LOCKHEED SR-71

SUPERSONIC/HYPERSONIC RESEARCH FACILITY

Researcher's Handbook

VOLUME II TECHNICAL DESCRIPTION

NASA'S EXTERNAL BURN EXPERIMENT

INTRODUCTION

Volume I of the Researcher's Handbook, an Executive Summary, provides an overview of the SR-71 as a supersonic/hypersonic research facility. Volume II compliments Volume I by providing detailed information on the aircraft, aircraft operations and performance, aircraft systems, and payload capabilities. Information is also provided regarding previous payloads, experiments, and studies connected with the SR-71 class of aircraft, together with a description of the SR-71's extended flight capabilities.

The SR-71 is, for sustained flight, the highest flying, fastest aircraft in the world today. No other aircraft comes even remotely close to matching its performance for cruise flight at altitudes of 85,000 feet, at speeds of Mach 3 plus. The SR-71 and its predecessor, the A-12, have unmatched records in their roles as reconnaissance aircraft. Likewise, the YF-12A achieved comparable performance as a prototype interceptor and as a test vehicle for NASA.

Now operated by NASA, the SR-71, in its new role, functions as a unique tool for basic and applied research and for testing and evaluation of materials and equipment under actual flight conditions. The aircraft is a mature, proven platform that, during over 25 years of service, has logged over 53,000 hours of flight, with over 11,000 hours at speeds above Mach 3. Under standard flight conditions, dynamic pressure "*q*" loads range up to 686 lbs/ft^2, with nominal Reynolds numbers ranging from 0.69 to 3.3 million per foot.

The SR-71 family of aircraft has demonstrated the ability to launch and guide air-to-air missiles of the Phoenix class; transport to 80,000 feet and launch at Mach 3 a 12,000 pound, ram-jet drone; provide an extended high-heat environment for composite material testing at up to 600 degrees F (316 degrees C) for periods of over an hour; and conduct other tests requiring high-altitude, high-Mach profiles. Various studies have been accomplished using the aircraft as a research or test platform; such as: air particulate sampling, exploration of Mach growth, and testing of supersonic inlets. To assist with RDT & E activities, the SR-71 has an on-board Mission Recorder System (MRS) with hundreds of data-monitor input points available for both ground and in-flight recording.

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GLOSSARY

The following abbreviations, acronyms, and terms are used in this handbook.

A1 & A2	Alpha Pneumatic Pressures
A/B	Afterburner
AC	Air conditioning
AD	Aerodynamic Disturbance
ADI	Attitude Director Indicator
ADS	Air Data System △1 Accessory Drive System
AFCS	Automatic Flight Control System △1
AICS	Air Inlet Control System △1
Alpha	Angle of Attack
ANS	Astroinertial Navigation System
AOA	Angle of Attack
A/P	Autopilot △1
APW	Automatic Pitch Warning (System) △1
ASARS	Advanced Synthetic Aperture Radar System
ATT	Attitude
B1 & B2	Beta Pneumatic Pressures
BDHI	Bearing, Distance, Heading Indicator
Beta	Angle of Side Slip
BIT	Built-In Test
BL	Butt Line
CCT	Computer Compatible Tape
CEP	Circular Error of Probability
CFD	Computer Fluid Dynamics
CIP	Compressor Inlet Pressure
CIS	Chemical Ignition System
CIT	Compressor Inlet Temperature
CSC	Control Stick Command
CSD	Constant Speed Drive
CP	Control Point
DAFICS	Digital Automatic Flight and Inlet Control System △1
DCU	Digital Computer Unit △1
DP	Destination Point
DPRT	Duct Pressure Ratio Transducer
ECS	Environmental Control System
EGT	Exhaust Gas Temperature
ESS	Essential (electrical term)
FP	Fix Point
FS	Fuselage Station
GTOW	Gross Takeoff Weight
HSI	Horizontal Situation Indicator
IAW	In Accordance With
IGV	Inlet Guide Vanes
INS	Inertial Navigation System
KEAS	Knots Equivalent Airspeed
KIAS	Knots Indicated Airspeed
LN_2	Liquid Nitrogen
LOX	Liquid Oxygen
LRU	Line Replaceable Unit
LVDT	Linear Variable Differential Transformer
MGFU	Mobile Ground Formatter Unit
MRS	Mission Recorder System
NASP	National Aero-Space Plane
NS	Nacelle Station
N MI	Nautical Miles
NWS	Nose Wheel Steering
OBC	Optical Bar Camera
PCM	Pulse Code Modulation
PTA	Pressure Transducer Assembly △1
PVD	Peripheral Vision Display
RHSO	Right-Hand Shutoff (fuel control switches)
RS	Radar (nose section) Station
RSO	Research Systems Operator △2

NOTE △1 Part of DAFICS △2 A flight test engineer

GLOSSARY (CONT)

SAS	Stability Augmentation System 1
S/O	Shutoff
SRA	Special Repair Activity
SS	Spike Station
TAS	True Airspeed
TEB	Triethylborane
TCU	Tape Copy Unit
V/H	Velocity divided by Height
WL	Water Line
WS	Wing Station
ZFW	Zero Fuel Weight

SECTION I

THE AIRCRAFT

1.1 GENERAL.

The SR-71, built by Lockheed, is a long-range, two-place aircraft, powered by two Pratt and Whitney J58 turbojet engines. (See figures 1-1 and 1-2.) The aircraft is characterized by its black color, long narrow fuselage extending forward from a large delta wing, and two prominent engine nacelles located in the wing on each side of the fuselage. Other unique features include twin canted rudders, mounted on top of the nacelles, and large pointed bodies (spikes), extending forward from the nacelles. The sides of the fuselage, from the nose to the wing, extend outward to form lifting surfaces (chines) and provide space for internally-mounted payload equipment.

The ability of the aircraft to achieve high-altitude high-Mach flight derives from its propulsion system and its structure, which is ninety three percent titanium with the balance consisting of high-temperature composite materials. Black paint, by radiation, assists in the reduction of internal temperatures during high-speed flight. Virtually every part and component of the aircraft and its systems are designed and fabricated to withstand the rigorous conditions and requirements dictated by its flight envelope. (See figure 1-3 for surface temperatures generated at cruise conditions.)

During its military career, the SR-71A was used for reconnaissance. The SR-71B is the trainer version of the SR-71A. (See figure 1-4.) Aside from having two essentially identical cockpits, one cockpit for the instructor pilot and the other for the student pilot, the flight characteristics and capabilities of the SR-71B are identical with those of the SR-71A. The SR-71B aircraft and aircraft systems, aside from the requirements associated with two pilot cockpits, are essentially the same as described for the SR-71A. Where used, the term SR-71 denotes both SR-71A and SR-71B aircraft.

1.2 THE AIRCRAFT.

1.2.1 General Arrangement. As shown in figure 1-1, the forward part of the fuselage contains the cockpits, nose landing gear, nose landing gear wheel well, and radio (R) and electrical (E) bays. An air conditioning (AC) bay, aerial refueling door and receptacle, and a drag chute compartment are located on the top centerline of the fuselage. The balance of the fuselage consists of fuel tanks. Mission (payload) equipment bays are located within the chines. The nose section, of which there are several configurations, is detachable. The inner wings, between the fuselage and nacelles, contain the main landing gear, main landing gear wheel wells, and fuel tanks. The nacelles support the rudders and contain the engines and air inlet systems.

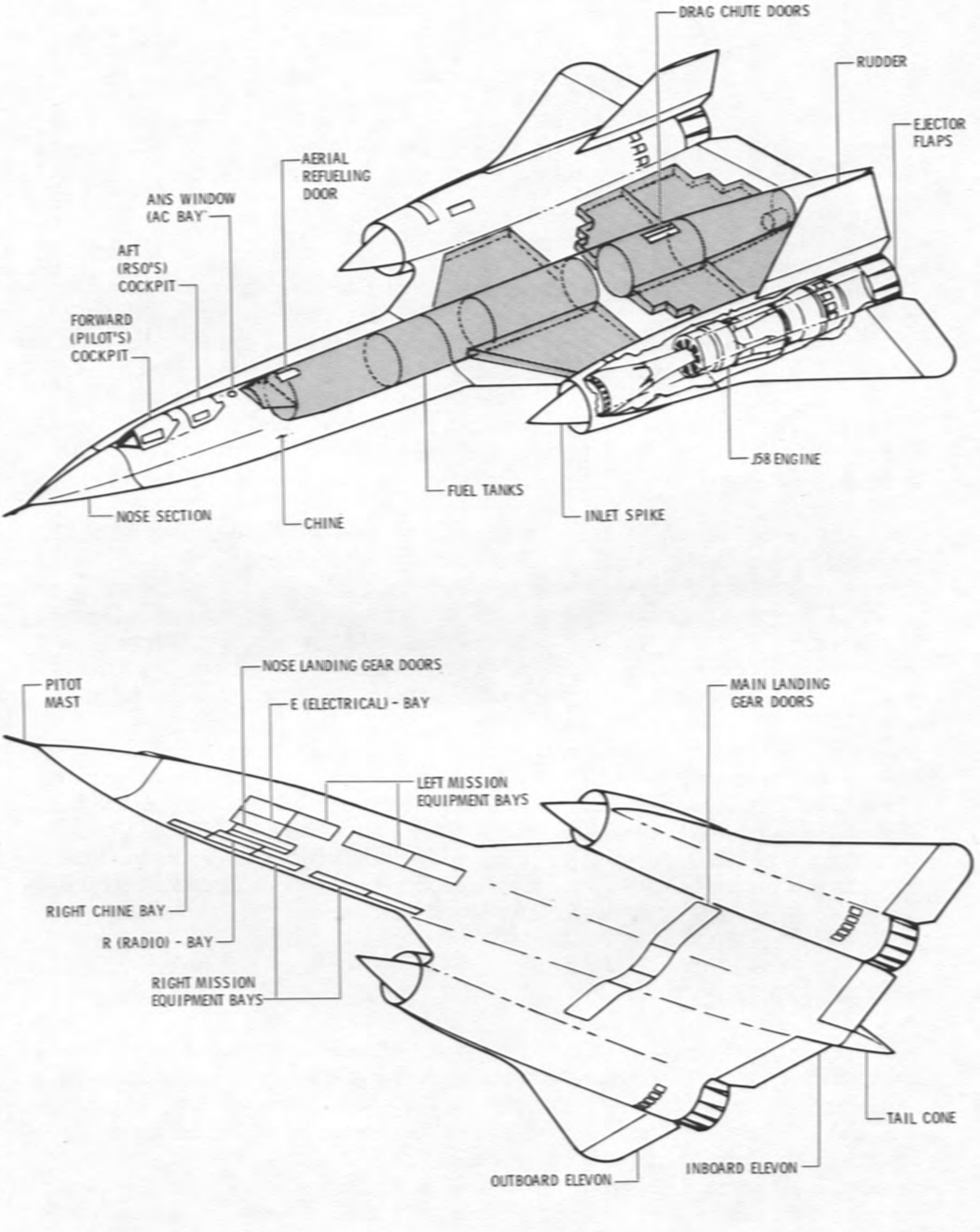


Figure 1-1. SR-71A General Arrangement

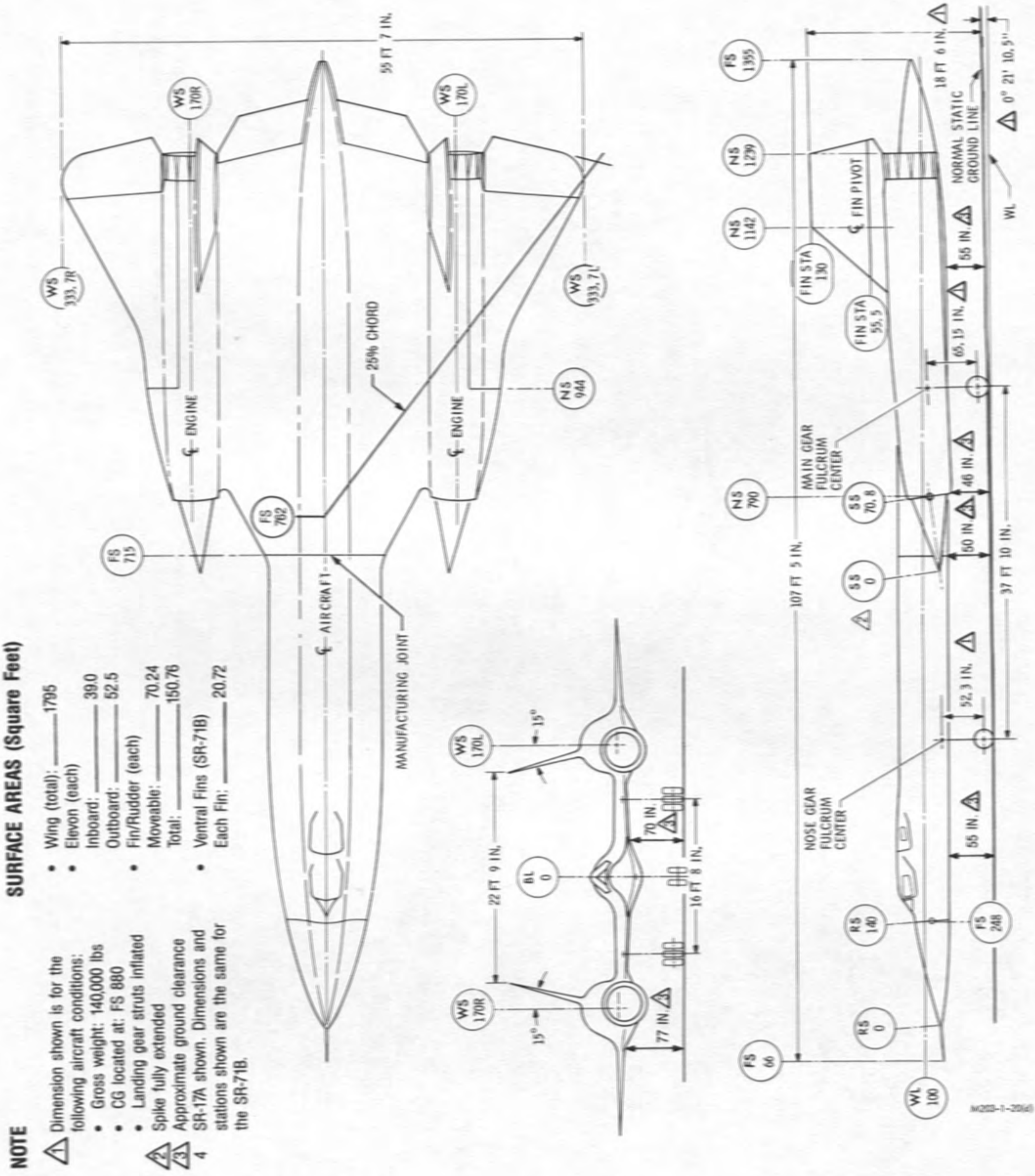


Figure 1-2. Principal Dimensions and Major Stations

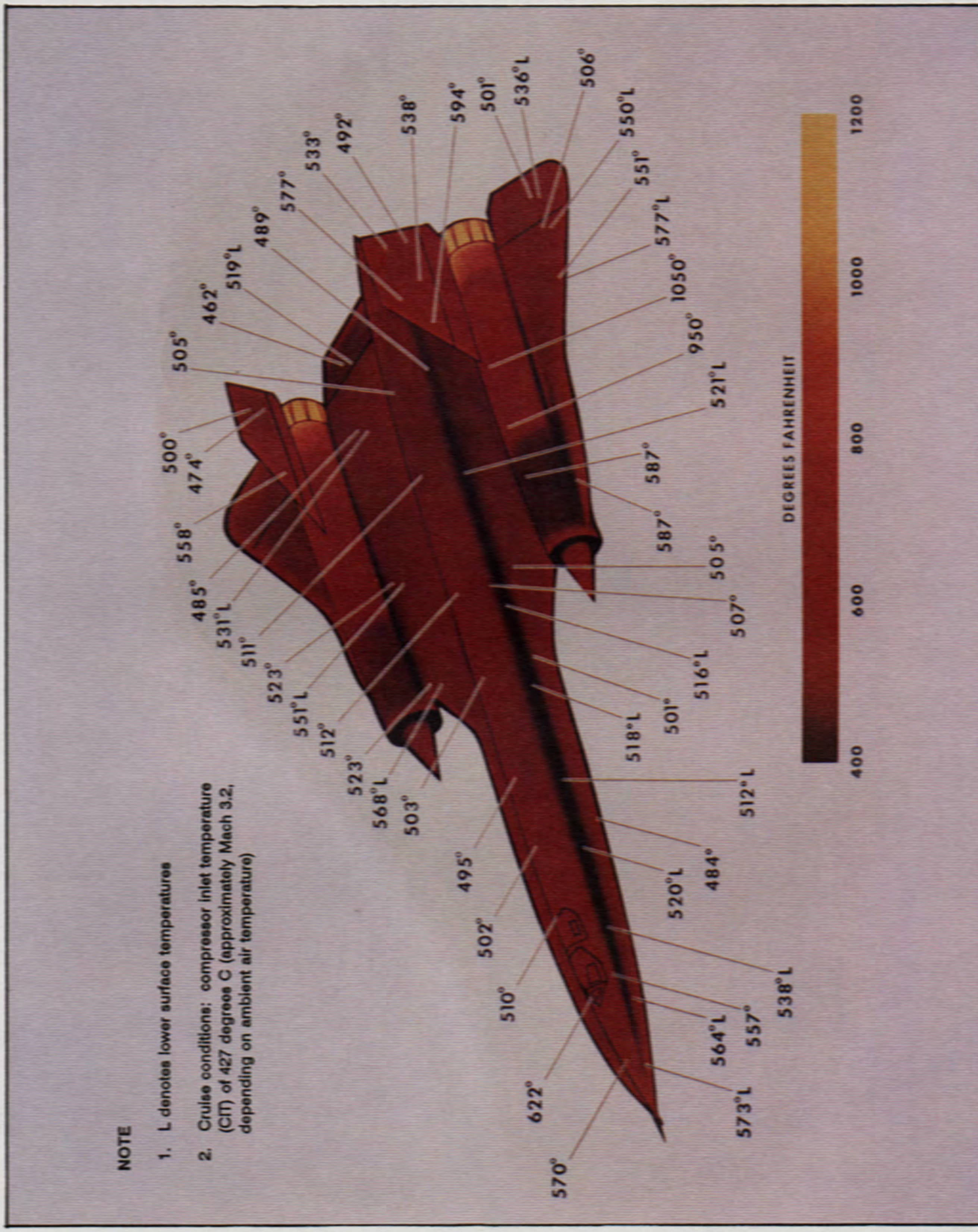


Figure 1-3. Surface Temperatures at Cruise Conditions

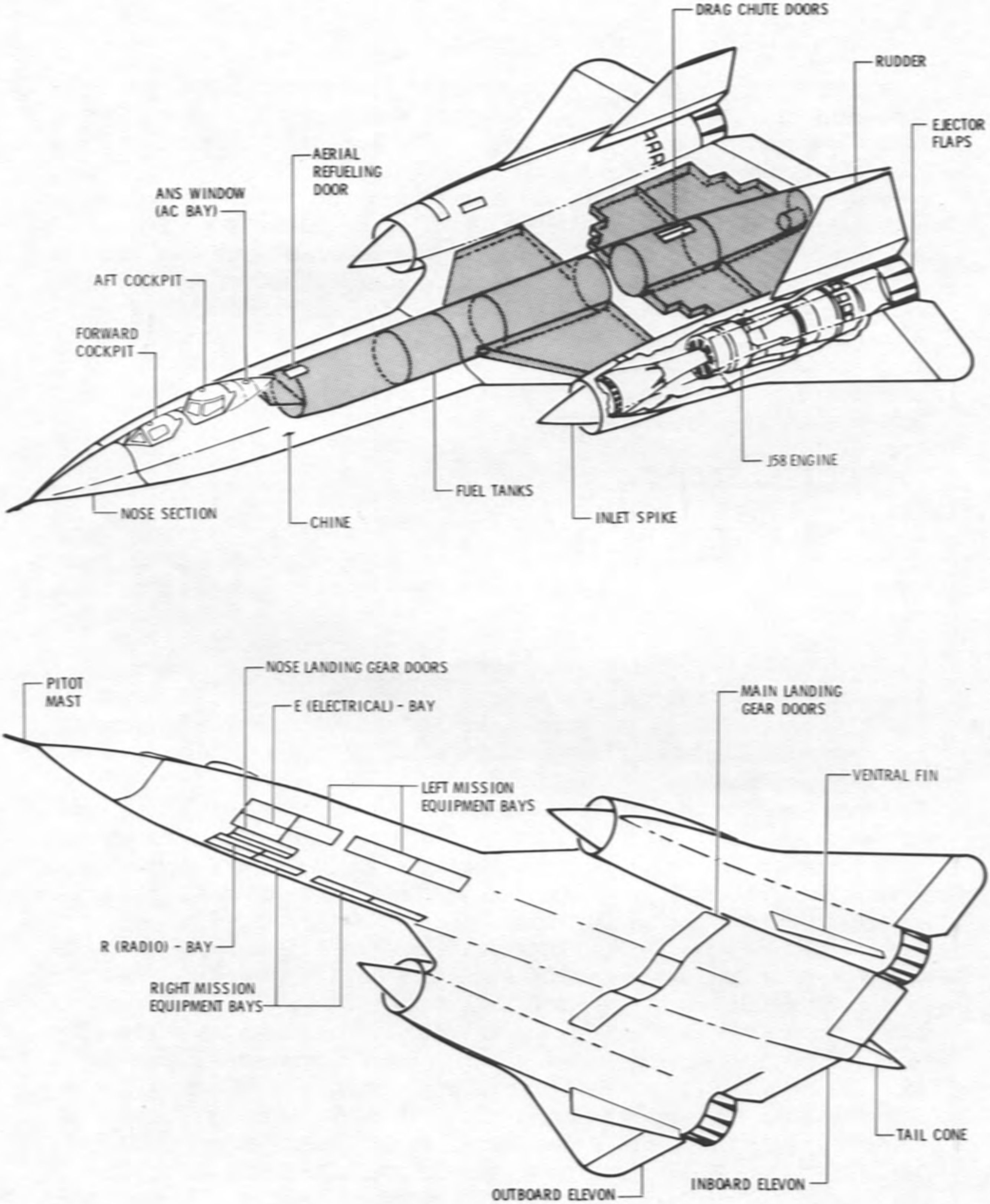


Figure 1-4. SR-71B General Arrangement

1.2.2 Airframe Components and Construction. Figure 1-5 shows the major components of the aircraft. The basic structure, essentially traditional in design, is fabricated from high strength titanium alloys. The fuselage forebody and aft body, joined during production at fuselage station (FS) 715, is a semi-monocoque design, utilizing fuselage rings and longerons for the framework. Non-metallic composite-honeycomb panels, secured to titanium ribs, form the chines. The wings are of multispar construction, with chord-wise stiffened skin panels attached to form box beams. Wing beams extend through the fuselage, from nacelle to nacelle. The upper and lower wing surfaces are corrugated with long longitudinal beads that permit differential expansion and contraction between the skin and wing structure during thermal heating and cooling. Non-metallic composite-honeycomb material is incorporated along the edges of the wing and in the outer surfaces of the spikes (refer to Section IV, figure 4-5). The rudders are of titanium.

1.3 AIRCRAFT SYSTEMS.

SR-71 aircraft systems are sophisticated and, considering the hostile flight environment and demanding operational requirements, elegantly simple in design and operation. The following paragraphs provide an overview of these systems. Refer to Section II for a more detailed description.

1.3.1 Propulsion, Flight Controls, and DAFICS Systems. In addition to the engines, the propulsion system includes moveable spikes and bypass doors, located at the forward end of the nacelles, to control engine inlet and bypass air. The inlet system, in the process of decelerating air flow at the face of the engine to subsonic speed, contributes substantially to overall thrust developed by the propulsion system. Of the total thrust produced at cruise speed (Mach 3.2), the engine delivers 17.6 percent, the ejector section (aft of the afterburner) 28.4 percent, and the inlet 54 percent. Flight control surfaces consist of elevons, located along the wing trailing edges, and two all-moving rudders (vertical fins). Engine air inlet spikes, forward bypass doors, and flight control surfaces are positioned by hydraulically-powered actuators and servos, controlled by pilot-initiated commands or by input signals from the Digital (computer-controlled) Automatic Flight and Inlet Control System (DAFICS). In addition to contributing to significantly improved flight performance and increased aircraft range, DAFICS virtually eliminates engine unstarts, once a problem. Pilot control of the flight control surfaces is through use of a conventional control stick and rudder pedals, the forces of which are mechanically transmitted to the electro-mechanically controlled elevon and rudder servos. Electronic control of the elevon and rudder servos is from the Automatic Flight Control System (AFCS) in the DAFICS. The AFCS includes a pitch autopilot, with Mach-hold and KEAS-hold functions, and a roll autopilot, with a heading-hold function. Coupled with navigation system steering input signals, the roll autopilot also has great-circle-steering capability. The AFCS also includes a three-axis Stability Augmentation System (SAS), which is normally in operation during flight.

