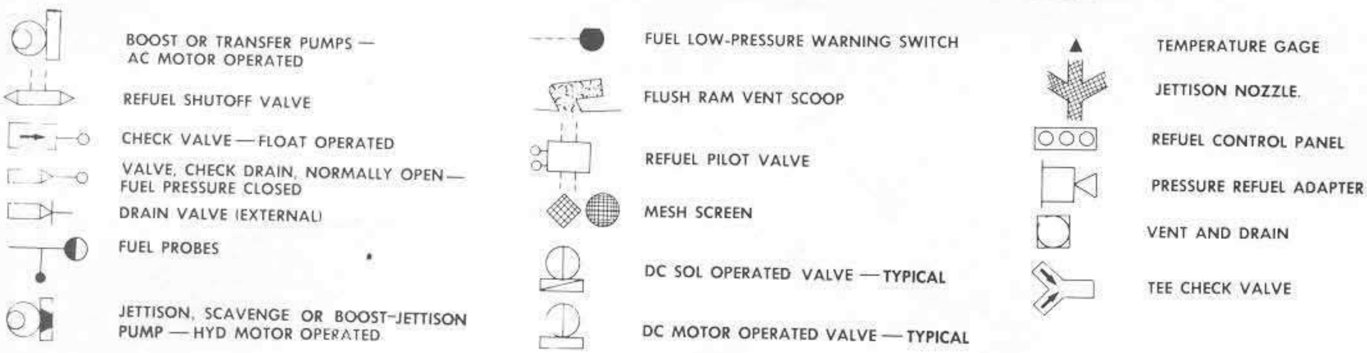
1. GROUND JACK
2. GRAVITY-FLOW REFUEL FILLER
3. CHECK VALVE—FLAPPER TYPE
4. LIQUID SIGHT GAGE
5. FLOAT OPERATED VENT VALVE
6. CHECK VALVE WITH RESTRICTOR
7. BOOST OR TRANSFER PUMP
8. REFUEL SHUTOFF VALVE
9. FLOAT OPERATED CHECK VALVE
10. CHECK DRAIN VALVE—DRAINS INTO TANK
11. DRAIN VALVE—DRAINS EXTERNALLY
12. FUEL PROBE
13. JETTISON OR SCAVENGE PUMP—HYDRAULICALLY OPERATED
14. FUEL LOW-PRESSURE WARNING SWITCH
15. RAM-VENT SCOOP
16. REFUEL PILOT VALVE
17. MESH SCREEN
18. STOP-FUEL VALVE
19. REFUEL-DEFUEL TRUNK LINE VENT SHUTOFF VALVE
20. TEMPERATURE GAGE
21. ENG. NO. 2 FUEL EMERGENCY SHUTOFF VALVE
22. ENGINE NO. 2 CROSSFEED VALVE
23. L/H DEFUEL VALVE
24. AUX CTR TANK & WING CROSSFEED VALVE
25. ENGINE NO. 1 CROSSFEED VALVE
26. AUX CTR SEC TANK JETTISON VALVE—L/H
27. JETTISON NOZZLE—L/H
28. ENGINE NO. 1 LINE (SHUTOFF) VALVE
29. ENG. NO. 1 FUEL EMERGENCY SHUTOFF VALVE
30. REFUEL CONTROL PANEL
31. PRESSURE REFUEL ADAPTER
32. BOOST-JETTISON PUMP—HYD MOTOR OPERATED
33. CTR TANK REFUEL SELECTOR VALVE
34. CTR SEC VENT AND DRAIN
35. TEE CHECK VALVE
36. JETTISON FUEL SHUTOFF VALVE—SPRING-LOADED POPPET TYPE CHECK
37. CAVITY PRESSURE DIFFERENTIAL VENT
38. ENGINE NO. 2 LINE (SHUTOFF) VALVE

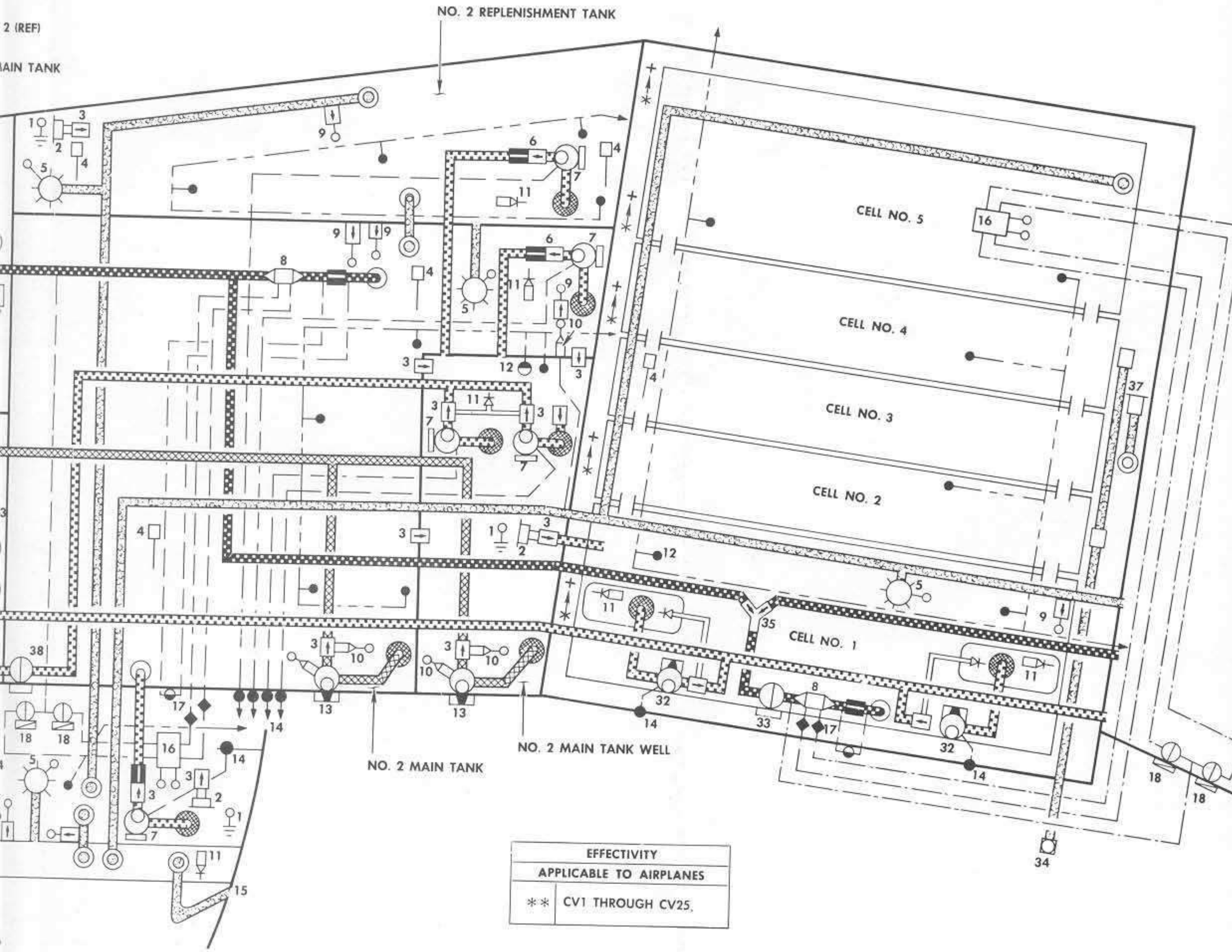
ENGINE NO. 2 (REF)

NO. 1 MAIN TANK



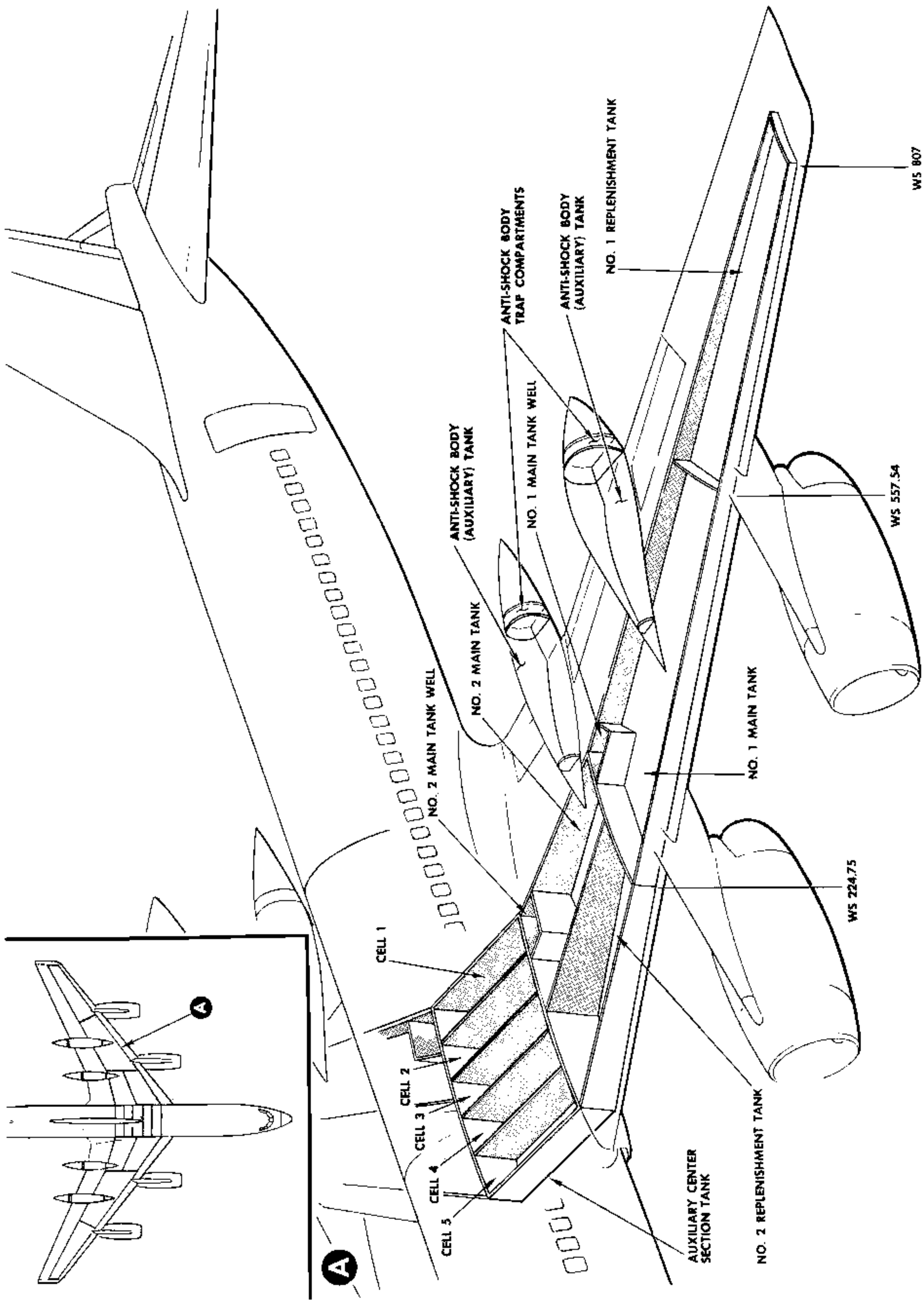
30.02.28.035A



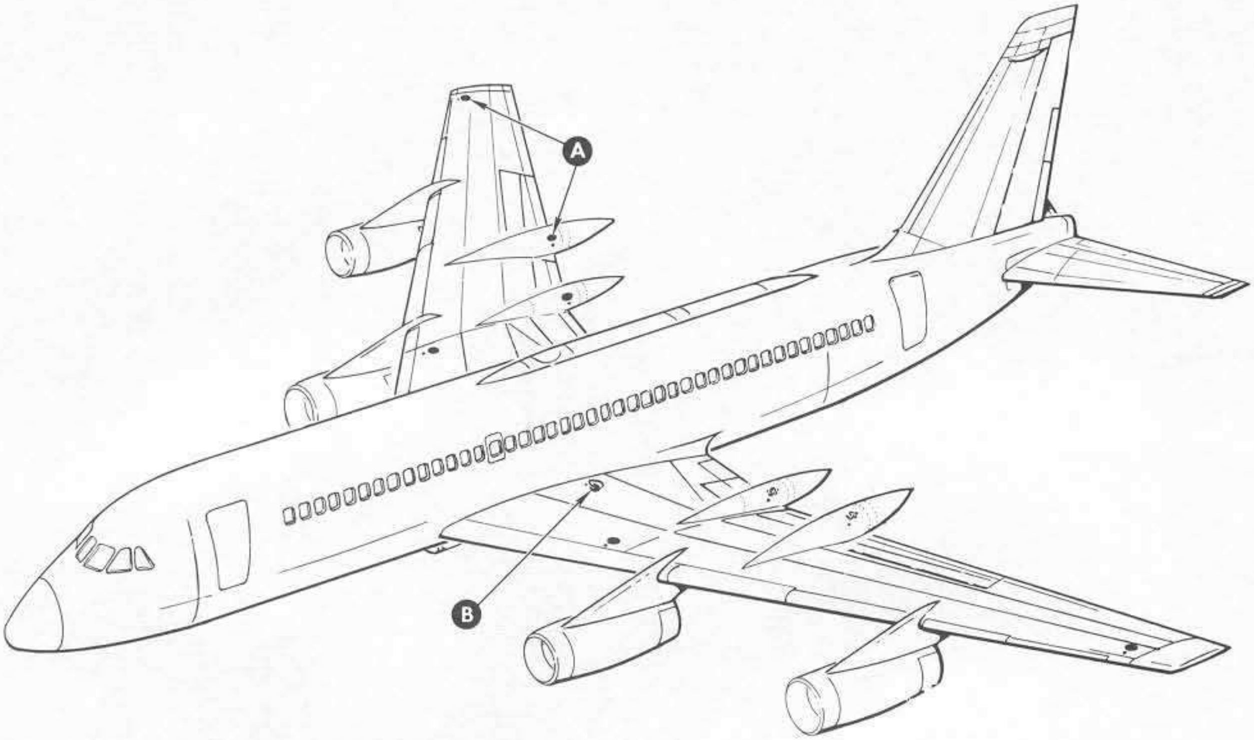


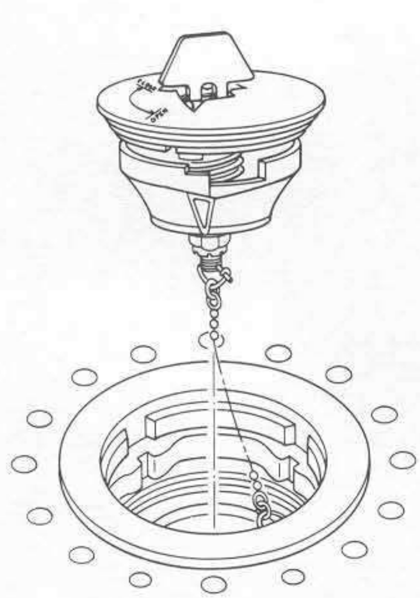
EFFECTIVITY	
APPLICABLE TO AIRPLANES	
**	CV1 THROUGH CV25.

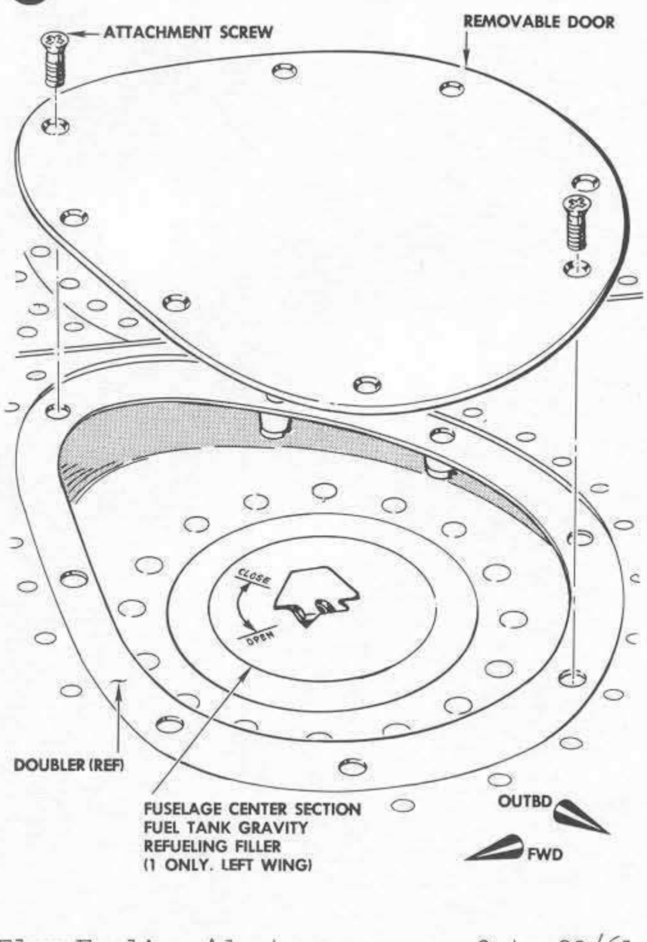
Fuel System Schematic
Figure 10-1



Fuel Tank Arrangement
Figure 10-2







30.30.28.011-2

Pressure and Gravity Flow Fueling Adapters
Figure 10-3

from each anti-shock body tank to its respective main tank by means of dual electric fuel transfer pumps in each outboard ASB and single electric fuel transfer pumps in each inboard ASB. A fuel jettison nozzle is located in the aft apex of each outboard anti-shock body.

Center Section Tank

The wing center section tank is made up of five fuel tight cells, each a two ply nylon fabric bag impregnated with buna-N rubber. The airplane structure supports the bags. They are held in place by nylon cords attached alternately to the bags and to the airplane structure. The center section tank area is accessible through an opening in the left side of the rear spar, forward in the left-hand main wheel well.

Five separate subsystems comprise the main fuel system: Refueling-Defueling, Engine Fuel Supply, Fuel Jettison, Fuel Tank Vent and Fuel Quantity Gaging.

PRESSURE REFUELING SYSTEM

Pressure refueling is the normal method of refueling. Pressure refueling adapters are located in the lower surfaces of the wings outboard of the No. 2 and No. 3 engine pylons. Pressure hoses connect to the individual adapters for pressure refueling and tanks may be filled one at a time or simultaneously. The total intake is 1200 gpm, based on a 300 gpm flow with 50 psig fuel pressure at each of the four adapters.

The outboard pressure refuel fittings are also used to refuel the center section tank. A center tank refueling selector valve is installed in the center tank refueling trunk line. With this valve closed, the wing outboard tank system can be refueled without refueling the center tank. With the valve open, both tanks will be refueled simultaneously.

CAUTION: MAXIMUM PRESSURE AT THE CONNECTORS SHOULD NEVER EXCEED 60 PSIG.

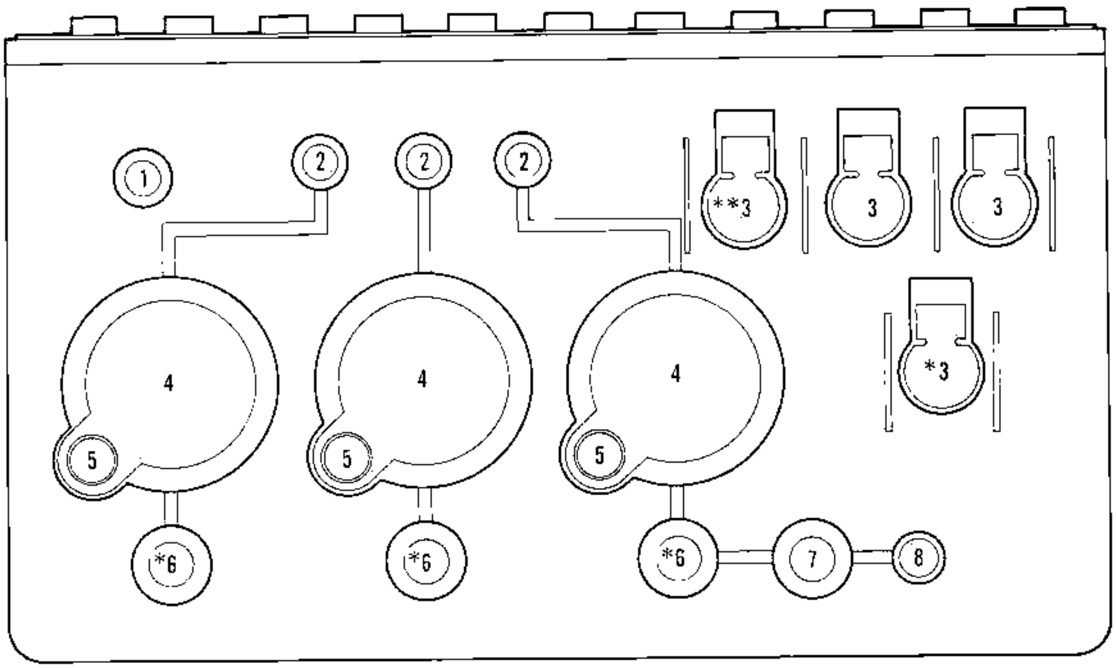
Two refueling control panels are provided, one on the aft side of the rear spar of each wing. With the airplane resting in a level attitude, the desired amount of fuel per tank is selected by presetting the quantity control knob located in the fuel control panel (Figure 10-4).

GRAVITY REFUELING SYSTEM

Three-inch openings located in each of the four replenishment tanks, and accessible from the upper wing surfaces, are used for gravity-flow refueling. The fillers are capable of receiving fuel from two-inch diameter nozzles at the rate of 150 gpm. As fuel enters the replenishment tanks, the fuel level is equalized in the main tanks and tank wells through check valves in the separating bulkheads. Gravity refueling of the center section is accomplished from an opening over the No. 2 Main tank. The Anti-shock bodies are gravity fueled through openings (one each) located on their upper surfaces.

DEFUELING SYSTEM

Defueling at the rate of 50 gpm is accomplished through any or all of the pressure fueling adapters by suction from the tanker. The defueling operation is controlled from the flight engineer's fuel control panel. Defueling of any or



1. COMPARTMENT LIGHT SWITCH (DOOR-OPERATED).
2. REFUEL PRECHECK CONTROL SWITCH.
3. COMMUNICATIONS EQUIPMENT JACK.
4. FUEL QUANTITY REPEATER INDICATOR.
5. REFUEL QUANTITY PRESET CONTROL.
6. REFUELING STOPPED INDICATOR LIGHT.
7. CENTER SECTION SELECTOR VALVE OPEN INDICATOR LIGHT.
8. CENTER SECTION SELECTOR VALVE CONTROL SWITCH.

EFFECTIVITY	
APPLICABLE TO AIRPLANES	
*	CV1 THROUGH CV25
**	CV26 AND ON

30.02.28.040A

Rear Spar Under-Wing Refuel Control Panels
Figure 10-4

all of the fuel tank systems may be accomplished by using the crossfeed valves. It is not necessary to close the corresponding emergency fuel shutoff valves in the pylons when defueling, since the positive displacement engine-driven fuel pumps will stop air leakage. Fuel is transferred from the desired tank through the defueling valve and into the fuel tanker or suitable receptacle. (See Figure 10-5.)

ENGINE FUEL SUPPLY SYSTEM

Fuel from the outboard replenishment and outboard and inboard main tanks is transferred by bulkhead check valves and transfer pump to the main tank wells. From the main tank wells, dual booster pumps supply fuel to each of the four engine-driven fuel pumps (Figure 10-6). Hydraulically operated boost/jettison pumps supply fuel from the center section tanks to the cross feed lines. ASB fuel is transferred by electric transfer pumps into their respective main tanks.

Fuel Transfer Pumps

Two fuel transfer pumps are provided for each fuel tank and each outboard ASB. Each inboard ASB has one fuel transfer pump. For the No. 1 and No. 4 tank systems (outboard), both pumps are located in the main tank; one in the aft outboard corner and one in the forward inboard corner. For the No. 2 and No. 3 tank systems (inboard), one pump is located in the aft inboard corner of the replenishment tank and one pump in the forward inboard corner of the main tank. Fuel transfer pumps are used to transfer fuel from their respective areas into the main tank wells.

Replenishment Tank Fuel Level Indication

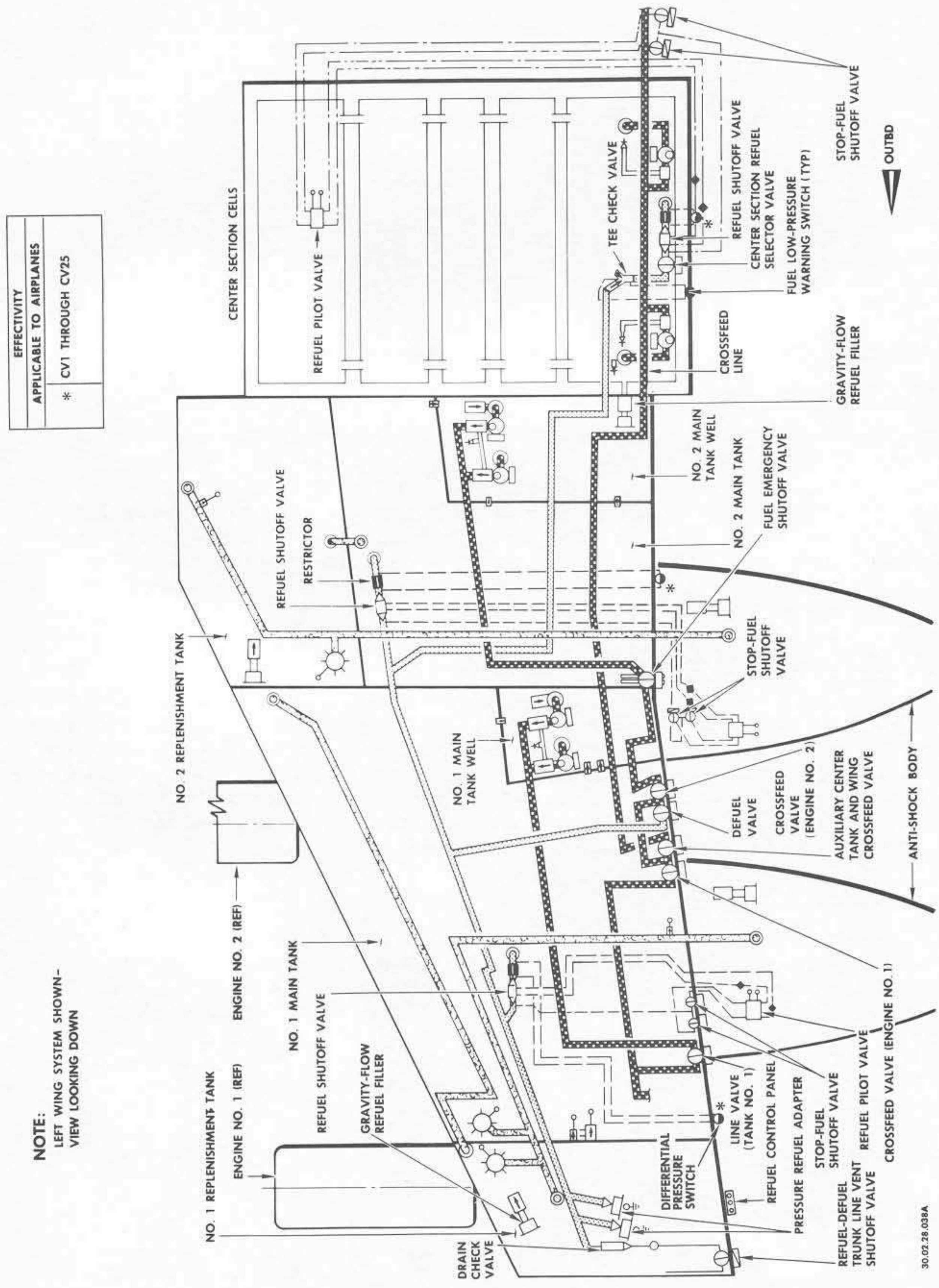
Low fuel level in the inboard replenishment tanks is indicated by the illumination of the pump low pressure warning light for the transfer pump of each inboard system. Illumination of the light indicates that the pump discharge pressure is low, either from lack of fuel or pump malfunction. Float switches in series with the pump low pressure warning light switches of all main tank transfer pumps prevent the corresponding warning light from illuminating unless the pressure loss is due to pump malfunction.

Fuel Boost Pumps

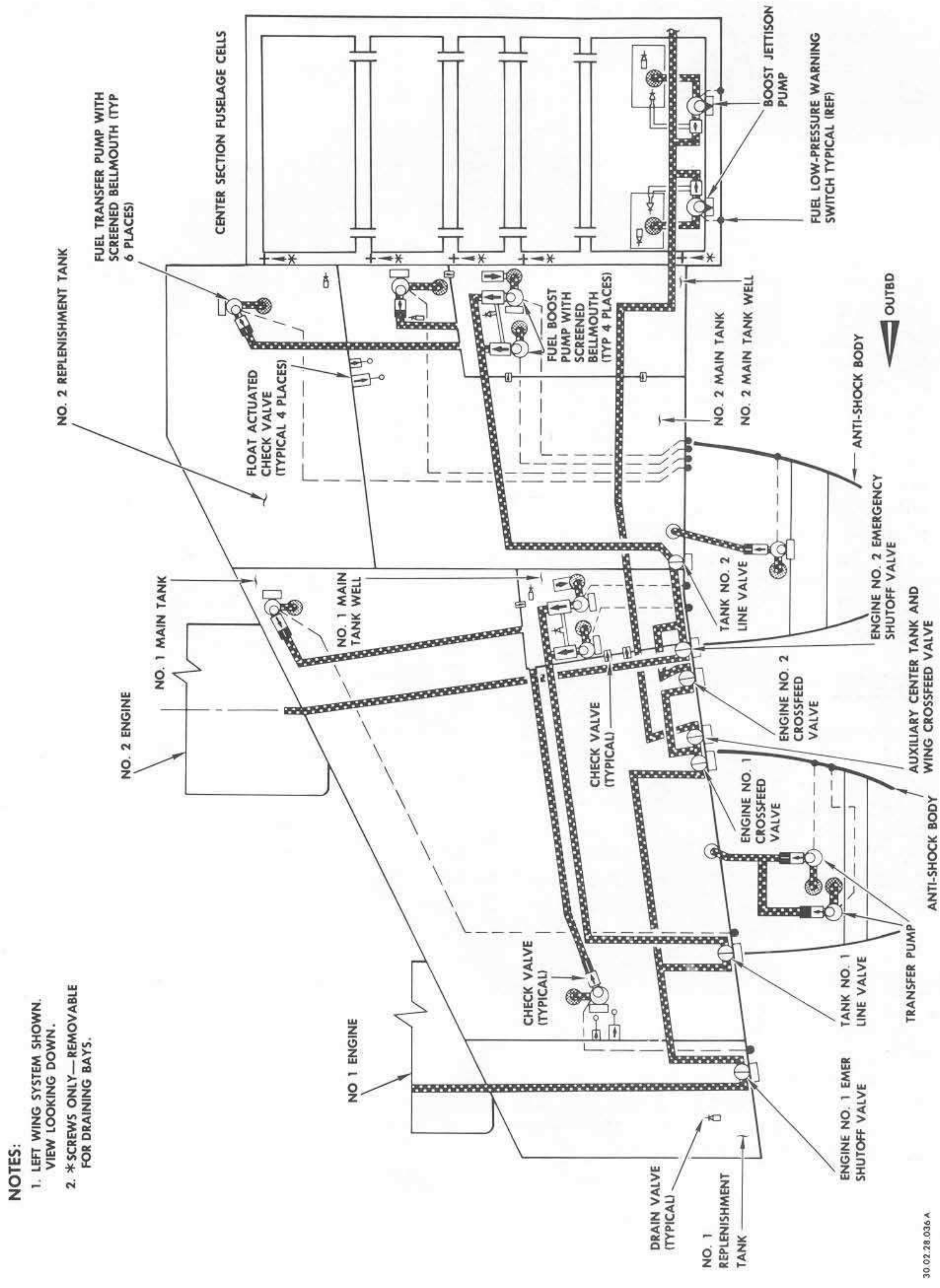
Each engine is normally supplied with fuel from two electric boost pumps located in each main tank well and the two hydraulic boost/jettison pumps in the center section tank. The pumps maintain between 15 psia (minimum) and 50 psig (maximum) fuel pressure at each engine fuel pump inlet during all airplane operating conditions. Both, or either pump alone, are capable of supplying a maximum fuel flow rate of 125 percent of takeoff requirements (standard day fuel consumption). Any pair of boost pumps is capable of supplying 100 percent maximum takeoff fuel flow to its own engine and to one engine in the opposite wing. Any pump pair is capable of supplying 100 percent of maximum cruise fuel flow to all four engines.

Emergency Fuel Shutoff Valves

For each engine an emergency fuel shutoff valve is installed on the rear spar. The valves are operated in connection with the fire protection system. Their



Refuel-Defueling System Schematic
Figure 10-5



Fuel Supply System Schematic
Figure 10-6

primary function is to close off fuel to the engines in event of a fire. Operating the FIRE PULL "T" handle of the fire protection system causes the corresponding emergency fuel shutoff valve to close.

Engine Fuel and Control System Block Diagram

Figure 10-7, Engine Fuel Supply Block Diagram, provides a ready reference as to engine fuel flow from the emergency fuel shutoff valve to the fuel nozzles.

Engine Fuel Pump

A primary boost element in the engine fuel pump increases the fuel tank boost pump pressure by 15 to 45 psi. The fuel flows from the primary boost element to two high pressure elements, increasing the fuel pressure 100 to 850 psig depending on engine speed and flight conditions. The pump discharges the fuel to the fuel heater. Fuel flows through the heater and into the fuel filter. From the filter the fuel goes into the fuel control, to the engine fuel-oil cooler, CSD fuel-oil cooler, the pressurization and drain valve, and into the fuel nozzles. The fuel pump always delivers more fuel to the control than is required to sustain the engine under any condition. The control uses some of the excess fuel to operate internal servo mechanisms and to position and cool the engine variable stator actuators. The remainder is returned to the inlet side of the high pressure elements of the fuel pump. In case of failure of both fuel tank boost pumps in any one tank, the engine-driven fuel pump supplies sufficient fuel to the engine to maintain all performance requirements up to 30,000 feet. Fuel temperature at the engine must be approximately 43 degrees C (110 degrees F).

Engine Fuel Heater

The engine fuel heater is a counter-flow air-to-liquid type heat exchanger. Fuel enters through the fuel heater inlet and flows through the fuel tubes. Seventeenth stage compressor air at approximately 63 degrees C (145 degrees F) flows around the fuel tubes warming the fuel. The fuel flows past a temperature sensor and is discharged to the fuel filter. Expansion or contraction of the thermostatic element in the fuel temperature sensor opens or closes the fuel heater compressor air shutoff valve. The valve closes at fuel discharge temperatures above 7 degrees C (45 degrees F) and opens at fuel discharge temperatures below 0 degrees C (32 degrees F).

Engine Fuel Filter

The fuel filter, located between the fuel heater and the fuel control, removes solid contaminants from the fuel. Fuel enters the filter and surrounds the filter screen, passes through the screen into an inner chamber, and flows out the discharge port. If the filter should become clogged, fuel will bypass through the relief valve which opens when the pressure differential across the filter is 20 to 24 psi.

To Be Furnished

Engine Fuel Supply Block Diagram
Figure 10-7

Engine Fuel Control

The engine fuel control is a hydro-mechanical system. High turbojet engine performance calls for rapid engine acceleration, but a conventional seventeen stage fixed compressor would tend to stall during throttle bursts. The problem is caused by critical combinations of air, temperature and engine speed which create air turbulence and consequently restrict the flow of air through the compressor during acceleration. When this condition develops, the turbojet engine usually stalls or fails in rapid acceleration. This difficulty is overcome in the CJ805-23 engine by making the twenty inlet guide vanes and the first six stages of compressor stator blades variable. As some point is reached during acceleration (depending on compressor inlet temperature), the variable stators rotate slightly to maintain an angle with the inlet air that encourages a smooth airflow and at the same time discourages air turbulence. The fuel control unit is instrumental in positioning the inlet guide vanes and the stator blades. Thus the fuel control system can be considered made up of two subsystems: the variable stator subsystem which is concerned with the continuous and automatic adjustment of the airflow path, and the fuel subsystem which is concerned with fuel metering and delivery to the combustion chambers.

Specific discussion of the complex engine fuel control system can be found in the appropriate engine manual. However, briefly, the fuel control system includes components which sense compressor inlet temperature (CIT), compressor discharge pressure (CDP), engine rpm, fuel temperature, specific density of the fuel being used, and power lever signals. During flight, as the ambient temperatures and pressure change, the fuel control system will automatically vary both fuel flow and compressor stator angle in order to operate the engine efficiently at the rpm selected by the pilot through setting of the power control lever.

Fuel Shutoff Levers

The fuel shutoff lever actuates a fuel shutoff valve in the discharge portion of the fuel control unit. The valve has two positions; RUN (open position), and OFF (closed position). Metered fuel from the fuel control flows through the fuel shutoff valve to the engine fuel-oil cooler. The engine must never be started or operated with the fuel shutoff lever in an intermediate position. Use as a fuel throttling means will damage the fuel control unit.

Engine Fuel-Oil Cooler

The engine fuel-oil cooler is a liquid-to-liquid heat exchanger using engine fuel as the coolant. (See Section 11, OIL SYSTEM for detailed information). From the engine fuel-oil cooler the fuel is directed through the fuel flow transmitter to the constant speed drive (CSD) fuel-oil cooler.

Fuel Flow Transmitter

The fuel flow transmitter is a propeller-driven synchro motor that is actuated by varying fuel flow against the propeller. The transmitter signal is displayed on the pilots' engine instrument panel in pounds per hour of fuel flow.

Constant Speed Drive Fuel-Oil Cooler

The constant speed drive fuel-oil cooler is a liquid-to-liquid type heat exchanger using engine fuel as the coolant. From the constant speed drive fuel-oil cooler, metered fuel is routed to the pressurizing and drain valve.

Fuel Pressurizing and Drain Valve

Metered fuel flows to the inlet of the fuel pressurizing and drain valve. When the inlet pressure rises to 80 to 100 psi above the reference pressure plus the spring force that is applied to the aft side of the pressurizing piston, the drain piston closes the drain port. The pressure also moves the pressurizing piston away from the drain piston, allowing fuel to flow to the engine. When the pressure differential across the piston drops below 80 psi, fuel flow to the engine is cut off and the drain port is opened to drain the fuel manifold.

The total drainage after engine shutdown or a wet prestart is 450 to 550 cc. No leakage should occur during other phases of engine operation.

Engine Fuel Nozzles

Metered fuel from the pressurizing and drain valve flows into the ten fuel nozzles which spray atomized fuel into the combustion chambers.

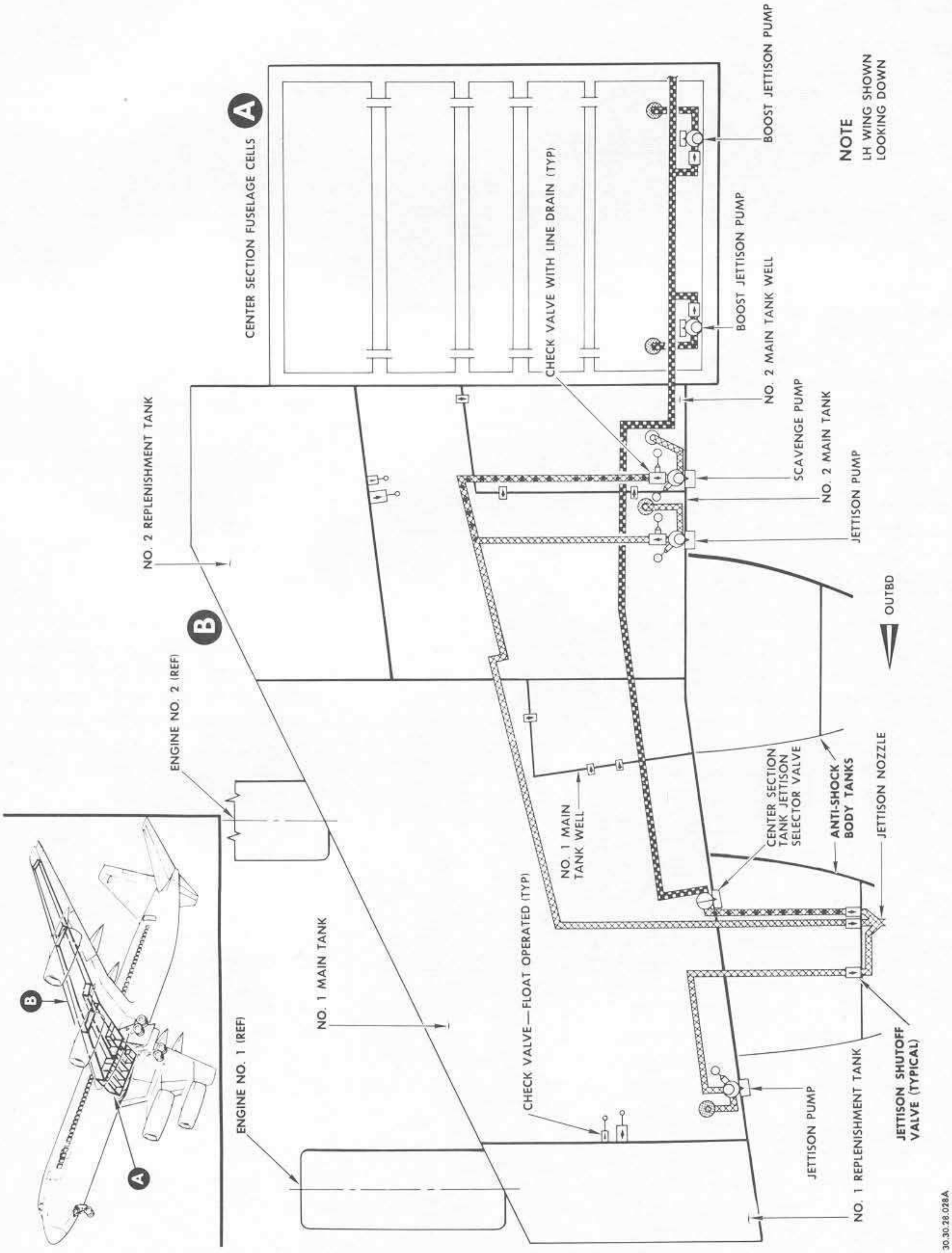
FUEL JETTISON SYSTEM

The fuel jettison system is designed for fuel jettisoning and for fuel scavenging. The fuel jettison system permits in-flight reduction of the airplane weight to the maximum permissible landing weight. The scavenge portion of the system completely empties the inboard tanks and the center section tanks, adjacent to the cabin area, in the event that an emergency wheels up landing is anticipated. A schematic of the system is shown in Figure 10-8. The jettison/scavenge system is a fixed installation useable at any airplane configuration of flaps or landing gear and at speeds up to V_{NO}/M_{NO}.

A hydraulically operated fuel jettison pump is provided in each of the four main fuel tanks. The jettison pumps are controlled from switches located on the fuel control panel. Moving the switches to the OPEN position energizes solenoid valves which port No. 2 hydraulic system fluid to the jettison pump drive motors. The LH center section boost/jettison pumps utilizes hydraulic power from No. 1 hydraulic system; the RH center section boost/jettison pump utilizes hydraulic power from the No. 2 hydraulic system. Each fuel tank jettison system is equipped with inlet bellmouth fittings positioned to assure a minimum of 8810 pounds total fuel remaining in the tanks after jettisoning. The jettison system is capable of dumping overboard a minimum of 49,800 pounds of fuel in 27 minutes, or an average of 1845 pounds per minute.

Scavenge System

A hydraulically operated fuel scavenge pump is located in each inboard main tank well. Two hydraulically operated combination boost, jettison, and scavenge pumps are located in the center section tank. All four pumps are controlled by switches located on the flight engineer's fuel control panel. The



Fuel Jettison System Schematic
Figure 10-8

center section pumps are used to jettison or scavenge fuel when the jettison valves are opened and the center tank and wing crossfeed valves are closed. Fuel is then diverted from the engines into the jettison system. Continued operation of the center section pumps results in fuel scavenging. The two scavenge pumps are controlled by switches located on the flight engineer's fuel control panel. These switches actuate solenoid valves which port No. 1 hydraulic system fluid to the scavenge pump motors. The scavenge pumps obtain fuel from the bottom of the wells through the downturned bellmouth inlets and scavenge the entire inboard system (tanks No. 2 and No. 3) of each wing. This leaves only the fuel remaining below the bellmouths in the outboard systems (tanks No. 1 and No. 4) for operation during emergency wheels up landing procedures.

Scavenge/Jettison Systems Hydraulic Power Source

The main tank fuel jettison system is operated by hydraulic power from the No. 2 hydraulic system. The main tank fuel scavenge system is operated by hydraulic power from the No. 1 hydraulic system. In the center section tank, the left boost/jettison pump is powered by the No. 1 hydraulic system, the right pump is powered by the No. 2 hydraulic system.

Fuel Jettison Nozzles

A fuel jettison nozzle is located in the aft apex of each outboard anti-shock body. Each nozzle is fed by two fuel jettison lines and one scavenge line. The scavenge fuel line is teed into the jettison line for the inboard tank. The nozzles supply a proper exit stream pattern and establish a minimum exit velocity of 30 feet per second to prevent reverse fuel flow in the wing boundary layer and to retard flame propagation. The nozzles are removable and streamlined to the apex of the anti-shock body tail cone.

FUEL TANK VENT SYSTEM

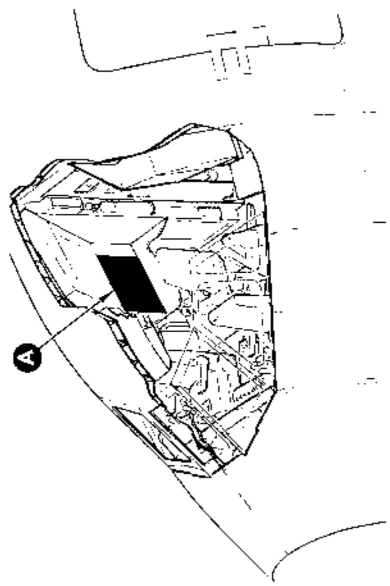
Three vent valves are used in each tank vent system. One float-type valve is located in each main tank, each replenishment tank, the center section tank, and each inboard anti-shock body, and one combination float-type and pressure relief valve is located in each replenishment tank and the center section tank. The tank venting systems connect with the ram-vent scoop in the lower surface of each wing tip.

Ram-Vent Scoop

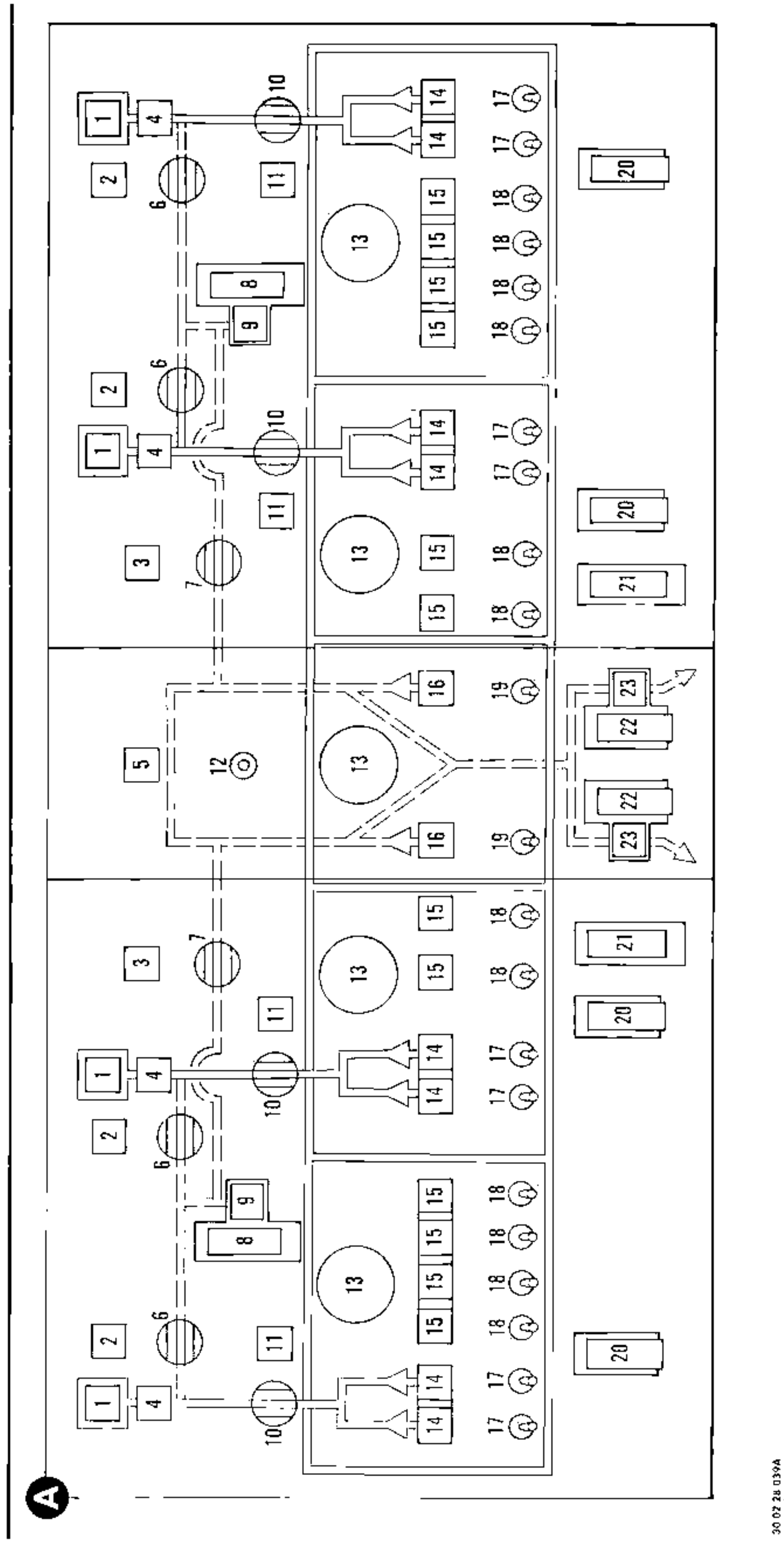
One ram-vent scoop is fitted flush with the inboard surface of each anti-shock body and serves two purposes. Its purpose is to vent vapor pressure from the tanks; in addition, the scoop has a plenum chamber which scoops air during flight to the extent that approximately two to three pounds of air pressure above ambient is maintained in the tank systems.

FUEL CONTROL PANEL

All functions of the fuel tank systems except refueling are controlled from the flight engineer's fuel control panel (see Figure 10-9). The fuel control panel contains fuel quantity indicators, valve position indicators, fuel pressure



1. ENGINE FUEL PUMP LOW PRESSURE WARNING LIGHT.
2. CROSSFEED VALVE IN TRANSIT LIGHT.
3. AUX CTR TANK & WING CROSSFEED VALVE IN TRANSIT LIGHT
4. EMERGENCY FUEL SHUTOFF VALVE CLOSED LIGHT.
5. OUTBOARD ANTI-SHOCK BODY EXCESS FUEL LIGHT.
6. CROSSFEED VALVE CONTROL SWITCH.
7. AUX CTR TANK & WING CROSSFEED VALVE CONTROL SWITCH
8. DEFUEL VALVE CONTROL SWITCH.
9. DEFUEL VALVE OPEN LIGHT.
10. LINE VALVE CONTROL SWITCH.
11. LINE VALVE IN-TRANSIT LIGHT.
12. FUEL QUANTITY SELECTOR SWITCH.
13. FUEL QUANTITY INDICATOR.
14. BOOST PUMP LOW PRESSURE WARNING LIGHT.
15. TRANSFER PUMP LOW PRESSURE WARNING LIGHT.
16. BOOST-JETTISON PUMP LOW PRESSURE WARNING LIGHT.
17. BOOST PUMP CONTROL SWITCH.
18. TRANSFER PUMP CONTROL SWITCH.
19. BOOST-JETTISON PUMP CONTROL VALVE SWITCH.
20. JETTISON PUMP CONTROL VALVE SWITCH.
21. JETTISON-SCAVENGE PUMP CONTROL VALVE SWITCH.
22. CENTER TANK JETTISON VALVE CONTROL SWITCH.
23. CENTER TANK JETTISON VALVE OPEN LIGHT.



Flight Engineer's Fuel Control Panel
Figure 10-9

warning lights, boost and transfer pump switches, and normal feed, crossfeed, defuel, scavenge and jettison valve switches. A single fuel temperature indicator is located on a separate panel adjacent to fuel control panel.

The arrangement of controls, indicators and warning lights permits an immediate interpretation of the fuel feed condition being used, as well as the immediate selection of any other configuration desired. A schematic diagram of the fuel line circuit is superimposed on the panel and is shown on an internally illuminated diagram. The fuel valve switch knobs are placed across the circuit to represent the corresponding valve location in the actual fuel system. The switch knobs incorporate an engraved solid line across their circumference. By aligning the line on a knob with the fuel circuit line, the corresponding valve is opened; when the knob reference line is set at right angles to the fuel line on the panel, the valve is closed.

Fuel Quantity Gaging System

The fuel quantity gaging system indicates fuel quantity on a weight basis for all five of the tank systems. Five indicators are located on the flight engineer's fuel control panel, one for each tank system. A switch, spring loaded to TOTAL, selects TOTAL, MAIN, REPLENISH, or ANTI-SHOCK BODY, for reading from the No. 1, 2, 3, and 4 tank indicators; the center section tank indicator reads only TOTAL fuel. A single indicator (totalizer) on the pilots' center instrument panel shows the combined airplane total fuel remaining. Repeater indicators located on the rear spar underwing refuel control panels, also reflect quantity of fuel indicated by the corresponding tank indicator on the flight engineer's fuel control panel. Twenty one liquid sight gages (drip sticks) are provided for manual checking of fuel quantity.

Pump Low Pressure Warning Lights

An amber PUMP LOW PRESS indicator light is provided for each of the twelve fuel transfer pumps and ten red PUMP LOW PRESS warning lights for the fuel boost pumps. The No. 2 main transfer pump and the No. 2 replenish tank transfer pump have the same warning light. The No. 3 main transfer pump and the No. 3 replenish tank transfer pump have the same warning light. Six amber PUMP LOW PRESS indicator lights are provided for the six anti-shock body transfer pumps.

The ten boost pump warning lights illuminate when the fuel pressure drops due to lack of fuel or malfunction of the pump.

Except for No. 2 and No. 3 replenishment tank transfer pumps, all transfer pump lights illuminate only when the pump malfunctions. Float switches prevent their illumination for low fuel conditions. The No. 2 and No. 3 replenishment tank transfer pump lights will illuminate for low fuel conditions as well as pump malfunctioning. The center section low pressure light will illuminate for a low fuel condition.

Engine Fuel Pump Warning Lights

One red PUMP LOW PRESS warning light is provided for each engine driven fuel pump. Illumination indicates either low fuel condition or pump malfunction.

Emergency Fuel Shutoff Indicator Lights

One green EMER VALVE CLOSED indicator light is provided for each engine. Illumination of this light indicates the emergency fuel shutoff valve has been closed by operation of a FIRE-PULL "T" handle.

Valve In Transit Indicator Lights

Each of the fuel crossfeed and line valves is provided with a blue VALVE IN TRANSIT indicator light. The valve in transit light illuminates only during the time the valve is in motion and extinguishes when movement is completed.

De-Fuel Valve Indicator Lights

A red VALVE OPEN warning light is provided for each de-fuel valve. Each light illuminates when the associated de-fuel valve is open.

Outboard Anti-Shock Body Warning Light

One amber EXCESS FUEL light is illuminated continuously if either outboard ASB has a "fuel remaining" level of more than 40 US gallons. There are two separate ASB fuel level and speed warning circuits; system No. 1 utilizes the alarm bell portion of the VNE and MNE warning circuit, system No. 2 is completely independent with its own warning bell. System No. 1 is actuated when the airplane exceeds .69 mach or 347 knots with any fuel in either outboard ASB. When actuated, the No. 1 system causes the VNE/MNE warning bell to ring. System No. 2 is actuated whenever the airplane exceeds .69 mach or 347 knots and more than 40 US gallons of fuel remains in either outboard ASB. System No. 2 actuates an independent warning bell. System No. 2 also causes the EXCESS FUEL light to remain illuminated if either outboard ASB has a fuel level above 40 US gallons.

De-Fuel Valve Switches

Two de-fuel valve guarded switches are provided and are safetied in the CLOSED position for normal operation. Moving the switches to the OPEN position opens the two de-fuel valves.

Line Valve Switches

Turning the line valve switches to align the knob marker with the circuit diagram fuel line opens the line valve. Turning the knob marker to a position across the circuit diagram fuel line closes the line valve.

Crossfeed Valve Switches

Turning the crossfeed valve switches to align the knob marker with the circuit diagram fuel line opens the crossfeed valve. Turning the knob marker to a position across the circuit diagram fuel line closes the crossfeed valve.

Boost Pump Control Switches

The boost pumps are started by ten boost pump control switches on the fuel control panel. Each switch incorporates an OFF and ON position. The two center section switches also are used to jettison center section fuel when the jettison valves are opened.

Tank No. 1 and Tank No. 4 Transfer Pump Switches

Two transfer pump switches are provided for tank No. 1 and two transfer pump switches for tank No. 4 Moving a switch from OFF to the "on" position starts the corresponding transfer pumps. Two transfer pump switches are provided for each outboard anti-shock body. Activating the switches to ON transfers fuel to the No. 1 and No. 4 main tanks.

Tank No. 2 and Tank No. 3 Transfer Pump Switches

Tank No. 2 and Tank No. 3 are each provided with a single three-position transfer pump switch. The center position is the OFF position. Movement to the MAIN position operates the transfer pump located in the main tank. Movement to REPLENISH position operates the transfer pump located in the replenishment tank. One transfer pump switch is provided for each inboard anti-shock body. Actuating the switch to ON transfers fuel to the No. 2 and No. 3 main tanks.

Jettison Switches

A tank jettison guarded switch is provided for each main tank and each switch is safetied in the CLOSE position. Movement of the switch to the OPEN position opens a solenoid valve and ports hydraulic power system fluid to the appropriate jettison pump.

Scavenge Switches

Two scavenge guarded switches are provided, one for tank No. 2 and one for tank No. 3. Both switches are safetied in the CLOSE position. Movement of the switches to the OPEN position opens a solenoid valve and ports No. 2 hydraulic power system fluid to the appropriate scavenge pump.

Center Section Jettison Valve Selector Switches

The center tank jettison valves are controlled by individual switches located on the lower portion of the flight engineer's fuel control panel. Placing the switch in the OPEN position will open the valve and illuminate a red VALVE OPEN warning light adjacent to each switch. When the valves are opened for jettisoning, the center tank and wing crossfeed valves must be closed.

FUEL TEMPERATURE INDICATOR

The fuel temperature gage is a subpanel of the flight engineer's control panel. It indicates fuel temperature in degrees Centigrade as read by any one of five temperature sensitive elements. One element is located in the No. 1 tank well and the other four are located, one to each engine, in the fuel stream near the engine-driven fuel pump. A five position selector switch is provided on the right side of the fuel control panel.

REAR SPAR UNDER-WING REFUELING CONTROL PANEL

The refueling control panel is located on the under-wing surface at the rear spar and outboard of No. 1 and 4 pylons. It contains fuel quantity gages, preset quantity selectors, and precheck shutoff controls. Jacks for microphone, handsets and headsets and installed for direct communication with the flight compartment only during defueling operations. During pressure refueling operations, personnel are not required in the flight compartment and the entire operation is conducted from the refueling control panel. It is impossible to replace the cover on the refueling control panel unless all controls, including removal of defueling communication equipment, are in proper position for flight (see Figure 10-4).

Fuel Quantity Indicators

Three fuel quantity gages are provided for each wing refueling control panel, one gage for the outboard tank system, one gage for the inboard tank system, and one for the center section tank. The inner pointer of each gage displays the total quantity of fuel in its associated tank. The outboard pointer of each gage is capable of being set to the desired refueling quantity by a knob at the lower left corner of each indicator. When the two pointers coincide, the refueling flow will be stopped.

CAUTION: WHEN THE PRESET FUEL QUANTITY HAS BEEN DELIVERED TO THE FUEL TANKS AND AUTOMATIC SHUTOFF HAS BEEN ACCOMPLISHED, THE FUEL SUPPLY SOURCE PRESSURE SHOULD BE RELEASED IMMEDIATELY.

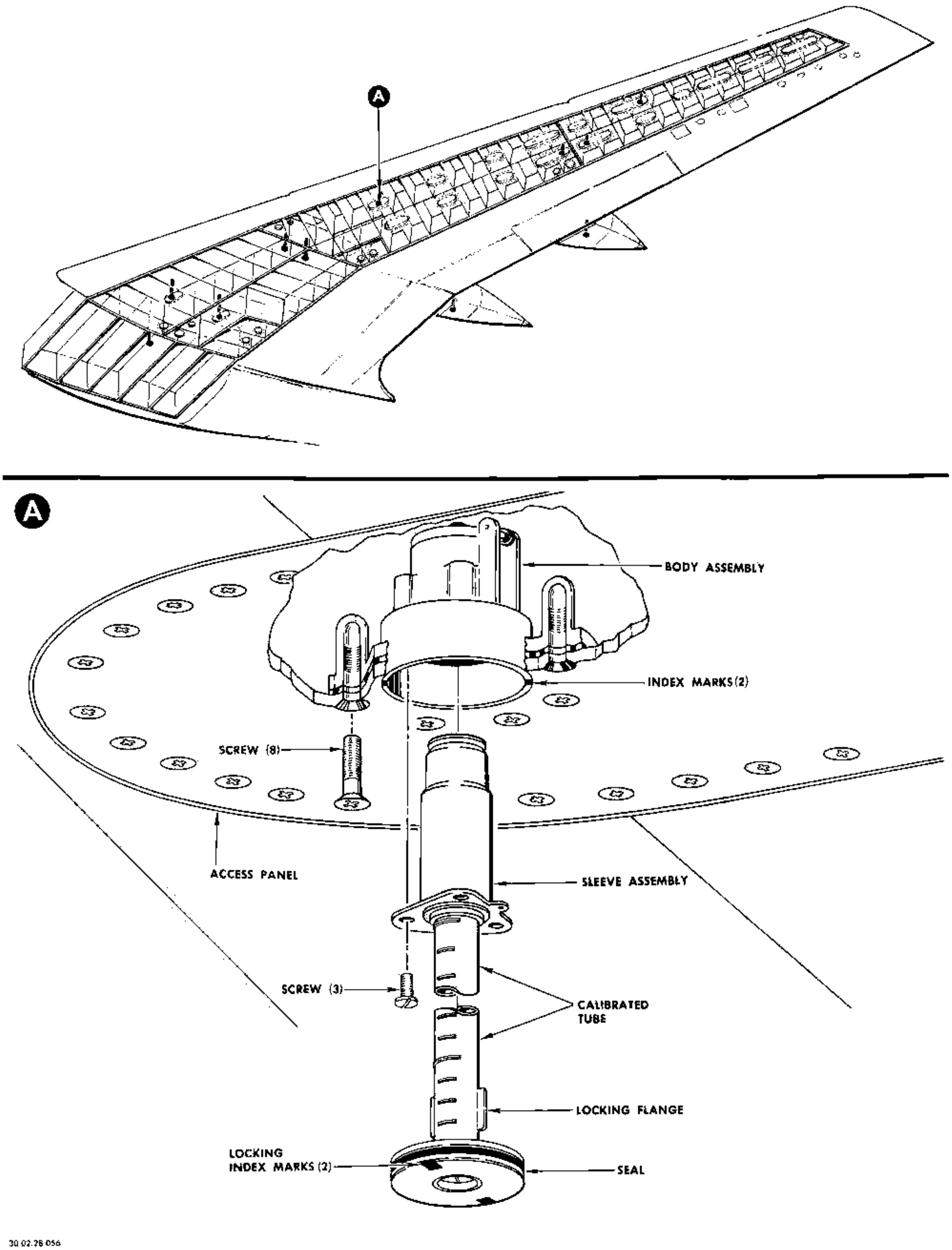
Precheck Switches

A three-position precheck switch is provided to test the automatic refueling system for each tank. Moving the switch to the STOP "PT" position will open the primary solenoid valve and port pressure fuel to the primary side of the dual pilot-level control valve, resulting in shutoff of fuel flow. Moving the switch to the STOP "ST" position will open the secondary solenoid valve and port pressure fuel to the secondary side of the dual pilot-level control valve, resulting in shutoff of fuel flow. Setting the switch in AUTO position allows the "stop" signal, determined by setting of the refuel-quantity pointer, to be applied to the primary and secondary solenoid valves after the selected amount of fuel has been delivered to the tank.

There is one refuel pilot valve in each tank, and one in the forward cell of the center section tank.

LIQUID SIGHT GAGES

A total of 21 liquid-sight gages are provided. The gages are installed in housings attached to the bottom of the tanks and extend up through the tanks, fitting into gage supports attached to the top of the tanks (see Figure 10-10). Each liquid sight gage consists of a calibrated tube, or drip stick, which is



Liquid Sight Gage - Typical
Figure 10-10

free to slide up and down within a sleeve attached to the gage housing. The bottom of the tube is attached to a butt plate which fits into a recess in the under surface of the wing.

The gage is unlocked by pushing the butt plate upward and turning it to the left with a special tool. The tube, with the tool attached, is lowered until fuel drips from the tube indicating that the top of the tube is below the fuel level in the tank. The tube is then raised until the dripping ceases and the quantity of fuel in the tank is read in pounds at the highest visible calibration on the tube. A trap attached to the tool catches the minor fuel leakage when the tube is lowered below the surface of the fuel.

The gage is closed by pushing the butt plate up into the recess in the wing surface and turning it to the right until the index marks on the butt plate and the collar are aligned. Because of the height of the sleeve and housing; the fuel level cannot be measured below 2.9 inches above the bottom of the tank.

The tube and sleeve assembly can be removed for replacement without draining the tank.

FUEL SYSTEM ELECTRICAL SOURCES

The control relay coils for each boost and transfer pump are energized through the individual switches from the following buses: (For detailed information, consult the WIRING DIAGRAM MANUAL.)

1. From the 115-volt, 400 cps, ac nonessential bus:
   No. 1 tank outboard boost pump
   No. 2 tank outboard boost pump
   No. 3 tank outboard boost pump
   No. 4 tank outboard boost pump
   No. 1 main tank inboard transfer pump
   No. 2 main tank transfer pump
   No. 3 main tank transfer pump
   No. 4 main tank inboard transfer pump
   No. 1 ASB (outboard) transfer pump
   No. 2 ASB transfer pump
   No. 3 ASB transfer pump
   No. 4 ASB (outboard) transfer pump

2. From the 115-volt, 400 cps, ac essential bus:
   No. 1 tank inboard boost pump
   No. 2 tank inboard boost pump
   No. 3 tank inboard boost pump
   No. 4 tank inboard boost pump
   No. 1 main tank outboard transfer pump
   No. 2 replenish tank transfer pump
   No. 3 replenish tank transfer pump
   No. 4 main tank outboard transfer pump
   No. 1 ASB (inboard) transfer pump
   No. 4 ASB (inboard) transfer pump

Boost and Transfer Pump Operating Power Sources

All main tank boost and transfer pumps are operated by 200/115-volt, 3-phase, 400 cps ac electrical power. Each phase to each pump is individually protected by circuit breakers. The center section tank boost/jettison pumps are hydraulically operated.

Defueling Valve Power Source

Both defueling valves are operated by 28-volt dc. If required, the airplane emergency battery dc power can be used.

Fuel Tank Shutoff Valve Power Source

All fuel tank shutoff valves are operated by 28-volt dc. If required, the emergency battery dc power can be used.

Crossfeed Valve Power Source

All crossfeed valves are operated from the 28-volt dc emergency bus. If required, emergency battery power can be used.

Emergency Fuel Shutoff Valve Power Source

All emergency fuel shutoff valves, operated by the fire-pull "T" handles, are operated from the 28-volt dc emergency bus. If required, the emergency battery power can be used.

Fuel Jettison System Power Source

All fuel jettison system pump solenoid valves are operated by 28-volt dc. Actual operation of the jettison pumps is by hydraulic fluid ported to the pumps by opening of the solenoid valves.

Fuel Quantity Gaging System Power Source

Power for the fuel quantity selector switch system is by 28-volt dc. Power for operation of the probe-indicating systems is 115-volt, 400 cps.

Fuel Pump Low Pressure Warning Light Power Source

The engine fuel pump low pressure warning lights obtain power from a 28-volt dc bus.

The fuel tank booster and transfer pump low pressure warning lights obtain power from a 28-volt dc bus.

Fuel Temperature Indicator Power Source

The fuel temperature indicator obtains power from a 28-volt dc essential bus.

Rear Spar Under-Wing Refueling System Power Source

The refueling system obtains power from a 28-volt dc bus.

Fuel Flow Indicator Power Source

The fuel flow indicators receive power from the pilot's essential ac bus, and the 28-volt dc emergency bus.

11 OIL SYSTEM

Section 11

OIL SYSTEM

TABLE OF CONTENTS

TITLE	PAGE
GENERAL OIL SYSTEM	11-1
Oil System Reservoir	11-1
Oil System Replenishment	11-1
Oil System Consumption Rate	11-1
ENGINE OIL SYSTEM	11-1
Engine Oil Scavenge System	11-4
Engine Fuel-Oil Cooler	11-4
CONSTANT SPEED DRIVE (CSD) OIL SYSTEM	11-4
CSD Scavenge Pump	11-4
CSD Oil Coolers	11-5
OIL SYSTEM PRESSURIZATION	11-5
Altitude Sensing and Pressurization	11-5
OIL SYSTEM DESIGN LIMITS	11-5
Oil System Flight Attitude Limits	11-6
THRUST REVERSER OIL SYSTEM	11-6
Thrust Reverser Indication System	11-6
OIL SYSTEM INDICATING ELEMENTS	11-6
Engine Oil Quantity Gages	11-6
CSD Oil Quantity Determination	11-7
Engine Oil Temperature Gages	11-7
Engine Oil Pressure Gages	11-7
Engine Oil Pressure Warning Lights	11-7
CSD Oil Temperature Warning Lights	11-7
OIL SYSTEM ELECTRICAL SOURCES	11-7

OIL SYSTEM

GENERAL OIL SYSTEM

The oil system is composed of three subsystems; the engine system, the constant speed drive (CSD) hydraulic system and the thrust reverser actuation system. The engine oil system is used to cool and lubricate the bearings and gears. With respect to the constant speed drive and the thrust reversers, the oil is utilized as a hydraulic fluid and lubricant for the CSD power drive and for hydraulic actuation purposes. (See Figure 11-1.)

Oil System Reservoir

The oil system reservoir is mounted on the upper forward right side of the engine (see Figure 11-2). The reservoir is of single unit construction with an internal vertical bulkhead that divides the interior into two separate reservoir sections. The two sections are connected by an internal pressure equalizing line in the uppermost portion of the expansion spaces. The larger and aft section of the reservoir holds 4.15 gallons of oil for the engine system. The forward section of the reservoir holds 1.72 gallons of oil for the constant speed drive and thrust reverser systems. There is no flow connection between the oil in the two reservoir sections or between the systems they supply.

Oil System Replenishment

Each section of the oil reservoir is provided with a gravity flow filler port at the top of the reservoir. A calibrated dipstick is provided as an integral portion of each gravity fill cap. Accidental overflow of oil is removed by a scupper drain opening between the two caps with the drain exit at the bottom of the reservoir.

When the engine oil section of the reservoir is filled to 4.15 gallon capacity, 1.26 gallons expansion space is provided. The CSD section, when filled to 1.72 gallon capacity, provides 0.50 gallons of expansion space.

Oil System Consumption Rate

With the oil reservoir filled to capacity, the engine has a flight capability of 13 hours. At the end of that period approximately 1.15 gallons of oil will be left in the engine section. With this quantity in the reservoir, the engine can be inclined 30 degrees up or down, or rolled to the right or left 20 degrees without uncovering the oil outlets.

ENGINE OIL SYSTEM

Oil from the engine section of the oil reservoir flows to the engine oil pump. The engine-driven positive displacement pump increases the oil pressure to 12 to 60 psig, depending on engine speed, and discharges the oil through a check valve to the engine oil filter. The filtered oil is then divided into flows supplying the front gearbox and the No. 1 bearing, the transfer and rear gear boxes, the horizontal drive shaft damper bearing, and the No.2, No.3, No.4 and No. 5 bearings. A pressure relief valve bypasses oil from the downstream side

TO BE FURNISHED

Oil System Schematic
Figure 11-1

TO BE FURNISHED

Oil System Components
Figure 11-2

of the oil filter to the inlet side of the oil pump in case of filter clogging. In this way, the maximum pressure delivered to the oil pressure indicating system is 200 psig. Actually, insofar as pump performance is concerned, during cold weather starting, engine oil pressure may range as high as 500 to 600 psig.

Engine Oil Scavenge System

The engine oil in the bearing sumps, damper bearing and accessory cases is scavenged, filtered, cooled, de-aerated and returned to the engine section of the oil reservoir by means of the three scavenge system oil pumps. The scavenge system pumping capacity is approximately 2.5 times the actual amount of oil supplied to any one area. The scavenge oil filter contains a magnetic chip detector which attracts any metal chips in the filter. An accumulation of chips will complete a metallic path which can be detected by use of an external continuity check. Thus the presence of metallic contamination can be determined without removing and inspecting the filter.

Engine Fuel-Oil Cooler

The engine fuel-oil cooler is a liquid-to-liquid heat exchanger which uses engine fuel as a coolant. When the scavenge oil temperature is below 38 degrees C (100 degrees F), a bypass valve is opened allowing a portion of the oil to flow directly to the cooler outlet without passing through the cooler. As the temperature increases, the valve closes, forcing all the oil to flow through the cooler core. A fuel temperature sensor opens the bypass valve when the fuel temperature reaches 116 degrees C (241 degrees F). Since certain engine operating conditions may demand greater fuel flow through the cooler than is possible through the normal core system, a bypass valve is provided that will open and allow large volumes of fuel to bypass the cooler.

CONSTANT SPEED DRIVE (CSD) OIL SYSTEM

The CSD oil system provides lubrication, control system hydraulic power and transmission hydraulic power. The oil from the CSD section of the reservoir is increased in pressure to 140 to 150 psig by the CSD internal oil pump. From the pump, the pressurized oil is directed through a filter unit, past a magnetic chip detector and into the control, lubrication and actuating portions of the CSD unit. A temperature sensing probe projects into the CSD sump area. The thermal switch and probe assembly will illuminate a DRIVE MALFUNCTION light on the flight engineer's panel when the oil temperature reaches or exceeds 177 degrees C (350 degrees F).

CSD Scavenge Pump

The CSD scavenge pump scavenges oil from the CSD case sump and returns it to the CSD internal reservoir. When the internal reservoir is full, a relief valve operating from 110 to 120 psi ports the excess oil to the CSD section of the main oil reservoir through the oil-to-air and oil-to-fuel coolers.

CSD Oil Coolers

CSD oil from the CSD internal reservoir is directed to the CSD oil-fuel cooler. However, the amount of heat the fuel can absorb at this point in the fuel system flow approaches zero under certain engine operating conditions. A fuel temperature sensor in the oil-fuel cooler opens the oil bypass valve when the fuel temperature reaches 116 degrees C (241 degrees F) and ports the oil directly to the CSD section of the oil reservoir.

OIL SYSTEM PRESSURIZATION

The sump pressurizing system regulates the pressure of the air in the oil tank, the gearboxes and bearing sumps. These units are pressurized primarily to assure a positive pressure at the inlet of the pressure and scavenge pumps. However, the method used to obtain air for pressurization is based upon the premise that airflow is from the rear side of the seals into the sump area. The results of this flow is a control of oil consumption--since, with air continuously flowing into the bearing sumps, only a small amount of oil is lost through the seals. By maintaining sump and tank pressure at 3 to 7 psi above ambient pressure, a balanced system is provided.

The air used for pressurization is obtained from two sources; leakage of air across the air and oil seals into the sumps and/or ambient air. The source of the air depends largely upon the efficiency of the seals; if sufficient air bleeds across them to prevent cavitation of the pumps, no ambient air is required. However, if the seals are exceptionally tight and virtually no air bleeds across them, ambient air must be drawn into the system.

Altitude Sensing and Pressurization

All of the scavenge pumps on the engine are capable of pumping a greater quantity of oil than is present in the sumps--the deficit is made up of air. As the scavenge oil-air mixture passes through the de-aerator in the oil tank, the entrained air is separated from the oil. The air, plus any excess air that may enter the tank from the sump vent line, pressurizes the oil tank. The only route of escape for the air in the tank is through the tank and sump pressurizing valves. The overboard vent of the sump pressurizing valve is full open up to 20,000 feet and pressure in the tank and sumps is established at 3 to 4 psi above ambient pressure by the action of the tank pressurizing valve. At 20,000 feet, the sump pressurizing valve begins to regulate to increase pressure in the system. At 28,000 feet, the valve attains full regulation and, added to the action of the tank pressurizing valve, establishes pressure in the tank and sumps at 5 to 7 psi above ambient pressure.

OIL SYSTEM DESIGN LIMITS

Normal limits shown below define a range that is encountered during normal operations. Extreme limits define a range that may be encountered less than 5 percent of the operating time.

1. Lubrication oil temperature into the engine
   Normal... 38 degrees C to 122 degrees C (100 degrees F to 250 degrees F)
   Extreme..-40 degrees C to 149 degrees C (-40 degrees F to 300 degrees F)

2. Lubrication oil temperature out of the engine
   Normal... 66 degrees C to 149 degrees C (150 degrees F to 300 degrees F)
   Extreme..-40 degrees C to 177 degrees C (-40 degrees F to 350 degrees F)

3. Lubrication oil pressure (Indicated)
   Normal.... 12 psig minimum at IDLE rpm
   35 to 65 psig at normal rated thrust
   Extreme (Cold Start)......200 psi (gauge limit)

Oil System Flight Attitude Limits

The oil system will perform satisfactorily during continuous engine operation in the following attitudes:

1. Level fore and aft, 20 degree roll attitude in either direction.
2. Dive or climb angle of 0 to 30 degrees, 10 degree roll attitude in either direction.

THRUST REVERSER OIL SYSTEM

The engine oil system provides hydraulic power to position the thrust reverser doors in either the stowed (forward thrust) or the extended (reverse thrust) position. It also provides the hydraulic power to latch (or unlatch) the reverser doors from the stowed position. Oil is gravity fed from the CSD portion of the engine oil tank to the engine mounted thrust reverser pump where it is increased to an operating pressure of 150 psi. From the pump it passes through a thrust reverser filter, a selector valve, interlock valve, metering valve and to the door latch and door actuators. Return oil is ported to the inlet side of the CSD fuel oil cooler.

Thrust Reverser Indication System

A blue valve IN TRANSIT and amber REVERSE THRUST light are provided for each engine reverse thrust system. These lights, located on the pilots' engine instrument panel, indicate the position of the "fail-safe" interlock valve.

OIL SYSTEM INDICATING ELEMENTS

Quantity, temperature and pressure gages are provided for the engine oil system. Warning lights are provided for both engine and CSD oil systems, and provisions have been made for preflight checking of the CSD oil quantity.

Engine Oil Quantity Gages

Four engine oil quantity gages, one for each engine, are located on the flight engineer's instrument panel. Each gage indicates the oil quantity in gallons in the engine section of the oil reservoirs. A dipstick, integral with the gravity fill cap on each engine oil section of the oil reservoir, is available for ground check on the oil quantity.

CSD Oil Quantity Determination

No provisions have been made for inflight measurement of the CSD section oil reservoir contents. A dipstick, integral with the gravity fill cap on each CSD section of the oil reservoir, is available for ground pre-flight check on CSD section oil quantity.

Engine Oil Temperature Gages

Four engine oil temperature gages, one for each engine, are provided on the flight engineer's instrument panel. Each gage indicates the oil temperature in degrees Centigrade. The temperature sensor probe is located between the engine oil scavenge filter and the engine oil-fuel cooler.

Engine Oil Pressure Gages

Four engine oil pressure gages, one for each engine, are provided on the flight engineer's instrument panel. Each gage indicates the oil pressure in psi units. The oil pressure takeoff point is immediately after the engine oil filter.

Engine Oil Pressure Warning Lights

Four engine oil pressure warning lights, one for each engine, are provided on the pilots' engine instrument panel. The oil pressure takeoff point is immediately after the engine oil filter and the pressure switches are set to predetermined critical pressure settings. The amber OIL PRESSURE LOW lights will illuminate when the critical low pressure setting is reached.

CSD Oil Temperature Warning Lights

Four warning lights, one for each CSD unit, are available for temperature indication purposes in connection with the CSD oil system. The red DRIVE MALFUNCTION lights are located in the electrical section of the flight engineer's instrument panel. The warning light will illuminate when a temperature sensitive probe in the CSD unit detects an oil temperature in excess of 177 degrees C (350 degrees F).

OIL SYSTEM ELECTRICAL SOURCES

The oil pressure gage system receives power from a 26-volt ac bus.

The oil temperature gage system receives power from a 28-volt dc bus.

The oil pressure warning lights receive power from a 28-volt dc bus.

The oil quantity gage system receives power from a 115-volt, 400 cps, ac bus.

The DRIVE MALFUNCTION warning lights receive power from a 28-volt dc bus.

For detailed information consult the WIRING DIAGRAM MANUAL.

12

HYDRAULIC SYSTEM

Section 12

HYDRAULIC SYSTEM

TABLE OF CONTENTS

TITLE	PAGE
HYDRAULIC POWER SYSTEMS	12-1
No. 1 HYDRAULIC POWER SYSTEM	12-1
No. 2 HYDRAULIC POWER SYSTEM	12-1
Reservoirs	12-1
Reservoir Filler Provisions	12-4
Hydraulic Reservoir Sight Gage	12-4
Pump Supply Line Boost Pumps	12-4
Engine-Driven Hydraulic Pumps	12-4
High and Low Pressure Filters	12-4
Pump Inlet Shutoff Valves	12-4
Pressure Line Accumulators	12-4
AUXILIARY HYDRAULIC POWER SYSTEM	12-6
Auxiliary Hydraulic Pump (Electric Driven)	12-6
Auxiliary Hydraulic Pump Filters	12-6
HYDRAULIC SYSTEM CONTROL AND INDICATING UNITS	12-6
System Low-Pressure Switches	12-6
Pump Supply Low-Pressure Switches	12-8
Remote Hydraulic Pressure Indicators	12-8
Temperature Transmitters and Indicators	12-8
Hydraulic Fluid Quantity Indicators	12-8
HYDRAULIC POWER SYSTEMS ELECTRICAL SOURCES	12-8

HYDRAULIC SYSTEM

HYDRAULIC POWER SYSTEMS

Hydraulic power is provided by two separate and independent systems, the No. 1 hydraulic power system and the No. 2 hydraulic power system. An electric hydraulic pump acts as an auxiliary hydraulic power system to operate both normal systems. The No. 1 and No. 2 systems are connected only at their reservoirs and through the auxiliary pump pressure supply line. The No. 1 system furnishes hydraulic power for nose landing gear operation, nose wheel steering, nose wheel brakes, slats and Kreuger flaps, power rudder, one-half the power for spoiler operation, the horizontal stabilizer trim system, fuel scavenge pumps and one center section tank boost/jettison pump. The No. 2 system furnishes hydraulic power for main landing gear operation, main gear wheel brakes, slats and Kreuger flaps, power rudder, one-half the power for spoiler operation, fuel jettison pumps and one jettison pump. The wing flaps can be supplied by either system, as selected by the pilots.

NO. 1 HYDRAULIC POWER SYSTEM

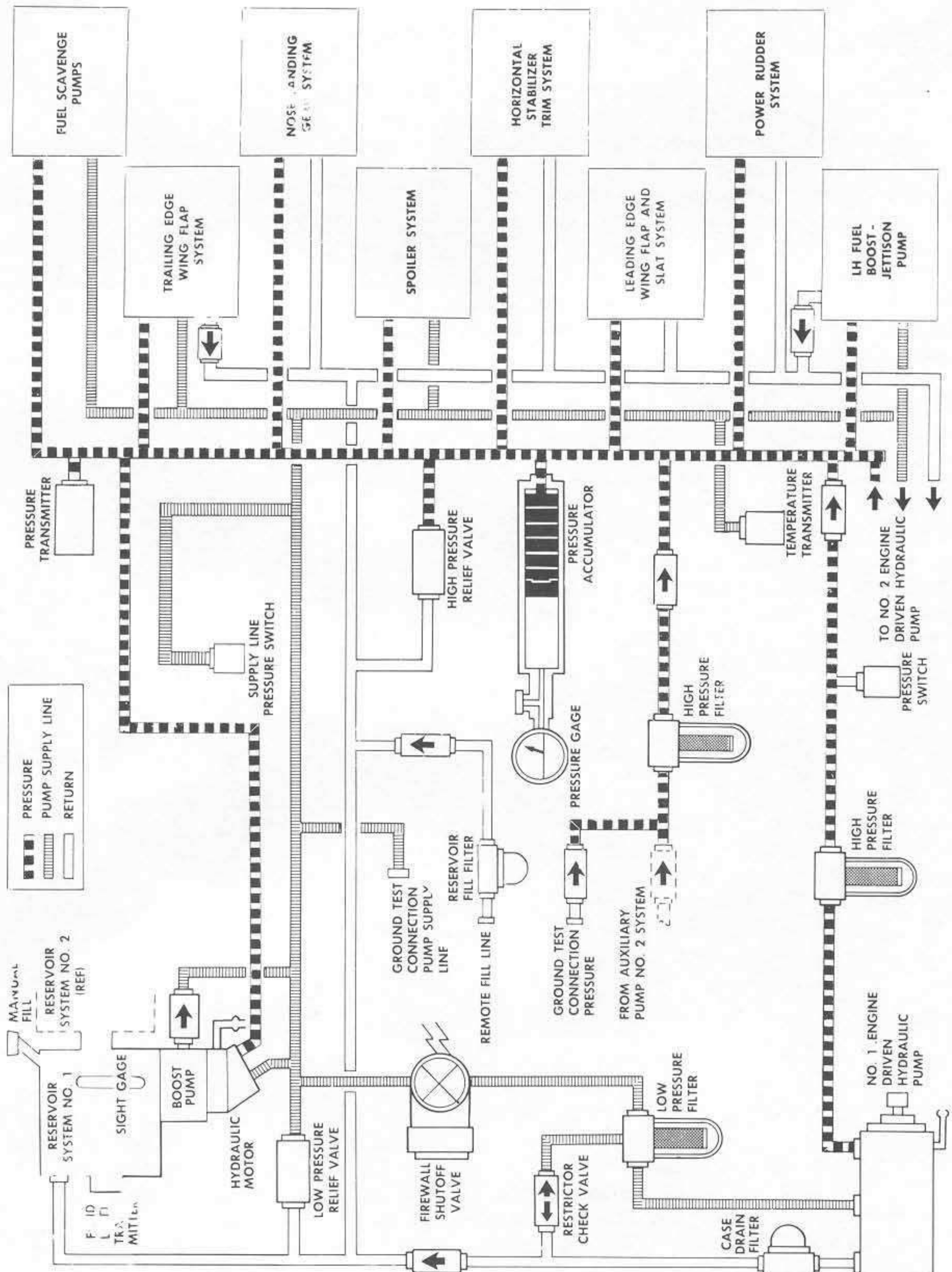
The No. 1 hydraulic power system (see Figure 12-1) consists of a hydraulic reservoir, a pump supply line boost pump, two engine-driven hydraulic pumps on engines 1 and 2, two emergency hydraulic shutoff valves, two low and three high-pressure filters, a pressure accumulator, low-pressure switch, pressure and temperature transmitter, a temperature control valve, pressure relief valves, check valves, and ground test connections. The system uses a pressurized pump supply line and a non-pressurized reservoir return line.