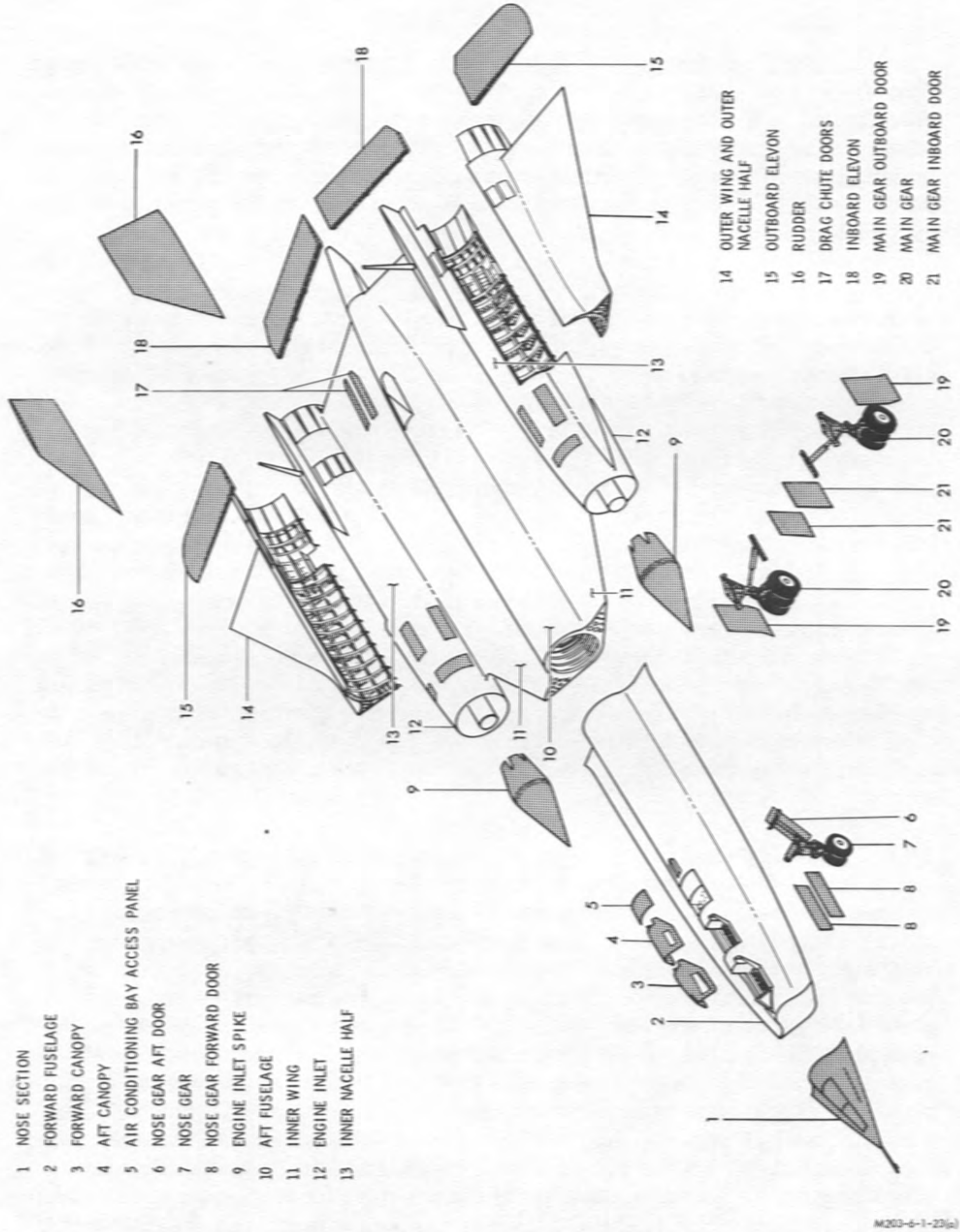


Figure 1-5. Major Airframe Components

1.3.2 Landing Gear, Drag Chute, and Hydraulic Systems. The tricycle landing gear consists of a nose gear, located beneath and just aft of the aft cockpit, and two main gear, located in the bottom of the wing between the fuselage and the nacelles. For full-stop landings, a pilot-actuated drag chute augments the main landing gear braking system. Four independent engine-driven hydraulic systems are paired to power the flight control surfaces, moveable inlet surfaces, landing gear, brakes, and air-refueling door.

1.3.3 Fuel, Environmental Control, Liquid Nitrogen, and Oxygen Systems. The fuel system includes six fuel tank groups, which take up most of the fuselage volume aft of the nose wheel well together with large areas within the inner wings. (See figure 1-1.) Fuel tank sequencing can be accomplished automatically, providing automatic control of aircraft c.g. (which is critical for flight stability and also affects range performance), or by pilot manual tank selection. Refueling can be accomplished in flight as well as on the ground. Due to its low vapor pressure (high flash point) JP-7 fuel, specially formulated for thermal stability to prevent coke deposits and varnish in fuel system passages, is ignited during engine start by a Chemical Ignition System (CIS). The Environmental Control System (ECS), powered by two identical heat exchanger/cooling turbine systems, air conditions and pressurizes the cockpits, provides ventilation air to crew pressure suits, cools and pressurizes equipment bays and compartments, and provides deicing air for the pilot's windshield. The liquid nitrogen system includes three liquid nitrogen (LN_2) Dewars on the SR-71A, two on the SR-71B. The purpose of the system is to pressurize and inert the fuel tanks and ancillary equipment with gaseous nitrogen. The oxygen system includes three liquid oxygen converters which provide gaseous oxygen to the crew members in both cockpits. Two of the converters are for normal oxygen supply, one is for standby supply.

1.3.4 Electrical, Communication, Navigation, and Instrument Systems. The electrical system is powered by two 60 KVA ac generators, with transformer rectifiers providing dc power. A static standby inverter and a dual battery provide emergency ac and dc power, respectively. Communication systems include an intercom system and HF, VHF, and dual UHF transceivers. Besides providing for intercockpit communications, the intercom system provides for communications between the cockpits and the ground crew and, during aerial refueling, between the aircraft and a compatibly-equipped tanker, via the refueling boom. Radio navigation equipment includes an ILS system and a TACAN transceiver. Also, the UHF systems have direction finding capability. Navigation systems include an Astroinertial Navigation System (ANS) and a backup Inertial Navigation System (INS). The forward cockpit of the SR-71A (both cockpits on the SR-71B) is equipped with controls and instruments necessary for controlling flight. An annunciator panel enables the pilot to monitor systems and conditions related to flight. The aft cockpit is equipped with controls and instruments associated with payload equipment control, communication, and navigation. Both cockpits have map projectors that provide navigation and communication data to assist in accomplishing a specific mission. Additionally, the forward cockpit on the SR-71A (both cockpits on the SR-71B) has a rear-view periscope that provides the pilot with an aft-looking view of the top of the fuselage, wings, and nacelles. This instrument can also be used to visually monitor a top-mounted payload.

1.3.5 Mission Recorder System (MRS). The MRS functions as both a mission and a maintenance recording system. The system continuously monitors and records hundreds of events and conditions occurring within the aircraft systems, and it has the capacity to monitor and record hundreds more for research and flight testing purposes. Recorded information, which includes crew voice audio and ANS navigation and time data, is used for detailed reconstruction of a specific mission as well as for maintenance analysis.

1.3.6 Internal Payload Areas. Payloads can be carried internally within the aircraft in the mission equipment bays, located within the fuselage chines, in a small compartment (C-bay) forward of the nose wheel well, and in the detachable nose section. (See figure 1-1.) These areas are supplied with cooling air from the environmental control system and, on SR-71A aircraft, have electrical wiring provisions. In addition to the equipment bays/compartments in the fuselage, adaptable to payload packages as large as 16 by 17 by 92 inches with weights up to 900 pounds, there are three distinctly different nose sections, any one of which can be installed for flight. The "lightweight" nose section can be used for generic payloads, as can the Optical Bar Camera (OBC) nose which is also adaptable to downward-looking equipment. The Advanced Synthetic Aperture Radar System (ASARS) nose, with its radome, is suitable for research and development of synthetic aperture radar equipment. Nose payload weights can range up to 550 pounds. Details on payload carrying capability are provided in Section III.

1.3.7 Ancillary Payload Systems. As part of its reconnaissance-role complement of equipment, the SR-71A has four systems which can be valuable in flight testing certain research payloads: a viewsight, a V/H system, a sensor event/frame count system, and an exposure control system. The viewsight provides a downward and forward view of terrain beneath the aircraft. The V/H system provides signals proportional to the angular velocity of the aircraft with respect to the earth. The sensor event/frame count system provides for MRS recording of certain events and related navigation data; permitting a correlation of these events (unique occurrences within a payload; e.g., a photographic exposure) with the exact time, location, heading, and attitude of the aircraft at the time of the event. The exposure control system supplies signals representing the setting, in degrees of sun angle, of an exposure control located in the aft cockpit.

1.4 CREW PRESSURE SUIT.

The model 1030 full-pressure suit provides crewmembers with a safe environment, regardless of cockpit pressure or temperature. The suit is required for flights above 50,000 feet. Although an oxygen mask assembly could be used for "shirt sleeve" flights below 50,000 feet, the pressure suit should be worn for all flights to ensure crew safety.

1.5 EGRESS SYSTEM.

1.5.1 Ejection Seat. The ejection seat is usable from zero speed and zero altitude to the limits of the flight envelope. The seat is a rocket-propelled, upward-ejecting unit which uses a drogue chute to stabilize seat descent. Man-seat separation occurs automatically on descent to, or at barometric altitudes below, 15,000 feet.

1.5.2 Canopy. In an emergency, the canopies can be ballistically jettisoned by operation of the external canopy release handle, externally mounted on the left side of the forward cockpit (fires both canopies); the internal canopy jettison handle, one in each cockpit (fires related canopy); or the D-ring on the ejection seat (fires related canopy then the seat).

1.6 OPERATIONS.

1.6.1 Weight. Zero fuel weight (ZFW) varies from approximately 56,000 to 60,000 pounds, depending on payload. Gross (loaded with fuel) takeoff weight (GTOW) varies from approximately 95,000 to 140,000 pounds. Maximum fuel load weight is 80,280 pounds.

1.6.2 Takeoff/Landing Distances. The following are typical distances for operation from Edwards AFB (2302 feet MSL) with a 60,000 pound (empty weight) aircraft on a dry 90 degree F day. With 35,000 pounds of fuel (95,000 pounds GTOW) the aircraft uses 5,000 feet of runway. With 80,000 pounds of fuel (140,000 pounds GTOW) the aircraft uses 7600 feet of runway. Landing distance, with 10,000 pounds of fuel remaining and drag chute deployed, is 4200 feet.

1.6.3 Flight Speeds. The aircraft is designed to achieve maximum cruise performance at speeds near Mach 3.2, at altitudes between 74,000 and 85,000 feet; however, it is capable of sustained flight at lower Mach and altitudes. Knots indicated airspeed (KIAS) are used to gauge subsonic speed, knots equivalent airspeed (KEAS) for supersonic speed. Typical takeoff and landing speeds are at 210 and 155 KIAS, respectively. Standard climbs are at 400 to 450 KEAS, although, current operating limit climbs can be made at 500 KEAS. Standard supersonic cruise speeds range from 310 to 400 KEAS; however, cruise speeds can be extended to 500 KEAS within current operating limits.

1.6.4 Mission Profiles. The aircraft is capable of taking off with a full fuel load, climbing to 80,000 feet at Mach 3.0, and remaining there for approximately 1 hour, depending on payload weight. Figure 1-6 show the profile for a maximum range flight without aerial refueling. Figure 1-7 shows the profile of a Mach 3.2 flight without aerial refueling. Figure 1-8 shows flight profiles from 60,000 feet, at Mach 2.4, to 85,000 feet, at Mach 3.2, using aerial refueling. The aerial refueling profiles originate at tanker hookup. Figure 1-9 shows the aircraft turn radius at various bank angles and cruise speeds.

1.6.5 Climb, Dynamic Pressure, and Compressor Inlet Temperature Profiles. Figure 1-10 shows the current speed/altitude envelope of the aircraft. (Refer to Section V for extended flight capabilities.) Minimum airspeeds are shown to the left of the envelope, maximum airspeeds to the right. A standard climb, at 450 KEAS to Mach 2.6 at 60,000 feet, is shown, with KEAS bleeding off to 390 at Mach 3.2 at 75,000 feet. Climb at 500 KEAS can be accomplished, with KEAS bleed to 440 starting at Mach 2.6 at 56,000 feet. As shown in figure 1-10, KEAS is normally bled off as the climb proceeds; however, it is not necessary to do so. A 450 or 500 KEAS climb can continue directly to Mach 3.2, which will be reached at a lower than normal altitude. Dynamic pressure "*q*" versus KEAS is shown in figure 1-11. Compressor inlet temperature (CIT) versus Mach is shown in figure 1-12.

1.6.6 Operating Limits. Figure 1-13 lists specific operating limits for flight of the aircraft.

1.6.7 Flight Operations and Safety. Flight operations are conducted and controlled by NASA. NASA follows their standard flight readiness review process to assure safety of the flight crew and aircraft. Lockheed and NASA will coordinate with the researcher to assure that safety is a fundamental ingredient in the installation and flight testing of experiments.

1.7 AERODYNAMIC DATA.

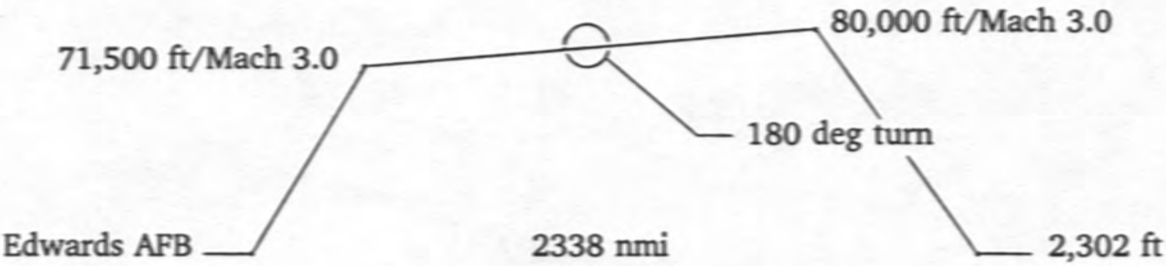
Complete flight-verified aerodynamic data is documented and available for experimental use and design. Complete computer fluid dynamics (CFD) data is in development and will be available for use. Lockheed and NASA are prepared to assist the researcher in using this data to optimize the installation of his experiment or payload.

1.8 FLIGHT PARAMETERS FOR EXPERIMENTS.

Time at Mach 3.2 relative to payload weight is shown in figure 1-14. Available excess thrust is shown in figure 1-15. Flight parameters associated with experiments mounted on top of the aircraft are shown in figures 1-16 through 1-18.

1.9 PROGRAM LIFE EXPECTANCY.

The useful life of the airframe is essentially without limit. Available spares ensure flight operations for approximately 10 years, at planned utilization rates. Lockheed Special Repair Activity (SRA) capability can extend the life of the program indefinitely.



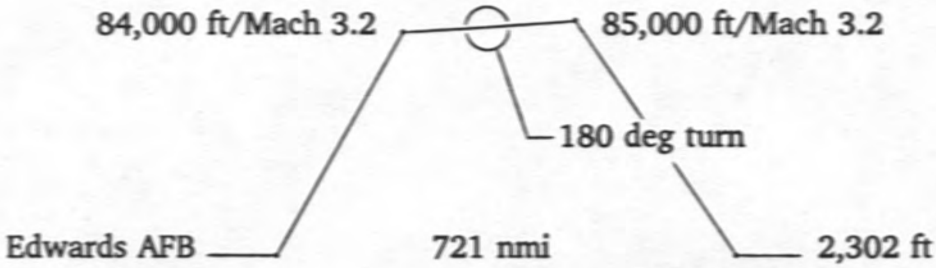
Mission Profile - Mach 3.0, Max Range Cruise

Depart/Return: Edwards AFB; Elevation, 2302 feet; Temperature, 90 deg F
Weight: Aircraft (ZFW), 60,000 lbs; Fuel, 80,000 lbs; Gross: 140,000 lbs
Temperature at Altitude: Standard Day

Climb Profile:	450 KEAS	Total Distance Traveled:	2,338 nmi
Start Cruise Alt.:	71,500 feet	Total Mission Time:	1 hr, 41 min
End Cruise Alt:	80,000 feet	Mach 3.0 Time:	1 hr, 4 min

PHASE	DISTANCE (NMi)	TIME: MINS (Phase/Total)	FUEL USED (Pounds)	FUEL REMAINING (Pounds)
Taxi	--	30/30	4,500	75,500
Climb to 30K Feet	34	4/34	6,000	69,500
Accel to Mach 1.25	31	4/38	4,100	65,400
Climb to Cruise	250	15/53	15,000	50,400
Cruise	803	28/81	16,464	33,936
Turn	217	8/89	4,704	29,323
Cruise	803	28/117	16,464	12,768
Descent	200	4/131	2,200	10,560

Figure 1-6. Maximum Range Mission Profile



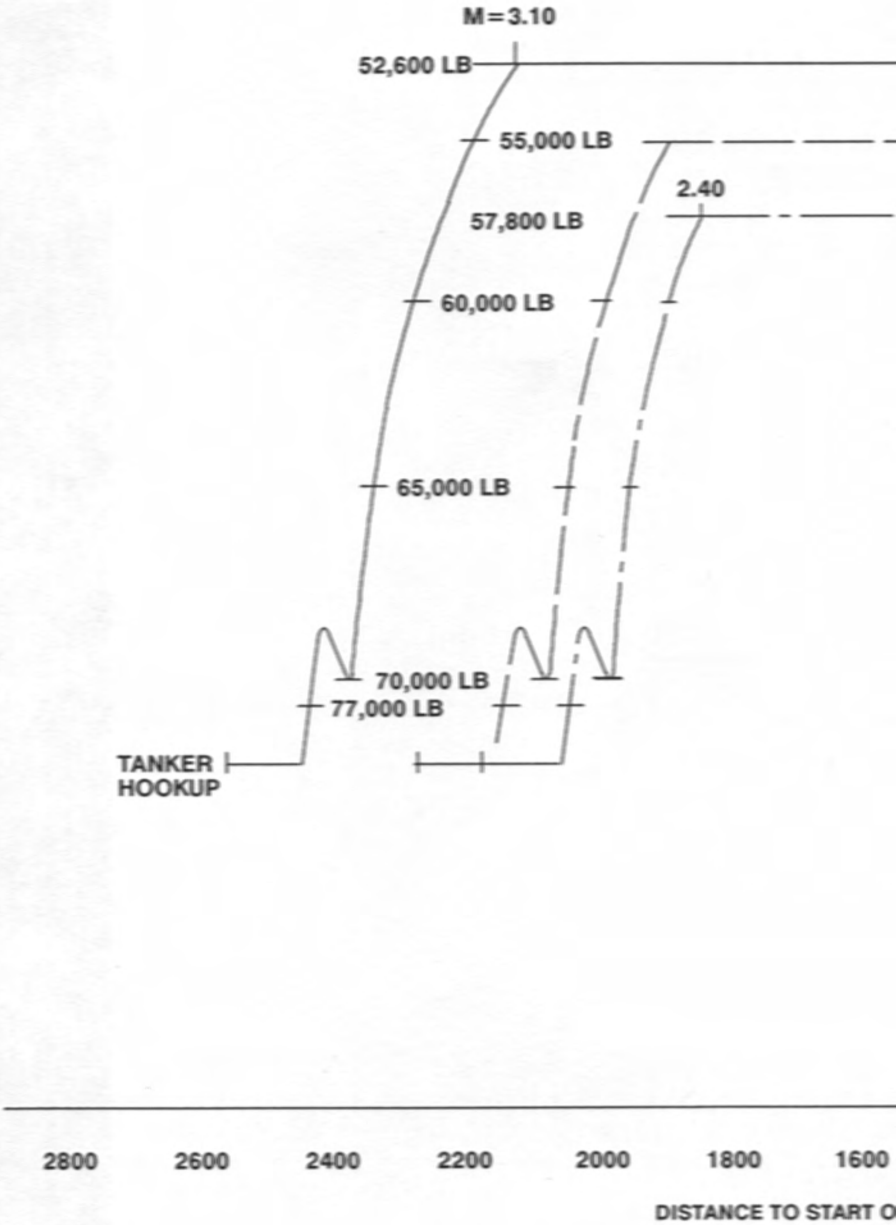
Mission Profile - Mach 3.2, Max Afterburner Cruise

Depart/Return: Edwards AFB; Elevation, 2302 feet; Temperature, 90 deg F
Weight: Aircraft (ZFW), 60,000 lbs; Fuel, 35,000 lbs; Gross: 95,000 lbs
Temperature at Altitude: Standard Day

Climb Profile:	450 KEAS	Total Distance Traveled:	721 nmi
Start Cruise Alt.:	84,000 feet	Total Mission Time:	53 min
End Cruise Alt:	85,000 feet	Mach 3.0 Time:	12 min

PHASE	DISTANCE (NMi)	TIME: MINS (Phase/Total)	FUEL USED (Pounds)	FUEL REMAINING (Pounds)
Taxi	--	15/15	2,250	32,750
Climb to 30K Ft	5	2.5/17.5	2,200	30,550
Accel to Mach 1.25	22	2.5/20	2,200	28,350
Climb to Cruise	175	7/27	10,000	18,350
Cruise	51	2/29	854	17,496
Turn	217	8/37	4,670	12,826
Cruise	51	2/39	854	11,971
Descent	200	14/53	2,200	9,771

Figure 1-7. Maximum Afterburner Mission Profile


M=3.10
52,600 LB
55,000 LB
2.40
57,800 LB
60,000 LB
65,000 LB
70,000 LB
77,000 LB
TANKER
HOOKUP
2800
2600
2400
2200
2000
1800
1600
DISTANCE TO START O

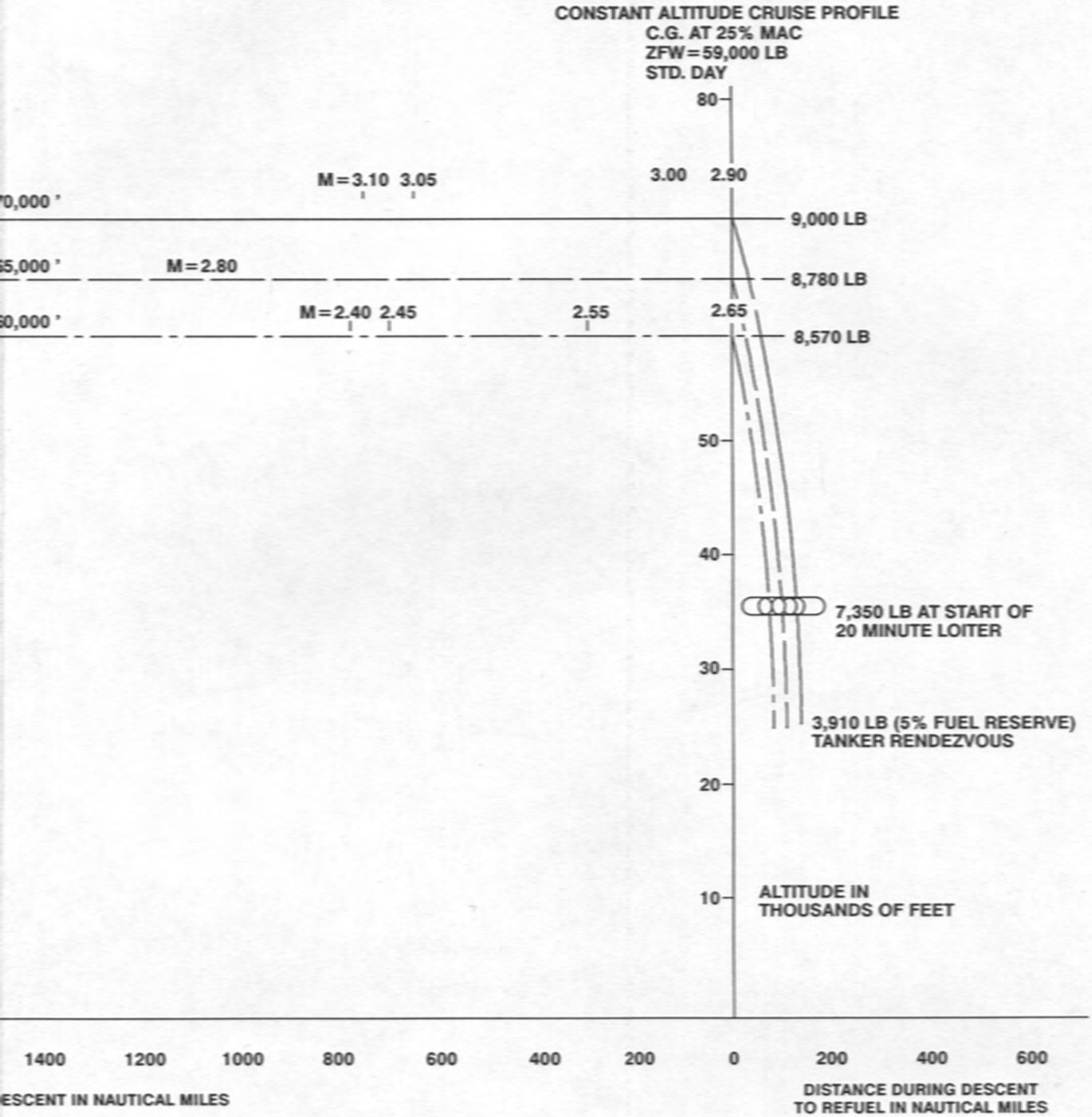
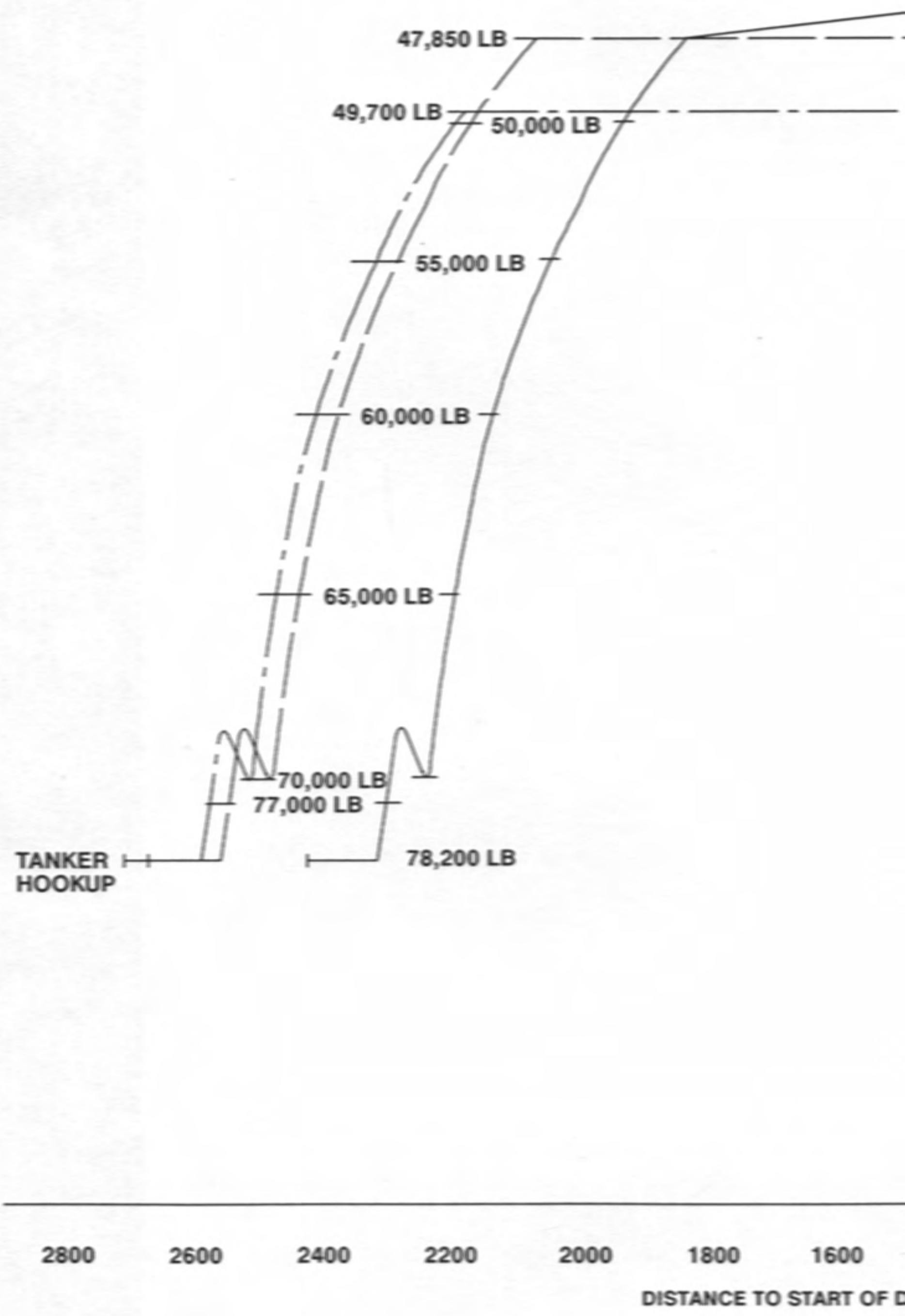


Figure 1-8.
Mission Profile Using Aerial Refueling
(Sheet 1 of 2)


47,850 LB
49,700 LB
50,000 LB
55,000 LB
60,000 LB
65,000 LB
70,000 LB
77,000 LB
78,200 LB
TANKER
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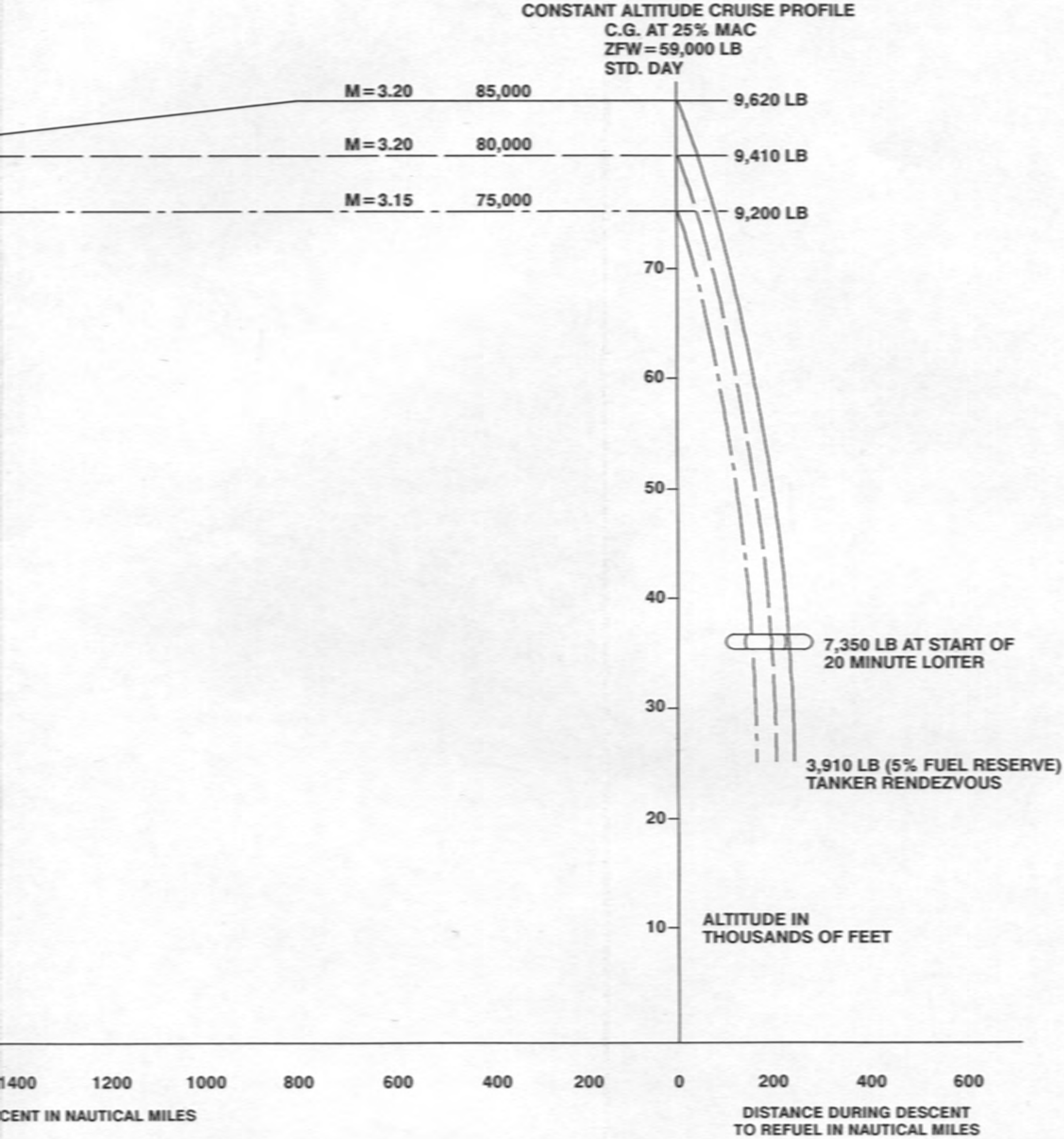


Figure 1-8.
Mission Profile Using Aerial Refueling
(Sheet 2 of 2)

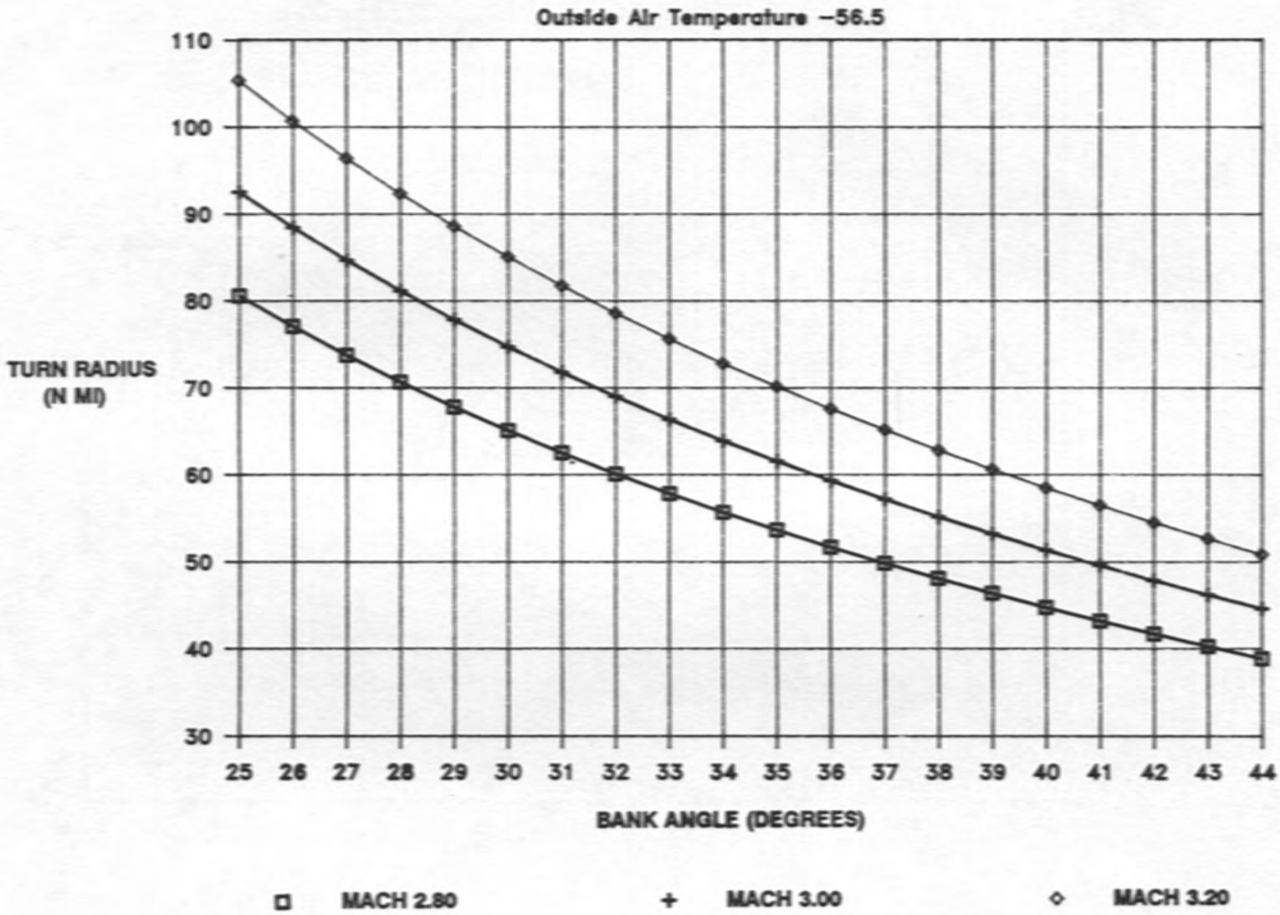


Figure 1-9. Radius of Turn

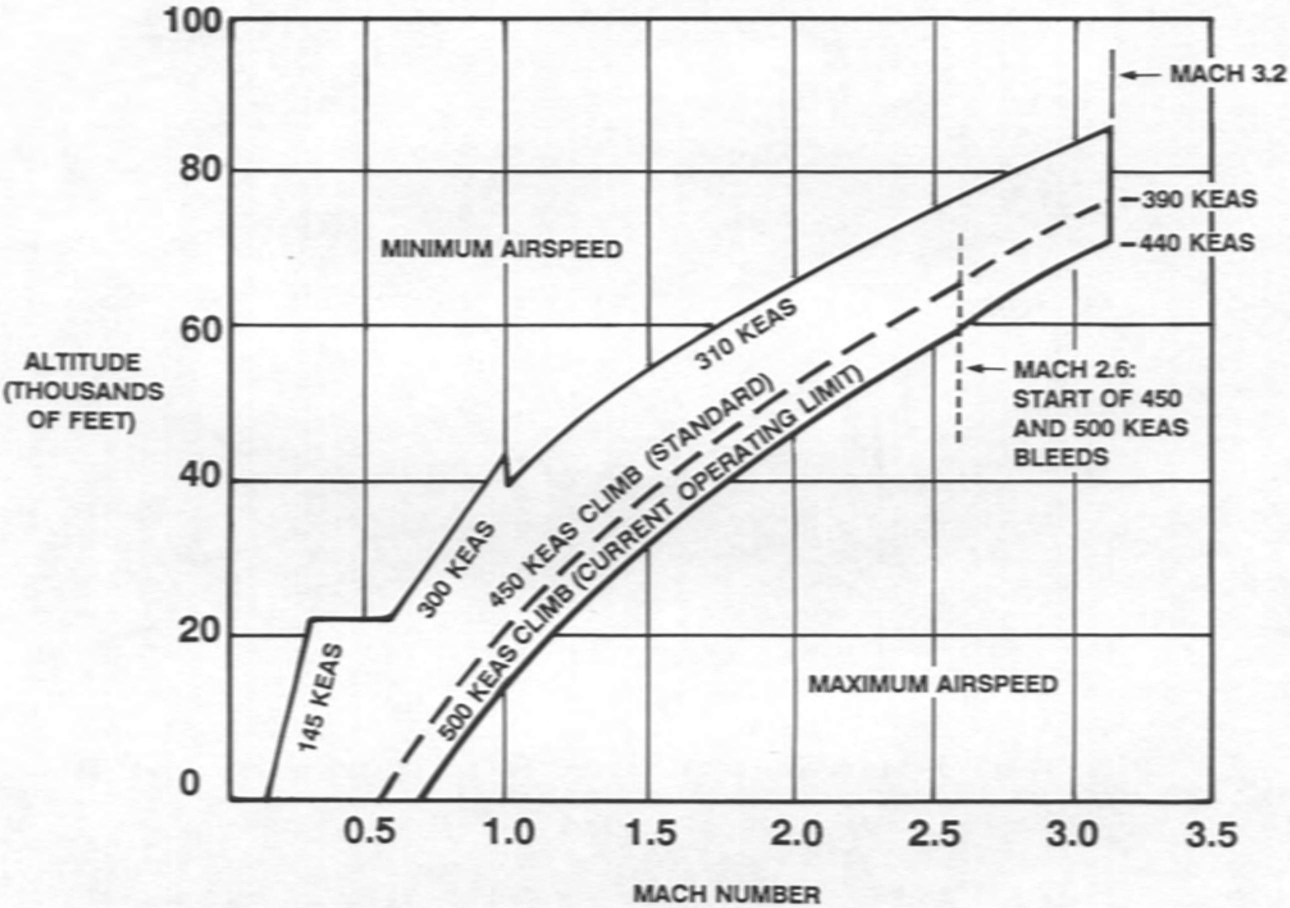
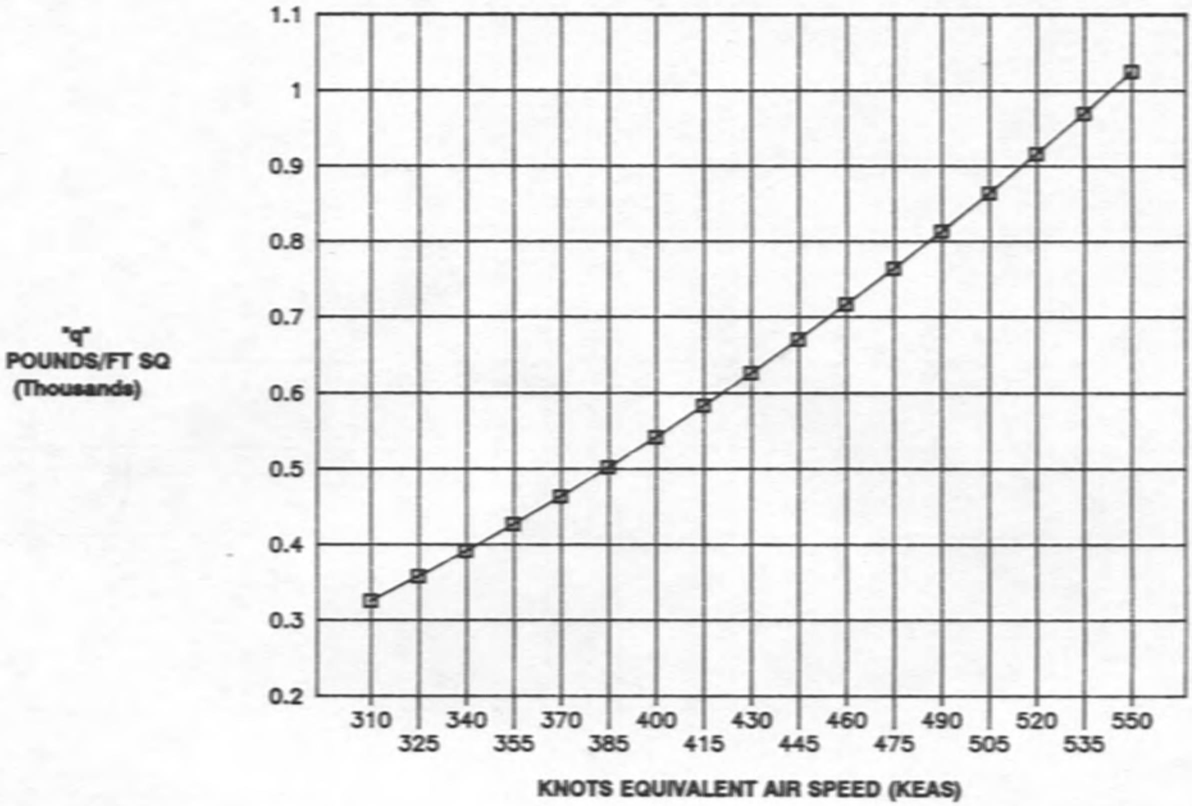


Figure 1-10. Current Flight Envelope



STANDARD CLIMB SCHEDULES

450 KEAS Climb Schedule △1

Mach	KEAS	ALT
0.90	450	15.0
to	to	
2.60	450	61.0
2.70	440	63.0
2.80	430	66.0
2.90	420	68.0
3.00	410	71.0
3.10	400	73.0
3.20	390	75.0

500 KEAS Climb Schedule △2

Mach	KEAS	ALT
0.90	500	10.0
to	to	
2.60	500	56.0
2.70	490	58.0
2.80	480	61.0
2.90	470	63.0
3.00	460	66.0
3.10	450	68.0
3.20	440	70.0

NOTE

△1 Max "*q*" continues up to 61K feet, then diminishes.
△2 Max "*q*" continues up to 56K feet, then diminishes.

Figure 1-11. Dynamic Pressure "*q*" Profiles

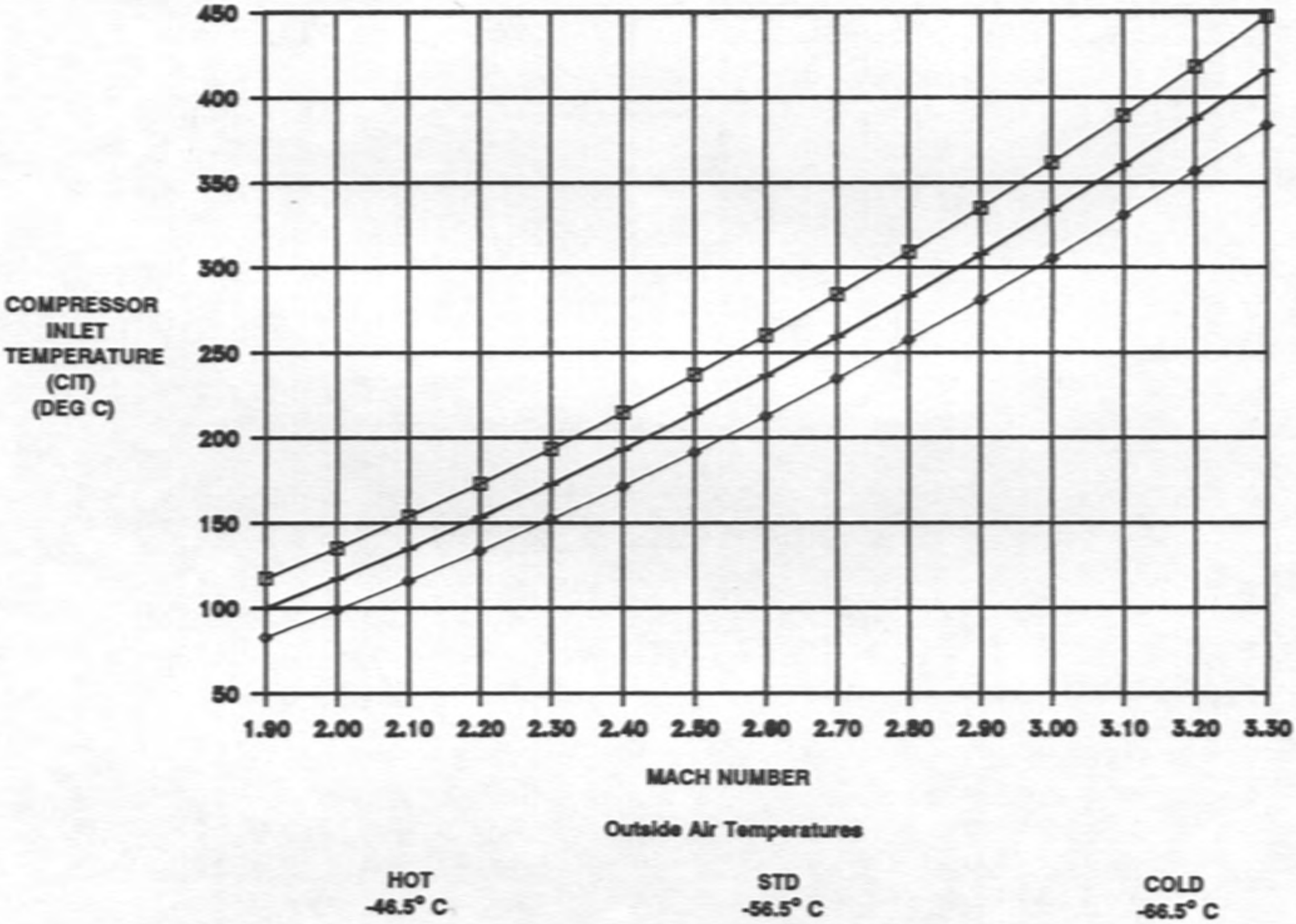


Figure 1-12. Compressor Inlet Temperature (CIT) versus Mach

Minimum Crew:	one; aircraft can be flown solo
Fuel:	JP-7 only
Tires, Maximum Ground Speed:	239 knots
Drag Chute Deployment, Maximum Airspeed:	210 knots

Engine Temperatures:

CIT: 427 deg C: normal operation - inlet guide vanes (IGV) cambered
150 deg C: IGV axial
(See figure 1-12 for CIT versus Mach profile.)