NO. 2 HYDRAULIC POWER SYSTEM

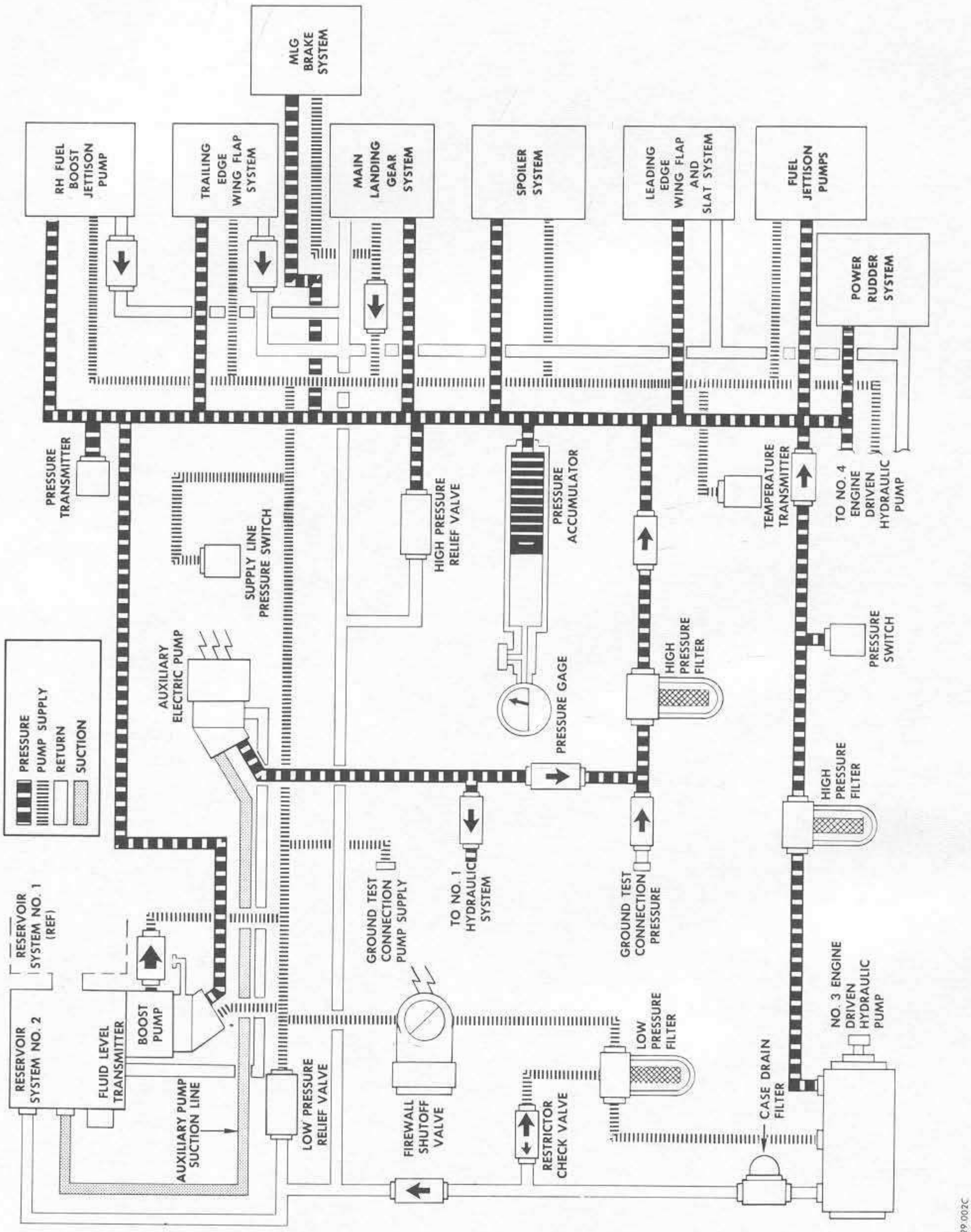
The No. 2 hydraulic power system (see Figure 12-2) is identical in operation and, with few exceptions, employs the same components as the No. 1 system. The hydraulic reservoir in the No. 2 power system is larger and is not equipped with a filler fitting. It is filled simultaneously when the No. 1 reservoir is replenished. The No. 2 reservoir also supplies non-pressurized hydraulic fluid to the auxiliary hydraulic pump. The engine-driven hydraulic pumps for system No. 2 are located on engines 3 and 4.

Reservoirs

Two stainless steel hydraulic reservoirs are used in the hydraulic power systems, one for each system. The No. 2 system reservoir has an indicatible fluid capacity of 6.7 gallons of hydraulic fluid. The No. 1 system reservoir has an indicatible fluid capacity of 2.4 gallons. The reservoirs are located on the right side of the hydraulic and pneumatic compartment. Both reservoirs are non-pressurized and are separately vented to the hydraulic compartment. A fluid connecting line between the two reservoirs permits simultaneous filling of both reservoirs from one filler opening. The connecting line is installed at the reservoir refill level to prevent a leak in either hydraulic system from depleting the fluid in the remaining system below an operational level.



No. 1 Hydraulic Power System
Figure 12-1



No. 2 Hydraulic Power System
Figure 12-2

Reservoir Filler Provisions

There are two methods of filling the reservoirs. A gravity manual fill opening is provided on the top of the No. 1 system reservoir. Both reservoirs can be hand filled through the one opening. Pressure filling is possible through the remote pressure filler connection located next to the hydraulic ground test connections (see Figure 12-3).

Hydraulic Reservoir Sight Gage

A hydraulic reservoir sight gage on the side of the No. 1 system reservoir indicates the fluid level in both reservoirs; either FULL or REFILL. In order to read the sight gage, both the No. 1 and No. 2 system accumulators must be uncharged, and the brake accumulator charged.

Pump Supply Line Boost Pumps

A reservoir pump supply line boost pump is located on the botton external surface of each reservoir. The boost pumps maintain a nominal 70 to 80 psi pressure in the supply lines to the engine-driven hydraulic pumps. Boost pump motors are operated by the 3000 psi hydraulic system pressure and normally operate in a near stall condition. When a drop in supply lines pressure exists, the pump automatically increases its output to meet the system demand.

Engine-Driven Hydraulic Pumps

Four variable-displacement engine-driven hydraulic pumps are used. One pump is installed on the forward left side of each engine accessory drive. The hydraulic pumps on engines No. 1 and No. 2 supply pressure to the No. 1 hydraulic power system. The hydraulic pumps on engines No. 3 and No. 4 supply pressure to the No. 2 hydraulic power system.

High and Low Pressure Filters

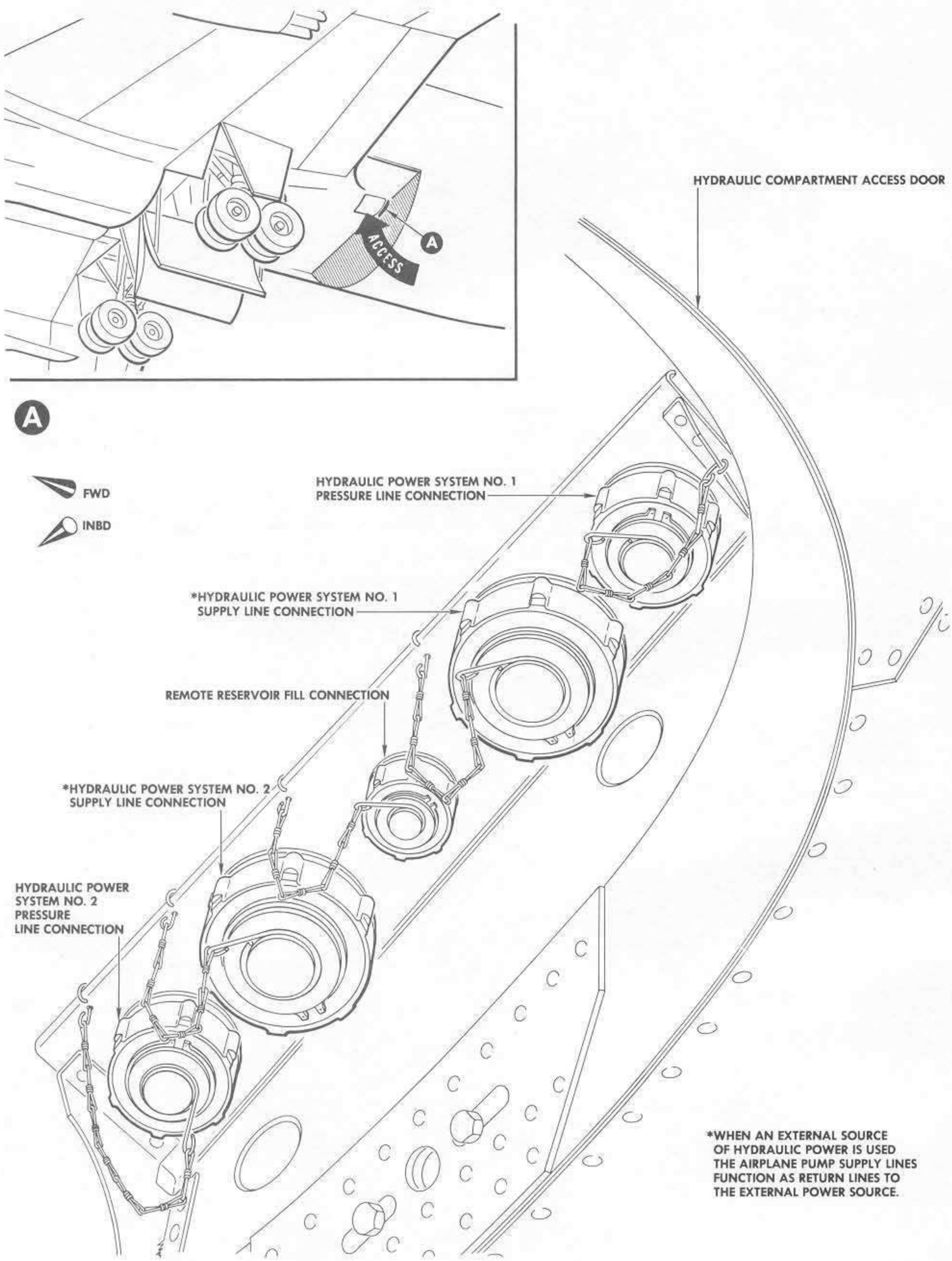
A high pressure filter is installed downstream of each engine-driven pump. These filters have no internal relief provisions. A low pressure filter is installed between the reservoirs and each engine-driven pump. A high pressure filter is also installed downstream of the ground test connection for each system.

Pump Inlet Shutoff Valves

A pump inlet shutoff valve is installed in the pressurized supply line to each engine-driven hydraulic pump. These valves are operated by a dc motor. When a FIRE-PULL "T" handle is pulled, the dc motor drives the valve to the closed position and shuts off the hydraulic fluid flow to the engine-driven pump.

Pressure Line Accumulators

A pressure line accumulator is installed in each main hydraulic system (No. 1 and No. 2). Each unit is pneumatically charged to 900 psi and dampens surges in the pressure lines. These units also store hydraulic fluid under pressure for momentary use by the operating systems until the pressure from the engine



30.30.29.001 A

Hydraulic System Remote Filler
and Ground Test Connections
Figure 12-3

pump is sufficient. The normal operating pressure of the accumulators is 3000 psi. A pressure gage is attached to each of the pressure line accumulators. The range markings are red from 0 to 900 psi and green from 900 psi to 3000 psi. Gage range is from 0 to 4000 psi. The gages indicate the amount of air pressure in the accumulators when the hydraulic pressure is 0 psi. They also indicate the pressure in the hydraulic system when the system is pressurized.

AUXILIARY HYDRAULIC POWER SYSTEM

The auxiliary hydraulic power system supplies pressure to both main systems (No. 1 and No. 2). The auxiliary system is normally used to provide hydraulic pressure for ground service checkout of the controls, and to maintain parking brake pressure.

The auxiliary hydraulic power system consists of an electric hydraulic pump, two filters, and the necessary connections and check valves to connect the system into the No. 1 and No. 2 hydraulic power systems.

Auxiliary Hydraulic Pump (Electric Driven)

The auxiliary hydraulic pump is a three-phase, 115/200-volt, ac operated, variable displacement pump controlled by the HYD PUMP switch on the flight engineer's panel. The pump is located in the hydraulic and pneumatic compartment. The pump operates at 3750 rpm and delivers a minimum of 2.45 gallons of hydraulic fluid per minute at 2600 psi. Hydraulic fluid is obtained from the No. 2 system reservoir through an auxiliary suction port located near the reservoir interconnect line level. An inverted standpipe prevents emptying the No. 2 system reservoir in the event the auxiliary pump is operating and a leak develops in system No. 1.

Auxiliary Hydraulic Pump Filters

A high-pressure filter is located in the auxiliary pump pressure line to each main hydraulic system.

HYDRAULIC SYSTEM CONTROL AND INDICATING UNITS

All No. 1 and No. 2 hydraulic power system control switches and indicators except the pressure indicators for the two power systems and the brake pressure indicator, are located on the upper left portion of the flight engineer's control panel (see Figure 12-4).

System Low-Pressure Switches

A pressure sensitive switch, located in each engine-driven hydraulic pump pressure line, detects excessive low output pressure. Each switch connects to a LOW PRESS warning light located on the flight engineer's panel. The switch closes at approximately 1000 psi.

TO BE FURNISHED

Flight Engineer's Hydraulic Control Panel
Figure 12-4

Pump Supply Low-Pressure Switches

A pressure sensitive switch is installed in the pressurized supply line to each main hydraulic power system. These switches energize a SUPPLY PRESS LOW warning light on the flight engineer's panel when the line pressure drops to 10 psi.

Remote Hydraulic Pressure Indicators

A hydraulic pressure transmitter in each main hydraulic power system pressure line transmits system pressure indications to their respective dial indicators located on the copilot's instrument panel. The indicators are placarded SYSTEM 1 and SYSTEM 2 and are calibrated from 0 to 4000 psi. A brake hydraulic pressure indicator is also provided.

Temperature Transmitters and Indicators

An electric temperature transmitter is installed in the pressurized pump supply line of each main hydraulic power system. The temperature transmitters indicate the hydraulic supply line fluid temperature by means of dial indicators on the flight engineer's panel. The temperature indicators are calibrated from minus 70 degrees to plus 150 degrees Centigrade.

Hydraulic Fluid Quantity Indicators

A fluid level transmitter in each hydraulic reservoir transmits fluid level signals to a dual-pointer indicator on the flight engineer's panel. The scale ranges are zero to 3.5 gallons for the No. 1 tank and zero to nine gallons for the No. 2 tank.

HYDRAULIC POWER SYSTEMS ELECTRICAL SOURCES

Emergency hydraulic shutoff valves	28-volt dc
Auxiliary hydraulic pump	115-volt ac
Low-pressure switches	28-volt dc
Pump supply line low-pressure switches	28-volt dc
Hydraulic system pressure indicators	26-volt ac
Temperature indicators	115-volt ac
Fluid quantity indicators	28-volt dc

For detailed information, consult the WIRING DIAGRAM MANUAL.

13 ELECTRICAL SYSTEM

Section 13

ELECTRICAL SYSTEM

TABLE OF CONTENTS

TITLE	PAGE
GENERAL AIRPLANE ELECTRICAL SYSTEM	13-1
CONSTANT SPEED DRIVE	13-1
Overspeed-Underspeed Control	13-4
Mechanical Disconnect	13-4
Overrunning Clutch	13-4
Thermal Switch	13-4
Load Controller	13-6
GENERAL AC ELECTRICAL SYSTEM	13-6
AC GENERATORS	13-10
Permanent Magnetic Generator	13-10
Static Exciter	13-10
AC Voltage Regulation	13-10
Reactive Biasing	13-10
Overvoltage-Undervoltage Fault Protection System	13-11
Generator/Feeder Faults	13-11
Unbalanced Current	13-11
Neutral Current and Open Phase	13-11
Load Faults	13-12
LOAD DISTRIBUTION	13-12
Essential Buses	13-12
Nonessential Buses	13-12
Synchronizing Bus	13-12
Pilot's Essential Bus	13-13
26-Volt AC Instrument Power	13-13
AUTOMATIC-PARALLELING	13-13
Auto-Paralleling Relay	13-13
Paralleling Sequence	13-15
FLIGHT ENGINEER'S AC CONTROL PANEL	13-15
Generator Control Switch	13-15
Generator Line Switch	13-15
Bus-Tie Switch	13-17
Generator Disconnect Switch	13-17
External Power Switch	13-17
Load Reduction Switch	13-17
26-Volt AC Power Switch	13-17
Frequency-Volt Selector Switch	13-17

TABLE OF CONTENTS (CONT)

TITLE	PAGE
AC SYSTEM WARNING LIGHTS	13-17
Generator Overheat Lights	13-18
Generator Off Lights	13-18
Generator Line Lights	13-18
Bus-Tie Lights	13-18
Drive Malfunction Lights	13-18
External-Power Lights	13-18
Load-Reduction Lights	13-18
26-Volt AC Power Lights	13-18
AC METER SYSTEM	13-19
Ammeters	13-19
Kilowatt Meters	13-19
Frequency Meter	13-19
Voltmeter	13-19
CIRCUIT PROTECTION	13-19
DC ELECTRICAL SYSTEM	13-19
Transformer-Rectifiers	13-19
Battery	13-21
Battery Charger	13-21
Reverse-Current Relays	13-21
Battery-Emergency Relay	13-21
Essential Bus Disconnect Relay	13-21
Current Limiters	13-21
Battery Distribution Bus	13-22
DC Power Distribution	13-22
FLIGHT ENGINEER'S DC CONTROL PANEL	13-22
Main T-R Unit Ammeters	13-22
Battery Ammeter	13-22
Voltmeter	13-24
DC SYSTEM WARNING LIGHTS	13-24

ELECTRICAL SYSTEM

GENERAL AIRPLANE ELECTRICAL SYSTEM

NOTE: The illustrations and discussions in this chapter include an auxiliary external power receptacle. The auxiliary receptacle is required only on those airplanes having electric freon system drive units.

The electrical system represents a marked departure from the low-voltage direct-current systems in general use for many years. All electric energy for flight operations is supplied by 115-volt, 400-cycle, 3-phase, ac generators, one mounted on each of the four turbojet engines (see Figure 13-1). Direct current for various airplane requirements is derived from transformer-rectifiers which convert the 115-volt ac generator current to 28-volt direct current. Low alternating-current voltage for instrument use is derived from voltage step-down transformers. A 27.5-volt storage battery provides an emergency source of dc power. External ac power is plugged into the system for use on the ground when the engines are not running.

To maintain constant frequency and also to combine the output of two or more ac generators, it is necessary to operate them at the same rotational speed. This is accomplished by connecting each generator to the engine through a hydraulically operated constant speed drive (CSD).

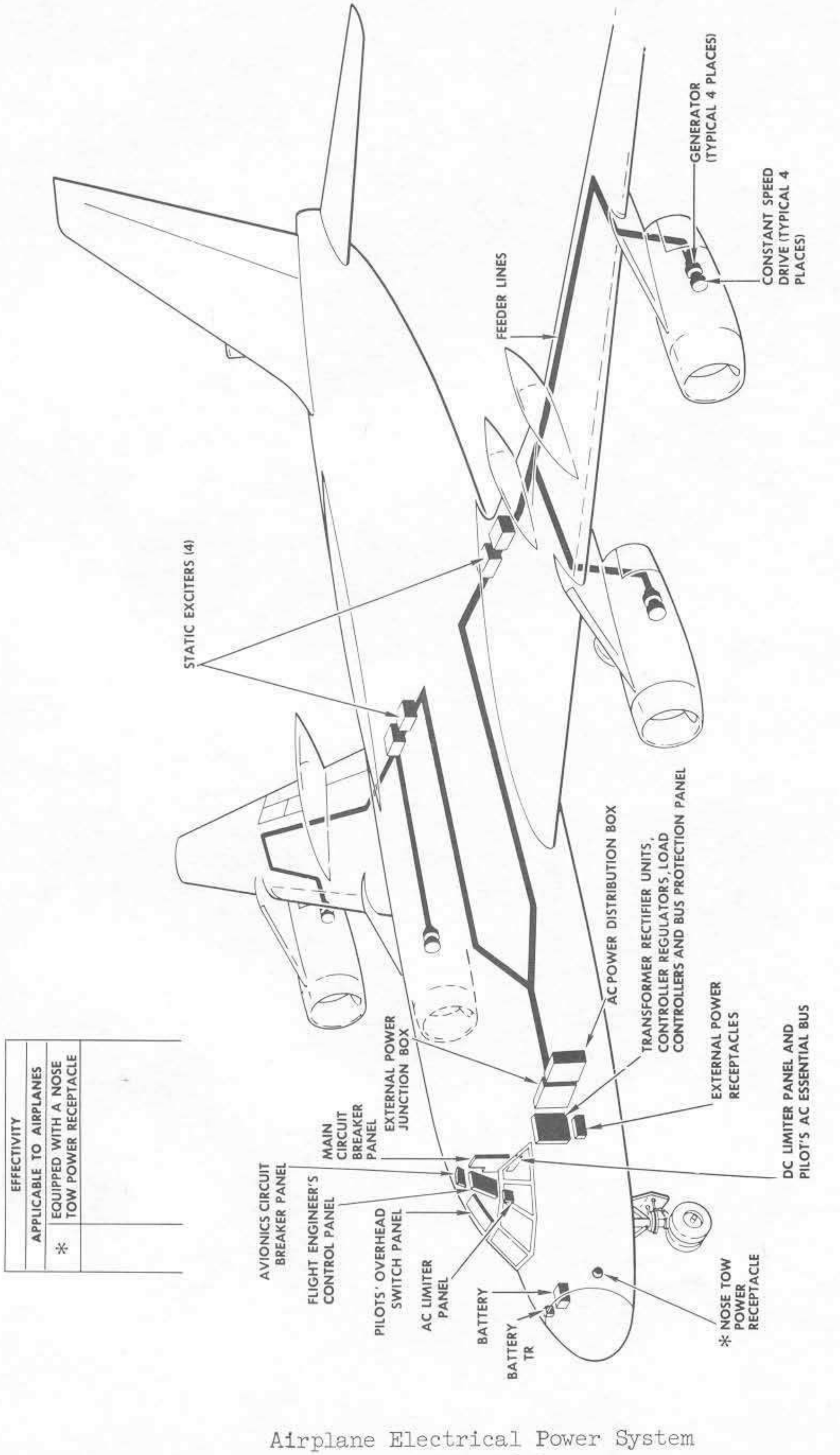
CONSTANT SPEED DRIVE

The output or generator drive speed of the CSD units is monitored and adjusted by the electrical system to keep the generators synchronized, divide the electric load equally among them, and disconnect a generator from the line and shut it down in case of faulty operation.

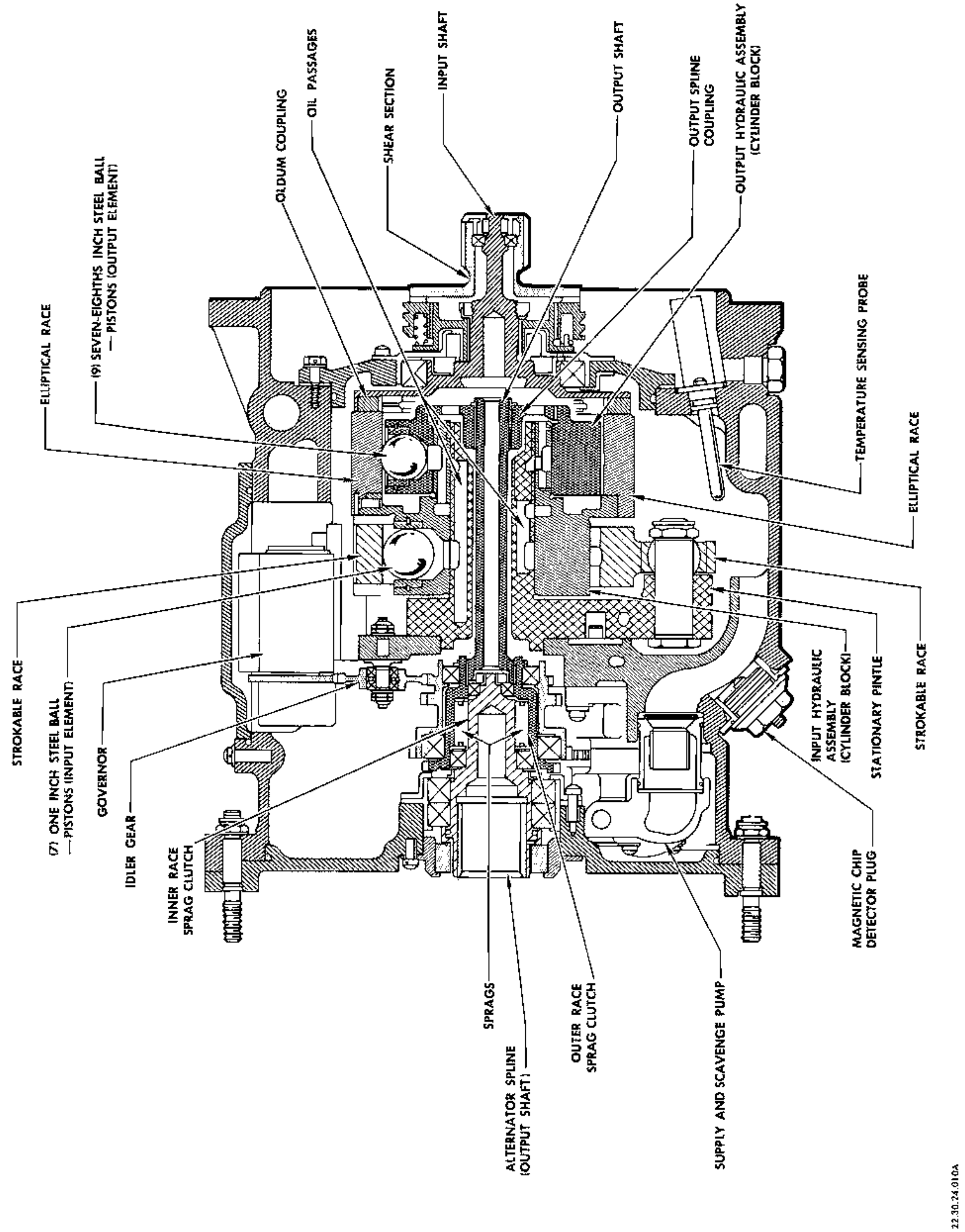
The CSD unit functions as an automatic differential transmission furnishing the driving speed increase or decrease required to maintain a constant generator speed throughout the engine operating range. It will convert any engine drive speed between 4300 and 7760 rpm to a constant 6000 rpm output speed to the generator. This speed corresponds to a line frequency of 400 cycles per second.

The operating mechanism of the CSD consists of a variable volume hydraulic pump connected to the engine drive, a hydraulic motor connected to the generator, two governors, a supply and scavenge oil pump, and electric units and circuits for equipment protection and synchronous control (see Figure 13-2). Each hydraulic unit contains a cylinder block, a tracking race, and ball pistons which are kept in firm contact with the tracking race by centrifugal force. The tracking race of the pump is circular but that of the motor is elliptical. Passages permit oil to flow back and forth between the pump and the motor.

The cylinder block of the pump and the tracking race of the motor form a single unit connected to the engine drive. The tracking race of the pump, which is pivoted to provide for varying the pump displacement, is positioned by the action of the control governor. The cylinder block of the motor is connected



Airplane Electrical Power System
Figure 13-1



CSD Cross Section View
Figure 13-2

to the generator drive. There is no mechanical connection between the engine drive and the generator.

When the engine is turning at the speed for which the CSD is set - in this case 6000 rpm - the control governor will maintain the pump tracking race in the neutral position. With the race in this position, there will be no pumping action and the motor block will rotate at the same speed as its tracking race. At lower engine speeds, the governor will move the pump race toward the increase-speed position, high pressure will be supplied to the motor, and the motor block will be driven forward within its rotating race. At higher engine speeds, the governor action will swing the pump race toward the decrease-speed side of neutral, the oil flow through the pump will be reversed, and the motor cylinder block will turn backwards within its rotating race reducing the speed of the generator.

Overspeed-Underspeed Control

An overspeed governor in each CSD unit protects the generator from excessive speed if the primary speed control should fail. If the output speed of the CSD should exceed 7200 rpm, the overspeed governor disconnects the control governor and actuates a hydraulic valve to move the pump race to the full decrease-speed position. Normal operation of the unit cannot be resumed until the engine has been shut down and the oil pressure within the hydraulic unit has dissipated.

A second set of flyweights within the governor operates an underspeed switch. This switch places the generator on the line when the output speed of the CSD increases to between 5625 and 5775 rpm and disconnects the generator when the speed decreases to between 5400 and 5250 rpm.

Mechanical Disconnect

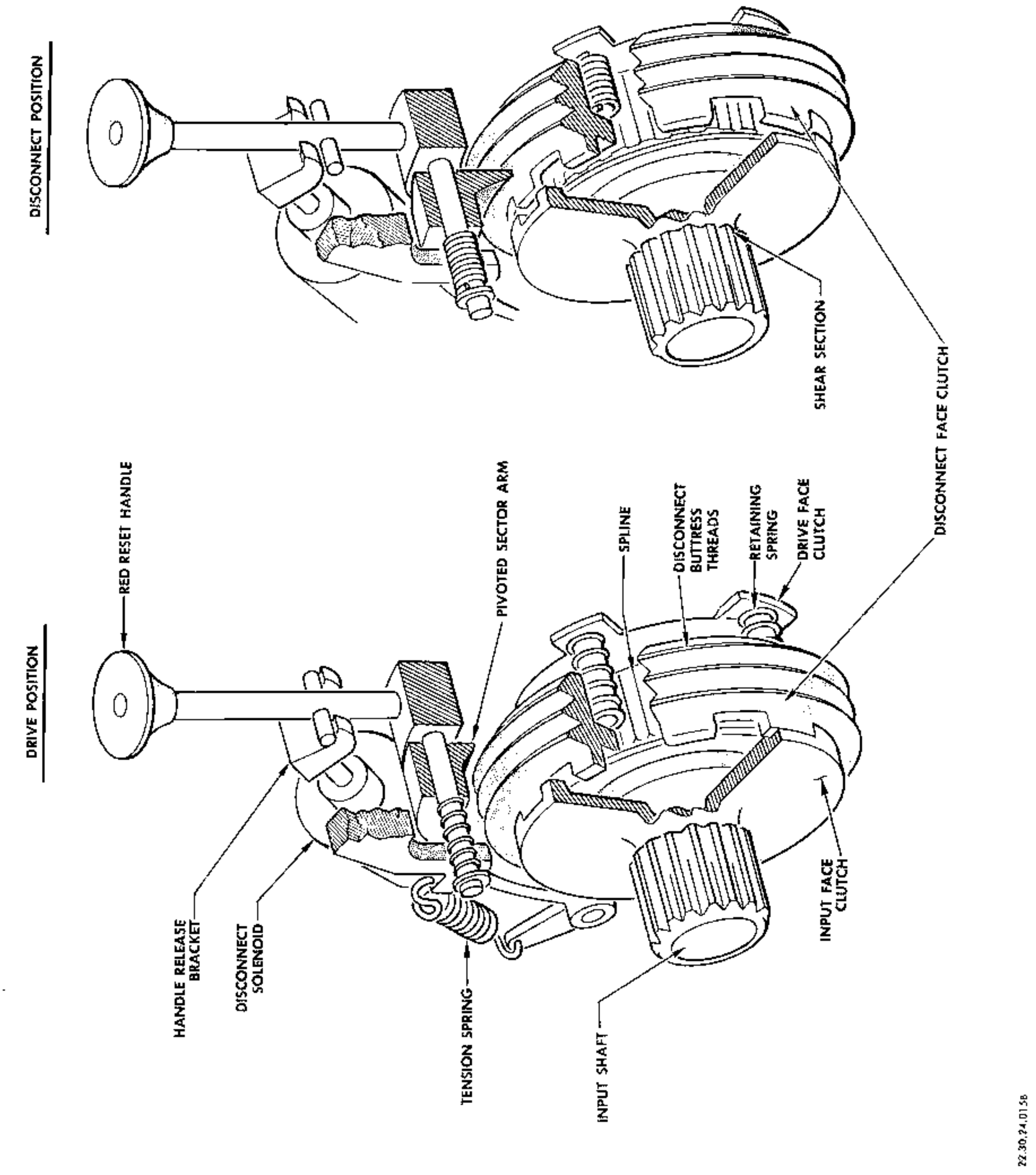
A switch on the flight engineer's panel actuates a mechanical disconnect mechanism, disengaging a face clutch in the drive shaft between the CSD unit and the engine. This clutch cannot be reengaged during flight. Manual reset is required by pulling the reset handle located at the CSD unit (see Figure 13-3).

Overrunning Clutch

An overrunning sprag-type clutch is located between each CSD output shaft and the generator drive. This free-wheeling clutch prevents the generator from driving the CSD if the generator should become motorized during an unbalanced load condition.

Thermal Switch

A thermal switch in the oil sump of each CSD provides an indication of an overheat condition. A red warning light on the flight engineer's panel illuminates when the oil temperature at the switch exceeds 177 degrees C (350 degrees F).



CSD Quick-Disconnect Unit
Figure 13-3

Load Controller

Four load controllers, one for each CSD unit, monitor the in-phase current of each generator and compare it with the average current of all the generators operating in parallel. The load controller modifies the action of the control governor in the CSD to vary the torque output, and correct uneven load distribution (see Figure 13-4).

GENERAL AC ELECTRICAL SYSTEM

The ac electrical system is a statically-excited four-generator system capable of producing 160 KVA continuously. The output of each generator is fed through its respective static exciter in the trailing edge of the wing, just inboard of the wheel wells. Here, the currents and voltages of the three phases are used to produce excitation for the generator field. From the exciter, the feeders continue through the generator line bus-tie contactor to the load bus.

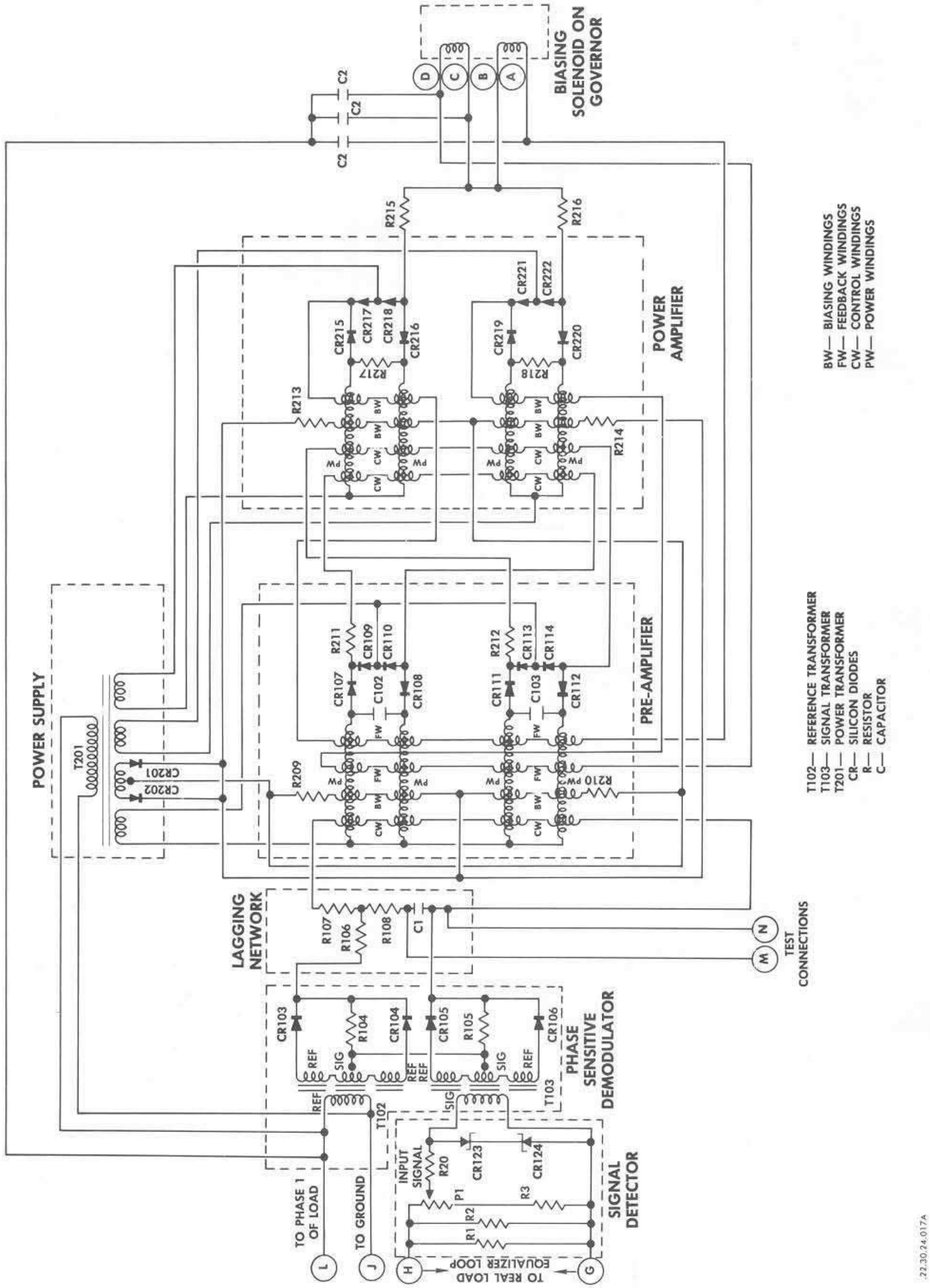
The ac system contains five load buses: two essential, two nonessential, and a pilot's essential bus. During normal operation, the generators and their respective load buses are paralleled by an automatic paralleling circuit. Bus-tie contactors connect the generator load buses to a synchronizing bus (see Figure 13-5).

Protective circuits are installed throughout the ac supply system. These protective circuits will automatically isolate a generator and its load bus from the synchronizing bus, or shut down the generator, depending on the fault or malfunction.

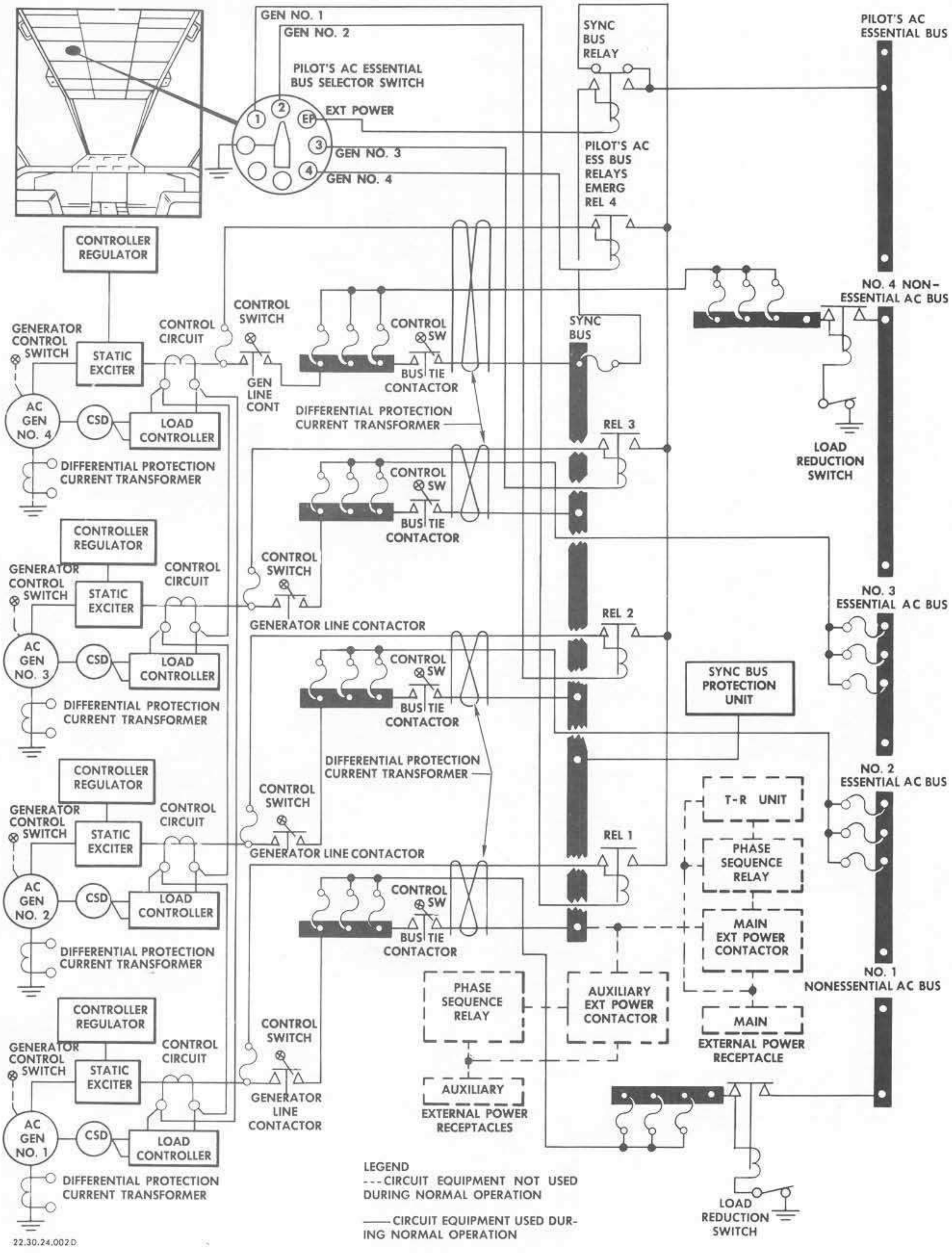
Warning lights, instruments, and switches are located on the flight engineer's ac panel for controlling and monitoring the ac electrical system. Through selective switching, a generator can be isolated without loss of electrical power to its load bus. The bus may be supplied electric power by the synchronizing bus.

For ground operation, an external ac power supply unit is plugged into the external power receptacle on the right side of the forward fuselage. The airplane's electrical system is protected from damage by a protective circuit which senses the ground power phase and frequency at the plug (see Figure 13-6).

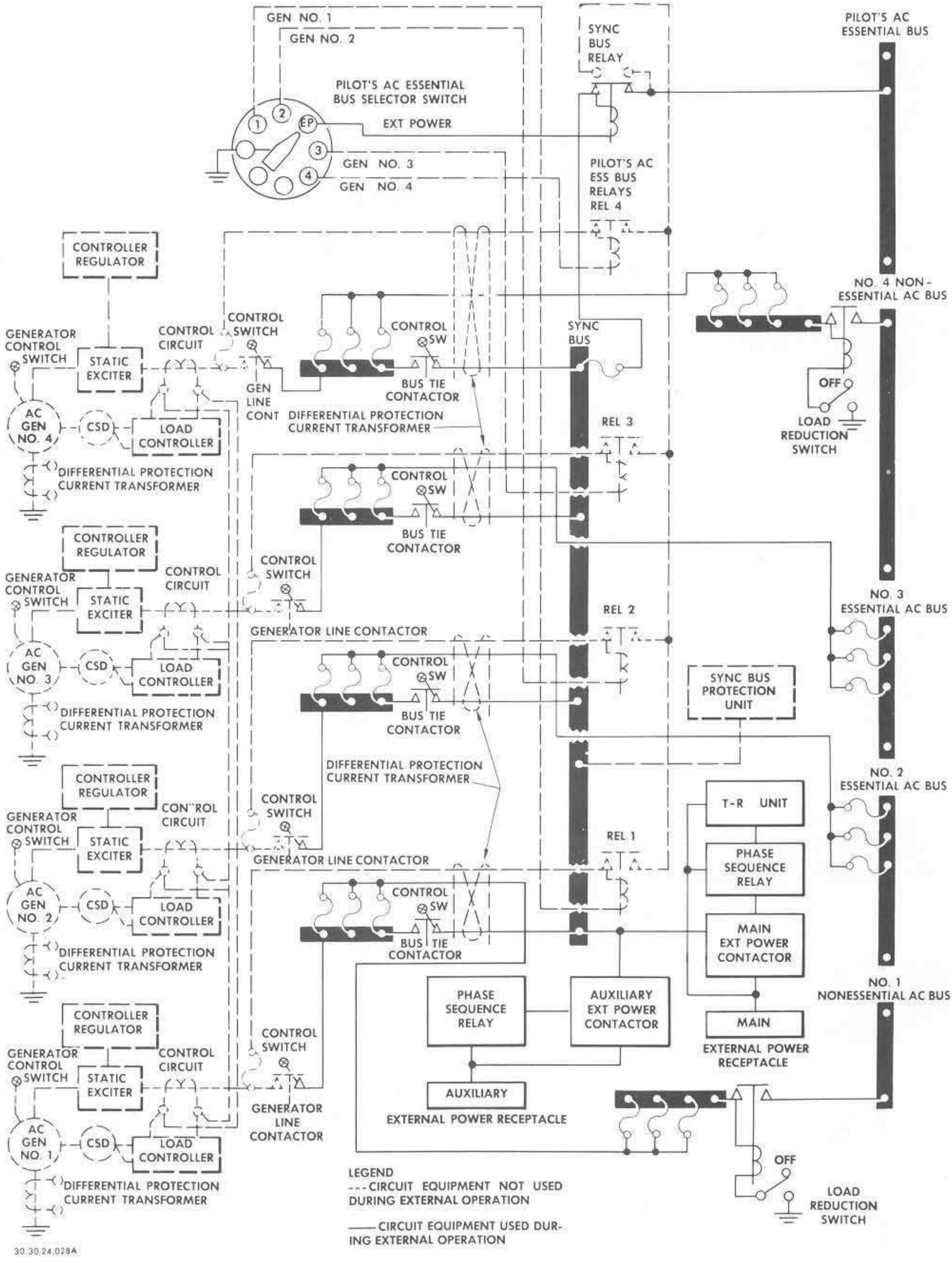
On some airplane versions a nose towing power receptacle and circuit are provided. The receptacle is located on the forward right side of the fuselage at station 198. Circuits for the nose towing power system and the external power system are interconnected to prevent simultaneous application of electrical loads to the systems.



Load Controller Schematic
Figure 13-4



Normal AC Power System
Figure 13-5



External AC Power System
Figure 13-6

AC GENERATORS

Four 120/208-volt, three-phase, Y-connected, 400-cycle, statically-excited ac generators are provided, one in the forward accessory compartment of each engine. Each generator is rated at 40 KVA continuous, with a capability of 60 KVA (150 percent of rated load) for five minutes, and 80 KVA (double load) for five seconds. The generators are self ventilated on the ground by means of integral fans. An air-inlet scoop on the left door of each engine nacelle provides ram air to cool the generators in flight.

Two thermal switches provide generator overheat and generator drive-end bearing overheat warning. One is mounted in the stator end turns and one in the front-bearing support. Overheat actuation of either switch illuminates a warning light on the flight engineer's panel.

Permanent Magnet Generator

A small permanent magnet generator (PMG) rotor is on the same shaft as the generator rotating field. Its stator provides 55 (±3) volt, 1600 cps, alternating current which is then rectified to 28-volt direct current for field flashing and control operations. When the generator is producing 115 volts, the PMG is removed automatically as the source of field excitation.

Static Exciter

A static exciter, consisting of saturable-current transformers, rectifiers, and a generator control relay, provides the normal excitation of the generator field. Increase in load results in an increase in exciter output to satisfy the added field current requirements. Conversely, decrease in the load results in a decrease in exciter output.

AC Voltage Regulation

The voltage output of the ac generators is regulated by voltage regulators located in the electrical compartment. The voltage is maintained within 2.5 volts of the rated line-to-ground voltage by varying the dc current from the static exciter to the field winding of the generator.

Reactive Biasing

For optimum performance in parallel generator operation, it is necessary to divide the load current equally between the generators connected to the synchronizing bus. Load current is made up of two components; a real component which is a direct measure of the work done in the system, and a reactive component which is the result of the inductive reactance of the load.

NOTE: This measure of the two components requires a definition of Power Factor. Power factor is defined as the cosine of the phase angle difference between the voltage and the current existing in the system. It may also be considered as the cosine of the angle between the system's apparent power (KVA) and the real power (KW).

The division of real power between the generators in the system is governed by the torque of the constant speed drive. The individual generator control system must sense and control the reactive component of the power. For parallel operation, a reactive load-division circuit is added to bias the voltage regulator to correct for differences in the reactive current between generators.

Overvoltage-Undervoltage Fault Protection System

An undervoltage relay is provided to protect the generator system and load equipment from sustained undervoltages or underexcitation. If an undervoltage condition (96 ±4 volts) exists on one or more phases for a period of five to ten seconds, the undervoltage relay will automatically disconnect the generator from its load bus.

The overvoltage relay protects the equipment from sustained overvoltages as well as high circulating currents caused by overexcitation. If the overvoltage relay detects an overvoltage condition in one or more phases, it will automatically disconnect the generator from its load bus after a time delay inversely proportional to the magnitude of overvoltage.

Generator/Feeder Faults

A differential protection relay is installed to remove a faulty generator from its load bus if any type of feeder or generator-stator short circuit occurs. A line-to-line or line-to-neutral current differential of 35 amperes or more between current transformers in the same phase will cause the differential current relay to be energized and the generator to be deenergized and disconnected from its load bus.

Unbalanced Current

An unbalanced current circuit, installed in each generator system, consists of a current transformer loop in phase "c" of each generator, a full wave bridge rectifier, a resistor, and an unbalanced current relay. The unbalanced current relay senses and protects the generator against loss of real load division by the constant speed drive. It is effective during parallel operation and provides back-up protection against failure of the control governor.

Loss of real load division will isolate the faulty generator in a paralleled system. When the current in phase "c" of the generators differs from phase "a" of "b" by more than 50 amperes, the unbalanced current relay will isolate this particular system, and the other systems will not be affected because of the reduced magnitude of the error signal. After isolation, the other protective circuitry will deenergize the faulty generator.

Neutral Current and Open Phase

Neutral-current sensing provides a means of detecting an open phase in the system, in addition to detecting a line-to-ground or a line-to-line-to-ground fault during parallel operation. The circuit for neutral-current sensing makes use of the same current transformer loop used for differential-current protection, except for an additional transformer in the neutral leg of the current transformer's secondary loop. For normally balanced 3-phase loads, the neutral leg does not carry current since the series aiding currents are restricted to the interphase loops.

When an open phase exists, current is produced in the neutral leg because of the missing signal in the transformer secondaries. The neutral current will flow in the primary of the transformer and energize the neutral-current relay, thus isolating one or more generators from parallel operation. Once the faulty generator has been isolated, it is the function of undervoltage sensing to trip the system.

Load Faults

With 50 to 80 amperes of neutral-line current resulting from a line-to-ground or from a line-to-line-to-ground fault, sufficient current will flow in the primary of the transformer to energize the neutral-current relay. The prime function of the neutral-current relay is to initiate action to split the system so that each generator assumes its own load (assuming that the load limiters have not cleared). Should the fault remain in the system once it is in isolated operation, the undervoltage relay will disconnect the affected generator.

LOAD DISTRIBUTION

There are three types of load buses which provide for ac power distribution: essential, nonessential, and pilot's essential bus. For detailed information, consult the WIRING DIAGRAM MANUAL.

Essential Buses

Essential buses are those required for safe flight. The essential buses are powered by No. 2 and No. 3 generators in isolated operation. If, during parallel operation the No. 2 or No. 3 generator fails, power for the essential bus is provided by the other generators through the synchronizing bus. The failed generator switch is turned off. The load will remain connected to the synchronizing bus as long as the bus-tie contactor switch is closed.

Nonessential Buses

The nonessential buses supply power to systems not required for safety of flight. The No. 1 and No. 4 generators power the nonessential buses in isolated operation. However, the nonessential buses may be energized by the other generators if No. 1 and No. 4 generators fail. The failed generator switch is turned off and the bus-tie contactor switch is left in the closed position.

Synchronizing Bus

The synchronizing bus is utilized for paralleling the generators, and for energizing an individual load bus when its generator has failed. This bus also supplies power to the pilot's essential bus for external power operation.

Pilot's Essential Bus

The pilot's essential bus is powered normally by any one of the four generators selected on the pilot's ESSENTIAL BUS SELECTOR panel. This power is taken from the generator before the line contactor. Thus, it is possible to have a generator operating and supplying power only to the pilot's essential bus. The pilot's essential bus supplies power to the essential flight instruments, to a transformer-rectifier for emergency 28-volt dc power, and to a second transformer-rectifier for battery charging (see Figure 13-7).

26-Volt AC Instrument Power

The 26-volt ac instrument power is delivered from a main or standby 26-volt ac single-phase, stepdown transformer. A two-position 26 V.A.C. POWER switch on the flight engineer's control panel controls the selection. The MAIN 26-volt transformer receives its power from the No. 3 essential bus and the STANDBY transformer from the pilot's essential bus.

AUTOMATIC-PARALLELING

The automatic-paralleling circuit ties each generator to the synchronizing bus at the proper instant to minimize system disturbances. The automatic-paralleling relay will synchronize the generators only when they are in the on-frequency range. It will not bring the generators within the on-frequency range. The generator output frequency is controlled by the governor in the constant speed drive.

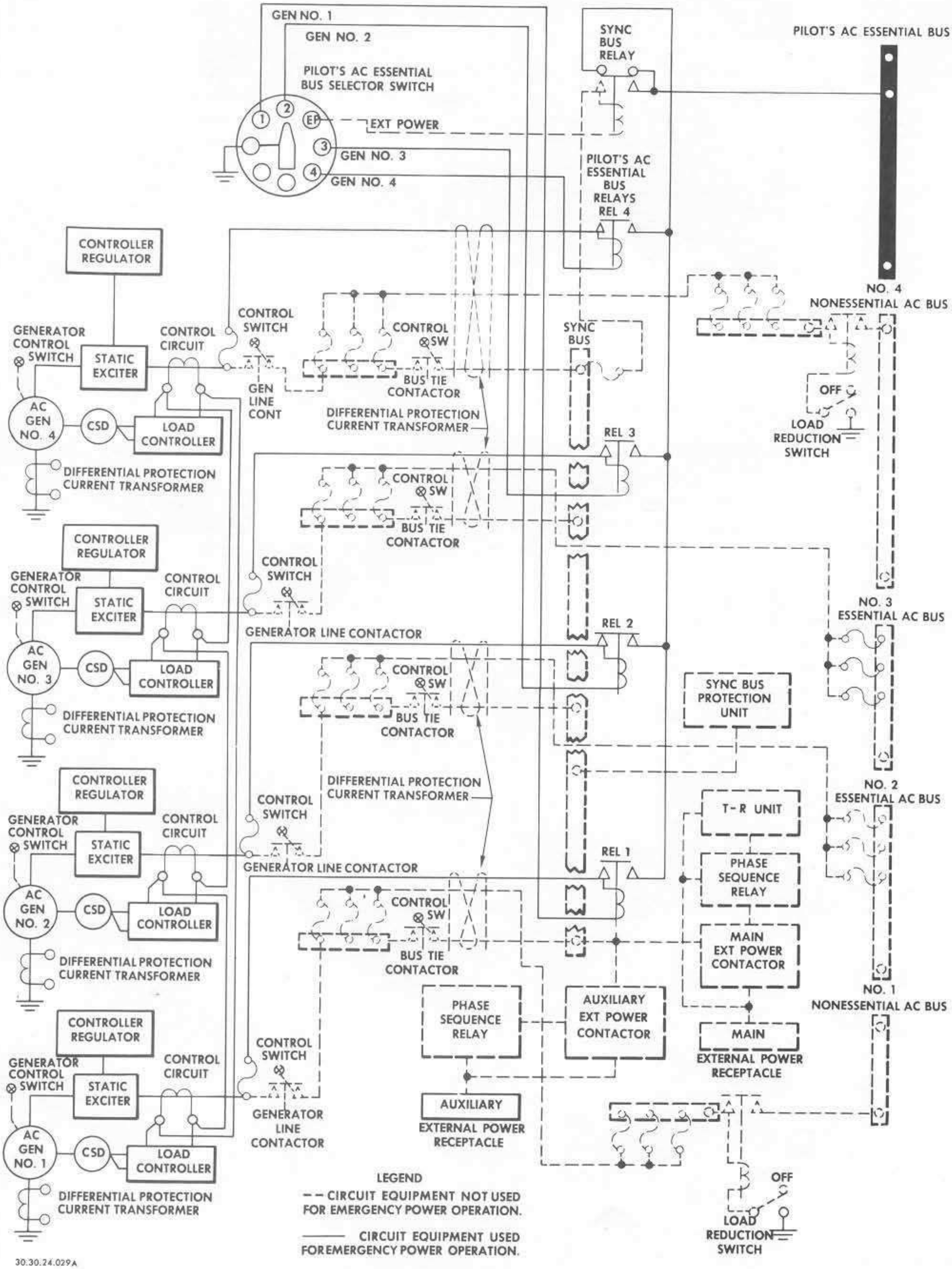
To synchronize generators, momentarily actuate the EXTERNAL POWER switch to the GEN PARALLEL position, thereby connecting one system to the synchronizing bus through the sequence-switching circuits. Each of the remaining systems will automatically be connected to the bus by its auto-paralleling relay.

Auto-Paralleling Relay

The auto-paralleling relay consists of a full-wave bridge circuit, an auto-paralleling relay, and a reference diode. The bridge circuit receives two signals: one from phase "c" of the synchronizing bus through a current limiting resistor, and a second from phase "c" of the system to be paralleled. If the two signals applied across the bridge network are at different frequencies, the voltage difference will vary as the difference in frequency. When the modulation (difference voltage) signal has reached the 90 degree point, the magnitude of the difference in voltage impressed is sufficient to trigger the breakdown diodes. The auto-paralleling relay is then actuated until the 270 degree point is reached, beyond which the reference diode ceases conducting and the relay will drop out. The drop-out initiates the paralleling action.

If the difference in frequency is less than 8 cps, there will be sufficient elapsed time to close the various contactors necessary to parallel. If the difference is greater than 8 cps, the auto-paralleling relay will energize before the contactors can close and the generator will not parallel.

The maximum phase difference between the synchronizing bus and the generator to be paralleled is 90 degrees. Beyond 90 degrees the system will not parallel.



Emergency Operation of the AC System
Figure 13-7

Paralleling Sequence

The No. 1 generator normally is connected to the bus first and the other generators are paralleled to it. If, for any reason, the No. 1 generator does not go on the bus but remains isolated, the next generator in sequence (the No. 2 generator) will be connected to the bus and the other generators will parallel to it. If the No. 2 generator is not ready to go on the bus, then the No. 3 generator will be connected to the bus and it in turn will be paralleled by the No. 4 generator.

For example, if the No. 1 generator is connected to the bus and the No. 2 generator will not come on the line because it is out of phase with the No. 1 generator by 100 degrees, then the No. 3 generator will connect to the bus and the No. 4 generator will parallel it. When the No. 2 generator gets back in phase with the No. 1 generator, or the synchronizing bus, it will then parallel.

FLIGHT ENGINEER'S AC CONTROL PANEL

The operation of the ac system is controlled through the ac control panel located at the flight engineer's station (see Figure 13-8). With the exception of the pilot's essential bus, this control panel is equipped with the necessary warning lights, instruments, and switches required for complete control of the ac system. The selector for the pilot's essential bus power source is on the pilot's overhead switch panel. On some airplane configurations provisions are made for the installation of an essential bus selector on the flight engineer's ac control panel.

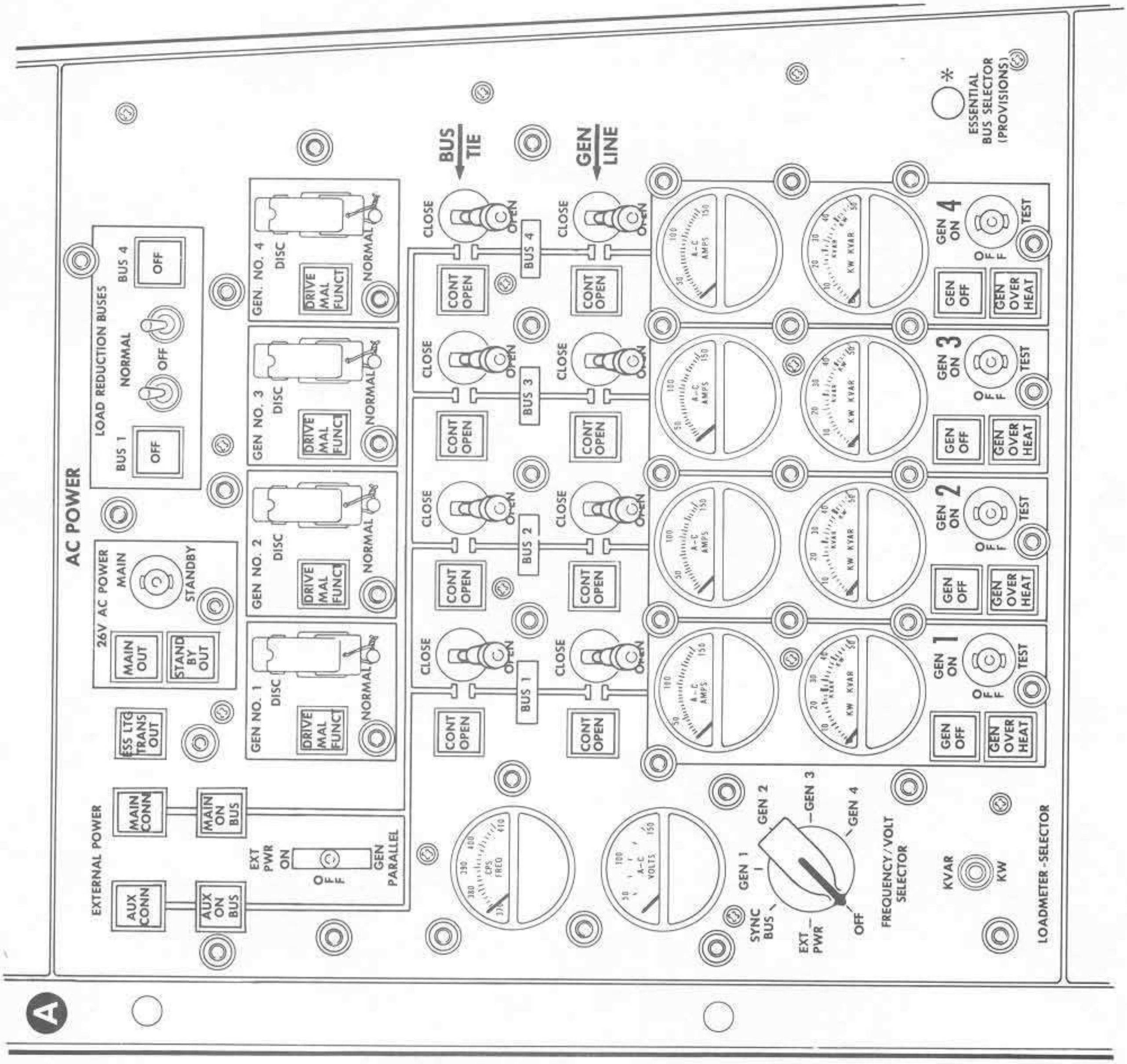
The ac system is operated automatically with all switches in the ON or CLOSE position. The system may be isolated by the action of the protective units or at the discretion of the flight engineer. The switches and warning lights are grouped to illustrate the flow and control of the power. There is a generator-control switch, a generator line-contactor switch, a bus-tie contactor switch, and a generator disconnect switch for each generator.

Generator Control Switch

Each GEN control switch is a three-position, ON-OFF-TEST, safety-type switch. The handle must be pulled out before the switch can be moved to the ON or OFF position. The TEST position is a spring-loaded momentary contact. In the ON position, the generator is allowed to build up and produce power. In the OFF position, the field of the generator does not receive excitation and the generator output is less than one volt (as a result of residual magnetism). The output of the generator can be checked by setting the selector switch to connect the generator to the voltage and frequency meters and holding the generator control switch in the TEST position.

Generator Line Switch

Each GEN LINE contactor switch is a two-position, OPEN-CLOSE, safety-type switch. The handle must be pulled out to move the switch to the OPEN or CLOSE position. The CLOSE position connects the generator to its load bus; the OPEN position disconnects the generator from its bus.



EFFECTIVITY	
	APPLICABLE TO AIRPLANES
*	CV1 THROUGH CV25

30.30.24.023A

Flight Engineer's AC Control Panel
Figure 13-8

Bus-Tie Switch

Each BUS-TIE contactor switch is a two-position, OPEN-CLOSE, safety-type switch which must be pulled out to move it to either position. This switch controls the circuitry for opening or closing the contactor between the individual load buses and the synchronizing bus.

Generator Disconnect Switch

Each GEN No. (x) disconnect switch is a guarded two position, NORMAL-DISC, switch. Actuation of the switch to DISC supplies power for disconnecting the constant speed drive input shaft from the engine.

External Power Switch

The EXTERNAL POWER switch is a three-position, ON-OFF-GEN PARALLEL, toggle switch. In the ON position, without the engines operating, external power is supplied to the synchronizing bus and, through the bus-tie connectors, to the load buses. With the engines operating and the switch in the ON position, external power is supplied to the synchronizing bus only. In the OFF position, no external power is applied to the buses. In the GEN PARALLEL position, external power is disconnected and the generators share the loads through the synchronizing bus.

Load Reduction Switch

Two LOAD REDUCTION, two-position, ON-OFF, toggle switches are provided. These switches control the No. 1 and No. 4 nonessential buses. In the ON position, ac power is supplied to the No. 1 or No. 4 nonessential bus, depending on the position of the corresponding switch. In the OFF position, those units operating from the nonessential bus will be inoperative. In the OFF position, the respective generators still supply power to the other buses through synchronizing bus.

26-Volt AC Power Switch

The 26 V.A.C. POWER switch is a two position, MAIN-STANDBY, toggle switch. Actuation of the switch determines which transformer will supply the 26-volt ac instrument bus.

Frequency-Volt Selector Switch

The FREQUENCY/VOLT SELECTOR switch is a seven-position switch allowing the flight engineer to read the frequency and voltage of the external power unit, the synchronizing bus, and each individual generator. An OFF position is provided.

AC SYSTEM WARNING LIGHTS

Warning lights are located on the flight engineer's ac instrument panel to warn of malfunction and indicate the position of the relays that control the operation of the ac system.

Generator Overheat Lights

Four red GEN OVERHEAT lights, one for each generator, are located along the bottom of the ac panel. A light will illuminate if a generator becomes overheated due to drive-end bearing friction or electrical load.

Generator Off Lights

Four amber GEN OFF lights, one for each generator, illuminate when the generators are not producing power and the dc emergency bus is energized.

Generator Line Lights

Four amber generator line CONT OPEN lights, one for each generator, illuminate when the generator line contactors are open and generator power is not connected to the load buses.

Bus-Tie Lights

Four amber bus-tie CONT OPEN lights, one for each generator, indicate the position of the bus-tie contactor for each generator. A light illuminates when the associated bus-tie contactor is open.

Drive Malfunction Lights

A red DRIVE MALFUNCT light is provided for each generator. A light will illuminate when the constant speed drive of the associated generator becomes overheated.

External-Power Lights

Two white lights, MAIN CONN and AUX CONN, illuminate when the main and the auxiliary power lines are connected to the airplane. Two blue lights, MAIN ON BUS and AUX ON BUS, illuminate, when main and auxiliary power is connected to the synchronizing bus.

Load-Reduction Lights

An amber load-reduction OFF light illuminates when No. 1 or No. 2 nonessential buses are disconnected.

26-Volt AC Power Lights

Two 26-volt ac power MAIN OUT and STANDBY OUT warning lights indicate the loss of 26-volt ac power. The main transformer indicator illuminates amber and the standby transformer indicator illuminates red upon failure of the respective unit.

AC METER SYSTEM

Ammeters, a frequency meter, voltmeter, and kilowatt meters, all located on the ac instrument panel, enable the flight engineer to monitor the operation of the ac electrical system.

Ammeters

Four ac ammeters indicate the current output of each generator.

Kilowatt Meters

Four ac kilowatt meters indicate the load output of each generator.

Frequency Meter

A single frequency meter is used to check the frequency of the external power, the synchronizing bus, and each individual generator. The frequency-volt selector switch controls the desired selection.

Voltmeter

A single ac voltmeter is provided to check the voltage of the external power, the synchronizing bus, or each individual generator. The voltmeter selection is accomplished simultaneously with frequency meter selection.

CIRCUIT PROTECTION

All circuits are protected by current limiters, fuses, or circuit breakers. The circuit breakers are located at the flight engineer's station. The fuses and current limiters are located in the electronic and electrical areas, and at the flight engineer's station. For detailed information, consult Chapter 24 of the MAINTENANCE MANUAL and also the WIRING DIAGRAM MANUAL.

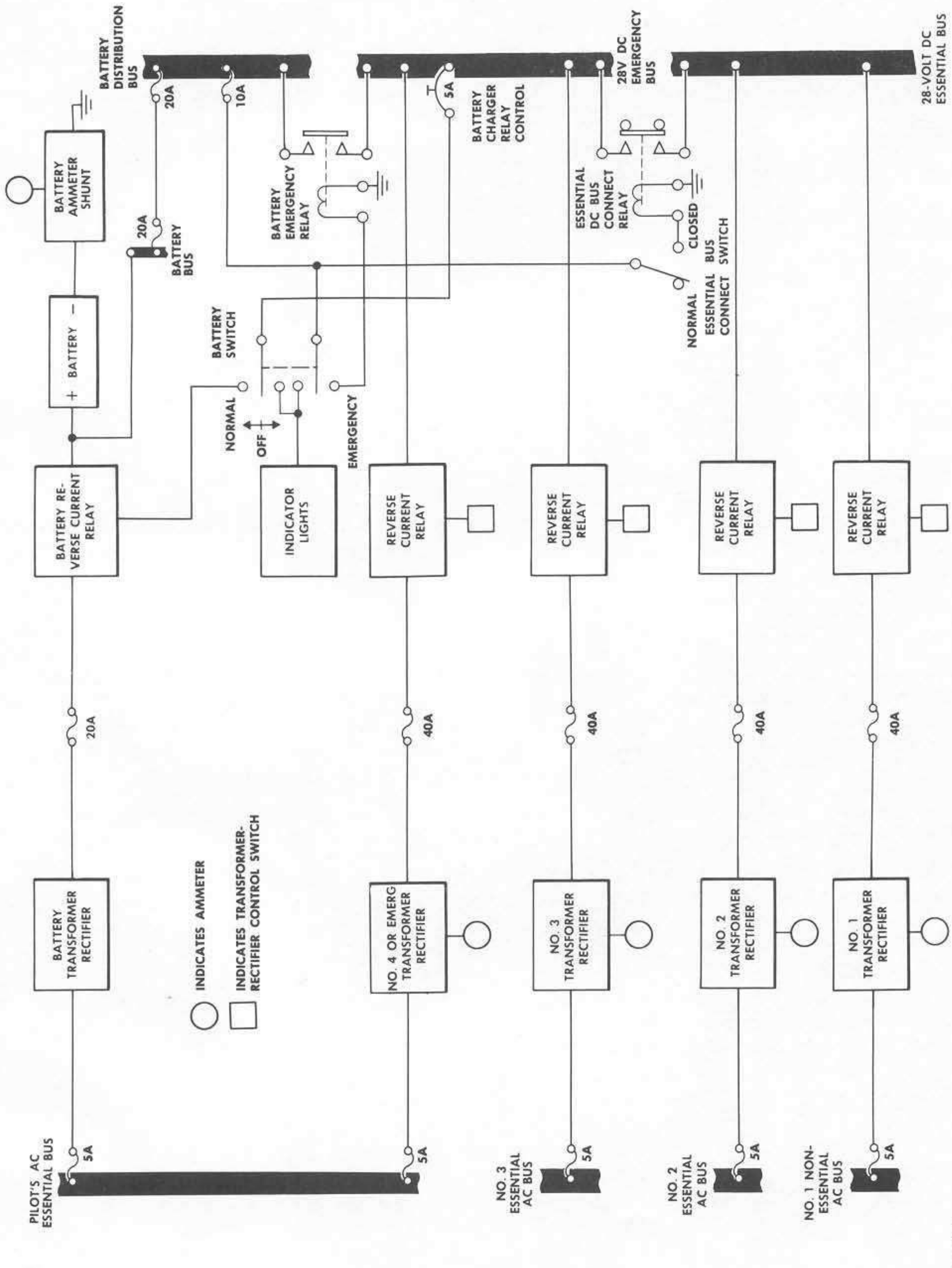
DC ELECTRICAL SYSTEM

The dc system does not use separate dc generators but obtains direct current through the use of transformer-rectifiers operating on power from the ac system. The dc system is automatic in operation when all transformer-rectifier (T-R) control switches and the battery switch are in their normal positions.

The dc system is composed of transformer-rectifiers, reverse current relays, a battery relay, an essential bus, an emergency bus, and a 27.5-volt 13.5 ampere-hour battery (see Figure 13-9).

Transformer-Rectifiers

Four 50-ampere transformer-rectifiers are installed in the right electrical equipment rack. The No. 1 main T-R unit is powered from the No. 1 nonessential ac bus, the No. 2 unit from the No. 2 essential ac bus, the No. 3 unit from the No. 3 essential ac bus, and the No. 4 unit from the pilot's essential ac bus.



DC Power System
Figure 13-9

The output voltage range of the T-R units is 24.0 to 27.5 volts for an individual load current range of 0 to 50 amperes. Outputs are paralleled during normal operation, thereby providing a 200-ampere, 28-volt dc bus. The T-R units are of the fan-cooled silicon rectifier type.

Battery

A 27.5 volt, 13.5 ampere-hour (at a 2-hour rate) nickel-cadmium battery is installed in the forward nose section of the airplane immediately inside the nose access door.

Battery Charger

The battery charger is a 29.25 to 33.25-volt, 0 to 20-ampere T-R unit of convection-cooled silicon rectifier design and is powered from the pilot's ac essential bus. It has an overload capacity of 30 amperes for 5 minutes. The unit is installed adjacent to the battery in the forward nose section of the airplane.

Reverse-Current Relays

Identical reverse-current relays are installed in the output lines of each main T-R unit and the battery charger. Those in the main T-R unit outputs serve as bus and wire protection against faults, and also as output control relays. The relay in the battery-charger output provides battery and wire protection against faults, and serves as the battery charger output control relay. The main T-R unit reverse-current relays are located at the dc buses in the electrical compartment. The battery charger reverse-current relay is installed in the battery relay box adjacent to the battery.

Battery-Emergency Relay

A 50-ampere relay, adjacent to the battery, connects the battery to the emergency bus for emergency operation.

Essential Bus Disconnect Relay

A 200-ampere capacity relay isolates the essential bus from the emergency bus to prevent excessive current drain on the battery during emergency operation. It is located in the dc junction box at the lower aft side of the flight engineer's station.

Current Limiters

A 40-ampere current limiter is installed adjacent to, and in the output of, each main T-R unit for protection of the wires to the main T-R unit reverse current relays. A 20-ampere limiter is installed adjacent to, and in the output of, the battery charger to protect the wire to the battery charger reverse-current relay. Three groups of three 5-ampere limiters, located in the ac power distribution box at the base of the flight engineer's station, protect the wires to the inputs of No. 1, No. 2, and No. 3 main T-R units. The input wires to the emergency main T-R unit (No. 4) and the battery charger are protected by 5 ampere limiters installed on the pilot's essential ac bus adjacent to the dc junction box. The triple wire feeder through which the battery supplies the emergency dc bus is protected by 20-ampere limiters. Various small limiters protect wires supplying power from the battery bus.

An emergency inverter, controlled by a switch at the Flight Engineer's station, provides emergency AC power for engine restart if all four engines should flame-out in flight.

Battery Distribution Bus

The battery bus, located in the battery relay box, supplies power to the battery control switch, the emergency and dome lighting circuit, the landing gear warning horn circuit, and the main fire extinguisher bus.

DC Power Distribution

For detailed dc power distribution consult the WIRING DIAGRAM MANUAL. The essential dc bus, located in the dc junction box, supplies power to all essential dc loads and to the nonessential dc loads through current limiters and circuit breakers. The emergency dc bus is located in the dc junction box. It supplies power from the main T-R units or the battery to all the emergency circuits through current limiters and circuit breakers.

FLIGHT ENGINEER'S DC CONTROL PANEL

The dc electrical system is monitored and controlled through the instruments and switches located on the flight engineer's dc control panel (see Figure 13-10).

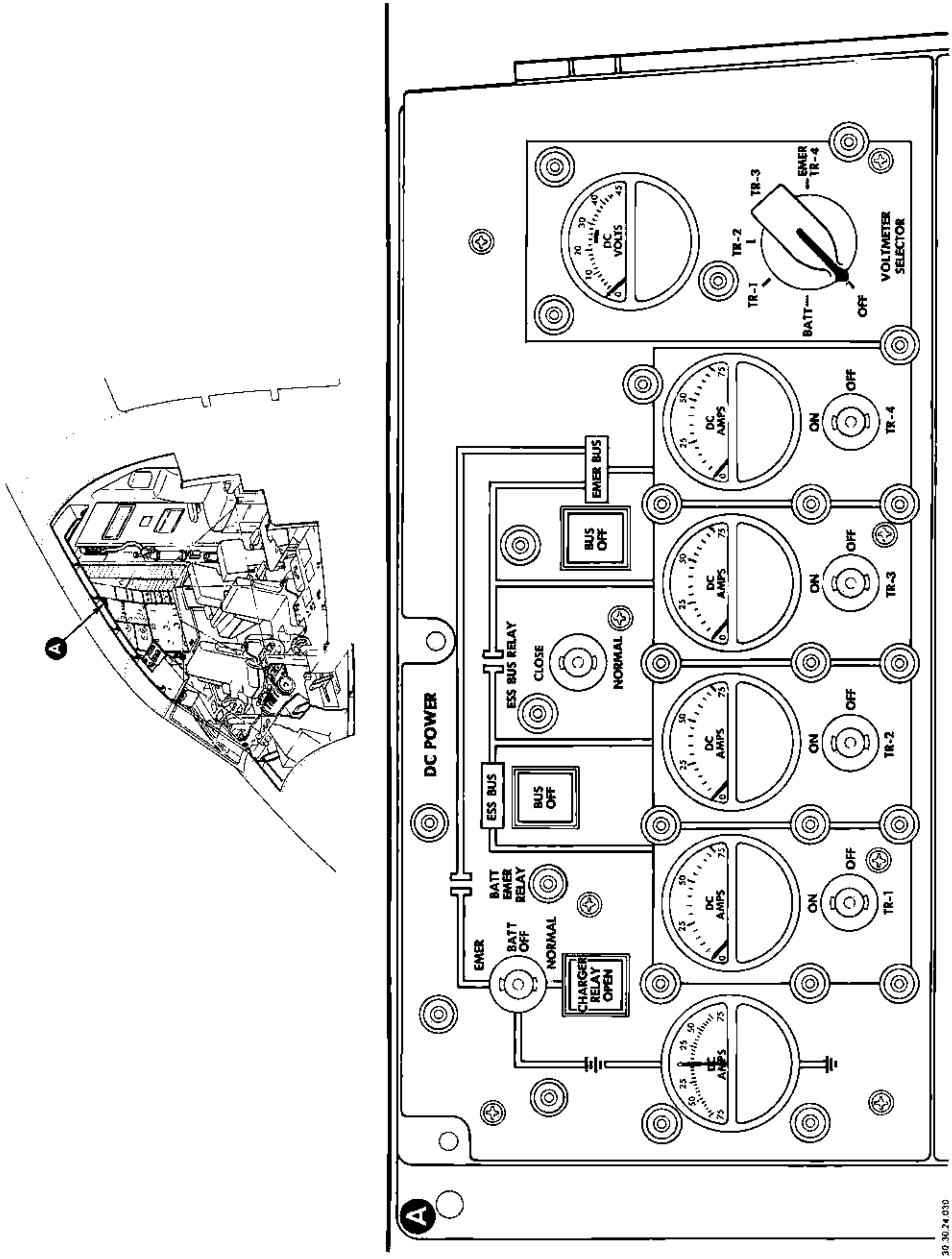
Main T-R Unit Ammeters

Four ammeters, one for each main T-R unit, are connected to the negative load of each main T-R unit. The meters are external shunt-type, calibrated from 0 to 75 amperes.

Battery Ammeter

The battery ammeter is an external shunt-type, 75-0-75 amperes, connected in the negative leg of the battery. One half of the scale indicates battery discharge current with the battery switch in the EMER position. The other half of the scale indicates battery charging current with the battery switch in NORMAL position. The meter is located on the flight engineer's panel.

NOTE: The battery ammeter can be interpreted to indicate state of charge of the battery. If the charging current is low (approximately 0.5 amperes), the battery is charged. If the meter indicates a charging rate of around 20 amperes, it is discharged. The charger will charge a dead battery in about one hour, during which the ammeter reading will vary from an initial value of about 20 amperes to the final value of 0.5 amperes.



Flight Engineer's DC Control Panel
Figure 13-10

Voltmeter

A voltmeter and a six-position selector switch OFF, BATT, TR-1, TR-2, TR-3, EMER TR-4 are provided on the flight engineer's dc panel. During normal system operation, the switch may be placed in the various positions and the corresponding voltages monitored.

DC SYSTEM WARNING LIGHTS

Three amber lights, located on the engineer's dc panel, indicate the electrical connections in the dc system.

When the battery switch is in the NORMAL position, and the pilot's essential bus is energized, the CHARGER RELAY OPEN light will be off. In the EMER position, the light will illuminate. The BUS OFF light in the essential bus circuit illuminates when the 28-volt dc essential bus is deenergized.

The BUS OFF light in the 28-volt dc emergency bus circuit illuminates when the emergency dc bus is deenergized.

14

AIR CONDITIONING AND
PRESSURIZATION SYSTEMS

Section 14

AIR CONDITIONING AND PRESSURIZATION SYSTEMS
(Electric Freon Drives Installed)

TABLE OF CONTENTS

TITLE	PAGE
GENERAL AIR CONDITIONING AND PRESSURIZATION	14-1
TURBOCOMPRESSOR UNIT	14-1
Turbine	14-5
Compressor	14-5
AIR-TO-AIR HEAT EXCHANGER	14-5
FREON COOLING SYSTEM	14-5
Freon Compressor	14-8
Freon Condensor	14-8
Freon Evaporator	14-10
Motor Cooling Loop	14-10
Surge Control	14-11
FLIGHT DECK ELECTRIC HEATER	14-11
CABIN ELECTRIC HEATERS	14-11
Cabin Differential Temperature Control System	14-13
CABIN AIR RECIRCULATING SYSTEM	14-13
ALTERNATE PRESSURIZATION	14-13
CABIN PRESSURE REGULATOR AND OUTFLOW VALVE	14-13
VALVES	14-16
Check Valves	14-16
Modulating Valves	14-17
Shutoff Valves	14-17
SYSTEM OPERATION	14-17
AIR CONDITIONING AND PRESSURIZATION SYSTEM CONTROLS	14-18
Instruments	14-18
Air Conditioning and Pressurization System Control Switches	14-22
Air Conditioning and Pressurization System Warning Lights	14-23
AIR CONDITIONING OPERATION	14-24
Inflight Heating	14-24
Inflight Intermediate	14-26
Inflight Cooling	14-26

TABLE OF CONTENTS (CONT)

TITLE	PAGE
AIR CONDITIONING OPERATION (CONT)	
Ground Heating	14-29
Ground Cooling	14-29
POWER SOURCES	14-32
Engine Bleed Air	14-32
Non-Essential AC Bus System	14-32
Essential AC Bus System	14-32
28-Volt DC Bus System	14-32
AIR CONDITIONING AND PRESSURIZATION SYSTEMS (Pneumatic Freon Drives Installed)	
GENERAL AIR CONDITIONING AND PRESSURIZATION	14-33
TURBOCOMPRESSOR UNIT	14-33
Turbine	14-37
Compressor	14-37
AIR-TO-AIR HEAT EXCHANGER	14-37
FREON COOLING SYSTEM	14-37
Freon Compressor	14-40
Freon Condenser	14-40
Freon Evaporator	14-42
Compressor Bearing Cooling Loop	14-42
CABIN ELECTRIC HEATERS	14-43
Cabin Differential Temperature Control System	14-43
CABIN AIR RECIRCULATING SYSTEM	14-43
ALTERNATE PRESSURIZATION	14-44
CABIN PRESSURE REGULATOR AND OUTFLOW VALVE	14-44
VALVES	14-47
Check Valves	14-47
Modulating Valves	14-47
Shutoff Valves	14-48
SYSTEM OPERATION	14-48
AIR CONDITIONING AND PRESSURIZATION SYSTEM CONTROLS	14-48
Instruments	14-48
Air Conditioning and Pressurization System Control Switches	14-53
Air Conditioning and Pressurization System Warning Lights	14-54

TABLE OF CONTENTS (CONT)

TITLE	PAGE
AIR CONDITIONING OPERATION	14-54
Inflight Heating	14-55
Inflight Intermediate	14-55
Inflight Cooling	14-58
Ground Heating	14-58
Ground Cooling	14-61
POWER SOURCES	14-61
Engine Bleed Air	14-61
Non-Essential AC Bus System	14-61
Essential AC Bus System	14-61
28-Volt DC Bus System	14-61

AIR CONDITIONING AND PRESSURIZATION SYSTEMS
(Electric Freon Drives Installed)

GENERAL AIR CONDITIONING AND PRESSURIZATION

The air conditioning and pressurization system provides passenger and crew comfort under all anticipated temperature conditions at airplane altitudes up to 41,000 feet. The pressurization equipment maintains sea level pressure in the cabin up to an airplane altitude of 21,300 feet, and an 8000-foot cabin pressure up to an airplane altitude of 41,000 feet. The air conditioning equipment supplies a complete change of air in the cabin every two and one-half minutes and in the crew compartment every minute. The inside temperature can be maintained between 18 degrees C and 33 degrees C (65 degrees F and 90 degrees F), with outside temperatures -54 degrees C to 38 degrees C (-65 degrees F to 100 degrees F). (See Figures 14-1 and 14-2.)