EGT: 775 to 805 deg C: normal operation

Maximum Weight, Takeoff/Landing:

Gross Weight:
GTOW: 143,000 lbs
Landing: 75,000 lbs

Maximum Fuel Weight:
Routine Full-Stop Landings: fuel < 10,000 lbs
Touch and Go Landings: fuel < 25,000 lbs

Load Factor:

	Weight	G Loads
At Mach < 2.0:	65,000 to 124,000 lbs	-0.2 to 2.5
	124,000 to 143,000 lbs	-0.2 to 2.0
At Mach 2.0 to 2.6:	All weights	-0.1 to 2.0
At Mach 2.6 to 3.2:	All weights	-0.1 to 1.5

Mach Number:

Manual Inlet Operation: Mach 3.0
Design Mach: Mach 3.2

Dynamic Pressure "*q*" (see figure 1-11.):

Normal cruise: 325 to 686 lbs/ft^2 at 310 to 450 KEAS
Current Operating Limit: 847 lbs/ft^2 at 500 KEAS
Extended Limit: 1000 lbs/ft^2 at 543 KEAS △1

Figure 1-13. Operating Limits (Sheet 1 of 2)

Angle of Attack (Alpha):

Critical angle of Attack: Angle of attack above 18 degrees can result in uncontrollable pitch up.

Subsonic:

- Below 25,000 Feet: 14 degrees
- Above 25,000 Feet: 10 degrees

Supersonic:

- Auto Inlet: 8 degrees
- Manual Inlet, above 70,000 Feet: 6 degrees

Angle of Sideslip (Beta):	450 KEAS	350 KEAS	SUBSONIC
	3 degrees	4.5 degrees	12 degrees

Airspeed, Minimum - Subsonic:

Below 25,000 feet: 145 KIAS or KIAS at 14 degrees alpha (max) △2
Above 25,000 feet: 300 KEAS or KIAS at 10 degrees alpha (max) △2

Airspeed, Minimum/Maximum - Supersonic:

Normal Operation:

- Minimum Speed: 310 KEAS or KEAS at 8 degrees alpha (max) △2
- Maximum Speed, with CG fwd of 17% ref chord: 400 KEAS
- Maximum speed, with CG within limits aft of 17% ref chord: 450 KEAS
- Current Operating Limit: 500 KEAS

Extended Limit Max Speed: 543 KEAS △1

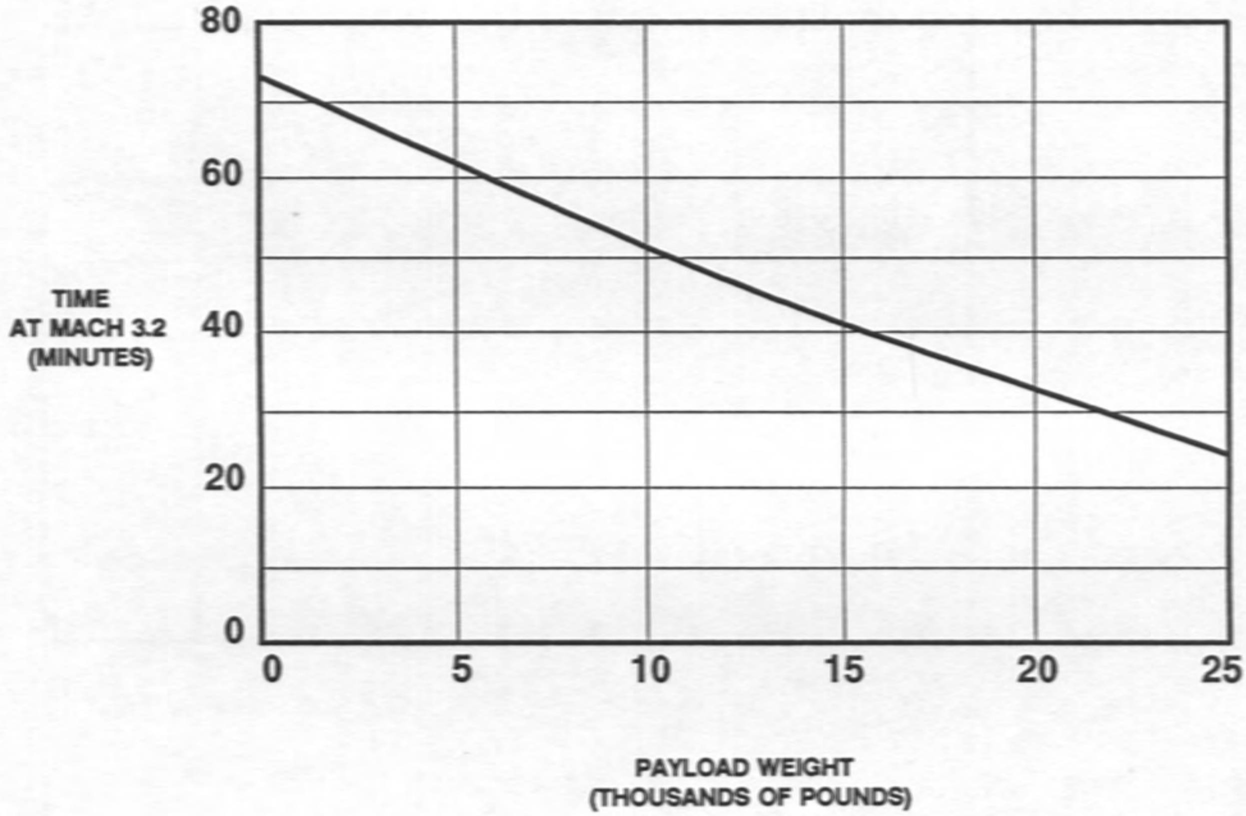
Center of Gravity (as a percent of reference chord):

Subsonic: 17 to 22%
Mach < 1.8, gross weight <100,000 lbs: between 14.5 and 17%
Supersonic - Maximum aft CG:

- Below Mach 3.2: between 22 and 25%
- Above Mach 3.2: forward of 25% by 0.7% for each 0.1 Mach increment increase

NOTE: △1 Controlled experimental flights only. See Section V.
△2 Whichever speed occurs first.

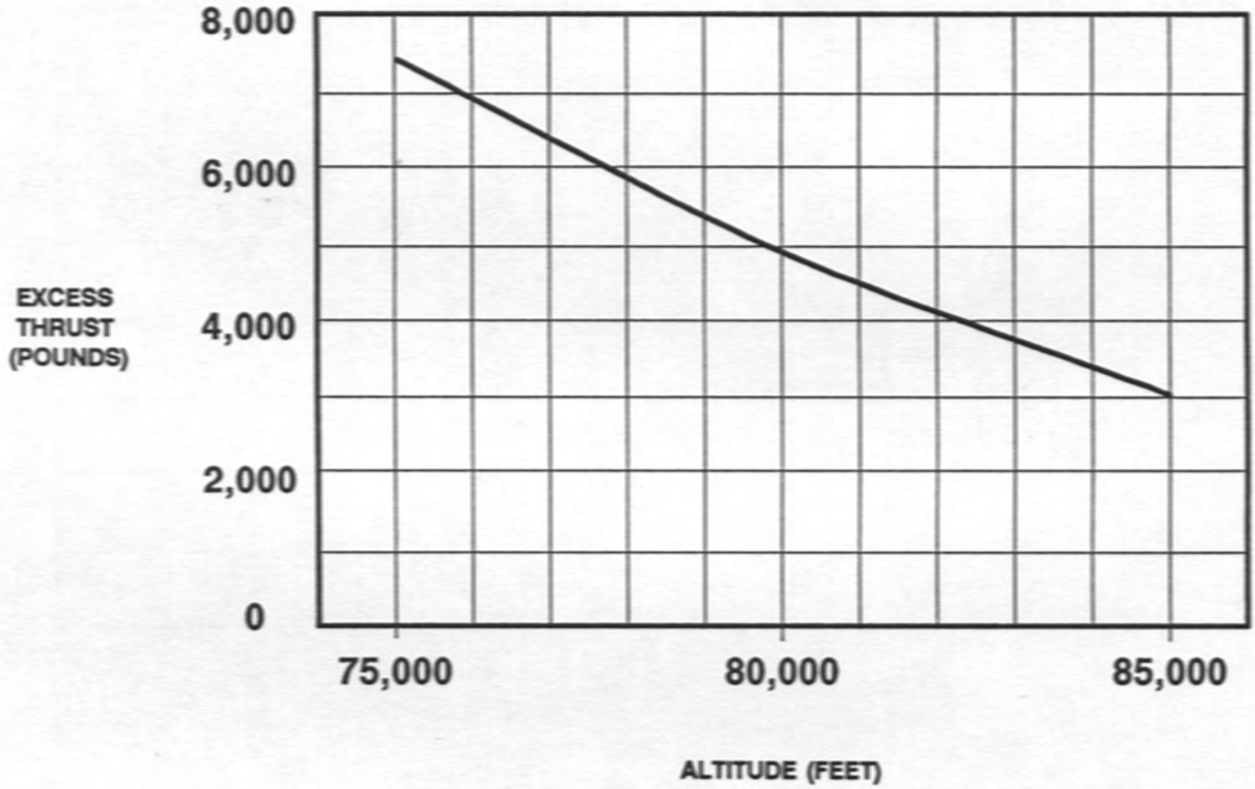
Figure 1-13. Operating Limits (Sheet 2 of 2)



CONDITIONS

GROSS TAKEOFF WEIGHT:	140,000 POUNDS
ZERO FUEL WEIGHT:	59,000 POUNDS
RESERVES:	10,000 POUNDS
STANDARD DAY	

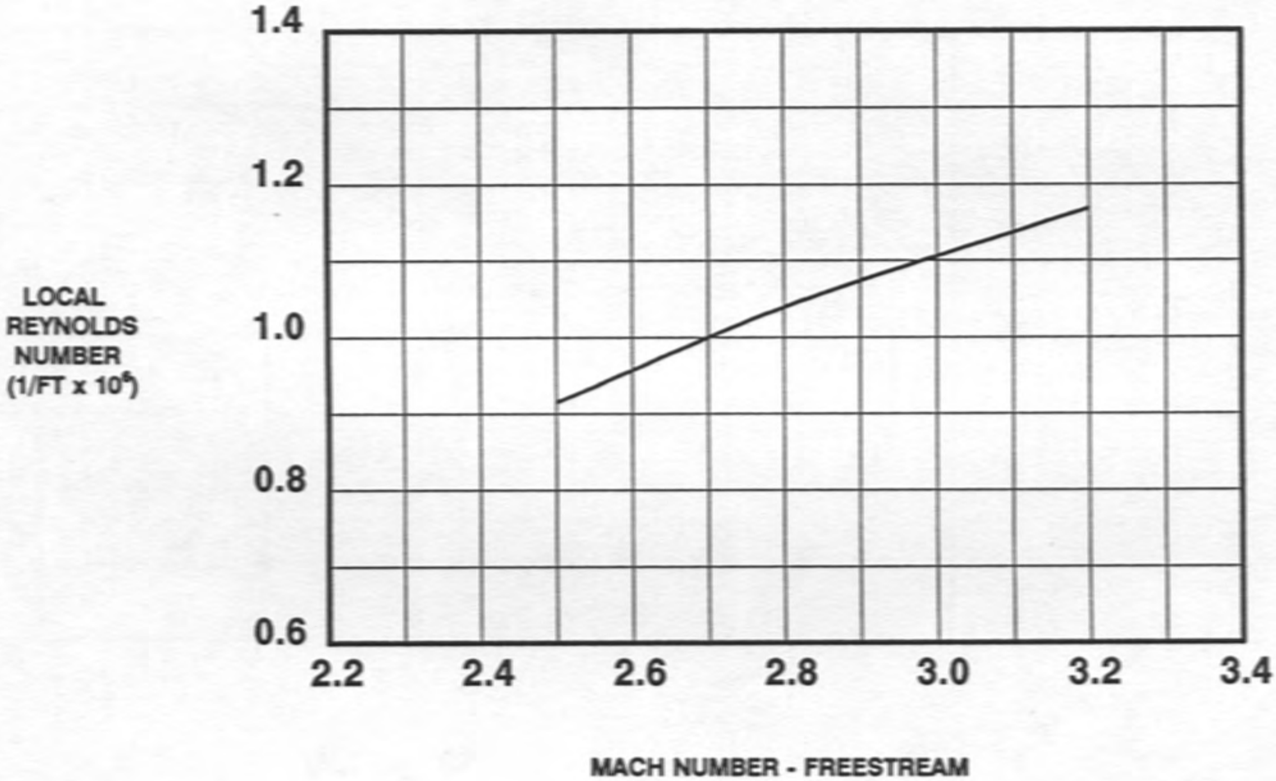
Figure 1-14. Time at Mach 3.2 versus Payload Weight



CONDITIONS

MAXIMUM AFTERBURNER
CRUISE C_L = .128
STANDARD DAY

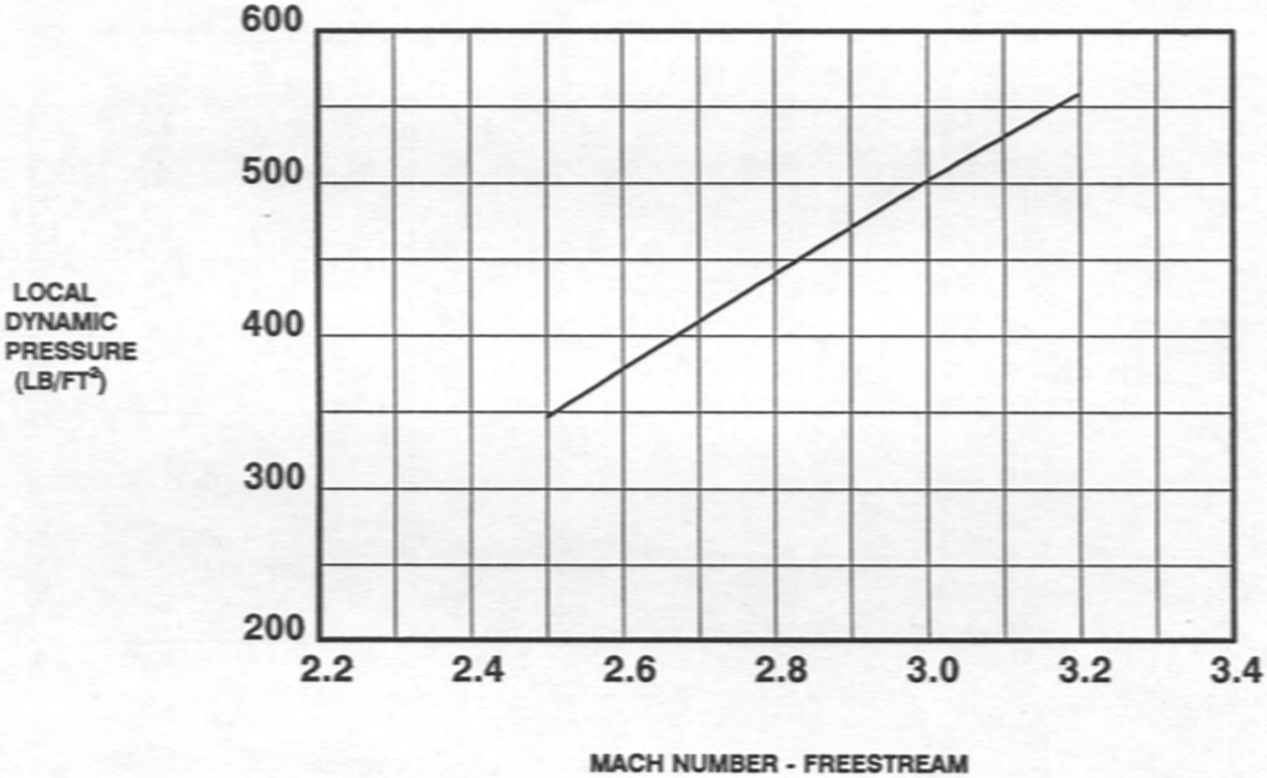
Figure 1-15. Excess Thrust versus Altitude at Mach 3.2



CONDITIONS

ALTITUDE: 75,000 FEET
DATA MEASUREMENT: 20 TO 80 INCHES ABOVE FUSELAGE

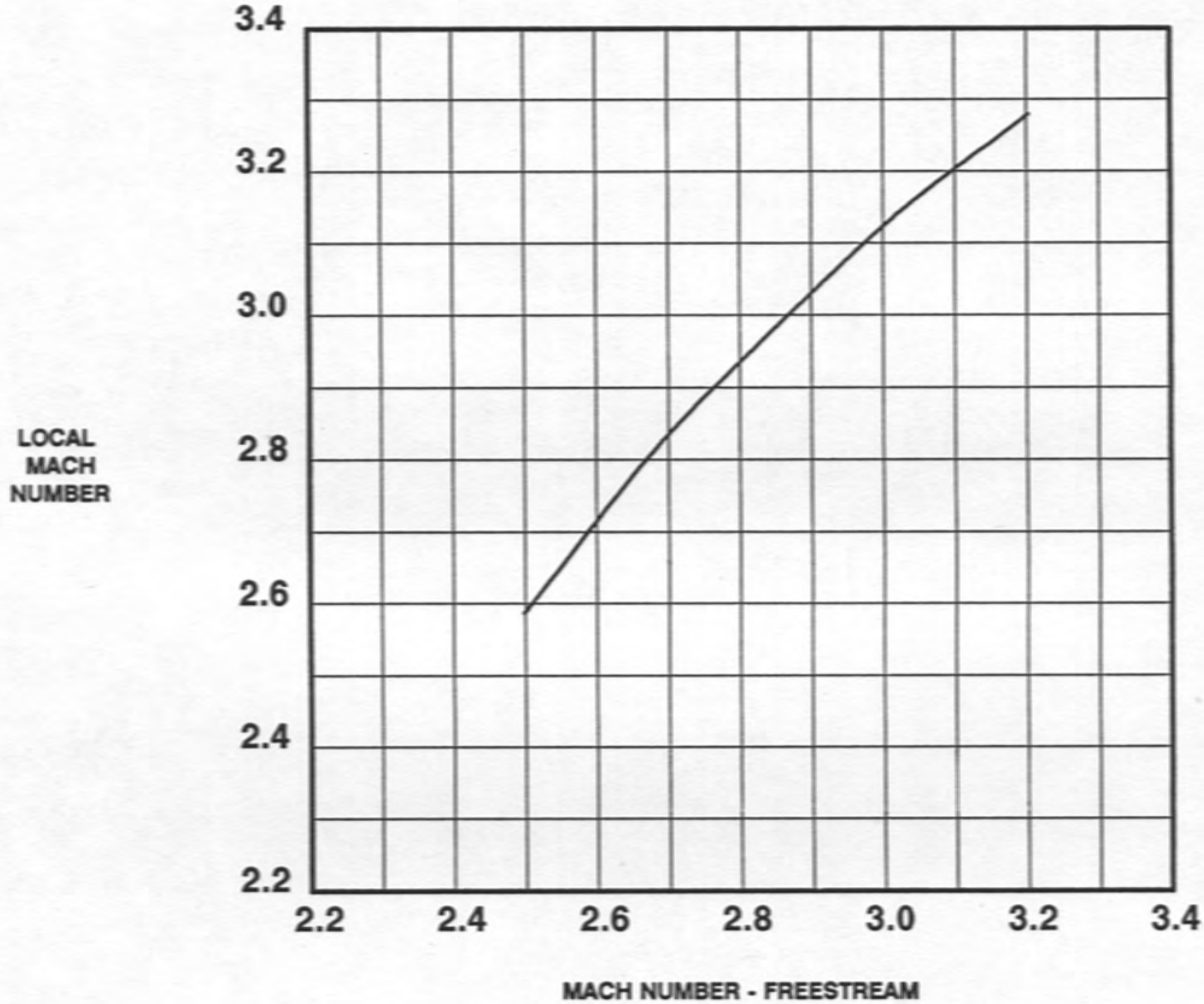
Figure 1-16. Flow Field Survey, Upper Surface Reynolds Number



CONDITIONS

ALTITUDE: 75,000 FEET
DATA MEASUREMENT: 20 TO 80 INCHES ABOVE FUSELAGE

Figure 1-17. Flow Field Survey, Upper Surface Dynamic Pressure



CONDITIONS

ALTITUDE: 75,000 FEET
DATA MEASUREMENT: 20 TO 80 INCHES ABOVE FUSELAGE

Figure 1-18. Flow Field Survey, Upper Surface Mach Number

SECTION II

AIRCRAFT SYSTEMS

2.1 GENERAL.

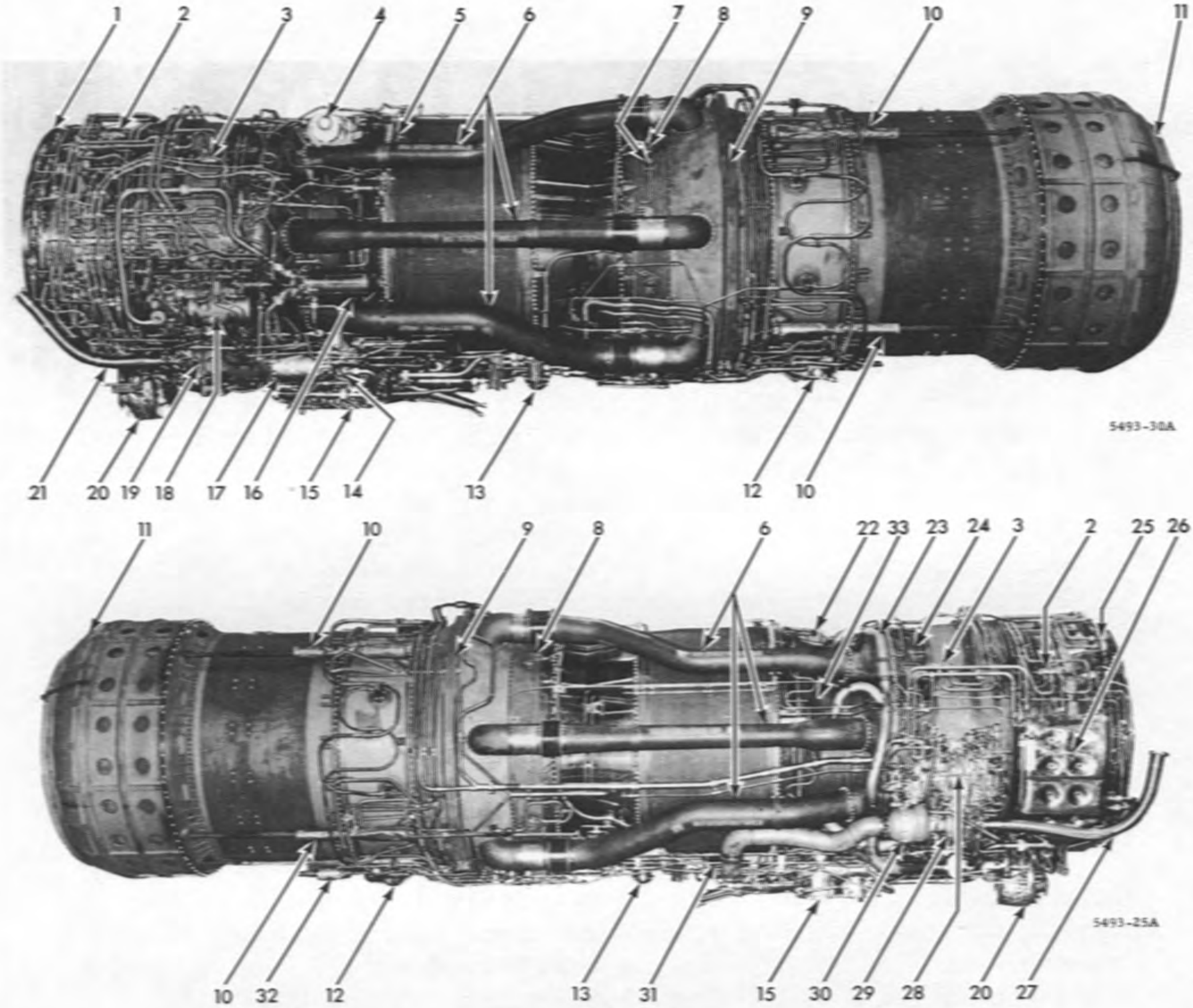
Information in this section is intended to provide an understanding of the purpose, scope, and capabilities of the aircraft and ancillary systems. If in-depth information is required, reference should be made to the SR-71A flight manual and to the SR-71-2 series of organizational (flight-line) maintenance manuals. The information provided in this section is exclusive of precautions and warnings regarding operation and maintenance of these systems. Such information, covering personnel safety and prevention of damage to the aircraft, aircraft systems, and equipment is provided in the -2 series manuals. References made to the pilot's instruments and controls in the forward cockpit of the SR-71A, also apply, unless otherwise stated, to both cockpits of the SR-71B.

2.2 POWERPLANT.

The aircraft is powered by two Pratt and Whitney J-58 engines, designed to operate continuously in afterburner at compressor inlet temperatures of up to 427 degrees C, for high-altitude high-Mach cruise. At sea level, static condition and standard day, the uninstalled engine, in afterburner, delivers 34,000 pounds of thrust. The J-58 engine consists of the following sections: compressor, diffuser, combustion, turbine, and afterburner. Engine and engine components are shown in figure 2-1. Engine instruments and throttle quadrant are shown in figure 2-2.

The engine incorporates moveable inlet guide vanes (IGV) to straighten airflow to the compressor. The trailing edges of the vanes change position from axial to cambered in response to the main fuel control. The vanes are normally in the axial position, providing more thrust during takeoff and acceleration to intermediate supersonic speed. Actuation to the cambered position occurs at a compressor inlet temperature (CIT) of 85 to 115 degrees C (about Mach 1.9) and is mandatory above a CIT of 150 degrees C (about Mach 2.0).

The compressor section is a nine-stage, axial-flow, single-spool, 8:8.1 pressure ratio compressor. The compressor incorporates a start bleed system, which is a bypass bleed air system that diverts fourth stage air downstream of the turbine, and mechanical linkage to external gear boxes. The compressor bypass bleed air enters the turbine exhaust near the front of the afterburner, where it provides cooling and increased thrust. Bypass bleed is scheduled by the main fuel control as a function of CIT and engine speed. Bleed air injection occurs at a CIT of 85 to 115 degrees C (about Mach 1.9).



1 COMPRESSOR INLET CASE
2 COMPRESSOR BYPASS ACTUATOR (4)
3 START BLEED DOOR ACTUATOR (3)
4 CHEMICAL IGNITION UNIT
5 CHEMICAL IGNITION PROBE
6 COMPRESSOR AIR BYPASS TUBES (6)
7 AFTERBURNER IGNITION PROBE
8 EGT THERMOCOUPLE (9)
9 REAR SUPPORT LUGS
10 EXHAUST NOZZLE ACTUATOR (4)
11 EXHAUST NOZZLE
12 ENGINE NOZZLE CONTROL
13 HYDRAULIC FUEL FILTER (2)
14 OIL PUMP
15 STARTER DRIVE PAD
16 FUEL-OIL HEAT EXCHANGER
17 MAIN GEAR BOX
18 MAIN FUEL CONTROL
19 MAIN FUEL PUMP
20 REDUCTION GEAR BOX
21 FUEL PUMP SUPPLY HOSE
22 FORWARD ENGINE MOUNT
23 BLEED AIR MANIFOLD
24 START BLEED DOOR (12)
25 IGV ACTUATOR (2)
26 OIL TANK
27 A/B FUEL SUPPLY HOSE
28 A/B FUEL CONTROL
29 A/B FUEL PUMP
30 A/B PUMP TURBINE
31 WINDMILL BYPASS VALVE
32 ENP TRANSDUCER
33 DIFFUSER CASE

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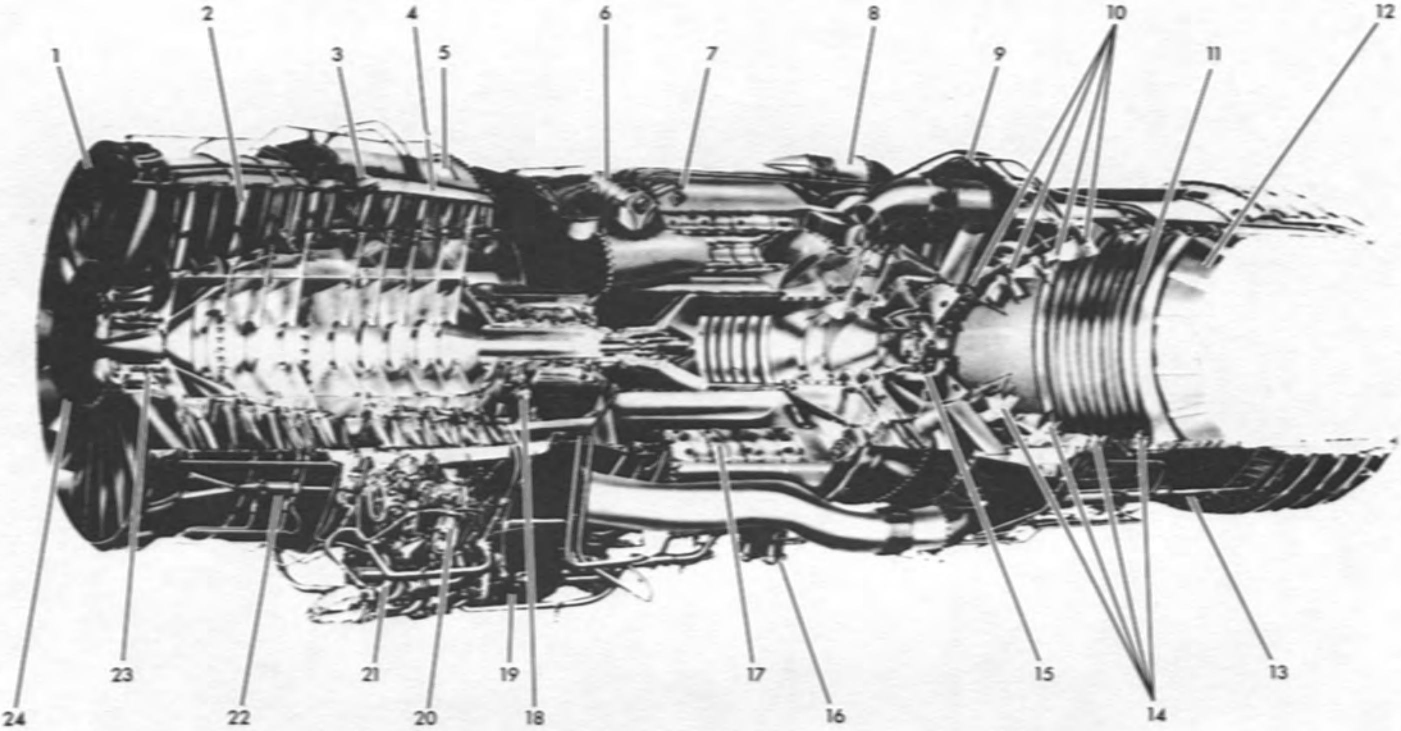
Figure 2-1. J58 Engine (Sheet 1 of 2)



1 INLET CASE
2 FORWARD COMPRESSOR SECTION (4 STAGES)
3 BLEED BYPASS DOORS (24)
4 BYPASS CHAMBER
5 START BLEED DOORS (12)
6 CHEMICAL IGNITION TANK (TEB)
7 MAIN BURNER INJECTOR PROBE
8 BLEED BYPASS TUBES (6)
9 AFT ENGINE MOUNT RING
10 AFTERBURNER SPRAY RINGS (4)
11 AFTERBURNER LINER
12 VARIABLE AREA EXHAUST NOZZLE
13 EXHAUST NOZZLE ACTUATORS (4)
14 FLAME HOLDERS (4)
15 TURBINE SECTION AND BEARING
16 HYDRAULIC FILTERS (2)
17 BURNER CAN (8)
18 AFT COMPRESSOR BEARING (DUPLEX BALL)
19 MAIN GEARBOX
20 MAIN FUEL CONTROL
21 MAIN FUEL PUMP
22 BYPASS BLEED DOOR ACTUATOR (4)
23 FRONT COMPRESSOR BEARING
24 INLET CASE ISLAND COVER

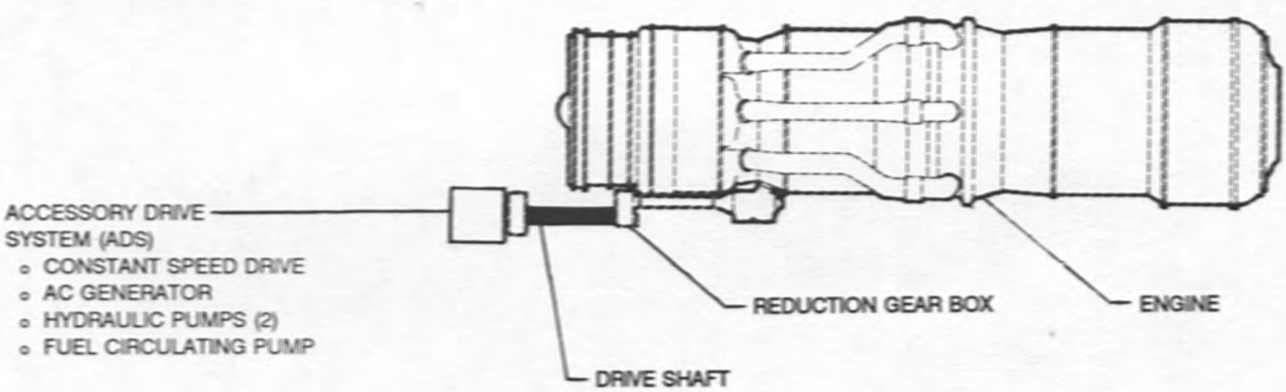


Figure 2-1. J-58 Engine (Sheet 2 of 2)

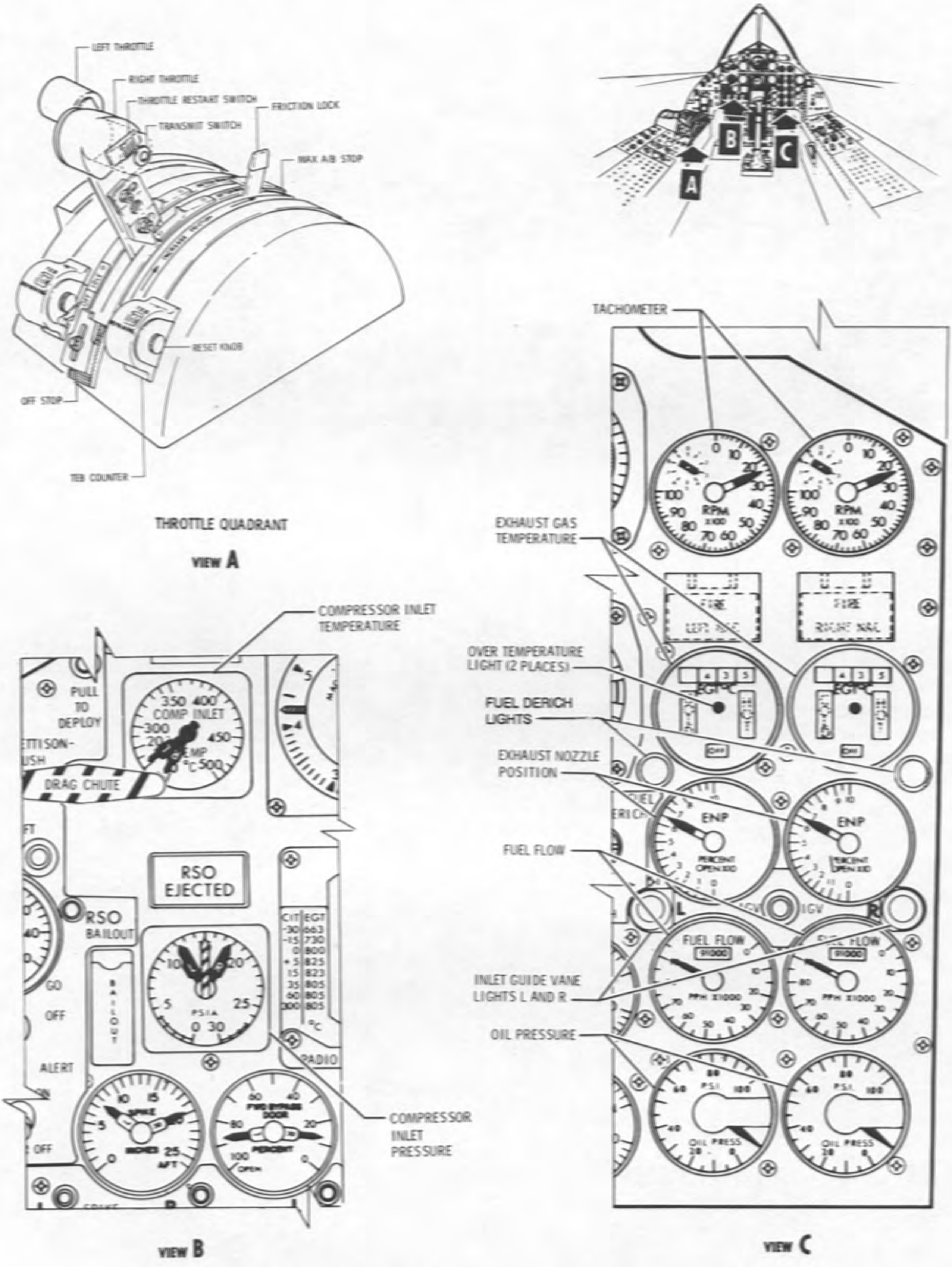


Figure 2-2. Engine Controls and Instruments

The diffuser section straightens and diffuses air from the compressor. It is the major structural unit of the engine, supporting the No. 2 bearing which accepts all the thrust and radial loads of the turbine shaft. The diffuser also supplies high pressure air for aircraft functions.

The combustion section consists of eight cylindrical combustion liners (cans), arranged in an annular configuration.

The two-stage turbine extracts power from fuel combustion and drives the turbine shaft and compressor blades.

The afterburner is enclosed in a convergent-divergent ejector nozzle, the aft portion of which is variable in area to regulate back pressure on the turbine and thereby maintain constant engine speed under all conditions scheduled by the main fuel control. Free-floating trailing edge flaps ("feathers") are attached to the aft end of the nozzle. These flaps open and close with changes in nozzle pressure, a function of Mach and engine thrust. The actuators that operate the variable inlet guide vanes, the bleed air system doors, and the variable-area exhaust nozzle, are hydraulically powered using fuel as the hydraulic fluid.

Two gearboxes, a main gearbox and a reduction gearbox, are mounted beneath the diffuser section. The gear boxes are mechanically linked to the compressor section. The Main gearbox accepts the external probes of an engine starter. The reduction gearbox provides mechanical power to an airframe-mounted Accessory Drive System (ADS). This system includes a constant speed drive and attached 60 KVA generator, two hydraulic pumps (to power the aircraft main hydraulic systems), and a fuel circulating pump (part of the fuel heat sink system).

Engine start is accomplished using air turbine-driven or engine-driven starters, probe-connected to the main gear box. During engine start, the Chemical Ignition System (CIS) injects hypergolic triethylborane (TEB) into the main and afterburner sections to achieve ignition of the high-flash-point JP-7 fuel. There is sufficient TEB for a minimum of 16 engine or afterburner starts for each engine. Engine control is from the engine main fuel control, mechanically linked to a throttle lever located on a quadrant on the pilot's forward left console. Three throttle positions are labeled, OFF, IDLE, and AFTERBURNER; with an unlabeled military power stop before AFTERBURNER. On the ground and at low Mach numbers, with throttle between IDLE and slightly below the military stop, engine speed varies with throttle position. At higher throttle settings, up to maximum afterburner, the main fuel control schedules engine speed as a function of CIT and modulates the variable-area exhaust nozzle to maintain essentially constant rpm. A manually-armed fuel derich system protects the engine against severe turbine over-temperature by automatically reducing the fuel/air ratio in the engine burner cans if exhaust gas temperature (EGT) reaches 860 degrees C. As shown in figure 2-2, engine instruments, except for CIT and compressor inlet pressure (CIP) indicators, are duplicated for each engine. Advisory lights for fuel derich and IGV are located adjacent to the engine instruments. Caution lights advising of low engine oil quantity are located on the pilot's annunciator panel.

2.3 HYDRAULIC SYSTEMS, LANDING GEAR, AND BRAKES.

2.3.1 Hydraulic Systems. Four separate hydraulic systems, each with its own pressurized reservoir and engine-driven hydraulic pump, supply power to hydraulically-actuated components in the aircraft. The normally independent systems are identified as A and B, and L and R. (See figure 2-3 for a schematic of the A and B hydraulic systems, figure 2-4 for the L and R systems.) The hydraulic pumps of the A and L systems are driven by the left engine Accessory Drive System (ADS); the B and R systems pumps are driven from the right engine ADS. The R system hydraulic reservoir is serviced with 4.5 gallons of hydraulic fluid, the other reservoirs with 2.8 gallons of fluid. The hydraulic fluid is cooled in oil/fuel heat exchangers that use the aircraft fuel supply for cooling. The hydraulic systems operate at a normal pressure of 3350 psig. Hydraulic fluid meets MIL-H-27601A and is specially formulated for the high-temperature requirements of the aircraft.

The A and B hydraulic systems operate in parallel to power the servos and actuators which move the elevon and rudder flight control surfaces, with each system supplying power for operation of one-half of the actuating cylinders. The A system also supplies hydraulic power to the stick pusher, part of the Automatic Pitch Warning (APW) System, which is part of DAFICS.

The L hydraulic system supplies power to operate the left engine air inlet and air bypass controls, normal brake system, landing gear, main gear inboard doors, normal nose wheel steering system, and aerial refueling door latch actuator.

The R hydraulic system supplies power to operate the right engine air inlet and air bypass controls, the alternate brake system, and the alternate nose wheel steering system. The R system also supplies power to retract the landing gear and close the main gear inboard doors if the L system fails prior to or during gear retraction.

As shown in figure 2-5, hydraulic system pressure indicators are located on the pilot's instrument panel, and hydraulic system caution lights, which advise of low fluid pressure or quantity, are located on the pilot's center stand.

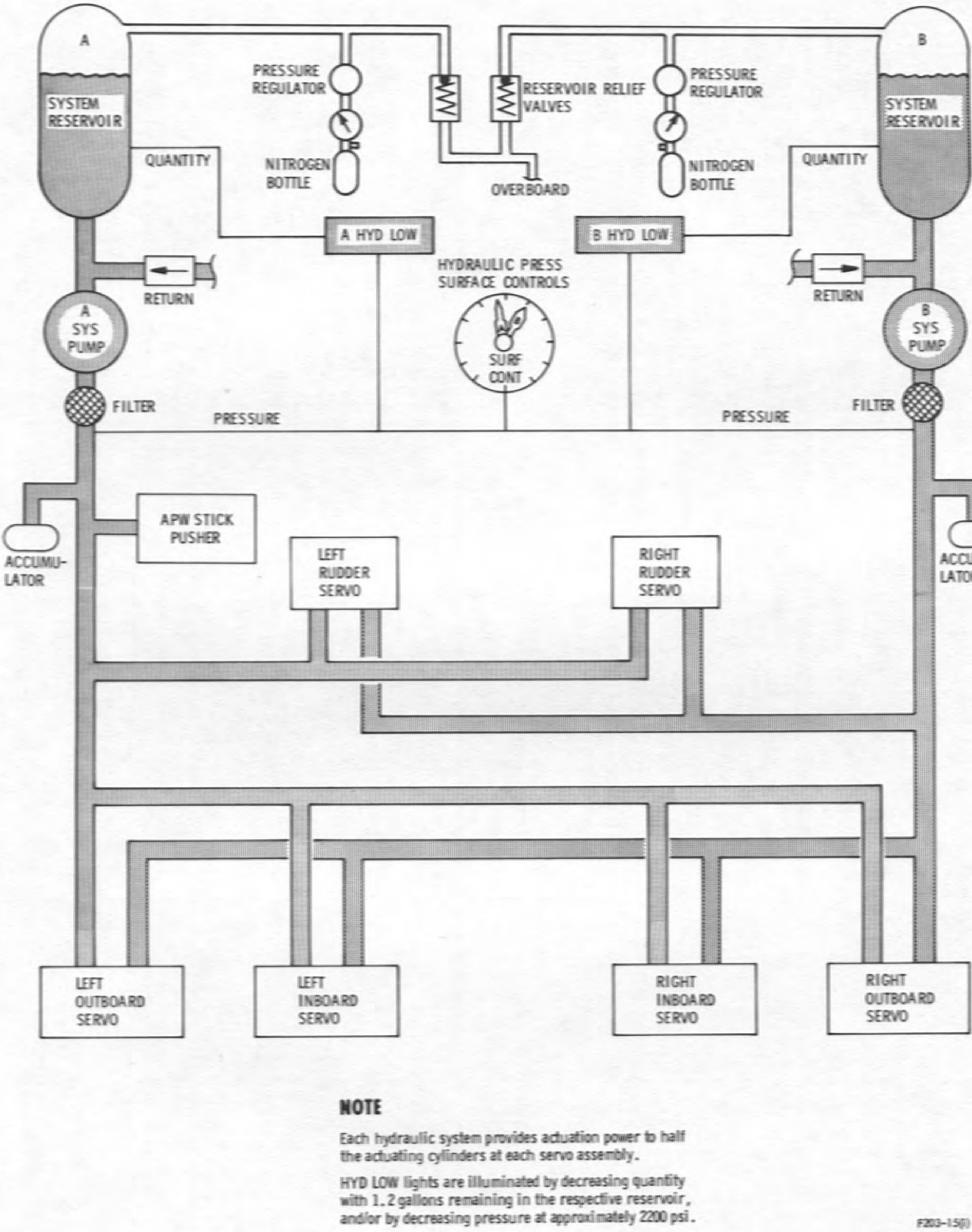
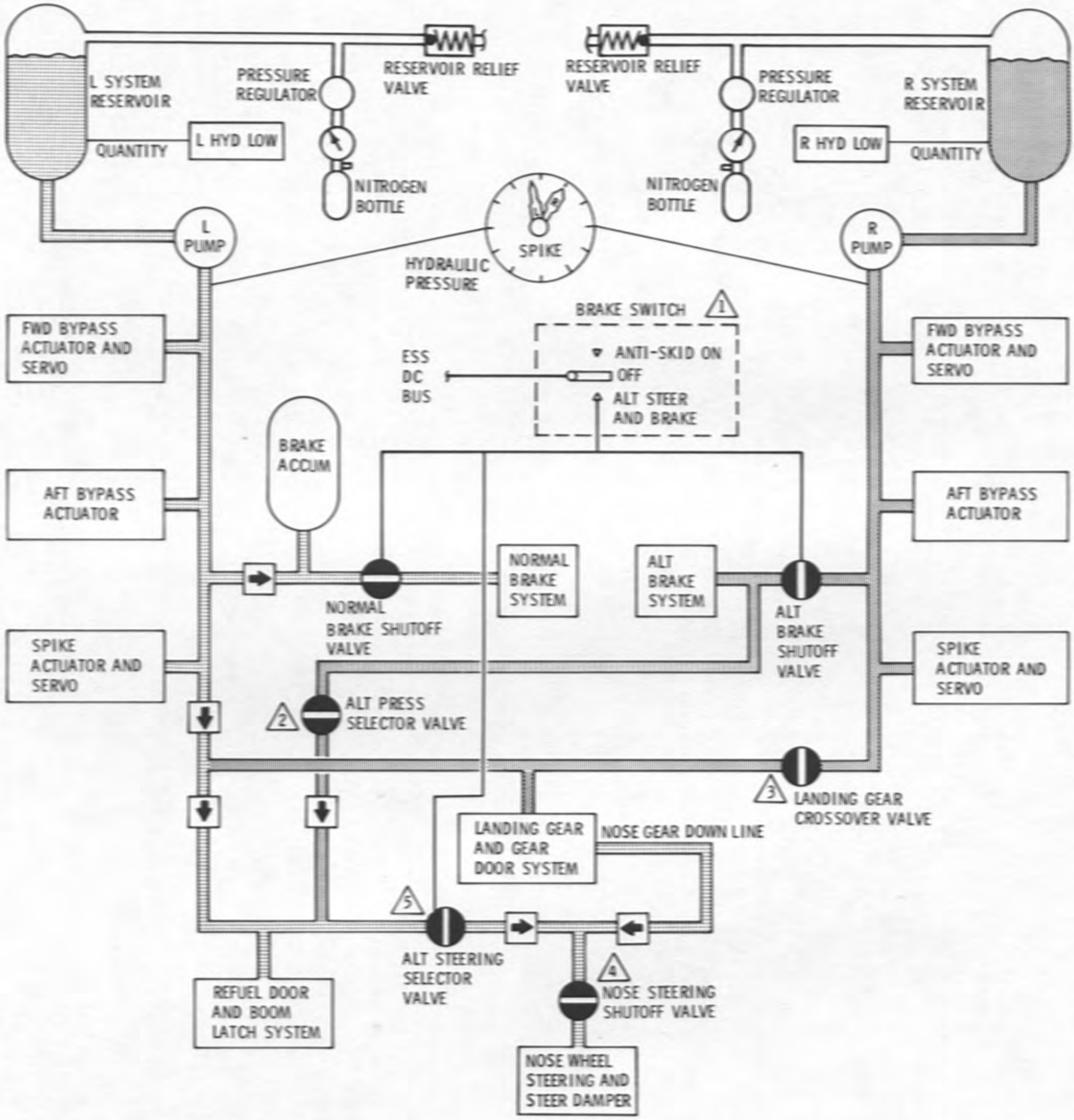