The cabin pressure and temperature are normally controlled automatically, but can be regulated manually at the discretion of the crew. The change of cabin pressure during rapid descents is controlled automatically at any preset rate between 50 to 2000 feet per minute.

The airplane is equipped with two independent and parallel pressurization and air conditioning systems. One system supplies air to the flight deck, the other to the cabin (see Figure 14-3). Two turbocompressors, driven by engine bleed air, provide an ample supply of compressed air to maintain the required cabin-air density during high-altitude operations. Inflight heating is provided by the compressing action of the turbocompressors. An electric heater is available for flight compartment heating (ground operation only). An air-to-air heat exchanger and a refrigeration unit in each system provide the cooling required to keep the cabin air at the selected temperature.

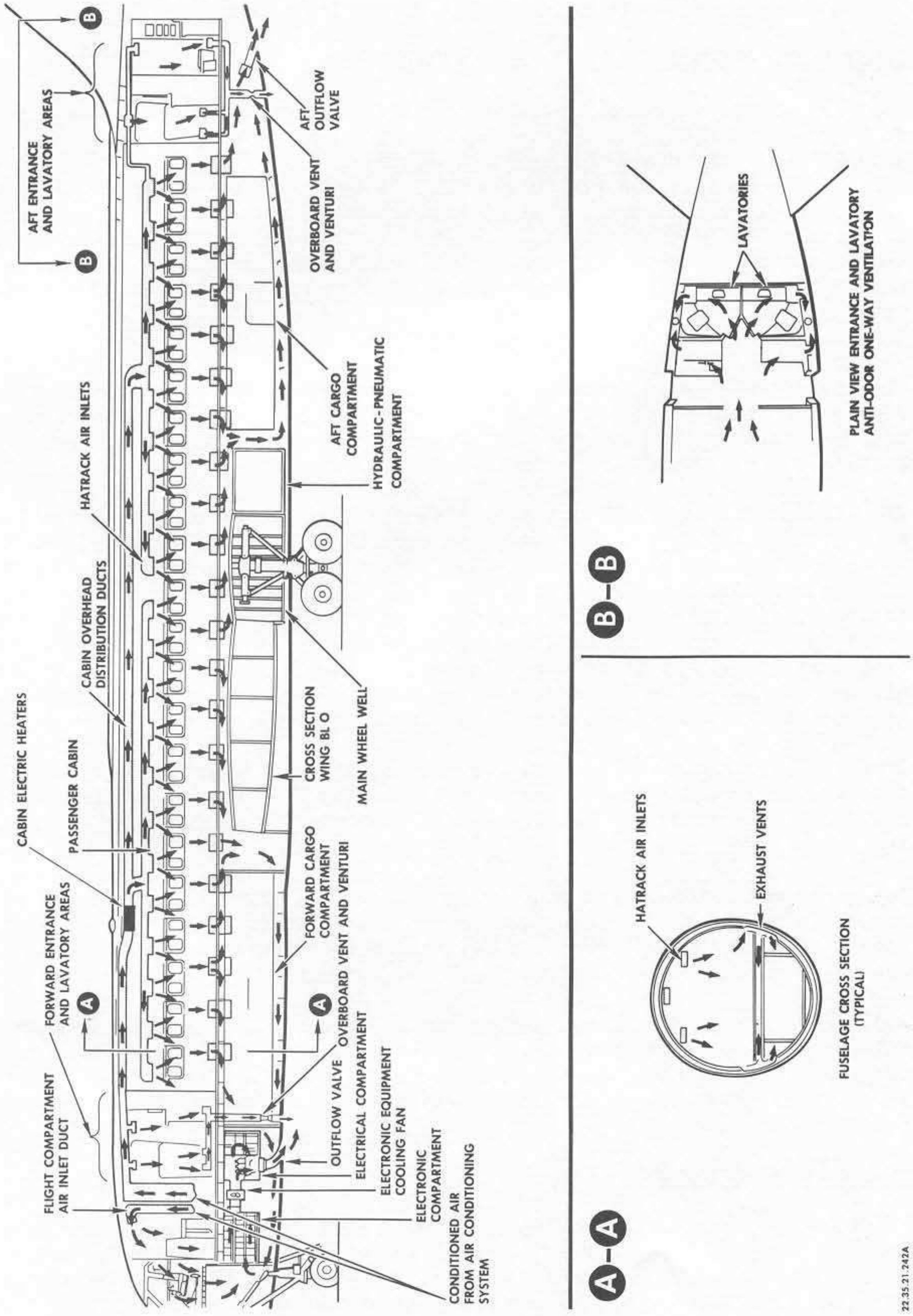
A recirculating fan can be used for recirculating the air through the cabin. A crossover duct provides a means of connecting the two systems if one turbocompressor should fail. Alternate pressurization is provided by bleed air from the engines.

The gages and switches for monitoring and controlling the operation of the air conditioning and pressurization system are located on the flight engineer's panel. The cabin pressure altitude, the cabin temperature, and the rate of change of cabin pressure can be regulated automatically or controlled manually.

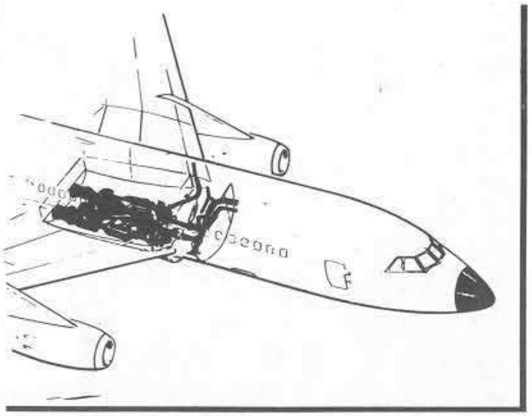
As the two air conditioning systems are identical, it will suffice to describe only one of them. Essentially, each system consists of a turbocompressor; a primary air-to-air heat exchanger; a refrigeration package; valves, regulators and switches for controlling the operation of the system; and the necessary air passages and ducts.

TURBOCOMPRESSOR UNIT

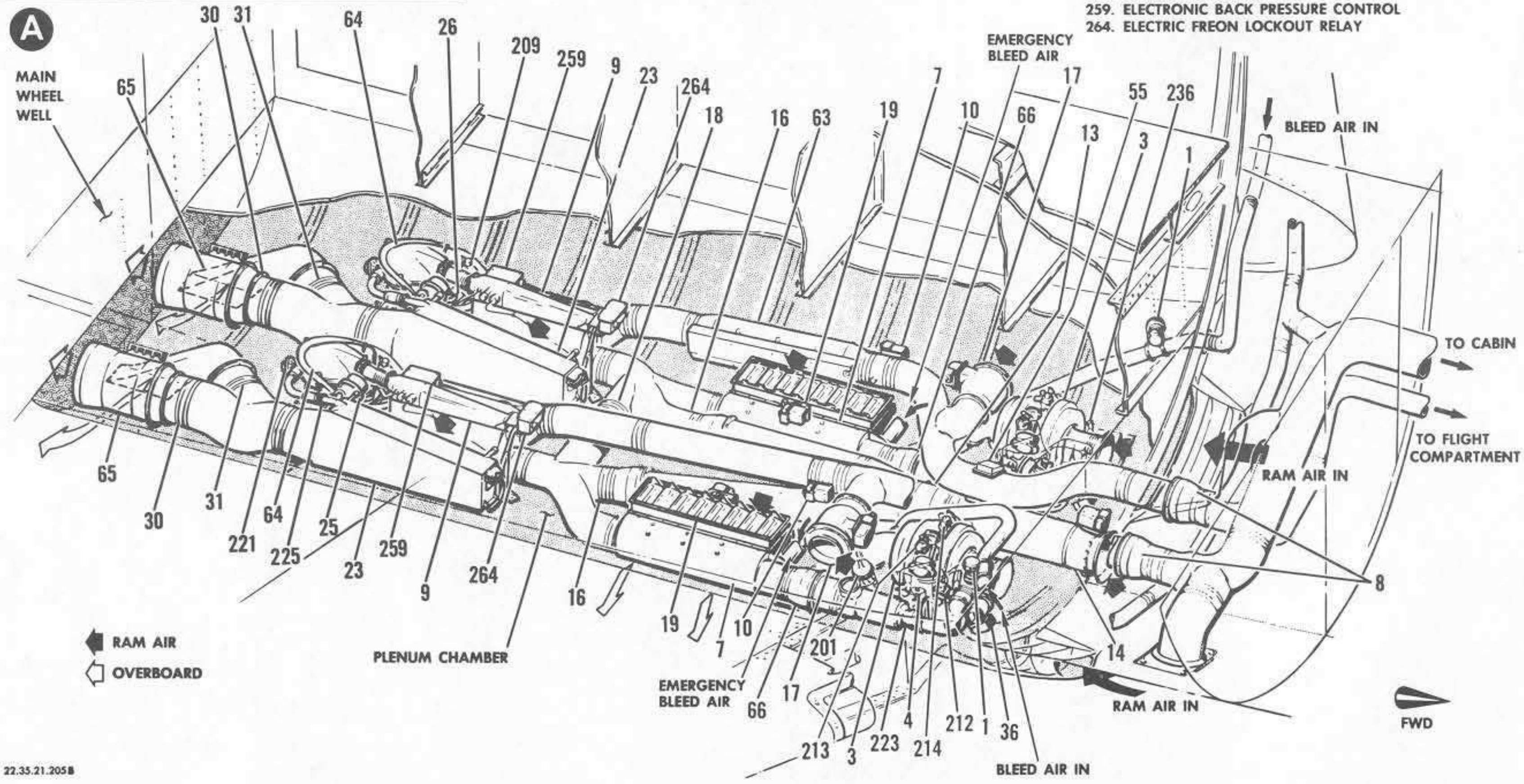
The turbocompressor supplies pressurized fresh air to the air conditioning system at the proper weight-flow and pressure to maintain the preselected cabin pressure. Each turbocompressor consists of a single-stage turbine and a centrifugal single-stage compressor mounted on a common shaft.



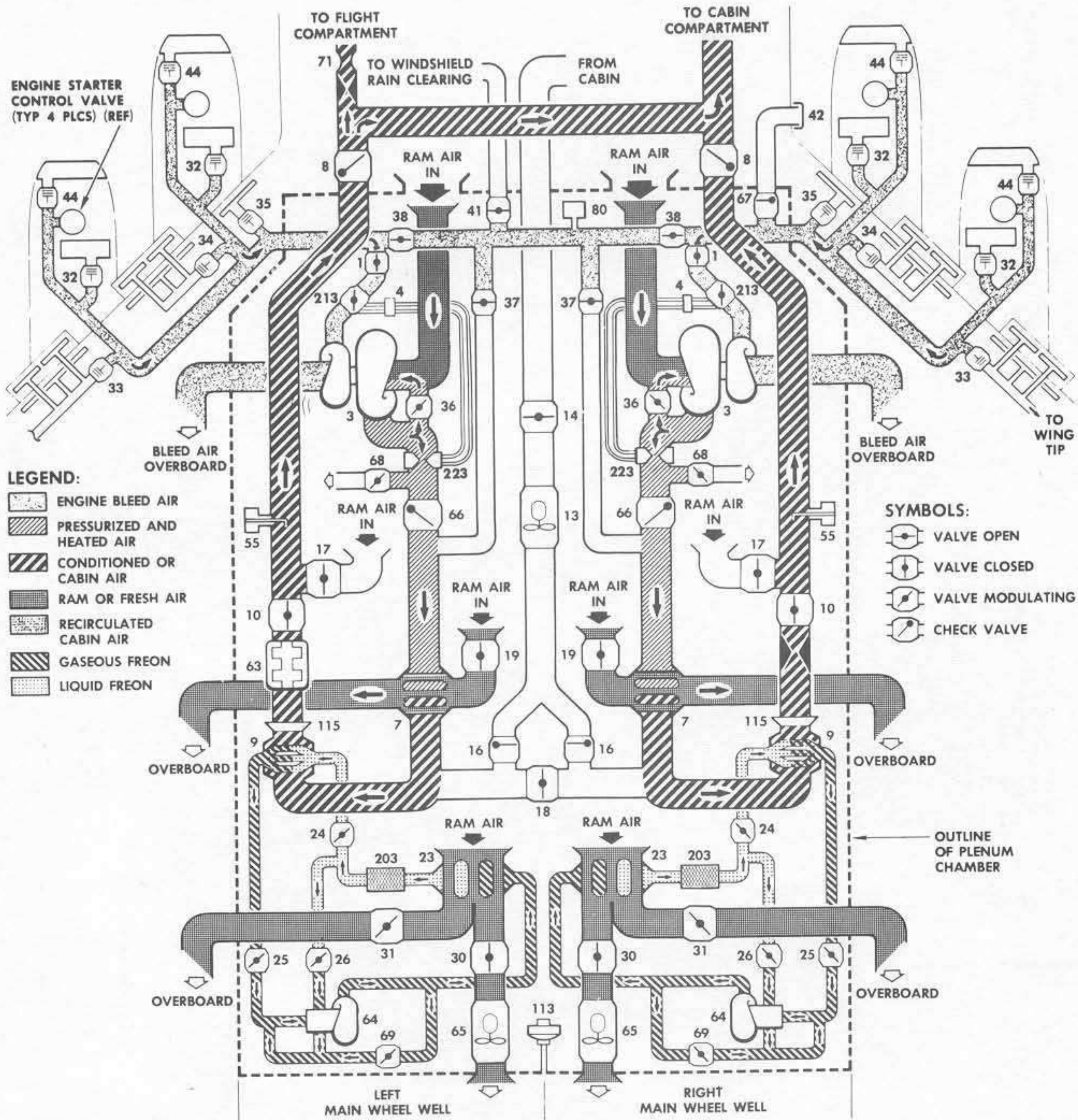
Air Conditioning and
Pressurization Airflow
Figure 14-1



1. TURBOCOMPRESSOR SHUTOFF VALVE
3. TURBOCOMPRESSOR
4. TURBOCOMPRESSOR FLOW CONTROL
7. HEAT EXCHANGER
8. CONDITIONED AIR CHECK VALVE
9. FREON EVAPORATOR
10. CONDITIONED AIR EMERGENCY SHUTOFF VALVE
13. RECIRCULATION FAN
14. RECIRCULATION CONTROL VALVE
16. RECIRCULATION AIR CHECK VALVE
17. RAM AIR SHUTOFF VALVE
18. CROSSOVER SHUTOFF VALVE
19. HEAT EXCHANGER COOLING AIR MODULATING VALVE
23. FREON CONDENSER
24. FREON EXPANSION VALVE (NOT SHOWN)
25. BACK PRESSURE REGULATOR VALVE
26. MOTOR COOLING EXPANSION VALVE
30. CONDENSER GROUND COOLING AIR SHUTOFF VALVE
31. CONDENSER COOLING AIR MODULATING VALVE
36. HEAT CONTROL MODULATING VALVE
55. FLIGHT DECK AIR FLOW SENSOR AND TRANSMITTER (CABIN SENSOR AND TRANSMITTER NOT SHOWN)
63. FLIGHT DECK ELECTRIC HEATER
64. FREON COMPRESSOR
65. CONDENSER FAN
66. TURBOCOMPRESSOR CHECK VALVE
69. FREON COMPRESSOR SURGE CONTROL VALVE (NOT SHOWN)
201. TURBOCOMPRESSOR SURGE VALVE
203. FREON DRIER AND STRAINER (NOT SHOWN)
204. FREON COMPRESSOR SURGE VALVE (NOT SHOWN)
209. CONDENSER TEMPERATURE CONTROL
212. PRESSURE LIMITER CONTROL
213. NOZZLE ACTUATOR
214. TURBOCOMPRESSOR SURGE CONTROL
221. FREON COMPRESSOR SURGE CONTROL
223. VENTURI DUCT
225. BELLOWS CONTROL
236. OVERSPEED CONTROL
259. ELECTRONIC BACK PRESSURE CONTROL
264. ELECTRIC FREON LOCKOUT RELAY



Air Conditioning Package
Figure 14-2



CODE:

1. TURBOCOMPRESSOR SHUTOFF VALVE
3. TURBOCOMPRESSOR
4. TURBOCOMPRESSOR FLOW CONTROL
7. HEAT EXCHANGER
8. CONDITIONED AIR CHECK VALVE
9. FREON EVAPORATOR
10. CONDITIONED AIR EMERGENCY SHUTOFF VALVE
13. RECIRCULATION FAN
14. RECIRCULATION CONTROL VALVE
16. RECIRCULATION AIR CHECK VALVE
17. RAM AIR SHUTOFF VALVE
18. CROSSOVER SHUTOFF VALVE
19. HEAT EXCHANGER COOLING AIR MODULATING VALVE
23. FREON CONDENSER
24. FREON EXPANSION VALVE
25. BACK PRESSURE REGULATOR VALVE
26. MOTOR COOLING EXPANSION VALVE
30. CONDENSER GROUND COOLING AIR SHUTOFF VALVE
31. CONDENSER COOLING AIR MODULATING VALVE
32. ENGINE BLEED AIR VALVE
33. OUTBOARD LEADING EDGE ANTI-ICING VALVE
34. MIDSECTION LEADING EDGE ANTI-ICING VALVE
35. INBOARD LEADING EDGE ANTI-ICING VALVE
36. HEAT CONTROL MODULATING VALVE
37. EMERGENCY CABIN AIR FLOW VALVE
38. BLEED AIR EMERGENCY WING ISOLATION VALVE
41. WINDSHIELD RAIN REMOVAL VALVE
42. ENGINE START AIR CONNECTION
44. ENGINE INLET DUCT LIP ANTI-ICING VALVE
55. AIR FLOW SENSOR AND TRANSMITTER
63. ELECTRIC HEATER—FLIGHT DECK
64. FREON COMPRESSOR
65. CONDENSER FAN
66. TURBOCOMPRESSOR CHECK VALVE
67. ENGINE START CHECK VALVE
68. TURBOCOMPRESSOR SURGE VALVE
69. FREON COMPRESSOR SURGE VALVE
71. FLIGHT COMPARTMENT AIRFLOW LIMITER
80. BLEED AIR OVERPRESSURE SWITCH
113. PLENUM CHAMBER PRESSURE SWITCH
115. WATER SEPARATOR
203. FREON DRIER AND STRAINER
213. NOZZLE ACTUATOR
223. VENTURI DUCT

30.02.21.122

Air Conditioning System Schematic
Figure 14-3

Turbine

The turbine is driven by bleed air from the main bleed-air manifold which receives air from the seventeenth compressor stage of each engine through a pressure regulator that reduces the pressure of the bleed air to 40 psig. After passing through the turbine, the bleed air is dumped overboard (see Figure 14-4).

Bleed air enters the turbine through a turbocompressor shutoff valve to a variable area nozzle controlled by a flow sensor at the compressor outlet. The sensor detects variations in airflow and automatically increases or decreases the nozzle opening to adjust the turbine speed accordingly. An rpm indicator, indicating turbine speed, is located on the flight engineer's panel. An overspeed control shuts off the supply of bleed air if the turbine speed increases to between 54,000 and 56,000 rpm. When the turbine overspeed cutout is tripped, the unit must be reset manually on the ground.

Compressor

Air conditioning ram air enters a plenum chamber through two large air scoops beneath the fuselage near the wing leading edge and passes directly into the compressor inlet. As the air passes through the compressor it is compressed and heated. If additional heating is required, a portion of the compressed air is recirculated through the compressor. A heat control modulating valve, regulated by the temperature control subsystem in the air conditioning system, controls the percentage of the compressor discharge air to be recirculated. A surge valve, also located at the compressor outlet, diverts compressed air back to the plenum chamber when a decrease in airflow through the compressor is accompanied by a marked increase in outlet pressure. This condition would arise in the case of a blocked duct. A check valve prevents reverse flow through the compressor when it is shut down.

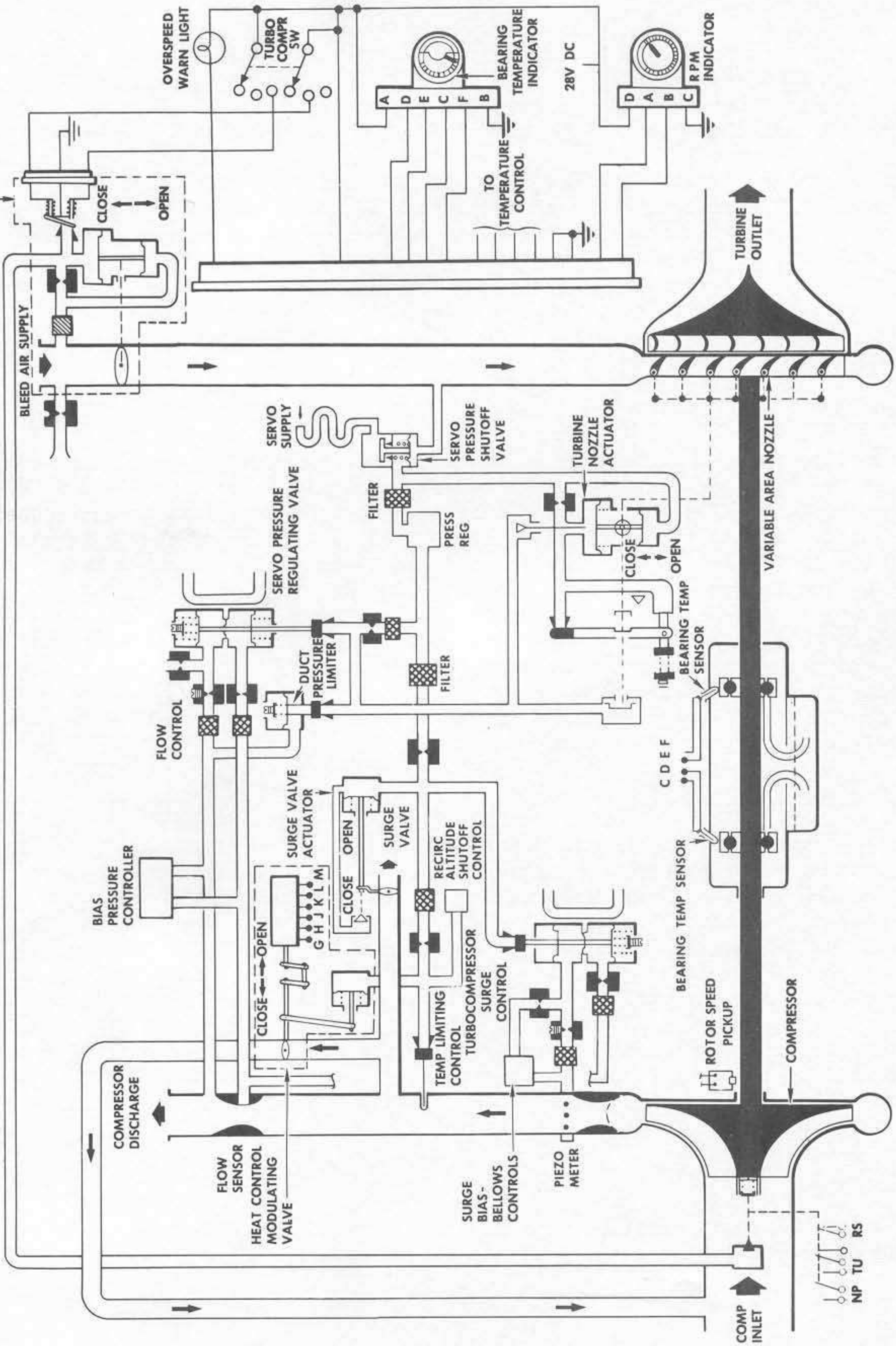
AIR-TO-AIR HEAT EXCHANGER

From the compressor, the heated compressed air passes through an air-to-air heat exchanger for primary cooling. The heat exchanger has a double-deck plate and fin core with vertical passages for ram air and horizontal passages for compressed cabin air (see Figure 14-5). The cabin air passes over and back through the two decks of the core. The ram air makes a single pass vertically through the core and absorbs heat from the plates and fins thus cooling the cabin air flowing through the horizontal passages.

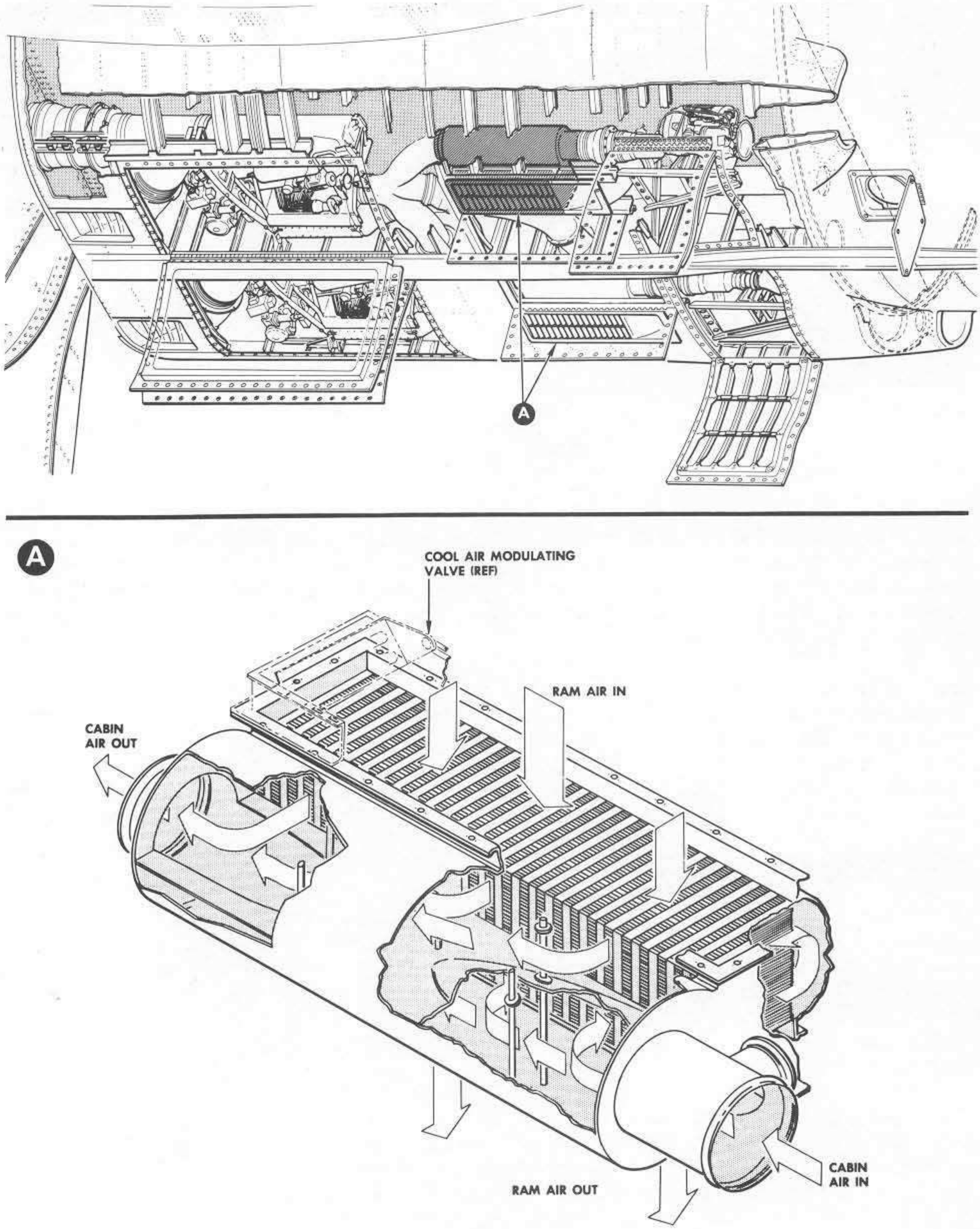
The ram air enters the heat exchanger through a cooling air modulating valve controlled by the temperature control subsystem of the air conditioning system. The amount of cooling varies from full capacity cooling to no cooling, depending on the position of the modulating valve.

FREON COOLING SYSTEM

Secondary cooling is provided by a closed vapor-cycle refrigeration unit located in the air conditioning compartment of the airplane. The unit consists of a



Turbocompressor Operational Schematic
Figure 14-4



22.35.21.228

Heat Exchanger
Figure 14-5

compressor, condenser, evaporator, and associated control devices (see Figure 14-6). Freon 114, a nonpoisonous fluid which boils at 3.5 degrees C (38.4 F), is used as the refrigerant. Special oil is mixed with the Freon, about 1 part in 20, to lubricate the bearings and other internal moving parts. The system is designed for either automatic or manual control and will maintain stable operation when adjusted for any amount of cooling between full capacity and essentially no cooling.

A vapor-cycle refrigeration depends on the latent heat of condenser and vaporization of a refrigerant to produce the desired drop in ventilating air temperature. The refrigerant, in a gaseous state, is passed through a compressor where it is compressed and heated. The compressed gas is then cooling and condensed in a heat exchanger that disposes of the latent heat released when the gas condenses. The refrigerant, now in the liquid state, flows through another heat exchanger where it changes back to gas. The heat needed to vaporize the liquid is extracted from the cabin air which the unit was designed to cool. From the evaporator, the gas is returned to the compressor and the cycle is repeated.

Freon Compressor

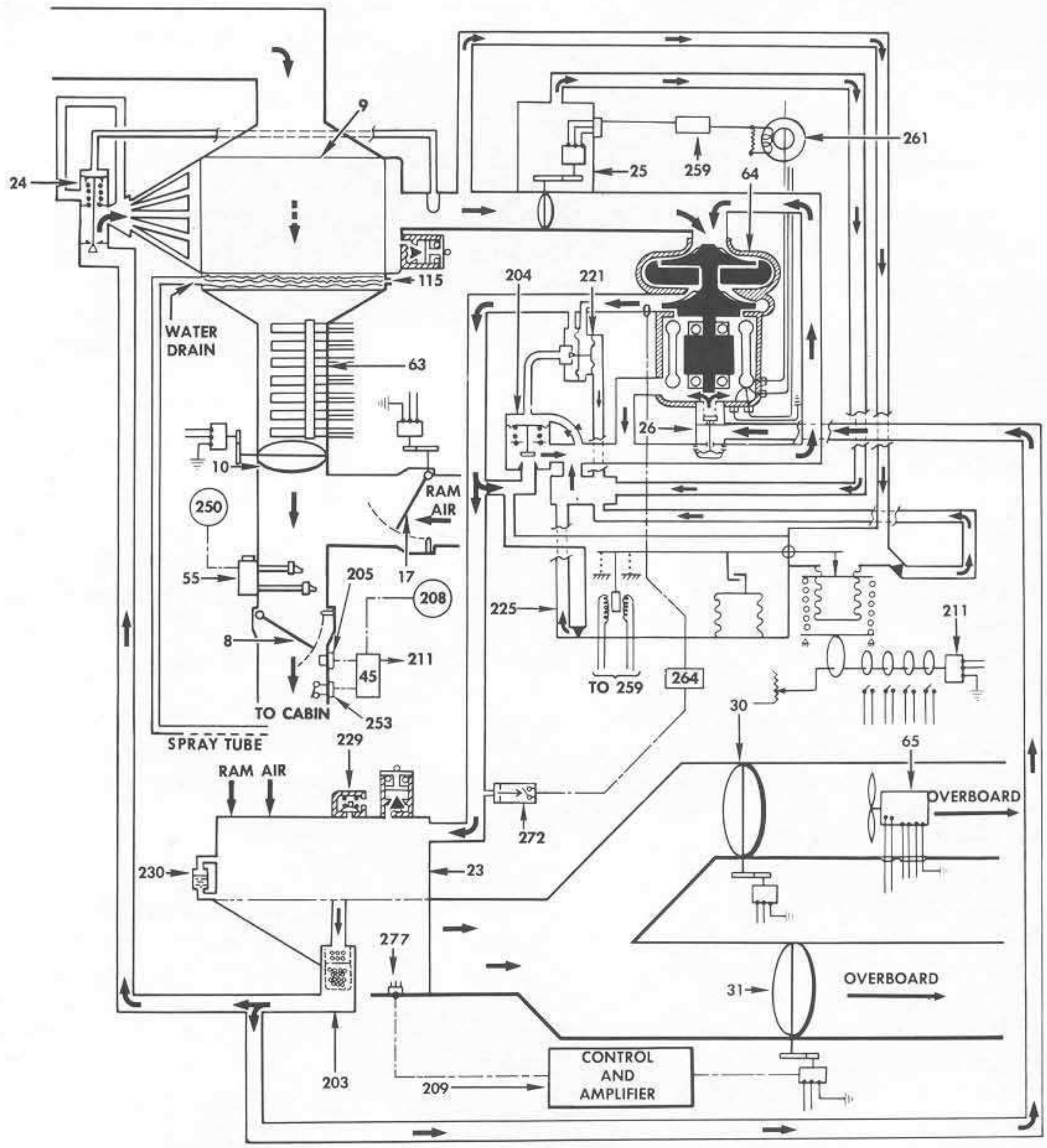
The Freon gas is compressed and heated in a two-stage centrifugal compressor with sweptback blades which permit a wide stable operating range for various compressor flows at nearly constant speed. The compressor is driven by a 3-phase, 400-cycle, 115/200-volt ac induction motor. The motor will develop about 25 hp and provide an 8-ton cooling capacity at a power factor of approximately 85 percent.

The motor is cooled and lubricated by circulating Freon through the bearings and passages in the motor housing. A back-pressure controller limits the operating current to 88 amperes per phase. Cutout switches, in conjunction with a lockout relay, protect the compressor from damage due to motor overheat, Freon discharge overheat or Freon overpressure. Another circuit protects the motor from burning out in the event a current limiter opens in the 3-phase power line.

Freon Condenser

The Freon condenser is a Freon-to-air type heat exchanger in which the hot compressed Freon gas is cooled and condensed to a liquid. Cooling is provided during most flight conditions by allowing ram air to pass through the condenser. For slow flight and ground cooling, plenum air is drawn through the condenser by an electric fan. Sufficient condenser cooling is provided to change the Freon gas to a liquid and subcool it to keep the liquid from flashing into a gas when it leaves the condenser.

The cooling air leaves the condenser through a Y duct. In flight, the cooling air discharges through the condenser cooling air modulating valve and is dumped overboard. On the ground, the cooling air leaves the condenser through the ground cooling air shutoff valve, passes through the fan, and empties into the main landing gear well. The modulating valves are regulated automatically to maintain the correct Freon temperature.



7. HEAT EXCHANGER
8. CONDITIONED AIR CHECK VALVE
9. FREON EVAPORATOR
10. CONDITIONED AIR EMERGENCY SHUTOFF VALVE
23. FREON CONDENSER
24. FREON EXPANSION VALVE
25. BACK PRESSURE REGULATOR VALVE
26. MOTOR COOLING EXPANSION VALVE
30. CONDENSER GROUND COOLING AIR SHUTOFF VALVE
31. CONDENSER COOLING AIR MODULATING VALVE
45. ELECTRONIC TEMPERATURE CONTROL
55. AIR FLOW SENSOR AND TRANSMITTER
63. ELECTRIC HEATER
64. FREON COMPRESSOR
65. CONDENSER FAN
115. WATER SEPARATOR
203. FREON DRIER AND STRAINER
204. FREON COMPRESSOR SURGE VALVE
205. THERMAL RESISTOR
208. TEMPERATURE SELECTOR
209. CONDENSER TEMPERATURE CONTROL
211. SEQUENCING DEVICE
221. FREON COMPRESSOR SURGE CONTROL
225. BELLOWS CONTROL
229. RELIEF VALVE
230. FREON LEVEL GAGE
250. MASS FLOW INDICATOR
253. 160° SWITCH
259. ELECTRONIC BACK PRESSURE CONTROL
261. CURRENT TRANSFORMER
264. LOCKOUT RELAY
272. PRESSURE SWITCH
277. THERMAL RESISTOR

22.35.21.223A

Freon Refrigeration System Schematic
Figure 14-6

A liquid level gage provides a visual indication of the amount of Freon in the condenser. For an accurate check, the reading should be taken with the Freon package operating and with the condenser as cold as possible. A spring-loaded relief valve on the top of the condenser will open if the pressure in the condenser reaches from 170 to 180 psig. This pressure is well above the normal operating range.

A Freon drier and strainer unit with a replaceable cartridge is located at the condenser outlet. The cartridge contains a chemical drying agent (desiccant) surrounded by a filter screen. All Freon leaving the condenser is strained by the filter screen to prevent the circulation of foreign particles through the system. A portion of the Freon also comes in contact with the desiccant which removes any moisture and prevents corrosion within the system.

Freon Evaporator

From the condenser, the cold liquid Freon flows through a thermostatic expansion valve to the evaporator where it is completely vaporized before it returns to the compressor to complete the vapor cycle. The heat to "boil" the liquid Freon is extracted from the incoming cabin air.

The evaporator is a Freon-to-air heat exchanger made up of an outer shell and two headers connected by a core containing tubes. The Freon flows through the tubes while the cabin air from the turbocompressor passes through the spaces between the tubes and the outer shell. As the cabin air is cooled to the dew point, the moisture in the air condenses on the core surfaces in the form of small water droplets. A fine mesh screen separates the small droplets from the cabin air. The water drains into a water collector where cabin air pressure forces it through a spray tube which sprays the water over the ram air inlet of the condenser core.

The thermostatic expansion valve in the Freon line from the condenser regulates the flow of Freon into the evaporator and the degree of superheat at the evaporator outlet. The operation of this valve is controlled by a temperature sensing element in the outlet header of the evaporator. The flow of Freon into the evaporator is restricted to a rate that will assure complete vaporization and additional heating of the vapor before it leaves the evaporator. This valve also prevents liquid Freon from flooding the evaporator when the Freon package is shut down.

The boiling temperature of the liquid Freon depends on the vapor pressure within the evaporator. The cooling provided by the evaporator is regulated by controlling the vapor pressure within the evaporator through the action of a back-pressure valve located in the Freon line between the evaporator and the compressor. The action of this valve is controlled automatically by the cabin temperature control system.

Motor Cooling Loop

A second Freon loop is provided to cool the motor and compressor. Liquid Freon from the condenser outlet passes through a thermostatic expansion valve into the section of the compressor housing enclosing the electric motor. From there it empties into the compressor inlet. This valve is controlled by a unit which senses the temperature of the Freon after it leaves the motor housing. In this loop, the liquid Freon is vaporized by heat extracted from the cabin air.

Surge Control

A third Freon loop connecting the compressor outlet with the compressor inlet controls pressure surging in the compressor. Freon flow through this loop is controlled by a surge valve. In case of limited weight-flow accompanied by an excessive pressure differential through the compressor, the surge valve will open and some of the Freon will be recirculated. The increase in turbine temperature due to recirculating the Freon causes the motor-cooling thermostatic valve to allow more Freon to circulate through the motor cooling loop. Thus, during times of Freon recirculation through the surge valve, the compressor, motor is overcooled to limit the Freon temperature at the compressor outlet.

FLIGHT DECK ELECTRIC HEATER

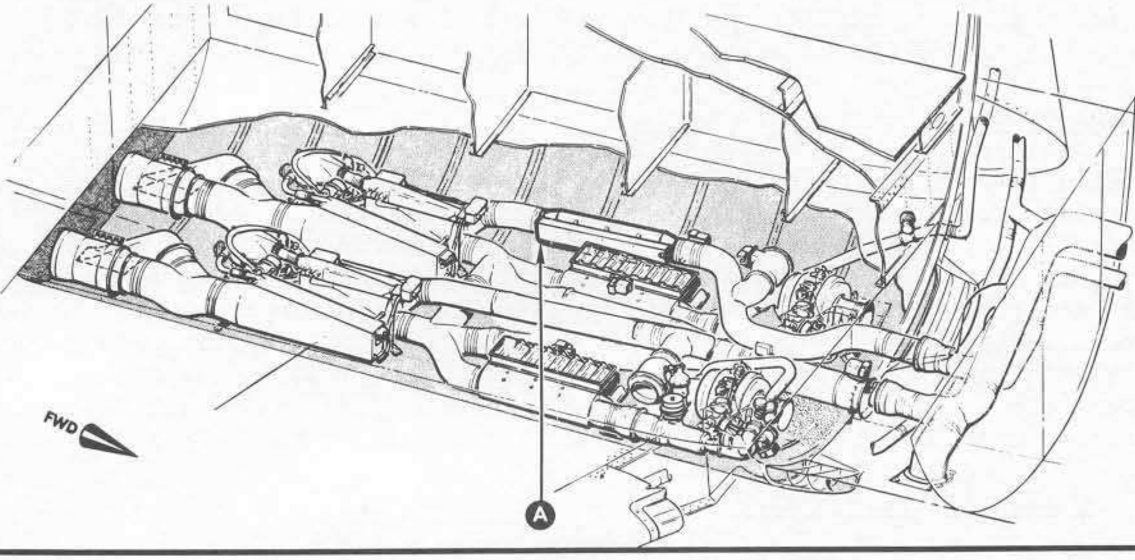
The flight deck electric heater is an expanded section of supply ducting containing seven heating elements and an overheat switch. The heater terminals are connected to 115/200-volt, 400-cycle, 3-phase, ac power through contacts of the heater relays. The overheat switch, consisting of three normally closed contacts in series around the three phases of the first heater element, connects the relay coil to ground through contacts of the recirculation fan motor relay. The heating elements are relay energized in seven increments, to provide modulated heating from zero to full capacity when heating of the recirculated air is scheduled. When low flow requirements, coupled with high scheduled heating, cause excessive element heating (to 350 degrees F 177°C) or if the recirculation fan is turned off, the duct heater will be inoperative, since actuating either the thermal switch or fan motor relay will interrupt the 28-volt dc relay ground removing power from the elements. The thermal protective switch will automatically reset when temperatures fall below 350 degrees F (177°C). The heater can only be used on the ground. (See Figure 14-7.)

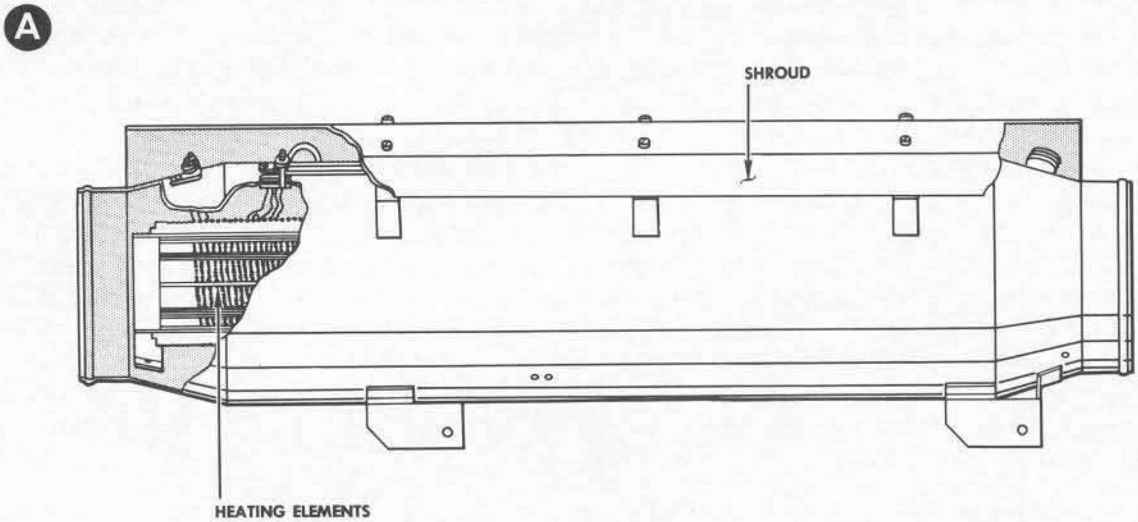
CABIN ELECTRIC HEATERS

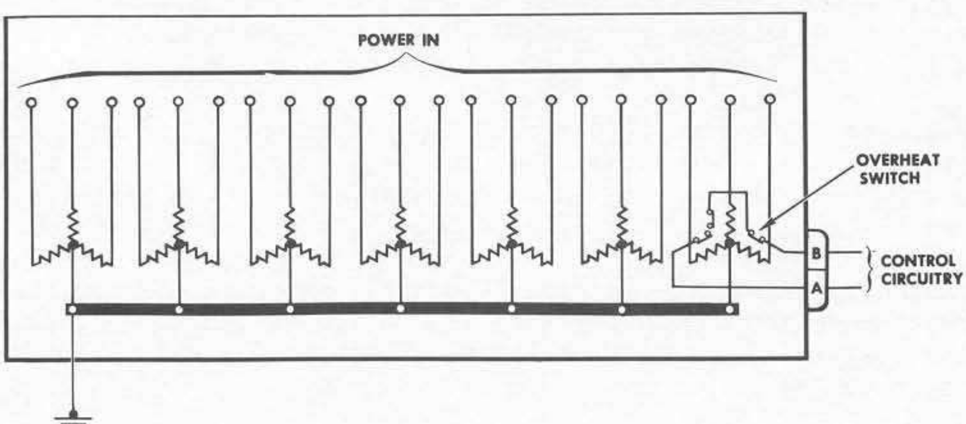
Two electrical heaters, located in the cabin overhead, heat conditioned air to a selected temperature prior to distribution in the cabin. The heaters are controlled by the cabin differential temperature control system.

Each heater is divided into two parallel sections. Air enters the heater from the forward side then passes through ducting to the forward and aft cabins. Bare wire heating elements in each heater, are wye-connected for 3-phase ac power. Each element is capable of heating 41 pounds of air per minute.

Two normally closed thermal switches are located in each heater. If the air temperature exceeds 130 degrees F (72°C) the switches open, breaking continuity to the heater elements. When the temperature goes below 130 degrees F (72°C), the switches close.







22.35.21.236 B

Flight Deck Electric Heater
Figure 14-7

Cabin Differential Temperature Control System

A maximum temperature differential of 2 degrees F (1.1°C) is maintained between the forward and aft cabin sections. Each of the heaters is controlled by the temperature differential control system. The cabin differential control system consists of the cabin temperature control panel, ground modulating power supply, modulating power supply, temperature differential controller, temperature sensors and related circuitry.

After the desired cabin temperature has been selected on the flight engineer's air conditioning panel, conditioned air from the under-wing turbocompressors enters ducting in the cabin overhead section. The air passes through the two heaters, and is ducted to the forward and aft cabin sections. The heater discharge air temperature and the cabin discharge air temperature are each monitored by sensors in both the forward and aft cabin sections. Signals from the sensors are sent to the temperature differential controller which compares the signals and schedules the system to supply warmer air to the colder cabin.

CABIN AIR RECIRCULATING SYSTEM

The recirculating system is used primarily to provide cabin ventilation when the airplane is on the ground and when taxiing on a hot day. In flight, the system can be used as a supplemental source of air when one of the turbocompressors is shut down. The system is connected to the flight deck and cabin air conditioning systems through the conditioned air crossover duct.

The recirculating system consists essentially of a two stage fan driven by a 400-cycle, 115/200-volt ac electric motor and a cabin air control valve. The fan is controlled through an AUTO-OFF-MAN toggle switch on the flight engineer's panel.

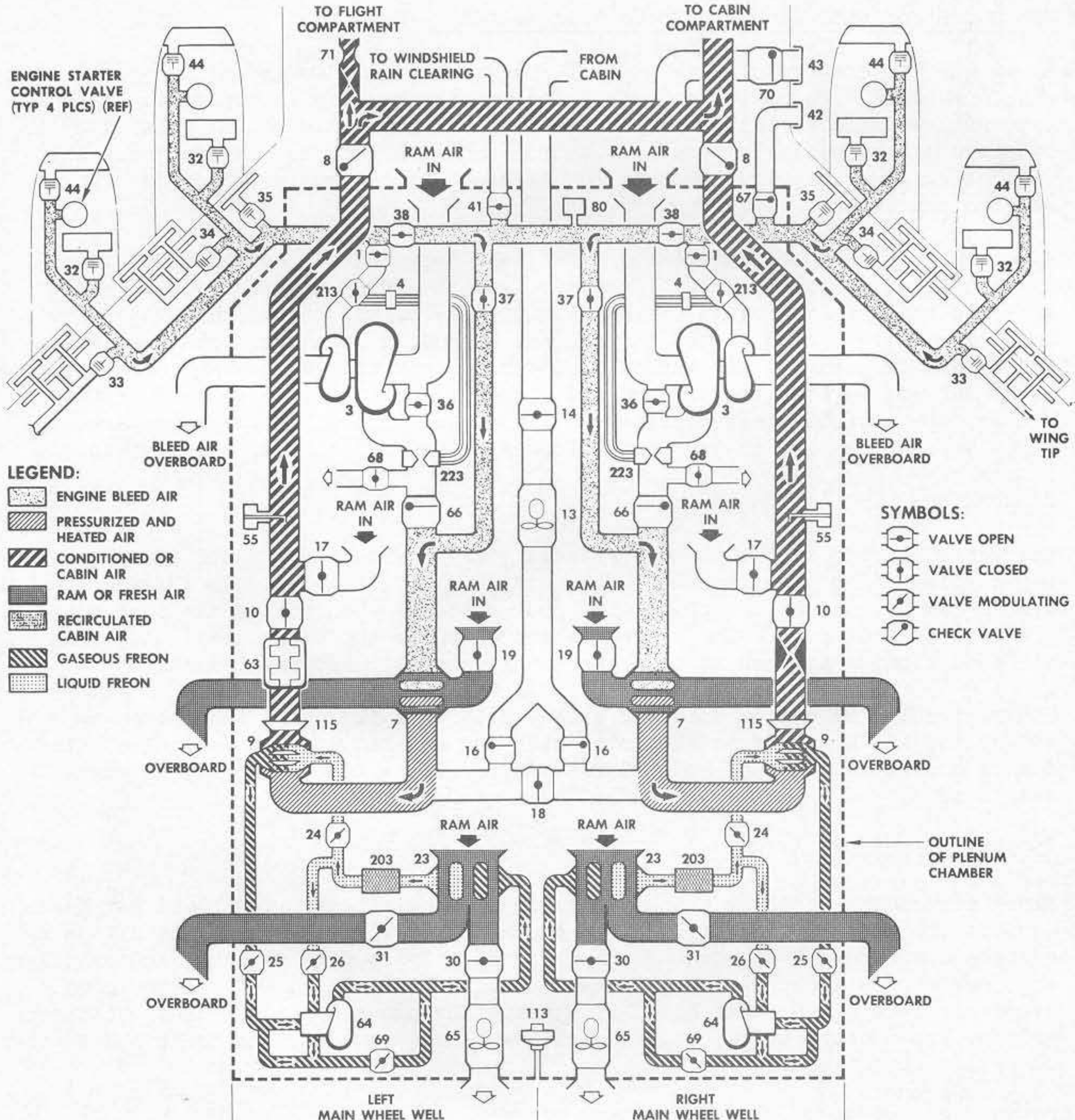
ALTERNATE PRESSURIZATION

Bleed air can be used for alternate cabin pressurization. Two bleed air lines connect the bleed air manifold with the cabin and flight deck air conditioning systems downstream from the turbocompressors. Two toggle switches on the flight engineer's panel control the alternate flow control and shutoff valve in each bleed air line. Since the bleed air passes through the primary heat exchangers and the Freon evaporators, its temperature is regulated by the cabin temperature control system (see Figure 14-8).

CABIN PRESSURE REGULATOR AND OUTFLOW VALVE

Two cabin pressure regulator and outflow valves maintain the desired pressure level within the airplane. The two valves control the pressurization level, or cabin altitude, by regulating the amount of air exahusted from the pressurized compartment. They also serve as pressure relief, vacuum relief, and dump valves (see Figure 14-9).

The forward outflow valve is located on the floor of the electrical compartment, and the aft outflow valve is located below the aft lavatory. Both valves are mounted on short transition ducts which adapt the circular valve outlet to

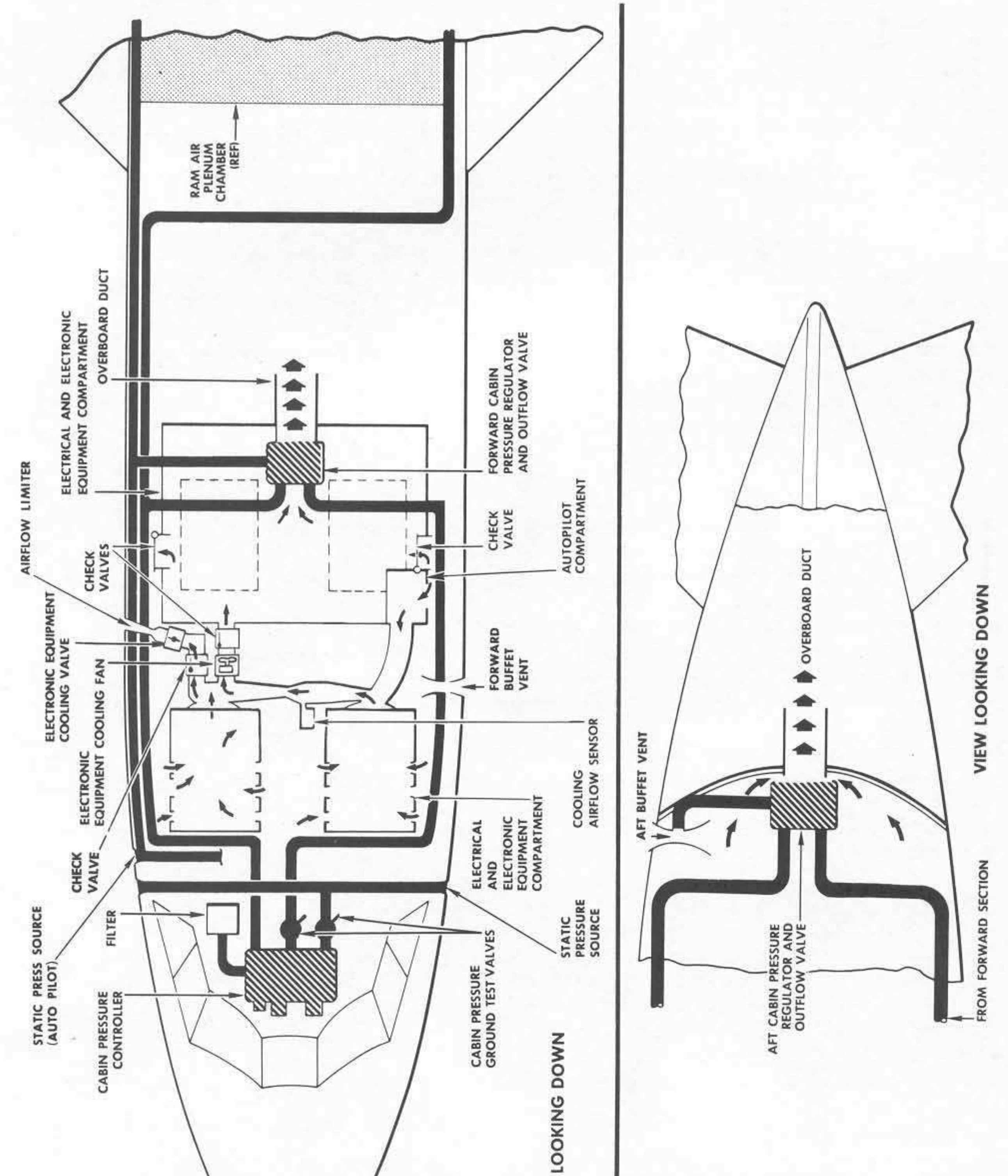


CODE:

1. TURBOCOMPRESSOR SHUTOFF VALVE
3. TURBOCOMPRESSOR
4. TURBOCOMPRESSOR FLOW CONTROL
7. HEAT EXCHANGER
8. CONDITIONED AIR CHECK VALVE
9. FREON EVAPORATOR
10. CONDITIONED AIR EMERGENCY SHUTOFF VALVE
13. RECIRCULATION FAN
14. RECIRCULATION CONTROL VALVE
16. RECIRCULATION AIR CHECK VALVE
17. RAM AIR SHUTOFF VALVE
18. CROSSOVER SHUTOFF VALVE
19. HEAT EXCHANGER COOLING AIR MODULATING VALVE
23. FREON CONDENSER
24. FREON EXPANSION VALVE
25. BACK PRESSURE REGULATOR VALVE
26. MOTOR COOLING EXPANSION VALVE
30. CONDENSER GROUND COOLING AIR SHUTOFF VALVE
31. CONDENSER COOLING AIR MODULATING VALVE
32. ENGINE BLEED AIR VALVE
33. OUTBOARD LEADING EDGE ANTI-ICING VALVE
34. MIDSECTION LEADING EDGE ANTI-ICING VALVE
35. INBOARD LEADING EDGE ANTI-ICING VALVE
36. HEAT CONTROL MODULATING VALVE
37. EMERGENCY CABIN AIR FLOW VALVE
38. BLEED AIR EMERGENCY WING ISOLATION VALVE
41. WINDSHIELD RAIN REMOVAL VALVE
42. ENGINE START AIR CONNECTION
43. GROUND AIR CONDITIONING CONNECTION
44. ENGINE INLET DUCT LIP ANTI-ICING VALVE
55. AIR FLOW SENSOR AND TRANSMITTER
63. ELECTRIC HEATER—FLIGHT DECK
64. FREON COMPRESSOR
65. CONDENSER FAN
66. TURBOCOMPRESSOR CHECK VALVE
67. ENGINE START CHECK VALVE
68. TURBOCOMPRESSOR SURGE VALVE
69. FREON COMPRESSOR SURGE VALVE
70. GROUND AIR CONDITIONING CHECK VALVE
71. FLIGHT COMPARTMENT AIRFLOW LIMITER
80. BLEED AIR OVERPRESSURE SWITCH
113. PLENUM CHAMBER PRESSURE SWITCH
115. WATER SEPARATOR
203. FREON DRIER AND STRAINER
213. NOZZLE ACTUATOR
223. VENTURI DUCT

22.35.21.250A

Alternate Pressurization System Schematic
Figure 14-8



Pressurization System Control Valves
and Vents
Figure 14-9

a rectangular discharge port in the skin. The two valves are identical except for a coarse screen which is wrapped around the forward valve. The screen will prevent the electrical compartment curtain from being sucked into the valve assembly should the curtain come loose. Primary control of both valves is maintained by the cabin pressure controller on the flight engineer's control panel.

When the PRESSURE REGULATORS control switches are placed in the AUTO position, the operation of the pressure regulator and outflow valves is completely pneumatic. When operated manually, an electric motor is used to position the outflow valve. If the valves are closed electrically the pressure and vacuum relief functions also require electric power. Many fail-safe features are incorporated in the design of the cabin pressure control subsystem. When operating in the automatic mode, a failure of one valve to the closed position will cause no change in pressurization because the other valve has the capacity to regulate the entire outflow. If both valves fail to the closed position, the pressure and vacuum relief functions will still operate. If one or both valves fail to the open position, a cabin pressure limiting device will close the valves enough to maintain a cabin altitude of 13,000 (±2,000) feet.

If two failures occur in one valve, such as a failure in the closed position and a failure of the pressure relief function - the manual controls can override the pneumatic system. Another safety feature is included in the wiring installation to prevent accidental dumping of cabin pressurization by a short circuit. The "manual open" control circuit is completely shielded with a grounded metallic sheath. The sheath will ground any "hot" wires that might come in contact with this critical wiring, and will actuate the appropriate overload protection to remove the power. Terminal connections on the manual control switches are "potted" to further reduce the possibility of trouble.

If the aft outflow valve should fail in the closed position, an inward-opening check valve allows more air to enter the electrical chamber directly from the cabin. If the forward outflow valve should fail in the closed position, an outward-opening relief valve allows air from the cooling fan to discharge into the cabin.

VALVES

The control and operation of the air conditioning and pressurization system depends primarily on the functioning of various valves located throughout the system. Check valves are placed at strategic points in the system to prevent reverse flow; modulating valves regulate the rate of flow through the passages in which they are located; and shutoff valves are utilized to start and stop certain operations. Some of the modulating valves also serve as shutoff valves.

Check Valves

A check valve is located between each turbocompressor and corresponding emergency bleed air inlet to prevent reverse flow through the turbocompressors when the bleed air is turned on. A second pair of check valves is located in the ducts leading to the flight deck and the cabin, upstream from the crossover passage connecting the two air conditioning systems. When one of the systems has been shut down, the corresponding valves will close and keep the conditioned air from entering the inoperative system. This also prevents loss of pressurization should a duct break occur in the plenum area. Another pair of check valves isolates the recirculating blower from the two air conditioning systems when the blower is not operating. These check valves also prevent engine bleed air from entering the recirculating blower housing when the alternate pressurization system is actuated.

Modulating Valves

The speed of each turbocompressor is regulated by a variable-area nozzle functioning as a modulating valve to control the flow of bleed air through the turbine. The recirculation of compressed air through the compressors is regulated by the cabin heat control modulating valves. Two modulating valves regulate the flow of cooling ram air through the primary heat exchangers and an additional pair of modulating valves controls the flow of ram air through the Freon condensers.

The air supply for the recirculating blower is controlled by a shutoff valve in the cabin air recirculating duct. This valve also functions as a shutoff valve to close the passage when not in use. The cabin air outflow valves modulate the rate at which the cabin air is released to the atmosphere and close the outflow openings when conditions warrant.

Four modulating valves in each Freon package regulate the flow of Freon through the three Freon loops. The Freon back pressure valve controls the cooling output of the unit; one expansion valve controls the flow of liquid Freon to the evaporator, another expansion valve controls the flow of liquid Freon through the compressor bearings for cooling and lubrication, and a fourth modulating valve regulates the recirculation of Freon gas through the compressor.

Shutoff Valves

Most of the air conditioning and pressurization equipment is turned on and off by shutoff valves. The turbocompressors are turned on by opening the turbocompressor bleed air shutoff valves. Two shutoff valves in the emergency bleed air ducts supply bleed air to the air conditioning systems when both turbocompressors are inoperative. An emergency shutoff valve in each system can be used to close either the conditioned air supply duct leading to the cabin or the one leading to the flight deck. Two motorized check valves, one in each ram air inlet, can be opened to admit ram air directly to the flight deck and the cabin. These valves cannot be opened when the cabin pressure is higher than that of the ram air.

SYSTEM OPERATION

Ram air from the plenum chamber is compressed and heated by the compressors. The cabin heat control modulating valves regulate the amount of air that will be recirculated through the compressors for additional heating. The compressed air then passes through check valves and enters the primary heat exchangers where it is cooled, as required, by ram air from the plenum chamber. The amount of cooling is regulated by modulating valves that control the flow of cooling ram air through the heat exchangers.

The compressed air then passes through the Freon evaporators where its temperature may be further reduced by the vaporizing action of the Freon. From the evaporator, the air flows through the electric heater (flight deck system) that can be used to heat the incoming air while the airplane is on the ground.

The conditioned air flows past the airflow sensors and transmitters that provide an indication of turbocompressor, or fan air-weight flow. It then flows through a flow limiter which permits about 10 percent of the air to flow to the flight compartment. The remaining airflow is routed to the cabin.

The recirculating blower is used for air conditioning on the ground and to improve ventilation in flight when one of the turbocompressors has been shut down. The outflow valves allow the air to escape from the cabin at a rate that will maintain the scheduled cabin pressure.

AIR CONDITIONING AND PRESSURIZATION SYSTEM CONTROLS

The instruments and controls for monitoring and regulating the air conditioning and pressurization systems are located on the flight engineer's panel (see Figure 14-10).

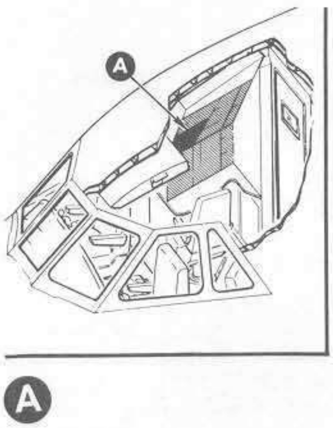
Instruments

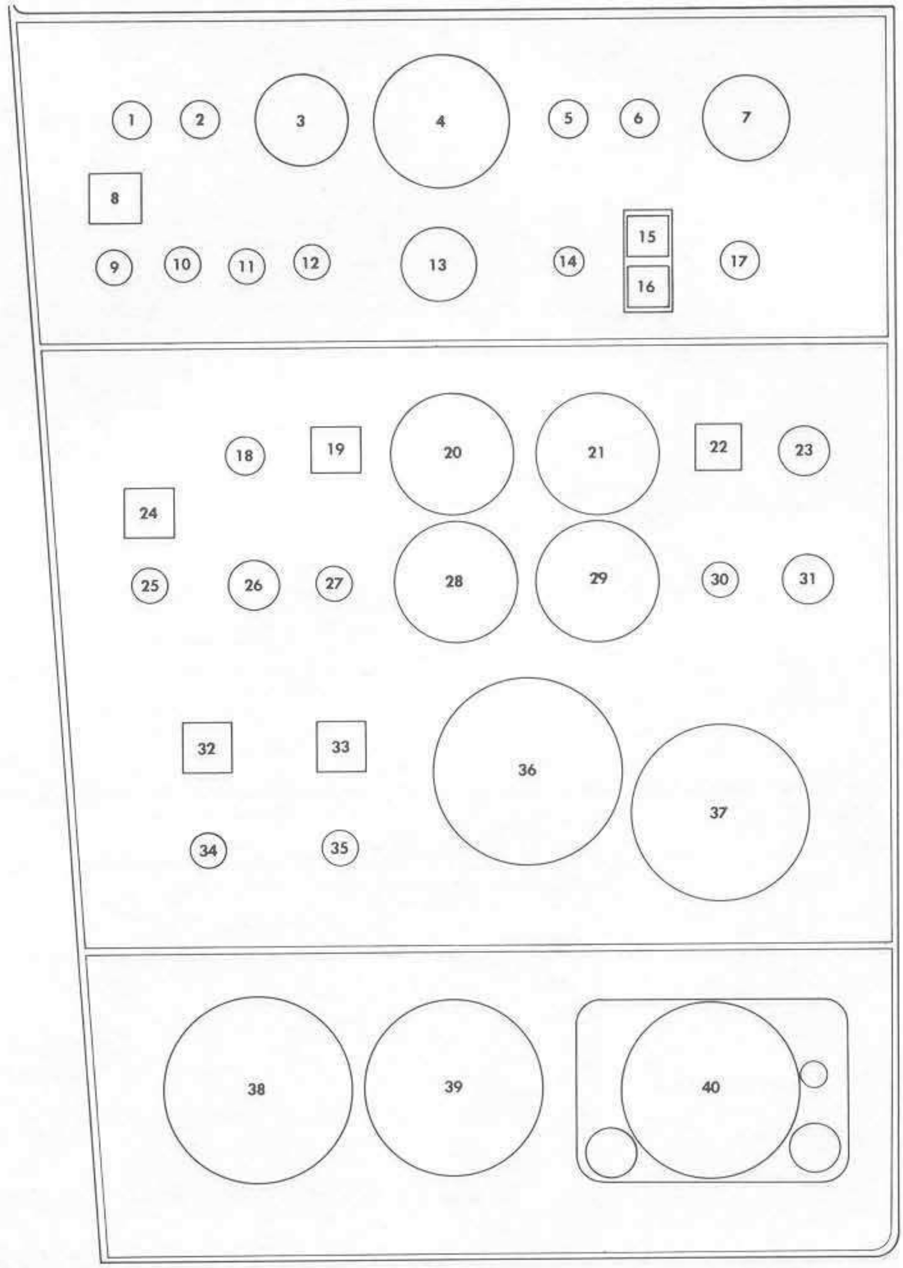
Two COMPR RPM gages indicate the speed of the turbocompressors. Two COMP BEARING TEMP gages indicate the temperature of the turbocompressor bearings. A dual COMPRESSOR AIR FLOW gage registers the air flow from each air conditioning unit in lbs/min. A CABIN TEMP gage provides an indication of cabin temperature and, by use of a rotary selector, the gage can also be used to monitor the operation of the Freon packages. Placing the selector in either the LH or RH FREON EVAP position provides an indication of the Freon evaporator discharge air temperature for the respective package.

A CABIN ALTIMETER and CABIN DIFFERENTIAL PRESSURE indicator, a cabin CLIMB indicator and a CABIN PRESSURE CONTROL are located in the lower half of the control panel.

The cabin pressure controller (see Figure 14-11) is used in conjunction with two cabin pressure regulating outflow valves to control cabin pressure. It controls the cabin pressure regulator and outflow valves by varying a reference pressure transmitted to the pneumatic relay in each valve.

The cabin pressure controller is located on the lower right side of the flight engineer's air conditioning and pressurization control panel. Three indicators and three control knobs are incorporated in the controller. The main dial and pointer indicate the selected cabin altitude within the range of minus 1,000 to plus 10,000 feet. The small inner dial at the top indicates the maximum flight altitude at which the selected cabin altitude can be maintained. The dial in the upper right side is the barometric correction scale, and is calibrated in inches of mercury. The three control knobs are the cabin altitude control, the barometric correction control, and the rate control.





EFFECTIVITY	
APPLICABLE TO AIRPLANES	
*	CV-1 THROUGH CV-25.
**	CV-35 AND ON.

1. FLIGHT DECK MANUAL TEMPERATURE CONTROL SWITCH
2. FLIGHT DECK AUTOMATIC TEMPERATURE CONTROL SWITCH
3. FLIGHT DECK TEMPERATURE SELECTOR
4. TEMPERATURE INDICATOR
5. CABIN MANUAL TEMPERATURE CONTROL SWITCH
6. CABIN AUTOMATIC TEMPERATURE CONTROL SWITCH
7. CABIN TEMPERATURE SELECTOR
8. ELECTRONIC COMPARTMENT COOLING LOW-AIRFLOW WARNING LIGHT
9. ELECTRONIC EQUIPMENT COOLING FAN SWITCH
10. ELECTRONIC COMPARTMENT COOLING VALVE SWITCH
11. CABIN/FLIGHT DECK FREON RESET SWITCH
12. CABIN TEMPERATURE EQUALIZATION CONTROL SWITCH
13. TEMPERATURE INDICATOR SELECTOR
14. FREON MASTER CONTROL SWITCH
15. FLIGHT DECK FREON COMPRESSOR FAIL WARNING LIGHT
16. CABIN FREON COMPRESSOR FAIL WARNING LIGHT
17. CABIN-BOTH-FLIGHT DECK FREON PACK CONTROL SWITCH
18. RECIRCULATION AIR BLOWER CONTROL SWITCH
19. FLIGHT DECK TURBOCOMPRESSOR OVERSPEED WARNING LIGHT
20. FLIGHT DECK TURBOCOMPRESSOR R. P. M. INDICATOR
21. CABIN TURBOCOMPRESSOR R. P. M. INDICATOR
22. CABIN TURBOCOMPRESSOR OVERSPEED WARNING LIGHT
23. RAM AIR VALVES CONTROL SWITCH
24. CABIN HIGH-ALTITUDE WARNING LIGHT
25. CABIN HIGH ALTITUDE WARNING HORN CUTOFF SWITCH
26. FLIGHT DECK EMERGENCY PRESSURE SOURCE SWITCH
27. FLIGHT DECK TURBOCOMPRESSOR CONTROL SWITCH
28. FLIGHT DECK TURBOCOMPRESSOR BEARING TEMPERATURE INDICATOR
29. CABIN TURBOCOMPRESSOR BEARING TEMPERATURE INDICATOR
30. CABIN TURBOCOMPRESSOR CONTROL SWITCH
31. CABIN EMERGENCY PRESSURE SOURCE SWITCH
32. FORWARD OUTFLOW VALVE VALVE-CLOSED WARNING LIGHT
33. AFT OUTFLOW VALVE VALVE-CLOSED WARNING LIGHT
34. FORWARD OUTFLOW VALVE CONTROL SWITCH
35. AFT OUTFLOW VALVE CONTROL SWITCH
36. CABIN/FLIGHT DECK TURBOCOMPRESSOR AIRFLOW INDICATOR

*37. CABIN DIFFERENTIAL PRESSURE INDICATOR

**37. CABIN DIFFERENTIAL PRESSURE AND CABIN ALTIMETER INDICATOR

*38. CABIN RATE-OF-CLIMB INDICATOR

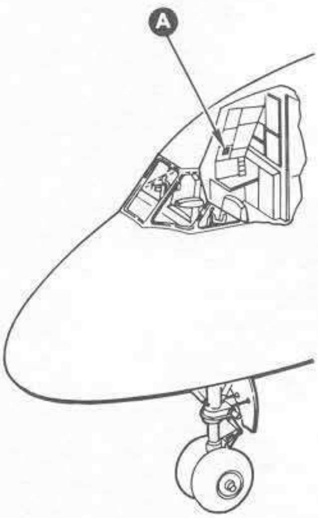
*39. CABIN ALTIMETER

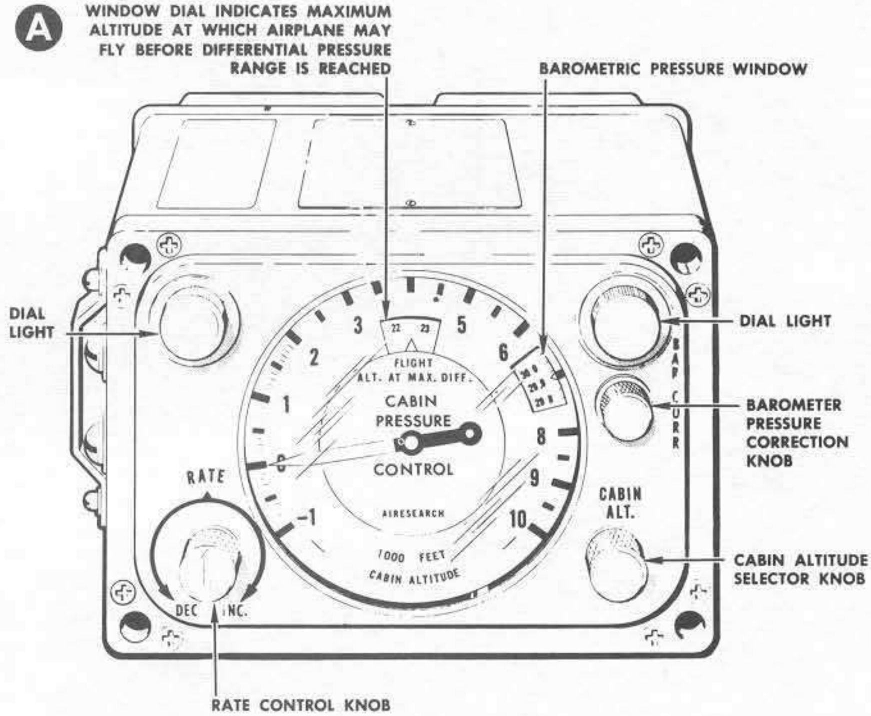
**39. CABIN RATE-OF-CLIMB INDICATOR

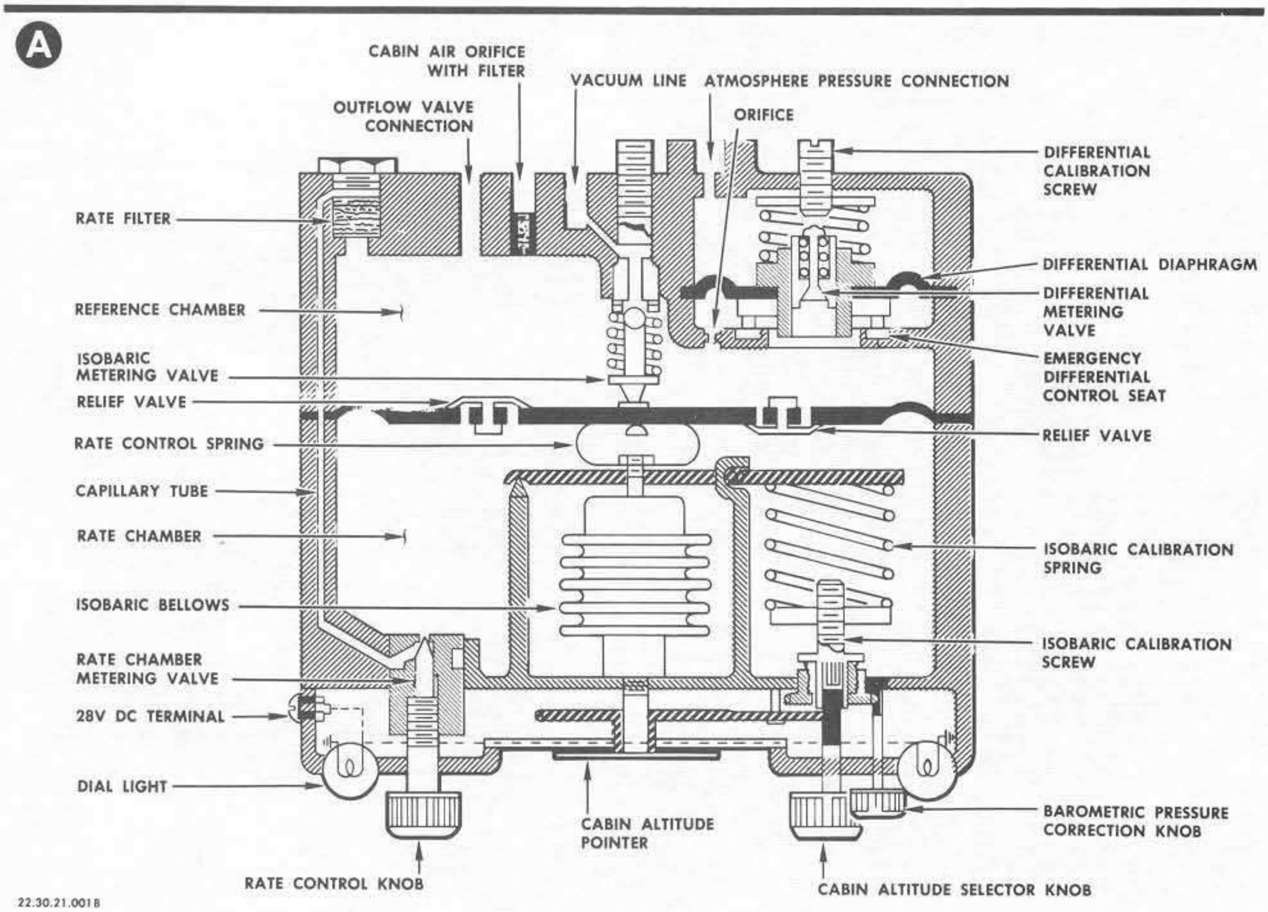
40. CABIN PRESSURE CONTROLLER (INCLUDES CABIN ALTITUDE CONTROL, RATE OF CLIMB OR DESCENT CONTROL, AND BAROMETRIC PRESSURE CONTROL)

30.02.21.106A

Air Conditioning and Pressurization
System Control Panel
Figure 14-10







Cabin Pressure Controller
Figure 14-11

The cabin altitude control knob is used to set the desired cabin altitude on the main dial. The barometric control is set at the local altimeter setting prior to take off or landing. The rate control knob is used to establish the rate of climb or descent in cabin altitude. The rate control knob can be set to maintain any rate between 50 and 2,000 feet per minute.

The cabin pressure controller has three separate pressure controlling systems or modes of operation. They are a selective pressure rate of change, a variable isobaric, and a fixed differential pressure system. These systems operate one at a time to establish the reference pressure, and the transition from one to another is automatic.

A. Selective Pressure Rate-of-Change Control.

The selective pressure rate-of-change control system consists of the rate control spring, a rate control diaphragm which separates the housing into two chambers, and a connecting drilled passage with filter, capillary tube, and rate chamber metering valve.

B. Isobaric Control.

The variable isobaric control system consists of an evacuated isobaric bellows and rocker arm which are connected to an isobaric metering valve through a rate control spring. An isobaric calibration spring and isobaric calibration screw are secured to the rocket arm, and are indexed to a cabin altitude pointer through a gear train.

C. Fixed Differential Control.

The fixed differential control system consists of a spring-loaded differential diaphragm, a differential metering valve and a differential calibration screw. One side of the differential diaphragm is vented to atmospheric pressure and the other side of the differential diaphragm is exposed to reference pressure. This system also incorporates a fail safe feature consisting of an emergency differential control seat and orifice.

D. Barometric Pressure Correction.

The barometric pressure correction mechanism consists of a barometric pressure correction knob, a gear train linked to the variable isobaric control spring, and a window dial calibrated over a range of 28.0 to 31.0 inches of mercury absolute.

E. Control Operation - Rate of Change.

Before control operation, pressures on both sides of the rate control diaphragm are equalized at existing atmospheric pressure. This compresses the isobaric bellows and opens the isobaric metering valve. Cabin air then flows into the reference chamber through the cabin air filter and orifice, and out through the isobaric metering valve to atmosphere. Pressure in the reference chamber is sensed by the outflow valve, which then operates according to the cabin-to-airplane altitude graph, see Section 21-0, to produce an essentially unpressurized cabin, provided the rate of ascent of the aircraft does not exceed the selected rate. If the airplane rate of climb exceeds the selected rate, reference pressure is reduced, resulting in a pressure differential across the rate control diaphragm because of the capillary tube and rate chamber metering valve position. The rate control diaphragm then moves the isobaric metering valve to limit flow of air from the reference chamber until reference pressure and rate pressure equalize. The outflow valve reacts to control the rate of change of cabin pressure accordingly.

F. Control Operation - Isobaric Range.

As the airplane enters the isobaric operating range, the isobaric bellows expands sufficiently to override the rate control diaphragm and assume control of the isobaric metering valve. With reference chamber pressure maintained essentially constant by the isobaric bellows and metering valve, reference pressure in the outflow valve remains constant. Variations in cabin pressure act directly to position the outflow valve, controlling cabin pressure at selected altitude.

G. Control Operation - Differential Range.

The differential control serves to limit reference pressure in order that the outflow valve will establish an essentially constant pressure differential between cabin and atmosphere. As the airplane enters the differential operating range, the pressure differential between reference chamber and atmosphere allows the differential diaphragm to open the differential metering valve and release reference chamber air to atmosphere. The fail safe feature prevents loss of reference pressure in the event of a ruptured differential diaphragm by sealing the differential diaphragm from the reference chamber, permitting only a small loss of reference chamber air through the orifice. During a rapid descent, the differential metering valve closes because of increased atmospheric pressure. If the resultant rate of increase in reference pressure exceeds the selected rate, the rate control diaphragm moves the isobaric metering valve to a metering position and air from the reference chamber is bled to atmosphere. The outflow valve reacts to the reference pressure by limiting the rate of increase in cabin pressure.

Before landing, the cabin altitude selector is set to the destination airport altitude. The barometric pressure correction, in the dial right window, is then set by turning the barometric pressure correction knob. This rotates the gear train, which changes the tension of the isobaric calibration spring and corrects the unit to barometric pressure at the flight destination.

Air Conditioning and Pressurization System Control Switches

Two turbocompressor ON-OFF switches control the operation of the flight deck and cabin turbocompressors.

Two PRESS SOURCE switches OPEN or CLOSE the bleed air valves for emergency pressurization.

The RECIRCULATING BLOWER is controlled by a single AUTO-OFF-MAN toggle switch.

The cabin temperature equalization control system is actuated by the single CABIN TEMP EQ CONT toggle switch.

The Freon compressor ON-OFF MASTER switch controls the Freon compressors as selected by the three-positioned FREON COMPRESSOR selector switch. This switch placarded FLT DECK OFF-BOTH ON-CABIN OFF, is used for manual selection of the desired Freon compressor operational mode.

The FLT DECK - CABIN FREON RESET switch is utilized for manual recycling of lockout relays which have opened a power circuit due to an overheat or overpressure condition.

A RAM AIR SOURCE locking type toggle switch is used to OPEN or CLOSE the ram air valve.

The electronic equipment cooling fan is controlled by an ON-OFF toggle switch placarded ELEC EQUIP COOL FAN. The electronic compartment cooling valve is controlled by an OPEN-CLOSE toggle switch placarded ELEC COMPT COOL VALVE.

Two PRESSURE REGULATOR switches placarded AUTO-OPEN-CLOSE-OFF provide for manual operation of the two outflow valves. In the AUTO position, cabin pressure is regulated by the cabin pressure controller. The OPEN and CLOSE positions are used when operated under manual control to "inch" the outflow valves toward the open or closed position.

Four toggle switches and two thermostatic controls are located at the top of the control panel. The two AUTO-OFF-MAN three-position switches are used to select the operating mode of the flight deck or cabin air conditioning systems. Two MAN HOT-MAN COLD momentary contact type switches are used to "inch" the temperature setting up or down when operating it in manual mode. Two thermostatic controls are used to regulate the temperature controls of the two systems when operating in the automatic mode.

Air Conditioning and Pressurization System Warning Lights

Eight air conditioning and pressurization system warning lights are located on the flight engineer's panel. Two amber OVERSPEED TRIP warning lights connect to the overspeed controls of the two turbocompressors. If one of the turbocompressors should overspeed and trip the control, the corresponding light will illuminate.

Two amber warning lights, LH FREON FAIL and RH FREON FAIL, connect to the compressor discharge overpressure and overtemperature switches, and motor overheat switch on the Freon compressors. If a compressor shuts down due to the actuation of any of the protection switches, the corresponding warning light will illuminate.

An amber FWD CLOSE and an amber AFT CLOSE light, above the pressure regulator switches provide a warning if one of the cabin outflow valves fails in the closed position. A single red HIGH ALT light illuminates when the pressure altitude in the cabin exceeds 10,000 feet. A warning horn sounds simultaneously with the light. An ALT WARN HORN CUT-OFF switch, placarded PUSH, is provided to shut off the warning horn.

An amber LOW AIR FLOW light above the ELEC EQUIP COOL FAN switch will illuminate if the airflow is too low to provide adequate cooling for the electronic equipment.

AIR CONDITIONING OPERATION

The air conditioning subsystem provides for control of three environmental factors in the airplane; cabin and flight compartment temperatures, replenishing of cabin and flight compartment air, and removal of moisture from the refrigerated air. The air conditioning subsystem is capable of cooling the compressed air, heating the compressed recirculated cabin air, removing moisture and controlling subsystem operation. Conditioned air is heated by two different methods: recirculating pressurized air through the turbocompressor and by electric heating. Air is cooled by two cooling subsystems: an air-to-air heat exchanger and a Freon refrigeration cooling subsystem. Cabin air is recirculated by an electrically-driven recirculation fan, ducting and valves. Moisture is extracted as a part of refrigeration cooling when the Freon refrigeration subsystem is in operation. The system is controlled by the temperature control subsystem.