Figure 2-3. A and B Hydraulic Systems Schematic



NOTE

1. OFF: "L" system powers brakes, no anti-skid protection.
   ANTI-SKID ON: Provides anti-skid protection on "Normal" brake system.
   ALT STEER AND BRAKE: Closes "Normal" and opens "Alt" brake shutoff valves, arms alternate system selector valves, and energizes "Alt. Anti-skid" system
2. With brake switch in "Alt Steer and Brake" position, valve is opened if "L" system pressure decreases below 2200 psi.
3. Crossover valve opens automatically if "L" system pressure decreases below 2200 psi, but only for gear retraction.
4. Steering controlled by CSC/NWS switch on control stick to provide nose steering on normal or alternate system pressure.
5. Valve opens when alternate steering and braking selected regardless of pressure in the L system

F203-16(b)

Figure 2-4. L and R Hydraulic Systems Schematic

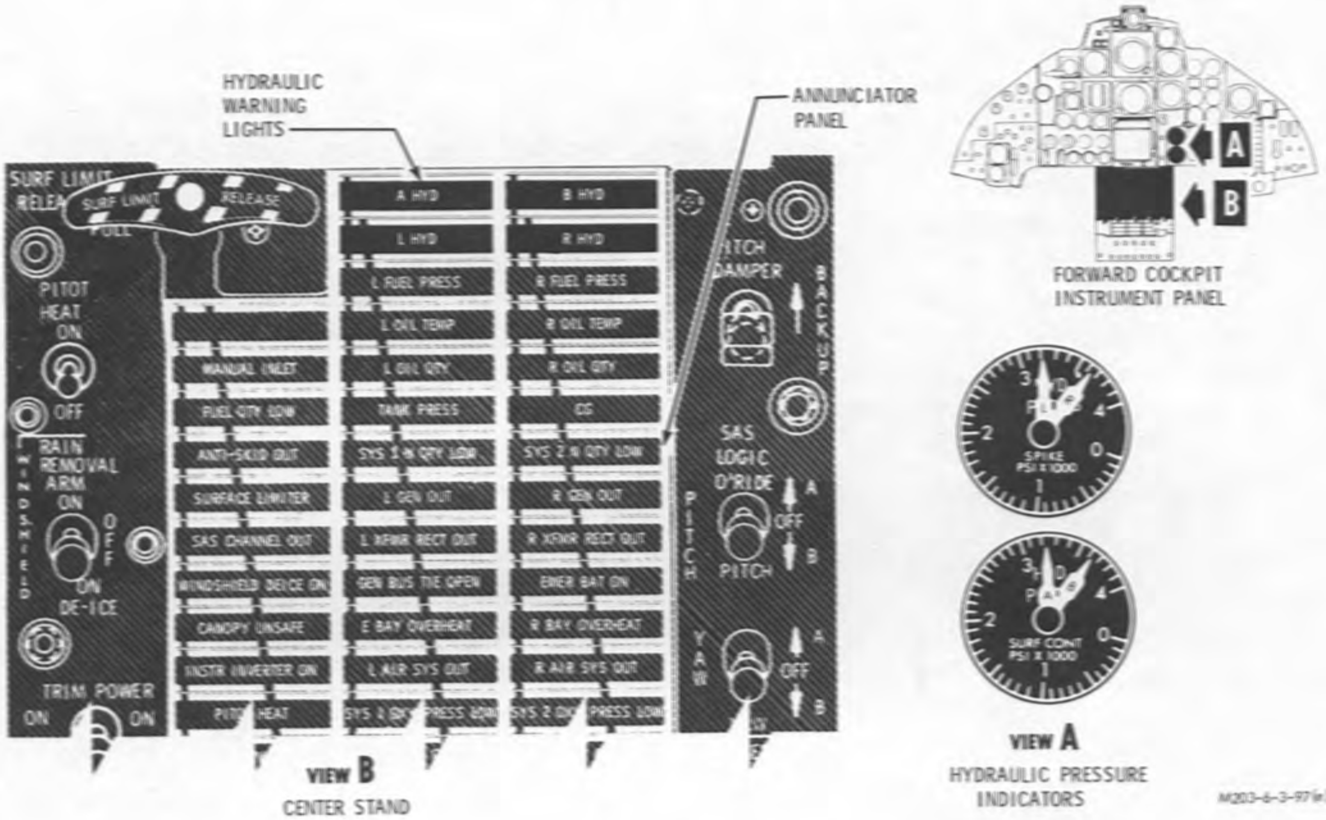


Figure 2-5. Hydraulic Systems Indicators and Warning Lights

2.3.2 Landing Gear. The tricycle landing gear and main gear inboard doors are electrically controlled and hydraulically actuated. The main gear outboard doors and the nose gear doors are mechanically linked to their respective struts. Gear control is achieved by operation of a handle on the pilot's instrument panel. During gear retraction, each of the three-wheeled main gear retracts inboard into the inner wing and fuselage, and the dual-wheeled nose gear retracts forward into the fuselage. The main gear are locked in the up position by the inboard main gear doors, which are mechanically latched in the up position. The nose gear is locked in the up position by an uplock which engages the strut. Hydraulic pressure, for normal gear operation and with the gear extended, is from the L hydraulic system. Should L system hydraulic pressure drop below 2200 psi, the R hydraulic system applies the required pressure. A cable-controlled gear release system, controlled by a handle on the pilot's center stand, permits free fall of the gear if the L hydraulic system fails or electrical gear control circuits become inoperative. Normal gear retraction and extension time is from 12 to 16 seconds. The aircraft has nose-steering capability, with the nose gear controlled by the pilot's rudder pedals.

2.3.3 Brakes. The aircraft has hydraulically operated brakes on the main landing gear wheels, controlled by toe-action of the pilot's rudder pedals. Artificial feel is provided. The brakes are multiple stator and rotor assemblies, installed on each main landing gear wheel. There are two interrelated brake systems, a normal system using L hydraulic system power and an alternate system using R hydraulic system power. Brake system selection is made with a switch on the pilot's instrument panel. The brake system is augmented with an antiskid system, featuring dry/wet runway condition selection, and a strut-damper system that dampens fore and aft main landing gear strut oscillation.

2.4 FUEL SYSTEM.

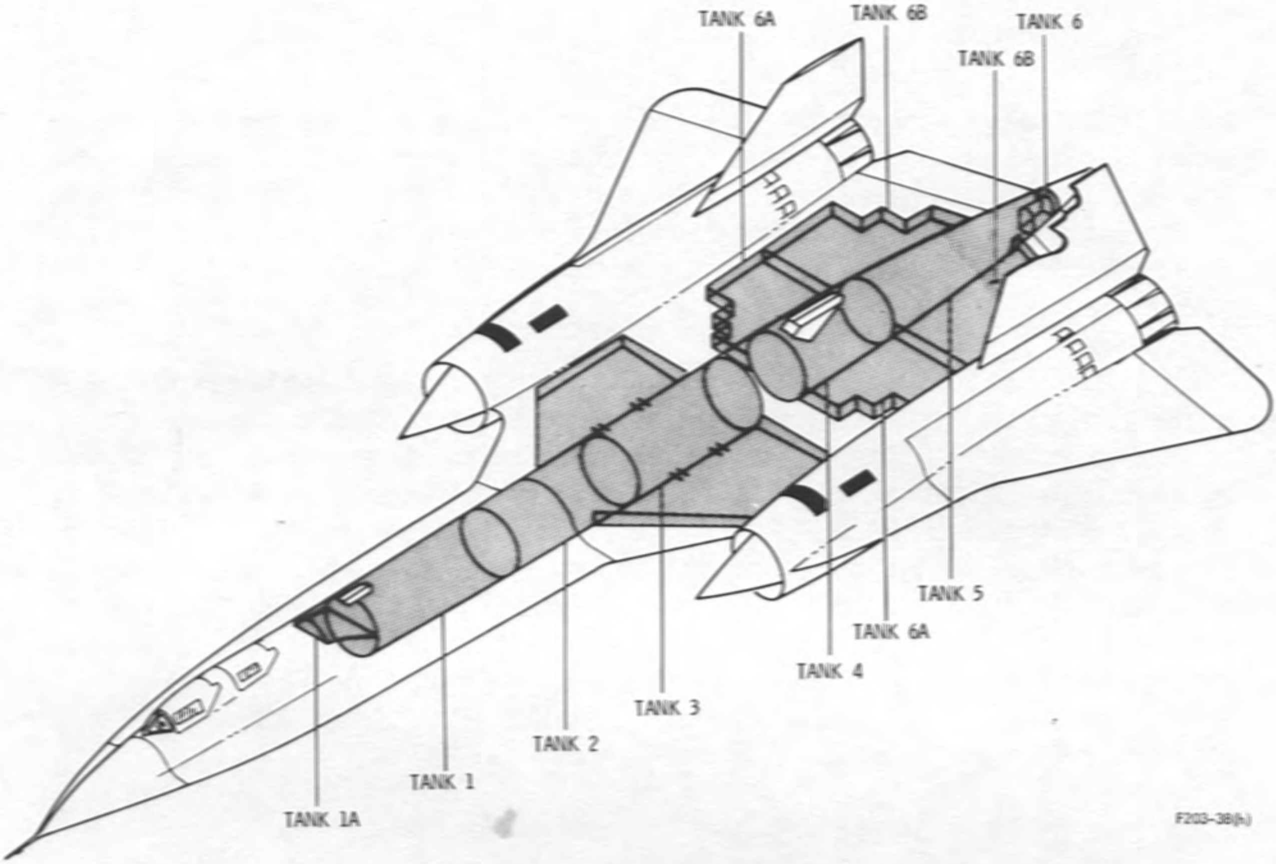
The fuel system consists of fuselage and wing fuel tanks, manifolds, boost pumps, valves, control circuitry, and other components. The system supplies the engines with fuel under all flight conditions. The system also acts as a heat sink that cools engine oil, Accessory Drive System (ADS) oil, hydraulic systems oil, and through operation of the Environmental Control System (ECS), cockpit and equipment compartment air. Additionally, the system assists in maintaining aircraft c.g. through fuel transfer between tanks. The system has the following capabilities and functions:

- Automatic sequence of fuel feed from tanks to engines to optimize aircraft c.g. for supersonic cruise
- Manual tank (pump) selection
- Automatic aft transfer of fuel for c.g. control
- Manual forward/aft transfer of fuel for c.g. control
- Manual control of fuel overboard dump
- Manual control of fuel shutoff to either or both engines
- Manual selection of crossfeed to interconnect left and right fuel feed manifolds
- Refueling through the ground refueling receptacle or the aerial refueling receptacle
- Defueling any or all tanks
- Filters fuel supplied to the engines
- Supplies fuel as a coolant to heat exchangers and other components.

2.4.1 Fuel Tanks. Fuel is stored in integral, sealed tanks. There are five fuselage tanks, identified as 1A, 1, 2, 4, and 5; a combined fuselage/wing tank, tank 3; and a wing tank group, divided into two separate tanks, tanks 6A and 6B. Fuel tank arrangement and capacities are shown in figure 2-6. All fuselage tanks are interconnected by a single vent line, with the wing tanks vented to the nearest fuselage tank.

2.4.2 Fuel Boost Pumps. Fuel boost pumps supply fuel to the engines and heat exchangers through left and right manifolds and are also used to transfer fuel for center of gravity control. Submerged, fuel-cooled, single-stage, centrifugal, ac-powered boost pumps are installed only in the fuselage tanks, with the boost pumps for wing tanks 6A and 6B located in a small fuselage sump tank, tank 6. There are sixteen pumps, with four each located in tanks 1 and 4, and two each in tanks 2, 3, 5, and 6. For reliability, half of the pumps are powered from the left generator ac bus, the other half from the right generator ac bus. At lower altitudes, two operating pumps are required for maximum afterburner fuel flow; however, if either pump of a pair fails, the other pump is capable of supplying sufficient pressure for continued engine operation at reduced afterburner thrust.

2.4.3 Fuel Feed and Sequencing. The left and right fuel feed manifolds supply fuel to the left and right engines respectively. (See figure 2-7.) Each manifold has a pilot-controlled fuel dump valve. A manually-controlled crossfeed valve permits connection of the manifolds. The left engine is normally supplied with fuel from tanks 1, 2, 3, and 4. The right engine is normally fed from tanks 1, 4, 5, and 6. Although crossfeed can be used to feed either engine from any tank, there is a normal automatic fuel tank sequencing schedule, with sequencing controlled by float switches. These switches function so as to maintain a flow of fuel to the manifolds through sequential boost pump operation in the appropriate tank(s) and also, as part of the automatic c.g. control system, to optimize aircraft center of gravity for supersonic cruise. The automatic sequence of fuel feed is essentially: first from tanks 1, 3, and 6; then from tanks 1, 3, and 5; then tanks 3 and 5; then 2 and 5; then tank 4. A stack of float switches in tank 1, called right-hand shutoff (RHSO) switches, are preset before flight to control the level at which fuel flow from tank 1 is reduced in order to limit the rapid aft shift in aircraft c.g. that occurs when tank 1 is supplying fuel at full flow, as it does from the beginning of a flight. The setting of these switches, accomplished remotely from the fuel tank during preflight operations, permits the automatic c.g. control system to compensate for variations in aircraft weight and c.g. due to payload loading, and for different mission profiles. Manual selection of tank (pump) operation supplements but does not terminate automatic tank sequencing and it does change the programmed c.g. schedule.



FUEL TANK CAPACITIES
Normal Flight Attitude

Tank	Fuel/Gal	Fuel (JP-7)
1A	251.1	1650 lb.
1	2095.9	13770 lb.
2	1974.1	12970 lb.
3	2459.7	16160 lb.
4	1453.6	9550 lb.
5	1758.0	11550 lb.
6A (forward)	1158.3	7610 lb.
6B (Aft)	1068.5	7020 lb.
Total	12219.2	80280 lb. *

* At average fuel density of 6.57 lb./gal.
(46.2° API, Fuel temperature = 78°F)

Figure 2-6. Fuel Tank Arrangement and Capacities

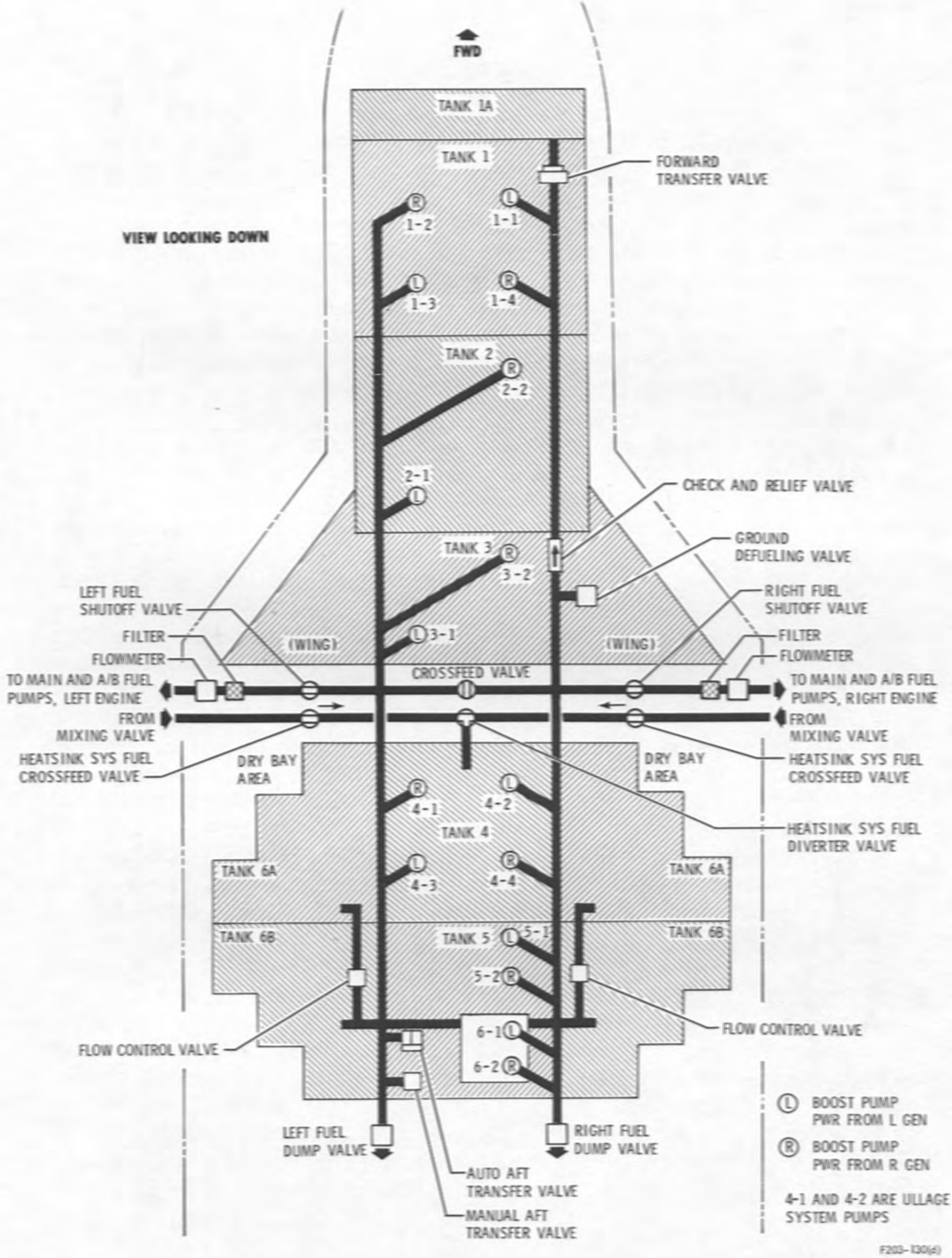


Figure 2-7. Fuel Feed System

2.4.4 Fuel Transfer. Fuel transfer systems are used to control aircraft c.g. There are automatic and manual aft transfer systems and a manual forward transfer system. The automatic aft transfer system transfers fuel from the left fuel manifold into tank 5, based on tank 2 having fuel above the low level float switch and tank 5 pumps operating. Automatic aft transfer rates are 65 pounds per minute, with both throttles in afterburner, and 23 pounds per minute, otherwise. Under certain conditions, to optimize c.g., the automatic aft transfer rate increases by 233 pounds per minute. Manual (pilot-controlled) aft transfer, which does not terminate automatic aft transfer, is at a rate of 233 pounds per minute. Manual (pilot-controlled) forward transfer moves fuel forward into tank 1 from tanks 4, 5, and 6 through the right fuel feed manifold. Fuel forward transfer rate is 950 pounds per minute.

2.4.5 Fuel Heat Sink System. The purpose of the heat sink system is to supply fuel as a coolant to various systems and components. The heat sink system is comprised of two identical systems, a left system and a right system. As shown in figure 2-8, fuel is supplied to heat changers for the air conditioning (environmental control) system; aircraft A, B, L, and R hydraulic systems oil; engine oil; ADS oil; and directly to cool the engine windmill and bypass valve, TEB tank, and control lines that actuate the afterburner nozzle. Fuel is also supplied as a coolant to pitch and yaw SAS gyros located in the bottom of tank 2. The fuel-to-fuel heat exchanger in tank 3 is common to both left and right heat exchanger systems. It equalizes fuel temperatures between the systems, which assists in stabilizing the operation of the dual (left and right) air conditioning systems. The mixing valve mixes return fuel from the heat sink loops. The temperature control valve directs return fuel either to the engine fuel line, at fuel temperatures below 270 degrees F, or back to the fuel tanks, through a crossover valve and the diverter valve, at temperatures over 300 degrees F. The crossover valve opens, in the event of engine failure or shutdown, to allow fuel flowing towards the inoperable engine to return to the tank and not stagnate. Whenever pressure in the crossfeed lines exceed sensed fuel manifold pressure, the diverter valve opens to pass the hot fuel back to tank 4 or, if tank 4 is full, to the refueling manifold to other tanks.

2.4.6 Fuel System Indicators and Controls. Fuel system indicators and controls are located on the pilot's instrument panel as shown in figure 2-9. They consist of a fuel quantity indicator and selector switch and a c.g. indicator (all duplicated in the aft cockpit), manual fuel tank select switches, fuel aft transfer and forward transfer switches, a fuel dump switch, and left and right emergency fuel shutoff switches. An air refueling control switch provides for automatic or manual control of aerial refueling (aerial refueling door opening/closing and tanker boom nozzle latching/ disconnect).

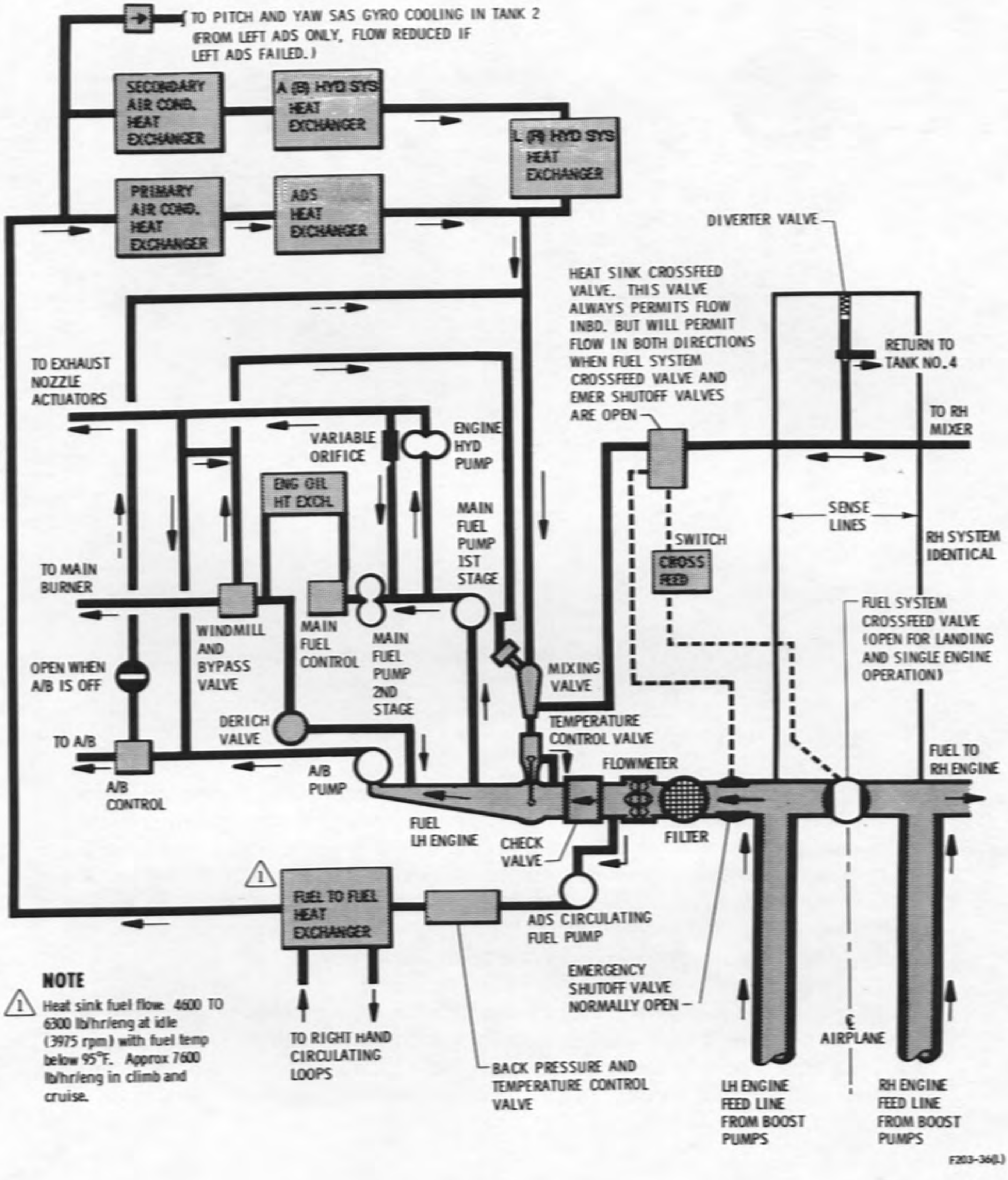


Figure 2-8. Fuel Heat Sink System

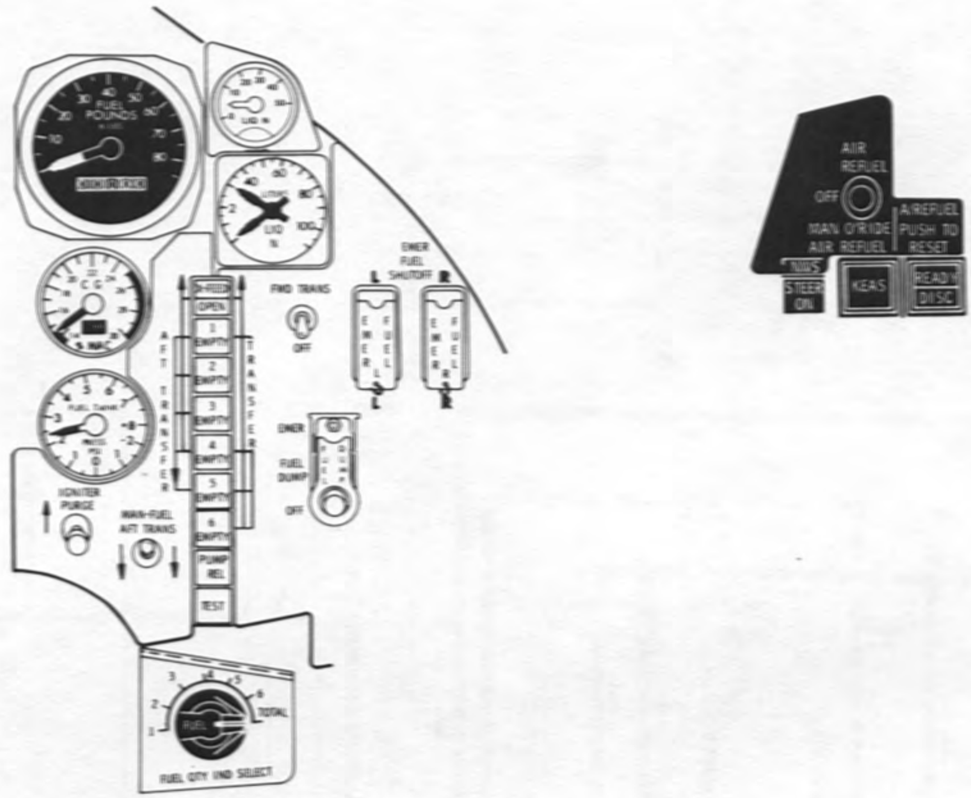


Figure 2-9. Fuel System Indicators and Controls

2.5 NITROGEN SYSTEM.

The purpose of the nitrogen system is to supply nitrogen gas to pressurize all fuel tanks to 1.5 (± 0.25) psid (differential pressure above ambient atmospheric pressure) and to inert the ullage space above the heated fuel to prevent autogenous ignition. The system also supplies gaseous nitrogen to inert the oil in the engine-powered Accessory Drive System (ADS). Figure 2-10 is a schematic of nitrogen system components and related pressure lines. Figure 2-11 shows system cockpit indicators and the location of the pressure lines and related components.