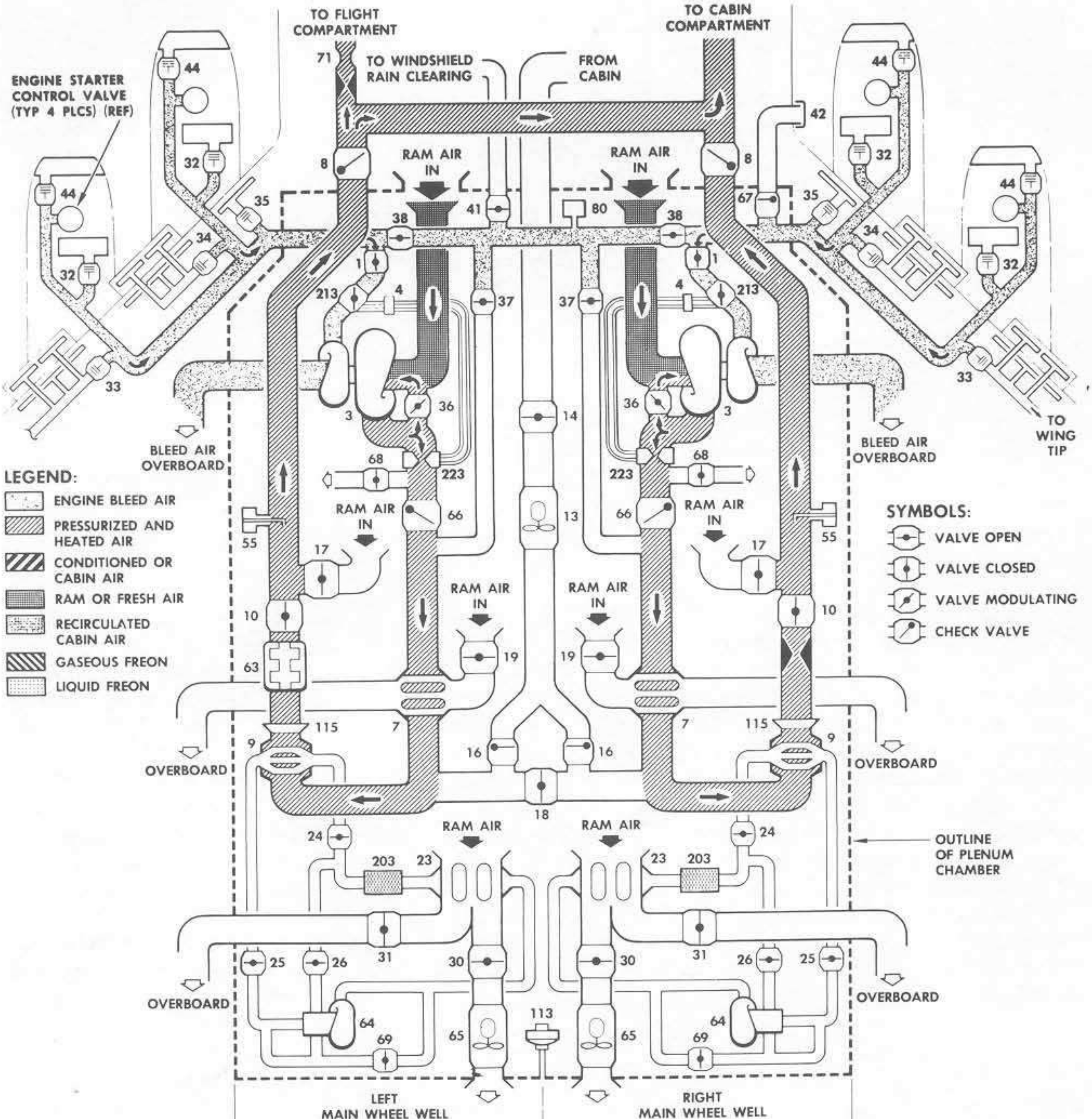
The ram air is compressed and heated in the turbocompressor. It is then cooled, if necessary. While flowing through the air-to-air heat exchanger and the Freon evaporator, and is distributed to the flight and cabin compartments. There are five normal modes of heating or cooling the flight and cabin compartments while the airplane is in flight or on the ground. These modes of operation are described in the following paragraphs.

Inflight Heating

During flight, the recirculation control valve and the flight deck electric heater are not subject to control by the temperature control system; the valve is closed and the flight deck electric heater is off. During flight the dual heaters located in the cabin ducts are used as required, to maintain a 2 degrees F (1.1°C) differential temperature between fore and aft cabin areas.

As shown in Figure 14-12, the turbocompressor shutoff valve is open, allowing engine bleed air to operate the turbocompressor. Ram air from the plenum chamber enters the compressor side of the turbocompressor, where it is compressed and heated. If one pass through the compressor does not heat the air enough, a portion of the air is recirculated through the compressor by the heat control modulating valve. The heated and compressed air then flows through the turbocompressor flow control and the turbocompressor check valve on to the heat exchanger.

The compressed and heated air enters the heat exchanger and flows through without any cooling exchange, because heating is scheduled. Since the heat exchanger cooling air modulating valve is closed, ram air for cooling cannot flow



CODE:

1. TURBOCOMPRESSOR SHUTOFF VALVE
3. TURBOCOMPRESSOR
4. TURBOCOMPRESSOR FLOW CONTROL
7. HEAT EXCHANGER
8. CONDITIONED AIR CHECK VALVE
9. FREON EVAPORATOR
10. CONDITIONED AIR EMERGENCY SHUTOFF VALVE
13. RECIRCULATION FAN
14. RECIRCULATION CONTROL VALVE
16. RECIRCULATION AIR CHECK VALVE
17. RAM AIR SHUTOFF VALVE
18. CROSSOVER SHUTOFF VALVE
19. HEAT EXCHANGER COOLING AIR MODULATING VALVE
23. FREON CONDENSER
24. FREON EXPANSION VALVE
25. BACK PRESSURE REGULATOR VALVE
26. MOTOR COOLING EXPANSION VALVE
30. CONDENSER GROUND COOLING AIR SHUTOFF VALVE
31. CONDENSER COOLING AIR MODULATING VALVE
32. ENGINE BLEED AIR VALVE
33. OUTBOARD LEADING EDGE ANTI-ICING VALVE
34. MIDSECTION LEADING EDGE ANTI-ICING VALVE
35. INBOARD LEADING EDGE ANTI-ICING VALVE
36. HEAT CONTROL MODULATING VALVE
37. EMERGENCY CABIN AIR FLOW VALVE
38. BLEED AIR EMERGENCY WING ISOLATION VALVE
41. WINDSHIELD RAIN REMOVAL VALVE
42. ENGINE START AIR CONNECTION
44. ENGINE INLET DUCT LIP ANTI-ICING VALVE
55. AIR FLOW SENSOR AND TRANSMITTER
63. ELECTRIC HEATER—FLIGHT DECK
64. FREON COMPRESSOR
65. CONDENSER FAN
66. TURBOCOMPRESSOR CHECK VALVE
67. ENGINE START CHECK VALVE
68. TURBOCOMPRESSOR SURGE VALVE
69. FREON COMPRESSOR SURGE VALVE
71. FLIGHT COMPARTMENT AIRFLOW LIMITER
80. BLEED AIR OVERPRESSURE SWITCH
113. PLENUM CHAMBER PRESSURE SWITCH
115. WATER SEPARATOR
203. FREON DRIER AND STRAINER
213. NOZZLE ACTUATOR
223. VENTURI DUCT

30.02.21.124

Air Conditioning System Schematic - Inflight Heating
Figure 14-12

through the heat exchanger. The conditioned air flows from the heat exchanger to the Freon evaporator. As the temperature control system is calling for maximum heat, the Freon evaporator is shut down and the valves in the Freon system are closed. The conditioned air flows through the Freon evaporator, without any cooling, and to the electric heater in the left hand system, (the right hand system heaters are located in the cabin ducting). The conditioned air flows through the electric heater with no added heating, because the heater is inoperative in flight. The conditioned air from the electric heater passes through the conditioned air emergency shutoff valve, which is in the open position, and on to the cabin and flight compartments, through the conditioned air check valves, where it is distributed by the air distribution subsystem. The cabin overhead electric heaters maintain the temperature within 2 degrees F (1.1 degree C) between the fore and aft cabin compartments.

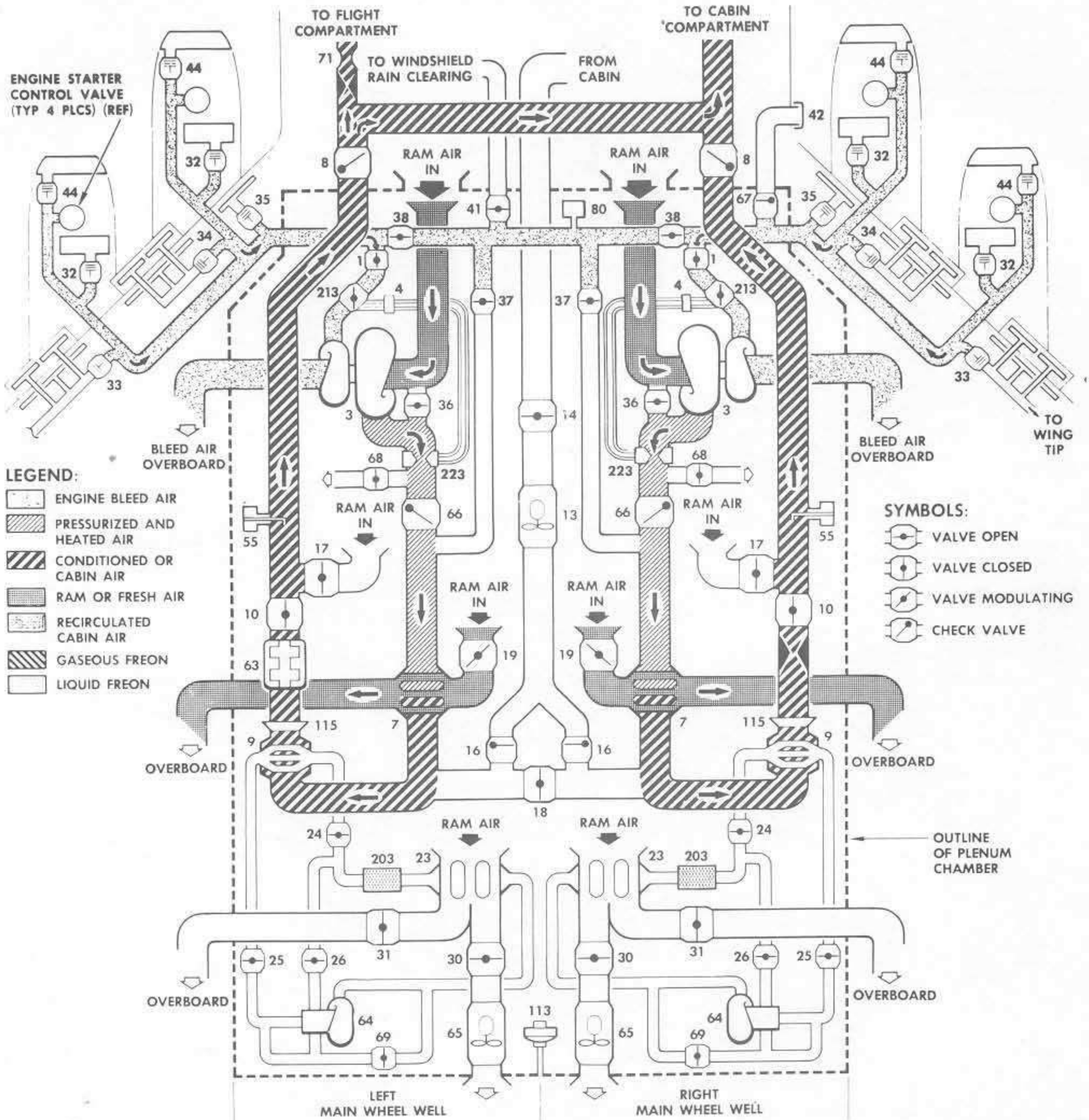
Inflight Intermediate

During the inflight intermediate range, the temperature control system is not calling for additional heating. As shown in Figure 14-13, the heat control modulating valve is closed, indicating that one pass through the compressor is heating the air sufficiently. The compressed air flows through the turbocompressor flow control valve and the turbocompressor check valve, on through to the air-to-air heat exchanger.

The heat exchanger cooling air modulating valve is in the modulating position, indicating that the temperature control system is calling for slightly cooler air than that coming from the turbocompressor. The heat exchanger cooling air modulating valve allows ram air to flow through the air-to-air heat exchanger and be dumped overboard. This ram air extracts heat from the heated air as required by the temperature control system. The compressed and conditioned air leaves the air-to-air heat exchanger and flows through the Freon evaporator. The compressed and conditioned air receives no cooling in the Freon evaporator because sufficient cooling has been obtained in the air-to-air heat exchanger, so the Freon compressor is shut down. The compressed air flows through the flight deck electric heater, which is inoperative during flight, through the conditioned air emergency shutoff valve to the cabin and flight compartment through the conditioned air check valve. The cabin overhead electric heaters maintain the temperature within 2 degrees F (1.1 degree C) between the fore and aft cabin compartments.

Inflight Cooling

When the temperature control system is calling for cooling, the heat control modulating valve is closed. The ram air is compressed in the turbocompressor and partially heated by compression. The compressed air flows through the turbocompressor flow control and the turbocompressor check valve to the heat exchanger. As shown in Figure 14-14, the heat exchanger cooling air modulating valve is full-open, because the temperature control system is calling for more cooling than the air-to-air heat exchanger can provide.

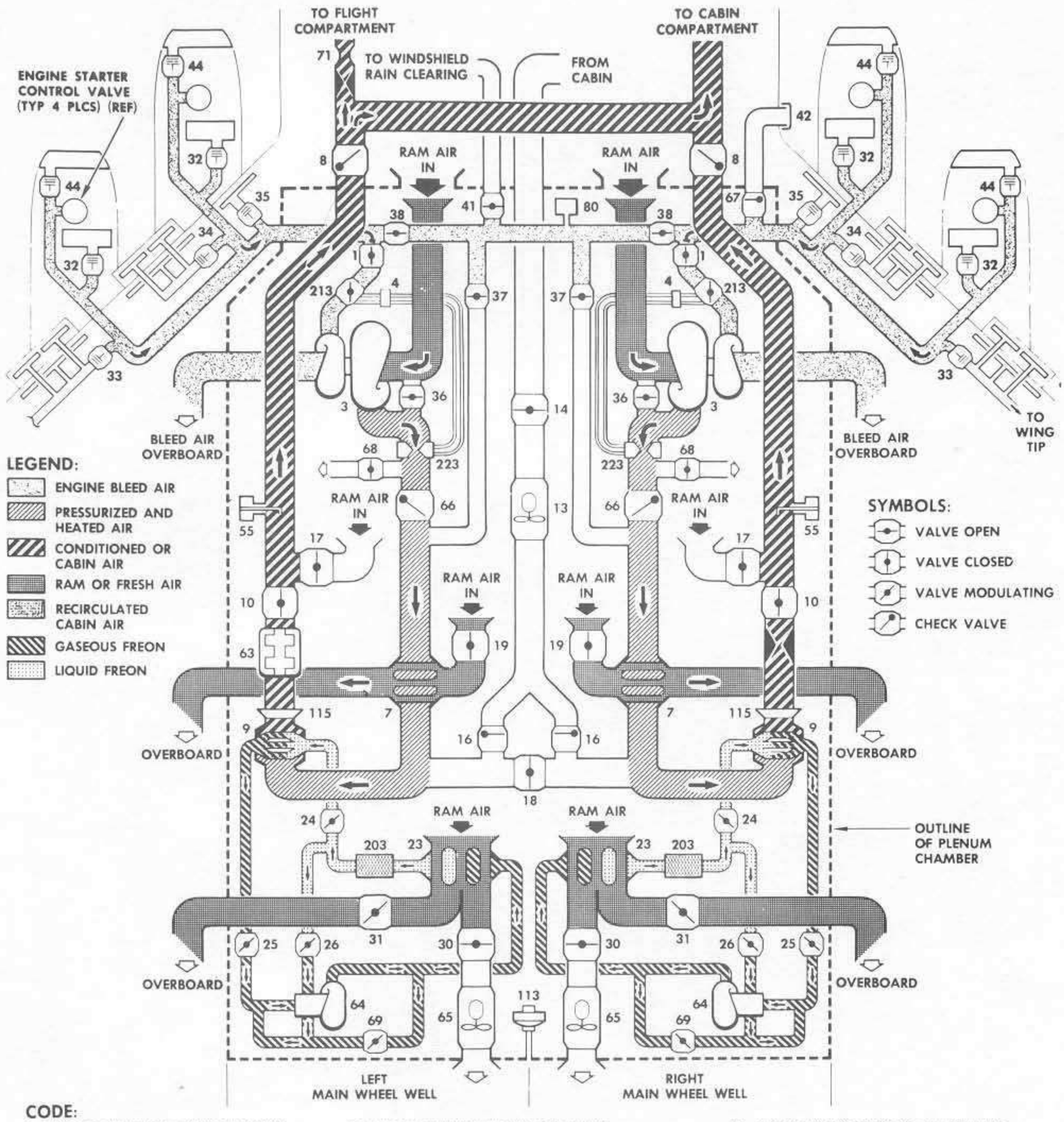


CODE:

1. TURBOCOMPRESSOR SHUTOFF VALVE
3. TURBOCOMPRESSOR
4. TURBOCOMPRESSOR FLOW CONTROL
7. HEAT EXCHANGER
8. CONDITIONED AIR CHECK VALVE
9. FREON EVAPORATOR
10. CONDITIONED AIR EMERGENCY SHUTOFF VALVE
13. RECIRCULATION FAN
14. RECIRCULATION CONTROL VALVE
16. RECIRCULATION AIR CHECK VALVE
17. RAM AIR SHUTOFF VALVE
18. CROSSOVER SHUTOFF VALVE
19. HEAT EXCHANGER COOLING AIR MODULATING VALVE
23. FREON CONDENSER
24. FREON EXPANSION VALVE
25. BACK PRESSURE REGULATOR VALVE
26. MOTOR COOLING EXPANSION VALVE
30. CONDENSER GROUND COOLING AIR SHUTOFF VALVE
31. CONDENSER COOLING AIR MODULATING VALVE
32. ENGINE BLEED AIR VALVE
33. OUTBOARD LEADING EDGE ANTI-ICING VALVE
34. MIDSECTION LEADING EDGE ANTI-ICING VALVE
35. INBOARD LEADING EDGE ANTI-ICING VALVE
36. HEAT CONTROL MODULATING VALVE
37. EMERGENCY CABIN AIR FLOW VALVE
38. BLEED AIR EMERGENCY WING ISOLATION VALVE
41. WINDSHIELD RAIN REMOVAL VALVE
42. ENGINE START AIR CONNECTION
44. ENGINE INLET DUCT LIP ANTI-ICING VALVE
55. AIR FLOW SENSOR AND TRANSMITTER
63. ELECTRIC HEATER—FLIGHT DECK
64. FREON COMPRESSOR
65. CONDENSER FAN
66. TURBOCOMPRESSOR CHECK VALVE
67. ENGINE START CHECK VALVE
68. TURBOCOMPRESSOR SURGE VALVE
69. FREON COMPRESSOR SURGE VALVE
71. FLIGHT COMPARTMENT AIRFLOW LIMITER
80. BLEED AIR OVERPRESSURE SWITCH
113. PLENUM CHAMBER PRESSURE SWITCH
115. WATER SEPARATOR
203. FREON DRIER AND STRAINER
213. NOZZLE ACTUATOR
223. VENTURI DUCT

30.02.21.125

Air Conditioning System Schematic - Inflight Intermediate
Figure 14-13



CODE:

1. TURBOCOMPRESSOR SHUTOFF VALVE
3. TURBOCOMPRESSOR
4. TURBOCOMPRESSOR FLOW CONTROL
7. HEAT EXCHANGER
8. CONDITIONED AIR CHECK VALVE
9. FREON EVAPORATOR
10. CONDITIONED AIR EMERGENCY SHUTOFF VALVE
13. RECIRCULATION FAN
14. RECIRCULATION CONTROL VALVE
16. RECIRCULATION AIR CHECK VALVE
17. RAM AIR SHUTOFF VALVE
18. CROSSOVER SHUTOFF VALVE
19. HEAT EXCHANGER COOLING AIR MODULATING VALVE
23. FREON CONDENSER
24. FREON EXPANSION VALVE
25. BACK PRESSURE REGULATOR VALVE
26. MOTOR COOLING EXPANSION VALVE
30. CONDENSER GROUND COOLING AIR SHUTOFF VALVE
31. CONDENSER COOLING AIR MODULATING VALVE
32. ENGINE BLEED AIR VALVE
33. OUTBOARD LEADING EDGE ANTI-ICING VALVE
34. MIDSECTION LEADING EDGE ANTI-ICING VALVE
35. INBOARD LEADING EDGE ANTI-ICING VALVE
36. HEAT CONTROL MODULATING VALVE
37. EMERGENCY CABIN AIR FLOW VALVE
38. BLEED AIR EMERGENCY WING ISOLATION VALVE
41. WINDSHIELD RAIN REMOVAL VALVE
42. ENGINE START AIR CONNECTION
44. ENGINE INLET DUCT LIP ANTI-ICING VALVE
55. AIR FLOW SENSOR AND TRANSMITTER
63. ELECTRIC HEATER—FLIGHT DECK
64. FREON COMPRESSOR
65. CONDENSER FAN
66. TURBOCOMPRESSOR CHECK VALVE
67. ENGINE START CHECK VALVE
68. TURBOCOMPRESSOR SURGE VALVE
69. FREON COMPRESSOR SURGE VALVE
71. FLIGHT COMPARTMENT AIRFLOW LIMITER
80. BLEED AIR OVERPRESSURE SWITCH
113. PLENUM CHAMBER PRESSURE SWITCH
115. WATER SEPARATOR
203. FREON DRIER AND STRAINER
213. NOZZLE ACTUATOR
223. VENTURI DUCT

30.02.21.126

Air Conditioning System Schematic -
Inflight Cooling
Figure 14-14

The compressed air flows through the air-to-air heat exchanger to the Freon evaporator. Because the heat exchanger is operating at full capacity and the temperature control system is calling for more cooling, the Freon refrigeration system is turned on. When the Freon compressor is operating, the Freon gas is passed through the compressor resulting in an increase in its temperature and pressure.

The compressed Freon gas then flows from the compressor and enters the condenser where the heat is removed by cool ram air blowing across the condenser. As the heat is removed from the gaseous Freon, it condenses into a liquid. The ram air used for cooling is then dumped overboard. The liquid Freon from the condenser enters the evaporator through the Freon expansion valve. Heat is absorbed by the Freon during evaporation. This heat is extracted from the air flowing to the cabin and flight compartment, thus the temperature of the air is reduced. The Freon gas from the evaporator then flows back through the compressor where the cycle is repeated.

Ground Heating

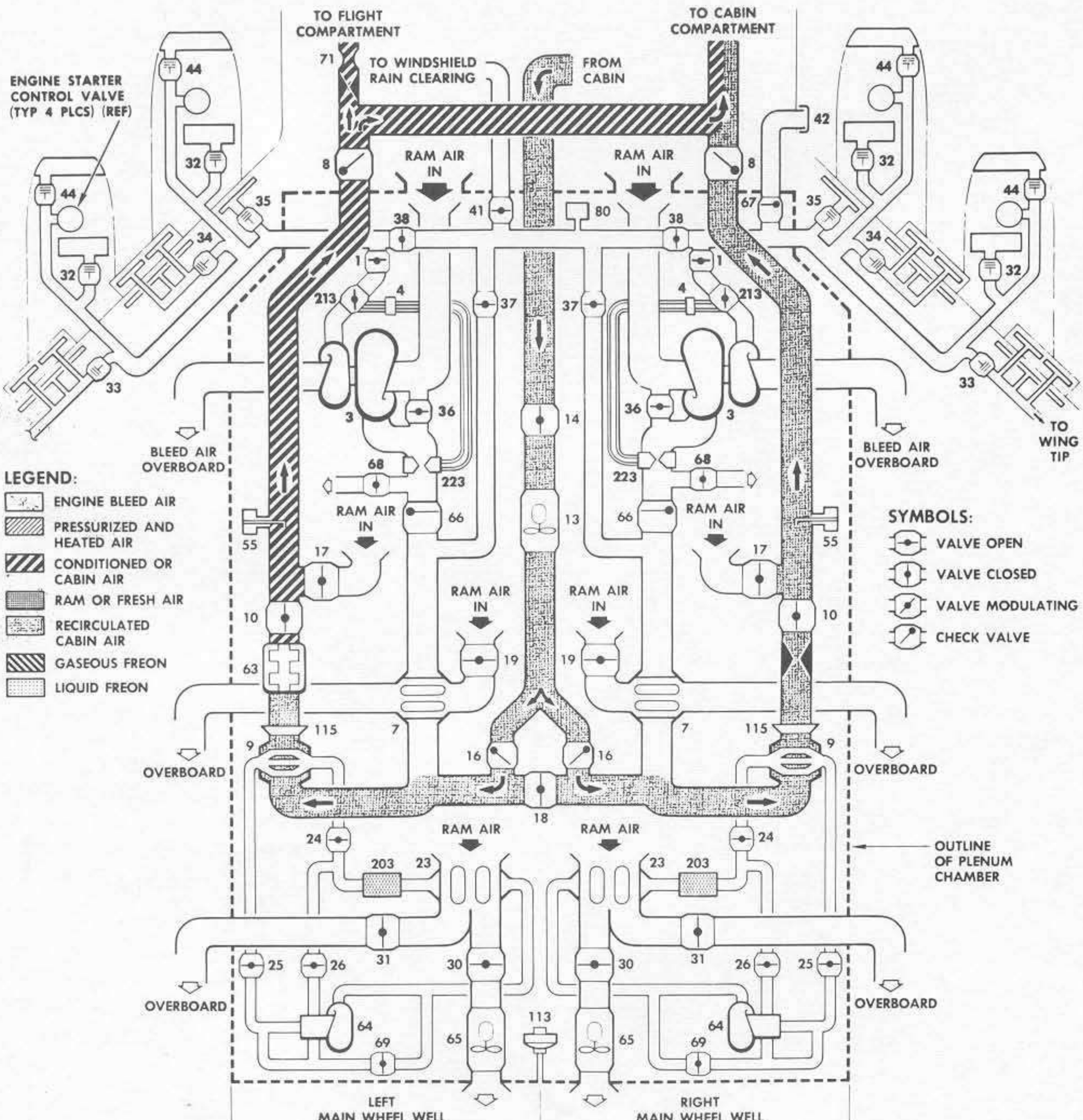
When the airplane is on the ground and the engines are not operating, no bleed air is available and the turbocompressors cannot operate; see Figure 14-15. The air from the cabin is picked up through the recirculation control valve and recirculated by the recirculation fan.

When heating is required, the Freon subsystem is shut down and all Freon subsystem valves are closed. For flight compartment heating, air flows through the Freon evaporator without being cooled, to the heater, where it is heated as required by the temperature control system. From the heater, the air flows through the conditioned air emergency shutoff valve and the conditioned air check valve to the flight compartment. For cabin heating, air flows through the Freon evaporator without being cooled, through the conditioned air emergency shutoff valve and the conditioned air check valve, to the heaters in the cabin ducts. The air is heated as required by the temperature control system and then flows into the cabin.

The flight deck electric heater contains seven electric heater elements. The number of heater elements used is determined by the sequencing device, which in turn is operated by the temperature control system. As more heating is required, the sequencing device will operate more heater elements. If less heating is required, the sequencing device will operate fewer heater elements.

Ground Cooling

With the airplane on the ground, the engines not operating, and the control system calling for maximum cooling, the system works as follows: As shown in Figure 14-16, the turbocompressors are inoperative, the recirculation fan is on, and the recirculation control valve is modulating. The cabin air is moved by the recirculation fan to the Freon evaporator. The temperature control system is calling for more cooling or maximum cooling; consequently, the Freon refrigeration subsystem is operating. The cabin air flows through the Freon evaporator, is cooled as scheduled by the temperature control, and flows to the electric heaters. As the temperature control system is calling for cooling, the electric heaters are inoperative and the cabin air moves through the open

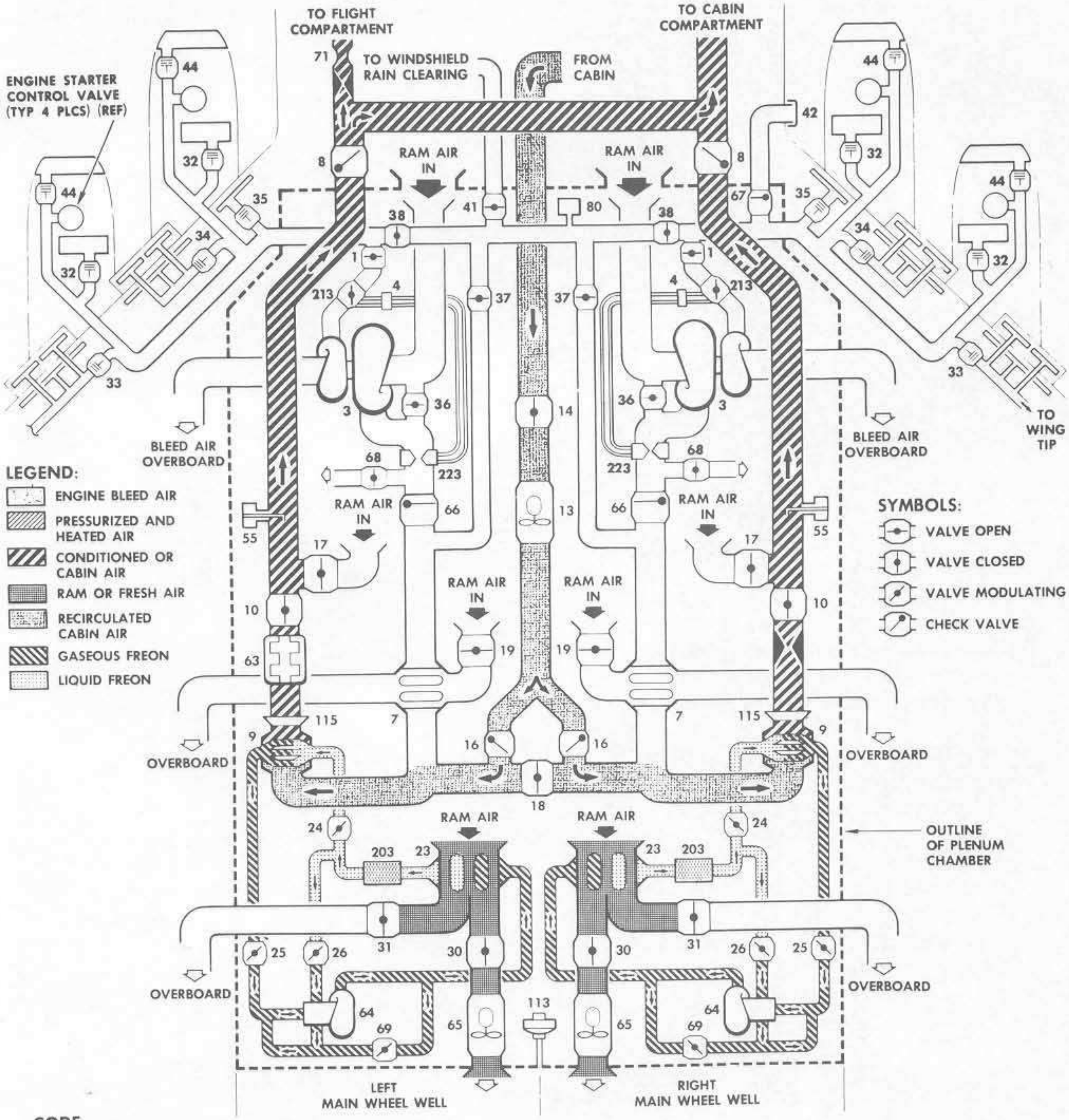


CODE:

1. TURBOCOMPRESSOR SHUTOFF VALVE
3. TURBOCOMPRESSOR
4. TURBOCOMPRESSOR FLOW CONTROL
7. HEAT EXCHANGER
8. CONDITIONED AIR CHECK VALVE
9. FREON EVAPORATOR
10. CONDITIONED AIR EMERGENCY SHUTOFF VALVE
13. RECIRCULATION FAN
14. RECIRCULATION CONTROL VALVE
16. RECIRCULATION AIR CHECK VALVE
17. RAM AIR SHUTOFF VALVE
18. CROSSOVER SHUTOFF VALVE
19. HEAT EXCHANGER COOLING AIR MODULATING VALVE
23. FREON CONDENSER
24. FREON EXPANSION VALVE
25. BACK PRESSURE REGULATOR VALVE
26. MOTOR COOLING EXPANSION VALVE
30. CONDENSER GROUND COOLING AIR SHUTOFF VALVE
31. CONDENSER COOLING AIR MODULATING VALVE
32. ENGINE BLEED AIR VALVE
33. OUTBOARD LEADING EDGE ANTI-ICING VALVE
34. MIDSECTION LEADING EDGE ANTI-ICING VALVE
35. INBOARD LEADING EDGE ANTI-ICING VALVE
36. HEAT CONTROL MODULATING VALVE
37. EMERGENCY CABIN AIR FLOW VALVE
38. BLEED AIR EMERGENCY WING ISOLATION VALVE
41. WINDSHIELD RAIN REMOVAL VALVE
42. ENGINE START AIR CONNECTION
44. ENGINE INLET DUCT LIP ANTI-ICING VALVE
55. AIR FLOW SENSOR AND TRANSMITTER
63. ELECTRIC HEATER—FLIGHT DECK
64. FREON COMPRESSOR
65. CONDENSER FAN
66. TURBOCOMPRESSOR CHECK VALVE
67. ENGINE START CHECK VALVE
68. TURBOCOMPRESSOR SURGE VALVE
69. FREON COMPRESSOR SURGE VALVE
71. FLIGHT COMPARTMENT AIRFLOW LIMITER
80. BLEED AIR OVERPRESSURE SWITCH
113. PLENUM CHAMBER PRESSURE SWITCH
115. WATER SEPARATOR
203. FREON DRIER AND STRAINER
213. NOZZLE ACTUATOR
223. VENTURI DUCT

30.02.21.127

Air Conditioning System Schematic -
Ground Heating
Figure 14-15



CODE:

1. TURBOCOMPRESSOR SHUTOFF VALVE
3. TURBOCOMPRESSOR
4. TURBOCOMPRESSOR FLOW CONTROL
7. HEAT EXCHANGER
8. CONDITIONED AIR CHECK VALVE
9. FREON EVAPORATOR
10. CONDITIONED AIR EMERGENCY SHUTOFF VALVE
13. RECIRCULATION FAN
14. RECIRCULATION CONTROL VALVE
16. RECIRCULATION AIR CHECK VALVE
17. RAM AIR SHUTOFF VALVE
18. CROSSOVER SHUTOFF VALVE
19. HEAT EXCHANGER COOLING AIR MODULATING VALVE
23. FREON CONDENSER
24. FREON EXPANSION VALVE
25. BACK PRESSURE REGULATOR VALVE
26. MOTOR COOLING EXPANSION VALVE
30. CONDENSER GROUND COOLING AIR SHUTOFF VALVE
31. CONDENSER COOLING AIR MODULATING VALVE
32. ENGINE BLEED AIR VALVE
33. OUTBOARD LEADING EDGE ANTI-ICING VALVE
34. MIDSECTION LEADING EDGE ANTI-ICING VALVE
35. INBOARD LEADING EDGE ANTI-ICING VALVE
36. HEAT CONTROL MODULATING VALVE
37. EMERGENCY CABIN AIR FLOW VALVE
38. BLEED AIR EMERGENCY WING ISOLATION VALVE
41. WINDSHIELD RAIN REMOVAL VALVE
42. ENGINE START AIR CONNECTION
44. ENGINE INLET DUCT LIP ANTI-ICING VALVE
55. AIR FLOW SENSOR AND TRANSMITTER
63. ELECTRIC HEATER—FLIGHT DECK
64. FREON COMPRESSOR
65. CONDENSER FAN
66. TURBOCOMPRESSOR CHECK VALVE
67. ENGINE START CHECK VALVE
68. TURBOCOMPRESSOR SURGE VALVE
69. FREON COMPRESSOR SURGE VALVE
71. FLIGHT COMPARTMENT AIRFLOW LIMITER
80. BLEED AIR OVERPRESSURE SWITCH
113. PLENUM CHAMBER PRESSURE SWITCH
115. WATER SEPARATOR
203. FREON DRIER AND STRAINER
213. NOZZLE ACTUATOR
223. VENTURI DUCT

30.02.21.128

Air Conditioning System Schematic - Ground Cooling
Figure 14-16

conditioned air emergency shutoff valve on to the conditioned air check valves and into the cabin and flight compartments.

POWER SOURCES

Power to operate the air conditioning and pressurization system is furnished by bleed air from the four turbojet engines and by the ac and dc electrical systems. For detailed information, consult the WIRING DIAGRAM MANUAL.

Engine Bleed Air
- Turbocompressors
- Alternate pressurization

Non-Essential AC Bus System
- Flight deck Freon package control components
- Electric heater, flight deck system
- Electronic temperature control, flight deck

Essential Ac Bus System
- Cabin Freon package control components
- Electric heaters, cabin system
- Electronic temperature control
- Ram air shutoff valves
- Crossover shutoff valve
- Cabin and flight deck emergency air shutoff valves
- Recirculating blower
- Electronic equipment cooling fan

28-Volt DC Bus System
- Warning lights
- Temperature and airflow indicators
- Turbocompressor shutoff valves and electronic cooling valve
- Alternate bleed air shutoff valves
- Controls for electric heaters, Freon refrigeration units, cabin air recirculating system, electronic equipment cooling fan, and electronic compartment cooling valve
- Cabin pressure regulators

AIR CONDITIONING AND PRESSURIZATION SYSTEMS
(Pneumatic Freon Drives Installed)

GENERAL AIR CONDITIONING AND PRESSURIZATION

The air conditioning and pressurization system provides passenger and crew comfort under all anticipated temperature conditions at airplane altitudes up to 41,000 feet. The pressurization equipment maintains sea level pressure in the cabin up to an airplane altitude of 21,300 feet, and an 8000-foot cabin pressure up to an airplane altitude of 41,000 feet. The air conditioning equipment supplies a complete change of air in the cabin every two and one-half minutes and in the crew compartment every minute. The inside temperature can be maintained between 18 degrees C and 33 degrees C (65 degrees F and 90 degrees F), with outside temperatures -54 degrees C to 38 degrees C (-65 degrees F to 100 degrees F). (See Figure 14-17 and 14-18.)

The cabin pressure and temperature are normally controlled automatically, but can be regulated manually at the discretion of the crew. The change of cabin pressure during rapid descents is controlled automatically at any preset rate between 50 to 2000 feet per minute.

The airplane is equipped with two independent and parallel pressurization and air conditioning systems. One system supplies air to the flight deck, the other to the cabin (see Figure 14-19). Two turbocompressors, driven by engine bleed air, provide an ample supply of compressed air to maintain the required cabin-air density during high-altitude operations. Inflight heating is provided by the compressing action of the turbocompressors. An air-to-air heat exchanger and a refrigeration unit in each system provide the cooling required to keep the cabin air at the selected temperature.

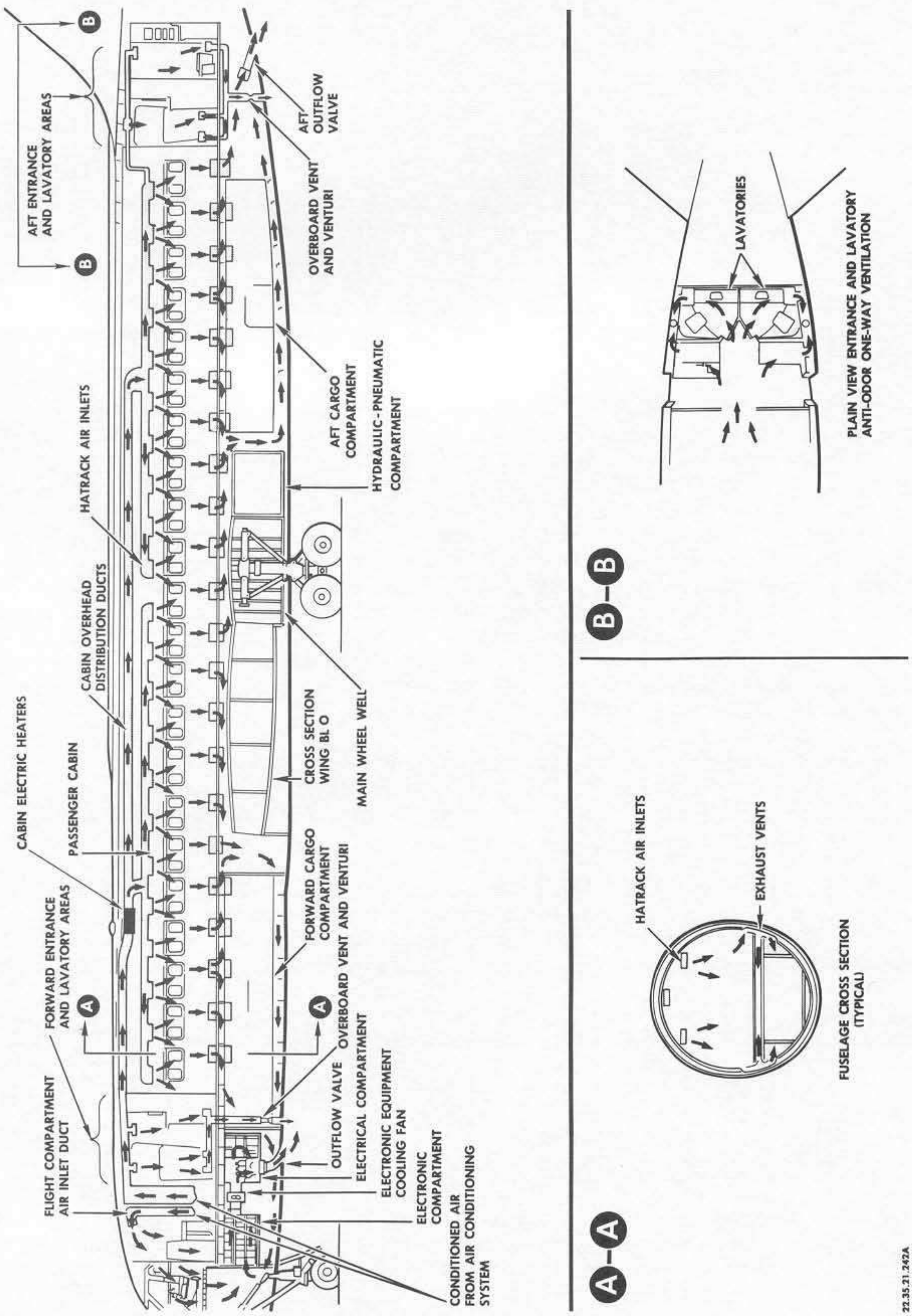
A recirculating fan can be used for recirculating the air through the cabin. A crossover duct provides a means of connecting the two systems if one turbocompressor should fail. Alternate pressurization is provided by bleed air from the engines.

The gages and switches for monitoring and controlling the operation of the air conditioning and pressurization system are located on the flight engineer's panel. The cabin pressure altitude, the cabin temperature, and the rate of change of cabin pressure can be regulated automatically or controlled manually.

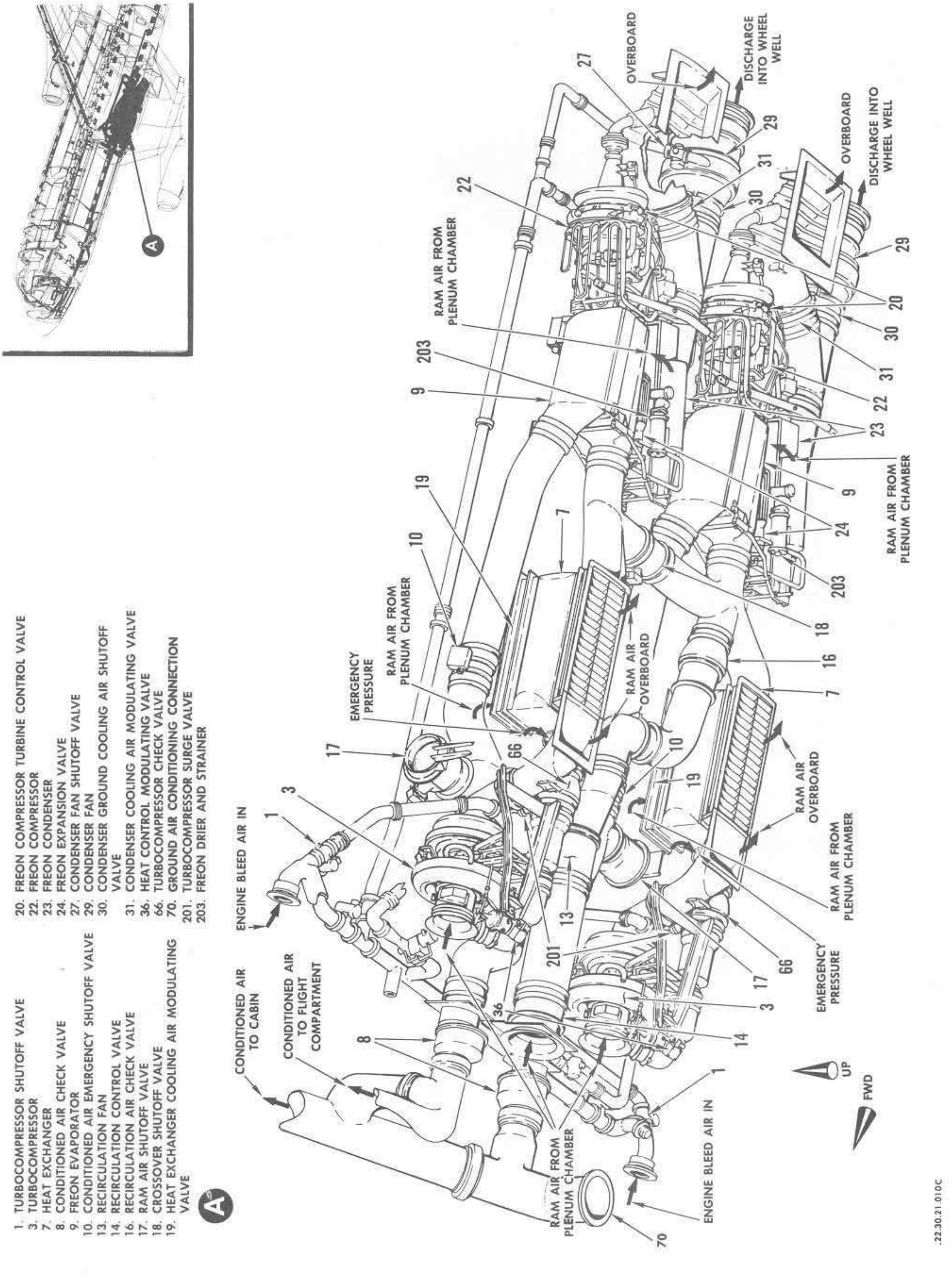
As the two air conditioning systems are identical, it will suffice to describe only one of them. Essentially, each system consists of a turbocompressor; a primary air-to-air heat exchanger; a refrigeration package; valves, regulators and switches for controlling the operation of the system; and the necessary air passages and ducts.

TURBOCOMPRESSOR UNIT

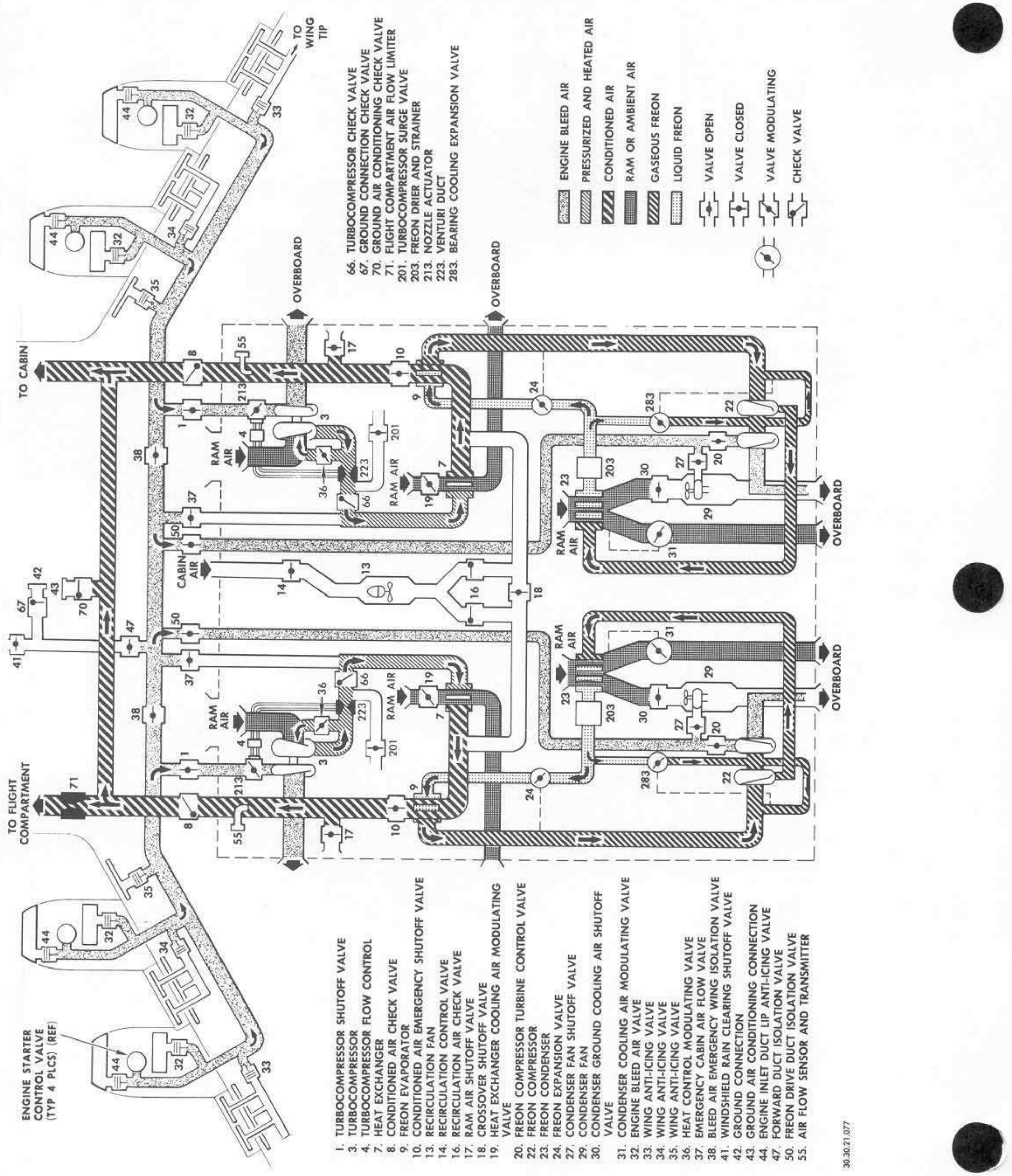
The turbocompressor supplies pressurized fresh air to the air conditioning system at the proper weight-flow and pressure to maintain the preselected cabin pressure. Each turbocompressor consists of a single-stage turbine and a centrifugal single-stage compressor mounted on a common shaft.



Air Conditioning and Pressurization Airflow
Figure 14-17



Air Conditioning Package
Figure 14-18



Air Conditioning System Schematic
Figure 14-19

Turbine

The turbine is driven by bleed air from the main bleed-air manifold which receives air from the seventeenth compressor stage of each engine through a pressure regulator that reduces the pressure of the bleed air to 40 psig. After passing through the turbine, the bleed air is dumped overboard (see Figure 14-20).

Bleed air enters the turbine through a turbocompressor shutoff valve to a variable area nozzle controlled by a flow sensor at the compressor outlet. The sensor detects variations in airflow and automatically increases or decreases the nozzle opening to adjust the turbine speed accordingly. An rpm indicator, indicating turbine speed, is located on the flight engineer's panel. An overspeed control shuts off the supply of bleed air if the turbine speed increases to between 54,000 and 56,000 rpm. When the turbine overspeed cutout is tripped, the unit must be reset manually on the ground. During reverse thrust, the turbocompressors are automatically shut down to prevent entry of exhaust gases into the cabin.

Compressor

Air conditioning ram air enters a plenum chamber through two large air scoops beneath the fuselage near the wing leading edge and passes directly into the compressor inlet. As the air passes through the compressor it is compressed and heated. If additional heating is required, a portion of the compressed air is recirculated through the compressor. A heat control modulating valve, regulated by the temperature control subsystem in the air conditioning system, controls the percentage of the compressor discharge air to be recirculated. A surge valve, also located at the compressor outlet, diverts compressed air back to the plenum chamber when a decrease in airflow through the compressor is accompanied by a marked increase in outlet pressure. This condition would arise in the case of a blocked duct. A check valve prevents reverse flow through the compressor when it is shut down.

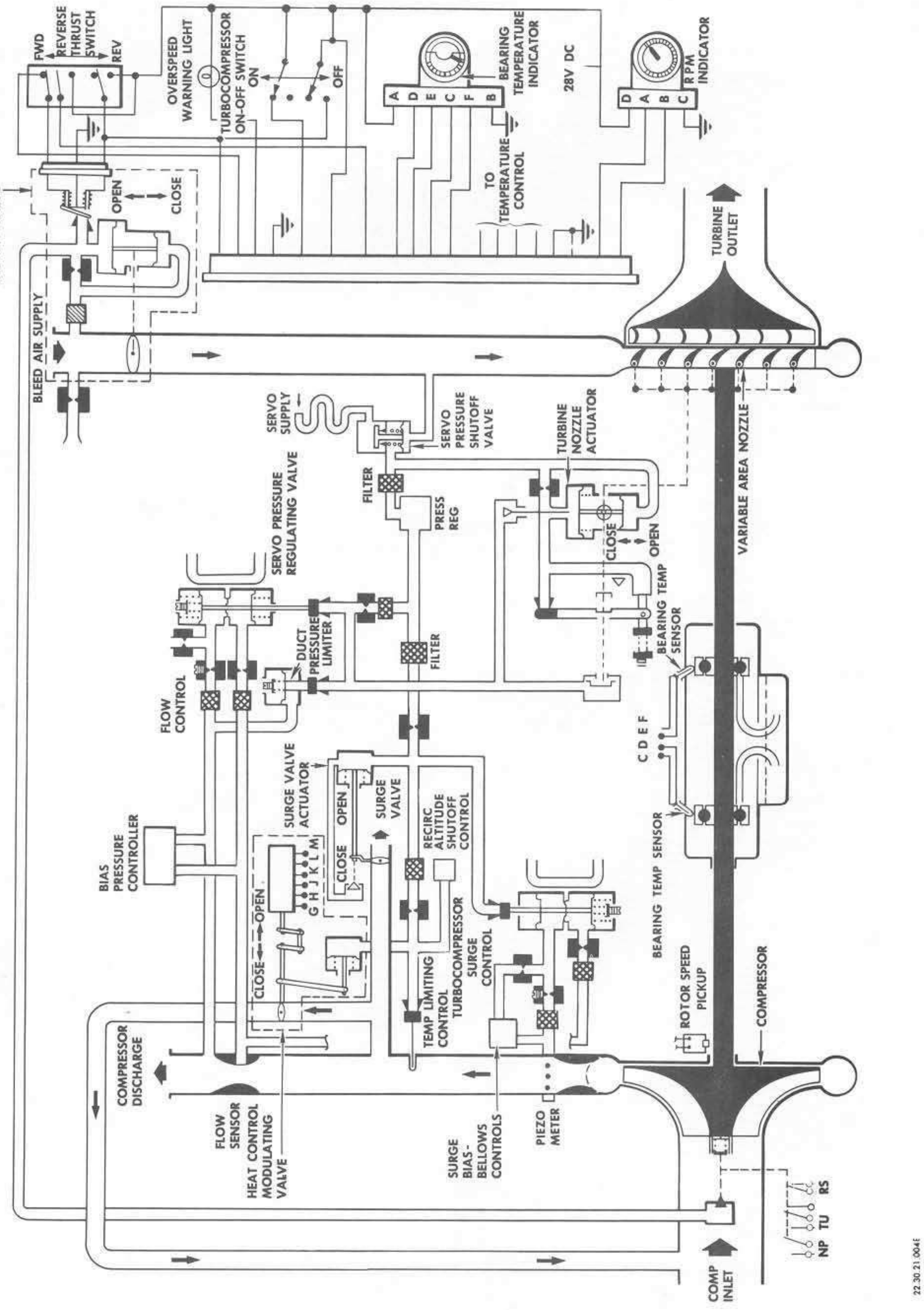
AIR-TO-AIR HEAT EXCHANGER

From the compressor, the heated compressed air passes through an air-to-air heat exchanger for primary cooling. The heat exchanger has a double-deck plate and fin core with vertical passages for ram air and horizontal passages for compressed cabin air (see Figure 14-21). The cabin air passes over and back through the two decks of the core. The ram air makes a single pass vertically through the core and absorbs heat from the plates and fins thus cooling the cabin air flowing through the horizontal passages.

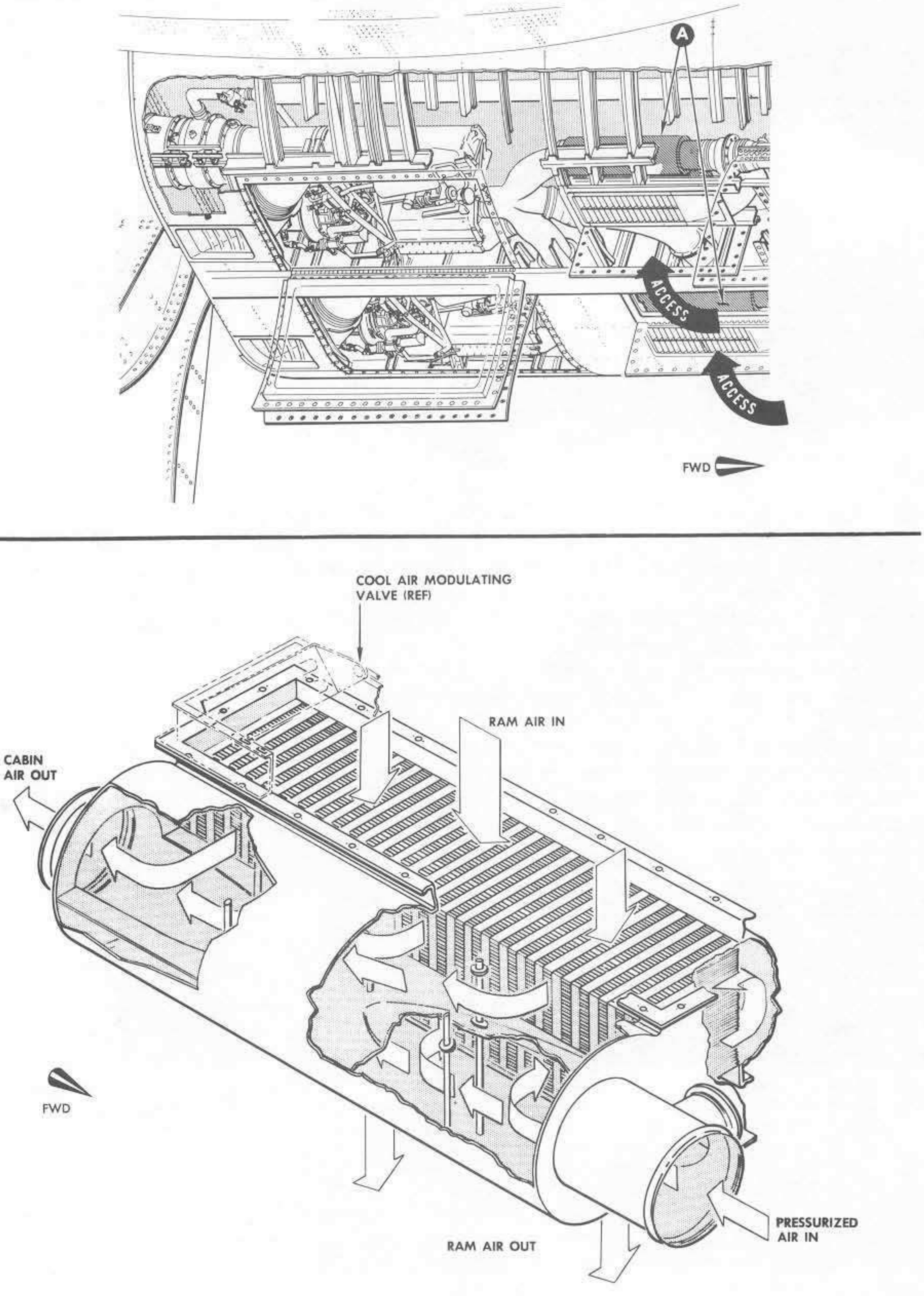
The ram air enters the heat exchanger through a cooling air modulating valve controlled by the temperature control subsystem of the air conditioning system. The amount of cooling varies from full capacity cooling to no cooling, depending on the position of the modulating valve.

FREON COOLING SYSTEM

Secondary cooling is provided by a closed vapor-cycle refrigeration unit located in the air conditioning compartment of the airplane. The unit consists of a



Turbocompressor Operational Schematic
Figure 14-20



22.30.21.0918

Heat Exchanger
Figure 14-21

compressor, condenser, evaporator, and associated control devices (see Figure 14-22). Freon 114, a nonpoisonous fluid which boils at 3.5 degrees C (38.4 F), is used as the refrigerant. Special oil is mixed with the Freon, about 1 part in 20, to lubricate the bearings and other internal moving parts. The system is deisnged for either automatic or manual control and will maintain stable operation when adjusted for any amount of cooling between full capacity and essentially no cooling.

A vapor-cycle refrigerator depends on the latent heat of condensation and vaporization of a refrigerant to produce the desired drop in ventilating air temperature. The refrigerant, in a gaseous state, is passed through a compressor where it is compressed and heated. The compressed gas is then cooled and condensed in a heat exchanger that disposes of the latent heat released when the gas condenses. The refrigerant, now in the liquid state, flows through another heat exchange where it changes back to gas. The heat needed to vaporize the liquid is extracted from the cabin air which the unit was designed to cool. From the evaporator, the gas is returned to the compressor and the cycle is repeated.

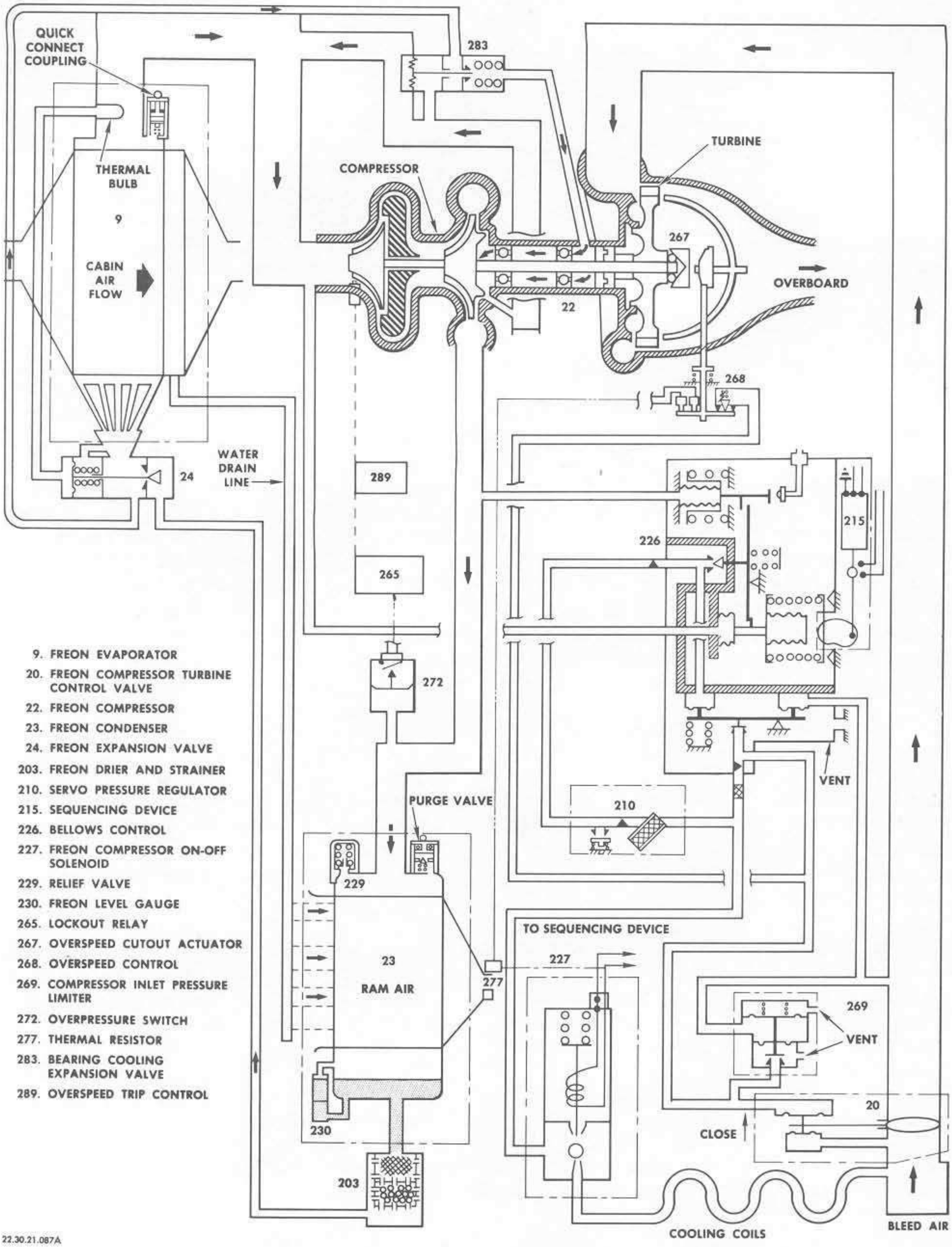
Freon Compressor

The Freon compressor is a turbine-driven two-stage centrifugal type compressor. The turbine rotor and compressor impellers are mechanically linked by a common drive shaft. The impellers have swept back blades which permit a wide stable operating range for various compressor speeds. Engine bleed air is used to drive the turbine. The bearings are cooled by vaporizing liquid Freon over them. Since the Freon contains a small quantity of lubricating oil, the cooling process deposits oil on the bearings for lubrication. The Freon compressor is protected against damage due to Freon overpressure. A cutout switch, in conjunction with a lockout relay, shuts down the compressor should an overpressure condition occur. An electronic topping speed control prevents the compressor from overspeeding. The control, in conjunction with the lockout relay and Freon on-off solenoid, shuts down the compressor when compressor speed reaches approximately 40,000 rpm. The control will restart the compressor when compressor speed reduces to about 32,000 rpm.

Freon Condenser

The Freon condenser is a Freon-to-air type heat exchanger in which the hot compressed Freon gas is cooled and condensed to a liquid. Cooling is provided during most flight conditions by allowing ram air to pass through the condenser. For slow flight and ground cooling, plenum air is drawn through the condenser by an electric fan. Sufficient condenser cooling is provided to change the Freon gas to a liquid and subcool it to keep the liquid from flashing into a gas when it leaves the condenser.

The cooling air leaves the condenser through a Y duct. In flight, the cooling air discharges through the condenser cooling air modulating valve and is dumped overboard. On the ground, the cooling air leaves the condenser through the ground cooling air shutoff valve, passes through the fan, and empties into the main landing gear well. The modulating valves are regulated automatically to maintain the correct Freon temperature.



9. FREON EVAPORATOR
20. FREON COMPRESSOR TURBINE CONTROL VALVE
22. FREON COMPRESSOR
23. FREON CONDENSER
24. FREON EXPANSION VALVE
203. FREON DRIER AND STRAINER
210. SERVO PRESSURE REGULATOR
215. SEQUENCING DEVICE
226. BELLOWS CONTROL
227. FREON COMPRESSOR ON-OFF SOLENOID
229. RELIEF VALVE
230. FREON LEVEL GAUGE
265. LOCKOUT RELAY
267. OVERSPEED CUTOUT ACTUATOR
268. OVERSPEED CONTROL
269. COMPRESSOR INLET PRESSURE LIMITER
272. OVERPRESSURE SWITCH
277. THERMAL RESISTOR
283. BEARING COOLING EXPANSION VALVE
289. OVERSPEED TRIP CONTROL

22.30.21.087A

Freon Refrigeration System Schematic
Figure 14-22

A liquid level gage provides a visual indication of the amount of Freon in the condenser. For an accurate check, the reading should be taken with the Freon package operating and with the condenser as cold as possible. A spring-loaded relief valve on the top of the condenser will open if the pressure in the condenser reaches from 170 to 180 psig. This pressure is well above the normal operating range.

A Freon drier and strainer unit with a replaceable cartridge is located at the condenser outlet. The cartridge contains a chemical drying agent (desiccant) surrounded by a filter screen. All Freon leaving the condenser is strained by the filter screen to prevent the circulation of foreign particles through the system. A portion of the Freon also comes in contact with the desiccant which removes any moisture and prevents corrosion within the system.

Freon Evaporator

From the condenser, the cold liquid Freon flows through a thermostatic expansion valve to the evaporator where it is completely vaporized before it returns to the compressor to complete the vapor cycle. The heat to "boil" the liquid Freon is extracted from the incoming cabin air.

The evaporator is a Freon-to-air heat exchanger made up of an outer shell and two headers connected by a core containing tubes. The Freon flows through the tubes while the cabin air from the turbocompressor passes through the spaces between the tubes and the outer shell. As the cabin air is cooled to the dew point, the moisture in the air condenses on the core surfaces in the form of small water droplets. A fine mesh screen separates the small droplets from the cabin air. The water drains into a water collector where cabin air pressure forces it through a spray tube which sprays the water over the ram air inlet of the condenser core.

The thermostatic expansion valve in the Freon line from the condenser regulates the flow of Freon into the evaporator and the degree of superheat at the evaporator outlet. The operation of this valve is controlled by a temperature sensing element in the outlet header of the evaporator. The flow of Freon into the evaporator is restricted to a rate that will assure complete vaporization and additional heating of the vapor before it leaves the evaporator. This valve also prevents liquid Freon from flooding the evaporator when the Freon package is shut down.

The boiling temperature of the liquid Freon depends on the vapor pressure within the evaporator. The cooling provided by the evaporator is regulated by controlling the vapor pressure within the evaporator through the action of the Freon compressor turbine control valve. The action of this valve is controlled automatically by the cabin temperature control system.

Compressor Bearing Cooling Loop

The compressor ball bearings are lubricated and cooled by vaporizing liquid Freon in the spaces surrounding the bearings. Liquid Freon from the condenser outlet passes into the bearing spaces through a bearing cooling expansion valve and empties into the bypass duct leading to the compressor inlet. The expansion valve is controlled by a pressure and temperature-sensitive thermal bellows located in the bypass duct where it comes in contact with the vaporized Freon leaving the bearings. The Freon flow through the bearings is automatically controlled to provide sufficient cooling and, at the same time, assure complete evaporation of the liquid Freon.

CABIN ELECTRIC HEATERS

Two electrical heaters, located in the cabin overhead, heat conditioned air to a selected temperature prior to distribution in the cabin. The heaters are controlled by the cabin differential temperature control system.

Each heater is divided into two parallel sections. Air enters the heater from the forward side then passes through ducting to the forward and aft cabins. Bare wire heating elements in each heater, are wye-connected for 3-phase ac power. Each element is capable of heating 41 pounds of air per minute.

Two normally closed thermal switches, are located in each heater. If the air temperature exceeds 130 degrees F (72°C) the switches open, breaking continuity of the heater elements. When the temperatures goes below 130 degrees F (72°C), the switches close.

Cabin Differential Temperature Control System

A maximum temperature differential of 2 degrees F (1.1°C) is maintained between the forward and aft cabin sections. Each of the heaters is controlled by the temperature differential control system. The cabin differential control system consists of the cabin temperature control panel, modulating power supply, temperature differential control, temperature sensors and two dual electric cabin overhead heaters.

After the desired cabin temperature has been selected on the flight engineer's air conditioning panel, conditioned air from the under-wing turbocompressors enters ducting in the cabin overhead section. The air passes through the two heaters, and is ducted to the forward and aft cabin sections. The heater discharge air temperature and the cabin discharge air temperature are each monitored by sensors in both the forward and aft cabin sections. Signals from the sensors are sent to the temperature differential controller which compares the signals and schedules the system to supply warmer air to the colder cabin.

CABIN AIR RECIRCULATING SYSTEM

The recirculating system is used primarily to provide cabin ventilation when the airplane is connected to a ground cart or when taxiing on a hot day. In flight, the system can be used as a supplemental source of air when one of the turbocompressors is shut down. The system is connected to the flight deck and cabin air conditioning systems through the conditioned air crossover duct.

The recirculating system consists essentially of a two stage fan driven by a 400-cycle, 115/200-volt ac electric motor and a cabin air control valve. The fan is controlled through an AUTO-OFF-MAN toggle switch on the flight engineer's panel.

ALTERNATE PRESSURIZATION

Bleed air can be used for alternate cabin pressurization. Two bleed air lines connect the bleed air manifold with the cabin and flight deck air conditioning systems downstream from the turbocompressors. Two toggle switches on the flight engineer's panel control the alternate flow control and shutoff valve in each bleed air line. Since the bleed air passes through the primary heat exchangers and the Freon evaporators, its temperature is regulated by the cabin temperature control system (see Figure 14-23).

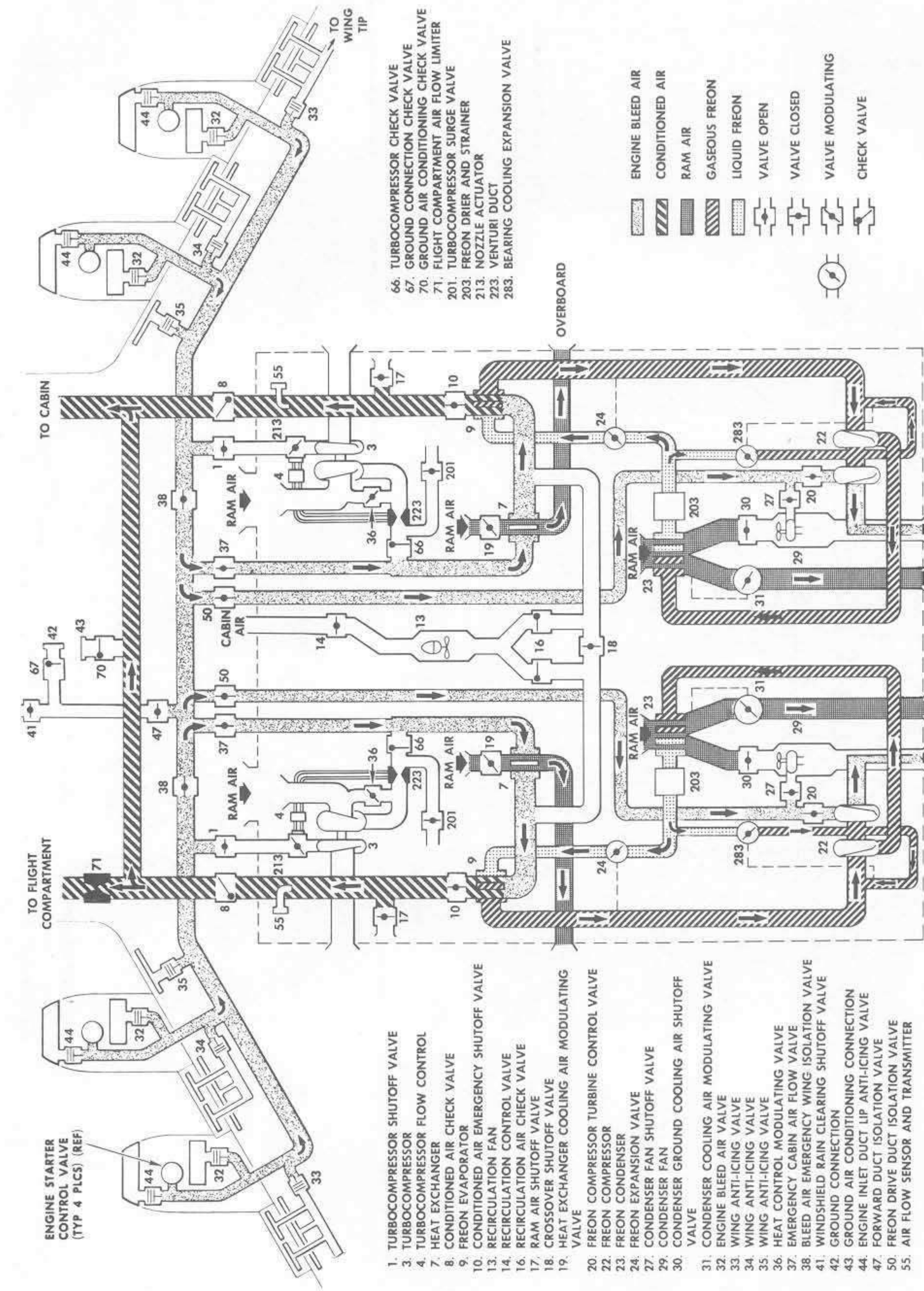
CABIN PRESSURE REGULATOR AND OUTFLOW VALVE

Two cabin pressure regulator and outflow valves maintain the desired pressure level within the airplane. The two valves control the pressurization level, or cabin altitude, by regulating the amount of air exhausted from the pressurized compartment. They also serve as pressure relief, vacuum relief, and dump valves (see Figure 14-24).

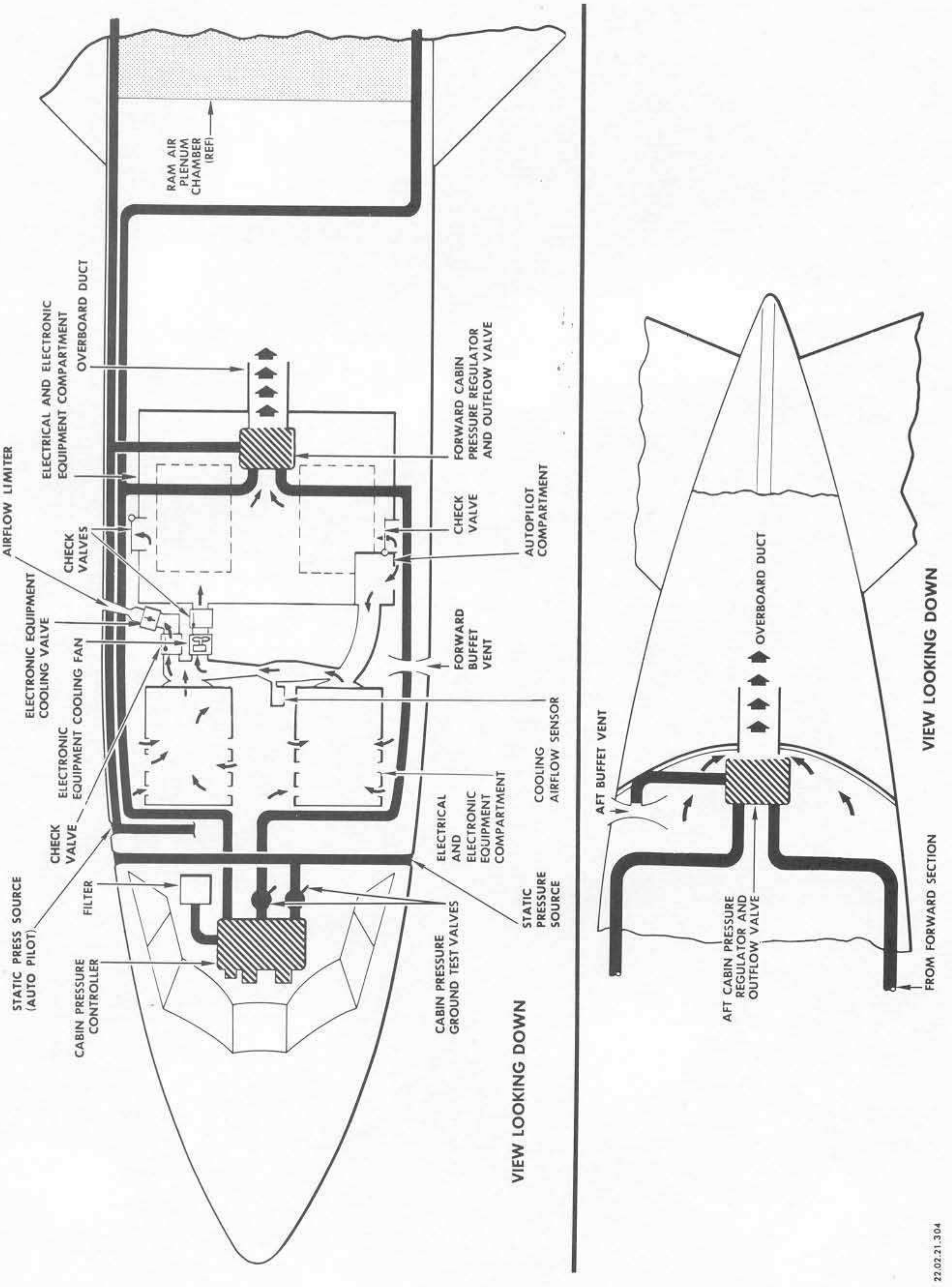
The forward outflow valve is located on the floor of the electrical compartment, and the aft outflow valve is located below the aft lavatory. Both valves are mounted on short transition ducts which adapt the circular valve outlet to a rectangular discharge port in the skin. The two valves are identical except for a coarse screen which is wrapped around the forward valve. The screen will prevent the electrical compartment curtain from being sucked into the valve assembly should the curtain come loose. Primary control of both valves is maintained by the cabin pressure controller on the flight engineer's control panel.

When the PRESSURE REGULATORS control switches are placed in the AUTO position, the operation of the pressure regulator and outflow valves is completely pneumatic. When operated manually, an electric motor is used to position the outflow valve. If the valves are closed electrically the pressure and vacuum relief functions also require electric power. Many fail-safe features are incorporated in the design of the cabin pressure control subsystem. When operating in the automatic mode, a failure of one valve to the closed position will cause no change in pressurization because the other valve has the capacity to regulate the entire outflow. If both valves fail to the closed position, the pressure and vacuum relief functions will still operate. If one or both valves fail to the open position, a cabin pressure limiting device will close the valves enough to maintain a cabin altitude of 13,000 (±2,000) feet.

If two failures occur in one valve, such as a failure in the closed position and a failure of the pressure relief function - the manual controls can override the pneumatic system. Another safety feature is included in the wiring installation to prevent accidental dumping of cabin pressurization by a short circuit. The "manual open" control circuit is completely shielded with a grounded metallic sheath. The sheath will ground any "hot" wires that might come in contact with this critical wiring, and will actuate the appropriate overload protection to remove the power. Terminal connections on the manual control switches are "potted" to further reduce the possibility of trouble.



Alternate Pressurization System Schematic
Figure 14-23



Pressurization System Control Valves
and Vents
Figure 14-24

If the aft outflow valve should fail in the closed position an inward-opening check valve allows more air to enter the electrical chamber directly from the cabin. If the forward outflow valve should fail in the closed position, an outward-opening relief valve allows air from the cooling fan to discharge into the cabin.

VALVES

The control and operation of the air conditioning and pressurization system depends primarily on the functioning of various valves located throughout the system. Check valves are placed at strategic points in the system to prevent reverse flow; modulating valves regulate the rate of flow through the passages in which they are located; and shutoff valves are utilized to start and stop certain operations. Some of the modulating valves also serve as shutoff valves.

Check Valves

A check valve is located between each turbocompressor and corresponding emergency bleed air inlet to prevent reverse flow through the turbocompressors when the bleed air is turned on. A second pair of check valves is located in the ducts leading to the flight deck and the cabin, upstream from the crossover passage connecting the two air conditioning systems. When one of the systems has been shut down, the corresponding valves will close and keep the conditioned air from entering the inoperative system. This also prevents loss of pressurization should a duct break occur in the plenum area. Another pair of check valves isolates the recirculating blower from the two air conditioning systems when the blower is not operating. These check valves also prevent engine bleed air from entering the recirculating blower housing when the alternate pressurization system is actuated.

Another check valve is located in the ground air conditioning connection to close this opening except when conditioned air is being furnished by a ground air conditioning unit.

Modulating Valves

The speed of each turbocompressor is regulated by a variable-area nozzle functioning as a modulating valve to control the flow of bleed air through the turbine. The recirculation of compressed air through the compressors is regulated by the cabin heat control modulating valves. Two modulating valves regulate the flow of cooling ram air through the primary heat exchangers and an additional pair of modulating valves controls the flow of ram air through the Freon condensers.

The air supply for the recirculating blower is controlled by a shutoff valve in the cabin air recirculating duct. This valve also functions as a shutoff valve to close the passage when not in use. The cabin air outflow valves modulate the rate at which the cabin air is released to the atmosphere and close the outflow openings when conditions warrant.

Modulating valves in each Freon package regulate the flow of Freon through the Freon loops. One expansion valve controls the flow of liquid Freon to the evaporator. Another expansion valve controls the flow of liquid Freon through the compressor bearings for cooling and lubrication.

Shutoff Valves

Most of the air conditioning and pressurization equipment is turned on and off by shutoff valves. The turbocompressors are turned on by opening the turbocompressor bleed air shutoff valves. Two shutoff valves in the emergency bleed air ducts supply bleed air to the air conditioning systems when both turbocompressors are inoperative. An emergency shutoff valve in each system can be used to close either the conditioned air supply duct leading to the cabin or the one leading to the flight deck. Two motorized check valves, one in each ram air inlet, can be opened to admit ram air directly to the flight deck and the cabin. These valves cannot be opened when the cabin pressure is higher than that of the ram air. Shutoff valves control the flow of engine bleed air to the Freon compressors and the condenser fans.

SYSTEM OPERATION

Ram air from the plenum chamber is compressed and heated by the compressors. The cabin heat control modulating valves regulate the amount of air that will be recirculated through the compressors for additional heating. The compressed air then passes through check valves and enters the primary heat exchangers where it is cooled, as required, by ram air from the plenum chamber. The amount of cooling is regulated by modulating valves that control the flow of cooling ram air through the heat exchangers.

The compressed air then passes through the Freon evaporators where its temperature may be further reduced by the vaporizing action of the Freon. From the evaporator, the air flows past the airflow sensors and transmitters that provide an indication of turbocompressor, or fan air-weight flow. It then flows through a flow limiter which permits about 10 percent of the air to flow to the flight compartment. The remaining airflow is routed to the cabin.

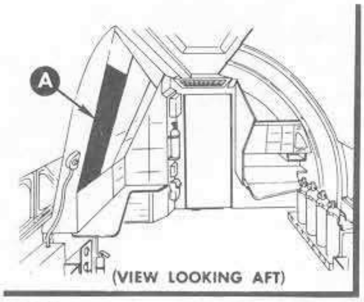
The recirculating blower is used for air conditioning on the ground and to improve ventilation in flight when one of the turbocompressors has been shut down. The outflow valves allow the air to escape from the cabin at a rate that will maintain the scheduled cabin pressure.

AIR CONDITIONING AND PRESSURIZATION SYSTEM CONTROLS

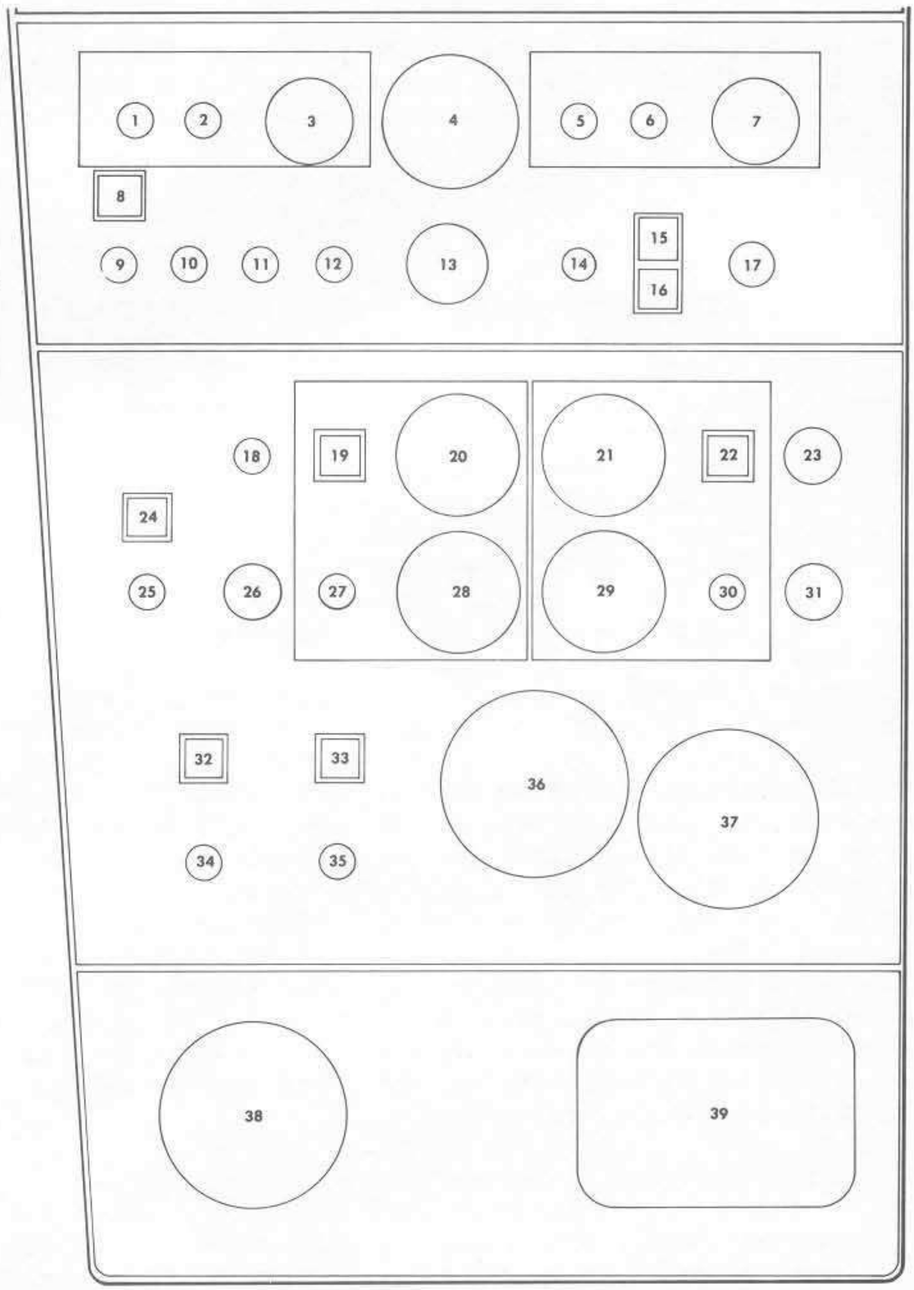
The instruments and controls for monitoring and regulating the air conditioning and pressurization systems are located on the flight engineer's panel (see Figure 14-25).

Instruments

Two COMPR RPM gages indicate the speed of the turbocompressors. Two COMP BEARING TEMP gages indicate the temperature of tue turbocompressor bearings. A dual COMPRESSOR AIR FLOW gage registers the air flow from each air conditioning unit in lbs/min. A CABIN TEMP gage provides an indication of cabin



A



1. FLIGHT DECK MANUAL TEMPERATURE CONTROL SWITCH
2. FLIGHT DECK AUTOMATIC TEMPERATURE CONTROL SWITCH
3. FLIGHT DECK TEMPERATURE SELECTOR
4. TEMPERATURE INDICATOR
5. CABIN MANUAL TEMPERATURE CONTROL SWITCH
6. CABIN AUTOMATIC TEMPERATURE CONTROL SWITCH
7. CABIN TEMPERATURE SELECTOR
8. ELECTRONIC COMPARTMENT COOLING LOW-AIRFLOW WARNING LIGHT
9. ELECTRONIC EQUIPMENT COOLING FAN SWITCH
10. ELECTRONIC COMPARTMENT COOLING VALVE SWITCH
11. CABIN/FLIGHT DECK FREON RESET SWITCH
12. CABIN TEMPERATURE EQUALIZATION CONTROL SWITCH
13. TEMPERATURE INDICATOR SELECTOR SWITCH
14. FREON MASTER CONTROL SWITCH
15. FLIGHT DECK FREON COMPRESSOR FAIL WARNING LIGHT
16. CABIN FREON COMPRESSOR FAIL WARNING LIGHT
17. CABIN-BOTH-FLIGHT DECK FREON PACK CONTROL SWITCH
18. RECIRCULATION AIR BLOWER CONTROL SWITCH
19. FLIGHT DECK TURBOCOMPRESSOR OVERSPEED WARNING LIGHT
20. FLIGHT DECK TURBOCOMPRESSOR R.P.M. INDICATOR
21. CABIN TURBOCOMPRESSOR R.P.M. INDICATOR
22. CABIN TURBOCOMPRESSOR OVERSPEED WARNING LIGHT
23. RAM AIR VALVES CONTROL SWITCH
24. CABIN HIGH-ALTITUDE WARNING LIGHT
25. CABIN HIGH-ALTITUDE WARNING HORN CUTOFF SWITCH
26. FLIGHT DECK PRESSURE SOURCE SWITCH
27. FLIGHT DECK TURBOCOMPRESSOR CONTROL SWITCH
28. FLIGHT DECK TURBOCOMPRESSOR BEARING TEMPERATURE INDICATOR
29. CABIN TURBOCOMPRESSOR BEARING TEMPERATURE INDICATOR
30. CABIN TURBOCOMPRESSOR CONTROL SWITCH
31. CABIN PRESSURE SOURCE SWITCH
32. FORWARD OUTFLOW VALVE VALVE-CLOSED WARNING LIGHT
33. AFT OUTFLOW VALVE VALVE-CLOSED WARNING LIGHT
34. FORWARD OUTFLOW VALVE CONTROL SWITCH
35. AFT OUTFLOW VALVE CONTROL SWITCH
36. CABIN/FLIGHT DECK TURBOCOMPRESSOR AIRFLOW INDICATOR
37. CABIN DIFFERENTIAL PRESSURE AND CABIN ALTIMETER INDICATOR
38. CABIN RATE-OF-CLIMB INDICATOR
39. CABIN PRESSURE CONTROLLER (INCLUDES CABIN ALTITUDE CONTROL, RATE-OF-CLIMB OR DESCENT CONTROL, AND BAROMETRIC PRESSURE CONTROL)

30.30.21.074 A

Air Conditioning and Pressurization System
Control Panel
Figure 14-25

temperature and, but use of a rotary selector, the gage can also be used to monitor the operation of the Freon packages. Placing the selector in either the LH or RH FREON EVAP position will provide an indication of the Freon evaporator discharge air temperature for the respective package.

A CABIN ALTIMETER and CABIN DIFFERENTIAL PRESSURE indicator, a cabin CLIMB indicator and a CABIN PRESSURE CONTROL are located in the lower half of the control panel.

The cabin pressure controller (see Figure 14-26) is used in conjunction with two cabin pressure regulating outflow valves to control cabin pressure. It controls the cabin pressure regulator and outflow valves by varying a reference pressure transmitted to the pneumatic relay in each valve.

The cabin pressure controller is located on the lower right side of the flight engineer's air conditioning and pressurization control panel. Three indicators and three control knobs are incorporated in the controller. The main dial and pointer indicate the selected cabin altitude within the range of minus 1,000 to plus 10,000 feet. The small inner dial at the top indicates the maximum flight altitude at which the selected cabin altitude can be maintained. The dial in the upper right side is the barometric correction scale, and is calibrated in inches of mercury. The three control knobs are the cabin altitude control, the barometric correction control, and the rate control.

The cabin altitude control knob is used to set the desired cabin altitude on the main dial. The barometric control is set at the local altimeter setting prior to take off or landing. The rate control knob is used to establish the rate of climb or descent in cabin altitude. The rate control knob can be set to maintain any rate between 50 and 2,000 feet per minute.

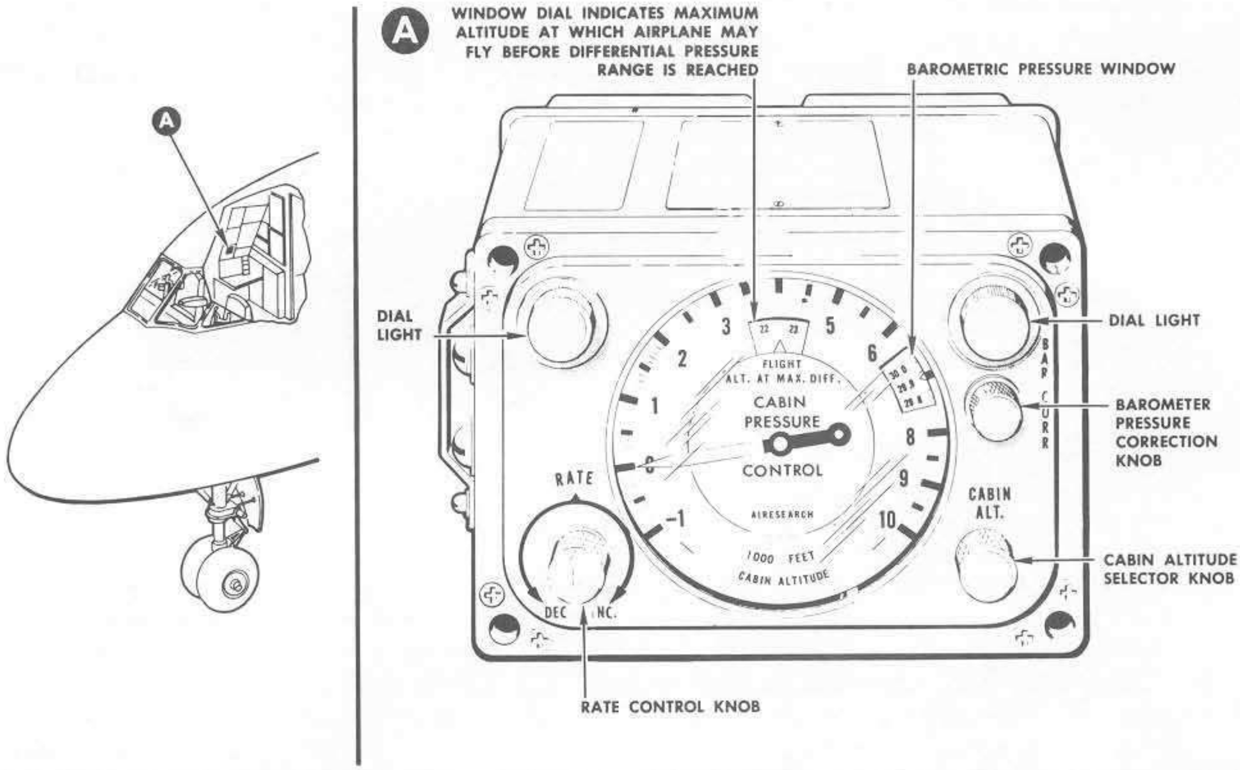
The cabin pressure controller has three separate pressure controlling systems or modes of operation. They are a selective pressure rate of change, a variable isobaric, and a fixed differential pressure system. These systems operate one at a time to establish the reference pressure, and the transition from one to another is automatic.

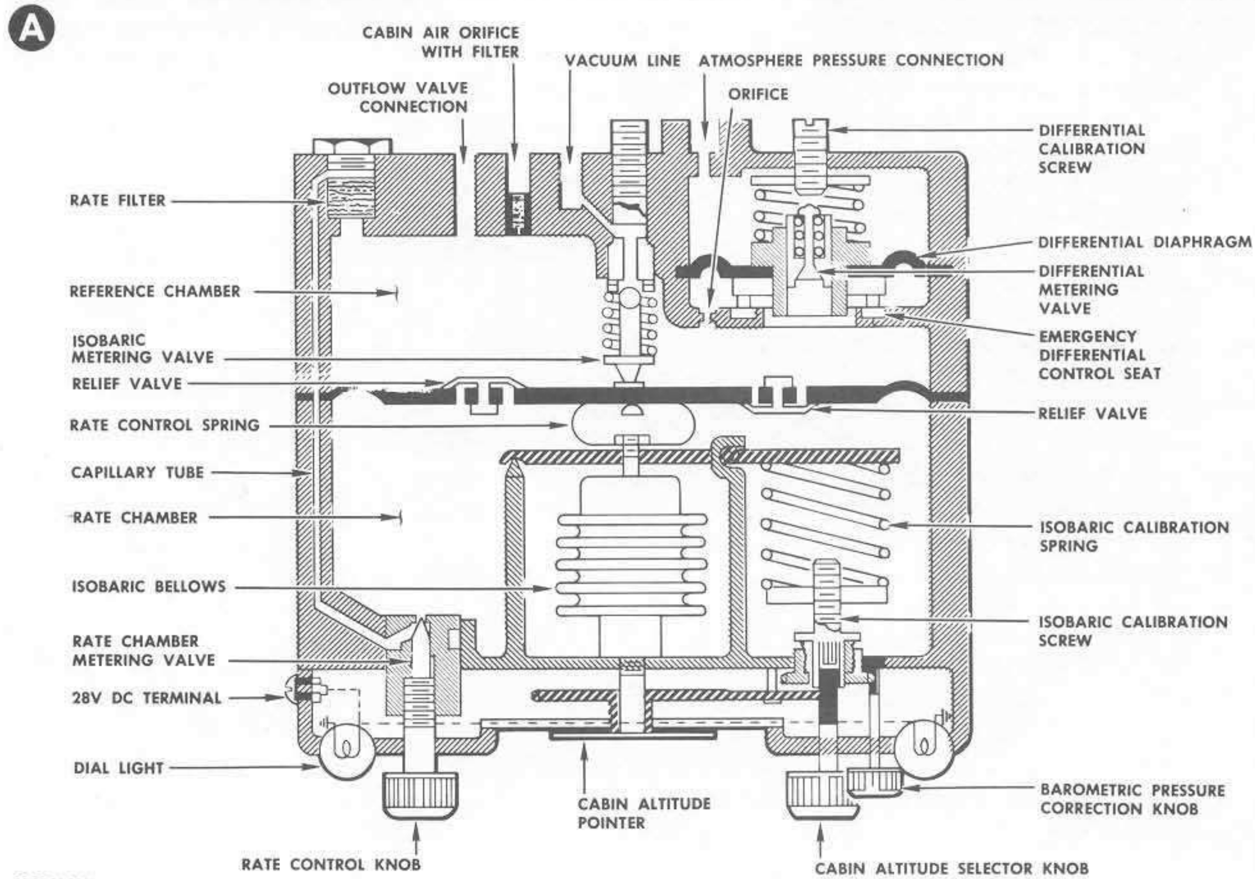
A. Selective Pressure Rate-of-Change Control.

The selective pressure rate-of-change control system consists of the rate control spring, a rate control diaphragm which separates the housing into two chambers, and a connecting drilled passage with filter, capillary tube, and rate chamber metering valve.

B. Isobaric Control.

The variable isobaric control system consists of an evacuated isobaric bellows and rocker arm which are connected to an isobaric metering valve through a rate control spring. An isobaric calibration spring and isobaric calibration screw are secured to the rocker arm, and are indexed to a cabin altitude pointer through a gear train.





22.30.21.001B

Cabin Pressure Controller
Figure 14-26

C. Fixed Differential Control.

The fixed differential control system consists of a spring-loaded differential diaphragm, a differential metering valve and a differential calibration screw. One side of the differential diaphragm is vented to atmospheric pressure and the other side of the differential diaphragm is exposed to reference pressure. This system also incorporates a fail safe feature consisting of an emergency differential control seat and orifice.

D. Barometric Pressure Correction.

The barometric pressure correction mechanism consists of a barometric pressure correction knob, a gear train linked to the variable isobaric control spring, and a window dial calibrated over a range of 28.0 to 31.0 inches of mercury absolute.

E. Control Operation - Rate of Change.

Before control operation, pressures on both sides of the rate control diaphragm are equalized at existing atmospheric pressure. This compresses the isobaric bellows and opens the isobaric metering valve. Cabin air then flows into the reference chamber through the cabin air filter and orifice, and out through the isobaric metering valve to atmosphere. Pressure in the reference chamber is sensed by the outflow valve, which then operates according to the cabin-to-airplane altitude graph (see Section 21-0) to produce an essentially unpressurized cabin, provided the rate of ascent of the aircraft does not exceed the selected rate. If the airplane rate of climb exceeds the selected rate, reference pressure is reduced, resulting in a pressure differential across the rate control diaphragm because of the capillary tube and rate chamber metering valve position. The rate control diaphragm then moves the isobaric metering valve to limit flow of air from the reference chamber until reference pressure and rate pressure equalize. The outflow valve reacts to control the rate of change of cabin pressure accordingly.

F. Control Operation - Isobaric Range.

As the airplane enters the isobaric operating range, the isobaric bellows expands sufficiently to override the rate control diaphragm and assume control of the isobaric metering valve. With reference chamber pressure maintained essentially constant by the isobaric bellows and metering valve, reference pressure in the outflow valve remains constant. Variations in cabin pressure act directly to position the outflow valve, controlling cabin pressure at selected altitude.

G. Control Operation - Differential Range.

The differential control serves to limit reference pressure in order that the outflow valve will establish an essentially constant pressure differential between cabin and atmosphere. As the airplane enters the differential operating range, the pressure differential between reference chamber and atmosphere allows the differential diaphragm to open the differential metering valve and release reference chamber air to atmosphere. The fail safe feature prevents loss of reference pressure in the event of a ruptured differential diaphragm by sealing the differential diaphragm from the reference chamber, permitting only a small loss of reference chamber air through the orifice. During a rapid descent, the differential metering valve closes because of increased atmospheric pressure. If the resultant rate of increase in reference pressure exceeds the selected rate, the rate control diaphragm moves the isobaric metering valve to a metering position and air from the reference chamber is bled to atmosphere. The outflow valve reacts to the reference pressure by limiting the rate of increase in cabin pressure.

Before landing, the cabin altitude selector is set to the destination airport altitude. The barometric pressure correction, in the dial right window, is then set by turning the barometric pressure correction knob. This rotates the gear train, which changes the tension of the isobaric calibration spring and corrects the unit to barometric pressure at the flight destination.

Air Conditioning and Pressurization System Control Switches

Two turbocompressor ON-OFF switches control the operation of the flight deck and cabin turbocompressors.

Two PRESS SOURCE switches OPEN or CLOSE the bleed air valves for emergency pressurization.

The RECIRCULATING BLOWER is controlled by a single AUTO-OFF-MAN toggle switch.

The cabin temperature equalization control system is actuated by the single CABIN TEMP EQ CONT toggle switch.

The Freon compressor ON-OFF MASTER switch controls the Freon compressors as selected by the three-positioned FREON COMPRESSOR selector switch. This switch placarded FLT DECK OFF-BOTH ON-CABIN OFF, is used for manual selection of the desired Freon compressor operational mode.

The FLT DECK - CABIN FREON RESET switch is utilized for manual recycling of lockout relays which have opened a power circuit due to an overheat or overpressure condition.

A RAM AIR SOURCE locking type toggle switch is used to OPEN or CLOSE the ram air valve.

The electronic equipment cooling fan is controlled by an ON-OFF toggle switch placarded ELEC EQUIP COOL FAN. The electronic compartment cooling valve is controlled by an OPEN-CLOSE toggle switch placarded ELEC COMPT COOL VALVE.

Two PRESSURE REGULATOR switches placarded AUTO-OPEN-CLOSE-OFF provide for manual operation of the two outflow valves. In the AUTO position, cabin pressure is regulated by the cabin pressure controller. The OPEN and CLOSE positions are used when operated under manual control to "inch" the outflow valves toward the open or closed position.

Four toggle switches and two thermostatic controls are located at the top of the control panel. The two AUTO-OFF-MAN three-position switches are used to select the operating mode of the flight deck or cabin air conditioning systems. Two MAN HOT-MAN COLD momentary contact type switches are used to "inch" the temperature setting up or down when operating it in manual mode. Two thermostatic controls are used to regulate the temperature controls of the two systems when operating in the automatic mode.

Air Conditioning and Pressurization System Warning Lights

Eight air conditioning and pressurization system warning lights are located on the flight engineer's panel. Two amber OVERSPEED TRIP warning lights connect to the overspeed controls of the two turbocompressors. If one of the turbocompressors should overspeed and trip the control, the corresponding light will illuminate.

Two amber warning lights, LH FREON FAIL and RH FREON FAIL, connect to the Freon compressor discharge overpressure switches through the Freon lockout relays. The "fail" lights will illuminate if the compressor(s) shut down due to an overpressure or overspeed condition.

An amber FWD CLOSE and an amber AFT CLOSE light, above the pressure regulator switches provide a warning if one of the cabin outflow valves fails in the closed position. A single red HIGH ALT light illuminates when the pressure altitude in the cabin exceeds 10,000 feet. A warning horn sounds simultaneously with the light. An ALT WARN HORN CUT-OFF switch, placarded PUSH, is provided to shut off the warning horn.

An amber LOW AIR FLOW light above the ELEC EQUIP COOL FAN switch will illuminate if the airflow is too low to provide adequate cooling for the electronic equipment.

AIR CONDITIONING OPERATION

The air conditioning subsystem provides for control of three environmental factors in the airplane; cabin and flight compartment temperatures, replenishing of cabin and flight compartment air, and removal of moisture from the refrigerated air. The air conditioning subsystem is capable of cooling the compressed air, heating the compressed recirculated cabin air, removing moisture and controlling subsystem operation. Conditioned air is heated by two different methods: recirculating pressurized air through the turbocompressor and by electric heating. Air is cooled by two cooling subsystems: an air-to-air heat exchanger and a Freon refrigeration cooling subsystem. Cabin air is recirculated by an electrically-driven recirculation fan, ducting and valves. Moisture is extracted as a part of refrigeration cooling when the Freon refrigeration subsystem is in operation. The system is controlled by the temperature control subsystem.

The ram air is compressed and heated in the turbocompressor. It is then cooled, if necessary, while flowing through the air-to-air heat exchanger and the Freon evaporator, and is distributed to the flight and cabin compartments. There are five normal modes of heating or cooling the flight and cabin compartments while the airplane is in flight or on the ground. These modes of operation are described in the following paragraphs.

Inflight Heating

During flight, the recirculation control valve is not subject to control by the temperature control system, thus, the valve is closed. During flight the dual heaters located in the cabin ducts are used as required, to maintain a 2 degrees F (1.1°C) differential temperature between fore and aft cabin areas.

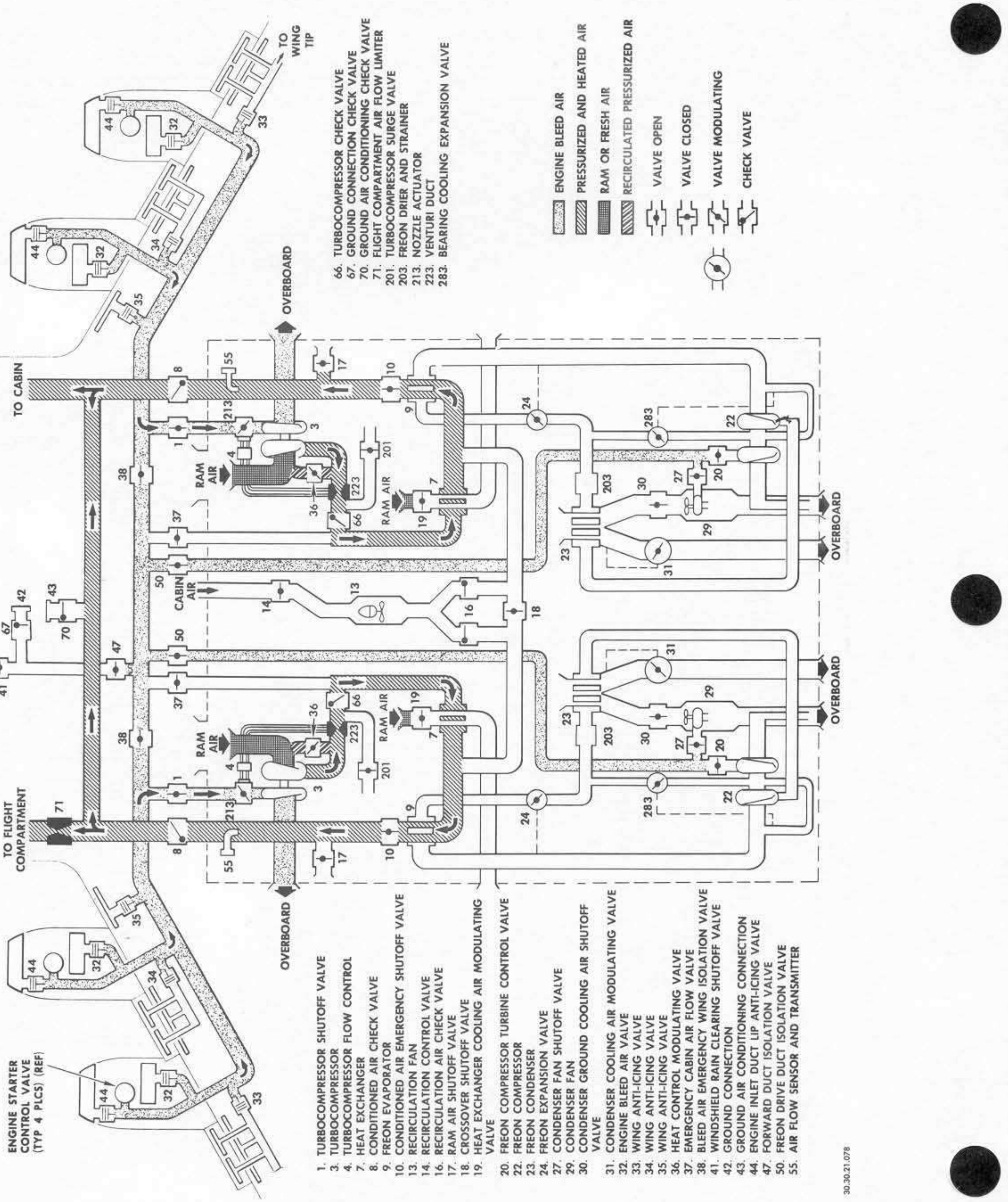
As shown in Figure 14-27, the turbocompressor shutoff valve is open, allowing engine bleed air to operate the turbocompressor. Ram air from the plenum chamber enters the compressor side of the turbocompressor, where it is compressed and heated. If one pass through the compressor does not heat the air enough, a portion of the air is recirculated through the compressor by the heat control modulating valve. The heated and compressed air then flows through the turbocompressor flow control and the turbocompressor check valve on to the heat exchanger.

The compressed and heated air enters the heat exchanger and flows through without any cooling exchange, because heating is scheduled. Since the heat exchanger cooling air modulating valve is closed, ram air for cooling cannot flow through the heat exchanger. The conditioned air flows from the heat exchanger to the Freon evaporator. As the temperature control system is calling for maximum heat, the Freon evaporator is shut down and the valves in the Freon system are closed. The conditioned air flows through the Freon evaporator, without any cooling, and through the conditioned air emergency shutoff valves. The air then flows to the cabin and flight compartments, through the conditioned air check valves, where it is distributed by the air distribution subsystem. The conditioned air delivered to the cabin is heated, if necessary, by the cabin dual heaters. Cabin heater operation is determined by the temperature equalization control system to maintain the air temperature between the forward and aft cabin areas within a maximum differential of 2 degrees F (1.1 degree C).

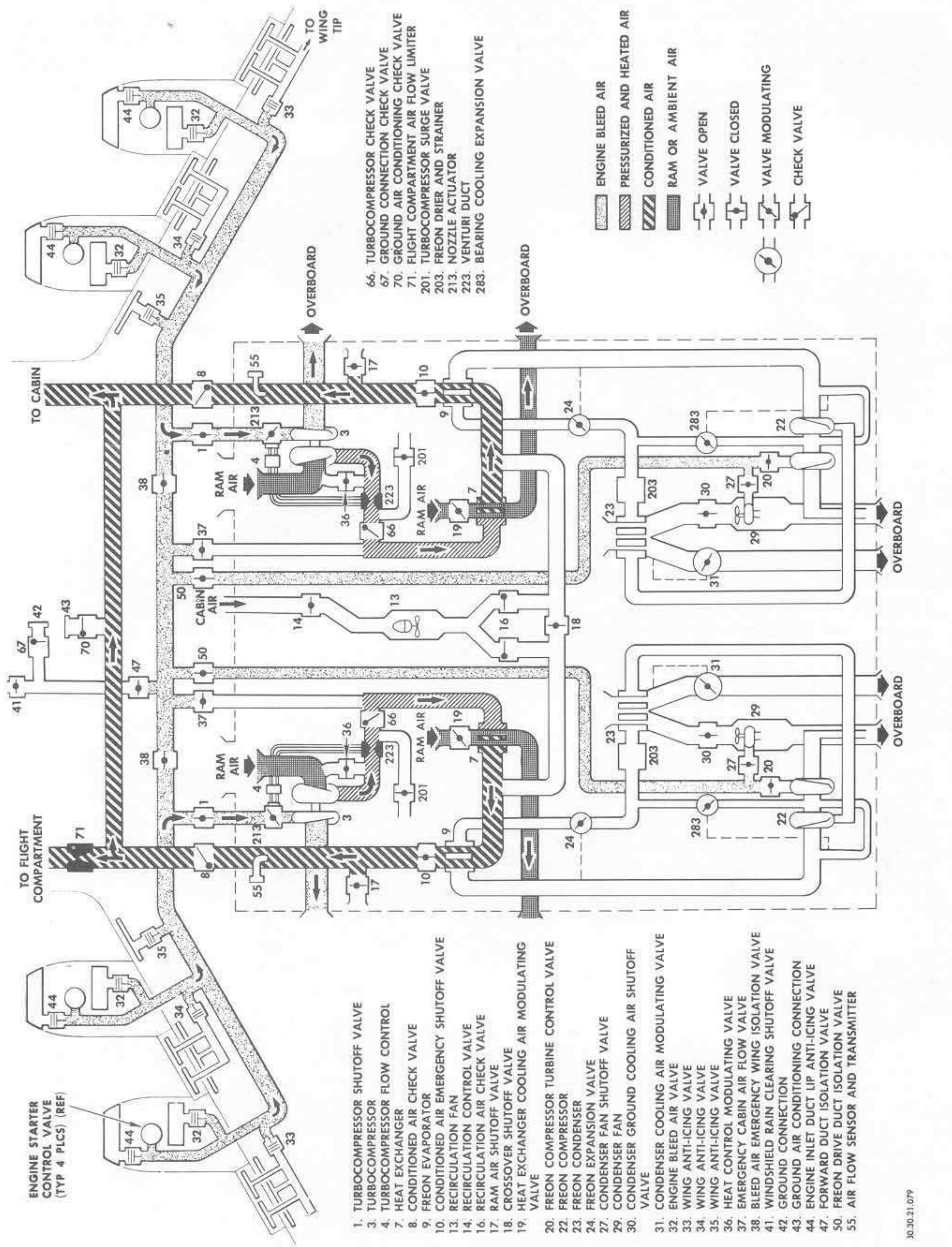
Inflight Intermediate

During the inflight intermediate range, the temperature control system is not calling for additional heating. As shown in Figure 14-28, the heat control modulating valve is closed, indicating that one pass through the compressor is heating the air sufficiently. The compressed air flows through the turbocompressor flow control valve and the turbocompressor check valve, on through to the air-to-air heat exchanger.

The heat exchanger cooling air modulating valve is in the modulating position, indicating that the temperature control system is calling for slightly cooler air than that coming from the turbocompressor. The heat exchanger cooling air modulating valve allows ram air to flow through the air-to-air heat exchanger and be dumped overboard. This ram air extracts heat from the heated air as



Air Conditioning System Schematic - Inflight Heating
Figure 14-27



Air Conditioning System Schematic - Inflight Intermediate
Figure 14-28

required by the temperature control system. The compressed and conditioned air leaves the air-to-air heat exchanger and flows through the Freon evaporator. The compressed and conditioned air receives no cooling in the Freon evaporator because sufficient cooling has been obtained in the air-to-air heat exchanger, so the Freon compressor is shut down. The compressed air flows through the open conditioned air emergency shutoff valves and conditioned air check valves. The conditioned air is then routed by the air distribution subsystem to the cabin and flight compartments. The conditioned air delivered to the cabin is heated, if necessary, by the cabin dual heaters. Cabin heater operation is determined by the temperature equalization control system to maintain the air temperature between the forward and aft cabin areas within a maximum differential of 2 degrees F (1.1 degree C).

Inflight Cooling

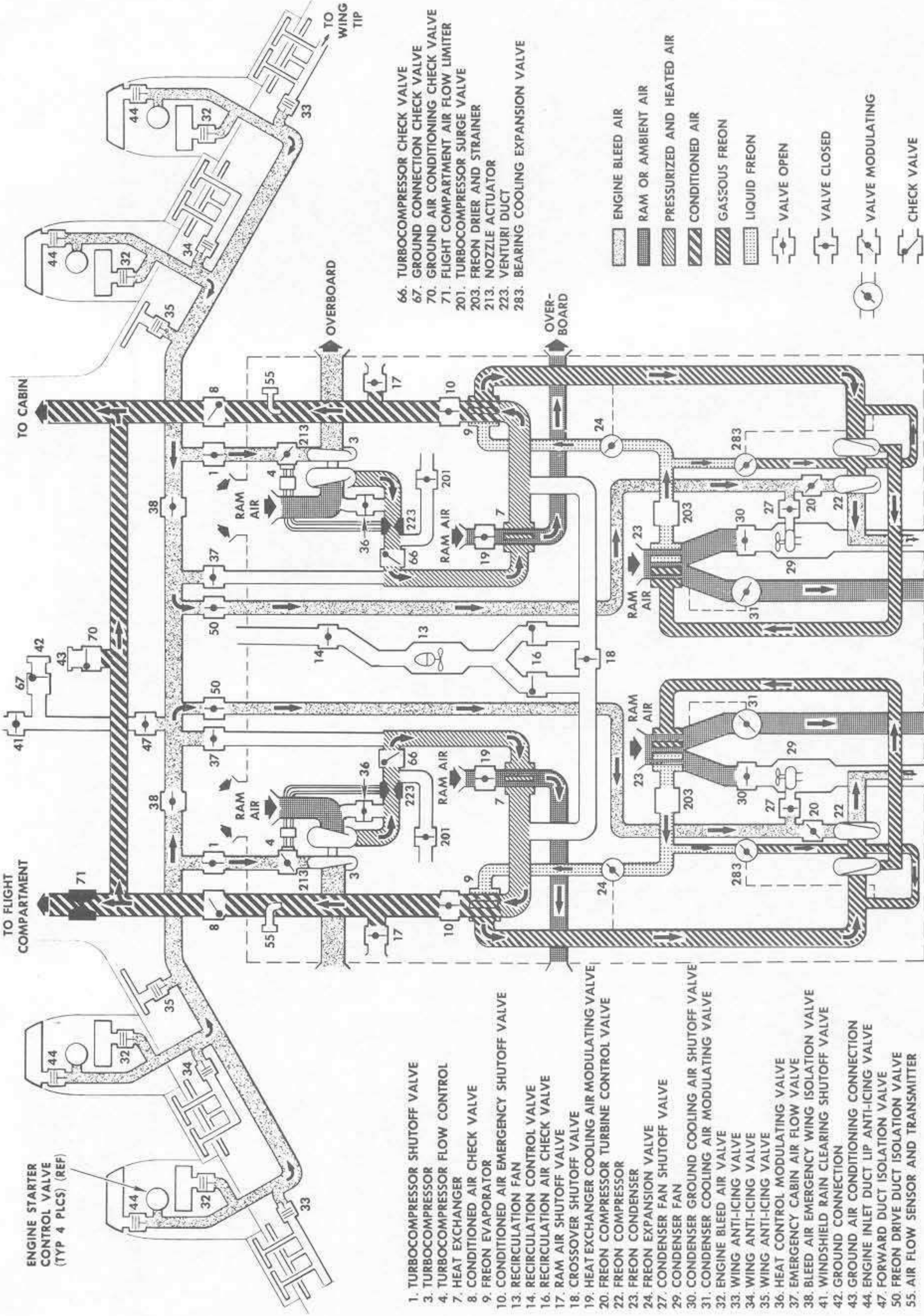
When the temperature control system is calling for cooling, the heat control modulating valve is closed. The ram air is compressed in the turbocompressor and partially heated by compression. The compressed air flows through the turbocompressor flow control and the turbocompressor check valve to the heat exchanger. As shown in Figure 14-29, the heat exchanger cooling air modulating valve is full-open, because the temperature control system is calling for more cooling than the air-to-air heat exchanger can provide.

The compressed air flows through the air-to-air heat exchanger to the Freon evaporator. Because the heat exchanger is operating at full capacity and the temperature control system is calling for more cooling, the Freon refrigeration system is turned on. When the Freon compressor is operating, the Freon gas is passed through the compressor resulting in an increase in its temperature and pressure.

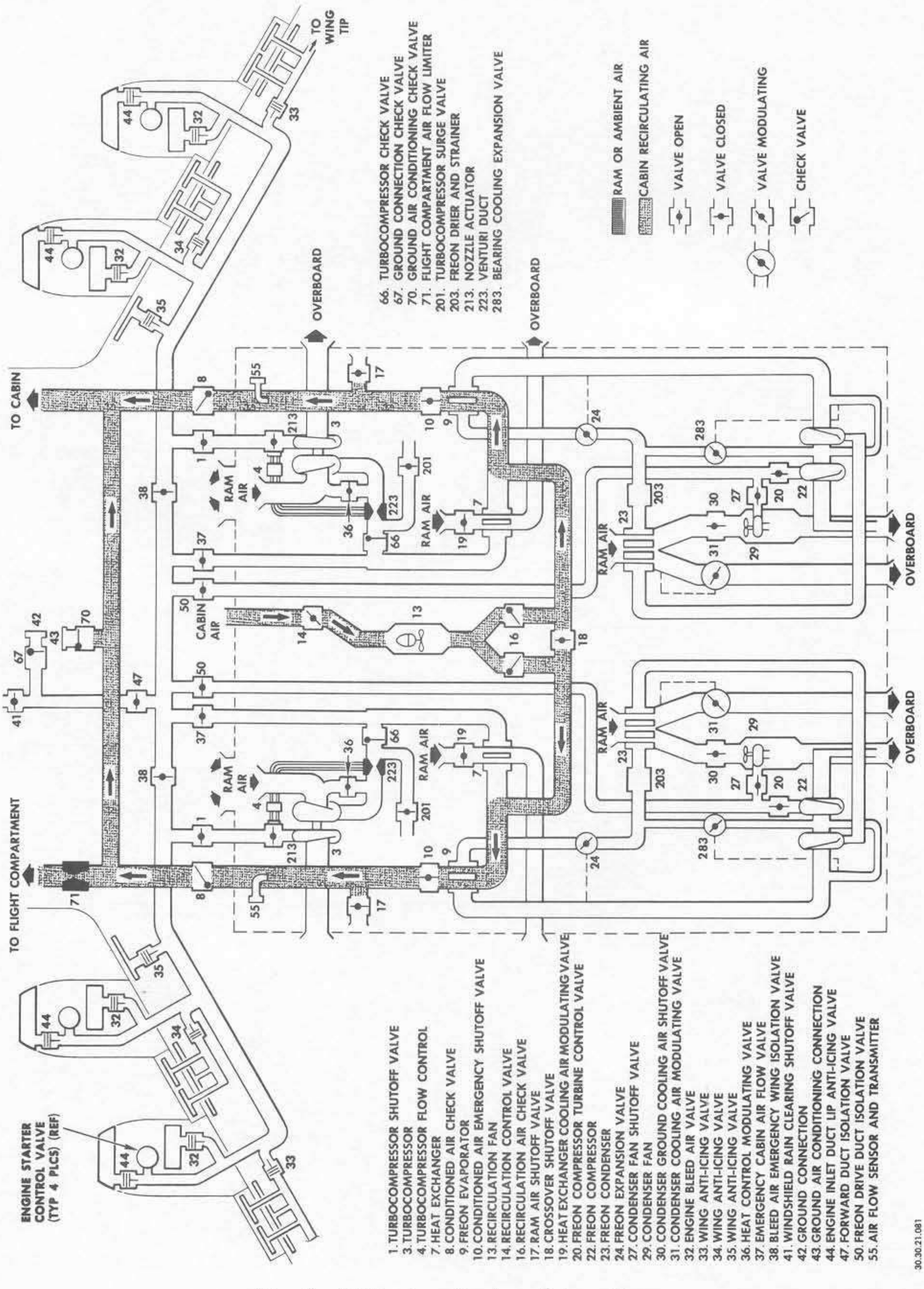
The compressed Freon gas then flows from the compressor and enters the condenser where the heat is removed by cool ram air blowing across the condenser. As the heat is removed from the gaseous Freon, it condenses into a liquid. The ram air used for cooling is then dumped overboard. The liquid Freon from the condenser enters the evaporator through the Freon expansion valve. Heat is absorbed by the Freon during evaporation. This heat is extracted from the air flowing to the cabin and flight compartments thus, the temperature of the air is reduced. The Freon gas in the evaporator then flows back through the compressor where the cycle is repeated.

Ground Heating

Ground heating can be divided into two phases. The primary phase, illustrated in Figure 14-30, provides comfortable temperatures by recirculating the cabin air. If this does not provide sufficient heat, the temperature control system will stop cabin air recirculation and start the pressurization turbocompressors. The turbocompressors heat the air as it is recirculated through the compressor. The amount of heating is controlled by the heat control modulating valve. When the engines are shut down, a ground source of air must be connected to drive the turbocompressors. When the turbocompressors are operating, ground heating is identical to inflight heating as shown in Figure 14-27.



Air Conditioning System Schematic -
Inflight Cooling
Figure 14-29



Air Conditioning System Schematic - Ground Heating
Figure 14-30

Ground Cooling

With the airplane on the ground, the engines not operating, and the control system calling for maximum cooling, the system works as follows: As shown in Figure 14-31, the turbocompressors are inoperative, the recirculation fan is on, and the recirculation control valve is open. The cabin air is moved by the recirculation fan to the Freon evaporator. The temperature control system is calling for more cooling or maximum cooling; consequently, the Freon refrigeration subsystem is operating. The cabin air flows through the Freon evaporator and is cooled to the temperature scheduled by the temperature control system. The cooled air leaves the evaporator and is distribured to the cabin and flight compartments. If cooling is required when the airplane is on the ground with the engines shut down, a ground cart is required to supply air (to simulate engine bleed air) to drive the Freon packages.

POWER SOURCES

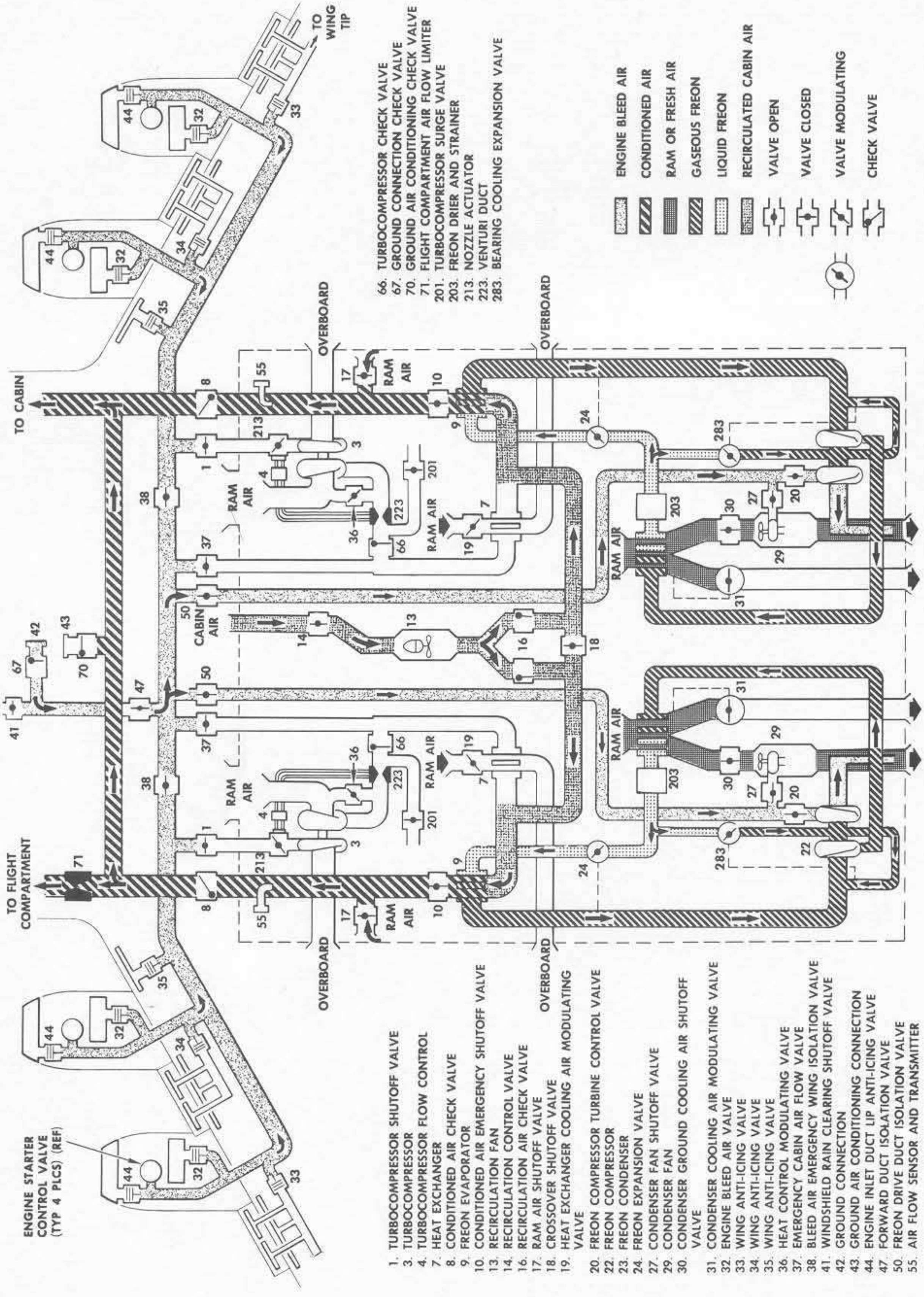
Power to operate the air conditioning and pressurization system is furnished by bleed air from the four turbojet engines and by the ac and dc electrical systems. For detailed information, consult the WIRING DIAGRAM MANUAL.

Engine Bleed Air
- Turbocompressors
- Alternate pressurization
- Freon compressors and condenser fans

Non-Essential Ac Bus System
- Flight deck Freon package control components
- Electronic temperature control, flight deck

Essential AC Bus System
- Cabin Freon package control components
- Electric heaters, cabin system
- Electronic temperature control, cabin
- Ram air shutoff valves
- Crossover shutoff valve
- Cabin and flight deck emergency air shutoff valves
- Recirculating blower
- Electronic equipment cooling fan

28-Volt DC Bus System
- Warning lights
- Temperature and airflow indicators
- Turbocompressor shutoff valves and electronic cooling valve
- Alternate bleed air shutoff valves
- Controls for electric heaters, Freon refrigerations units, cabin air recirculating system, electronic equipment cooling fan, and electronic compartment cooling valve
- Cabin pressure regulators



Air Conditioning System Schematic -
Ground Cooling
Figure 14-31

15

BLEED AIR AND ADVERSE WEATHER SYSTEMS

Section 15

BLEED AIR AND ADVERSE WEATHER SYSTEMS

TABLE OF CONTENTS

TITLE	PAGE
BLEED AIR SYSTEM	15-1
Bleed Air Sources and Flow Routing	15-1
Bleed Air Control Switches	15-1
Bleed Air Control System Indicator Lights	15-1
EXCESS HEAT AND ISOLATION SYSTEM	15-4
Excess Heat and Isolation System Operation	15-4
LEADING EDGE AND DUCT SPACE TEMPERATURE SYSTEM	15-6
Leading Edge and Duct Space Temperature System Operation	15-6
WING SPAR VENTILATION SYSTEM	15-7
Ventilation Duct Lip Anti-Icing	15-7
ADVERSE WEATHER SYSTEMS	15-7
WING AND LEADING EDGE SLATS ANTI-ICE SYSTEM	15-10
Wing Anti-Ice Valves	15-10
Wing Anti-Ice Switches	15-12
Wing Anti-Ice Indicator Lights	15-12
TAIL DE-ICING SYSTEM	15-12
Tail De-Icing Overheat Protection	15-12
Heat Sensors	15-12
Tail De-Icing Controller Unit	15-14
Controller Overload Protection	15-14
Tail De-Icing Control Switch	15-14
Tail De-Icing Malfunction Indicator	15-14
WINDSHIELD ANTI-ICE AND ANTI-FOG SYSTEMS	15-15
Windshield Overheat Protection	15-15
Windshield Anti-Ice-Fog Control Switches	15-15
PITOT TUBE ANTI-ICING SYSTEM	15-17
ELEVATOR BALANCE BOARD AIR INLET ANTI-ICING SYSTEM	15-17
ENGINE ANTI-ICING SYSTEM	15-17
Engine Duct Lip Anti-Icing	15-17
Engine Anti-Icing	15-17
Aft Fan Anti-Icing	15-20
Engine Anti-Ice Control Switches	15-20
Engine Anti-Ice Malfunction Indicators	15-20

TABLE OF CONTENTS (CONT)

TITLE — PAGE

WINDSHIELD RAIN CLEARING SYSTEM.......... 15-20
Rain Clearing Control Switch.......... 15-20
Overheat Protection.......... 15-22

ADVERSE WEATHER AND BLEED AIR SYSTEMS POWER SOURCES.......... 15-22
115-Volt, 400 cps, ac bus system.......... 15-22
28-Volt, dc bus system.......... 15-22

BLEED AIR AND ADVERSE WEATHER SYSTEMS

BLEED AIR SYSTEM

The engine bleed air system provides hot air to operate air conditioning and pressurization equipment, wing and leading edge slat anti-icing, engine anti-icing, windshield rain clearing and alternate cabin pressurization. Two overheat protection systems are considered a part of the engine bleed air system, the Leading Edge and Duct Space Temperature system and the Excess Heat and Isolation system.

Bleed Air Sources and Flow Routing

Bleed air is obtained from the seventeenth stage of the engine compressor at 237 psig maximum pressure and 427 degrees C (800 degrees F) maximum temperature. The hot air is manifolded from four ports in each engine into a single duct (see Figure 15-1). A bleed air pressure regulator shutoff valve limits down stream bleed air pressure to 40 psig. It also acts as a check valve to permit engine starting and as a shutoff valve for restricting bleed air at the source. The bleed air is routed through the pod and pylon into the wing bleed air manifold attached to the front spar. The right and left wing bleed air manifolds are connected by a fuselage crossover duct. A bleed air emergency wing isolation valve is located at each end of the fuselage crossover duct. The windshield rain clearing bleed air duct leads forward from the fuselage crossover duct. A windshield rain removal shutoff and isolation valve is provided close to the crossover duct.

Bleed Air Control Switches

Four two-position OPEN-CLOSE toggle switches are installed on the engine bleed air control panel located on the pilots' overhead switch panel. The switches control the operation of the engine bleed air regulator shutoff valves. In the OPEN position, the valves are opened unless closed by overriding signals from the excess heat and isolation system. In the CLOSE position, the valves remain spring-loaded closed regardless of the demands of any airplane system. Engine bleed air switches must be OPEN to use the anti-icing, de-icing, rain clearing and various air conditioning systems (see Figure 15-2).

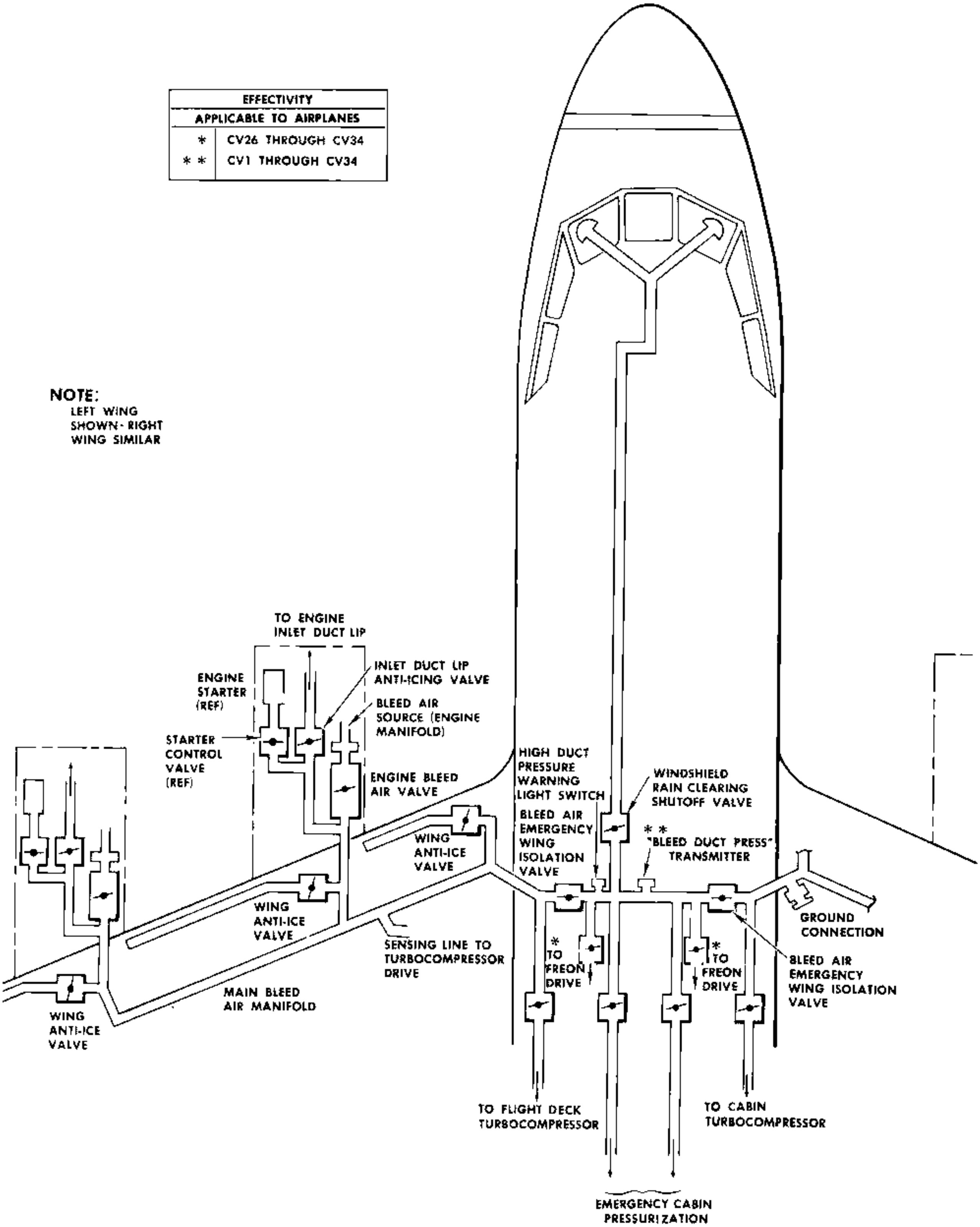
Bleed Air Control System Indicator Lights

Four amber warning lights, labelled CLOSED, are located above the four engine bleed air control switches. The lights will illuminate when the engine bleed air switches are in the OPEN position and the bleed air valves close due to malfunction or excess heat signals. A single red HIGH DUCT PRESSURE warning light, located above the four amber lights, illuminates if bleed air pressure in the crossover duct increases above 60 psig.

NOTE: On airplanes equipped with a bleed air pressure transmitter, the transmitter is located on the fuselage bleed air crossover duct. A pressure gage is located on the pilots' overhead switch panel to provide bleed air pressure readings for flight crew monitoring.

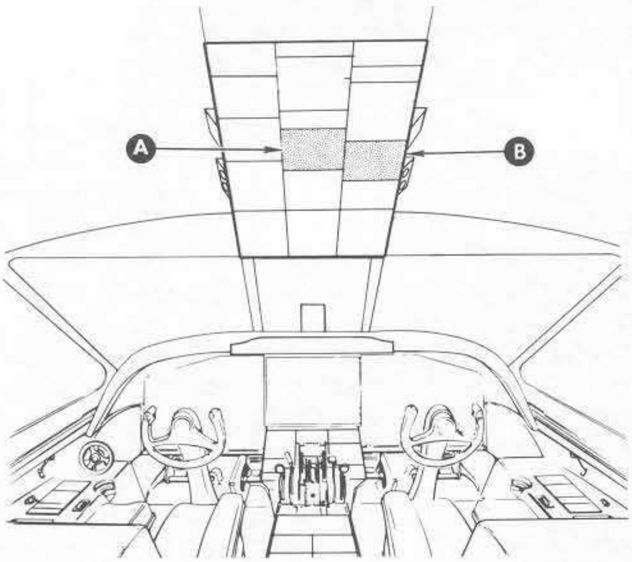
EFFECTIVITY	
APPLICABLE TO AIRPLANES	
*	CV26 THROUGH CV34
* *	CV1 THROUGH CV34

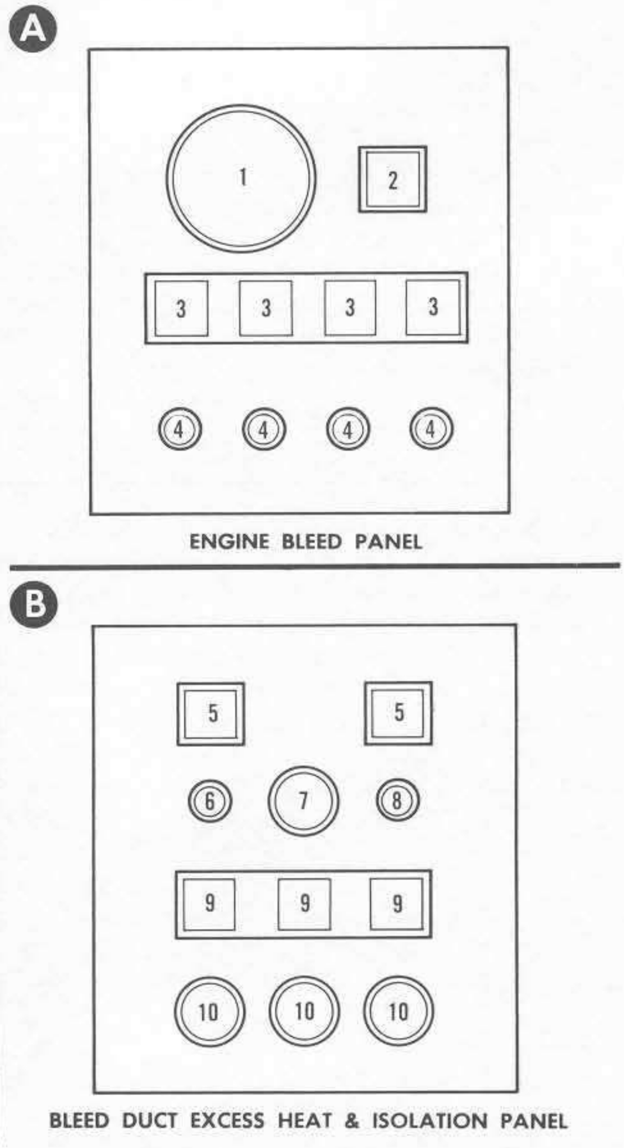
NOTE: LEFT WING SHOWN - RIGHT WING SIMILAR



30 02 36 017

Bleed Air Supply System
Figure 15-1





* 1. BLEED AIR DUCT PRESSURE INDICATOR
2. HIGH DUCT-PRESSURE WARNING LIGHT
3. ENGINE BLEED AIR VALVE VALVE-CLOSED LIGHT
4. ENGINE BLEED AIR VALVE OPEN-CLOSE SWITCH
** 5. FREON DRIVE DUCT EXCESS HEAT WARN LIGHT
** 6. LH FREON DRIVE AUTO-OFF-OPEN SWITCH
7. TEST SWITCH
** 8. RH FREON DRIVE DUCT EXCESS HEAT WARN LIGHT
9. LEFT WING, FUSELAGE AND RIGHT WING EXCESS HEAT WARN LIGHTS
10. LEFT WING, FUSELAGE AND RIGHT WING BLEED DUCT ISOLATION VALVE AUTO-OFF-OPEN SWITCHES

EFFECTIVITY	
APPLICABLE TO AIRPLANES	
*	CV1 THROUGH CV34
**	CV26 THROUGH CV34

30.30.36.007A

Engine Bleed Air Control Panel
Figure 15-2

EXCESS HEAT AND ISOLATION SYSTEM

The excess heat and isolation system is a protective system that monitors the ambient temperatures in the wing duct spaces ahead of the front spar and around the rain clearing duct in the fuselage area. Three continuous temperature-sensitive loops are provided. Two follow the basic bleed air duct routing, one loop in each wing. The wing loop paths follow the ducting into the engine pod and pylon areas. The wing temperature-sensitive loops are set to react to a temperature of 154 degrees C (310 degrees F) or higher. The third temperature-sensitive loop follows the fuselage crossover and rain clearing bleed air ducts and is set to react to a temperature of 154 degrees C (310 degrees F) or higher.

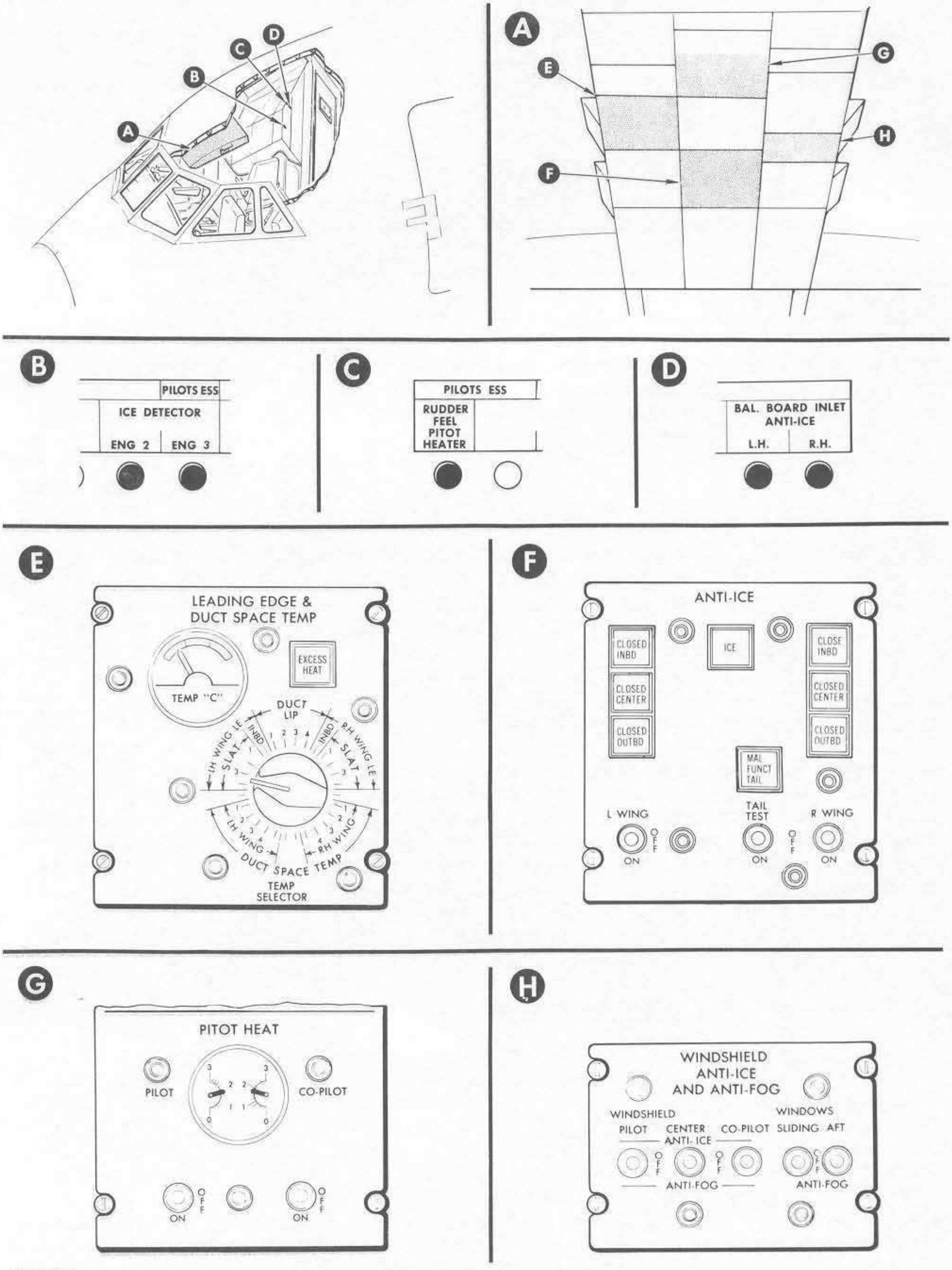
The wing portion of the excess heat and isolation system is designed for in-flight operation. The wing temperature-sensitive loops are rendered inoperative while on the ground. The temperatures at which the system will react are set higher than the reaction temperatures of the Leading Edge and Duct Space Temperature system. In normal operation, the latter system will indicate high temperature troubles well in advance of the indications from the excess heat and isolation system. The intent of the design is that the flight crew will be alerted to a high temperature condition, and be able to locate the trouble spot, by using the Leading Edge and Duct Space Temperature system. Should the flight crew miss an over-temperature condition, through other requirements on their time, the excess heat and isolation system will provide warning and automatically isolate the complete area involved in the over-temperature (see Figure 15-3).

Excess Heat and Isolation System Operation

The Excess Heat and Isolation system control panel is located on the pilots' overhead switch panel. Three three-position AUTO-OFF-MAN OPEN switches are located on the panel. Directly above each switch (L WING - FUS - R WING) are three red warning lights labelled EXCESS HEAT. Each warning light is connected to its respective temperature-sensitive loop. During normal operation, all three switches are placed in the AUTO position. An over-temperature condition in the alarm range of any loop will result in illumination of the associated warning light. If a wing loop is concerned, the engine bleed air valves for both engines in that wing and the emergency isolation valve will close, thus shutting off bleed airflow through the affected wing.

NOTE: As a result of this valve action, the engine bleed air warning lights for the engines concerned will illuminate. If the airplane anti-icing systems are operating, the engine anti-ice warning lights for the engines concerned and the three anti-ice wing valve warning lights for the wing concerned will also illuminate.

NOTE: On airplanes equipped with pneumatic Freon drives, five switches are provided. Two additional loops, one for each Freon drive duct in the under wing plenum chamber, are set to react at 154 degrees C (310 degrees F) and, on overheat, will automatically close the bleed air source valve to the applicable Freon drive duct.



Ice Protection System Controls
Figure 15-3

If an over-temperature condition in the fuselage crossover or rain clearing duct is involved, the EXCESS HEAT light will illuminate. At the same time, both wing emergency isolation valves will close. The engine bleed air valves will not close. The turbocompressors, being supplied from outboard of the isolation valves, will remain in operation.

NOTE: As a result of this action, if the airplane adverse weather systems are operating, the windshield rain clearing duct shutoff valve will close from lack of bleed air pressure, thus extinguishing the rain clearing ON light.

Should the external icing conditions be such that manual override of the automatic heat isolation system is desirable, the appropriate Excess Heat and Isolation system control switch can be placed in the MAN OPEN position for short periods of time. Placing any of the switches in MAN OPEN will open all isolation and engine bleed air regulator valves that have been closed automatically. Placing any of the switches in the OFF position will extinguish the appropriate warning lights and position the isolation and bleed air valves concerned to the closed position. Thus in the OFF position, the same valves are closed as are closed in the AUTOMATIC position by an overheat condition. However, the EXCESS HEAT warning light is extinguished.

CAUTION: PLACING AN ISOLATION SWITCH IN THE MAN-OPEN POSITION BYPASSES THE AUTOMATIC OVERHEAT PROTECTION AND EXCESS HEATING OF THE AIRPLANE STRUCTURE MAY RESULT. PLACING THE FUSELAGE SWITCH IN MAN-OPEN WILL OVERRIDE BOTH AUTOMATIC AND MANUAL CLOSING OF THE WING ISOLATION VALVES. THIS WILL MAKE IT IMPOSSIBLE TO ISOLATE ONE WING SINCE CROSSFEED OCCURS FROM THE OTHER WING.

LEADING EDGE AND DUCT SPACE TEMPERATURE SYSTEM

The Leading Edge and Duct Space Temperature system provides a means of monitoring temperatures at 18 places throughout the wing and engine inlet duct lips. Eight of the temperature sensing units, four to each wing, are installed in the wing duct space. One unit senses temperature in the LH wing inboard leading edge and one in the RH inboard leading edge. Two units sense temperatures in the LH slats 1 and 3 and two in the RH slats 1 and 3. One unit senses temperature in each engine duct lip. The wing area thermistors are set at an alarm temperature of 149 degrees C (300 degrees F) and the duct lip thermistors have an alarm temperature of 250 degrees F (121 degrees C).

Leading Edge and Duct Space Temperature System Operation

The Leading Edge and Duct Temperature system control panel is located in the pilots' overhead switch panel. The rotary eighteen-position switch connects each temperature sensing unit to a temperature indicator on the panel. The switch sectors are divided into four major divisions, LH WING LE, RH WING LE, DUCT SPACE TEMP and DUCT LIP. The LH WING LE and RH WING LE are divided into three positions, INBD, and SLAT 1 and 3. The DUCT SPACE TEMP segment divides into four LH WING and four RH WING temperature points. The DUCT LIP segment divides into four positions, one for each engine. An amber EXCESS HEAT warning light is provided and the light will illuminate when any temperature-sensitive element detects a temperature condition in excess of its alarm setting. The particular element sending the signal can be located by checking the temperature of the eighteen points. Manual isolation of the overtemperature section may then be accomplished.

WING SPAR VENTILATION SYSTEM

The face of each wing front spar has a fiberglass baffle attached to the wing leading edge enclosing an area in which the wing bleed air manifold is routed. The areas around the manifolds are vented by ram air entering through two flush type inlets located on the lower surface of the wing adjacent to the inboard engine pylons. Half of the air flows outboard and is discharged through ports on the lower surface of each wing near the tip, while the remainder of the air flows inboard and is ducted into the fuselage and dumped overboard through a discharge port (see Figure 15-4).

Ventilating air removes any fuel vapors which may be present, and limits the maximum temperature on the outside of the bleed air ducting to approximately 135 degrees C (275 degrees F).

Ventilation Duct Lip Anti-Icing

The ventilation inlet duct lip is protected from ice by electric heaters. These heaters are controlled by the right and left wing switches on the anti-ice control panel. They remain on continuously as long as the wing anti-ice system is actuated.

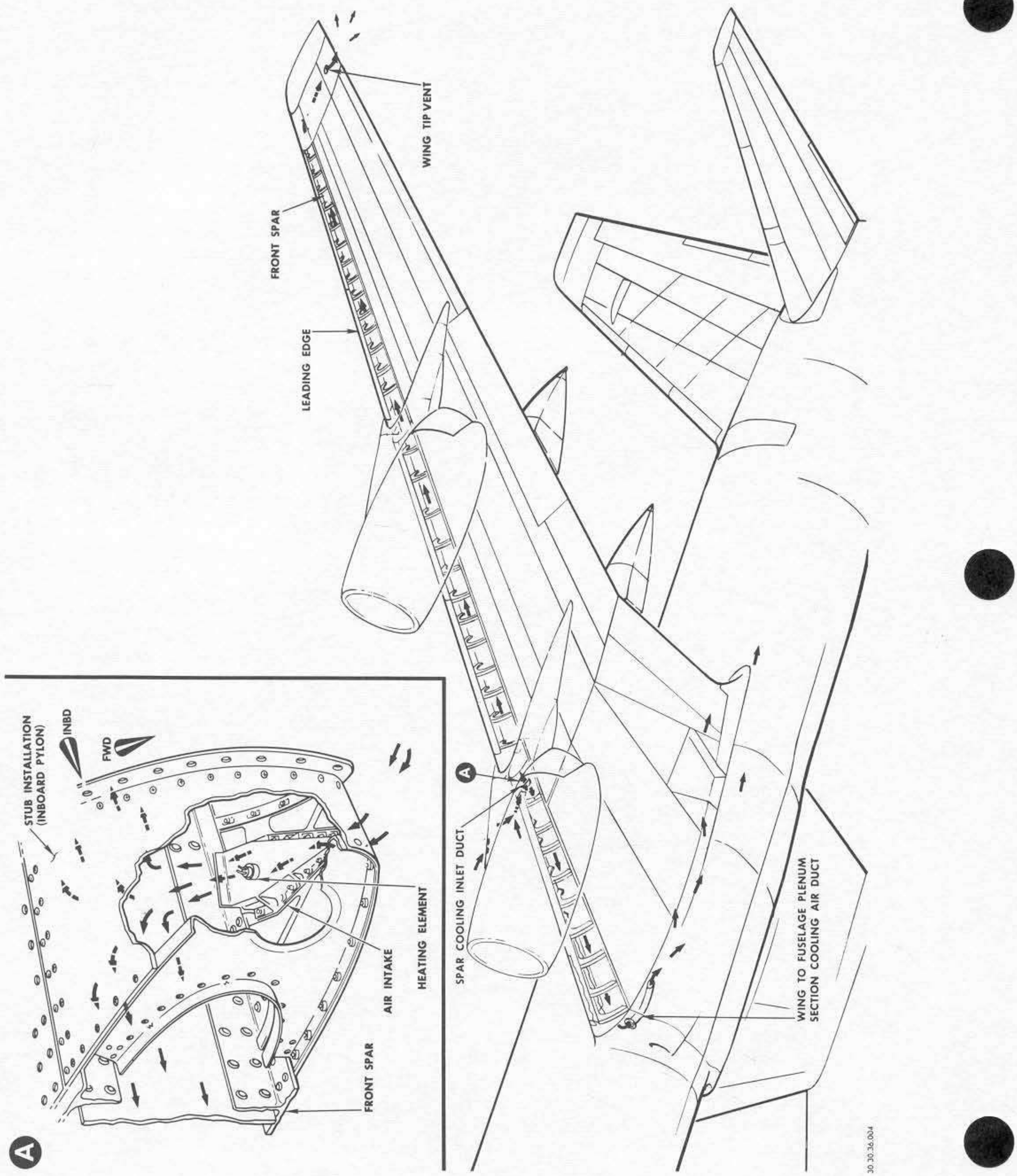
ADVERSE WEATHER SYSTEMS

Anti-icing and de-icing systems, and a windshield rain removal system provide complete adverse weather protection. The ice and rain protection systems are as follows: (Figure 15-5).

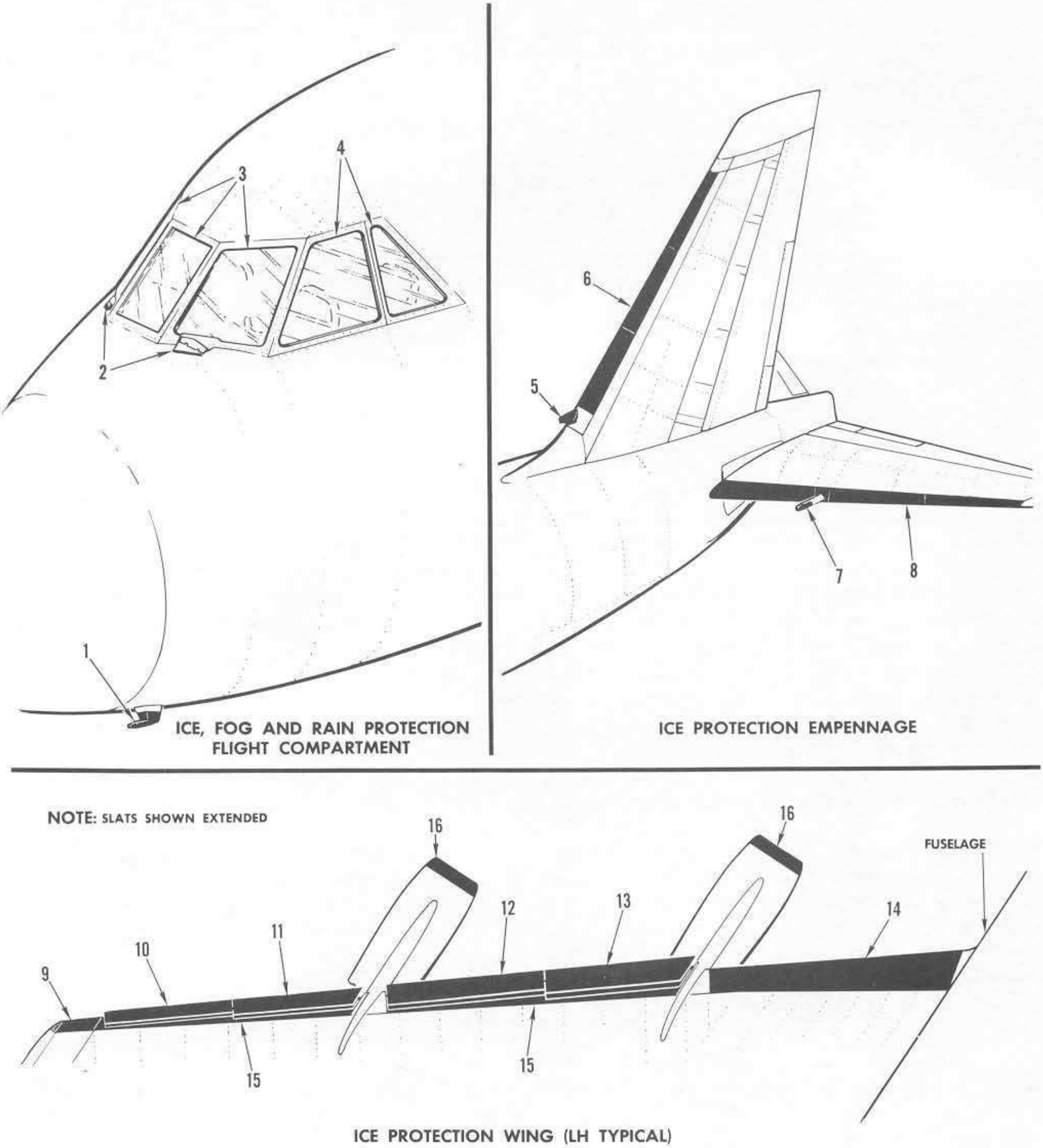
1. Bleed air anti-icing system for the wing and slat leading edges.
2. Bleed air rain clearing system for the windshields.
3. Bleed air anti-ice system for the engine inlet ducts.
4. Electric anti-ice and anti-fog systems for the flight compartment windshields and sliding windows.
5. Electric de-icing for the tail leading edges.
6. Electric anti-ice system for the flight instrument pitot tubes.
7. Electric anti-ice system for the rudder boost "Q" cylinder pitot tube.
8. Electric anti-ice system for the elevator pressure boards air inlets.

Although some systems are called "anti-icing," all systems have de-icing capability.

The wings are protected against ice formation by a double-skin, hot-air, anti-icing system installed in the wing leading edges. The tail de-icing system uses integral, electrically heated blankets on the horizontal and vertical stabilizer leading edges. The center and main windshields are anti-iced by electrically heating a high density transparent conductive coating on the inner surface of the outer glass ply. The main windshields are anti-fogged by electrically heating a low density coating parallel to the high density coating. Anti-fogging



Wing Spar Ventilating System
Figure 15-4



ICE, FOG AND RAIN PROTECTION FLIGHT COMPARTMENT

ICE PROTECTION EMPENNAGE

ICE PROTECTION WING (LH TYPICAL)

1. FLIGHT INSTRUMENT PITOT TUBES ANTI-ICE (TYPICAL 2 PLACES)
2. WINDSHIELD RAIN CLEAR
3. WINDSHIELD ANTI-ICE AND ANTI-FOG
4. SLIDE AND AFT WINDOW ANTI-FOG
5. RUDDER Q FEEL SYSTEM PITOT TUBE ANTI-ICE
6. VERTICAL STABILIZER LEADING EDGE DE-ICE
7. ELEVATOR PRESSURE BOARD AIR INLET ANTI-ICE
8. HORIZONTAL STABILIZER LEADING EDGE DE-ICE (TYPICAL)
9. OUTBOARD WING LEADING EDGE ANTI-ICE (TYPICAL)
10. SLAT NO. 4 ANTI-ICE (TYPICAL)
11. SLAT NO. 3 ANTI-ICE (TYPICAL)
12. SLAT NO. 2 ANTI-ICE (TYPICAL)
13. SLAT NO. 1 ANTI-ICE (TYPICAL)
14. INBOARD WING LEADING EDGE ANTI-ICE (TYPICAL)
15. WING LEADING EDGE ANTI-ICE (TYPICAL)
16. ENGINE AND POD ANTI-ICE (TYPICAL 4 PLACES) (REFER TO CHAPTER 75, AIR)

30.02.30.019

of the sliding and aft windows in the flight compartment is accomplished by electrically heating a low density conductive coating on the inner surface of the center glass ply.

The rain clearing system routes hot air to the windshield area and injects it parallel to the airstream through nozzles. The pitot system probes on each side of the fuselage, "Q" feel pitot tube, and the elevator pressure boards air inlets are thermally anti-iced by electrical heating units integral with the probe heads. The engine air inlet duct lip, engine compressor nose cone and guide vanes and the aft fan air inlet and outlet guide vanes on each engine are anti-iced by the application of hot air.

All control of the anti-icing and de-icing system is manual. Control switches for all the anti-icing and de-icing systems are located on the pilots' overhead switch panel (see Figure 15-3).

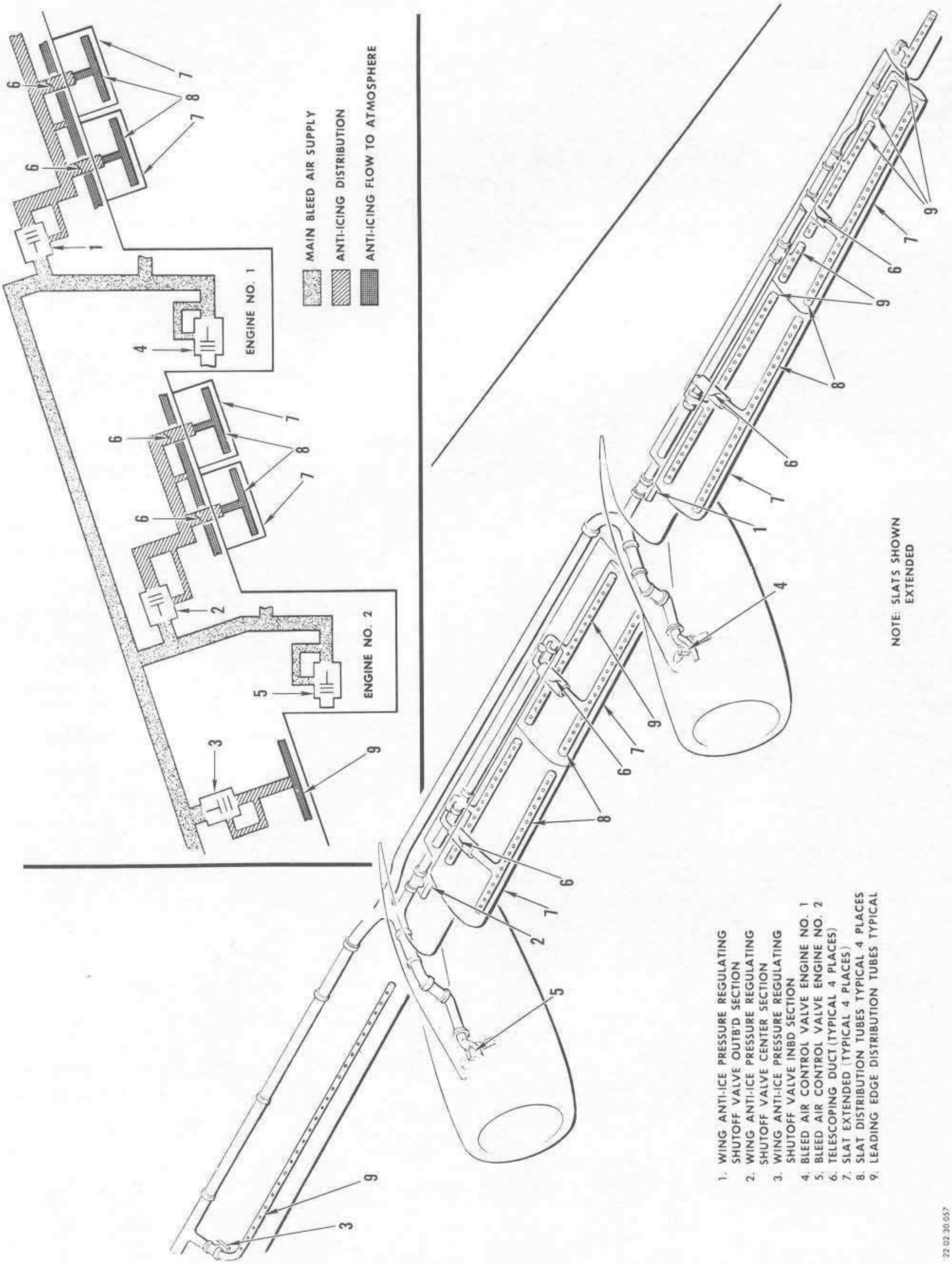
WING AND LEADING EDGE SLATS ANTI-ICE SYSTEM

Each wing leading edge is divided into three sections for anti-icing purposes: an inboard section between the inboard pylon and the fuselage, a center section between the inboard and outboard pylons, and the outboard section between the outboard pylon and the wing tip. Each of the six sections receives bleed air through the six wing anti-ice valves located in ducts connected to the bleed air system. The wing anti-ice valve reduces the 40 psig duct pressure to 13 psig and passes it into the distribution tube where it is ported to the leading edge skin passages. The distribution tube ports are small drilled holes which reduce the 13 psig bleed air pressure to approximately 2 psig. After passing chordwise through the leading edge passages, the air is dumped into the area between the forward and aft baffles in the leading edge just forward to the spar. The air is then dumped overboard through slots located on the lower surface of the leading edge at the front spar (see Figure 15-6).