A small quantity of nitrogen is used during taxi, runup, and takeoff. After takeoff, little or no nitrogen is used until descent for landing or refueling. During descent, increasing atmospheric pressure causes a demand on the nitrogen system in order to keep fuel tank internal pressure higher than the increasing external pressure.

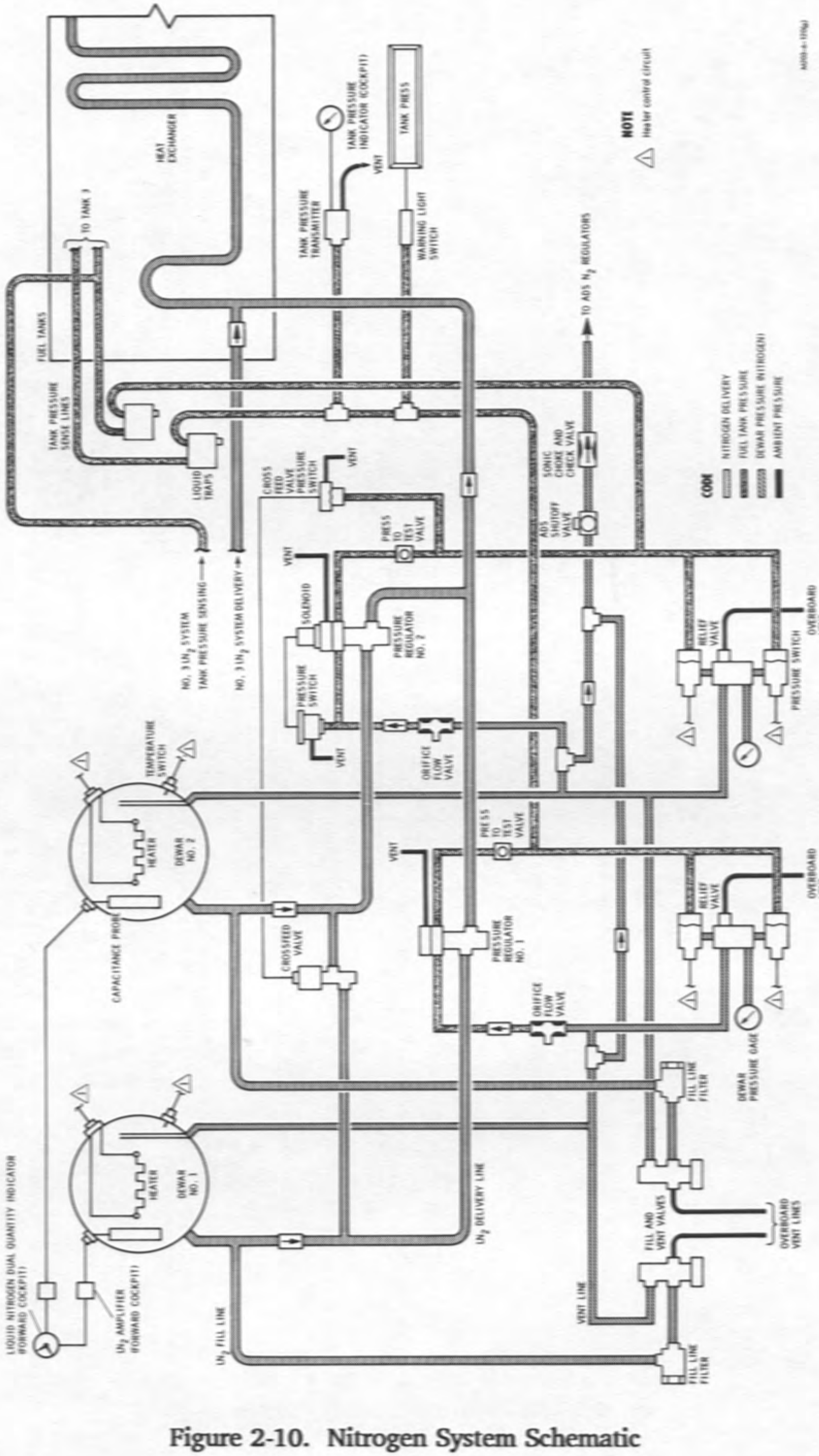
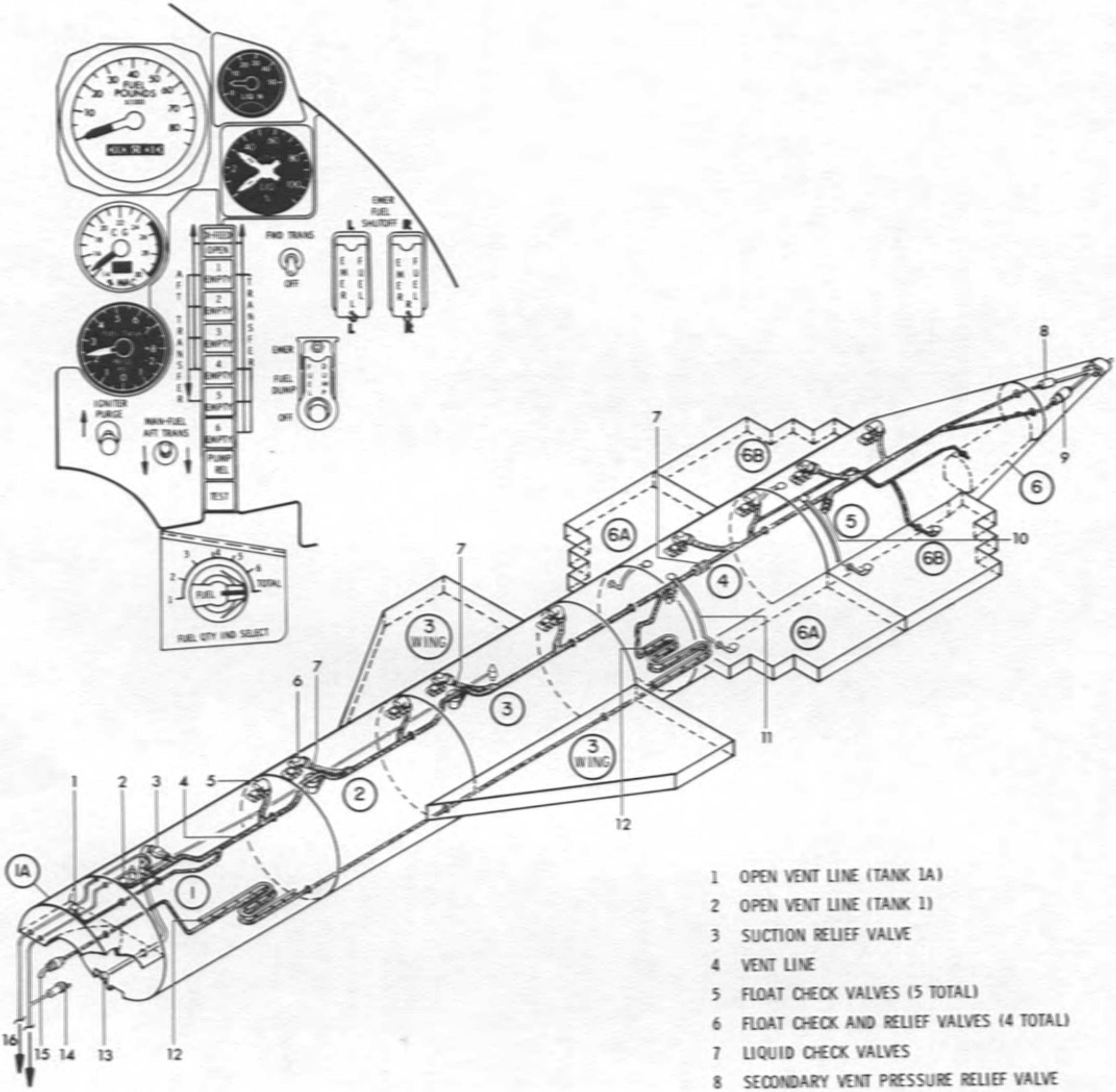


Figure 2-10. Nitrogen System Schematic



1 OPEN VENT LINE (TANK 1A)
2 OPEN VENT LINE (TANK 1)
3 SUCTION RELIEF VALVE
4 VENT LINE
5 FLOAT CHECK VALVES (5 TOTAL)
6 FLOAT CHECK AND RELIEF VALVES (4 TOTAL)
7 LIQUID CHECK VALVES
8 SECONDARY VENT PRESSURE RELIEF VALVE
9 PRIMARY VENT PRESSURE RELIEF VALVE
10 VENT DRAIN VALVE
11 VENT LINE, WING TANK TO TOP OF FUSELAGE TANK (4 TOTAL)
12 FUEL LINE FROM FUEL PUMP OUTLETS
13 SUCTION RELIEF ORIFICE (NOSE WHEEL WELL)
14 TANK PRESSURE TRANSMITTER
15 LN_2 FLOW FROM DEWAR TANKS
16 TO NITROGEN TANK PRESSURE SENSORS

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Figure 2-11. Nitrogen System Indicators and Pressure and Sensing Lines

Nitrogen system components include two 105-liter Dewar flasks, pressure regulators, pressure lines, various valves, fuel-to-nitrogen heat exchangers, and other components. The 105-liter Dewars and related components are functionally arranged into two independent systems, each capable of supplying the required flow of nitrogen. Additionally, in SR-71A aircraft, there is a third, 50-liter Dewar supplemental system.

The 105-liter Dewar flasks, for systems 1 and 2, are located overhead in the nose wheel well. The 50-liter Dewar, for system 3 in SR-71A aircraft, is located in the B2-bay, in the left chine (refer to Section III, figure 3-2). The Dewars have pressure and temperature controlled ac-powered heaters to assist in the production of nitrogen gas. Gaseous nitrogen, from standpipes in the Dewars, is supplied to the ADS units through a shutoff valve and a sonic choke and check valve. The much greater quantity of nitrogen required for fuel tank inerting is also supplied by the Dewars. Liquid nitrogen is supplied, through pressure regulators, to twisted-tube fuel-to-nitrogen heat exchangers in fuel tanks 1 and 4. The twisted-tube design causes maximum impingement of the liquid nitrogen on the walls of the tubes, efficiently converting the liquid nitrogen to a gas. The nitrogen gas is ported to the common fuel vent line that interconnects all tanks.

Fuel tank pressure is sensed in tank 3, with sensing lines extending forward to pressure regulators for systems 1, 2, and 3. Orifice flow valves, connected to the gaseous nitrogen lines of systems 1 and 2, allow a small quantity of nitrogen gas to flow through the sensing lines, purging them of fuel and moisture. Fuel tank pressure below 1.5 (± 0.25) psid causes the pressure regulators to supply a regulated flow of liquid nitrogen. Fuel tank pressure is prevented from falling below 1.0 (± 0.25) psid by the action of the No. 2 regulator, which opens to supply unregulated liquid nitrogen until tank pressure builds up to 1.75 psid. The regulator resumes normal control of liquid nitrogen when pressure falls to the 1.5 psid level. A normally-closed pressure-regulated crossfeed valve opens to allow Dewars No. 1 and 2 to supply a single pressure regulator in the event the other pressure regulator fails closed. If tank pressure exceeds 1.5 (± 0.25) psid, all pressure regulators close, preventing a flow of liquid nitrogen until tank pressure returns to that limit. During aircraft ascent, pressure relief valves in the tail cone open to vent fuel tank pressure when the pressure differential exceeds 3.25 (± 0.25) psi. During supersonic cruise, the fuel warms up and fuel vapor pressure tends to maintain the 1.5 psid level.

There are no cockpit controls for the nitrogen system. Cockpit indicators are shown in figure 2-11 and consist of a fuel tank pressure indicator, a dual LN_2 quantity indicator, for systems 1 and 2, and a separate quantity indicator for system 3. A caution light (see figure 2-10), on the pilot's annunciator panel, comes on if tank pressure falls below 0.25 (± 0.10) psid.

2.6 ENVIRONMENTAL CONTROL SYSTEM (ECS).

The SR-71 operates under extremely adverse environmental conditions where ram air temperatures can exceed 400 degrees C and ambient static air pressure can be less than 1/3 psia. In addition to the aircraft's black color, which radiates heat away from the vehicle, and special insulating materials, which minimize temperature buildup in sensitive areas, the Environmental Control System (ECS) maintains the cockpits, equipment areas, and heat-sensitive critical components within safe operating temperatures. The environmental control system is comprised of three subsystems: an air cycle and air distribution system, a temperature control system, and a pressurization system. The ECS performs the following functions:

- Air conditions and pressurizes both cockpits.
- Supplies cooling air to payload equipment bays and compartments, detachable nose section, and nose wheel well.
- Supplies cooling air to DAFICS pressure transducer assembly (PTA), in B1-bay, and to liquid oxygen converters and LN_2 Dewar, in B2-bay.
- Supplies cooling air to Astroinertial Navigation System (ANS) guidance group, in the AC-bay.
- Provides pressure suit ventilation air.
- Provides windshield defog air.
- Provides left windshield hot deicing air.
- Provides canopy seal pressurization.

2.6.1 Air Cycle and Air Distribution System. The air cycle and air distribution system develops required hot and cold air supplies, and controls and distributes this air to the cockpits, equipment bays and compartments, and various components. Due to the operating environment, most air lines and ducts are insulated.

Through operation of the heat sink system, aircraft fuel functions as the major heat sink for the air cycle systems. The air cycle systems, two identical left and right systems, are functionally in two parts; a hot air system and a cooling air system. The hot air system supplies air, partially cooled but still hot, to engine inlet system components and to the cooling compressor/turbine. The partially cooled air is also used for canopy seal pressurization and, when selected, for defog or deicing of the pilot's windshield. From the cooling turbine, the cooling air system distributes air to the cockpits, equipment compartments, and bay areas.

Figure 2-12 is a schematic of the hot air system. As shown, high pressure engine bleed air, from the ninth compressor stage, is initially cooled in an air-to-air heat exchanger then further cooled in the primary air-to-fuel heat exchanger, located in the heat sink package. At low altitudes, the temperature of the air exiting the package is pressure and temperature controlled by a hot air bypass system composed of a thermostat, valves, and other components. Pressure and temperature control are deactivated above 44,000 feet in order to utilize all available air for cooling. The pressure regulator and shutoff valve in the heat sink package regulates downstream air pressure when the air cycle system is in operation and closes, shutting off air flow and shutting down the system, when the pilot-controlled refrigeration switch is set to OFF or, automatically, if excessive air temperatures occur at the air inputs to the compressor/turbine. The partially cooled hot air from the heat sink package is supplied to the inlet system duct pressure ratio transducer (DPRT) and forward bypass door actuator linear variable differential transducer (LVDT) and to a filter which removes particles that could damage system components downstream.

Components shown in figure 2-13 are located in the air conditioning bay, just aft of the aft cockpit. As shown in the figure, filtered hot air is supplied to the windshield deicing system and is used for canopy seal pressure before it is applied to an air control valve. Regulated air from the air control valve is supplied to the hot air manifold, which, in turn, supplies air to the cockpit temperature valve, to the windshield defog valve, and to the cooling compressor/turbine. The compressor/turbine consists of a compressor section and an expansion turbine coupled by a common shaft. The unit is powered by the pressure of the input air and the expansion of the compressed air. Air, from the hot air manifold, is compressed then ducted externally to a secondary air-to-fuel heat exchanger. Partially cooled air from the heat exchanger is ducted back to the cooling turbine which drives the compressor section and allows the input compressed air to expand and cool further.

Air from the compressor/turbine is applied to the cold air manifold. A manifold temperature control system maintains the air in the cold air manifold at temperatures that prevent icing at lower altitudes and ensures a supply of constant-temperature air at high altitudes. As shown in figure 2-13, the cold air manifold supplies cooling air to the ANS guidance group, viewsight, E-bay, R-bay, and, through bay air shutoff valves, to the payload equipment bays. The cold air manifold also supplies cooling air to the cockpits and crew pressure suits. Exhaust air from the cockpits is used to cool the B1 and B2-bays in the forward left chine, the C-bay, the nose wheel well, and, through a shutoff valve in parallel with the payload equipment bays shutoff valves, the nose section. A ground air connector in the nose wheel well is part of the cold air manifold. The connector provides for the application of ground cooling air to areas and equipment connected to the manifold when cooling air is needed for ground operations. Water separators are provided to remove moisture from the cooling air before it is applied to the ANS guidance group and to electrical/electronic equipment in the payload equipment bays.

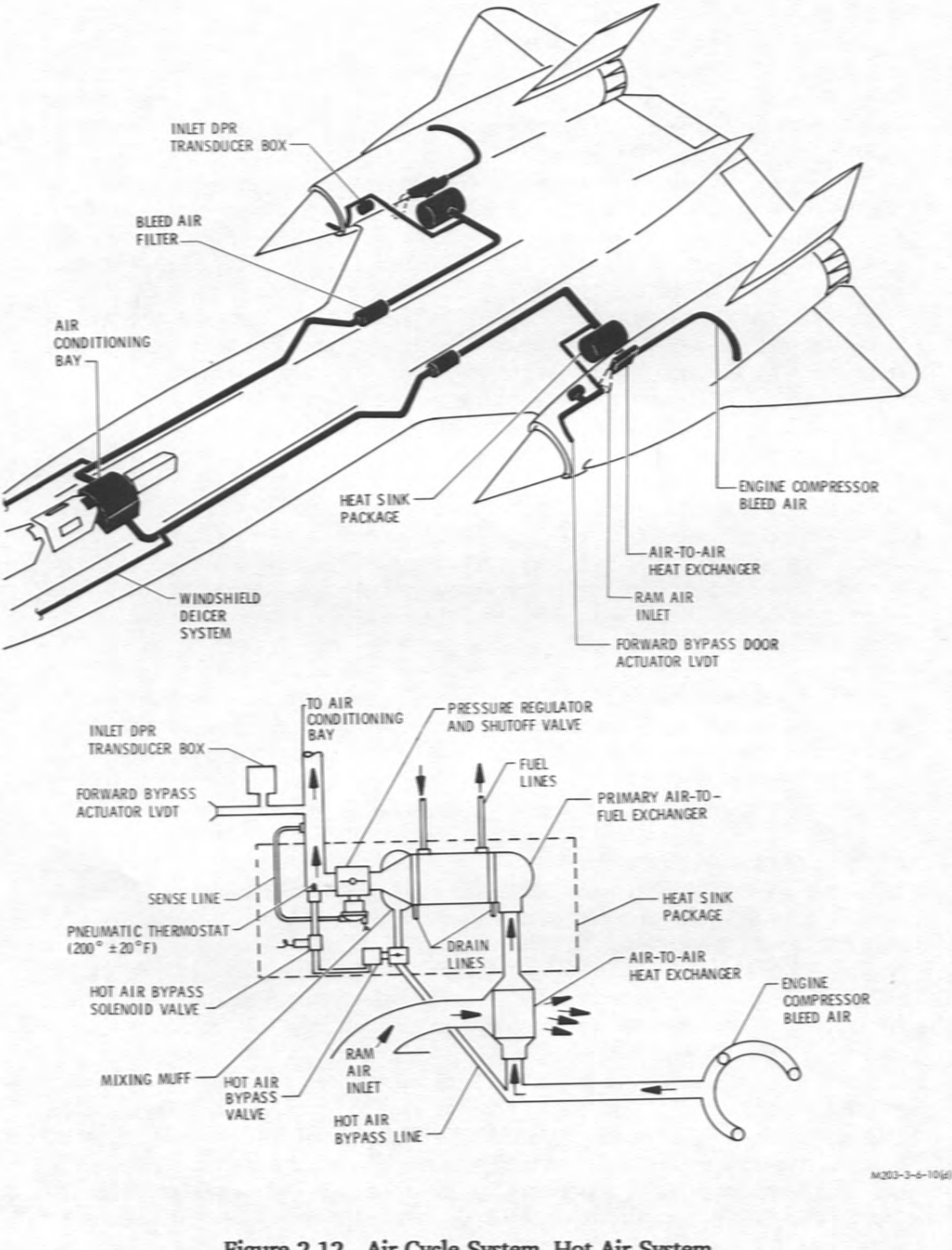


Figure 2-12. Air Cycle System, Hot Air System

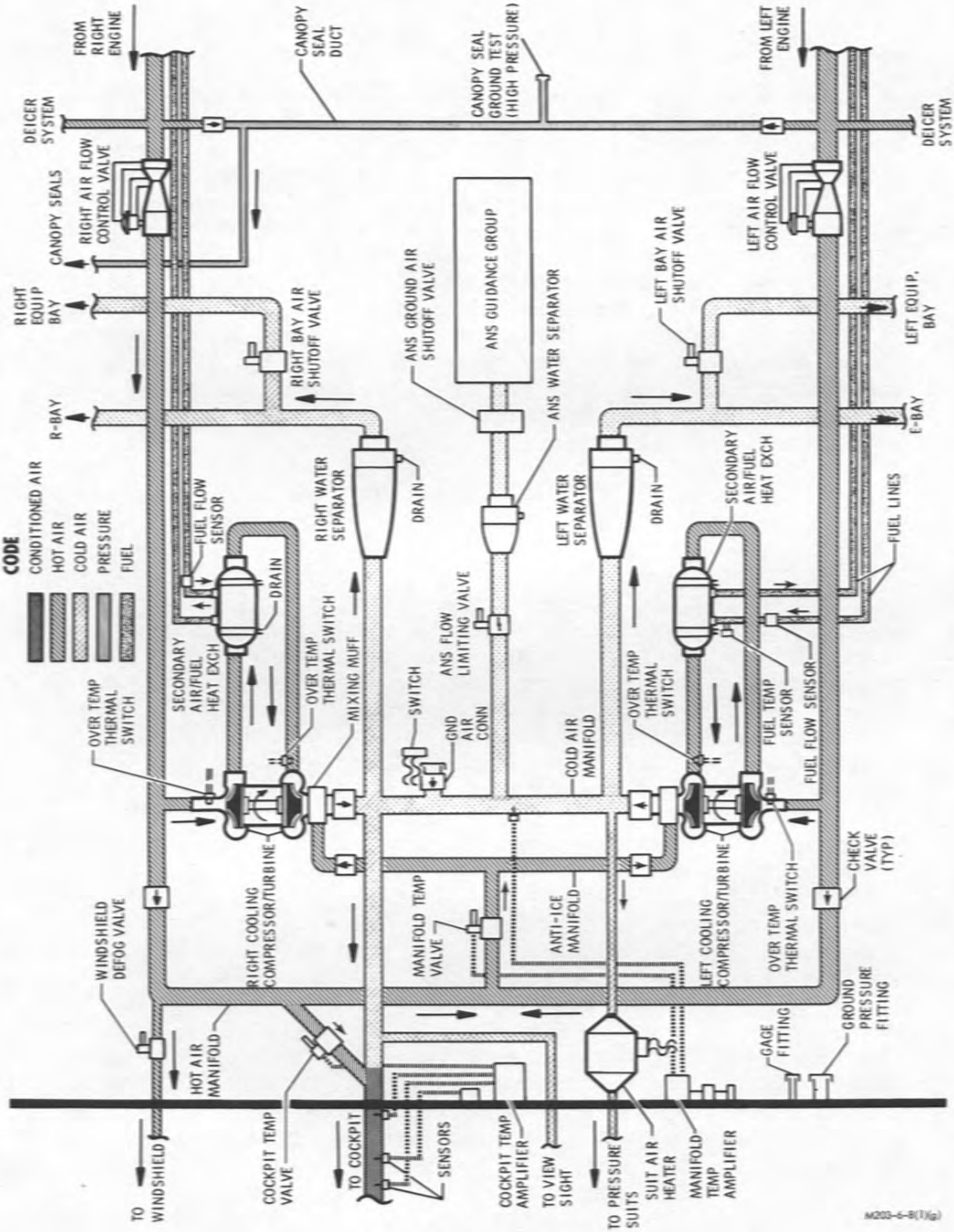


Figure 2-13. Air Conditioning Bay Air Flow Schematic

Manual controls for the air cycle and distribution system are located on the pilot's instrument panel. (See figure 2-14.) The L and R REFRIG switches control operation of the pressure regulator and shutoff valve in the respective heat sink packages and are normally, except during ground operations, retained in the ON position, permitting normal operation of the systems. A BAY AIR switch controls cooling air to the payload equipment bays and to the nose, through operation of air shutoff valves. The switch, normally set to ON, is set to OFF to shut off equipment bay and nose air in the event of air cycle system or engine failure. This conserves cooling air for the cockpits, crew pressure suits, and other flight-critical areas (E-bay, R-bay, and ANS guidance group). A DEFOG switch controls a windshield defog valve which injects hot air, from the hot air manifold, into the windshield cooling air manifold. A DE-ICE switch controls operation of the deice valve which supplies hot air to the pilot's left windshield de-ice manifold.