Bleed air tapped from the bleed air duct prevents formation of ice on the wing leading edge slats. The bleed air passes to piccolo tubes via a telescoping duct and bellows assembly. The tubes extend spanwise forward of the slat baffle; air from the piccolo tubes circulates between the inner and outer skins of the slat and into the area between the leading edge front and rear baffles. Temperature sensors are provided in the leading edge slats.

Wing Anti-Ice Valves

In addition to regulating the flow of bleed air to the distribution tubes, the six wing anti-ice valves can be closed automatically or manually when an overheat condition exists. The six skin patch temperature sensors (those paired with the six patches connected to the leading edge and duct space temperature system) are connected to their associated wing anti-icing valve. The valve control sensor will, upon detecting a skin temperature of 157 degrees C (315 degrees F), cause its associated wing anti-icing valve to close. Upon sensing a temperature drop to 46 degrees C (115 degrees F), the valve control sensor will re-signal the wing anti-ice valve to open. No provisions have been made for manual opening of any wing anti-icing valves against a closing signal from the sensor patches. However, all the valves (3) in each wing can be manually closed by placing the appropriate wing anti-icing switch in the OFF position.



Wing Leading Edge Anti-Ice Air Flow
Figure 15-6

Wing Anti-Ice Switches

Two two-position ON-OFF switches are provided for wing anti-icing control. One is located in the lower left side of the anti-ice panel, placarded L.WING, and one in the lower right side placarded R. WING. In the ON position, the respective wing anti-ice pressure regulator valves are opened and hot air flows into the wing and slat anti-ice air distribution tubes. A WING ICE LTS switch in the EXTERIOR LTS control panel, activates wing leading edge flood lights for night inspection purposes.

Wing Anti-Ice Indicator Lights

Directly above each wing anti-ice switch are three amber lights labeled CLOSED INBD, CLOSED CENTER and CLOSED OUTBD. Each light represents its respective wing anti-ice valve. With the wing anti-ice switches ON, a light will illuminate if a valve fails to open or is closed by an overheat condition.

CAUTION: OPERATING THE WING ANTI-ICE SYSTEM WHILE ON THE GROUND IS NOT RECOMMENDED DUE TO THE POSSIBILITY OF OVERHEATING THE LEADING EDGES, FRONT SPARS OR VENT SCOOPS. OPERATION FOR TEST PURPOSES UP TO 20 SECONDS IS PERMISSIBLE.

TAIL DE-ICING SYSTEM

The tail de-icing system employs integral cyclic-electrical leading edge heating blankets for periodic removal of ice (see Figure 15-7). The cyclic heating blankets are separated by parting strips which are heated continuously when the system is energized. Eighteen heaters are used, each having an area of approximately 400 square inches. A controller cycles power to the individual heaters for a maximum heat-on time of 12 seconds per segment, and a fixed heat-off time of 180 seconds. This sequence provides optimum de-icing performance with minimum runback ice formation behind the heated area.

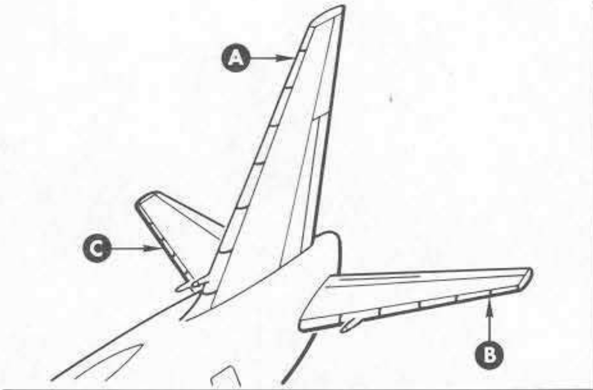
Tail De-Icing Overheat Protection

Overheat protection is provided by a microswitch on the main landing gear which prevents operation of the system on the ground, and by two thermoswitches which remove ac power from the parting strips at ambient air temperature above 7 degrees C (45 degrees F).

Heat Sensors

There are 36 temperature sensors in the heater system, two embedded in each heater pad. Only one sensor in each heater pad is used in the circuit; the others are spares that may be used to replace failed units without the removal of the boot installation. The sensors signal area temperature to the controller.

They are set at a value of 27 degrees C (80 degrees F) and, upon reaching this temperature, will shut down the respective heater pad and switch power to the next heater.



CYCLED AREAS

FIN

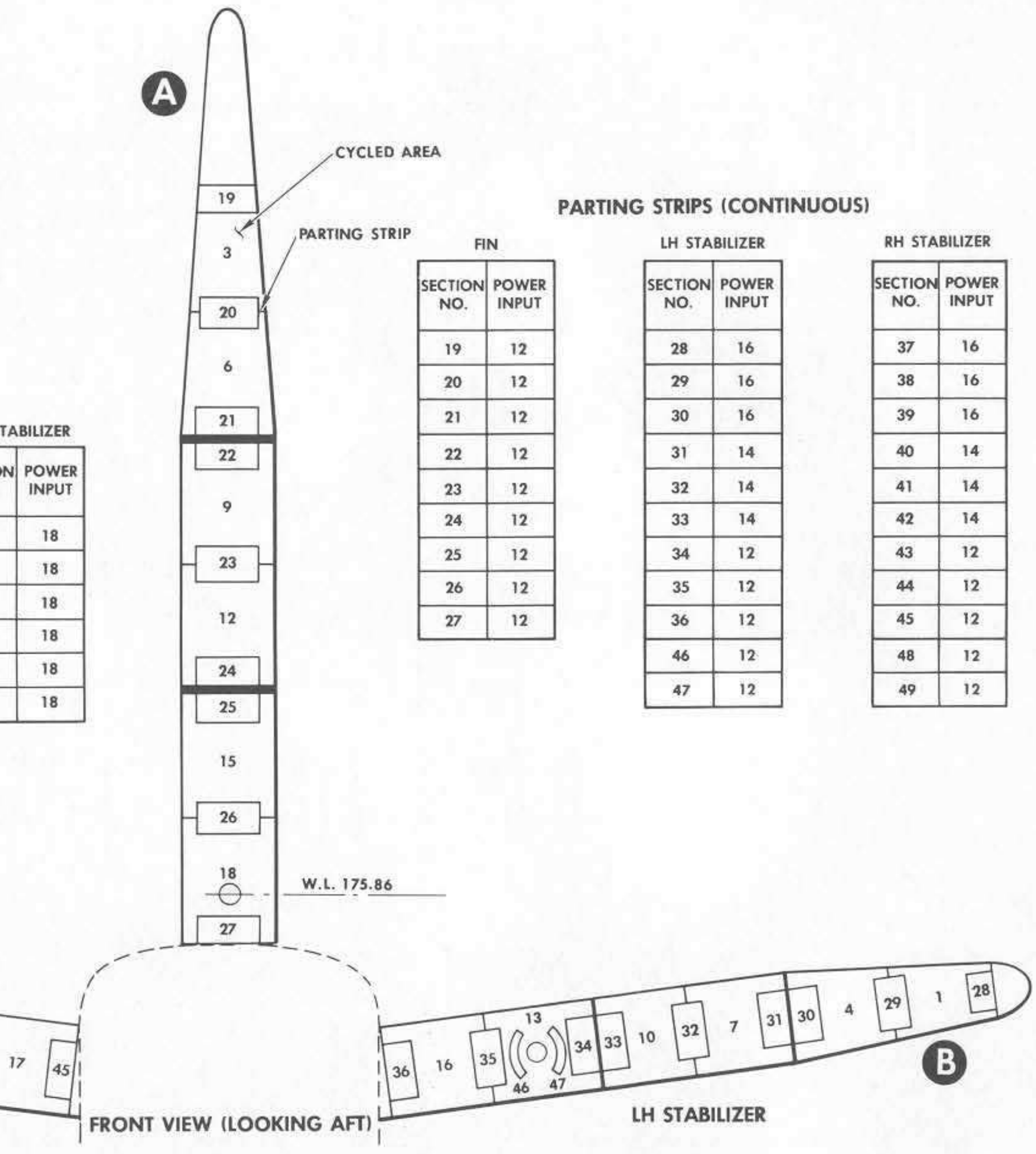
SECTION NO.	POWER INPUT
3	18
6	18
9	18
12	18
15	18
18	18

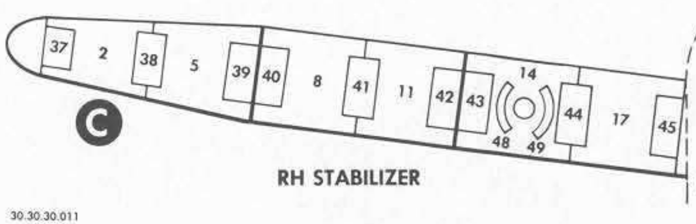
LH STABILIZER

SECTION NO.	POWER INPUT
1	18
4	18
7	18
10	18
13	18
16	18

RH STABILIZER

SECTION NO.	POWER INPUT
2	18
5	18
8	18
11	18
14	18
17	18





PARTING STRIPS (CONTINUOUS)

FIN

SECTION NO.	POWER INPUT
19	12
20	12
21	12
22	12
23	12
24	12
25	12
26	12
27	12

LH STABILIZER

SECTION NO.	POWER INPUT
28	16
29	16
30	16
31	14
32	14
33	14
34	12
35	12
36	12
46	12
47	12

RH STABILIZER

SECTION NO.	POWER INPUT
37	16
38	16
39	16
40	14
41	14
42	14
43	12
44	12
45	12
48	12
49	12

Tail De-Icing Heater Pads
Figure 15-7

Tail De-Icing Controller Unit

An electro-mechanical controller is located in the tail area aft of the pressure bulkhead. Upon receipt of a 28-volt dc control signal (TAIL anti-ice switch, pilots' overhead switch panel), the controller closes a 3-phase, 115-volt ac circuit, energizing the parting strips and connecting the No. 1 heating pad to its appropriate ac terminal. A full voltage is then applied to the No. 1 pad for a maximum of 12 seconds. The voltage then returns to zero, the No. 1 heating pad is removed from the circuit, and the No. 2 heating pad is connected to its appropriate terminal. This process is repeated until all 18 heater pads have been cycled.

The time interval of voltage application to the heater pads will vary from a maximum of 12 seconds to some shorter interval, depending on the ambient temperature. The controller will also select individual sensor signals which are synchronized to correspond with the heating pad selected.

Normal time for a complete cycle is 180 seconds. If, however, due to the ambient temperature the individual heater pad operation times are shortened, the controller will shut off cycling until 180 seconds have elapsed from the start of the heating cycle. However, ac power will continue to be impressed upon the parting strips throughout the entire 180 second period. The unit will cease to operate if ac power fails during a heating cycle. The unit will resume normal operation upon restoration of ac power.

Controller Overload Protection

The controller incorporates overload protection of all three phases of both the parting strips and the heater pads and will disconnect any overloaded phase or phases from the power supply. The parting strip heater circuit protection device provides visual evidence of the overloaded phase. It can be reset by manual means only at the controller itself. The heating pad circuit can be automatically reset by the controller when it selects the first subsequent heating pad that caused the overloaded condition. Fuses are not used in the overload protection devices. Any malfunction of the controller will not result in sustained heating of a heater pad which might cause damage.

Tail De-Icing Control Switch

The tail de-icing system is controlled by a toggle switch located in the anti-ice section of the pilot's overhead switch panel. This switch is labeled TAIL and has three positions, ON-OFF-TEST.

The TEST position provides for ground checking the controller without applying de-icing voltage. An amber light labeled MALFUNCT-TAIL is located directly above the switch.

Tail De-Icing Malfunction Indicator

The MALFUNCT-TAIL warning light will illuminate during the following malfunctions:

1. Overload of any phase in the parting strip circuits by continuous indication while the controller is operating.
2. Overload of any phase of the heater pad circuits by an indication only while the faulty pad is selected for heating.
3. Failure of the controller motor to start.
4. During ground test of the de-icing system.

The controller circuit may be tested prior to flight by placing the control switch in the spring-loaded momentary contact TEST position and observing 18 "blinks" on the MALFUNCT-TAIL light over a 10 to 15 second period of time.

WINDSHIELD ANTI-ICE AND ANTI-FOG SYSTEMS

The windshield anti-icing and anti-fogging systems provide optimum visibility under extreme weather conditions. Five-ply sandwich construction consisting of three layers of glass separated by two layers of vinyl plastic is used in both the windshields and side panels. Windshield anti-icing is accomplished by supplying electrical current to a high-density conductive coating on the inner surface of the outer glass panels. Anti-fogging for the windshields is provided by a parallel low-density coating on the inner surface of the center glass panels. The side panels and sliding windows are provided with anti-fogging only, accomplished by the same method as that used for the windshields (see Figure 15-8). The anti-fogging system is designed to operate at all times during flight. The heat supplied to the low-density coating of the windshields maintains the aft vinyl layers at approximately 38 degrees C (100 degrees F) and tends to increase the structural integrity of the windshields, making them more resistant to impact.

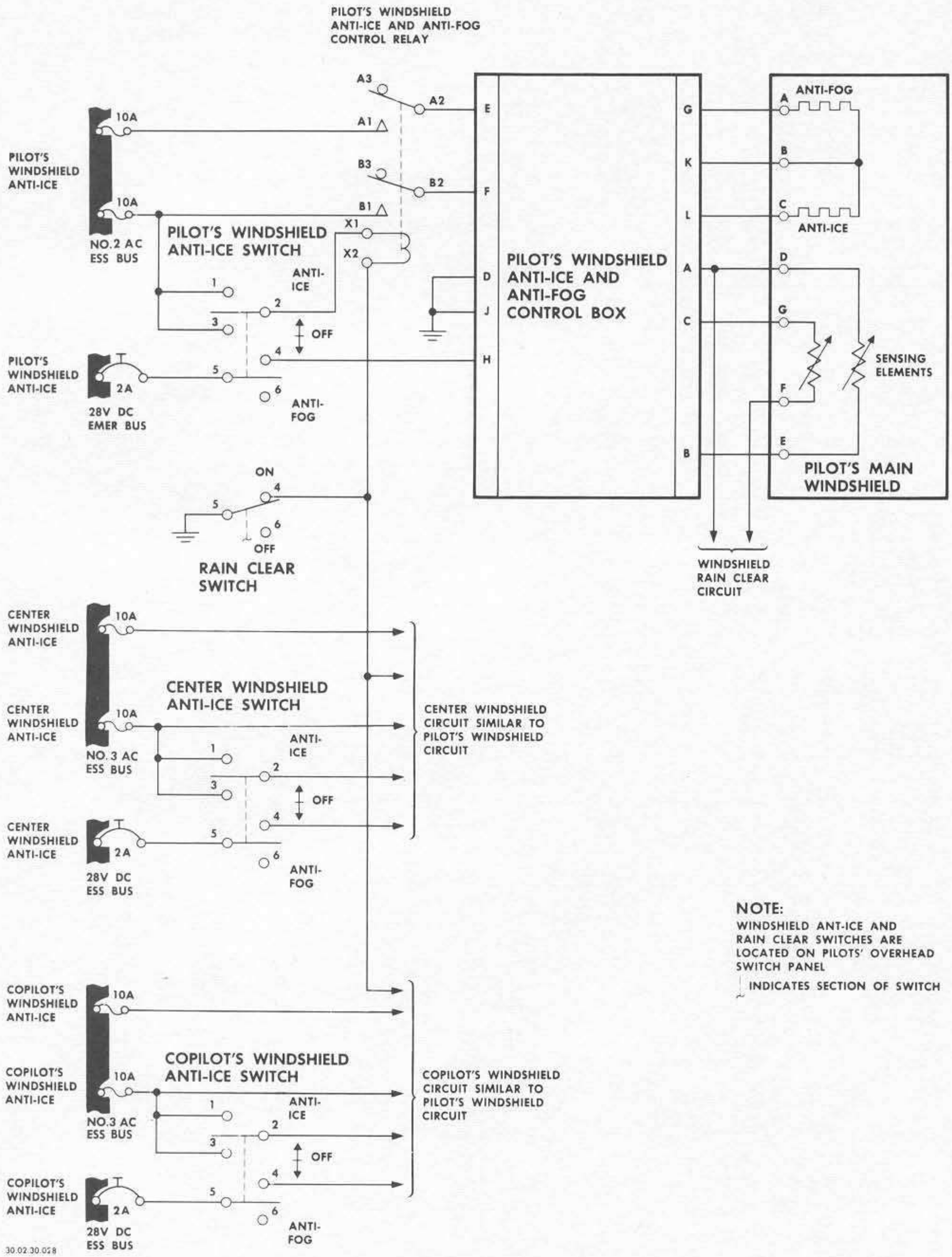
Windshield Overheat Protection

Both the windshield anti-icing and anti-fogging coatings are protected against overheating by sensing elements located in the vinyl layer next to the conductive coating. The sensing element controls the heat input by maintaining a constant temperature, thus precluding an overheated windshield. Protection against overheat by the rain clearing system is also provided.

Windshield Anti-Ice-Fog Control Switches

The windshield anti-icing and anti-fogging control switches are located on the pilots' overhead switch panel. Under normal operating conditions, all windshields and side windows will utilize anti-fogging continuously, except during anti-icing operations. The three windshields are controlled by three three-position switches labeled PILOT, CENTER and COPILOT. The three positions of the switches are ANTI-ICE, OFF and ANTI-FOG. The two sliding windows are controlled by two two-position switches labeled WINDOWS SLIDING AFT. The two positions of the switches are ANTI-FOG and OFF.

When all the switches are placed in the ANTI-FOG position, all of the anti-fog conductive coatings are energized and cause the control boxes to increase the temperature of the inner glass on all panels above the dew point. At the same time the inner layers of vinyl are maintained at a temperature of 27 to 38 degrees C (80 to 100 degrees F).



Windshield Anti-Ice and Anti-Fog System
Figure 15-8

When the three windshield control switches are placed in the ANTI-ICE position, current from the control boxes increases the outside surface temperature of the windshield panels to maintain a temperature of 1.1 degrees C (35 degrees F). The slide windows make no provision for anti-icing.

PITOT TUBE ANTI-ICING SYSTEM

The pitot system probes, located on the lower left and right sides of the fuselage are equipped with integral electric heaters for the prevention of ice accumulation. The heaters are controlled by two ON-OFF toggle switches located in the forward left corner of the pilots' overhead switch panel. Two pitot heater ammeters are located above the switches and indicate when their respective pitot probe heaters are energized (see Figure 15-9). The rudder "Q" feel pitot heater located just above the fuselage fairing on the vertical stabilizer, is anti-iced in the same manner as the flight instrument pitot tube heaters. No control switch is provided for the "Q" feel pitot tube. Opening and closing the RUDDER "Q" FEEL PITOT HEATER circuit breaker is the only circuit control available to the flight crew. An ammeter located on the pilots' overhead switch panel provides positive indication the "Q" feel circuit is energized.

ELEVATOR BALANCE BOARD AIR INLET ANTI-ICING SYSTEM

One elevator balance board air inlet is located on the right-hand and left-hand horizontal stabilizer leading edges. Each inlet is anti-iced by integral heating elements. No control switches are provided for the two inlet heaters. Opening and closing the LH and RH BAL BOARD INLET ANTI-ICE circuit breakers is the only circuit control available to the flight crew.

ENGINE ANTI-ICING SYSTEM

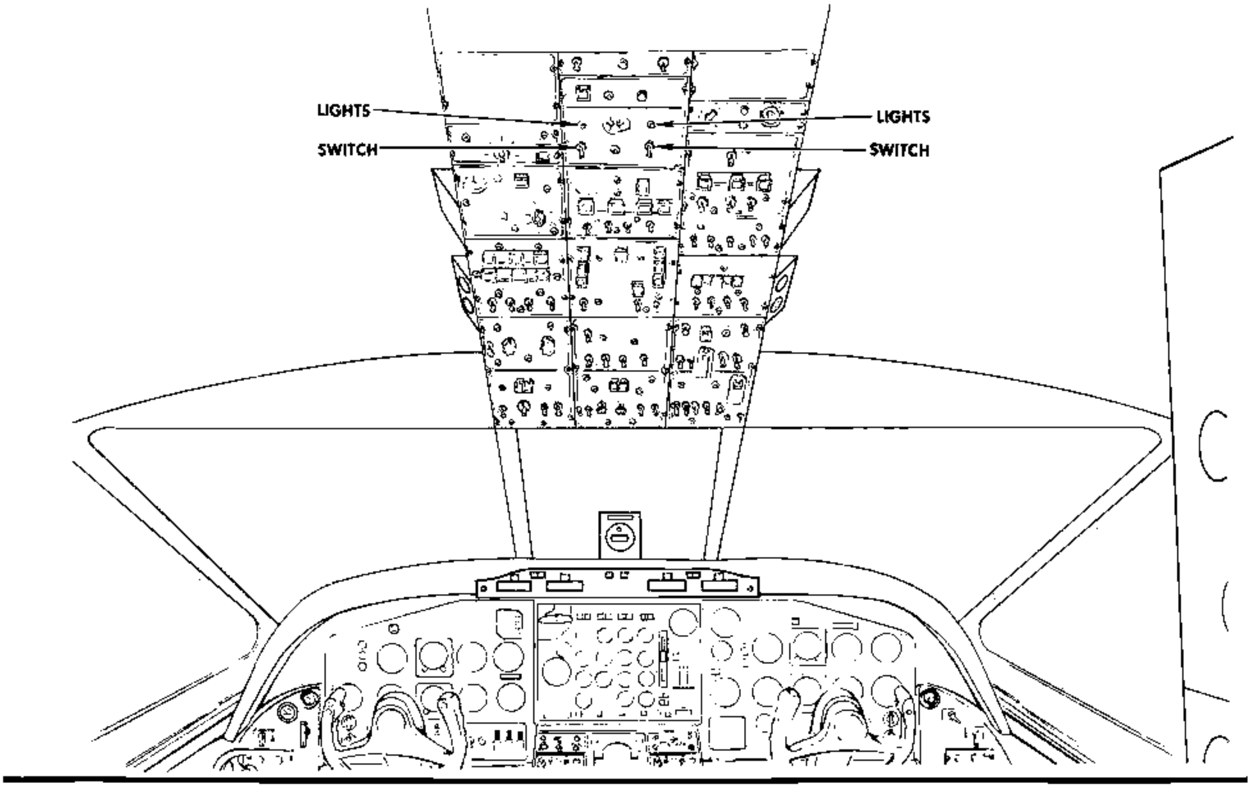
The engine air inlet duct lip, the engine compressor nose cone, the inlet guide vanes and the aft fan inlet on each engine are anti-iced by hot bleed air from the engine (see Figure 15-10).

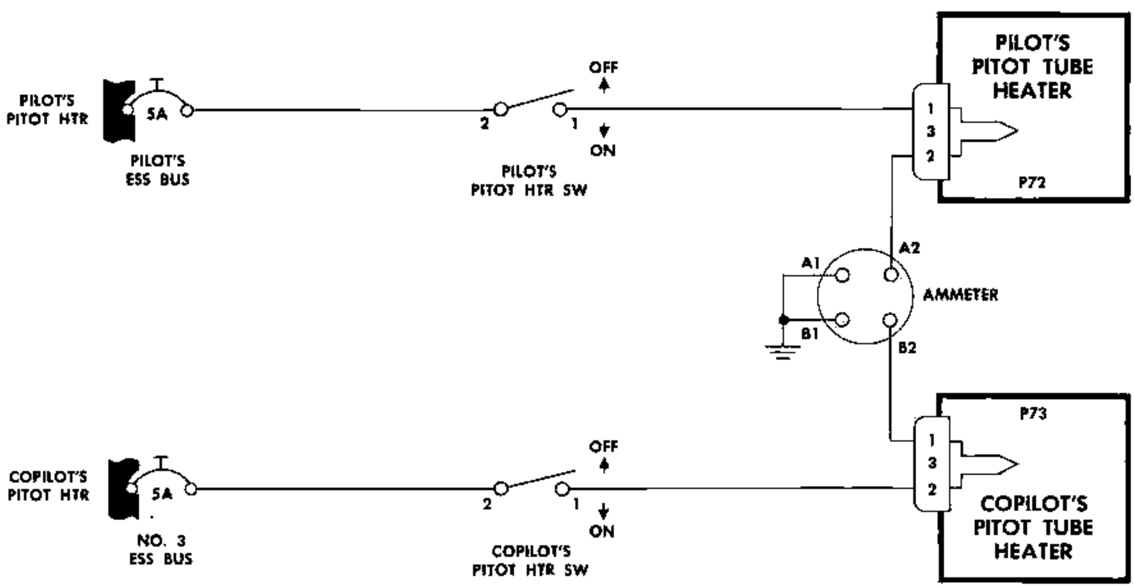
Engine Duct Lip Anti-Icing

Bleed air at 40 psig is directed from the engine bleed air pressure regulator-shutoff valve through a duct into the nose cowl of each engine through a pressure regulating and shutoff valve. From this valve the air discharges forward into the duct lip distribution plenum chamber at a pressure of 11 psig.

Engine Anti-Icing

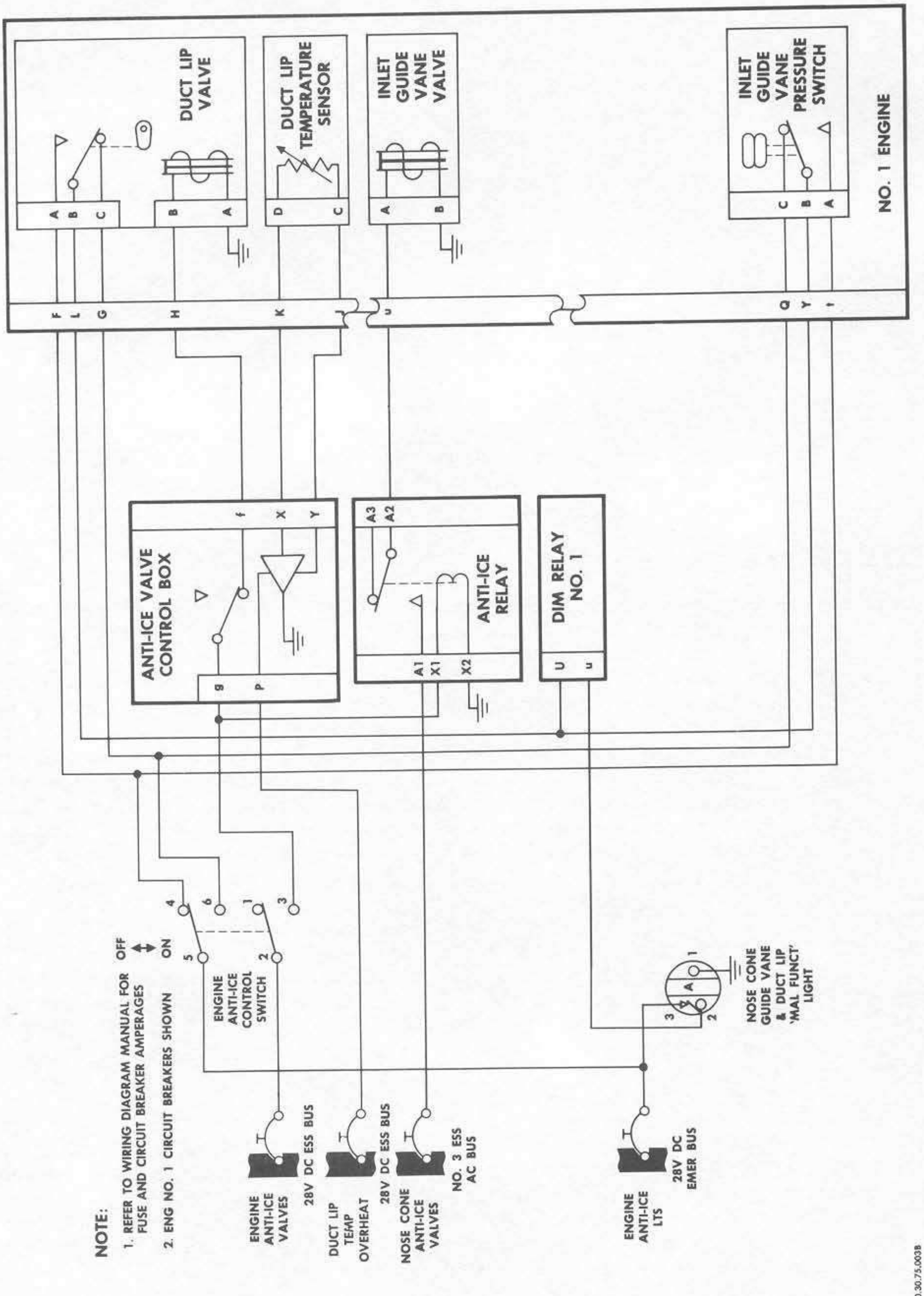
The nose cone and inlet guide vanes of each engine are protected from ice by bleed air taken directly from the engine's bleed air manifold. The bleed air passes through an engine anti-ice temperature regulator-shutoff valve and is then circulated through passages in the guide vanes and nose cone. The engine anti-ice valves are controlled by the same switches that control the duct lip anti-ice valves.





30.30.30.015

Pitot Heat Schematic
Figure 15-9



Number One Engine Anti-Ice Circuit
Schematic - Typical
Figure 15-10

Aft Fan Anti-Icing

A portion of the engine turbine exhaust gas is bled through holes in the inner strut baffles and flows into the middle hot section of the fan front frame. The air then flows outward through the leading and trailing edges of struts No. 2, 3, 4, 6, 7, and 8 of the fan front frame. This air anti-ices the struts and is vented overboard through two holes at the outer end of each strut. Strut No. 1 and No. 5 (the top and bottom struts) do not require anti-icing, since they are covered by the fan inlet ducts, and thus, are not subjected to icing conditions.

Engine Anti-Ice Control Switches

For operation of the engine anti-icing system, the four OFF-ON toggle switches in the engine anti-ice switch panel are placed in the ON position.

Engine Anti-Ice Malfunction Indicators

Four amber warning lights directly above the control switches illuminate MAL-FUNCT if their respective valves malfunction open or closed when the control signals from the switches dictate otherwise.

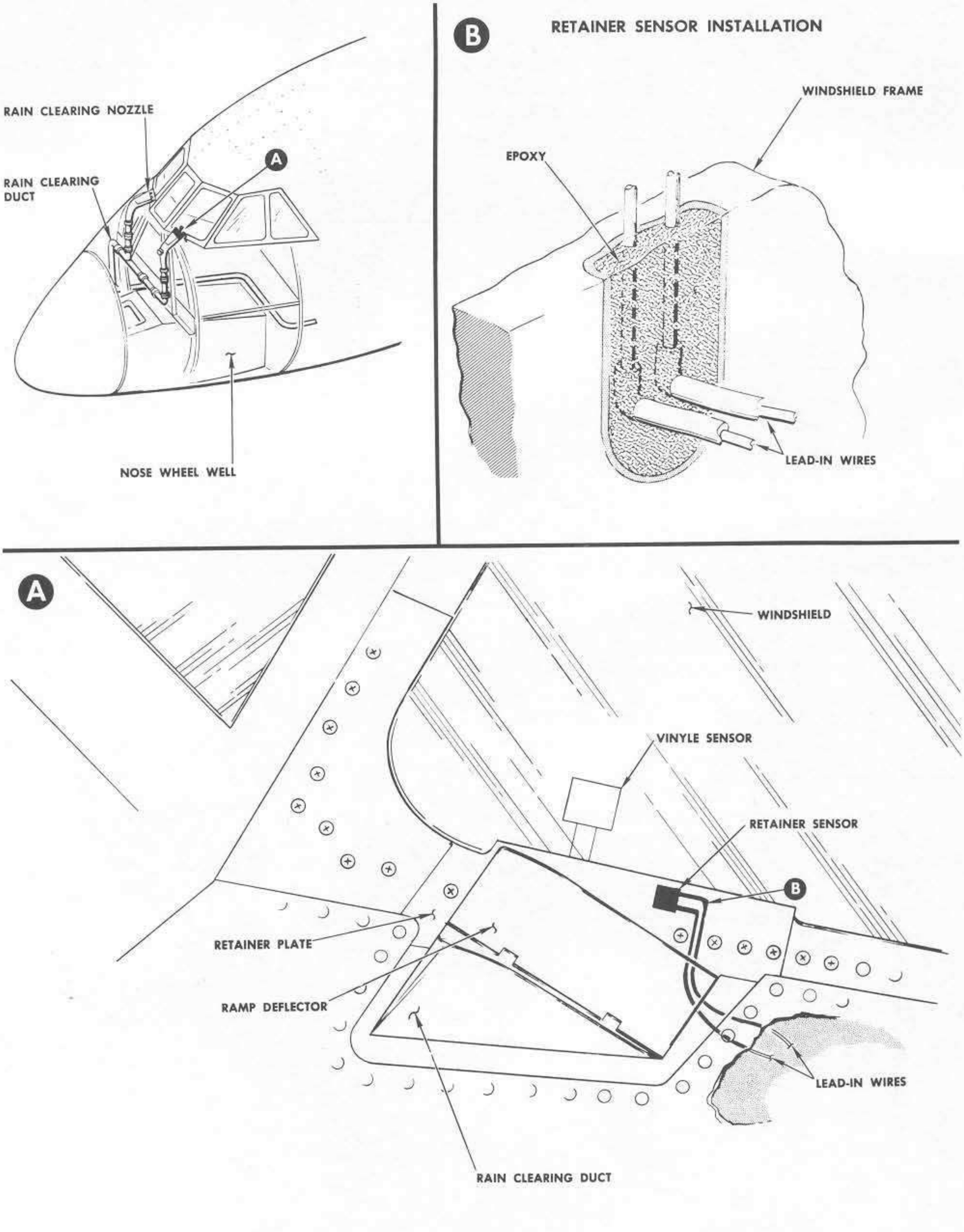
WINDSHIELD RAIN CLEARING SYSTEM

Proper visibility during rain conditions is assured by a bleed air rain clearing system. Bleed air is routed forward under the flight deck floor to the pilot's and copilot's windshields from the fuselage bleed air duct. The air passes through a rain clearing valve which also acts as the fuselage emergency bleed air isolation valve for that portion of fuselage ducting forward of the valve. From the rain clearing valve, the air continues through the duct to the sonic nozzles located at the lower inboard corner of the pilot's and copilot's windshields. If both the rain clearing and the anti-ice systems are turned on, only the rain clearing system will operate on the pilot's and copilot's windshield panels. The anti-ice system will operate on the center panel, which is not served by the rain clearing system.

Rain Clearing Control Switch

The rain clearing system is controlled by an ON-OFF switch located on the pilot's overhead switch panel and by two temperature sensors located in the windshield panels (see Figure 15-11). Hot air may be deflected from the windshield panels by two ramps, operated by pneumatic actuators. The actuators are controlled by vinyl temperature sensors. A temperature in excess of 200°F (93°C) at the vinyl sensor, i.e: during run-up or taxiing, will allow the ramps to raise, deflecting hot air from the windshield.

Placing the RAIN CLEAR switch in the ON position opens the rain clearing control valve. Regulated bleed air pressure at 40 psig is then ejected through sonic nozzles parallel with the airflow over the windshield panels. Because of the high air velocity across the surfaces, a barrier is formed which prevents rain impingement. Any rain that might actually penetrate the barrier will evaporate due to the heat of the windshield boundary layer.



22.30.30.018C

TEMPERATURE SENSOR LOCATIONS

Windshield Rain Clearing System
Figure 15-11

Overheat Protection

In case of an overheated condition, two temperature sensors located in the windshields are set to close the rain clearing valve automatically at 104 degrees C (220 degrees F). A blue indicator light above the rain clearing control switch on the overhead panel illuminates when bleed air is flowing through the rain clearing duct valve.

ADVERSE WEATHER AND BLEED AIR SYSTEMS POWER SOURCES

The buses that supply electric power to the components of the adverse weather system are as follows: (For detailed information, consult the WIRING DIAGRAM MANUAL.)

115-Volt, 400 cps, ac bus system

Pilot's pitot heater
Engine duct lip anti-icing valves, engines No. 1 and No. 4
Anti-fog coating, sliding and aft window panels
Anti-fog and anti-ice coating, left main windshield
Tail anti-ice heaters
Anti-fog and anti-ice coating, center and right windshields
Copilot's pitot heater
Bleed air overheat loop detectors, fuselage and wings
Skin overheat detectors, right and left wings
Engine duct lip overheat detectors
Space overheat detectors, right and left wings
Engine duct lip anti-icing valves, engines No. 2 and No. 3
Rudder "Q" feel pitot heater
Elevator balance board air inlets (2)

28-volt dc bus system

Bleed air overheat warning light, left wing
Bleed air regulator-shutoff valves, engines No. 1 and No. 2
Bleed air regulator-shutoff valve lights, engines No. 1 and No. 2
Bleed air regulator-shutoff valves, engines No. 2 and No. 4
Bleed air regulator-shutoff valve lights, engines No. 3 and No. 4
Bleed air high-pressure light
Bleed air excess heat warning light, fuselage and right wing
Space overheat warning light
Anti-ice ON indicator light
Tail de-ice controller and indicator
Windshield rain clearing bleed air valve
Rain clearing temperature sensors

16 OXYGEN SYSTEM

Section 16

OXYGEN SYSTEMS

TABLE OF CONTENTS

TITLE	PAGE
GENERAL OXYGEN SYSTEMS	16-1
FLIGHT CREW OXYGEN SYSTEM	16-1
Oxygen Flow Path	16-1
Normal Oxygen Mask	16-1
Oxygen Storage Bottles	16-1
Oxygen Storage Bottle Safety Disc	16-4
High Pressure Line Shutoff Valves	16-4
PASSENGER OXYGEN SYSTEM	16-4
Automatic Actuating System	16-4
Passenger Oxygen Masks	16-4
PORTABLE OXYGEN BOTTLES	16-5

OXYGEN SYSTEMS

GENERAL OXYGEN SYSTEMS

Two high pressure, 1800 psi gaseous oxygen systems and various numbers of portable oxygen storage bottles are available for flight crew and passenger use. A diluter-demand type gaseous oxygen system supplies oxygen to all the crew members and the observer. The other gaseous oxygen system is a continuous flow type supplying oxygen to the passengers and the stewardess. The passenger oxygen system is automatic in operation when the cabin altitude exceeds 14,500 (±1,000). (See Figure 16-1.) Portable oxygen bottles provide first aid oxygen for the passengers, and smoke protection or supplemental oxygen for the crew members.

FLIGHT CREW OXYGEN SYSTEM

The flight crew oxygen system consists of two 1800 psi oxygen storage bottles, a system pressure reducer, diluter-demand regulators for all the crew members and observer, and normal oxygen masks and smoke goggles for use with each diluter-demand regulator. A high pressure line shutoff valve is also included which permits the flight crew to divert oxygen from the passenger oxygen system storage bottles to the flight crew oxygen system.

Oxygen Flow Path

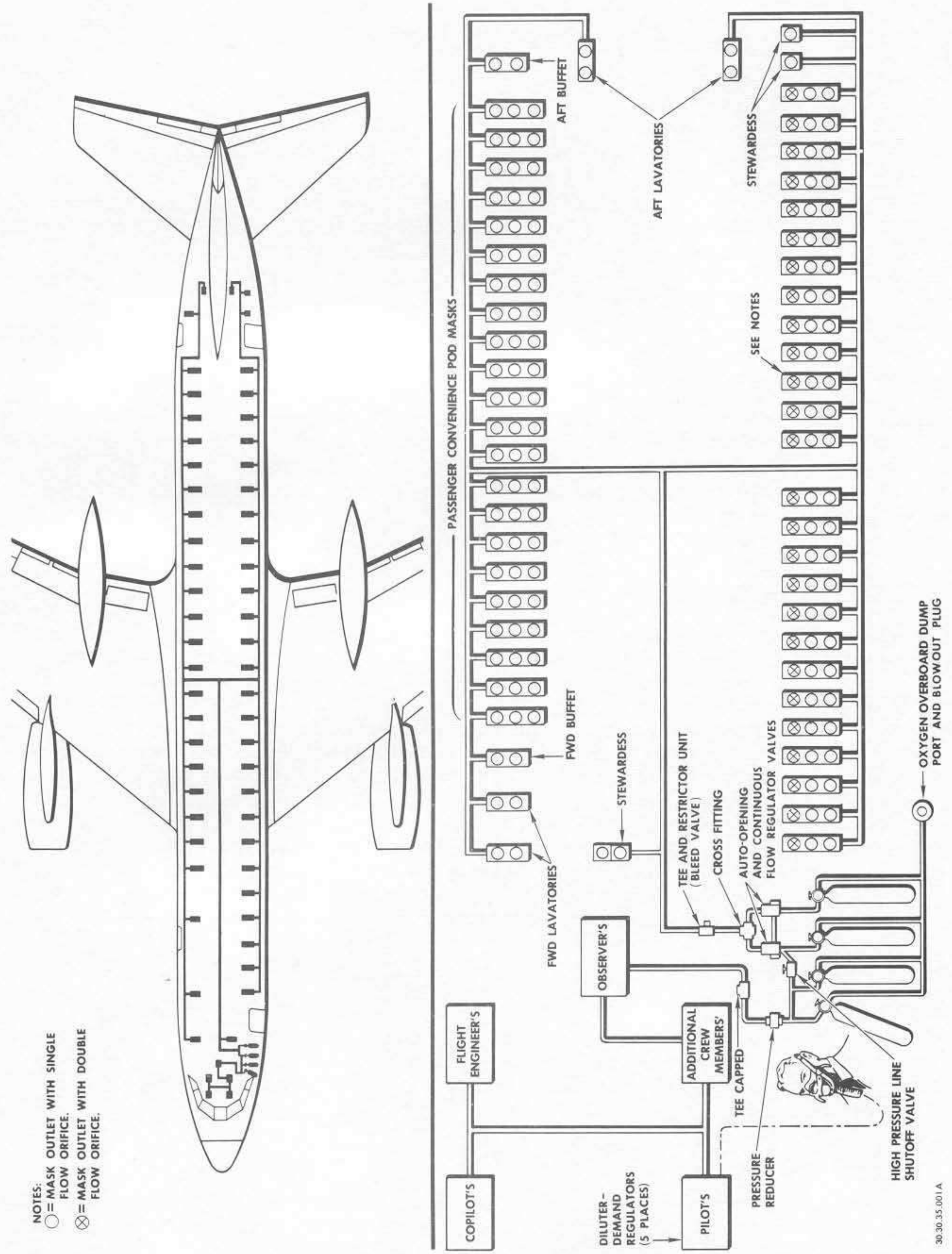
Oxygen flows from the source of supply through the pressure reducer where it is decreased in pressure from 1800 psi to 50-70 psig. From the reducer, oxygen flows to all the crew members' and observer's diluter-demand regulators. With the regulator supply switches in the ON, NORMAL OXYGEN, and EMERGENCY OFF positions, the regulator will supply a diluted amount of oxygen on demand up to 32,000 feet, at which point the regulator will be supplying 100 percent oxygen. With the switches in the ON, 100% OXYGEN, and EMERGENCY OFF positions, the regulator's demand feature is eliminated and 100 percent oxygen is supplied at all times. With the switches in the ON, 100% OXYGEN, and EMERGENCY ON positions, the regulator will deliver an oxygen flow at a positive pressure equal to 3.5 inches of water. The EMERGENCY ON position is used when added pressure of oxygen is required to flush the smoke mask after donning, when a positive pressure is necessary because of high cabin altitude, or to test the mask's oxygen flow.

Normal Oxygen Mask

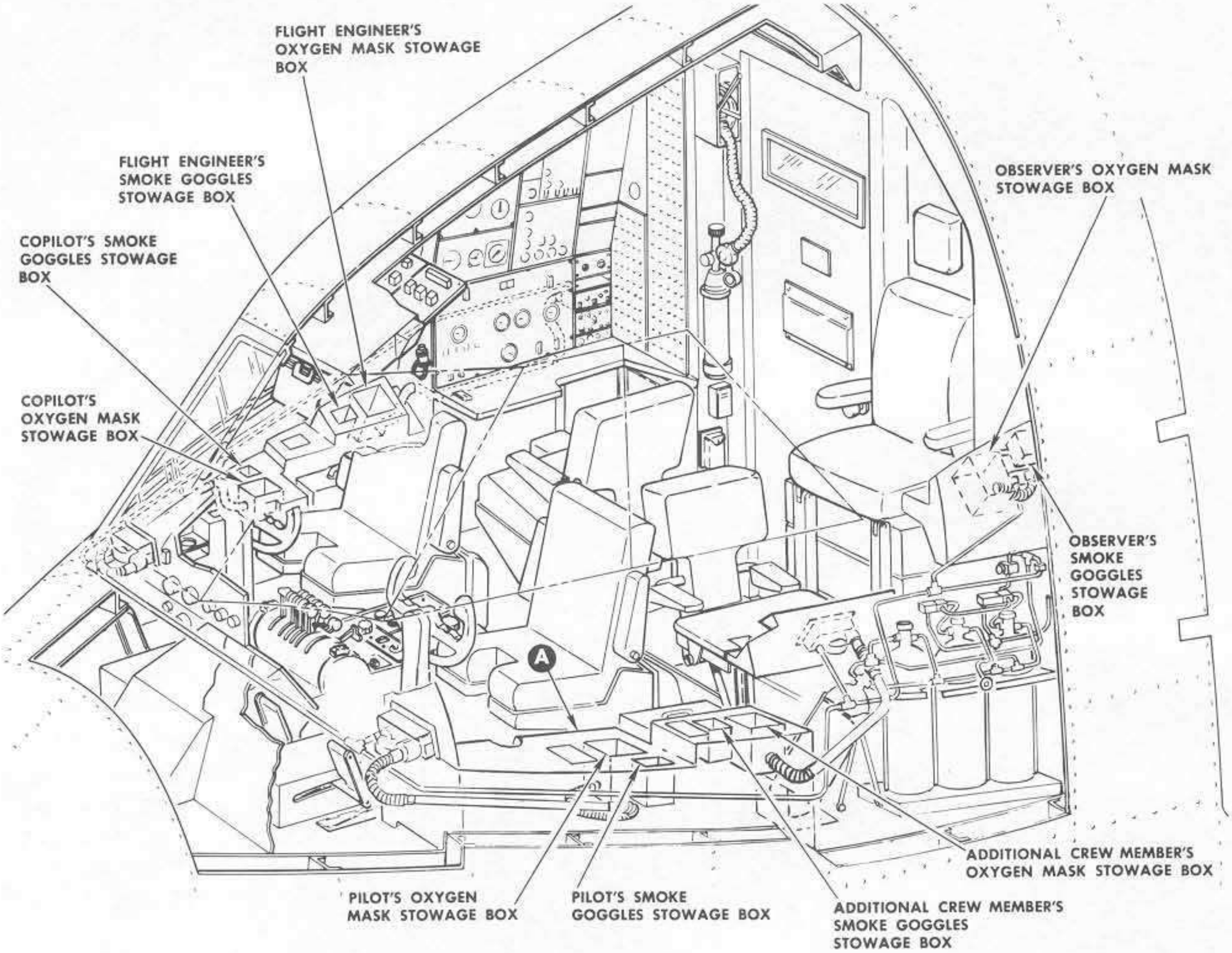
The normal oxygen mask is connected to the system at each crew member's mask stowage box. Oxygen is routed from the crew member's respective regulator to the mask stowage box and normal oxygen mask connector fittings. Smoke goggles are provided for use with each mask. The goggles are stowed adjacent to each mask stowage box. (see Figure 16-2).

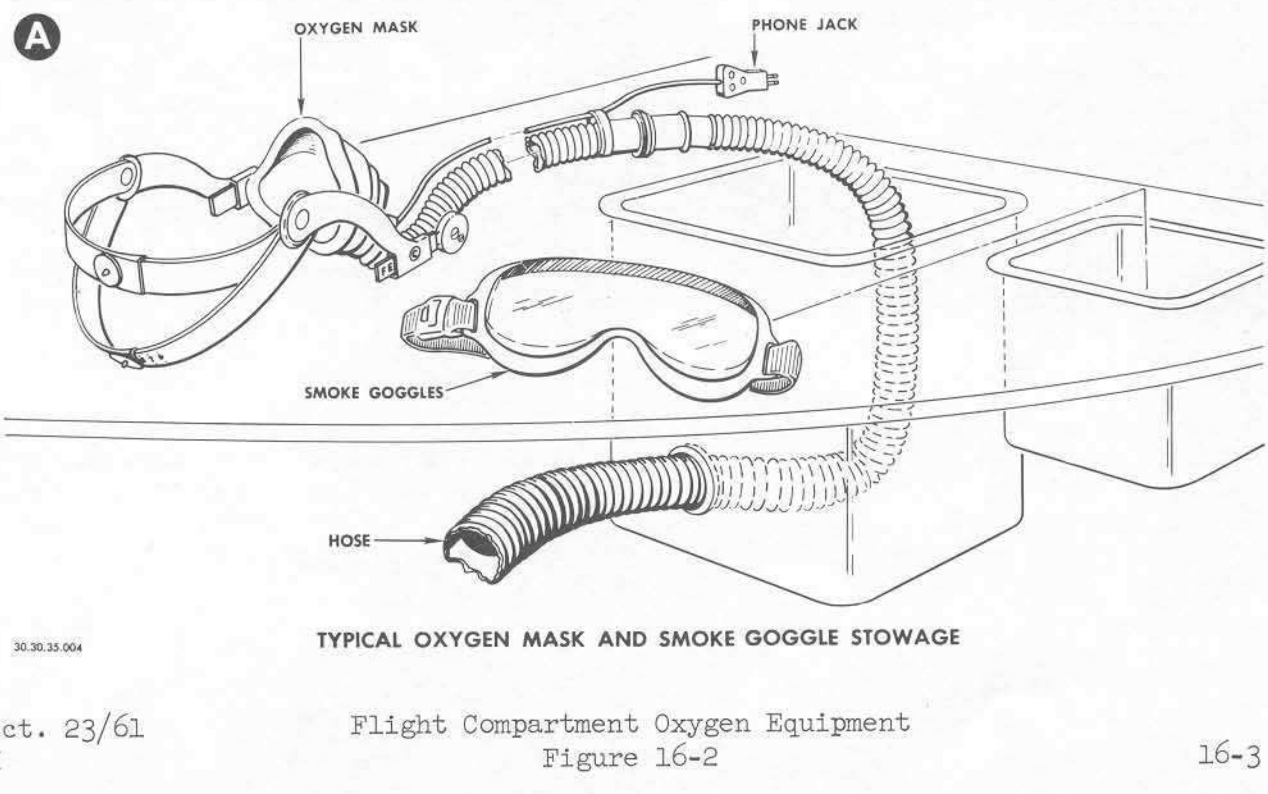
Oxygen Storage Bottles

The four oxygen storage bottles that furnish oxygen for the flight crew and passenger oxygen systems are mounted on the aft left side of the flight compartment. Each light-weight steel bottle has a capacity of 107.0 cubic feet of oxygen at



Typical Oxygen System Schematic
Figure 16-1





30.30.35.004

TYPICAL OXYGEN MASK AND SMOKE GOGGLE STOWAGE

Flight Compartment Oxygen Equipment
Figure 16-2

1800 psi. The contents of each bottle is indicated by a pressure gage mounted on the bottle. The gage is an integral part of the manually operated slow-opening valve that prevents pressure surges in the oxygen system.

Oxygen Storage Bottle Safety Disc

Each oxygen storage bottle is equipped with a built-in safety disc that will fracture at a bottle pressure of 2775 psig, or at a temperature of 160 degrees Fahrenheit. When a safety disc is blown out, the entire contents of the bottle will dump overboard and eject a green blow-out disc mounted in the fuselage skin. The green blow-out disc provides a visual indication from the exterior of the airplane as to the condition of the oxygen bottle safety disc. The blow-out disc is located in the skin directly outboard of the oxygen storage bottles.

High Pressure Line Shutoff Valves

The high pressure line shutoff valve is located in the interconnecting line between the two passenger oxygen storage bottles and the flight crew oxygen storage bottles. The purpose of this valve is to allow the flight crew to use the passenger oxygen supply when necessary.

PASSENGER OXYGEN SYSTEM

The passenger oxygen system is the continuous-flow type supplied from two oxygen storage bottles in the flight compartment. Oxygen from each storage bottle flows through an auto-opening continuous flow regulator valve. The flow from the two valves is combined into one single line into the passenger compartment area where the line tees into two lines connecting to the overhead convenience pods on each side of the airplane. At each pod, the lines connect to a rotary valve for each mask and to a pneumatic latch. Plastic tubes connect the rotary valves to the passenger face masks.

Automatic Actuating System

The two auto-opening continuous flow regulator valves are controlled by barometric pressure and are adjusted to open at a pre-determined altitude. The regulators reduce the 1800 psig supply oxygen to a pressure of 13-55 psig. When oxygen flow occurs, the pneumatic latches are released and the passenger oxygen masks are dropped. A manual override toggle switch at each valve allows oxygen to be released to the passenger masks manually. However, between cabin altitudes of 10,000 feet and 14,500 feet (±1,000 feet), the masks must be manually removed from their stowage positions.

Passenger Oxygen Masks

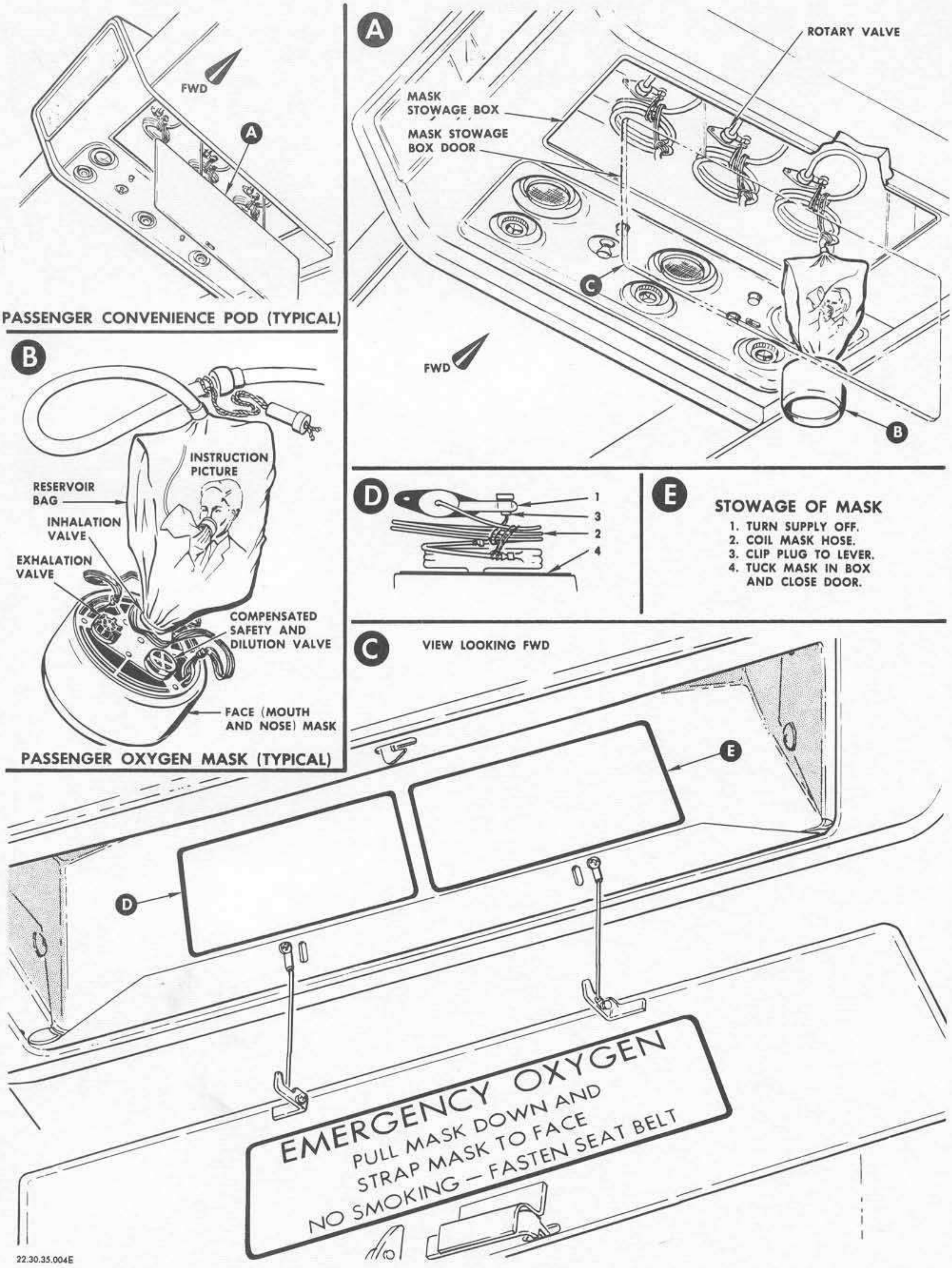
The number of masks in the passenger oxygen system, and their locations, varies with different interior configurations of the airplane. Each mask assembly consists of a face mask, various valves, a reservoir bag and connecting tubing. One additional mask for each seat row is provided to supply oxygen for children in arms. Each lavatory also has available two automatically released oxygen masks.

The passenger's and stewardess's oxygen masks are supplied with oxygen by manually pulling the masks downward after they have been released from the mask stowage panel. The downward pull actuates the rotary valves that permit oxygen to flow into the face masks (see Figure 16-3).

PORTABLE OXYGEN BOTTLES

One portable high pressure oxygen bottle, regulator and mask is located in the flight compartment.

Other portable high pressure oxygen bottles, varying from three to six in number depending on interior configuration, with demand regulators and continuous flow masks are located in the passenger compartment.



Passenger Oxygen Mask Stowage
Figure 16-3

17

FIRE PROTECTION
SYSTEM

Section 17

FIRE PROTECTION

TABLE OF CONTENTS

TITLE	PAGE
GENERAL FIRE PREVENTION	17-1
FIRE PROTECTION SYSTEM	17-1
FIRE DETECTION SYSTEM	17-1
Steady Illumination	17-5
Flashing Illumination	17-5
FIRE EXTINGUISHING SYSTEM	17-5
Fire Extinguishing Agent	17-5
Discharge Rate	17-5
FIRE EXTINGUISHING SYSTEM CONTROLS	17-6
Fire-Pull "T" Handles	17-6
Extinguishing Agent Switches	17-6
Extinguishing Agent Out Lights	17-6
Fire and Overheat Test Switch	17-6
Fire Bell Reset Switch	17-6
EXTINGUISHING AGENT STORAGE CONTAINER PRESSURE INDICATORS	17-7
GROUND FIRE FIGHTING PROVISIONS	17-7
HAND FIRE EXTINGUISHING EQUIPMENT WEIGHT AND CAPACITIES	17-7
EMERGENCY FIRE AXE	17-7
SMOKE GOGGLES	17-7
FIRE PROTECTION SYSTEM ELECTRICAL SOURCES	17-7

FIRE PROTECTION

GENERAL FIRE PREVENTION

In order to prevent any possible spread of a fire in the engine areas, the pod and pylon of each engine installation is divided into four separate compartments as follows:

1. The compressor and accessory section is divided into a forward compartment.
2. The combustion liner and turbine sections are divided into an aft compartment.
3. The pylon is isolated from the pod by a liquid and vapor tight titanium firewall capable of resisting a 2000 degree F flame for a period of 15 minutes.
4. The pylons are isolated from the under surface of the wing by means of a draft seal.

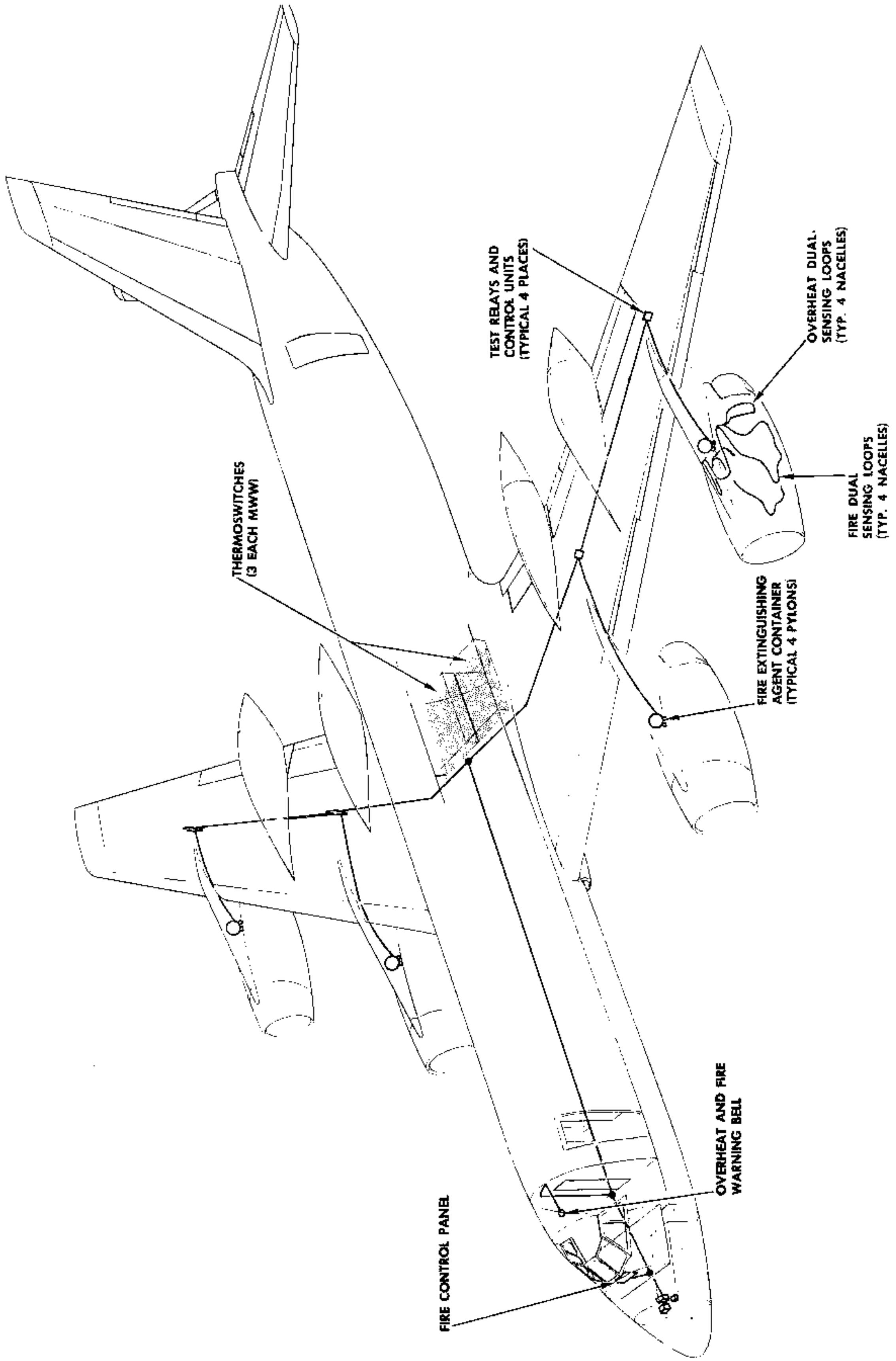
FIRE PROTECTION SYSTEM

The fire protection system consists of two subsystems: fire detection and fire extinguishing (see Figures 17-1 and 17-2). Continuous loops in the engine areas detect an overheat or fire condition, and transmit a signal through control units and power relays to a warning bell and warning lights in the flight compartment. A FIRE PULL "T" handle is then pulled to shut off the supply of fluids to the engine in which the fire exists. Pulling the fire-pull handle also exposes a previously inaccessible extinguishing agent release switch. When actuated, this switch releases the extinguishing agent to the affected area. If a fire has been extinguished and then re-occurs, a two-shot configuration makes it possible to direct the contents of another container to the area (see Figure 17-3).

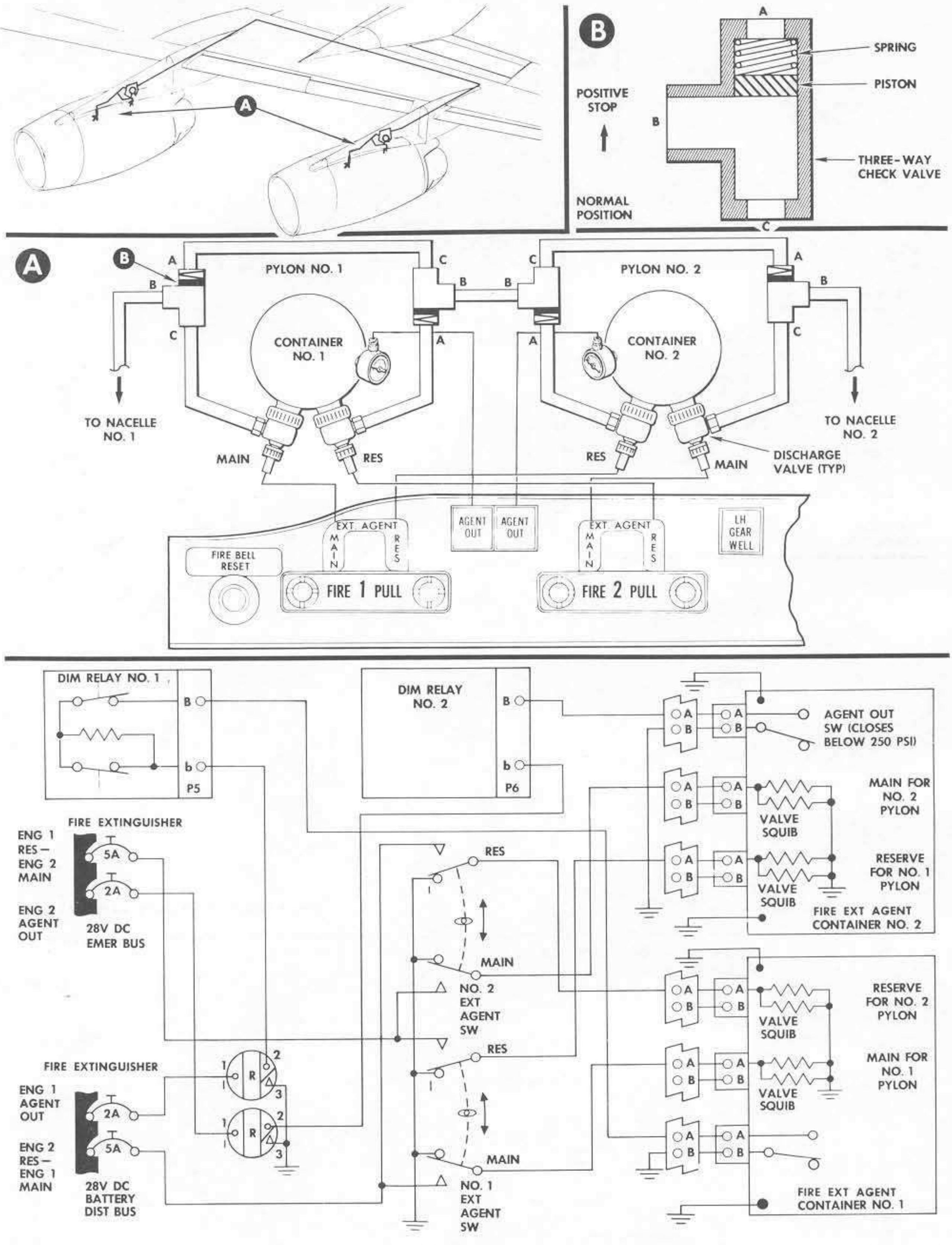
FIRE DETECTION SYSTEM

Each engine pod and pylon contains a fire detection circuit. The overheat and fire detection continuous loop is routed in a "maximum hazard" pattern around the burner and turbine portion of the engine in the aft pod compartment. A second fire detection loop is routed in the same manner around the compressor and accessory portions of the engine in the forward pod compartment and also extends into the pylon.

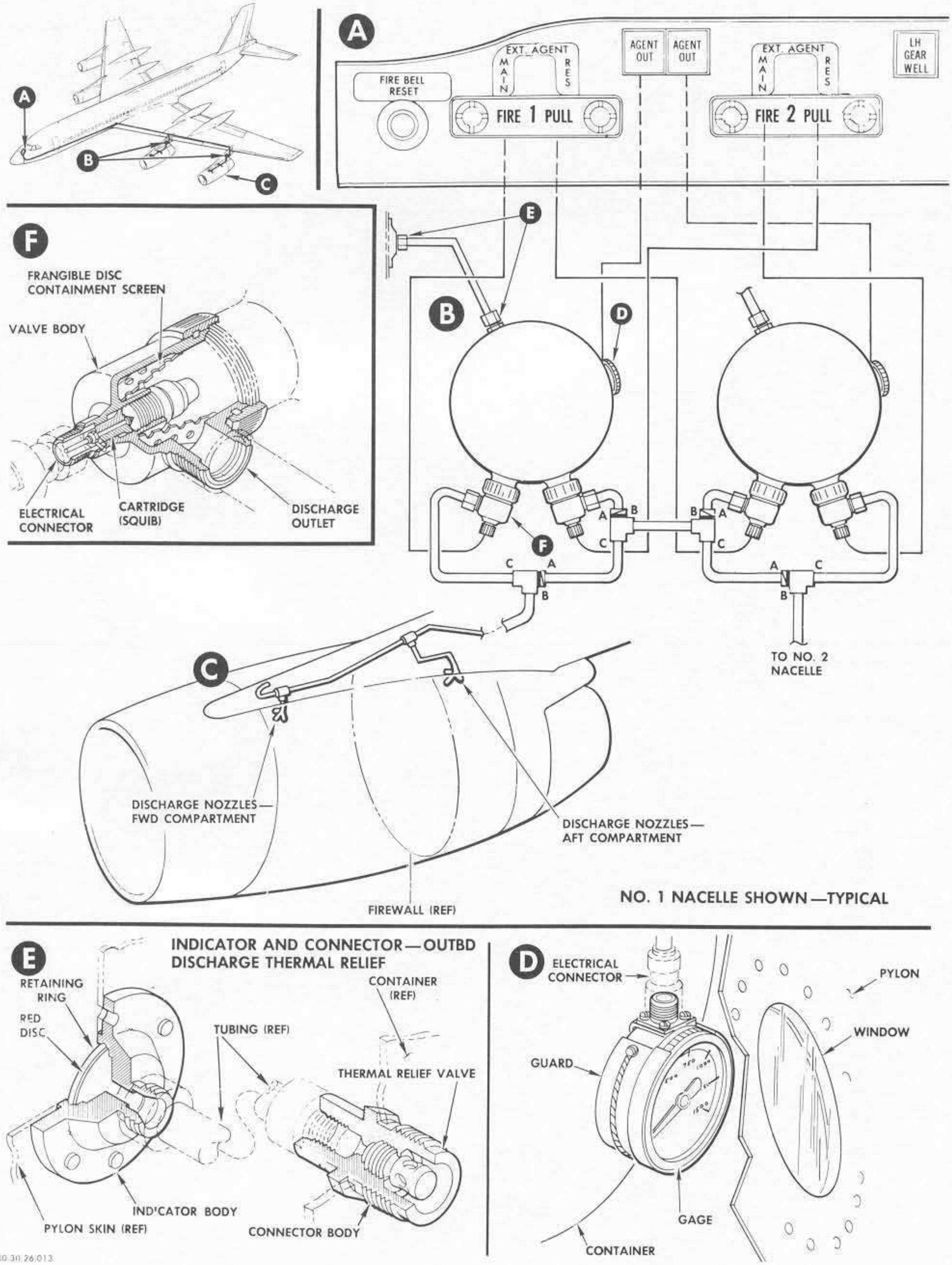
Whenever any part of the continuous loop circuit in the forward compartment is exposed to fire temperature, the detector warning lights, located in the fire-pull handle, illuminate and the fire warning bell will sound. When any part of the continuous loop circuit in the aft pod compartment is exposed to overheat or fire temperature, the detector warning lights, located in the fire-pull handle, flash intermittently and the warning bell sounds continuously.



Fire Protection and Detection Systems
Basic Components Locations
Figure 17-1



Fire Protection System Schematic
Figure 17-2



Fire Extinguishing System Components
Figure 17-3

In addition to the overheat and fire detection system for the nacelle and pylon areas, a "three-spot" thermoswitch system is installed in each main wheel well. If the thermoswitches detect excessive heat in the wheel well areas, a flasher circuit is excited and the appropriate LH GEAR WELL or RH GEAR WELL warning light flashes and the fire warning bell sounds continuously. The alarm temperature for the thermoswitches is 275 degrees F (135 degrees C). The two gear well warning lights are located on the fire control panel between the No. 2 and No. 3 engines fire pull handles.

Steady Illumination

Steady illumination of any of the four fire-pull "T" handle warning lights and steady sounding of the fire warning bell should be assumed by the flight crew to be an indication of engine fire and the fire fighting procedure should be performed immediately.

Flashing Illumination

Flashing illumination of any of the four fire-pull "T" handle warning lights and continuous sounding of the fire warning bell should first be assumed to be caused by an excessive aft compartment temperature. Retard the appropriate power control lever and observe if the warning signals cease. If not, assume an actual fire exists and follow the fire fighting procedure. Flashing of either gear well warning light should be assumed to be an overheat condition of the main landing gear brakes or tires. No distinction is made between overheat conditions and actual fire in the wheel well area, and no provisions are made for fire extinguishing. Therefore, in the event the gear well warning light flashes and the fire warning bell sounds, the landing gear should be extended for cooling.

FIRE EXTINGUISHING SYSTEM

The fire extinguishing portion of the fire protection system includes four spherical containers for storage of the extinguishing agent. One container is located in each pylon, just forward of the fuel box beam area in each pylon. Two discharge valves operated by electrically discharged cartridges, a main and reserve control and corresponding agent outlet tubing, attached to each container, release and route the extinguishing agent either to the corresponding pod and pylon on the same wing. This two-shot, crossfeed configuration makes it possible to release a second charge of fire extinguishing agent to the same pod and pylon if another fire breaks out, without having two containers for each engine.

Fire Extinguishing Agent

The fire extinguishing agent utilized in Bromotrifluoromethane (CF_3Br) and each container is charged with 5.5 lbs of the agent.

Discharge Rate

Each container will rapidly discharge so as to provide a concentration of 15 percent by volume throughout the compartments for a duration of 1/2 second. This concentration occurs within 3 to 4 seconds.

FIRE EXTINGUISHING SYSTEM CONTROLS

The fire extinguishing system controls consist of four fire-pull "T" handles, four agent-out indicator lights, and four switches to select main or reserve release of the fire extinguishing agent. A test switch is provided for test of the fire and overheat system. A fire warning reset button is also provided. All controls are on the fire control panel located in the center of the pilots' instrument panel anti-glare shield.

Fire-Pull "T" Handles

Four FIRE-PULL "T" handles are provided, one for each engine area. Each handle contains the fire detection system warning lights. Pulling the "T" handle exposes a previously inaccessible extinguishing agent switch and also actuates microswitches which route 28-volt dc power directly to the emergency fuel shutoff valve, engine bleed air shutoff valve, and the hydraulic suction shutoff valve. Also, hydraulic low pressure warning lights are disarmed.

Extinguishing Agent Switches

Pulling the "T" handle exposes the EXTINGUISHING AGENT three position spring-loaded switches. Actuating the switch momentarily to MAIN releases the extinguishing agent for the pod and pylon involved. Actuating the switch momentarily to RESERVE releases the extinguishing agent from the adjacent "same wing" engine fire protection system to the pod and pylon involved.

Extinguishing Agent Out Lights

A pressure sensitive switch in each container illuminates the AGENT OUT light when the contents of the container have been released or pressure should leak below 250 psi. Movement of the extinguishing agent switch to the reserve position illuminates the AGENT OUT light of the adjacent engine fire extinguishing agent container when its contents have been released.

Fire and Overheat Test Switch

A three position TEST switch is located on the right side of the fire control panel for pre-flight test of the fire detection system. Moving the switch to FIRE position illuminates the steady burning fire warning lights, and sounds the fire warning bell, if the circuit is operating properly. The main gear well warning lights will only illuminate in the FIRE position. The lights will flash indicating continuity of the circuit. Moving the switch to O'HEAT position illuminates the flashing overheat warning lights, and activates the fire warning bell, if the circuit is operating properly.

Fire Bell Reset Switch

A momentary push-button switch is provided on the left side of the fire control panel to reset the fire warning bell after it has been activated.

EXTINGUISHING AGENT STORAGE CONTAINER PRESSURE INDICATORS

Each extinguishing agent storage container is provided with a pressure gage marked in 50 psi increments from 0 to 1500 psi. Storage containers are normally pressurized to 600 lbs at 21 degrees C (70 degrees F) with nitrogen used as the pressurizing media.

GROUND FIRE FIGHTING PROVISIONS

Two fire access doors are installed in the left side of each engine nacelle. These doors are clearly marked, pop-in types, and are faired to the skin of the nacelles when not in use. Portable fire extinguishers, usually the cone type, can be thrust against one or both of the doors for ground fire extinguishing purposes (see Figure 17-4).

WARNING: CONSIDERABLE CARE SHOULD BE TAKEN BY GROUND PERSONNEL IN GROUND FIRE EXTINGUISHING PROCEDURES TO AVOID DEATH OR INJURY FROM OPERATING JET ENGINE INTAKES OR EXHAUSTS.

HAND FIRE EXTINGUISHING EQUIPMENT WEIGHT AND CAPACITIES

Three winterized hand water fire extinguishers are provided: one at the forward main entrance area and two on the forward side of the partition immediately forward of the aft entrance. A hand type CO_2 extinguisher is located in the flight compartment near the circuit breaker panel.

EMERGENCY FIRE AXE

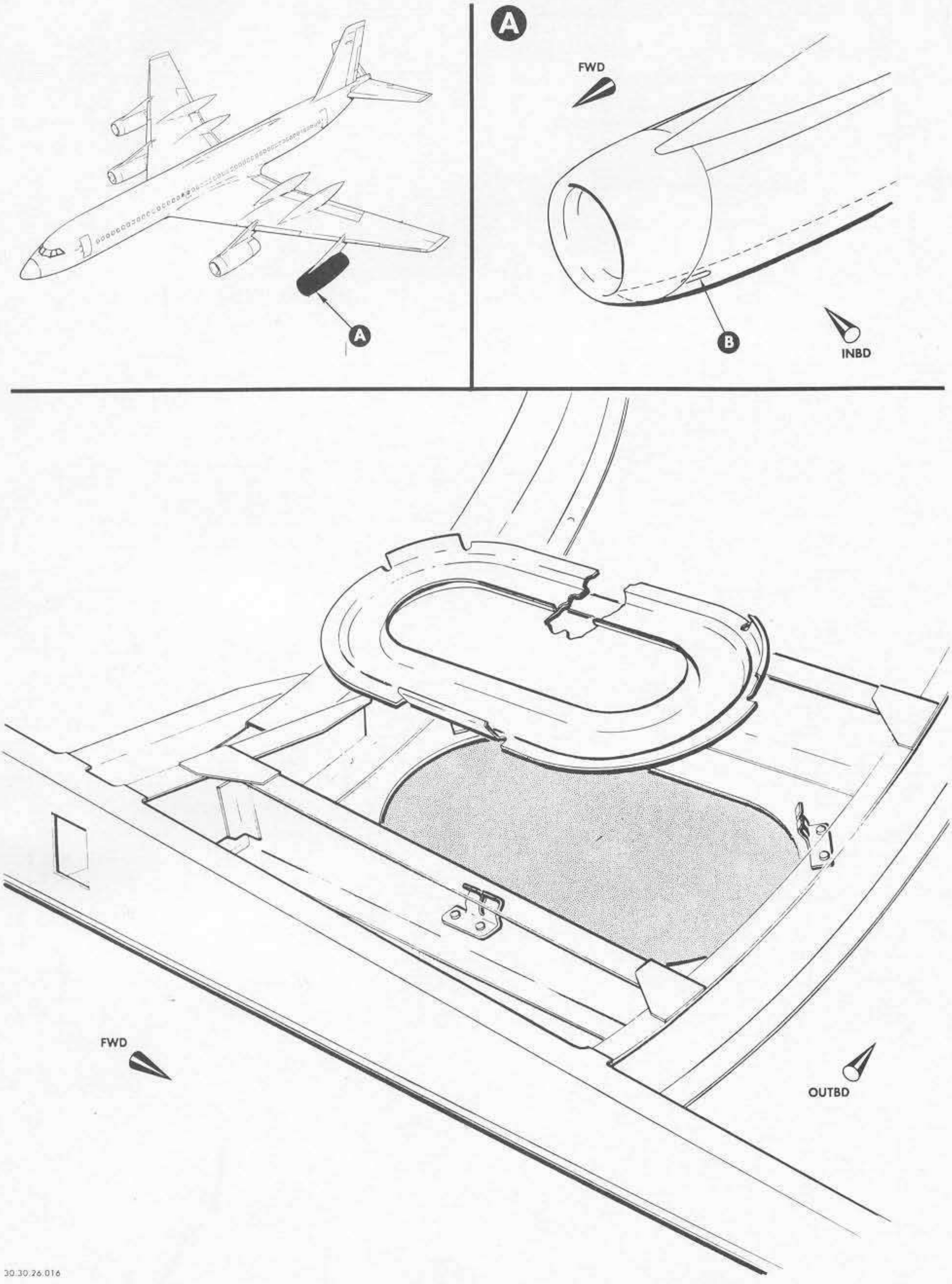
An emergency fire axe is provided in the flight compartment, mounted in a holder on the flight compartment door.

SMOKE GOGGLES

Smoke goggles are provided for use by all flight crew members and the observer. The goggles are stowed adjacent to each crew member's oxygen mask stowage box.

FIRE PROTECTION SYSTEM ELECTRICAL SOURCES

Fire detection system pilot's essential ac bus and the 28-volt dc emergency bus. Fire extinguishing system (including all shutoff valve electrical power, fire and overheat warning lights, and bell), 28-volt dc emergency bus or 28-volt dc battery bus.



Ground Fire Fighting Equipment Access Doors
Figure 17-4

18 FLIGHT AND NAVIGATION SYSTEMS

Section 18

FLIGHT AND NAVIGATION INSTRUMENTS

TABLE OF CONTENTS

TITLE	PAGE
GENERAL DESCRIPTION - FLIGHT AND NAVIGATION SYSTEMS	18-1
PITOT-STATIC SYSTEM	18-1
Pilot's Pitot System	18-1
Copilot's Pitot System	18-3
Pilot's Static System	18-3
Copilot's Static System	18-3
Auxiliary Equipment Static System	18-3
Autopilot Static System	18-3
Static Pressure Selector Switches	18-3
Pitot-Pressure Shutoff Valve	18-4
KOLLSMAN INTEGRATED FLIGHT INSTRUMENT SYSTEMS (KIFIS)	18-4
Airspeed Indicator	18-4
Angle of Attack Sensor	18-4
Altimeters	18-6
Altimeter Scale Error Correctors	18-6
Machmeter	18-7
True Airspeed Indicator	18-7
Static Air Temperature Indicator	18-7
KIFIS Test Switches	18-7
VHF NAVIGATION SYSTEMS	18-7
FLIGHT DIRECTOR SYSTEM (SPERRY)	18-8
Course Deviation Indicator (CDI)	18-10
Compass Card	18-10
Heading Knob	18-10
Course Knob	18-10
Course Deviation Bar	18-12
Ambiguity Flags	18-12
Fixed Airplane Reference	18-12
Horizon Director Indicators (HDI)	18-12
Attitude Sphere and Fixed Airplane Reference	18-13
Roll Angle Presentation	18-13
Pitch Trim Knob	18-13
Steering Needle	18-13
Warning Flags	18-13
Vertical Gyros	18-13
Flight Director Computer	18-14
Flight Director System Controls	18-14
Horizon Director Switch	18-14
Normal-Final Approach Switch	18-14
Magnetic Heading VOR/ILS Switch	18-14
Back and Front Switch	18-14
Compass Normal Alternate Switch	18-15

TABLE OF CONTENTS (CONT)

TITLE	PAGE
FLIGHT DIRECTOR SYSTEM (HORIZON DIRECTOR INDICATORS)	18-15
Horizon Director Indicator	18-15
Flight Director Computer	18-18
Mode Selector Switch	18-18
Vertical Gyro	18-19
SEARCH-WEATHER RADAR SYSTEM (RCA)	18-19
Receiver-Transmitter	18-21
Accessory Unit	18-21
Indicator Unit	18-21
Antenna Reflector Assembly	18-21
TERRAIN WARNING SYSTEM	18-21
Transmitter - Receiver Unit	18-21
Antenna	18-22
Distance Selector Switch and Indicator Lights	18-22
Test Switch	18-22
AUTOMATIC DIRECTION FINDING (ADF) SYSTEMS (COLLINS)	18-22
ADF Control Panels	18-23
Radio Magnetic Direction Indicator	18-26
ADF Receivers	18-26
ADF Loop Antennas	18-26
ADF Sense Antennas	18-26
MARKER BEACON SYSTEM	18-26
TRANSPONDER BEACON SYSTEM	18-28
LORAN NAVIGATION SYSTEM (Edo Model 345)	18-28
DISTANCE MEASURING EQUIPMENT-TACAN (ARINC)	18-28
Power Requirements	18-31
DOPPLER RADAR (PROVISIONS ONLY)	18-31
REMOTE COMPASS SYSTEMS	18-33
Operation (Slaved Gyro Operation)	18-33
Free Gyro Operation	18-35
Flux Valve	18-35
Directional Gyro	18-36
Remote Compass Mode Selector Switch	18-36
Radio Magnetic Direction Indicator	18-36
Course Deviation Indicator	18-37
Integrated Flight and Navigation Instrument Amplifier Racks	18-37
GLIDE SLOPE SYSTEMS	18-37
Control Panels	18-38
Antenna	18-38
MAGNETIC COMPASS	18-38
TURN AND SLIP INDICATOR	18-40
RATE OF CLIMB INDICATORS	18-40
ELECTRICAL POWER SOURCES	18-40

FLIGHT AND NAVIGATION INSTRUMENTS

GENERAL DESCRIPTION - FLIGHT AND NAVIGATION SYSTEMS

The pilot's and copilot's instrument panels each displays a full set of flight and navigation instruments. The flight instruments present altitude, attitude, and airspeed information. The navigation system includes magnetic and radio equipment for indicating heading, bearing, and position information. The flight crew is equipped to take advantage of the aids to air navigation provided by the federal airways system. All such equipment that is used on various versions of the Convair 990 are described in this section.

Some of the flight instruments are refinements of standard types that have been in common use for many years while others represent new concepts or combinations of two or more types of information in a single instrument.

Those flight instruments which indicate airspeed and altitude are combined in a Kollsman Integrated Flight Instrument System (KIFIS) which automatically compensates for the errors inherent in instruments that depend on air pressure for their operation. Stable directional information is provided by a compass system which combines gyro stability with the directional properties of the magnetic compass. VOR/ILS, heading, and attitude information and computed attitude commands are displayed in two instruments in the Flight Director System. Radio bearing and geographic position information is provided by the Automatic Direction Finding and the Marker Beacon Systems. DME-TACAN Systems and an EDO-Loran System are provided.

Search-weather radar furnishes information for avoiding areas of heavy precipitation which is associated with severe turbulance.

PITOT-STATIC SYSTEM

Pitot and static pressures are used in most of the flight instrument indicators, the autopilot system, the cabin differential pressure indicating system, the air conditioning system, the flight recorders, the airspeed warning system, the anti-shock body (ASB) fuel level and speed warning system, and the mach trim system.

Pilot's and copilot's pitot systems are completely separate. Pilot's and copilot's static systems are also separate for normal operation. However, static pressure selector switches in the systems permit selection of the alternate static pressure source for the pilot's or copilot's static system or for both systems at the same time. The block diagram of Figure 18-1 shows the arrangement of the pitot and static systems.

A pitot shutoff valve enables the copilot to cutoff pitot pressure to the autopilot, the flight recorders, the mach trim sensor, the airspeed warning control switch, the ASB fuel warning system.

Pilot's Pitot System

The left pitot tube supplies pitot pressure to the airspeed indicator and the Mach number indicator on the pilot's flight instrument panel.

To Be Furnished

Pitot-Static System Block Diagram
Figure 18-1

Copilot's Pitot System

The right pitot tube supplies pitot pressure to the airspeed indicator and the Mach number indicator on the copilot's flight instrument panel, and to the autopilot systems, the flight recorders, the airspeed warning control switch, the ASB fuel warning and mach trim sensors, if the pitot shutoff valve is not in the shutoff position.

Pilot's Static System

Two side static ports supply static pressure to the rate of climb indicator, the altimeter, the airspeed indicator, and the Mach number indicator on the pilot's flight instrument panel. If the pilot's static pressure selector is in the ALT position, these instruments receive static pressure from the alternate static source in the vertical stabilizer.

Copilot's Static System

Two side static ports supply static pressure to the following if the copilot's static pressure selector is in NORMAL position: copilot's rate of climb indicator, altimeter, airspeed indicator, and Mach number indicator. If the copilot's static pressure selector is in the ALT position, the copilot's static pressure requirements are transferred to the alternate static pressure source.

Auxiliary Equipment Static System

Two side static ports supply static pressure to the following: the cabin differential pressure indicator, the flight recorders, the mach trim sensor, the airspeed warning control switch, the ASB fuel warning, and the navigator/radio operator's altimeter.

Autopilot Static System

Two side static pressure ports supply static pressure to the air data sensor of the autopilot system.

Static Pressure Selector Switches

A static pressure selector switch enables either pilot to select the alternate static pressure source for indicators and equipment. A switch is located on each pilot's auxiliary instrument panel. When the ALT STATIC SELECT switch is actuated to ALT position, alternate static system pressure is in use. This selection can be made by either pilot, or by both pilots at the same time.

An electrical switch in the static pressure selector, which is mechanically linked to the ALT and NORMAL positions of the auxiliary instrument panel control, introduces a correction signal into the computer of the integrated flight instrument system.

Pitot-Pressure Shutoff Valve

Pitot pressure to the autopilot, the flight recorders, the airspeed warning control and fuel warning sensors is cut off by turning the AUX EQUIP PITOT SHUTOFF control on the copilot's auxiliary instrument panel to the OFF position. The instrument panel control actuates the pitot shutoff valve on the forward side of the panel.

KOLLSMAN INTEGRATED FLIGHT INSTRUMENT SYSTEMS (KIFIS)

The Kollsman Integrated Flight Instrument System, also known as the Air Data System, provides continuous indications of indicated and true airspeed, Mach number, altitude, and static air temperature. The system includes computer units which correct for variations in angle of attack, altitude, airspeed, and temperature. The computer units are located in the control chassis assembly in the electronics compartment. The system has an operating range from sea level to 50,000 feet with temperature ranging from -30 degrees C to +50 degrees C (see Figure 18-2).

The instruments included in the KIFIS are: airspeed indicators, machmeters, altimeters, a true airspeed indicator, and an outside air temperature indicator. Switches to test the system are also included.

WARNING: THE AIR DATA SYSTEM IS POWERED BY 115-VOLT, 400 CPS, AC CURRENT. WITH LOSS OF ELECTRIC POWER, THE ALTIMETERS, MACH INDICATORS AND AIRSPEED INDICATORS WILL OPERATE CONVENTIONALLY BUT THE RESULTANT INDICATIONS WILL BE UNCORRECTED FOR OPERATING VARIABLES. THE STATIC AIR TEMPERATURE AND TRUE AIRSPEED INDICATORS WILL BE INOPERATIVE.

Airspeed Indicator

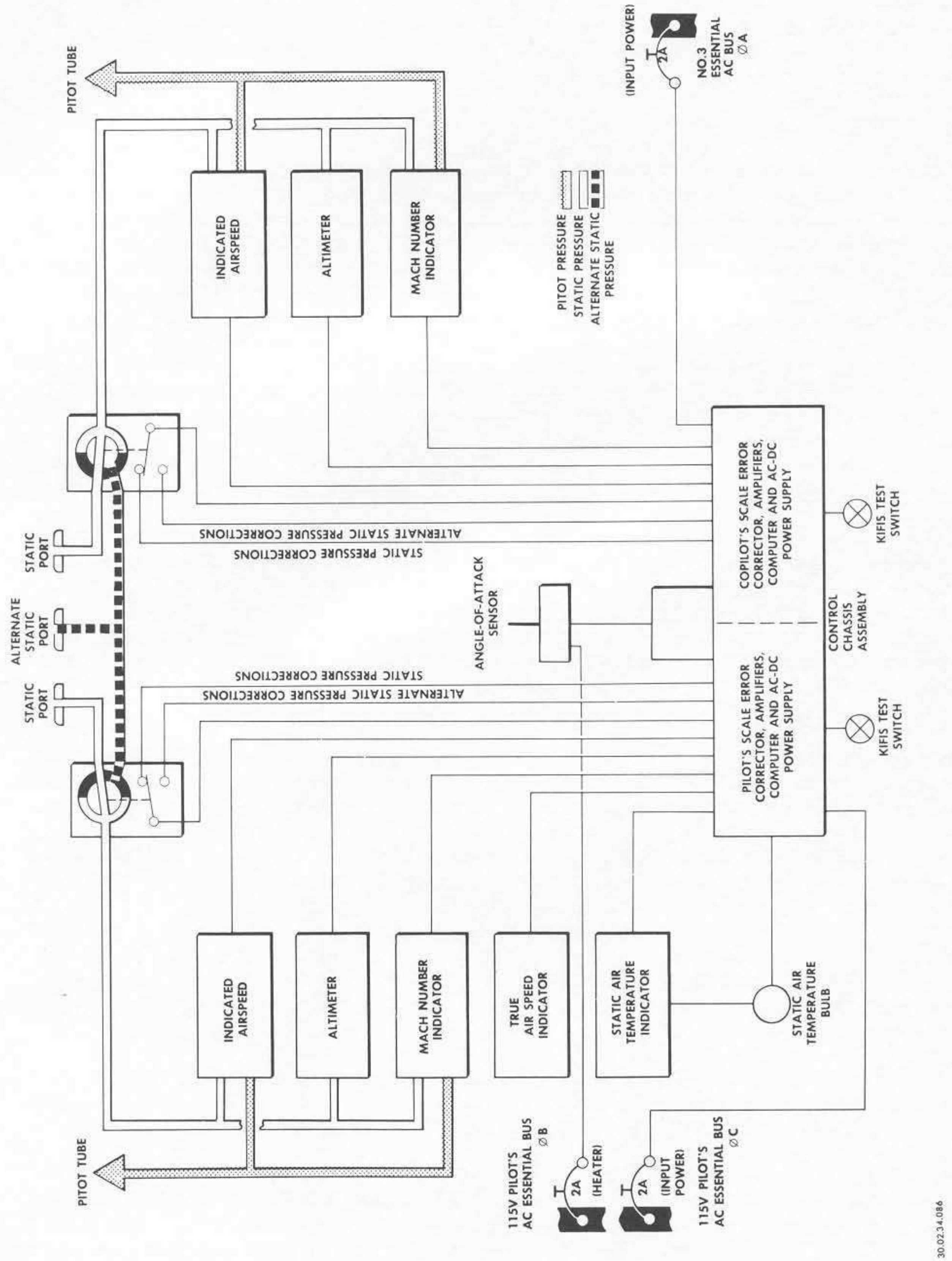
The pilot's and copilot's airspeed indicators are two-in-one instruments providing a continuous indication of indicated airspeed and maximum allowable airspeed. Each instrument includes an airspeed dial calibrated in knots, a white pointer to indicate airspeed, and a black pointer with red stripes to indicate the maximum allowable airspeed under all operating altitude and temperature conditions.

The airspeed mechanism consists of a pressure differential diaphragm which compares the static and pitot pressures and positions the white pointer to read indicated airspeed against the calibrations on the airspeed dial. The red striped black pointer is positioned with respect to the same dial by a static pressure diaphragm and indicates the maximum allowable airspeed.

The range of the airspeed indicator is 450 knots. A warning bell sounds if the indicated airspeed exceeds the maximum allowable airspeed. The bell circuit can be tested by a test button located on the pilot's flight instrument panel.

Angle of Attack Sensor

The angle of attack sensor is mounted on the left side of the fuselage near Station 473. The unit consists of a probe with two horizontal slots exposed to the ram air, a pressure differential diaphragm, and dual transmitting synchros.



Kollsman Integrated Flight Instrument
System Diagram
Figure 18-2

The slots sense the direction of the relative wind in relation to the attitude of the airplane and the synchros provide an angle of attack signal to the computer units. An integral heating unit in the sensor automatically maintains an adequate temperature to prevent icing.

Altimeters

A separate altimeter is provided for the pilot and copilot. Each instrument is the drum-pointer type with a range of -1000 feet to +50,000 feet. A radial pointer indicates hundreds of feet. Thousands of feet are indicated by a rotating drum visible through an opening in the face of the instrument. Two cutouts in the face of the instrument expose additional drums indicating barometric pressure in inches of mercury and in millibars. The barometric drums are positioned by a knob at the lower left corner of the case.

The pressure sensing mechanism of the altimeter incorporates a self-balancing dual static pressure diaphragm system which provides a high-torque precision output with minimum friction. A servo mechanism compensates for scale and static pressure errors.

Altimeter Scale Error Correctors

A scale error corrector provides a correction for scale or calibration errors inherent in each individual altimeter. Each correction unit contains an adjustable cam that is rotated as a function of altitude. Scale error corrections generated by cam action are transmitted to a differential synchro which combines the scale correction signal with a correction signal from the static system error corrector. The combined signal is transmitted to the servo mechanism in the altimeter which applies the appropriate correction to the altitude indication.

An automatic spring return device within the altimeter removes all error corrections in case of a power failure and a chevron appears on the altimeter dial. The instrument continues to operate as a conventional altimeter.

With the altimeter functioning as an integral part of the KIFIS and receiving correction signals for scale and static system errors, the total altimeter errors are reduced to the following:

Altitude (Feet)	Total Error (± Feet)
0	25
10,000	40
20,000	50
30,000	70
40,000	70

NOTE: Each altimeter and its scale error corrector are calibrated as a set and have the same serial number. If either unit is removed from the airplane, both units must be replaced with a matched set.

Machmeter

Two Machmeters are provided, one for the pilot and one for the copilot. Each Machmeter gives a continuous indication of airspeed in terms of Mach number.

A differential pressure diaphragm and a static pressure diaphragm, interconnected by a mechanical linkage, sense the two pressures, solve the ratio between them mechanically and indicate the Mach number by a radial pointer moving over a dial calibrated in percentage of Mach number. A synchro transmitter mounted on the pointer shaft transmits a Mach signal to the static system error corrector, the true airspeed indicator, and the static air temperature indicator of the KIFIS. The mechanism is compensated for variations in temperature ranging from -55 degrees C to +70 degrees C (-67 degrees F to 158 degrees F).

True Airspeed Indicator

A true airspeed indicator is located on the pilot's instrument panel. The true airspeed is indicated by a digital display through a window in the face of the instrument.

True airspeed is a function of Mach number and static or true outside air temperature. An electromechanical computer in the control chassis assembly combines these functions to actuate the motor which operates the counters in the instrument.

WARNING: THE TRUE AIRSPEED INDICATOR IS NOT TO BE USED BELOW 200 KNOTS.

Static Air Temperature Indicator

A static air temperature indicator is located in the copilot's instrument panel. The true outside air temperature is indicated by a radial pointer moving over a scale with a range of -100 degrees C to +50 degrees C (-147 degrees F to 122 degrees F).

The outside air temperature is sensed by a temperature bulb located in the slipstream of the airplane. In flight, the temperature indication would be higher than the true static temperature due to the compression of the air around the bulb. An electromechanical computer in the instrument applies a Mach correction signal to the signal from the temperature bulb to provide an accurate indication of the true air temperature.

KIFIS Test Switches

Test switches, marked KIFIS, are located on both the pilot's and copilot's instrument panels. These switches have TEST and NORMAL positions. In the TEST position, certain preset values for altitude, true airspeed, and static air temperature will be presented on the respective instruments if they are operating properly.

VHF NAVIGATION SYSTEMS

Two VHF navigation systems provide reception on 280 channels of all the VHF navigational facilities in the frequency band from 108 to 135.9 mc. These facilities include localizer beam transmissions in the 108 to 111.9 mc band, omnirange transmissions in the 112.0 to 117.9 mc band, tower communications in the 118.0 to 121.9 mc band, and voice communications in the 122.0 to 135.9 mc bank.

The upper halves of the VHF control panels on the pedestal between the pilots control the frequency of the VHF navigation receivers. VHF navigational data is presented on the pilot's and copilot's distance radio magnetic indicators (DRMI) and course indicators. VHF navigation system No. 1 drives the single-barred pointers of the distance radio magnetic indicators; No. 2 system drives the double-barred pointers. The DRMI pointers show the magnetic bearing of radio stations from the airplane. The course indicators show the magnetic heading of the airplane and the relative position of a VOR radial or a localizer beam. VHF navigation system No. 1 deflects the course bar of the pilot's course indicator; No. 2 system deflects the course bar of the copilot's course indicator.

VHF navigation system receiver audio is selected by the 1-VOR-2 toggle switch on the audio selector panel. The receivers of the two systems use common antennas installed in two horizontal fairings on the upper part of the vertical stabilizer.

No. 1 system receiver, power relay, and instrument transformer are installed on shelf B in the electronics compartment. No. 2 system receiver, power relay, instrument transformer, and the common accessory unit are installed on shelf E. The indicators and the selector switches are located on the pilots' flight instrument panels. Audio selector panels are installed for the pilots and for the flight engineer, and an audio selector panel is installed in the electronics compartment for maintenance uses.

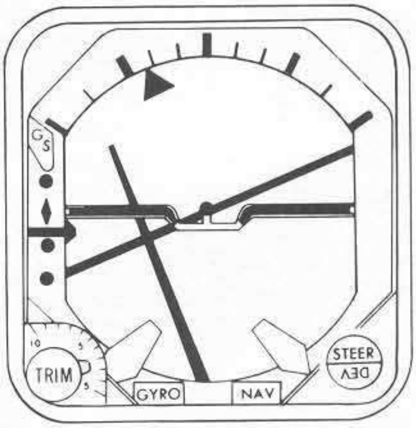
FLIGHT DIRECTOR SYSTEM (SPERRY)

The Sperry flight director system (FDS) coordinates compass heading, vertical gyro, pressure computer, and VHF navigational receiver signals and develops pitch and roll command signals. These command signals are presented on the horizon director indicators in the pilot's and copilot's instrument panels. FDS command signal displays on the indicators give pilots a picture of the best flight path as computed by the flight director system, and show the actual deviations of the airplane in attitude and/or direction from the computed path.

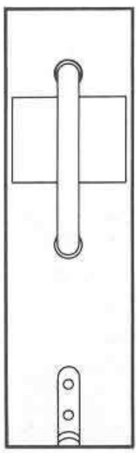
The flight director system (see Figure 18-3) comprises a flight director computer, two instrument amplifiers, two vertical gyros, two horizon director indicators, two course deviation indicators, and a flight director system control panel. Two radio magnetic direction indicators (RMDI's) not part of the flight director system, supplement FDS information to aid the pilots in maintaining a flight plan. The pressure computer is part of the SP-30 autopilot system.

The flight director computer, the instrument amplifiers, and the pressure computer are installed in the electrical equipment rack at the forward left side of the electrical compartment. The vertical gyros are located beneath the equipment rack. One horizon director indicator (HDI) and one course deviation indicator CDI are located in the top and center rows of instruments on each

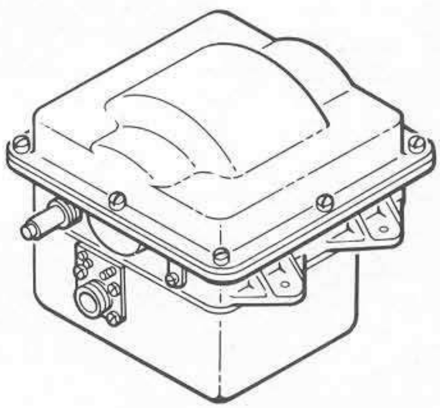
COURSE DEVIATION INDICATOR



HORIZON DIRECTOR INDICATOR



FLIGHT DIRECTOR COMPUTER



VERTICAL GYRO

30.02.34.101

Flight Director System Equipment
Figure 18-3

pilot's instrument panel. The RMDI's are located to the left of the CDI's in the center rows of instruments. The flight director system control panel is positioned in the lower right corner of the pilot's instrument panel.

The copilot's HDI and CDI displays are from the output of the No. 2 VHF and No. 2 compass systems at all times. See Figure 18-4 for a block diagram of the flight director system.

Course Deviation Indicator (CDI)

The course deviation indicator shows the position of the airplane in relation to a selected VOR or localizer radial. The indicator combines compass heading, present heading, preset course, VOR or localizer and glide slope bearings to or from indications, a NAV warning flag and a power OFF flag. The indicator has a preset course knob and a preset heading knob in the lower left and right corners. Four screws secure the indicator to the instrument panel. Electrical connections are made through receptacles on the rear of the case. The course deviation indicator consists of the following:

Compass Card

The servo-driven compass card rotates with changes in airplane heading to show magnetic heading. Synchros in the remote compass system transmit compass heading signals to the indicator and position the compass card. Since the cursors, the flags, and the course deviation bar rotate with the compass card, these displays are presented in positions relative to the compass card.

The compass card is graduated from zero to 360 degrees in 5-degree increments. Heading is read at a lubber line at the top of the indicator, and fixed markings are provided at 45 and 90 degrees.

Heading Knob

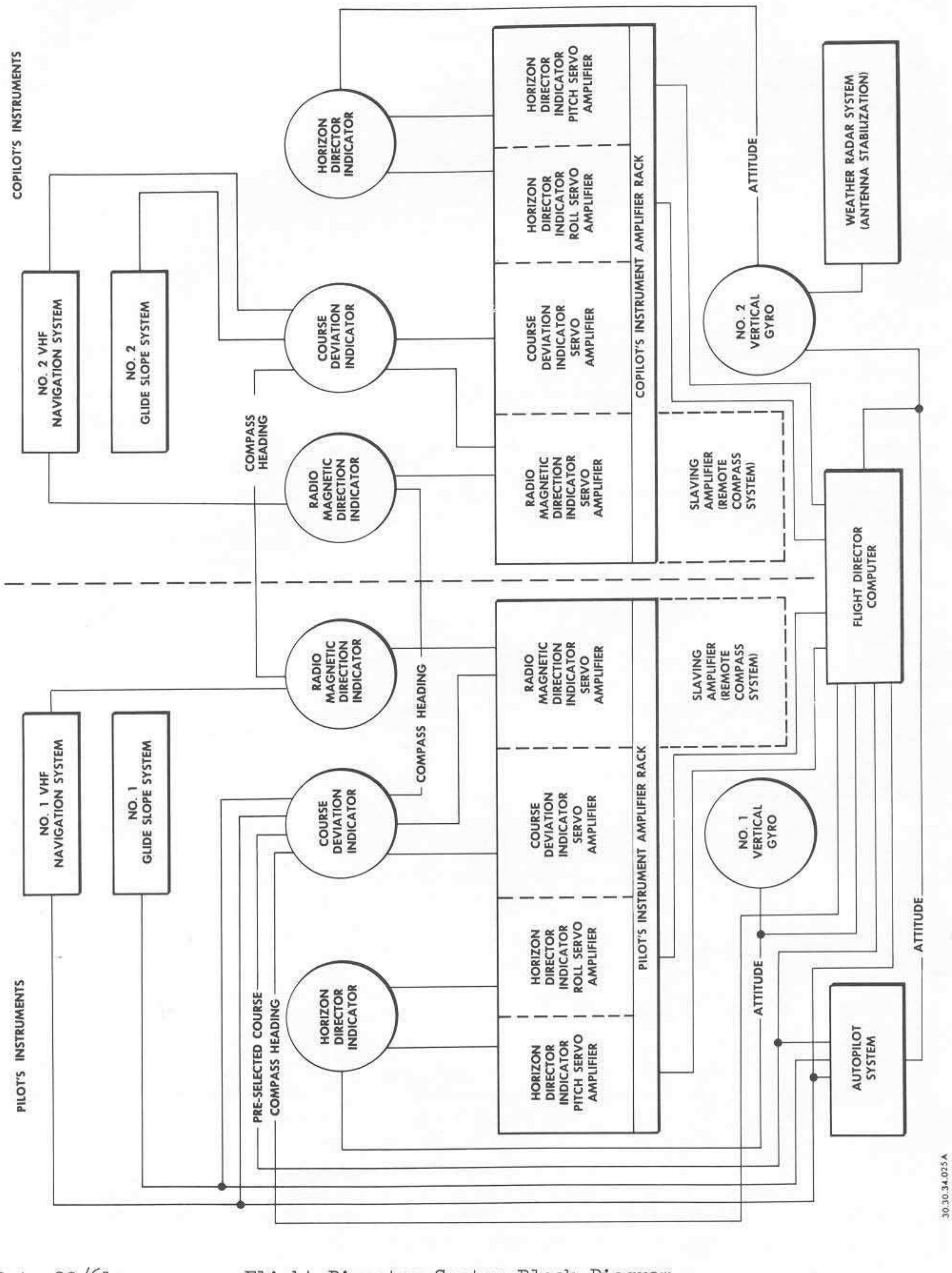
The heading knob at the lower right corner of the indicator selects the heading reference for the autopilot and for the command bar display in the horizon director indicator. The rectangular-shaped cursor on the compass dial is positioned by the HDG knob to indicate the preset heading selection.

Course Knob

The course knob enables the pilot to select the desired VOR radial or the runway heading for VOR or ILS operation. The course knob sets the course numbers into the window of the COURSE counter and positions the inverted "T" cursor to a corresponding position around the calibrated dial of the compass card. After being set, the course cursor rotates with the compass dial.

The information set into the system by the course knob is used by the flight computer and the autopilot system to turn the airplane smoothly into the VOR radial or into the localizer beam for an ILS approach.

The course knob has a normal and an emergency position. The knob when pushed in, is in normal position. For the emergency position the course knob is pulled



Flight Director System Block Diagram
Figure 18-4

out. If the compass system fails, the course knob is pulled out to its emergency position. The course deviation indicator display is released from its relationship to the compass dial, and the pilot can rotate the course deviation bar to a position parallel to the vertical axis of the indicator to continue using the signal from the VOR localizer receiver. Thus, if the remote compass system fails and the compass card of the course deviation indicator does not indicate magnetic heading of the aircraft, the pilot can orient the deviation bar to the line of flight. Deviations from course will then be relative to the line of flight, not relative to magnetic heading.

Course Deviation Bar

The course deviation bar is actuated by a dc meter movement. Radio signals from the VHF navigation receiver deflect the course deviation bar to show the azimuthal relationship of the airplane to the selected localizer heading or VOR radial. The course deviation bar also moves with the preset course cursor and assumes a position parallel to the course cursor and its reciprocal. Fixed course deviation dots on the indicator face indicate the amount of deviation from a selected course. The relative angle of the localizer beam or VOR radial is shown by the angular position of the course deviation bar with respect to the vertical centerline of the indicator. A NAV warning flag appears if the VHF navigation receiver becomes inoperative or if the VOR or localizer signal is absent or unreliable.

Ambiguity Flags

Two solid triangular ambiguity flags provide to and from indications to show the direction of the selected VOR station from the airplane. These flags are positioned by the VHF navigation receiver. Only one flag is visible at a time and it points in the direction of the station the receiver is tuned to. When the flight director system is used for ILS approach or is inoperative, both ambiguity flags are hidden.

Fixed Airplane Reference

The fixed airplane reference is a symbol located on the vertical centerline of the indicator face. This reference shows the flight path of the airplane relative to the selected VOR radial or localizer beam. The mode selector introduces signals that control the steering needle in the horizon director indicators. When the mode selector is in OFF position the steering needle remains out of sight.

Horizon Director Indicators (HDI)

The flight director system comprises two horizon director indicators, one for the pilot and one for the copilot. The horizon director indicator displays pitch and roll steering command signals from the flight steering computer; and shows airplane attitude by conventional gyro horizon indications. The horizon director indicator also displays a GYRO OFF warning flag and a NAV OFF warning flag. The horizon gyro indicators have a pitch TRIM knob in the lower left corner, and a STEER/DEV knob in the lower right corner. Four screws secure the indicator to the instrument panel. Electrical connections are made through a receptacle on the rear of the indicator case.

Attitude Sphere and Fixed Airplane Reference

The attitude sphere displays airplane pitch and roll indications against a fixed airplane reference on the indicator face. The attitude sphere, which is free to rotate 360 degrees, is divided by a horizon line. The upper half of the sphere is finished in light blue and the lower half is in black.

The attitude sphere is positioned in pitch and roll by servo mechanisms in the indicator, which are controlled through synchros in the vertical gyros, to show airplane attitude. The fixed airplane reference represents the airplane's attitude with reference to the horizon.

Roll Angle Presentation

A roll angle indicator at the top of the attitude sphere moves across reference markings on the face of the indicator to show the amount of airplane roll. The reference markings are 10-degree graduations ranging from zero to 30 degrees of roll with markings provided at 45 and 60 degrees of roll.

Pitch Trim Knob

The pitch trim knob in the lower left corner of the indicator trims the attitude sphere in the pitch axis to reposition the sphere horizon line to correspond to a sustained pitch attitude. The indicator is calibrated so the sphere can be trimmed to an attitude from 11 degrees airplane nose up to six degrees nose down.

Steering Needle

The steering needle on the face of the indicator displays flight steering computer "command" signals. The position of the steering needle shows the pilot at a glance how he must steer the airplane to maintain the flight patch calculated by the computer. The steering needle is out of view when the mode selector switch is in OFF.

Warning Flags

The horizon director indicator has a GYRO OFF warning flag and a NAV OFF warning flag. The gyro warning flag indicates gyro malfunction. The NAV warning flag indicates a flight steering computer malfunction.

Vertical Gyros

The vertical gyros supply roll and pitch attitude signals for the flight director system. The signals are transmitted by autosyn synchros in the gyros.

The No. 1 vertical gyro is mounted on the lower shelf left side, of the electrical compartment. The No. 2 vertical gyro is mounted on the lower shelf just aft of the No. 1 vertical gyro. The No. 1 vertical gyro supplies roll attitude signals to the pilots HDI's and flight director computer, and pitch attitude signals to the pilot's HDI only. The No. 2 vertical gyro supplies pitch attitude signals to the copilot's HDI, and provides information for the auto-pilot system. The pilot's HDI may be switched to the No. 2 vertical gyro, by the HORIZON DIRECTOR switch on the steering computer control panel.

Flight Director Computer

The flight steering computer combines and converts compass heading, vertical gyro, and radio signals to provide roll and pitch steering signals for the steering needle of the horizon director indicators. The computer receives information from No. 1 vertical gyro or No. 2 vertical gyro via the bootstrap autosyn in the pilot's horizon director indicator; VOR system No. 1, and selected course and heading information from the pilot's course deviation indicator. The computer contains an ac and dc power supply and plug-in modules for computing steering command signals. The 115-volt, ac No. 3 essential bus normally supplies electrical power to operate the computer. If the No. 3 bus fails, the load is transferred to the pilot's essential bus. Electrical connections are made through a receptacle mounted on the rear of the chassis.

Flight Director System Controls

A flight director system control panel is installed in the bottom right corner of the pilot's instrument panel. The following switches are located on the panel.

Horizon Director Switch

The HORIZON DIRECTOR switch is a two-position covered switch marked NORMAL and ALT. The NORMAL position provides attitude signals from No. 1 vertical gyro to operate the horizon display of the pilot's HDI. The ALT position connects the pilot's HDI signal source to the copilot's HDI, for use as an alternate signal source. The copilot's HDI RECEIVES PITCH ATTITUDE SIGNALS ONLY, FROM THE No. 2 vertical gyro.

Normal-Final Approach Switch

The NORMAL - FINAL APPROACH switch controls the sensitivity of the horizon director indicator steering needle.

Magnetic Heading VOR/ILS Switch

The MAG HDG VOR/ILS switch selects either the chosen magnetic heading, or VOR/ILS as an input to the computer. The frequency selector of the VOR receiver No. 1 serves to select either the VOR or the ILS localizer as an input to the computer.

Back and Front Switch

A covered BACK and FRONT switch reverses the polarity of the VOR deviation input signal to the computer and the pilots HDI's when selected to BACK, and is used in the VOR/ILS mode only. The auto-pilot is interlocked to prevent engagement of the VOR/ILS or GS modes, when the switch is in the BACK position. In addition, the BACK beam switch will reverse the sense of the course selection signal to the steering computer. An amber light mounted above the HDI on the copilot's instrument panel will illuminate when the "back beam" mode has been selected.

Compass Normal Alternate Switch

The COMPASS NORMAL or ALT switch may be used to select either No. 1 or No. 2 compass system.

FLIGHT DIRECTOR SYSTEM (HORIZON DIRECTOR INDICATORS)

The flight navigation instrument system integrates the functions of the VHF navigation sytem, the remote compass system, the glide slope system, and the vertical gyro. Depending upon the position of the mode selector switch, the information from these systems is displayed on the horizon director indicators (HDI). The HDI displays pitch and roll attitude of the airplane by a horizon sphere positioned with respect to the vertical gyro and read against a fixed reference on the face of the instrument. The HDI also displays steering commands by a vertical roll command bar and a horizontal pitch command bar. Depending upon the selection made by the pilot at the mode selector switch, the roll command bar is positioned with respect to outputs from the vertical gyro, the remote compass system, or the VOR-localizer VHF navigation system, and the pitch command bar is positioned with respect to the vertical gyro, the glide slope system or a fixed fly-up signal for go around operation. When the command bars are not in use, they are positioned out of view.

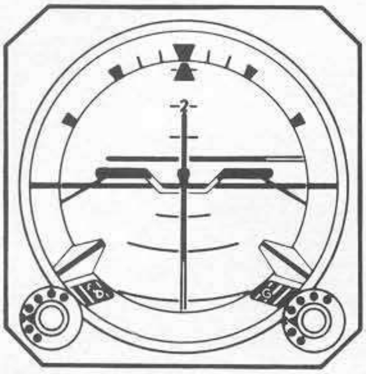
Included in the flight navigation instrument system are the mode selector switch, the flight director computer, the vertical gyro, the horizon director indicator, and the instrument amplifier. See Figure 18-5. Figure 18-6 is a block diagram of the system.

Horizon Director Indicator

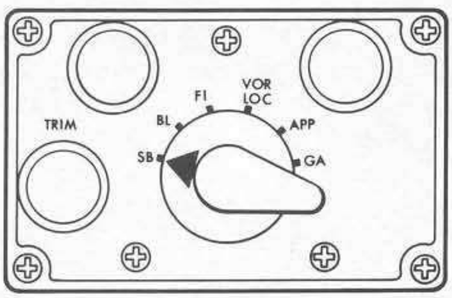
The horizon director indicator displays airplane attitude in pitch and roll by a gyro-controlled sphere. A line, which represents the earth's horizon, divides the light upper part of the sphere from the dark lower part, and the sphere is calibrated above and below the dividing line to indicate the airplane pitch attitude. These pitch calibrations are referenced to index lines at the right and left sides of the indicator center. The indicator face is calibrated at the top. A triangular index on the sphere indicates airplane bank angle by these calibrations.

Two lines across the indicator center, represent the airplane wings. These lines can be moved up and down by rotating the knob at the lower left side of the indicator. Positioning these lines helps the pilot to maintain a sustained pitch attitude. After the desired rate of climb or descent is established, the pilot can set the reference to the horizon line of the sphere; change from the desired climb or descent attitude will then show as deviation of the horizon line from the reference.

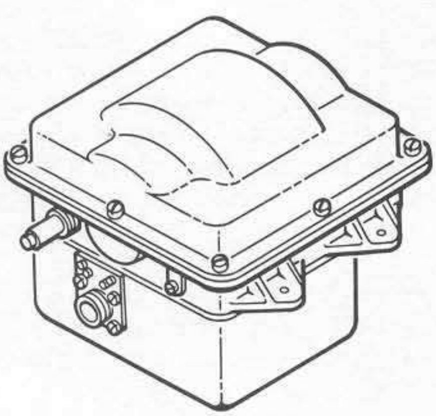
Two black flags cover the warning indications painted on the indicator mask when proper three-phase power is supplied to the vertical gyro, erection is complete, and the correct voltage is supplied to the servo amplifier and the flight director computer. Upon power failure to the flight director computer the FD painted



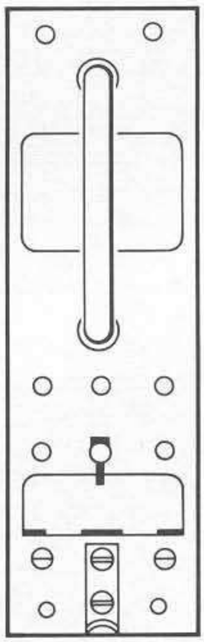
HORIZON DIRECTOR INDICATOR



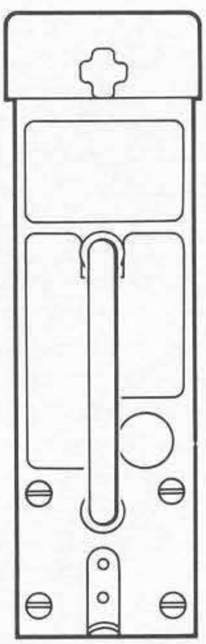
MODE SELECTOR SWITCH



VERTICAL GYRO



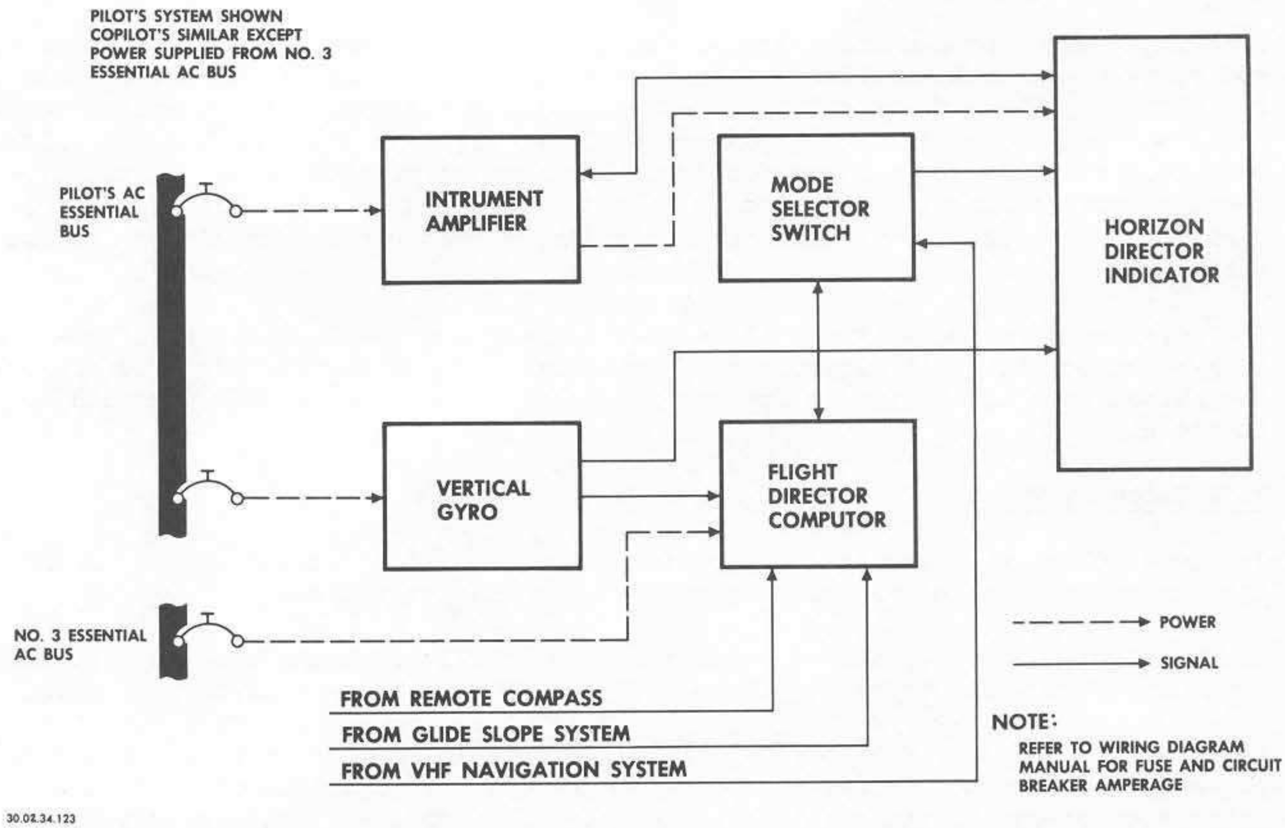
FLIGHT DIRECTOR COMPUTER



INSTRUMENT AMPLIFIER

30.02.34.124

Flight Navigation Instrument System Components
Figure 18-5



Flight Navigation Instrument
System - Block Diagram
Figure 18-6

on the lower left edge of the indicator mask becomes visible due to retraction of the black flag. Upon power failure to the servo amplifier the G painted on the lower right edge of the indicator mask becomes visible due to retraction of the black flag.

A horizontal and a vertical bar move across the face of the instrument to indicate steering commands. When both bars are centered on the fixed center reference, steering is correct. Movement of the vertical bar left or right from the center indicates a required roll to left or right. Movement of the horizontal bar up or down from the center reference indicates a required change in pitch for fly-up or fly-down. The command bars are positioned by gyro, radio, or magnetic compass references depending on the position of the mode selector switch.

Flight Director Computer

The flight director computer receives heading, attitude and radio information and computes correct steering commands to position the bars of the HDI so that the pilot may fly a desired flight path by zeroing the bars. The computer functions not only to determine displacement of the airplane from the desired flight path, but also rate of approach or departure from the flight path. If the airplane is approaching the flight path at the correct rate, the steering command bars will be centered even though the airplane is not yet on the correct flight path.

The reference for the flight path, either preselected heading, radio information, or gyro information, is determined by the mode selector switch. The mode selector switch selects the inputs for the flight director computer and the outputs to the HDI.

Mode Selector Switch

The mode selector switch is a six position rotary switch located on the engine instrument panel. The function of each position is as follows:

SB (standby) position places a voltage on the steering command bars to position them out of view and opens the circuit to the FD flag mask to expose the flag.

BL (blue left) position supplies glide slope information to the computer and connects the computer integrated pitch and glide slope command output to the pitch command bar of the HDI. This position also connects VOR-localizer signals to the computer but reverses normal polarity of the deviation signal. BL position is used when an approach is made to a runway from the opposite direction of the normal glide path approach which reverses polarity of deviation signals. Output to the roll command bar of the HDI in this position is integrated VOR-localizer information, heading, and roll rate command signals.

FI (flight instruments) position disconnects all radio inputs to the computer. Computer output to the HDI is preselected heading integrated with roll to the roll command bar, and pitch attitude to the pitch command bar. Pitch attitude indicated by the command bar is a more sensitive display than the normal pitch attitude displayed by the horizon sphere.

VOR-LOC (visual omnirange - localizer) position is identical to BL position except that roll deviation signals are connected in normal polarity for flying omnirange or localizer radials. In both BL and VOR-LOC operation, if the VHF navigation equipment is not tuned to a glide slope transmitter as in VOR operation, the glide slope bar responds to pitch information only to aid the pilot in maintaining level flight.

APP (approach) position connect VOR-localizer, and glide slope information to the computer. Unlike the VOR-LOC position, glide slope information is not integrated with pitch since during approach the normal attitude for the airplane will be nose down. However, a pitch rate signal is integrated with glide slope information so that the pitch command bar indicates correct rate of descent rather than simple airplane position relative to the glide path. This position also disconnects heading input to the computer so that the roll command bar responds to localizer-roll signals only.

GA (go around) position disconnects radio input to the computer, connects heading integrated with roll to the roll command bar of the HDI, and inject a fixed fly-up signal to displace the pitch command bar upward.

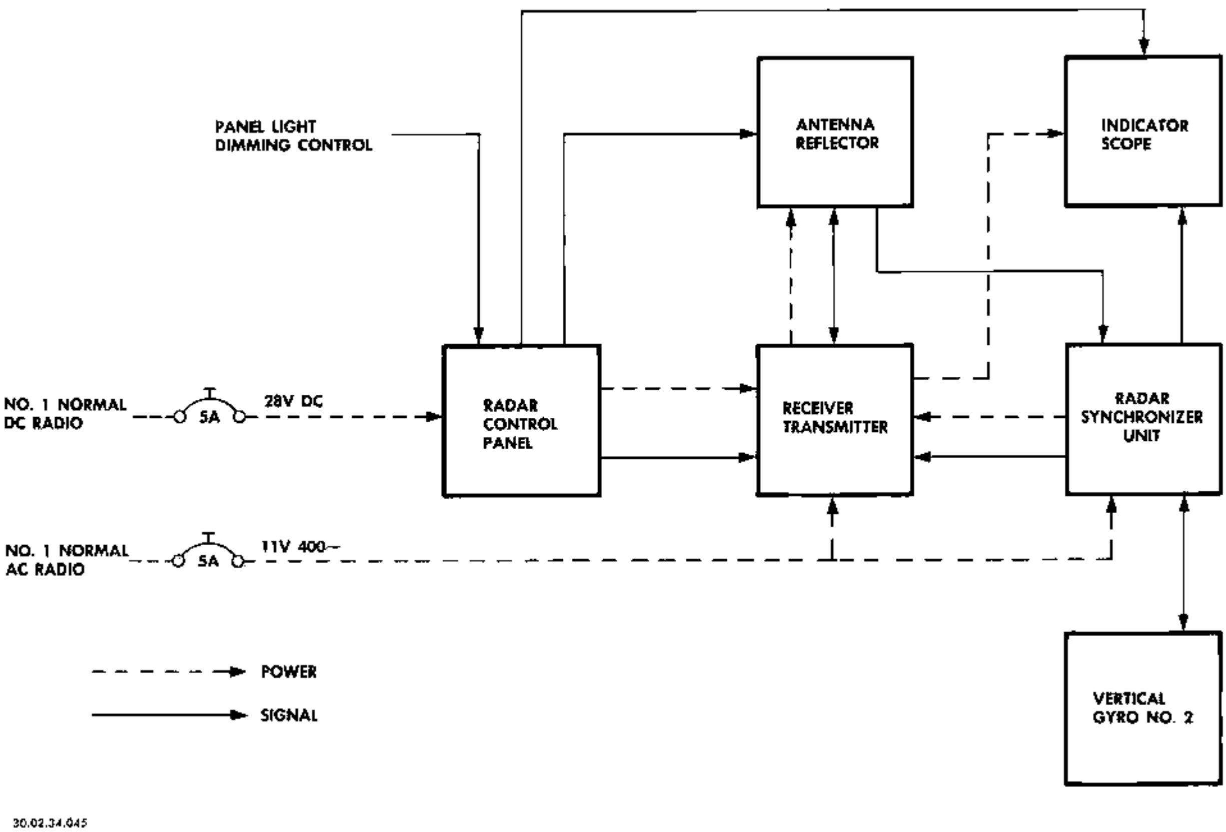
Vertical Gyro

The vertical gyro establishes roll and pitch references for the horizon indicator and for the flight director computer in the flight navigation instrument system, for the autopilot system and for the radar system. Synchros on the roll and pitch axes detect movement in these axes. The output signals from these synchros are routed to the horizon indicator roll and pitch synchros. These signals control the position of the horizon indicator sphere, maintaining the sphere in the position that corresponds to the vertical gyro position. The airplane therefore pitches and rolls about the horizon indicator sphere. The calibration on the sphere and on the fixed face of the indicator continuously display the airplane attitude. The vertical gyros operate from 115-volt, single-phase, ac power. No. 1 vertical drives the pilot's horizon indicator, and supplies vertical gyro information to the autopilot and weather radar systems.

SEARCH-WEATHER RADAR SYSTEM (RCA)

The search-weather radar system functions primarily to detect severe storm areas. It also serves as an anti-collision warning device through detection of objects in the flight path of the airplane. It may also serve as a navigational aid of providing terrain mapping. The search-weather radar system is comprised of an antenna-reflector, receiver-transmitter, accessory unit, control panel, and an indicator scope (see Figure 18-7).

Short high-power bursts of electrical energy are fed to the antenna from the transmitter and are propagated outward in straight lines with reference to the reflector position. When these energy impulses encounter an object of sufficient density, the pulses are reflected back to the antenna. These reflected pulses are amplified by the receiver and accessory units and are measured for display on the indicator-scope. The time interval between the transmitted and received pulses provides the basis for distance determination.



Search-Weather Radar Block Diagram
Figure 18-7

Simultaneously, the direction in which the reflector is pointed causes an azimuth indication to appear on the indicator scope. Since the pulses are transmitted in the horizontal plane as the indicator sweep is rotated about its origin, the distance of the target and its direction away from the airplane are displayed on the indicator scope.

Receiver-Transmitter

The transmitter supplies 75 KW minimum peak power to the antenna at a frequency of 5400 megaxycles. In addition, the transmitter supplies the accessory unit with the automatic frequency-controlled received signal information. A running time meter, a test meter, and a switch with eight metering positions are installed on the front panel. The range can be set by a RANGE switch on the control panel at 20, 50 or 150 miles.

Accessory Unit

The accessory unit contains the synchronizing, video, iso-contour, I-F, and stabilization circuitry for the radar system. In addition, the accessory unit provides filament and plate power for one or both of the indicators as well as the preamplifier and automatic frequency control unit in the receiver-transmitter. Two fuses, a test meter and an eight position meter selector switch are provided on the control panel.

Indicator Unit

The indicator unit houses the five-inch cathode-ray tube on which target information is displayed. The indicator also contains the final video amplifier, yoke-rotation mechanism, range marker lamps, and the high voltage power supply for the cathode-ray tube. The four adjustments on the indicator front assembly provide for RANGE MARKS, LIGHT, SWEEP, CURSOR, and INTENSITY GAIN.

Antenna-Reflector Assembly

The antenna-reflector is a lightweight transmit-receive unit that is line-of-sight stabilized against airplane pitch and roll. The antenna assembly consists of a base assembly housing, a synchro motor and transmission, a reflector frame, and a reflector feed dust cover on which the reflector assembly is mounted. The reflector assembly consists of the reflector, feed, matching plates, and mounting hardware.

TERRAIN WARNING SYSTEM

The terrain warning system utilizes a radar beam transmitter-receiver and an antenna combination to provide visual and aural warnings of terrain obstacles.

Transmitter-Receiver Unit

The transmitter-receiver consists of a transmitter whose RH out-put radiates from the antenna and a receiver which picks up reflections on the same antenna from objects on the ground. The receiver converts the reflected signals into a form suitable to operate the indicator lights or the aural warning device.

Antenna

The antenna is mounted on the bottom of the forward fuselage and serves as a receiving and transmitting unit. The body of the airplane serves as an effective shield against upward radiation from the antenna.

Distance Selector Switch and Indicator Lights

The distance selector switch and indicator lights are mounted at the left side of the pilot's flight instrument panel. The selector switch is a 3-position switch labeled 2000, 1000 and 500 feet. Three associated indicator lights are labeled 2000, 1000 and 500 feet. With the switch set at any of the three positions, the corresponding warning light will come on when the airplane comes within the selected distance from the ground. A warning bell will sound when one of the warning lights illuminates.

Test Switch

The warning lights will illuminate and the warning horn will sound when the test switch is placed in the ON position and the system is operation properly.

AUTOMATIC DIRECTION FINDING (ADF) SYSTEMS (COLLINS)

Two radio compass systems give ADF indications from radio frequency signals in the range from 90 to 1800 kc. This band of frequencies includes transmissions by low frequency ranges, homing beacons, and broadcast stations. The magnetic bearing of a radio station from the airplane is indicated by the pointers on the radio magnetic indicators on the pilot's and copilot's instrument panels when the RMDI selector switches are positioned to ADF. Two stations driving the pointers of an RMDI provide a running fix of position.

The ADF receivers combine signals from directional loop antennas and non-directional sense antennas to indicate the direction from which radio station signals reach the airplane. The remote compass system positions the dial card on the radio magnetic indicator so that the pointers show the magnetic heading to fly to reach the station tuned in on the receiver. The loop antenna can also be controlled manually and the direction of a station ascertained by listening for an aural null; but the aural null is ambiguous since it does not indicate whether, for example, the station is ahead of or behind the airplane.

Reception in the relatively low frequencies covered by the ADF receivers is nearly independent of altitude; but these frequencies are sensitive to atmospheric conditions and static. Modulated and unmodulated signals can be received. ADF receiver audio reaches the headsets through the flight interphone system and the 1-ADF-2 toggle switches on the audio selector panels. Range filters in the interphone system enable headset reception of range modulation or voice signals.

Each ADF radio compass system consists of a control panel, an ADF receiver, a loop antenna, a sense antenna, a sense antenna coupler, and two indicators. The control panels are located on the pedestal between the pilots, the receivers are installed in the racks in the electronics compartment, the loop antennas are installed on the bottom of the fuselage, and the sense antennas and antenna couplers form part of or are located in the dorsal fairing. The radio magnetic indicators on the pilots' panels also receive input signals from the remote compass system. Figures 18-8 and 18-9 show components and a block diagram of ADF navigation system No. 1.

ADF Control Panels

Two Collins control panels, located on the panel forward of the pedestal, control the ADF systems. The left panel controls the No. 1 ADF radio compass system; the right panel controls the No. 2 ADF system.

The ADF control panels carry the following controls: gain control; BFO ON-OFF toggle switch SHARP-BROAD SELECT toggle switch, three frequency selector knobs, loop control switch, and function control switch. The selected frequency appears in the window near the center of the control panel; and the tuning meter in the upper right corner of the control panel facilitates sharp tuning of the ADF receiver.

The gain control controls the gain of the ADF receiver.

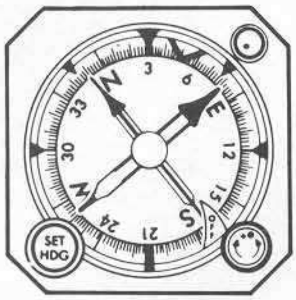
The BFO (beat frequency oscillator) control gives a beat note that enables reception of CW signals and helps to tune in distant or weak stations.

The SHARP BROAD SELECT toggle switch varies the selectivity of the receiver by selecting either a broad or a narrow bandpass filter in the receiver. If the receiver is correctly tuned while the switch is in SHARP position, the tuning will be correct if the switch is actuated to BROAD position. If the receiver is tuned with the switch in BROAD position, however, it may have to be retuned slightly when the switch is actuated to SHARP position.

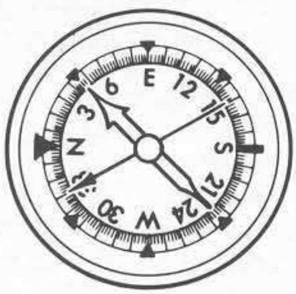
The three knobs in the center of the panel near the frequency window select the frequency that appears in the window. The left knob selects the hundreds figures; the center knob selects the tens figure; and the right knob selects the units figure in continuous tuning, and therefore permits sharp, accurate tuning.

The LOOP control switch at the lower left side of the control panel is used to rotate the loop for all positions of the mode switch at the right side of the panel. The LOOP control has two positions in each direction. When the switch is actuated toward either L or R, the loop rotates left or right slowly. When the LOOP switch is actuated to the extreme position, the loop rotates rapidly. The LOOP switch is used to find either null or maximum signal position, usually null, when the mode switch is in LOOP position; that is, when manual direction finding is employed. When precipitation static is intense, the electrostatically shielded loop sometimes improves communications. To improve voice reception during static, the mode selector switch is actuated to ANT position and the LOOP switch is used to rotate the loop to maximum receiver signal strength.

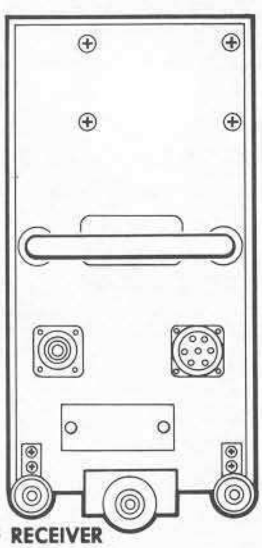
The mode selector switch at the lower right side of the panel turns the ADF receiver on and off. ADF position of the switch sets up the system for automatic direction finding indications of the direction from which the radio signal is arriving at the airplane. ANT position of the switch makes the receiver function with the non-directional antenna, and provides no directional indications. LOOP position of the mode selector switch makes the receiver use only



RADIO MAGNETIC DIRECTION INDICATOR

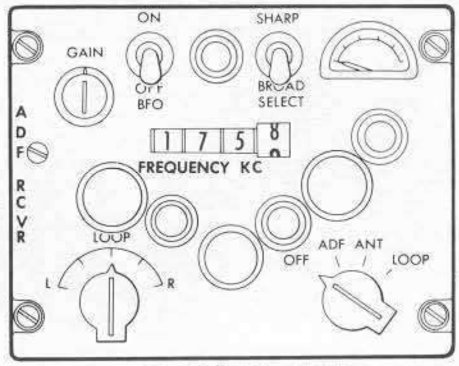


RADIO MAGNETIC INDICATOR

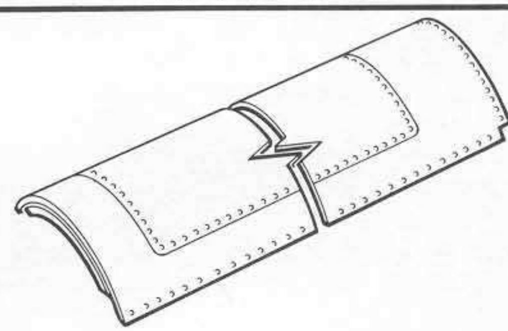


ADF RECEIVER

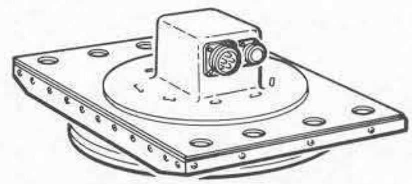
30.02.34.127



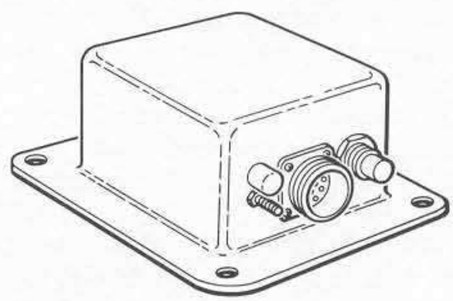
ADF CONTROL PANEL



ADF SENSE ANTENNA

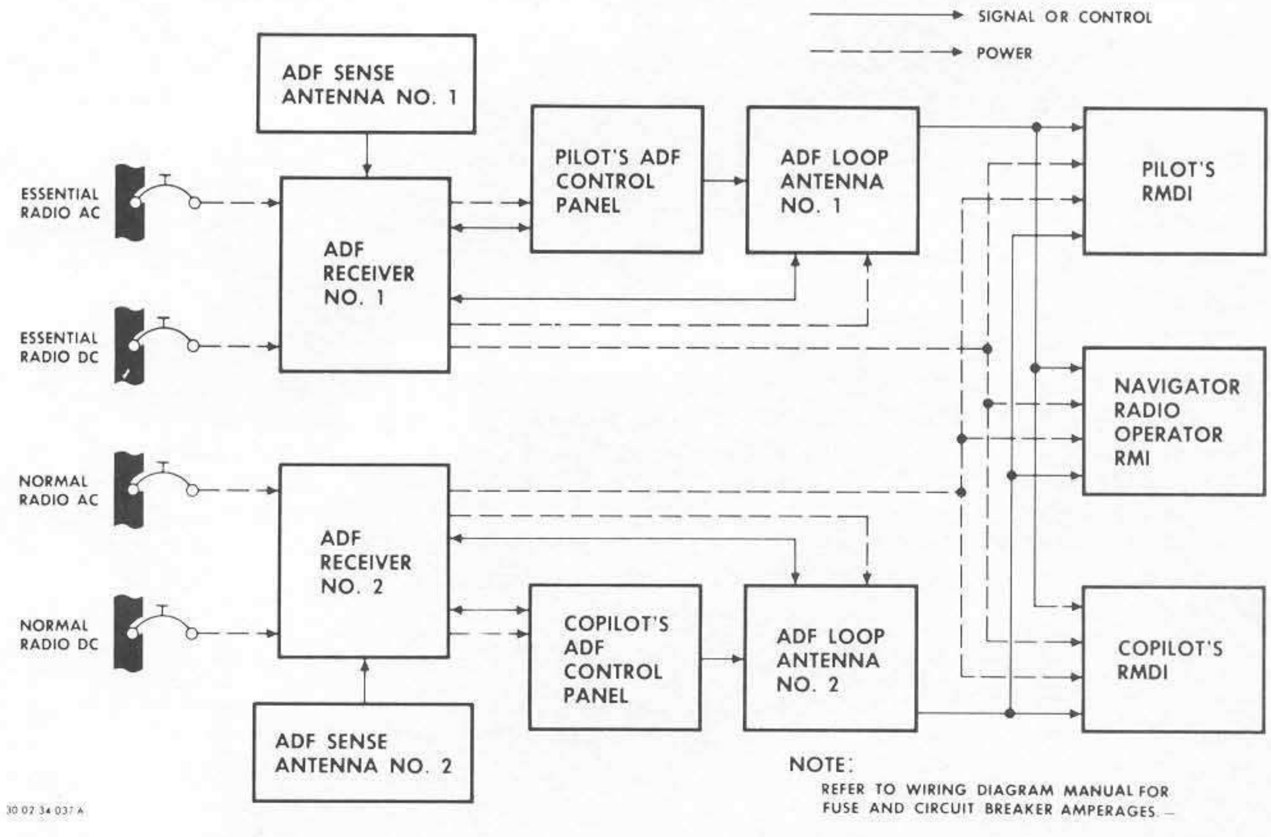


ADF LOOP ANTENNA



ANTENNA COUPLER

ADF System Components
Figure 18-8



ADF System No. 1 Block Diagram
Figure 18-9

loop antenna signals. LOOP position of the mode switch is used for manual direction finding by using headset aural null positions of the loop antenna, and for communications when severe static conditions prevail, as described in the preceding paragraph.

The control panel tuning meter aids in accurate tuning. Best tuning is indicated by maximum deflection of the meter needle to the right. This tuning meter also aids in tuning weak stations while using the BFO switch function.

Radio Magnetic Direction Indicator

A radio magnetic direction indicator is located at the right side of the center row of instruments on each pilot's instrument panel. The radio magnetic direction indicator consists of a rotatable compass card and two pointers. The compass card is positioned by the compass to display heading information. The single- and double-barred pointers are positioned by the ADF receivers. The No. 1 ADF receiver positions the single-barred pointer; the No. 2 ADF receiver positions the double-barred pointer.

The compass card rotates with changes in airplane heading to show heading at a lubber line at the top of the indicator. Pointer indications are therefore so related to compass card position that the pointers show the magnetic bearing from the airplane to the radio station tuned in on the ADF receivers.

ADF Receivers

Two Collins receivers cover the radio frequency range from 90 to 1800 kc and provide automatic direction finder indications on the radio magnetic indicators. The operation and functions of the receiver are controlled remotely by the control panel on the lower part of the pilots' center instrument panels. Frequency selection at the panel, automatically tunes the receiver. The BFO switch is used to tune unmodulated stations or to help locate weak stations. Maximum swing of the tuning indicator needle indicates best tuning.

ADF Loop Antennas

Two Collins flush-mounted loop antennas are installed on the bottom of the fuselage about four to seven feet forward of the wing leading edge. The loops are removed and replaced from the exterior of the airplane. Electrical connections are made through two receptacles on the loops.

Loop rotation is automatically controlled by the receiver during automatic direction finding operation. The loop can also be rotated by the LOOP control knob on the control panel in the flight compartment for aural null operation. The loop rotates rapidly when the LOOP control knob is turned to the extreme position. The loop rotates slowly if the control knob is turned only to the first index mark. For ADF indications loop antenna signals are combined with sense antenna signals in the receiver, and the loop rotates automatically toward no-signal or null position.

ADF Sense Antennas

Two sections of the antenna fairing between stations 899 and 1125, which are insulated from the structure of the airplane, are sense antennas for the ADF systems. Signal inputs to the ADF receivers from these sense antennas are nearly free of directional characteristics. Receivers combine loop and sense antenna signals to develop the automatic direction finding function. The sense antennas offer low input capacitance to the receivers which are designed for high input capacitance. The sense antennas are therefore matched to the receiver inputs by antenna couplers.

The insulated metal plate of the center part of the antenna fairing is connected to a sense antenna coupler, and the coupler supplies signal input to ADF No. 1 receiver. Similarly, the aft part of the antenna fairing serves as a sense antenna for the ADF No. 2 receiver.

MARKER BEACON SYSTEM

The marker beacon system receives signals from 75 mc marker beacon transmitters and provides pilots with audible and indicator light indications of the exact position of the airplane. Marker beacon stations are located at known points along glide slope paths, along the legs of range stations, and at range station positions. Marker beacons radiate signals vertically in cone, fan, or bone-shaped patterns. Z-type markers are located in the cone of silence of range stations to indicate that the airplane is over the station.

Outer, inner, and airways station marker signals are modulated by 400, 1300, and 3000 cps, respectively. Frequency-discriminating circuits in the airplane receiver activate relays and illuminate indicator lights on the right side of the pilot's instrument panel when the airplane is over a station. A blue light indicates the outer marker, an amber light indicates an inner marker, and a white light indicates the airways marker. The audio frequency modulation is available at the pilot's headsets and serves to further identify marker locations. Outer and inner marker signals give intermittent light and audio tone indications, and may be keyed in the Morse code to identify range station legs. The airways marker gives a steady light and audio tone indication.

The marker beacon systems consists of a marker beacon receiver, a marker beacon antenna, two marker beacon panel light indicator assemblies, and a control panel. The marker beacon system is controlled by the marker OFF-LO-HI switch and the volume control of the ATC control panel on the pedestal. Audio frequency to the headset is selected by the MKR toggle switch on the audio selector panel. The OFF-LO-HI switch on the ATC panel turns the receiver on and off and controls the sensitivity of the system to enable sharper indications when in a strong signal field, as on a glide slope path approach.

The audio frequencies of 400, 1300 and 3000 cps that modulate marker beacon transmitters are made available for headset reception and aural identification of marker beacon stations. The MKR toggle switch on the audio selector panel connects marker beacon signals to the headset. The three frequencies identify outer, inner, and airways markers, respectively. The outer and inner markers provide coded tone and light signals; the airways marker provides a continuous tone and a steady light while the airplane is over the marker beacon station.

TRANSPONDER BEACON SYSTEM

The transponder beacon system assists the air traffic controller in identifying aircraft within his control area and in tracking aircraft through heavy ground clutter and precipitation.

The transponder beacon system utilizes an interrogator unit located at the air traffic control center and a transponder unit installed in the airplane. The ground interrogator is coupled to the airport surveillance radar and its antenna rotates in synchronism with the ASR antenna.

The three components of the transponder system are the antenna, the transponder unit in the electrical compartment, and the ATC control panel in the flight compartment.

LORAN NAVIGATION SYSTEM (Edo Model 345)

LORAN is an abbreviation for Long Range Navigation. The LORAN system may be used at ranges up to 900 miles during the day and 1400 miles at night, depending on atmospheric conditions.

LORAN makes use of pulsed radio signals broadcast alternately from pairs of ground transmitting stations having fixed geographic locations. The station originating the pulse is called the MASTER while the station which transmits after a fixed time interval is called the SLAVE. The function of the LORAN receiver is to measure the time interval between the reception of signals from the master and slave stations. The elapsed time between the arrival of the two signals will be equivalent to a measure of the difference in distance from the transmitting stations to the receiver. When two or more time difference readings have been made from the sets of stations, the navigator may determine his position by comparing his readings with those appearing on standard charts.

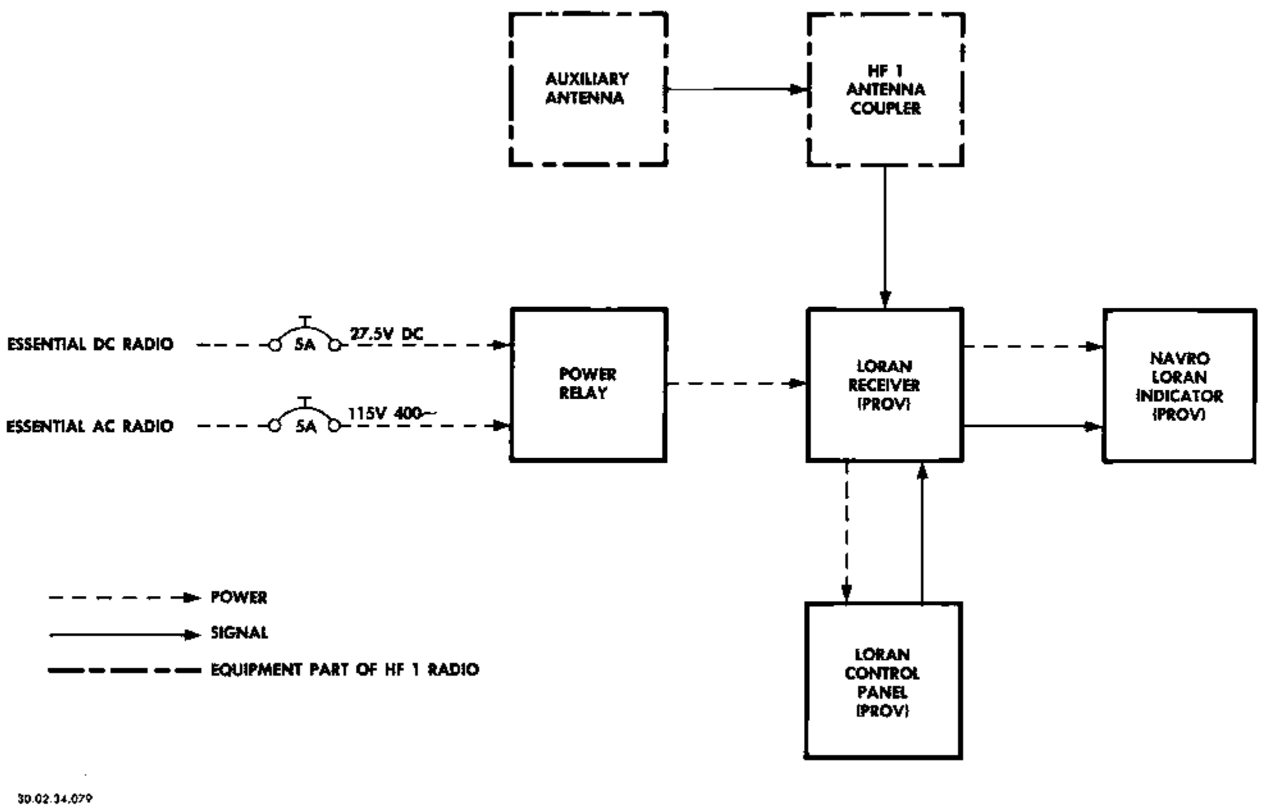
The signals may be received under all weather conditions, subject only to interference from severe electrical storms, to man-made interference, and to limitations imposed by the distance of the receiver from the transmitting stations. Under favorable conditions, with the modern direct reading loran equipment, a fix may be obtained in approximately 30 seconds.

The LORAN system consists of three major units, the receiver, the control unit and the indicator (see Figure 18-10).

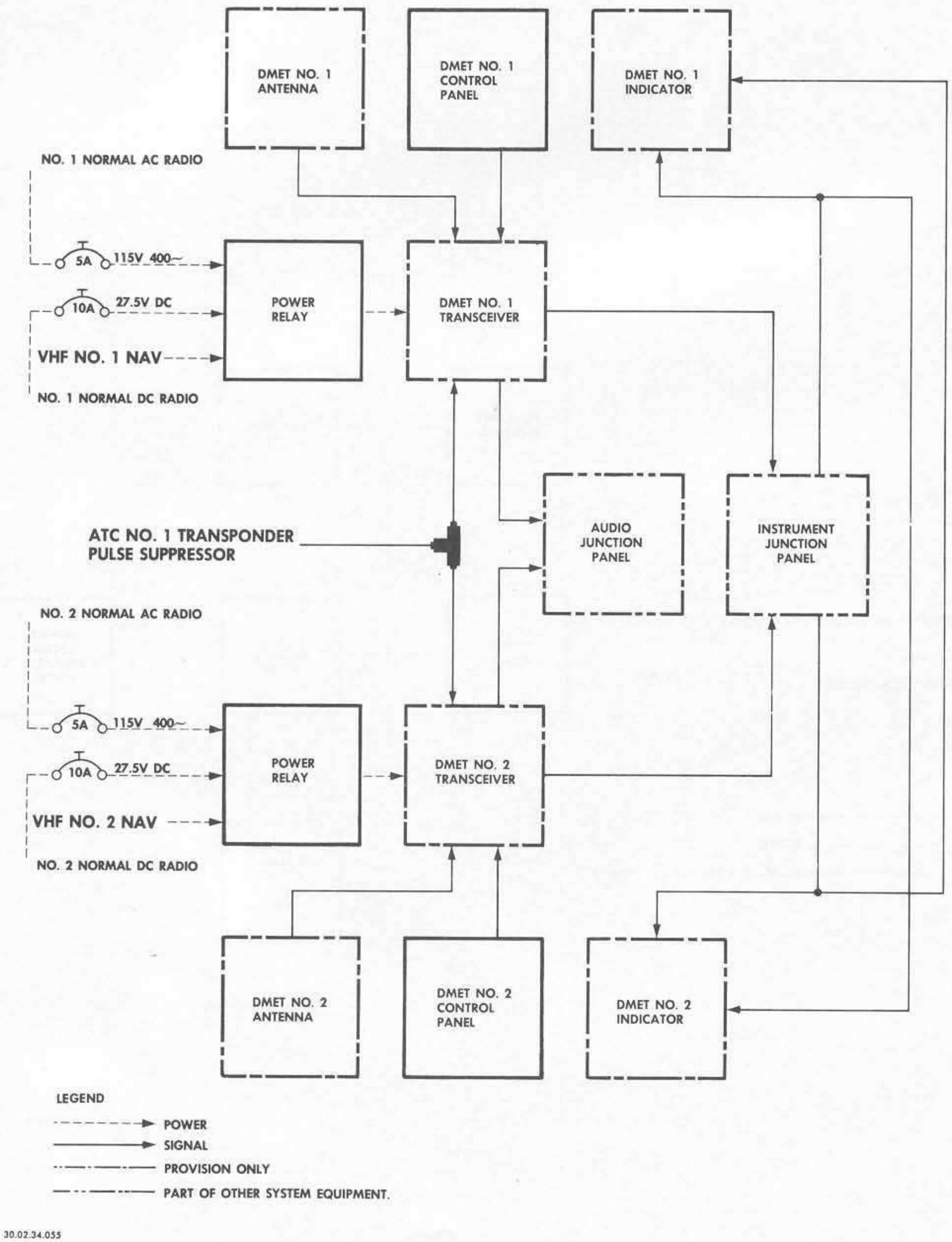
DISTANCE MEASURING EQUIPMENT-TACAN (ARINC)

The two DMET systems provide distance measuring facilities. Each system comprises a receiver-transmitter, a control panel, a distance indicator, and an antenna. (See Figure 18-11.)

The No. 1 DMET system receiver-transmitter unit is located beside the VHF No. 1 navigation receiver with whose frequency control the DMET unit is associated. The No. 2 DMET system receiver-transmitter is located beside the VHF No. 2 receiver.



EDO LORAN System
Figure 18-10



DME Equipment - TACAN System No. 1 and
No. 2 - Block Diagram
Figure 18-11

The control panels are located on the pilots' pedestal. The left VHF control panel, which controls VHF No. 1 communications sytem and VHF No. 1 navigation system, carries a toggle switch for control of the DMET No. 1 system. This toggle switch is marked DMET-STDBY. Similarly, the right VHF-VOR control panel controls DMET No. 2 system simultaneously with VHF communications and navigation systems No. 2. DMET and VHF navigation systems are associated because they form the airborne part of the airways VORTAC system. Selection of the VHF frequency of an omnirange station also selects the correct Tacan channel if VORTAC facilities are available. VORTAC enables a pilot to determine at once his direction and distance from an omnirange station.

The DMET receiver-transmitter is a transponder that interrogates a ground beacon. Distance in formation is obtained from beacon replies to coded interrogating pulses; the receiver admits only signals triggered by the transmitter interrogations.

DMET measures the time interval between transmission of interrogating pulses and reception of the replies and presents the information as the distance of the airplane from the beacon in nautical miles. This distance is displayed in numbers from zero to 195 on dual indicators located at the right side of the lower rows of instruments in the pilots' instrument panels. Readout can also be displayed in a window in the RMI units in lieu of separate indicators. (See Figure 18-12.)

Beacons are identified by signals in Morse code. DMET receiver audio is supplied to headsets through toggle switches on the audio selector panels. The switches are marked DMET-1 and DMET-2.

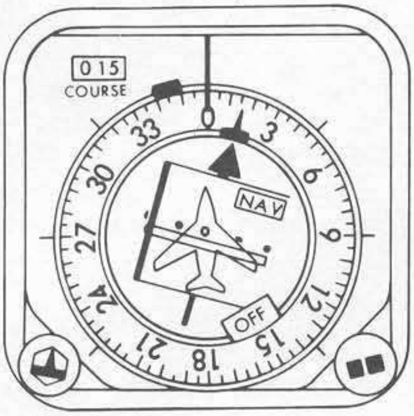
Power Requirements

The No. 1 DMET system takes electrical power from the No. 3 essential ac bus and from the 28-volt dc essential bus. The No. 2 DMET system takes electrical power from the No. 3 essential ac bus and from the 28-volt dc essential bus.

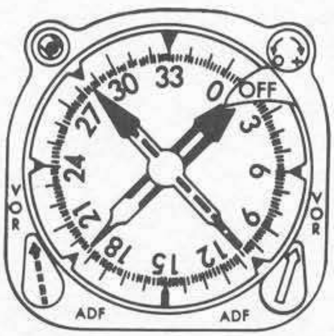
DOPPLER RADAR (PROVISIONS ONLY)

Doppler radar (dopplerad) is an electronic aid to navigation. It assists the pilots and navigator/radio operator by presenting ground speed and actual track information on an indicator. The dopplerad antenna radiates bursts of high power energy toward the ground. Part of this energy is reflected back into the antenna. Through receiver circuitry an apparent change in frequency is interpolated to give drift angle and ground speed. Figure 1 shows a block diagram of the doppler radar system with connections from the doppler navigation computer. This computer incorporates signals from other flight navigation systems to present the pilots a combined picture of actual airplane progress and speed. The system also works over water where no land marks are visible to navigate by.

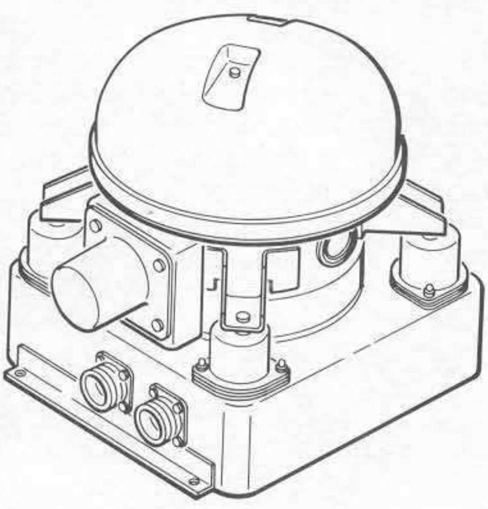
Except for the antenna the doppler system is not installed but all wiring and mounting provisions are in the airplane.



COURSE DEVIATION INDICATOR

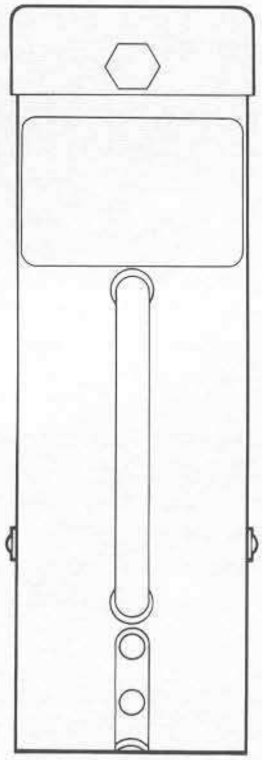


RADIO MAGNETIC DIRECTON INDICATOR

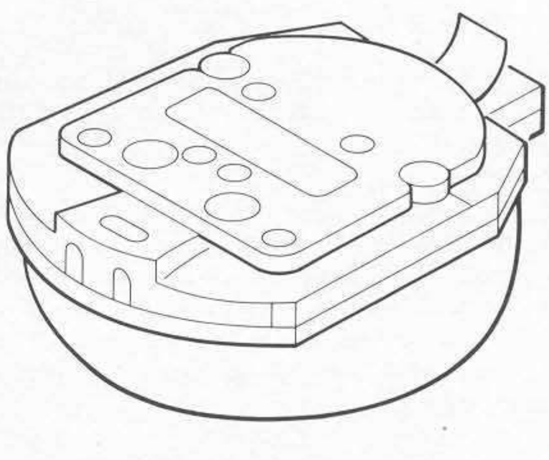


DIRECTIONAL GYRO

30.02.34 098



INSTRUMENT AMPLIFIER RACK



FLUX VALVE AND COMPENSATOR

Remote Compass System Equipment
Figure 18-12

REMOTE COMPASS SYSTEMS

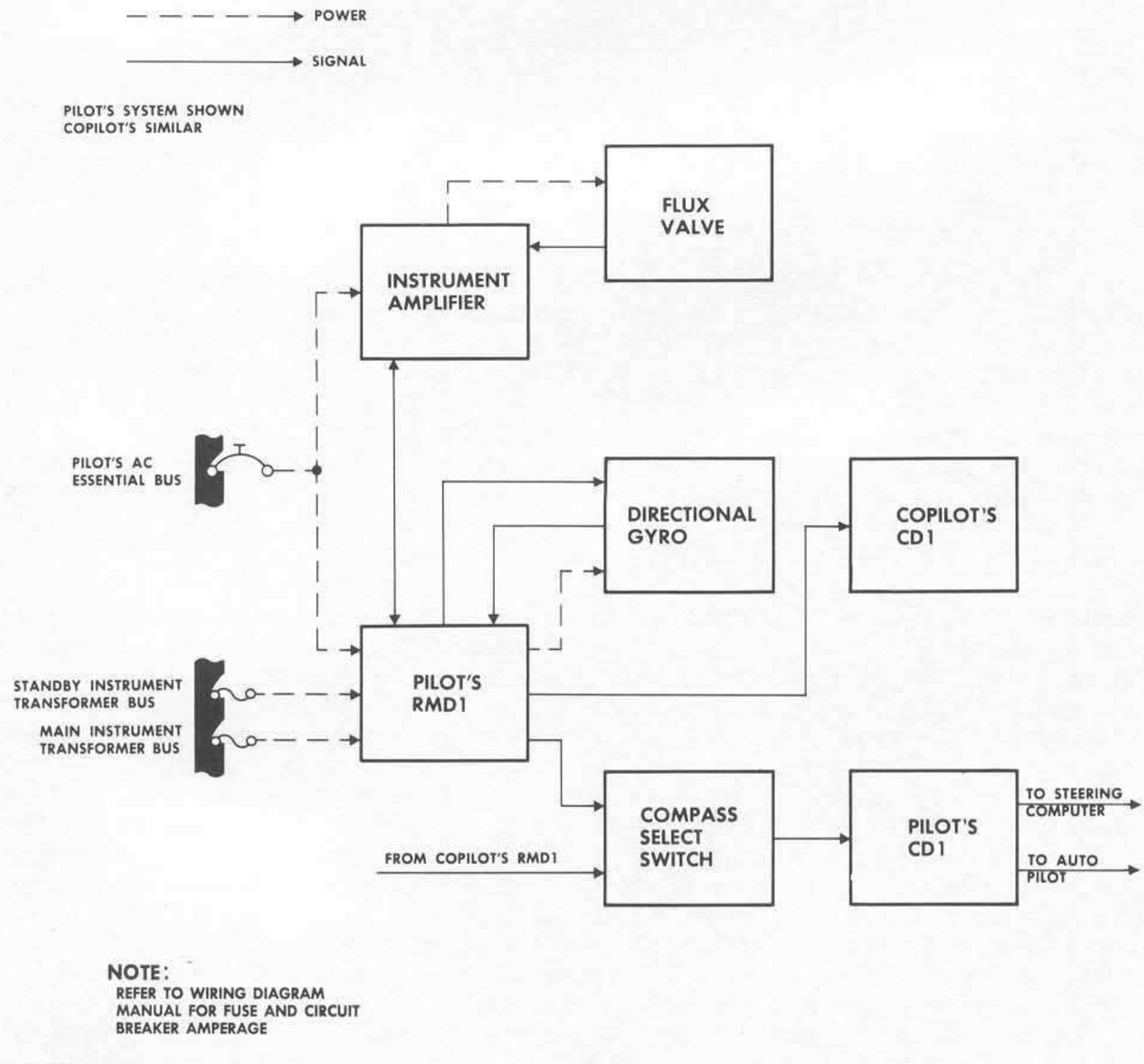
The primary function of the remote compass system is to sense and display magnetic heading. The sensor for the system is a flux valve mounted in the airplane wing tip, and the display component is a radio magnetic direction indicator (RMDI) counted on the pilots' instrument panel. A course devation indicator (CDI) mounted on the pilots' instrument panel provides a secondary display. A directional gyro stabilizes flux valve information to damp transient output variations and oscillation.

A secondary function of the system is to supply gyro heading reference when magnetic heading reference becomes unreliable or unusable. In high latitudes, values for variation become extreme and magnetic heading changes rapidly as the airplane progresses along its flight path. In these conditions the operator may select free gyro mode in which heading is referenced to the stable gyro spin axis. The indicators then display gyro heading which may be related to an arbitrary grid imposed over charts of the flight area.

Each compass system includes a flux valve and compenstor, directional gyro (DG), RMDI, CDI, and integrated flight and navigation instrument amplifier rack, and a compass mode selector switch. The flux valve for each system is located in the right wing tip. The amplifier racks are located in the electrical compartment. The compass mode selector switch is on the flight director system control panel. The indicating instruments are located on the pilots' instrument panels. The directional gyro is located in the autopilot compartment. (See figure 18-12).

Operation (Slaved Gyro Operation)

The directional gyro establishes gyro heading and transmits this information by means of a servo loop to the RMDI. (See Figure 18-13.) A gear train in the RMDI mechanically positions the rotor of a servo mechanism on the basis of gyro heading. Magnetic heading sensed by the flux valve is applied to the stator of the stator of the servo mechanism positioned to gyro heading. Null for this servo mechanism exists when gyro heading is equal to magnetic heading. The RMDI gear train positions the compass card of the RMDI and when the card is read against the fixed lubber line the reading is gyro heading or, when the servo mechanism is at null, gyro heading is equal to magnetic heading and the reading is magnetic heading. When gyro heading is different from magnetic heading, the servo mechanism on the RMDI gear train will not be at null and will transmit an error output. This output is amplified and applied to a torquer motor in the directional gyro. The torquer motor precesses the gyro in a direction to cancel the error between gyro heading and magnetic heading. As the gyro is precessed, the gyro heading servo output from the gyro causes the RMDI followup gear train to drive the servo mechanism toward null, and when gyro heading is equal to magnetic heading, the torquing signal collapses and the loop comes to rest. Except to cancel relatively small errors caused by gyro drift, once this alignment is established, further correction is unnecessary even through the airplane heading changes since the gyro will maintain a fixed directional relationship between its spin axis and the earth's lines of magnetic flux.



Remote Compass System - Block Diagram
Figure 18-13

If a torquing signal from the servo mechanism in the RMDI exists, the compass card is not indicating accurate magnetic heading. The torquing signal is applied to an annunciator on the face of the RMDI to inform the pilot that an error exists and in which direction. The pilot may then manually synchronize the RMDI gear train by the synchronizing knob on the RMDI. Rotating the synchronizing knob in the direction indicated by the annunciator advances the RMDI gear train through a mechanical differential to override the servo motor which normally dirves the gear train. The pilot uses the synchronizing knob at the start of operation to reduce the time it would normally take for the gyro to precess to the correct position.

The RMDI gear train positions three transmitting servomechanisms to supply magnetic or gyro heading to other equipment. Output of the pilot's RMDI is used to position the compass card of the copilot's CDI and to supply magnetic heading to the VHF navigation system. Output from the copilot's RMDI is used to position the compass card on the pilot's CDI and to supply magnetic heading information to the pilot's CDI to be used to generate heading and course error signals for the autopilot and flight director systems.

Free Gyro Operation

In free gyro operation the slaving torquer amplifier is deenergized and the torquer motor is disabled. The gyro then maintains a fixed position in the horizontal plane and the RMDI displays gyro heading without reference to the earth's magnetic field. The pilot may choose any arbitrary heading reference by manually positioning the RMDI gear train and display with the synchronizing knob.

Flux Valve

The thin type flux valve detects the direction of the horizontal component of the earth's magnetic flux with respect to the longitudinal axis of the airplane. Single phase voltage of 115-volts ac, 400 cps is supplied to the flux valve. The output windings of the flux valve are wye-connected and provide an output signal voltage of 800 cps. This signal voltage is compared in the RMDI with the heading shown on the card, and a signal indicating direction of error is sent to the slaving amplifier which activates the directional gyro and servo loop to rotate the compass card of the compass indicator and show magnetic heading of the airplane. The flux valve magnetic detector unit is mounted in damping fluid and is kept level by gravity within 30-degree limits. The flux valve has a graduated scale (±10) on the case aft mounting flange. Electrical connections are made through cable connectors. Both flux valve units are located in the right wing tip.

A magnetic compensator mounted on the top of the unit forms part of the flux valve. The compensator provides compensation for the magnetic influence of the airplane on the flux valve. Four adjusting screws on the compensator permit independent north-south and east-west compensation. The forward flux valve is in the pilot's remote compass system, the aft flux valve in the copilot's remote compass system.

Directional Gyro

The electrically-driven directional gyro is mounted on an inner and an outer gimbal. The outer gimbal allows full 360 degree rotation of the gyro in azimuth. Longitudinal tipping of the gyro within the inner gimbal is limited to 85 degrees. A liquid level switch controls current to a torquing motor that maintains the gyro level within plus or minus one degree. The directional gyros for both remote compass systems are located beneath the electronic equipment rack in the forward left side of the electrical compartment. Electrical connections to the gyros are made through two electrical receptacles on each unit. The gyros are driven by 115-volt 400-cps ac electrical power.

Slaved gyro mode of operation places the directional gyro under the control of the flux valve. Amplified flux valve signals activate the gyro torquing motor and cause the gyro to rotate in azimuth until the compass card reaches the correct magnetic heading.

Free-gyro mode of operation uses the characteristic of rigidity of the gyro to provide a heading reference; the random drift rate of the gyro is low. After the gyro erects it may be set to a desired heading by a magnetic compass or other reference, and thereafter maintains a stable heading reference. The directional gyro positions the compass card of the RMDI; the card can be rotated by any heading by turning the upper right knob on the instrument.

The flux valve and the directional gyro are synchronized when the heading displayed by the compass card corresponds to the heading sensed by the flux valve. This synchronization is shown on the annunciator on the compass indicator. In flight the annunciator shows a dot or cross alternately because of small yawing movements of the airplane. If either the dot or the cross remains in the annunciator, the system is not synchronized and the compass card indication is in error. Slaving circuits operate all the time in slaved mode to bring the directional gyro to the magnetic heading of the flux valve. If the compass system is far out of synchronization the system can be synchronized at a higher than normal rate by turning the upper right knob of the RMDI toward the dot or toward the cross, whichever appears in the annunciator.

Remote Compass Mode Selector Switch

A selector switch to show either NORMAL or ALT position is mounted on the flight director system control panel located in the bottom right corner of the pilot's instrument panel. When the switch is in the NORMAL position, the pilot's CDI receives heading information from the copilot's remote compass system. When the switch is in the ALT position, the pilot's CDI receives heading information from the pilot's remote compass system. The copilot's CDI always receives heading from the pilot's remote compass system.

Radio Magnetic Direction Indicator

The radio magnetic direction indicators are located on the pilot's and copilot's instrument panels. The RMDI has a calibrated rotatable compass card, two pointers, two selector switches, a control knob to allow rapid manual alignment of the card on start-up, a power failure flag and an annunciator.

The two pointers on the indicator are driven by the output signals of either the VHF (VOR) or ADF receivers, depending upon the position of the ADF-VOR selector knobs. The single-barred pointer receives information from ADF No. 1 or VOR No. 1 and the double-barred pointer receives information from ADF No. 2 or VOR No. 2. The pointers show the magnetic bearing of the ADF or VHF stations to which the receivers are tuned.

An annunciator to the upper left side of the indicator shows if the compass system is synchronized while operating with the compass mode selector switch in NORMAL position. A dot or cross shows in the annunciator if the compass system is not synchronized. In slaved mode, the system slowly synchronizes automatically, synchronization being indicated when the dot or cross alternately appear in the annunciator. When the airplane is stationary on the ground and the system is aligned, the annunciator will be blank.

The upper right knob on the indicator is engraved with two arrows, a dot and cross. The knob can be used to accelerate synchronization. Steady appearance of the dot or cross in the annunciator signifies that the compass card heading is not the correct magnetic heading. If the pilot desires to synchronize the compass system rapidly, the knob is turned in the direction of the dot or cross that appears in the annunciator. See Figure 4 for illustration of the RMDI.

Course Deviation Indicator

The course deviation indicators are located on the pilot's and copilot's flight instrument panels. The CDI used remote compass information in its displays of flight information, and is a repeater for the RMDI. This indicator, however, is primarily utilized in the flight director system.

Integrated Flight and Navigation Instrument Amplifier Racks

The flight and navigation instrument amplifier racks for each remote compass system are located in the electronic equipment rack at the forward left side of the electrical compartment. Each rack includes interconnections for five plug-in modules, four servo amplifiers and a slaving amplifier. The two racks are identical and interchangeable. The slaving amplifier provides the annunciator signal that appears on the RMDI, and slaves the directional gyro to flux valve signals. The amplifiers are fully transistorized units.

GLIDE SLOPE SYSTEMS

The glide slope systems provide reception of radio facilities of instrument landing systems (ILS) in the frequency range of 329.3 through 335.0 mc. Glide slope frequencies are paired with localizer frequencies; and the correct glide slope frequency is automatically selected when the localizer frequency is selected.

Two complete identical glide slope receiver systems are installed in the airplane. These receivers are electrically interconnected with the VHF navigation systems so that selection of a localizer frequency on the VHF receiver switches on the glide slope receiver and prepares it for operation on the corresponding glide slope frequency. No. 1 VHF navigation control panel switches on No. 1 glide slope receiver; No. 2 VHF navigation control panel switches on No. 2 glide slope receiver. Figure 18-14 shows a block diagram of glide slope system No. 1.

Glide slope transmitters, located near touchdown point of runways, transmit beams on the selected angle of descent. The upper part of the beam is modulated by 90 cps, the lower part of the beam is 150 cps. The position of the airplane along the beam is visually indicated on the course deviation indicator by the horizontal bar. If the airplane is too high on approach, 90 cps modulation predominates, and the horizontal bar is displaced downward to indicate to the pilot that he should lose altitude. If the airplane is too low on approach, 150 cps modulation predominates, and the indicator bar is displaced upward. In the center of the beam the indicator bear remains centered to show that a correct rate of descent is being maintained.

Glide slope No. 1 receiver output connects into the autopilot system. In automatic glide slope mode of operation the autopilot turns the airplane into the glide slope and guides the airplane at the correct rate of descent along the glide slope beam.

Control Panels

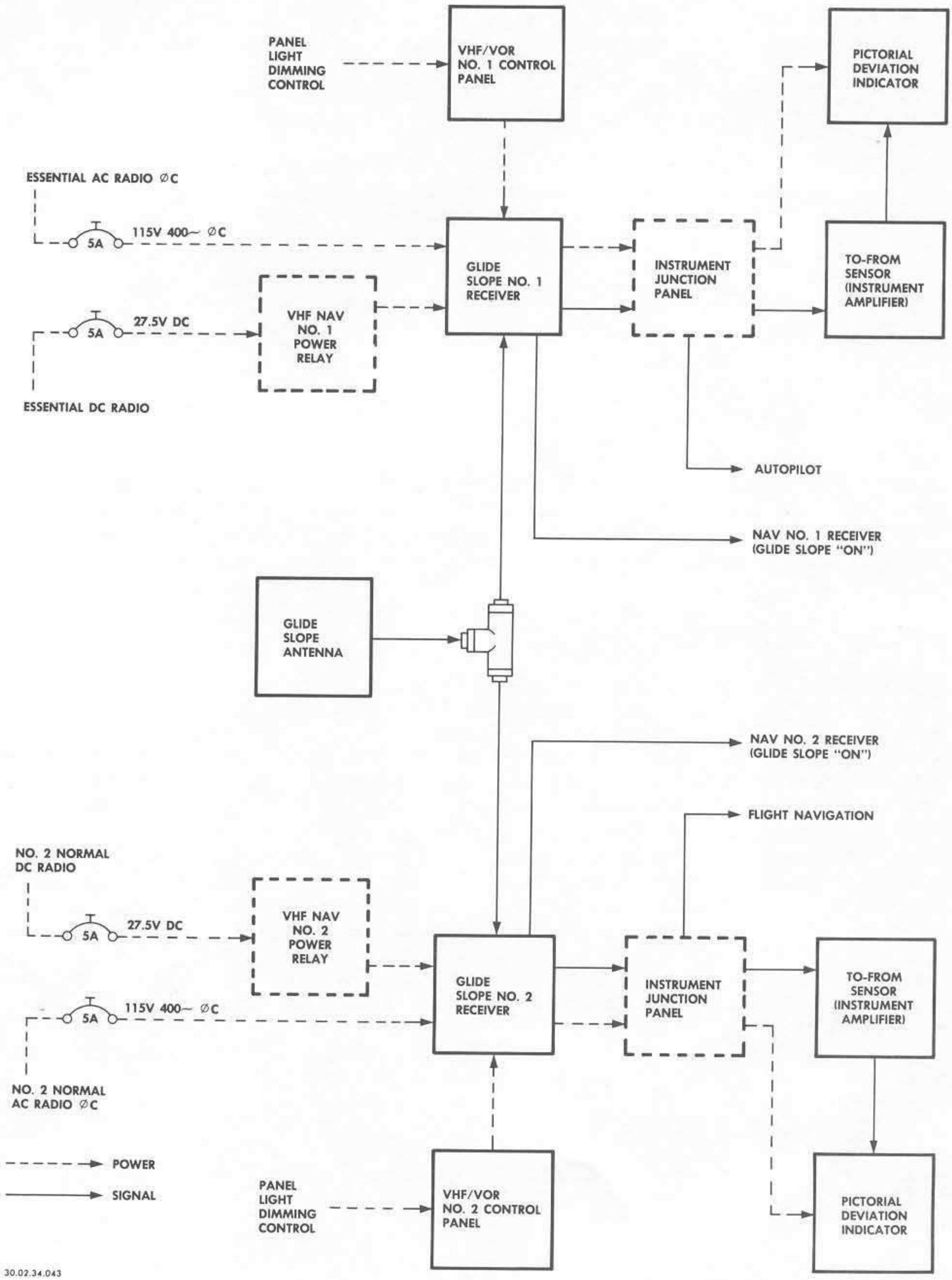
The control panels for the VHF navigation systems control the operation of the glide slope systems. No. 1 glide slope system is linked to the No. 1 VHF navigation system, and No. 2 glide slope system is linked to No. 2 VHF navigation system. Selection of a localizer frequency on a VHF navigation receiver selects the corresponding paired glide slope frequency on the glide slope receiver and switches on electrical power for the glide slope receiver. Glide slope receivers receive electrical power only when the VHF navigation receivers are tuned to localizer frequencies.

Antenna

The Collins glide slope antenna is installed on the forward side of the bulkhead in the radome above the radar antenna. The antenna coaxial cable is connected to a fitting on the aft side of the bulkhead. Both glide slope receivers connect to and use this antenna by means of a tee-fitting.

MAGNETIC COMPASS

A standby magnetic compass is provided in the glare shield above the pilot's instrument panel. The compass is stowed in the glare shield when not in use. To use the instrument, it is lifted into the line of vision above the glare shield.



Glide Slope System - Block Diagram
Figure 18-14

The standby compass indicates the heading of the airplane with reference to magnetic north. The compass card is graduated in 5-degree increments and numbered every 30 degrees. Damping fluid surrounds the compass card assembly to reduce oscillations and a small light illuminates the card. The light is turned on by a switch on the copilot's flight instrument panel.

The compass is compensated for deviation by adjusting a pair of permanent magnets located in the compass compensator below the compass bowl. The compass correction card showing the deviation remaining after the compass has been compensated is installed above the compass dial.

TURN AND SLIP INDICATOR

The turn and slip indicator is a combination of two instruments, a turn indicator and a laterally mounted inclinometer. The turn indicator consists of an electric-driven gyro and a pointer which indicates the direction and the rate at which the heading of the airplane is changing.

The inclinometer consists of a black ball free to move from side to side in a curved glass tube filled with a damping liquid. The position of the ball indicates the net result of the lateral forces acting on the airplane. When the ball is centered, the airplane is in perfect lateral balance. When the ball is displaced from center, the lateral forces are unbalanced and the airplane will slip or slide in the direction in which the ball is displaced from center.

RATE OF CLIMB INDICATORS

A rate of climb indicator is installed in the center row of instruments of the pilot's and copilot's instrument panels. The rate of climb indicator shows the rate at which the airplane is changing altitude. The scale is calibrated in thousands of feet per minute, from zero to six thousand fpm. The upper half of the dial registers ascent, the lower half descent. The rest position of the indicator pointer is horizontal.

Static pressure from the static pressure system is admitted to the interior of a bellows housed in a case to which static pressure is admitted through a capillary tube of controlled leakage. When the airplane changes altitude, the pressure inside the case changes more slowly than the pressure inside the bellows, since air leaks into and out of the case slowly. The pointer, which is linked to the bellows, is deflected from zero position as long as a pressure differential exists between the interior of the bellows and the interior of the case.

The dial is illuminated by a penal light; brilliance is controlled by the instrument lighting rheostat on the copilot's instrument panel.

ELECTRICAL POWER SOURCES

For detailed information concerning the power sources for the flight and navigation instruments, consult the WIRING DIAGRAM MANUAL.

19

RADIO AND COMMUNICATION SYSTEMS

Section 19

RADIO AND COMMUNICATION SYSTEMS

TABLE OF CONTENTS

TITLE	PAGE
GENERAL RADIO AND COMMUNICATION SYSTEMS	19-1
RADIO POWER CONTROL PANEL	19-1
AUDIO SELECTOR PANELS	19-1
Microphone Jack Panels	19-2
RADIO COMMUNICATION SYSTEMS	19-2
VHF Communication Systems	19-2
VHF Control Panels	19-3
HF Communication Systems	19-3
Push-to-Talk Microphone Switches	19-3
SELECTIVE CALLING SYSTEM	19-4
SELCAL Airborne Unit	19-4
INTERPHONE SYSTEM	19-4
Interphone Flight Mode	19-6
Interphone Service Mode	19-6
Interphone Call System	19-6
Handsets	19-6
PUBLIC ADDRESS SYSTEM	19-6
Public Address Control Panels	19-7
Public Address Handsets	19-7
PASSENGER CALL SYSTEM	19-7
TAPE REPRODUCER SYSTEM	19-8
ELECTRICAL POWER SOURCES	19-8

RADIO AND COMMUNICATION SYSTEMS

GENERAL RADIO AND COMMUNICATION SYSTEMS

The communication and radio systems include two HF systems, two VHF systems, a public address system, a flight and service interphone system, an airborne tape reproducer, dual ATC transponder beacons and a dual SELCAL system.

Receivers, transmitters and audio amplifiers are installed on shock-mounted shelves in the electronics compartment located below the flight deck floor. This compartment is accessible during flight emergencies through a door in the floor of the forward coat closet.

Controls for the radio communication systems are located on the pilot's and co-pilot's consoles, on the forward and aft areas of the pilots' pedestal, on the pilot's overhead switch panel and at the flight engineer's station. Interphone and public address system controls are located in the flight compartment and in the passenger cabin. Jacks and jackboxes for service interphone communication are provided in the flight compartment, electronics compartment, wheel wells, and in the engine pods and other service areas of the airplane.

RADIO POWER CONTROL PANEL

Two ac and two dc airplane electrical buses supply power to the radio communication and navigation systems. Electric power from these buses is controlled by two ON-OFF toggle switches in the RADIO POWER panel located on the pilots' overhead switch panel. (The flight and service interphone systems and the P.A. system operate without benefit of switches.)

The NORMAL switch connects the No. 3 essential ac and the 28-volt dc essential buses of the electrical system of the airplane to the No. 1 and No. 2 normal ac and dc radio buses.

The ESSENTIAL switch connects the pilot's ac essential and the dc emergency buses of the electrical system to the essential ac and dc radio buses. The ESSENTIAL switch can be overridden by two circuit breakers in the main circuit breaker panel in the flight compartment.

AUDIO SELECTOR PANELS

Five Gables G-605 audio selector panels select radio receiver, interphone, and microphone audio. Each panel incorporates five pushbutton microphone selector switches, eleven toggle audio selector switches, an ADF filter selector switch, an emergency switch, a master volume control, two lights which illuminate to indicate microphone switch selection, and lights for panel illumination. A transistorized isolation amplifier plugs into the back of the panel. The isolation amplifier permits the selection of one or more audio sources at any panel without interfering with or reducing the level at any of the other four panels. If the isolation amplifier malfunctions, any audio output channel can be heard by moving the EMER-NORMAL selector switch to the EMER position. This switch action by-passes the isolation amplifier. With the switch in the EMER position, the operator should select only one audio source at a time.

The pushbutton microphone selector switches connect the microphones to the VHF, HF, or interphone systems. Actuating a microphone selector switch connects that microphone to a system, illuminates the PB LITES, and releases any previously engaged microphone switches.

With the FILTER switch actuated, a range filter is switched into the ADF audio output circuit.

The SPKR toggle switch on the pilot's and copilot's console panels switches headset audio into the speaker amplifier, enabling the pilot to communicate on interphone or radio without wearing a headset. The corresponding speaker is muted when the pilot or copilot keys his microphone.

Microphone Jack Panels

A Gables G-942 microphone jack panel is installed at each of the four flight compartment stations. Each panel incorporates jacks for connecting a hand-held microphone, a mask microphone, a boom microphone, and a headset to the associated audio selector panel. When the microphones are properly connected to the jack panel and any one of the five microphone selector switches on the audio panel is actuated, the output of mask, boom, or hand microphone is routed to the system selected; no further selection is necessary. Each microphone is keyed by actuating its press-to-talk switch or the yoke control switch. Each microphone panel uses transistor preamplifiers for the boom and mask microphones.

RADIO COMMUNICATION SYSTEMS

The airplane is equipped with two VHF and two HF radio communication systems. The VHF systems provide static-free reception with range limited to line-of-sight. The HF systems provide long range operation in the 2 to 25 mc. radio spectrum.

VHF Communication Systems

Two separate VHF communication systems provide two-way, static-free communication with FAA and company radio stations. Each system includes a crystal-controlled transmitter and a receiver combination designed to operate in 360 channels between 118.0 and 135.95 mc.

The VHF systems are designated VHF-1 and VHF-2. Receiver audio output is selected by the two VHF toggle switches on the audio selector panels. The microphones can be connected to either system by the MIC SEL switches.

Each VHF system includes a receiver, transmitter, remote control panel, power relay, and an antenna.

The VHF-1 system is connected to a blade antenna installed on the underside of the forward fuselage; the VHF-2 system is connected to a flush antenna in the dorsal fairing on the top of the fuselage. The remote control panel is installed on the pilots' pedestal.

The VHF-1 receiver and transmitter are designed to operate in either a single-channel or double-channel mode. In double-channel operation, the transmitter operates on a frequency six mc above that of the receiver. During double-channel operation, the receiver is tuned to the selected frequency and the transmitter adjusts automatically to a frequency six mc higher. Double-channel operation is limited to receiver frequencies from 118.0 to 120.95 mc, and from 127.0 to 129.95 mc.

VHF Control Panels

A control panel is provided for each VHF communication system. The controls for the VHF-1 communication and the VOR-1 navigation systems are combined in a single panel. An identical panel contains the controls for the VHF/VOR-2 communication and navigation system. A white line through the center of the panel separates the two functions. Both panels are installed on the pilots' pedestal.

The controls for each VHF communication system are located on the upper half of the control panels. The control knobs are white and the frequency window is marked VHF. The selector knob at the left rotates an 18-position switch for selecting the whole numbers in the frequency range from 118 to 135 mc. The knob at the right controls a 10-position switch for selecting the decimal settings from 0.0 to 0.9. This arrangement provides for tuning each radio set to 180 different frequencies spaced 100 kc apart in the band between 118.0 and 135.9 mc.

A toggle switch labeled DC-SC is located adjacent to the left selector knob. This switch is used to select the double-channel or single-channel mode of operation. A small SENS knob mounted on the top of the left selector knob governs the sensitivity of the receiver.

HF Communication Systems

Two high frequency communication systems provide voice communication in the frequency range from 2 to 25 mc. These systems enable relatively long-range communication, not being limited by line-of-sight characteristics as is VHF equipment. Communications in the high frequency range, however, are sensitive to atmospheric disturbance and rain static. Both systems use the same fin tip antenna for sending and receiving.

A HF communication system comprises a control panel that selects transmitter and receiver frequencies simultaneously, a transmitter-receiver, two power relays, two antenna relays, two antenna coupling controls, an antenna coupling assembly, and a high frequency communications antenna. Since both systems use the same antenna, simultaneous operation is not possible. The control panel is installed on the right forward side of the pedestal between the pilots. The transceiver unit and the power and antenna relays are installed in the electronics compartment and the coupler controls in the aft section of the airplane.

Push-to-Talk Microphone Switches

Thumb-operated push-to-talk switches on the aft outboard horn of each control wheel are provided for radio or interphone communication. A push-to-talk switch is located on the flight engineer's microphone selector panel.

SELECTIVE CALLING SYSTEM

The SELCAL control panel is located on the pilots' pedestal anc contains two receiver selector switches and a flashing light associated with each switch. The switches, marked VHF-1, VHF-2, HF-1 and HF-2, are used to connect the audio output of any two communications receivers to two detector channels in the SELCAL unit. When the airplane is being called on the frequency of one of the selected radio receivers, the corresponding light will flash and a chime will sound. Before answering the call, the pilot can activate a reset switch to extinguish the light and silence the chime by depressing the light. The system is then ready for the next call.

SELCAL Airborne Unit

The SELCAL airborne unit consists of two decoder channels enclosed in a metal case mounted on the aft bulkhead in the flight compartment. Removing the cover plate exposes eight selector knobs on the front panel of the case, a set of four knobs for each channel. Each knob rotates through 12 lettered positions from A to M omitting the letter I. The code call for each channel is preset by positioning the selector knobs to correspond to the code assigned to the airplane, starting at the upper left corner of each set of knobs.

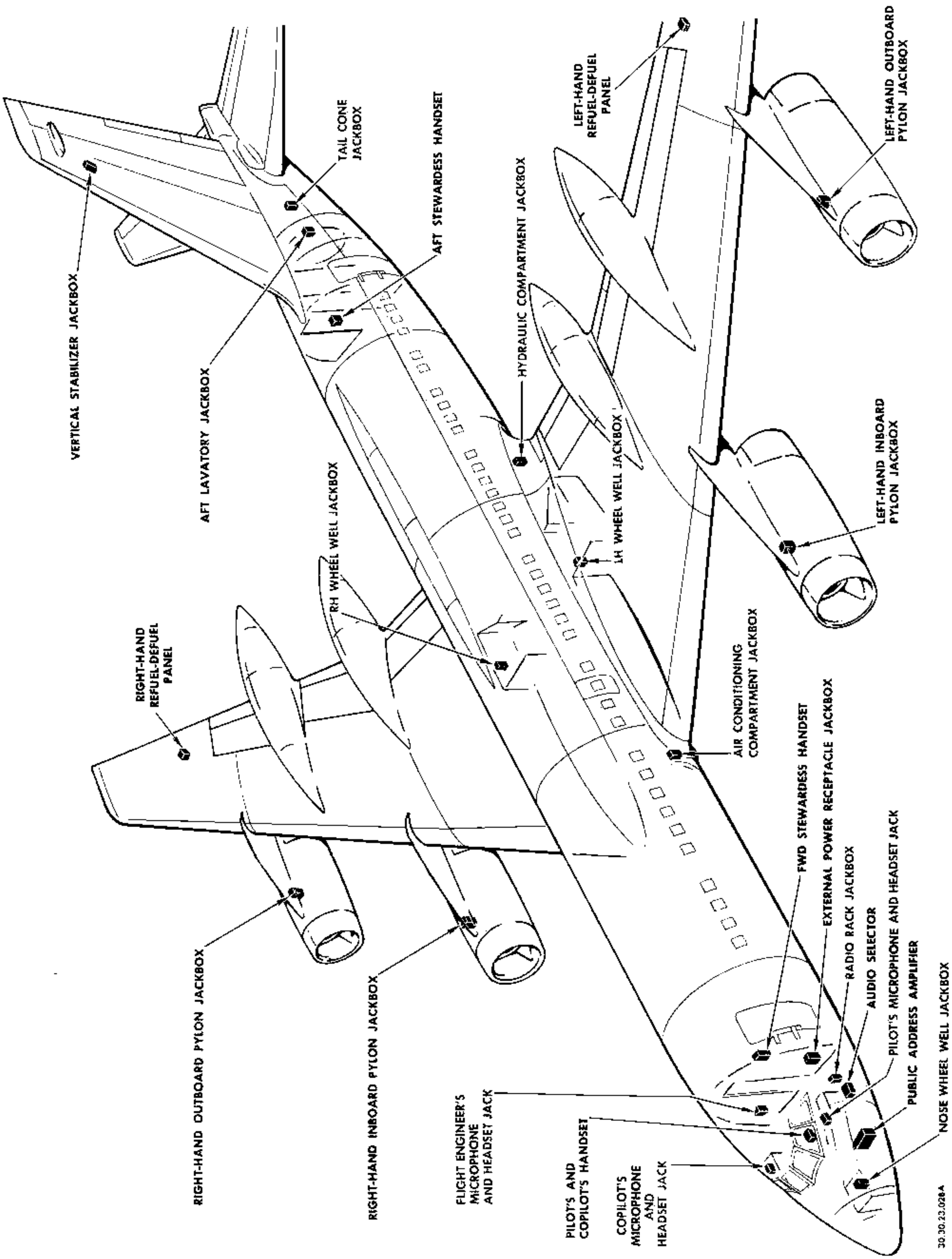
The radio receivers receive SELCAL tone signals from ground stations and relay the signals to the decoders of the SELCAL unit. If a coded tone signal is received which corresponds to the code set in one of the decoders of the airborne SELCAL unit, the corresponding light and chime circuits will be energized and the pilots alerted to the incoming call. Each preset decoder channel will respond to only one of 1438 possible tone codes.

The SELCAL chime is mounted in the overhead of the flight compartment adjacent to the interphone chime. Each chime is distinguished from the other by a different distinctive tone.

INTERPHONE SYSTEM

An interphone system is provided for communication between crew members during flight and between members of the ground crew when the airplane is being serviced. Interphone communication is possible from each flight crew station, the fore and aft cabin attendants' stations, the electronics compartment, the nose wheel well, and the principal points in and around the airplane (see Figure 19-1). The crew can converse with one another using the same microphones and audio equipment as used for radio transmitting and receiving. The cabin attendants' stations are equipped with handsets. The service interphone locations are provided with jackboxes for plugging in headsets and microphones or handset combinations.

The interphone system is connected to the emergency dc radio bus and is turned on whenever there is dc available at the airplane emergency bus. Communications are established at the pilot's, copilot's and flight engineer's stations and in the electronics compartment through the switches on the audio selector panel at each station. The audio is turned on by placing the INT toggle switch in the



Flight and Service Interphone Locations
Figure 19-1

ON position; the microphone is connected to the system by rotating the MIC SEL switch on the audio selector panel to the INT position. The remaining stations are connected automatically when the communications equipment is plugged in.

The interphone system can be operated in either a flight or service mode. The operating mode is selected by a 2-position ON-OFF toggle switch on the flight engineer's interphone microphone selector panel.

Interphone Flight Mode

In the flight mode, only the pilot's, copilot's, and flight engineer's stations and the outlets in the electronics and left aft lavatory compartments are connected in the system. The system is placed in the flight mode by placing the flight engineer's service interphone switch in the OFF position.

The electronics outlet is used when checking the electronics equipment either in flight or on the ground. This mode of operation is normally used in flight for communication among the members of the crew including the cabin attendants.

Interphone Service Mode

The service mode of operation enables the ground crew to communicate with each other from widely separated areas around and in the airplane. When the flight engineer's service interphone switch is placed in the ON position, all of the outlets in the system are interconnected. Communications are established at the service outlets by plugging in a microphone and headset or a handset unit.

Interphone Call System

A call system is provided to notify the flight crew and the cabin attendants when they are wanted on the interphone system. A pilot call combination light and switch is installed on each cabin attendant's panel. When the switch is depressed, a light will glow and a chime will sound in the cockpit indicating a call from one of the cabin attendants. A STEW call switch is installed in the pilots' overhead switch panel. When this switch is depressed, a light glows on each cabin attendant's panel and the cabin chimes sound, indicating a call from the flight compartment. These lights remain on until a cabin attendant removes a handset from its cradle at either control panel.

Handsets

Three handsets are provided; one in the flight compartment mounted on the aft side of the pilots' pedestal and one at each attendant's station in the cabin.

PUBLIC ADDRESS SYSTEM

The public address (PA) system enables the pilots and cabin attendants to address the passengers over the loudspeaker system. Magnetic tape recordings can be played through the public address system amplifier.

The PA system consists of a pilot's handset and control panel installation, forward and aft cabin handset and control panel installations, a priority relay, a high fidelity public address amplifier, and 22 loudspeakers installed in the hatracks in the cabin.

Pilots' voice communications have priority over music and voice announcements from either of the cabin control positions. The forward cabin control position has priority over music and voice announcements from the aft cabin control position. The aft cabin control position has priority over music for making voice announcements.

Public Address Control Panels

A PA control panel is provided for the pilots and each of the two cabin attendants. Each panel carries a vu-meter, volume controls, and call indicator lights. The panels are edge-lighted by lamps in the panels.

The pilots' control panel is installed on the left aft part of the control pedestal. This panel has a volume control switch for microphone audio only. An amber light PUSH PA switch connects the pilot's handset with the public address system. Communications from the pilot interrupt cabin music and voice communications being made from either of the cabin PA positions.

The forward cabin PA control panel is installed in the forward buffet area. This panel contains a MUSIC ON-OFF switch and the volume control for adjusting the level of music in the cabin. A voice volume control adjusts the audio level for voice announcements. PILOT CALL and HOSTESS CALL warning lights indicate when the position is being paged by another position. The call light will extinguish automatically when the handset is removed from the cradle.

The aft cabin PA control panel is installed in the aft buffet area. This control panel contains a volume control for handset voice announcements, PILOT CALL and HOSTESS CALL warning lights, and a PUSH PA control switch.

Public Address Handsets

A handset is installed at each control panel. The handset is normally connected to the interphone system. To talk over the PA system, the operator must depress the amber light PUSH PA switch on the PA control panel. The amber light will illuminate indicating that the handset is connected to the PA system. When the handset is restored to the holder, the handset is automatically returned to the interphone system.

PASSENGER CALL SYSTEM

Pushbutton switches and light assemblies installed near the passenger seats and in the lavatories enable the passengers to call the cabin attendants. A call by a passenger illuminates a light near the passenger area, call lights on the forward and aft cabin attendants' panels, and sounds a chime in the cabin ceiling near each buffet area.

Forward and aft cabin attendant's light panels are identical. Passenger call lights are grouped by rows of seats. Three lights indicate the general area in which the calling passenger is seated. Two lights indicate calls from the forward and aft lavatories. One light signals a call from the flight compartment and one light indicates a call from either cabin attendant's position to the other cabin attendant.

Each light is part of a switch assembly. Pulling the light extinguishes it and resets the system for another call. Cabin light control panels are interconnected so that resetting a light at one panel resets the other panel.

TAPE REPRODUCER SYSTEM

The tape reproducer system reproduces prerecorded music over the public address system for the entertainment of passengers. Operation of the tape reproducer and the listening level of music in the cabin are controlled remotely by an ON-OFF switch and a volume control on the public address control panel of the forward stewardess.

ELECTRICAL POWER SOURCES

The radio and communications equipment receives electrical power from four ac and four dc radio buses. AC power is supplied to the ac radio buses by the No. 3 essential ac bus and the pilot's ac essential bus of the electrical system of the airplane. DC power is provided for the dc radio buses by the 28-volt dc essential and 28-volt dc emergency buses. For detailed information, consult the WIRING DIAGRAM MANUAL.

20

MISCELLANEOUS

Section 20

MISCELLANEOUS SYSTEMS

TABLE OF CONTENTS

TITLE	PAGE
AIRPLANE LIGHTING SYSTEM	20-1
FLIGHT COMPARTMENT LIGHTS	20-1
Instrument Panel Lights	20-1
Instrument Panel Integral Lights	20-1
Pilot's Console Lights	20-2
Pilot's Overhead Switch Panel Lights	20-2
Pilot's Pedestal Lights	20-2
Flight Engineer's Panel Lights	20-2
Map Reading Lights	20-3
Standby Compass Light	20-3
Aldis Lamp Circuit	20-3
PASSENGER AND LOUNGE COMPARTMENT LIGHTING	20-3
Cabin General Lights	20-3
Lounge Overhead Lights	20-5
Aisle Lights	20-5
Passenger Reading Lights	20-5
Passenger Entrance Lights	20-5
Coat Closet Lights	20-5
Emergency Lighting System	20-5
Emergency Light Inertia Switches	20-7
Lavatory Lights	20-7
Buffet Area Lights	20-7
Cabin Attendant's Control Panel Lights	20-7
CARGO AND MISCELLANEOUS COMPARTMENT LIGHTING	20-7
Forward and Aft Cargo Compartment Lights	20-7
Electrical and Electronic Compartment Lights	20-10
Air Conditioning Compartment Lights	20-10
Wheel Well Lights	20-10
Hydraulic and Pneumatic Compartment Lights	20-10
Aft Fuselage Lights	20-10
Under-Wing Rear Spar Refuel Panel Lights	20-10
EXTERIOR LIGHTS	20-10
Landing Lights	20-10
Taxi Lights	20-12
Wing Illumination Lights	20-12
Position Lights	20-12
Anticollision Lights	20-12

TABLE OF CONTENTS (CONT)

TITLE	PAGE
AIRPLANE LIGHTING SYSTEMS POWER SOURCES	20-12
WATER SYSTEM	20-13
Pressurization	20-13
Heating	20-13
Water System Operation	20-13
WASTE SYSTEM	20-13

MISCELLANEOUS SYSTEMS

AIRPLANE LIGHTING SYSTEM

The airplane lighting system includes all the necessary lights for flight crew members, passengers and maintenance personnel, and airplane navigation requirements. The lighting systems are divided into four subsystems:

1. Flight compartment lights.
2. Passenger and lounge compartment lights.
3. Lavatory and buffet area lights.
4. Cargo and miscellaneous compartment lights.
5. Exterior airplane lights.

Alternating and direct current power sources are used for the various subsystems. Circuit breakers are provided to protect each individual light circuit from any electrical overload condition that may occur.

FLIGHT COMPARTMENT LIGHTS

Two dome lights are located in the flight compartment ceiling aft of the pilots' overhead switch panel. Each dome light has a red and white bulb. A three position COMPT LT, RED-OFF-WHITE, toggle switch on the overhead switch panel controls the lights. A pushbutton thunderstorm switch is mounted in each pilot's control wheel. When pressed, these switches will illuminate the white dome lights regardless of the position of the toggle switch. When the thunderstorm switches are released, the dome lights will return to the illumination mode called for by the toggle switch position.

Instrument Panel Lights

The pilots' instrument panel is illuminated by sixteen red and twelve white floodlights mounted in the glare shield above the instrument panel. Three high intensity white fluorescent floodlights, located in the glare shield provide additional instrument panel illumination. Eleven of the sixteen red lights illuminate the pilot's flight instrument panel and the engine instrument panel. The eleven red lights are controlled by a rheostat type PILOT PANEL RED LIGHT switch located on the pilot's console. The remaining five red lights illuminate the copilot's flight instrument panel and are controlled by a rheostat type CO-PILOT PANEL RED LIGHT switch located on the copilot's flight instrument panel. The twelve white floodlights in the glare shield are controlled by a rheostat type PANEL WHITE LIGHT switch located on the pilot's flight instrument panel. This switch also illuminates the high intensity white fluorescent floodlights when rotated to the full BRT position.

Instrument Panel Integral Lights

On some versions of the Model 990 airplane, all instruments on the pilots' flight instrument panel, the emergency airbrakes pressure gage on the pilot's auxiliary instrument panel, the engine instrument panel, and the copilot's flight instrument panel are equipped with integral red lights. These lights are divided into three subpanel circuits -- the pilot's flight instrument panel circuit (the emergency airbrake pressure gage light is electrically connected to the pilot's flight instrument panel lighting circuit), the engine instrument panel lighting circuit and the copilot's flight instrument panel lighting circuit. The integral instrument lights on the copilot's flight instrument panel and the engine instrument panel are controlled by a rheostat-type switch on the pilot's flight instrument panel. The integral instrument lights on the copilot's flight instrument panel are also controlled by a rheostat-type switch mounted on the copilot's panel. Each subpanel circuit is protected by its respective INST LTS fuse located on the pilot's and copilot's flight instrument panels above the light control switches.

Pilot's Console Lights

The pilot's and copilot's consoles are each illuminated by four red lights in the console and controlled by a rheostat type CONSOLE LTS switch on each console. Four red floodlights are provided above each console. These lights are controlled by a three position OFF-INCREASE switch mounted on each console panel. Two additional lights, for the oxygen regulators on each console, are connected with the integral red lights circuits.

Pilots' Overhead Switch Panel Lights

Each section of the pilots' overhead switch panel is internally illuminated by red lights mounted in the panel. A rheostat type OVERHEAD PANEL LIGHTS switch on the overhead switch panel controls the light intensity.

Pilots' Pedestal Lights

Each section of the pilots' pedestal is internally illuminated by red lights mounted in the panel. The lights are controlled by a rheostat type PEDESTAL LIGHTS switch located on the pilots' pedestal. A red semispotlight is located in the flight compartment ceiling at station 300 to provide additional pedestal and overhead switch panel illumination. An OFF-INCREASE switch, PEDESTAL FLOOD LT, on the overhead switch panel controls the floodlight.

Flight Engineer's Panel Lights

Each section of the flight engineer's instrument panel is internally illuminated by red lights mounted in the panel. A rheostat type FLIGHT ENGINEERS PANEL lights switch located on the flight engineer's panel controls the light intensity. In addition, the flight engineer's panel has two red and two white floodlights mounted in the flight compartment ceiling. These lights are controlled by individual rheostat type ENGINEERS FLOODLIGHT switches located on the aft side of the flight engineer's panel. Lights are also provided to illuminate the circuit breaker panels and the limiter and ac panels under the flight engineer's table.

A utility light is also provided on the flight engineer's panel. An integral ON-OFF switch is included on the light assembly.

Map Reading Lights

Map reading lights are provided for the pilot and copilot. The pilot's map reading light is located to the left of the overhead switch panel and the copilot's light is to the right of the switch panel. Each light has a built-in OFF-INCREASE switch and light intensity control. Each light can be adjusted from a 14-inch diameter spot to a 2-inch diameter spot when directed on a map in the crew members lap.

Standby Compass Light

A red light is provided for the magnetic standby compass on the glare shield. The rheostat type STANDBY COMPASS LIGHT switch is located on the copilot's flight instrument panel. A spare lamp and fuse stowage box is mounted on the flight compartment aft bulkhead. Spare lamps, two of each type used for the flight compartment lights, are stored in a shock-absorbing, foamed plastic retainer inside the box (see Figure 20-1).

Aldis Lamp Circuit

The aldis lamp circuit consists of a circuit breaker and lamp power receptacle. Closing the ALDIS LAMP OUTLET circuit breaker sends 28-volt dc power (emergency bus) to a lamp receptacle located aft of the pilot's seat.

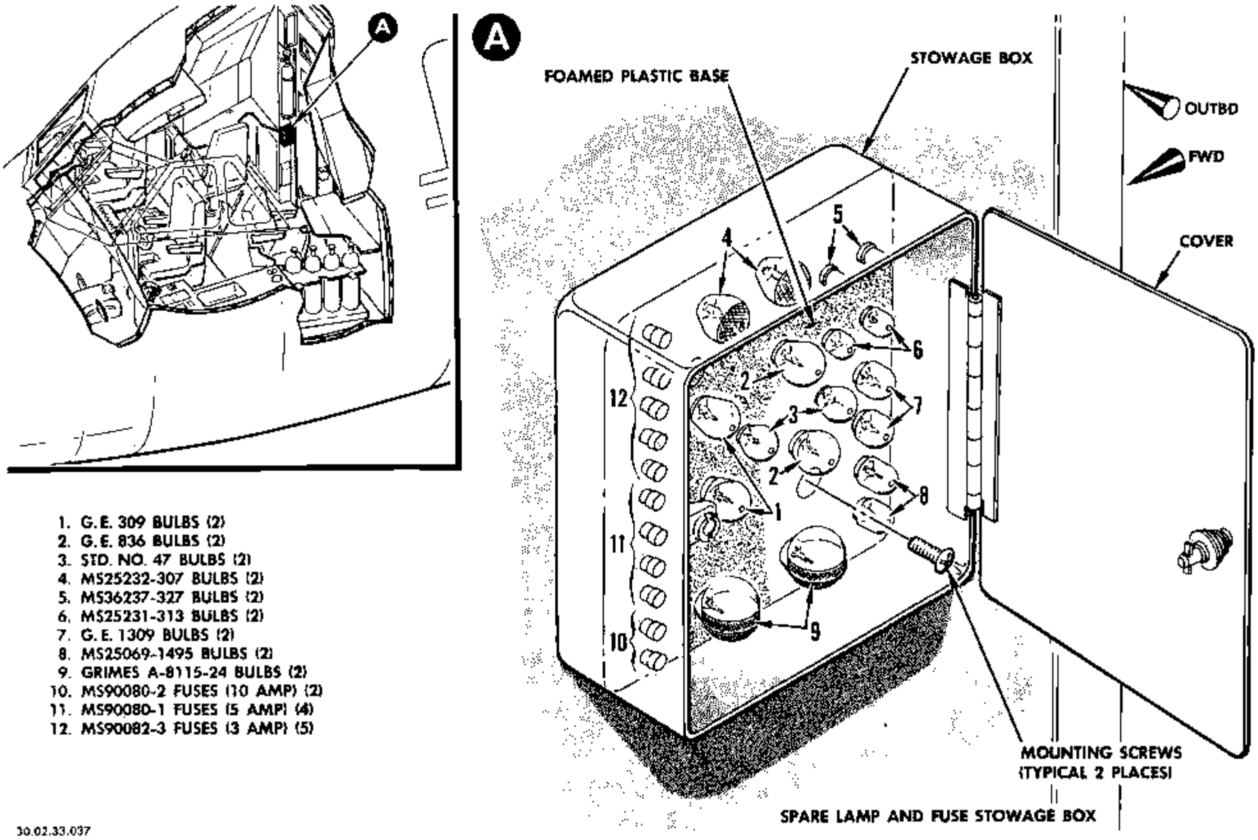
PASSENGER AND LOUNGE COMPARTMENT LIGHTING

The passenger and lounge compartment lighting includes the cabin general lights, lounge overhead and table lights, aisle lights, passenger reading lights, passenger entrance lights, coat closet lights, emergency lighting system, and lavatory and buffet lights. Some variation in lighting between different versions is encountered but basic lighting is as noted in the following descriptions.

Cabin General Lights

Fluorescent lights located in the cabin overhead and above the cabin windows provide general illumination in the passenger cabin. The overhead lighting is indirect and consists of twenty-eight, 13-watt fluorescent lights mounted on each side of the ceiling. The window lighting consists of twenty-two lights installed on each side of the passenger cabin, one above each pair of windows. All of the windows lights, except the light directly aft of each emergency exit, are 13-watt units. The two fluorescent lights directly aft of the left and right emergency exits are 8-watt units.

The general cabin light intensity is controlled by two-level switching-type ballasts. Light intensity is controlled from either of the two stewardess's panels by a three-position DIM-NEUTRAL-BRIGHT toggle switch, spring-loaded to the NEUTRAL position. The toggle switch is held in the DIM or BRIGHT position until the light intensity is as desired. When the switch is in the DIM position, a control switch actuates which turns the cabin lights off. The general cabin lights can be extinguished from the flight compartment by means of an



1. G.E. 309 BULBS (2)
2. G.E. 836 BULBS (2)
3. STD. NO. 47 BULBS (2)
4. MS25232-307 BULBS (2)
5. MS36237-327 BULBS (2)
6. MS25231-313 BULBS (2)
7. G.E. 1309 BULBS (2)
8. MS25069-1495 BULBS (2)
9. GRIMES A-8115-24 BULBS (2)
10. MS90080-2 FUSES (10 AMP) (2)
11. MS90080-1 FUSES (5 AMP) (4)
12. MS90082-3 FUSES (3 AMP) (5)

30.02.33.037

Spare Lamp Stowage Box
Figure 20-1

override switch located on the pilots' overhead switch panel. The cabin window lights can be separately controlled, if desired, by an ON-OFF toggle switch on each stewardess's control panel. These switches are wired in a three-way circuit to provide full operation from either panel (see Figure 20-2).

Lounge Overhead Lights

Lounge lighting is provided by three overhead fluorescent lights. The lounge lights are controlled by a circuit breaker type switch located on the forward stewardess's control panel.

Aisle Lights

Nine aisle lights are installed in the cabin ceiling to illuminate the aisle between the passenger cabin seats. The aisle lights are controlled by two ON-OFF-DIM toggle switches, one on each stewardess's control panel. These switches are wired in a three-way circuit to provide full operation from either panel.

Passenger Reading Lights

The passenger reading lights are located in the convenience pods, one above each passenger seat. Pushbutton switches (push to illuminate, push again to extinguish) are located adjacent to each light. An override switch, PASSENGER READING LIGHTS in the pilots' overhead switch panel permits the pilots to extinguish the passenger reading lights. Five READING LIGHTS switches on the aft stewardess's panel must be ON to arm the individual control switches.

Passenger Entrance Lights

Two threshold lights and two overhead lights are located at each main passenger entrance. The lights are controlled by a switch in the corresponding main entrance door sill and are extinguished when the doors are closed and locked. This switch also controls the door open warning light in the flight compartment. The lights illuminate when the main entrance doors are opened. An override ENTRANCE, ON-OFF, switch is provided on the respective stewardess's control panel. Also located at each main entrance is a RAMP RECEPTACLE which is supplied with airplane electrical power.

Coat Closet Lights

The forward and aft coat closets are each provided with one incandescent light controlled by a circuit breaker type ON-OFF switch located on the side of each coat closet.

Emergency Lighting System

The emergency lighting system includes the lights at the forward and aft main entrance and service areas, and the left and right passenger cabin emergency exits, and the white dome lights in the flight compartment. These lights may be energized by either the 28-volt dc emergency bus or the battery bus. The EMERGENCY EXIT LIGHT switch in the pilots' overhead switch panel is a

To Be Furnished

Typical Stewardess's Control Panel
Figure 20-2

three-position, ON-ARMED-OFF toggle switch. When the flight compartment switch is placed in the ON and ARMED position, all emergency lights will illuminate. When the flight compartment switch is placed in the OFF position, electrical power from the battery bus is interrupted. In the event of an electrical power failure on the dc emergency bus, a transfer relay deenergizes and transfers battery bus electrical power to the emergency lights. The lights may be extinguished by placing the flight compartment switch in the OFF position (see Figure 20-3).

Emergency Light Inertia Switches

If a shock force of 1.5 G's or more is experienced, two inertia switches will close and energize the emergency lighting circuit, illuminating all emergency lights from the battery bus.

Lavatory Lights

One 13-watt and four 8-watt fluorescent lights in each forward lavatory are controlled by a switch on the forward stewardess's control panel. A lavatory door switch automatically extinguishes all lavatory lights except one over the toilet when the door is opened. Both aft lavatories have two 13-watt and two 8-watt fluorescent lights each, all controlled by a switch on the aft stewardess's control panel. Each aft lavatory door switch automatically extinguishes three of the lights when the respective door is opened.

Buffet Area Lights

Four lights are located in the ceiling of both the forward and aft buffet areas. Switches of the stewardess's panel in their respective areas control the lights.

Cabin Attendant's Control Panel Lights

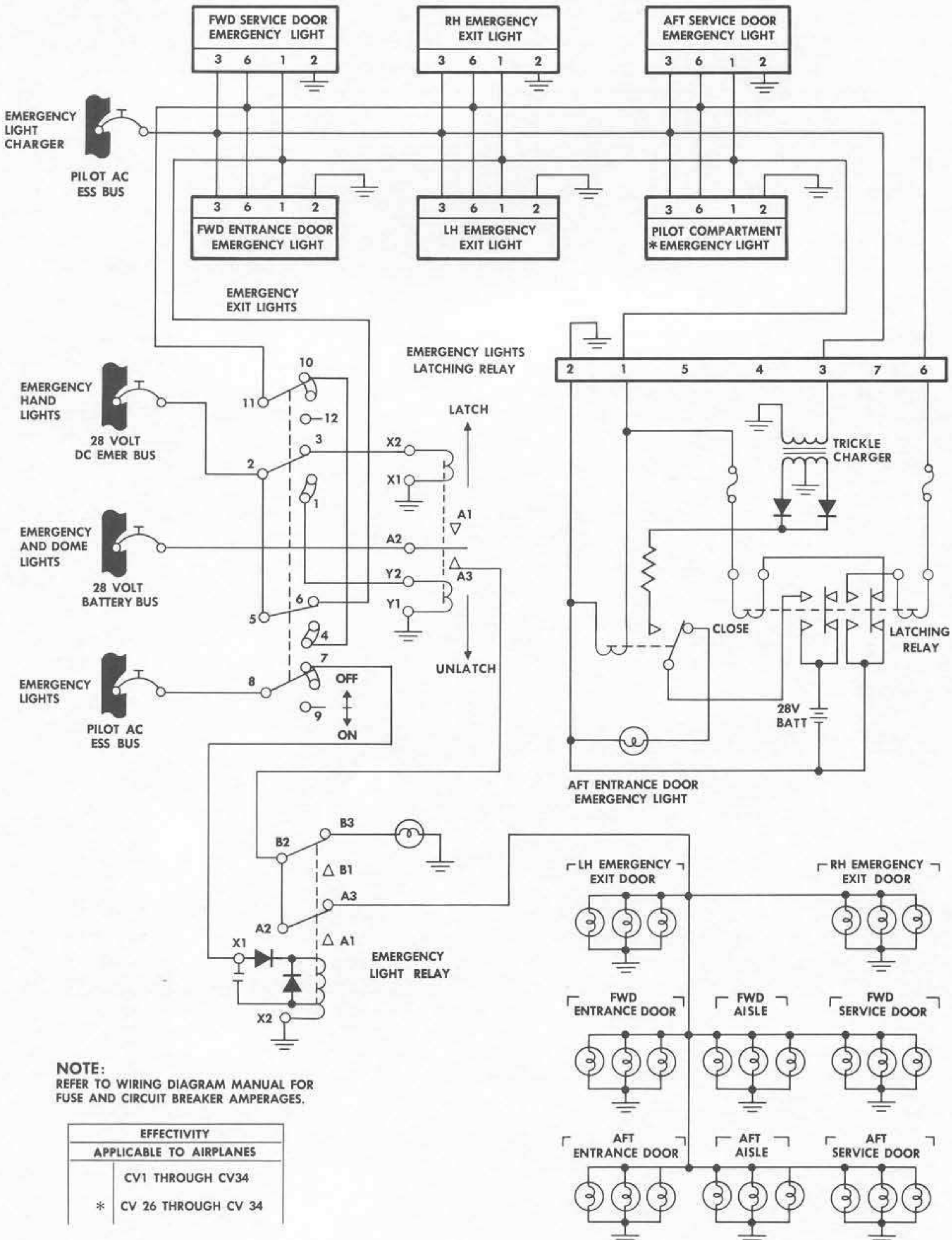
The forward stewardess's control panel is illuminated by nine integral red lights. A control switch, PANEL, ON-OFF, is located on each panel.

CARGO AND MISCELLANEOUS COMPARTMENT LIGHTING

The cargo and miscellaneous compartment lighting includes the lights in the forward and aft cargo compartments, the electrical and electronic compartments, the air conditioning compartment, main and nose gear wheel wells and the hydraulic and pneumatic compartment. Also included are the lighting circuits for the aft fuselage interior and the under-wing rear spar refueling panels.

Forward and Aft Cargo Compartment Lights

Three dome lights and one threshold light are located in each forward and aft cargo compartment. The lights are controlled by automatic door switches, which also actuates the door open warning lights in the flight compartment.

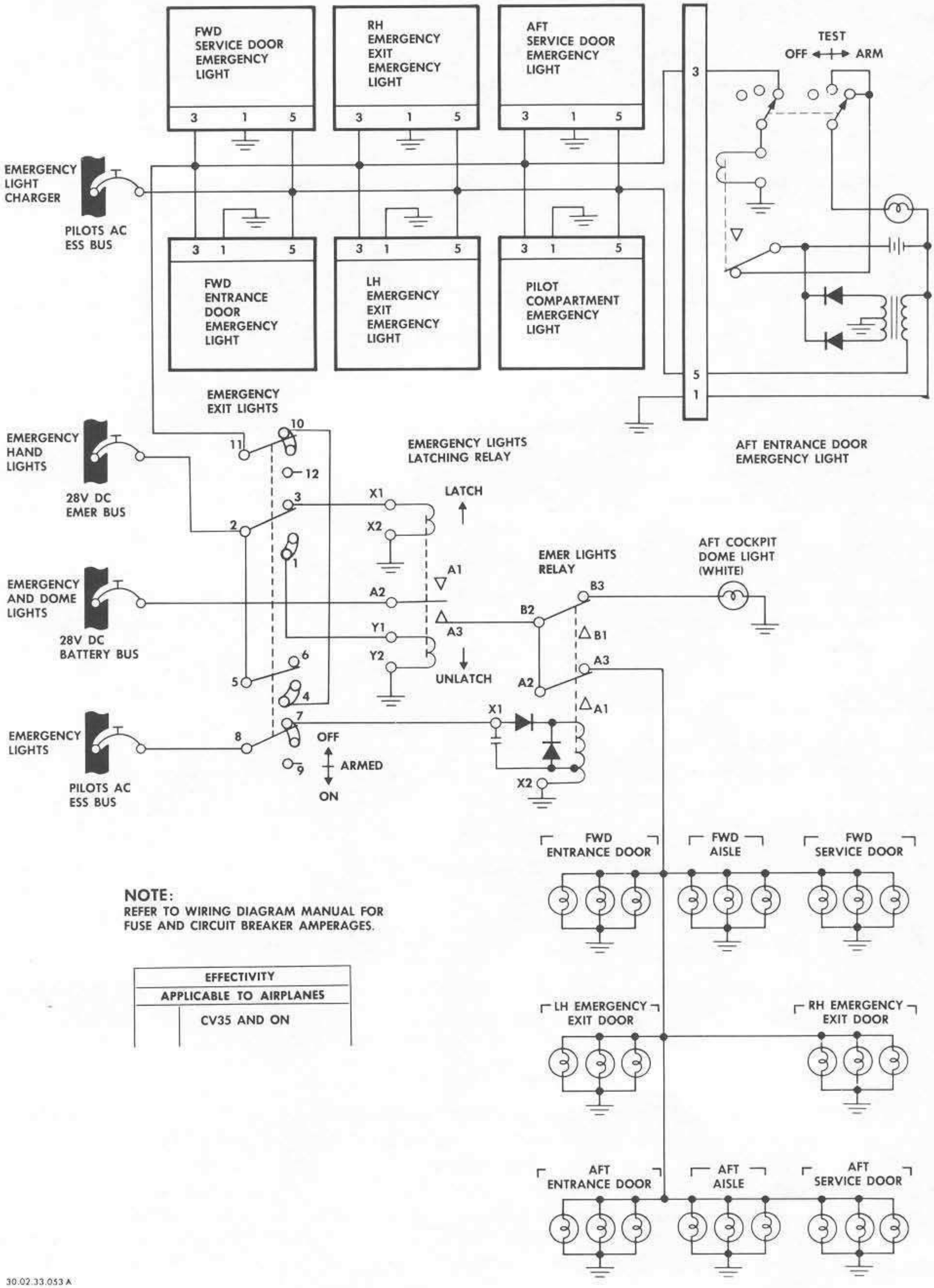


NOTE:
REFER TO WIRING DIAGRAM MANUAL FOR FUSE AND CIRCUIT BREAKER AMPERAGES.

EFFECTIVITY	
APPLICABLE TO AIRPLANES	
	CV1 THROUGH CV34
*	CV 26 THROUGH CV 34

10.02.33.094

Emergency Lighting System Schematic
Figure 20-3 (Sheet 1 of 2)



NOTE:
REFER TO WIRING DIAGRAM MANUAL FOR FUSE AND CIRCUIT BREAKER AMPERAGES.

EFFECTIVITY	
APPLICABLE TO AIRPLANES	
	CV35 AND ON

30.02.33.053 A

Emergency Lighting System Schematic
Figure 20-3 (Sheet 2 of 2)

Electrical and Electronic Compartment Lights

Four lights provide illumination in the electrical and electronic compartment. The lights are controlled by the access door switch or an ON-OFF toggle switch adjacent to the inner surface of the access door. (The toggle switch is used when gaining entry to the compartment through the flight compartment floor door.) The electrical and electronic compartment door is connected into the door open warning light.

Air Conditioning Compartment Lights

Two lights are located in each of the forward and aft air conditioning compartments. The lights are controlled by automatic door switches which also actuate the door open warning light.

Wheel Well Lights

One light is located in each wheel well. These lights are connected to the position lights circuit and are armed through the landing gear door "up" and "locked" warning switches. All of the wheel well lights may be controlled by a WHEEL LTS, ON-OFF switch, located on the pilots' overhead switch panel.

Hydraulic and Pneumatic Compartment Light

One light, automatically controlled by the access door, illuminates the hydraulic and pneumatic compartment. This door is connected into the door open warning light.

Aft Fuselage Lights

Two lights provide illumination of the area aft of the aft pressure bulkhead. Both lights are controlled by automatic door open switches at either door.

Under-Wing Rear Spar Refuel Panel Lights

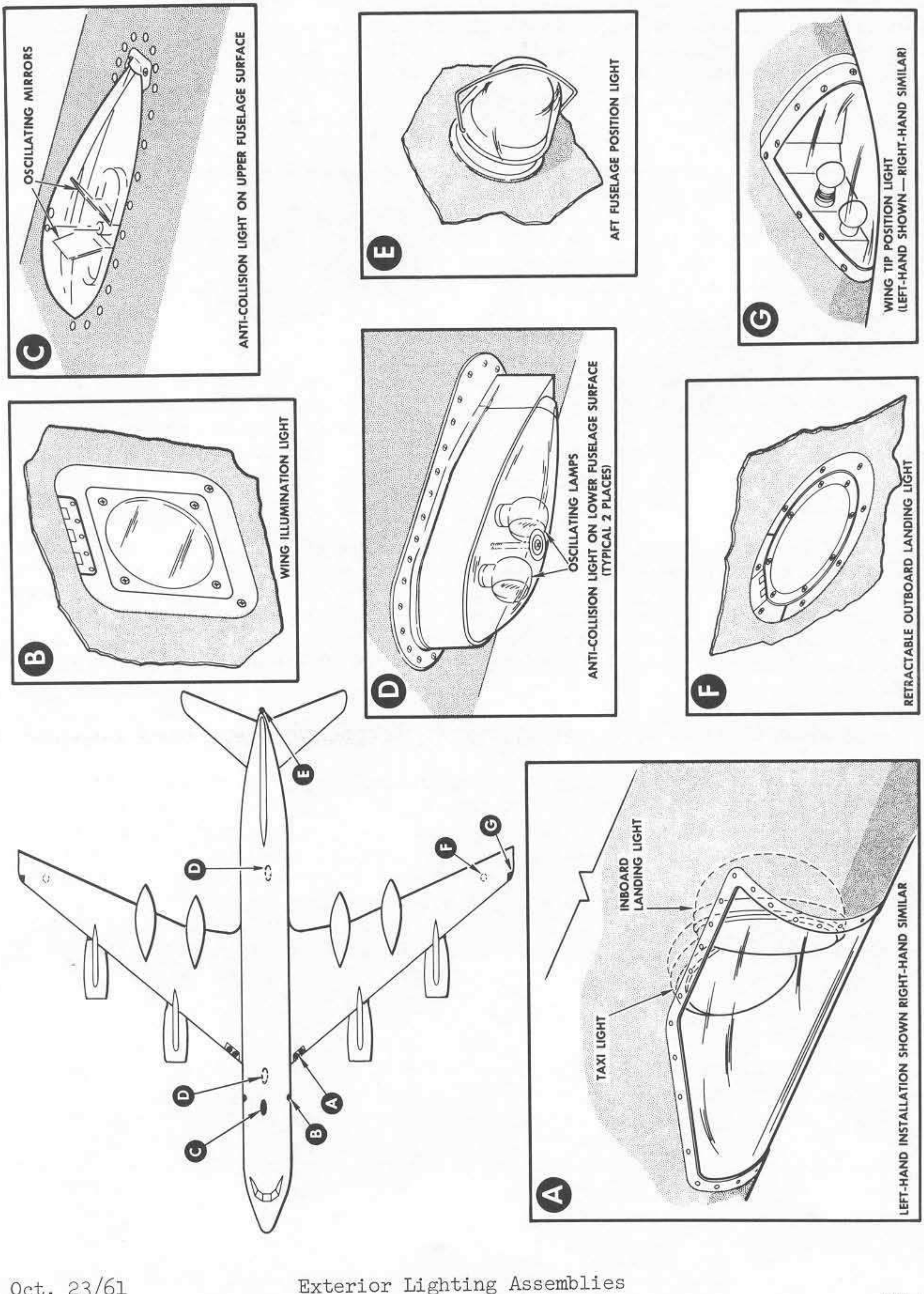
Internal lights illuminate each refuel panel. The lights are controlled by door actuated microswitches.

EXTERIOR LIGHTS

The exterior lights include the landing lights, taxi lights, wing illumination lights, position lights, and the anti-collision lights (see Figure 20-4).

Landing Lights

Two inboard and two outboard landing lights are provided. The fixed position inboard landing lights are located in the leading edge of each wing, just outboard of the fuselage. The outboard landing lights are located in the lower outboard surface of each wing at the rear spar (wing station 744). These lights are retractable units that flush with the wing when not in use. The outboard landing lights use 600-watt lamps and each light is controlled by a separate three-position EXTEND-OFF-RETRACT toggle switch. In the EXTEND



Exterior Lighting Assemblies
Figure 20-4

position, the light is extended and illuminated. The light can be held in an extended but extinguished position by placing the switch in the OFF position after extending the light. Two amber warning lights indicate when the outboard landing lights are extended. The inboard landing lights use 600-watt lamps and each light is controlled by a separate ON-OFF toggle switch. All landing light switches and warning lights are located on the pilots' overhead switch panel.

Taxi Lights

For taxiing purposes, two 250-watt lamps are located in the leading edge of each wing. Each light is controlled by a separate ON-OFF switch located on the pilots' overhead switch panel.

Wing Illumination Lights

A 100-watt wing illumination light is installed on each side of the fuselage forward of the wing at fuselage station 574. These lights are used to detect ice formation on the wing leading edge. A WING ICE LIGHTS "ON-OFF" switch is located on the pilots' overhead switch panel.

Position Lights

The position lights consist of a green light on the right wing tip, a red light on the left wing tip and a white light on the aft tip of the fuselage. Each wing position light incorporates two light bulbs. An ON-OFF toggle switch located on the pilots' overhead switch panel illuminates the position lights. The position light control switch also controls the dimming of the master warning system lights. A flasher unit is not included in the position light circuitry.

Anticollision Lights

Three 40-watt anti-collision lights are provided, one on the upper surface of the fuselage at station 499 and two on the bottom surface of the fuselage at stations 594 and 1316. The upper light consists of two horizontally mounted lamps directed into two oscillating diagonal mirrors. The two lower lights each consists of two 40-watt lamps with integral reflectors to direct the light in beams. Each lamp assembly rotates in 180 degree arcs about the centerline of the airplane. One lamp faces aft and the other forward. Two ON-OFF toggle switches, ANTI-COL UPPER and ANTI-COL LOWER, located on the pilots' overhead switch panel control the operation of the three anti-collision lights.

AIRPLANE LIGHTING SYSTEMS POWER SOURCES

For detailed information regarding power sources for the lighting systems, consult the WIRING DIAGRAM MANUAL.

WATER SYSTEM

The water system consists of a fiberglass water supply tank, a ground service panel which is used in connection with maintenance and servicing of the system, and numerous valves and interconnecting tubing to provide the means of delivering water to the proper places.

Pressurization

The pressurization circuit consists of a motor-driven air pump, one pump control relay, a manual control switch, a low-pressure warning switch, a low-pressure pump control switch and warning light (see Figures 20-5 and 20-6). Some versions of the Convair 990 include an auxiliary pump and related circuitry.

Heating

The water heater circuit consists of three water heaters, two control switches, six circuit breakers and interconnecting wiring (see Figure 20-7).

Water System Operation

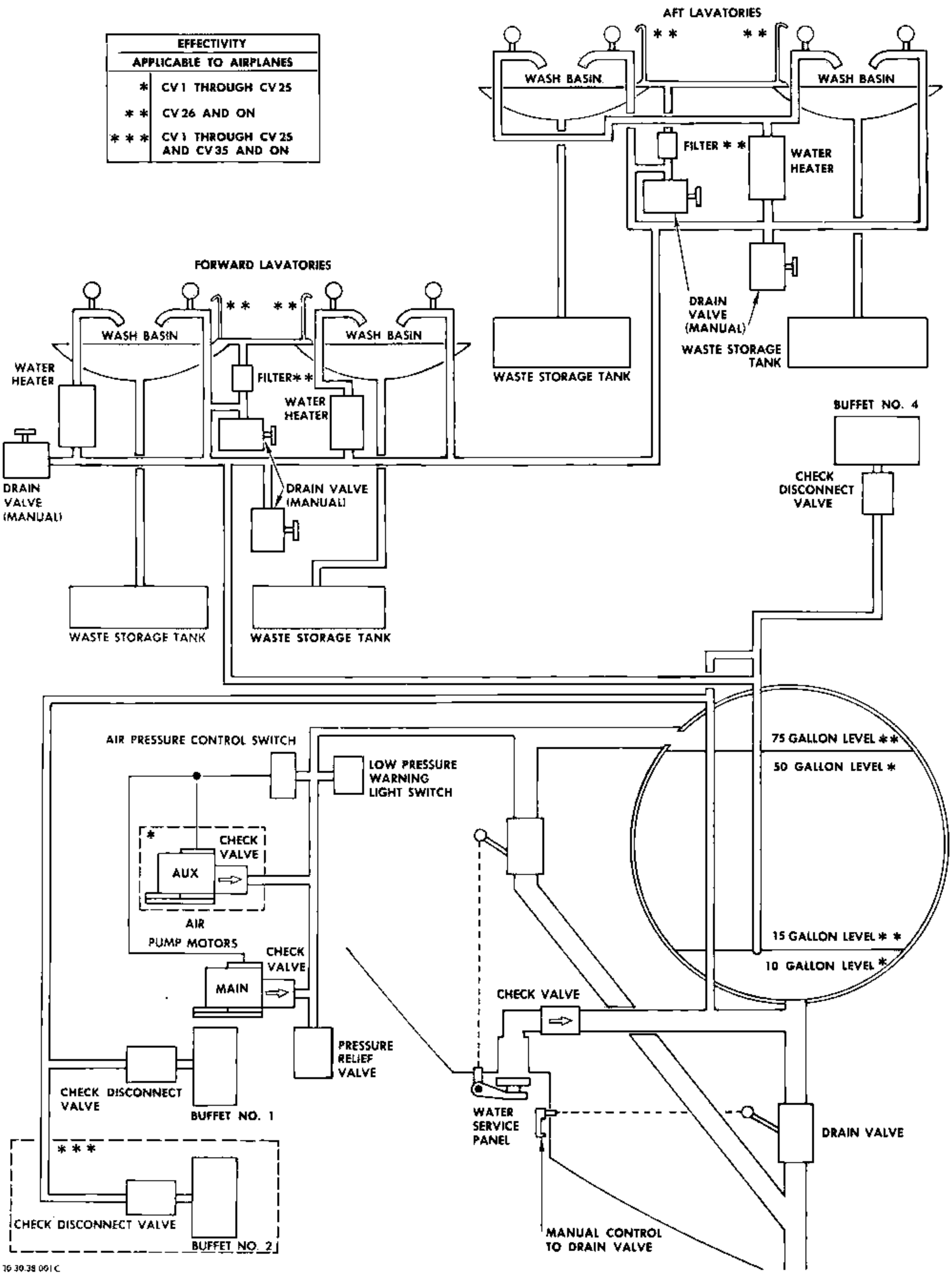
The water for the system is contained under pressure in the supply tank and is routed to the lavatories and buffets on demand when the spigots are opened. Once the pump control switch, located on the flight engineer's panel, is turned "ON", the pressurization circuit automatically maintains system pressure through the operation of the low-pressure pump control switch. The low-pressure warning switch monitors the pressurization circuit by illuminating the warning light when the system pressure is too low. As the system water supply is depleted, the air pressure decreases until it reaches a certain point. The decreased pressure allows the low-pressure pump control switch to complete the control circuit for turning on the air pump motor and rebuilding the system pressure.

Hot water for use in the lavatories is heated by the three water heaters installed in the water supply lines. Once the heater control switches on the stewardess's control panels are placed in the ON position, the water temperature is automatically maintained by one bi-metallic switch within the units.

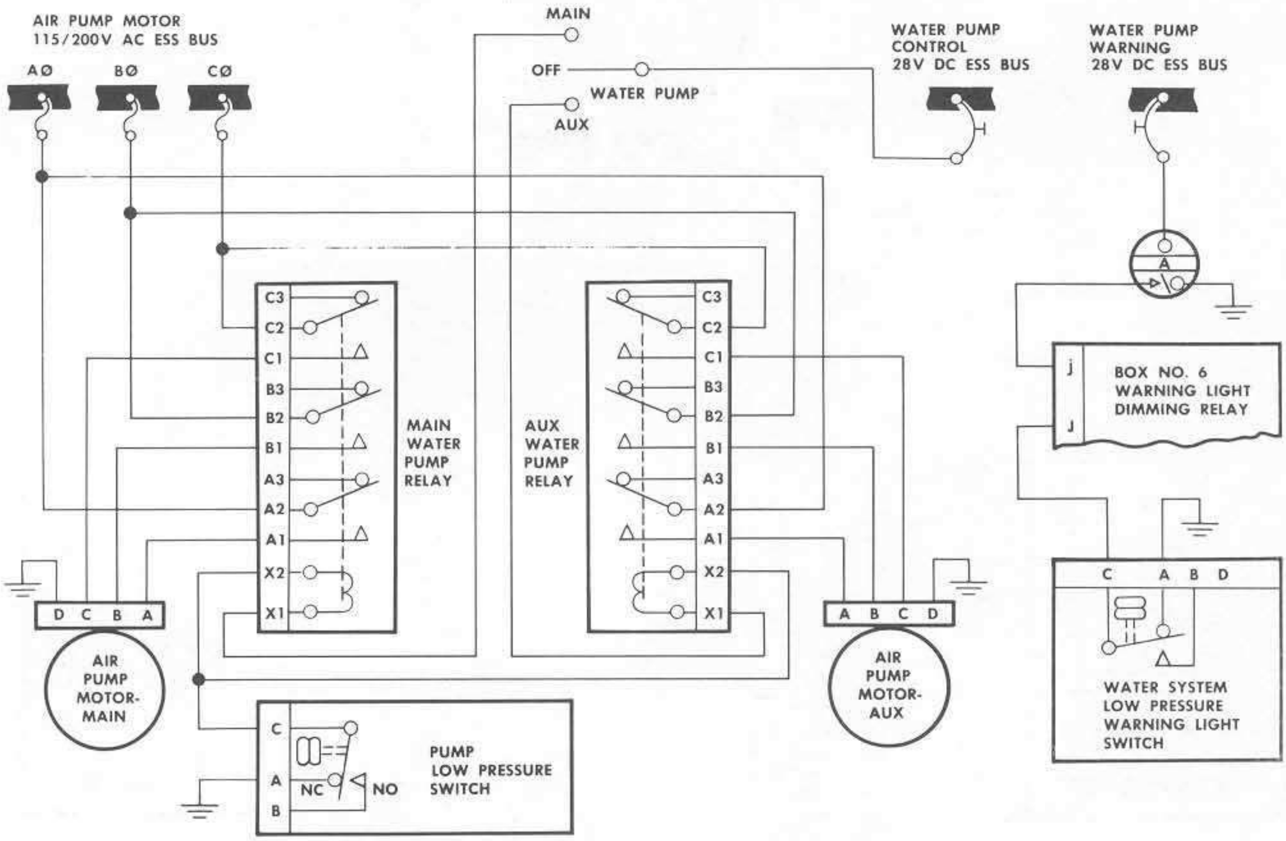
WASTE SYSTEM

The waste system consists of four waste storage tanks, two service panels and the associated plumbing from the lavatory wash basins (see Figure 20-8). The tanks are located below the toilet in each lavatory and are used as storage containers for all toilet waste and for waste water from the wash basins. Waste from the toilet goes directly to the storage tank during normal use, and flushing rings that are integral with the top of the bowl and tank are utilized to wash waste from the bowl and tank during ground servicing. Waste water from the wash basins is drained into the storage tank when the sink stopper is manually operated. Two external ground service panels, one for the aft lavatories and one for the forward lavatory, provide draining and flushing facilities for ground servicing.

EFFECTIVITY	
APPLICABLE TO AIRPLANES	
*	CV1 THROUGH CV25
**	CV26 AND ON
***	CV1 THROUGH CV25 AND CV35 AND ON



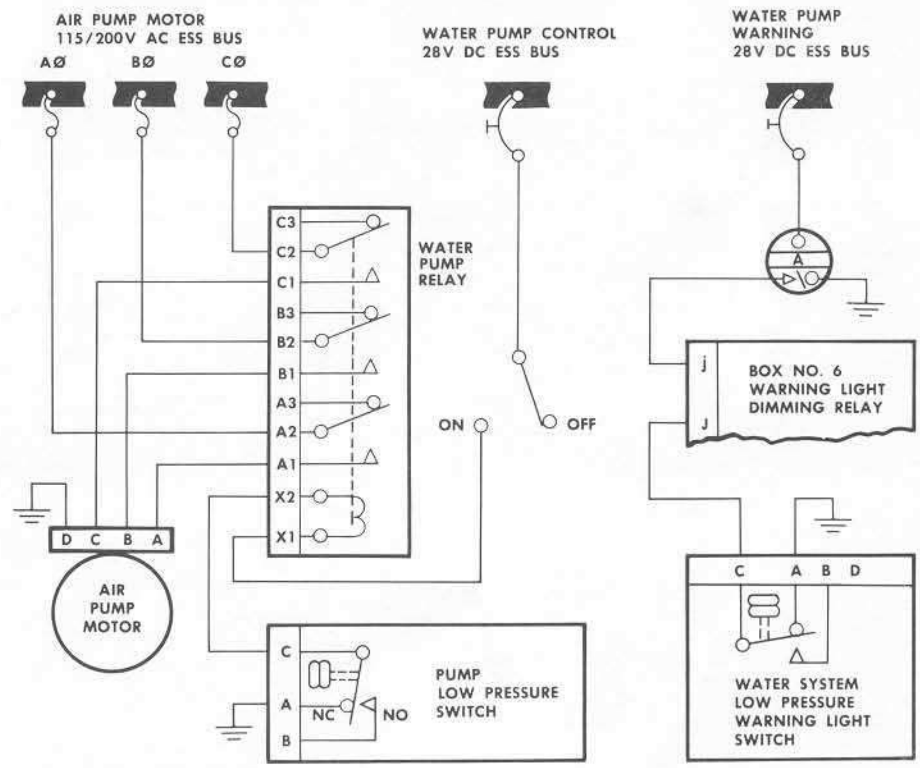
Water System Schematic
Figure 20-5



* WATER PRESSURIZATION CIRCUIT SCHEMATIC

NOTE:
REFER TO WIRING DIAGRAM MANUAL FOR FUSE AND CIRCUIT BREAKER AMPERAGES

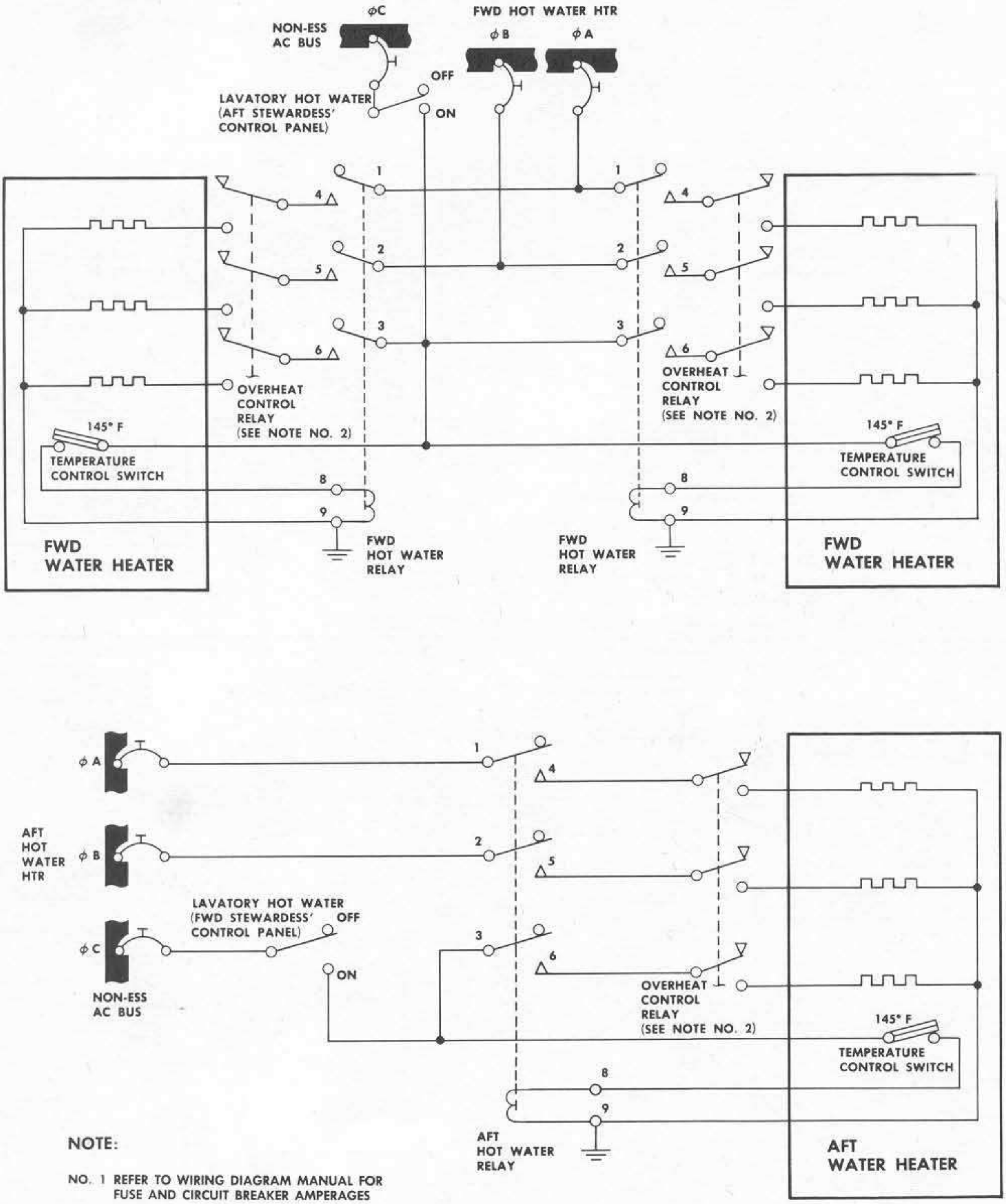
EFFECTIVITY	
APPLICABLE TO AIRPLANES	
*	CV 1 THROUGH CV 25
* *	CV 26 AND ON



30.30.38.005B

* * WATER PRESSURIZATION CIRCUIT SCHEMATIC

Water Pressurization System Circuit
Figure 20-6



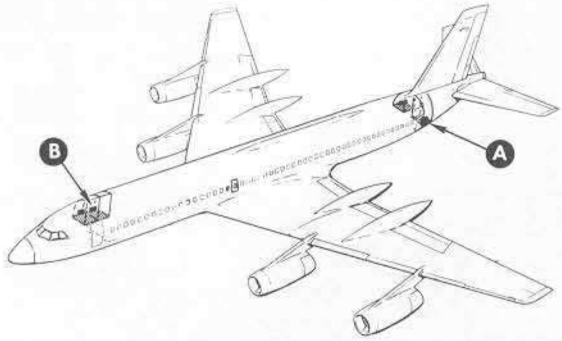
NOTE:

NO. 1 REFER TO WIRING DIAGRAM MANUAL FOR FUSE AND CIRCUIT BREAKER AMPERAGES

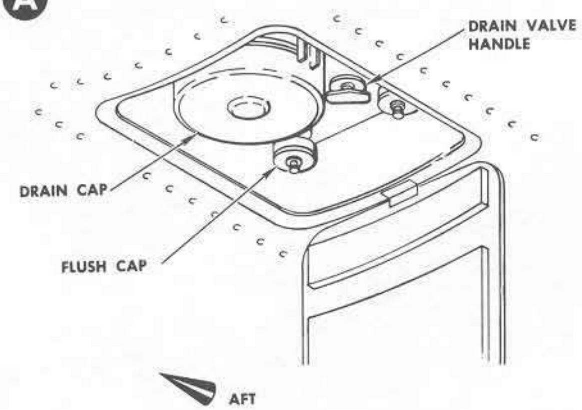
NO. 2 OVERHEAT CONTROL RELAY ACTUATED BY FUSIBLE LINK

30.02.38.011A

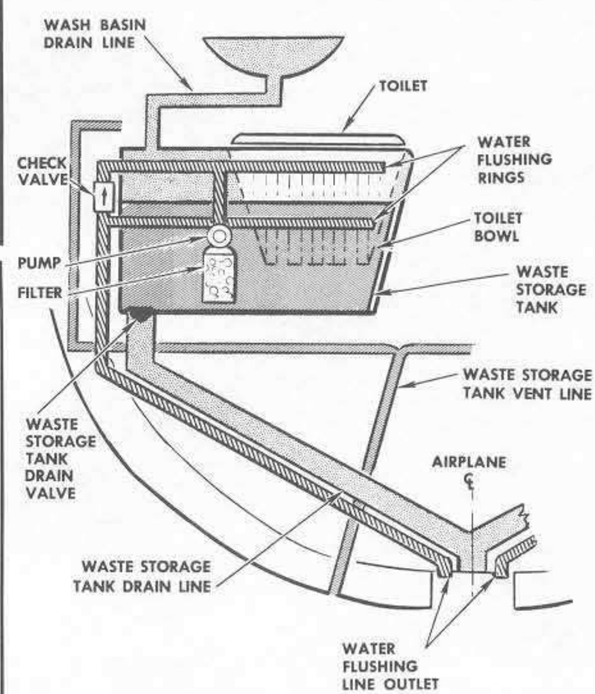
Water Heater Circuit
Figure 20-7



A



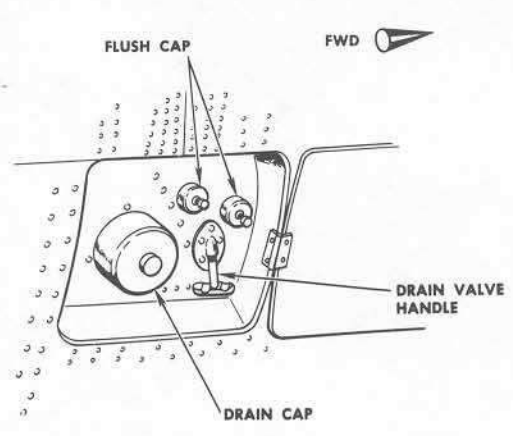
AFT LAVATORY (TYPICAL)



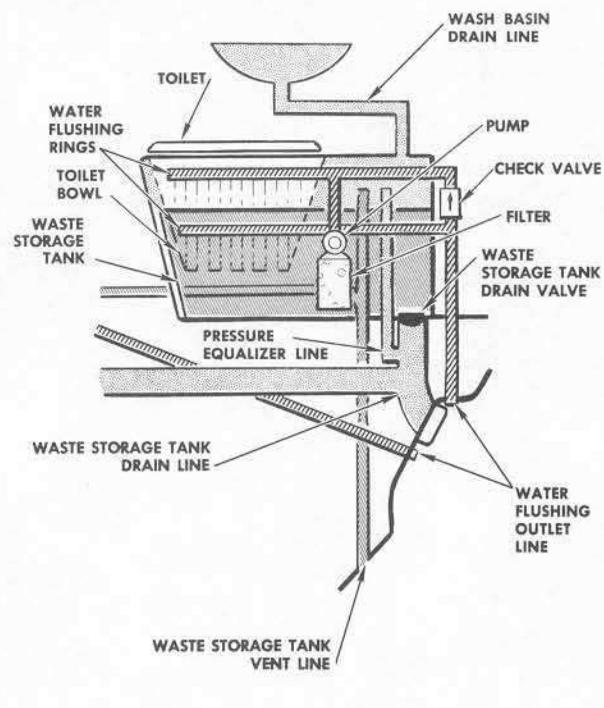
B

CAUTION: DO NOT PULL DRAIN HANDLE UNTIL DRAIN HOSE HAS BEEN CONNECTED

1. REMOVE DRAIN AND FLUSH CAPS.
2. CONNECT DRAIN AND FLUSH HOSES.
3. OPEN DRAIN VALVE BY DEPRESSING BUTTON IN CENTER OF HANDLE AND PULL (PULL HANDLE ALL THE WAY DOWN BEFORE RELEASING BUTTON.)
4. FLUSH TANK.
5. CLOSE DRAIN VALVE BY DEPRESSING BUTTON IN CENTER OF HANDLE AND PUSHING HANDLE ALL THE WAY UP.
6. DISCONNECT HOSES AND REPLACE DRAIN AND FLUSH CAPS.



FORWARD LAVATORY (TYPICAL)



30.30.38.006B

Waste System Schematic - Typical
Figure 20-8

Airplanes CVL through CV34 are equipped with flushing type toilets. The toilets recirculate the waste tank water and disinfectant solution by means of a motor driven mechanical filter and pump assembly. The flushing mechanism is actuated by a microswitch which is cam-linked to a flushing handle. The flushing handle is located in the toilet shroud.