Figure 2-15 is a schematic of the air cycle and air distribution system. Detailed information on radar and OBC noses relate to prior use of the aircraft and are for reference purposes only.

2.6.2 Temperature Control Systems. The temperature control systems control the temperature of air supplied to various areas and equipment. The four subsystems are: hot air bypass system, cockpit temperature system, manifold temperature system, and pressure suit ventilating air temperature system. The hot air bypass system is automatic, with no manual control. The remaining three systems are pilot controlled. Controls and indicators for the temperature control systems, located on the pilot's left instrument panel, are shown in figure 2-14. Payload compartments cooling air data, for cruise flight conditions, is provided in Section III.

The hot air bypass system controls hot air bypass around the primary heat exchanger as described in paragraph 2.6.1.

The cockpit temperature system provides automatic and manual control of cockpit temperature by metering hot air, from the hot air manifold, with cold air, from the cold air manifold, as shown in figure 2-13. The conditioned air is supplied to both cockpits. Cockpit controls include a mode switch which permits the pilot to select automatic or manual control of cockpit air temperature. With the automatic (AUTO TEMP) mode selected, the system automatically controls air temperature to a setting manually established with a rotary temperature control. The manual modes permit the pilot to manually raise, lower, or maintain the temperature of cockpit air. Cockpit, E-bay, and R-bay air temperatures are selectable for display on a temperature indicator.

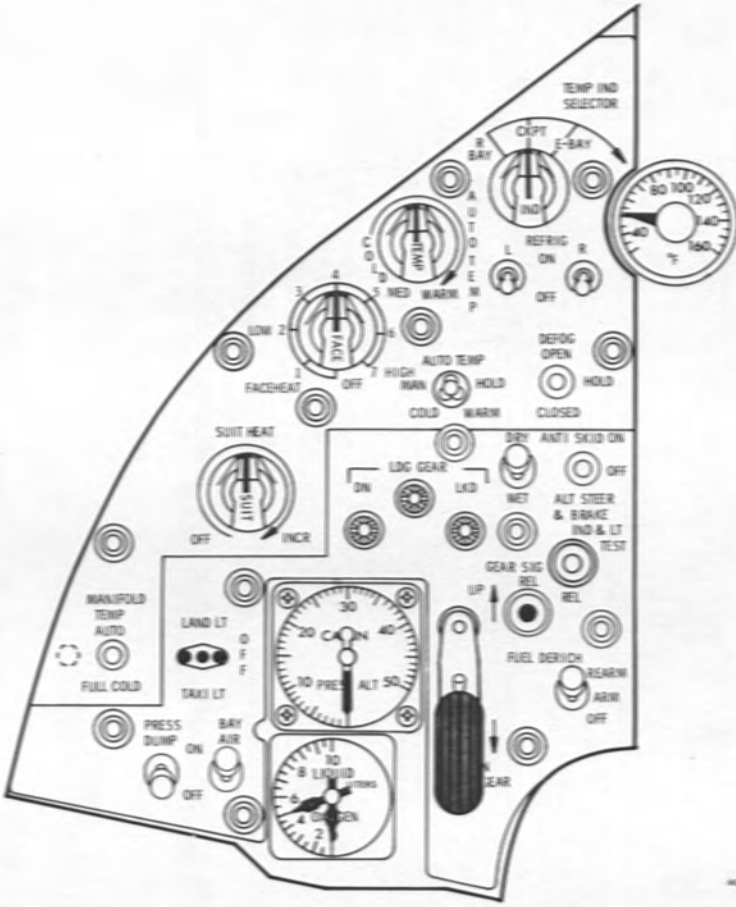


Figure 2-14. Environmental Control System Controls and Indicators

The manifold temperature system provides anti-icing protection for the cold air manifold. Air temperature is held at +45 (± 3) degrees F at lower altitudes, where moisture is more prevalent (atmospheric air pressures greater than 6 inches of mercury) and at -30 (± 3) degrees F at higher altitudes (air pressures less than 5 inches of mercury). The system has two modes, selectable with the MANIFOLD TEMP switch. The automatic mode selection causes the system to be automatically controlled. The full cold mode is used in the event of automatic mode failure and stops hot air introduction into the cold air manifold.

The pressure suit ventilating air temperature system controls heating of air supplied from the cold air manifold. An in-line electrical heater controls air temperature in response to the setting of a rotary SUIT HEAT switch on the pilot's instrument panel. The pilot controls suit air temperature for both crew members.

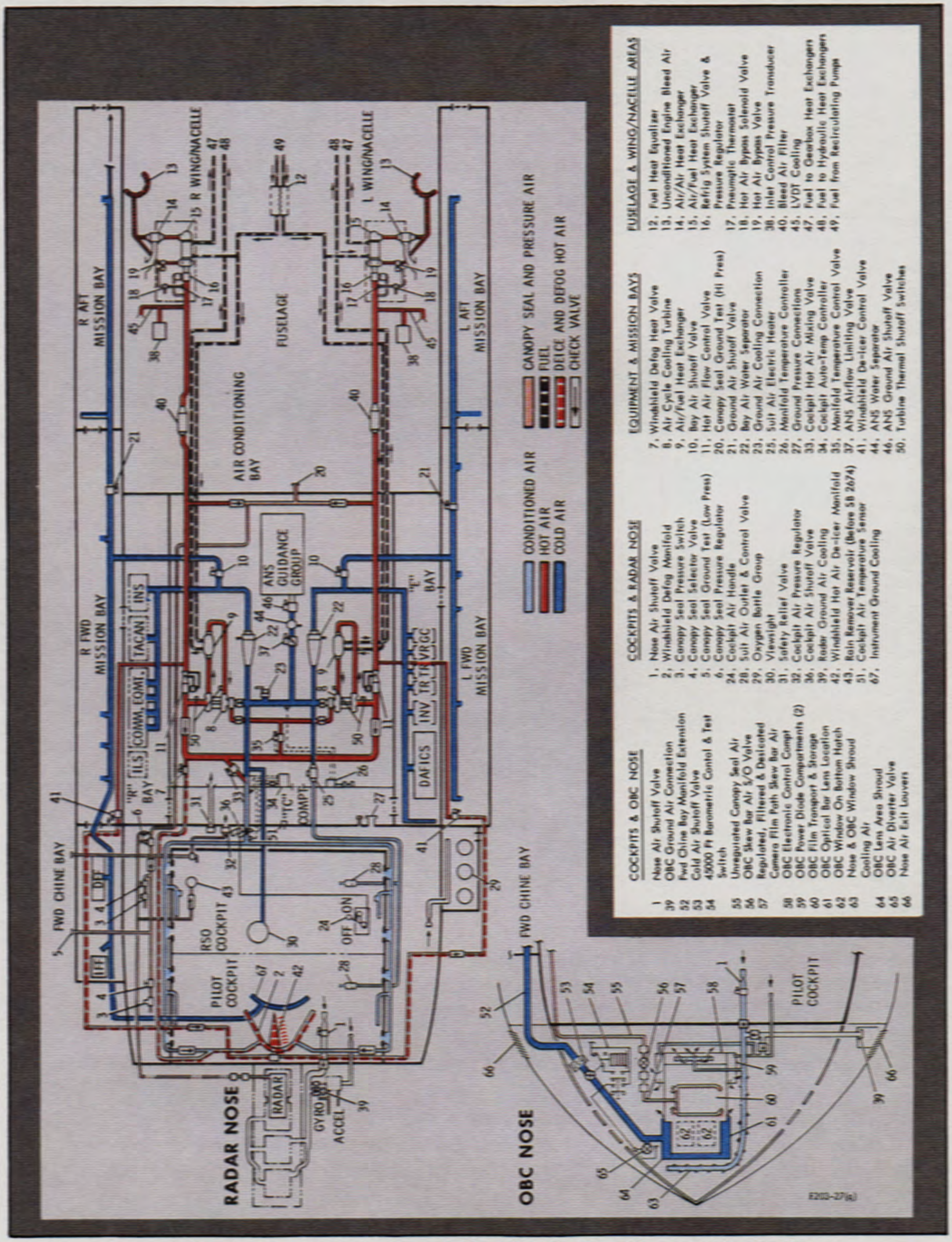


Figure 2-15. Air Cycle and Air Distribution System Schematic

2.6.3 Pressurization System. The pressurization system includes the cockpit pressurization system and the canopy seal system.

2.6.3.1 Cockpit Pressurization System. (See figure 2-16.) Crewmembers operate within sealed, insulated, and pressurized cockpit compartments. The cockpit pressurization system automatically controls air pressure in both cockpits according to manually selected schedules. Pressurization is accomplished by controlling cockpit exhaust air through a pressure regulator. A CABIN PRESS switch in each cockpit allows selection of either 10,000 FT or 26,000 FT pressure schedules. The cockpits are essentially unpressurized at altitudes below the pressure altitude selection. With 26,000 feet selected, the preferred selection as it provides for greater airflow and cooling, 26,000-foot pressurization is maintained at altitudes above 28,000 feet. With 10,000 feet selected, 10,000-foot pressurization is maintained to 28,000 feet, above which a positive pressure differential to ambient of 5 psig is maintained. A safety relief valve provides pressure differential protection in the event of pressure regulator failure. The valve also opens fully, for rapid cockpit depressurization, when the pilot's PRESS DUMP switch is set to ON (see figure 2-14).

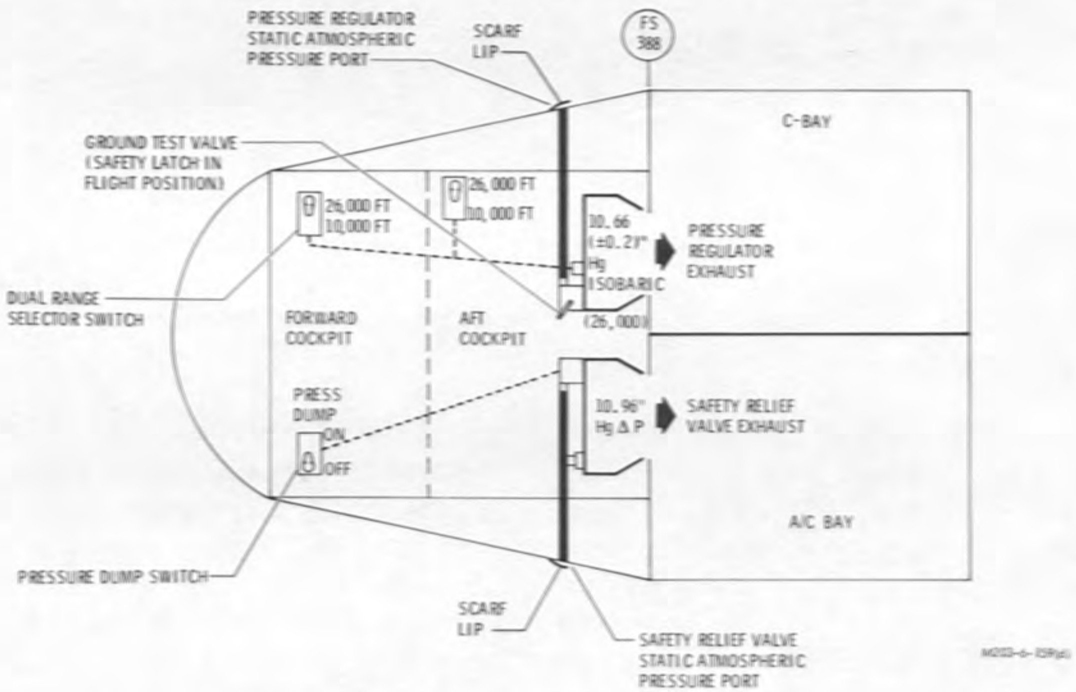


Figure 2-16. Cockpit Pressurization

2.6.3.2 Canopy Seal System. (See figure 2-17.) The canopy seal system uses air pressure from the hot bleed air manifold (see figure 2-13) to pressurize canopy seals. Seal pressure is regulated at 20 psi above cockpit pressure. Low pressure and high pressure ground air connectors are provided for ground checkout of the system.

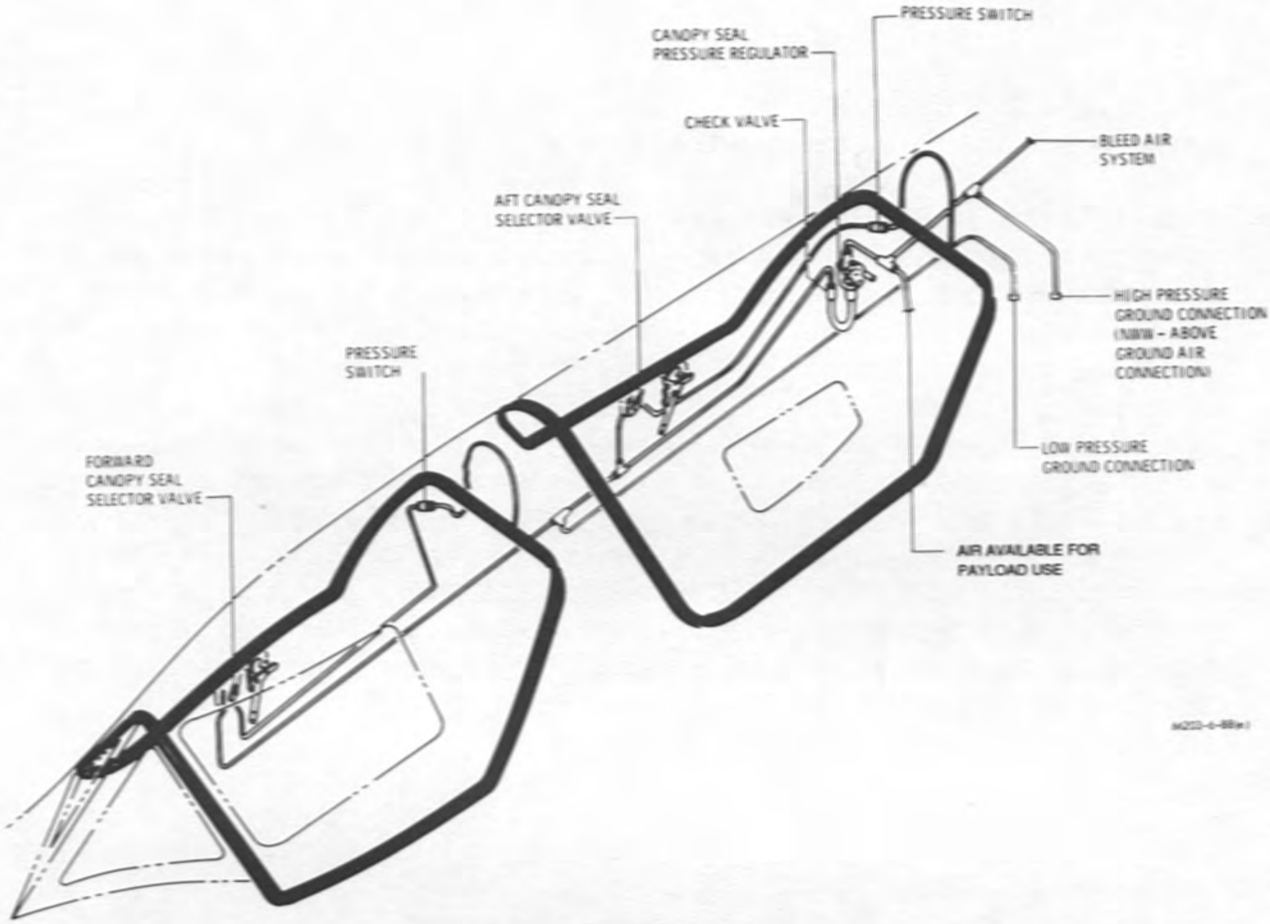


Figure 2-17. Canopy Seal Pressurization System

2.6.4 ECS Caution Lights. The following caution lights, located on the pilot's annunciator panel, are associated with operation of the environmental control system. The L (R) AIR SYS OUT light comes on when the shutoff valve in the corresponding heat sink package is automatically or manually (REFRIG switch set to OFF) commanded to close, shutting down the system. E-BAY or R-BAY OVERHEAT lights come on if air temperature in the corresponding bay exceeds 150 degrees F. The COCKPIT AIR OFF light comes on when the cockpit air valve is closed because the aft canopy latch handle is in the unlocked position or aft cockpit air handle is set to OFF. WINDSHIELD DEICE ON light is on when windshield DE-ICE switch is set to ON. CANOPY UNSAFE light comes on if either canopy is not properly closed on its sill, either canopy latch handle is not latched, or if canopy seals are not fully inflated. The BAY AIR OFF light comes on if the BAY AIR switch is OFF, shutting off air to the payload equipment bays and nose, or if the ground air receptacle is not properly capped.

2.7 OXYGEN SYSTEM.

Crew members breath oxygen for the entire duration of the flight, from the time they are seated in the cockpits, and pressure suit/helmet connections are made, until they are disconnected from the aircraft at the end of the flight. Oxygen is not only needed for breathing during high-altitude flight but is required in order to purge nitrogen from crewmembers' blood so that nitrogen embolisms (bends) do not occur during ascent and flight at altitude. The average rate of oxygen consumption, at a selected cockpit altitude of 26,000 feet, is 1 liter of liquid oxygen per hour for two persons. With 10,000 feet selected, the average rate is 1.3 liters per hour.

As shown in figure 2-18, oxygen is supplied by two "normal" oxygen systems and one standby system. Each system has a 10-liter liquid-oxygen converter which produces gaseous oxygen, at 65 to 100 psi, through a regulated evaporative process. In addition to the converters, located in the B2-bay in the left chine (refer to Section III, figure 3-2.), there are three control panels. Control panels for both normal systems 1 and 2 and the standby system are located on the pilot's left console. The aft cockpit only has a control panel for normal systems 1 and 2, located on the left console. The normal oxygen control panels have two selector valves, for systems 1 and 2, operated by latchable-ON levers. Dual-reading gages indicate pressure in the systems when the selector levers are set to ON. The standby panel also has two lever-operated selector valves, one for system 1 and one for system 2, and a single needle gage which continuously indicates only standby system pressure. The standby system is normally used only if pressure remaining in system 1 or system 2 is low or if either system fails. The standby system is connected to either system 1 and/or system 2, using the selector valve levers on the standby control panel.

A dual gage in each cockpit monitors the liquid oxygen quantity remaining in each converter. The number 2 needle of each gage indicates the oxygen quantity in converter No. 2. A two-position selector switch, on the pilot's left console, permits the number 1 needle of each gage to indicate oxygen quantity in converter No. 1 or in the standby converter. Four caution lights, on the annunciator panels in each cockpit, illuminate to warn of a low-level or low-pressure condition in system 1 or system 2. The low-level lights come on when oxygen quantity in the associated system is less than 1 liter. The low-pressure lights come on when supply pressure falls below 50 (± 3) psi. An emergency oxygen supply, part of each crewmember's ejection seat survival kit, can be activated without ejection.

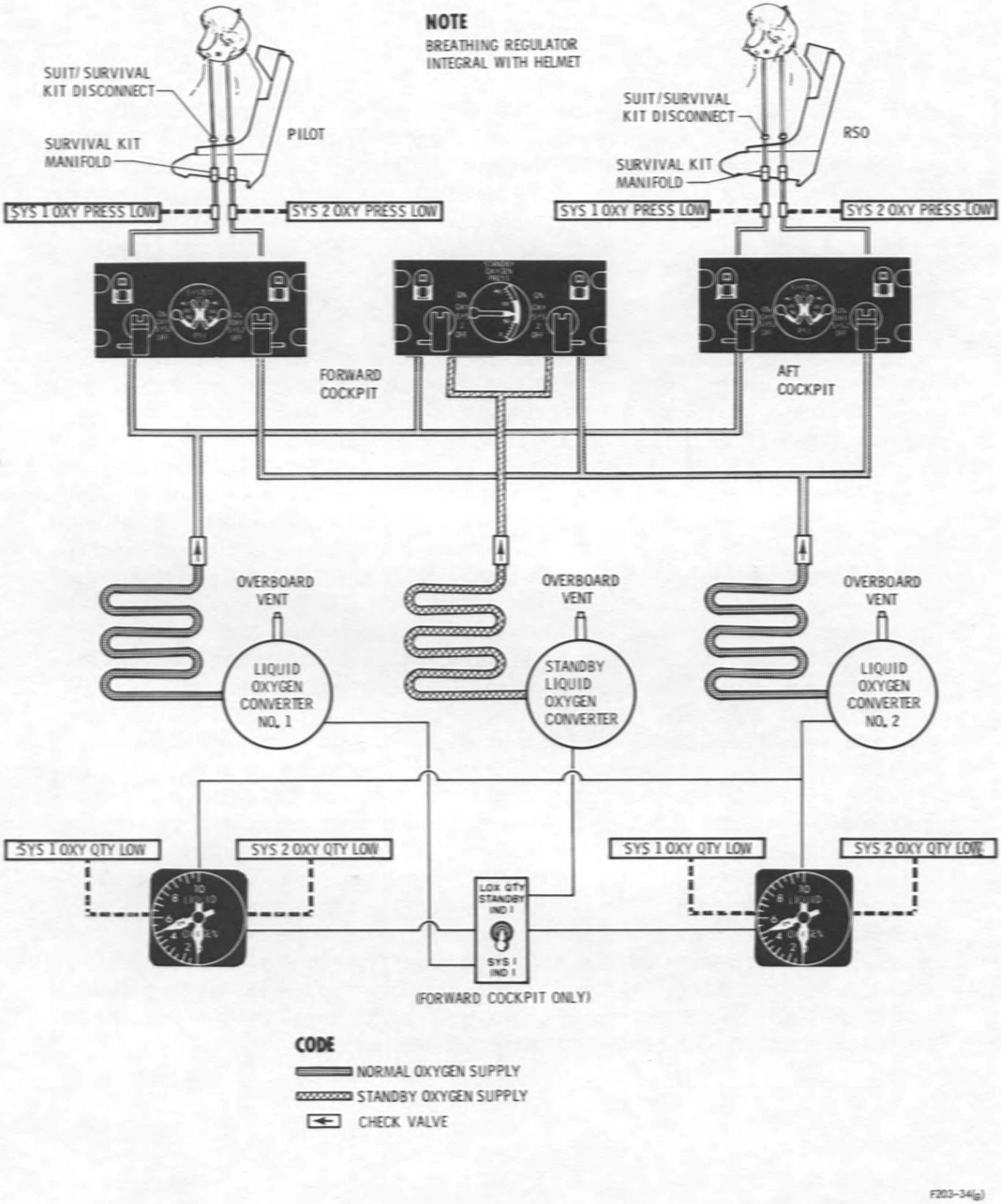


Figure 2-18. Oxygen System Schematic

2.8 FLIGHT CONTROLS.

The delta wing configuration of the aircraft employs elevons (combination elevators and ailerons), mounted along the trailing edge of the wing, for aircraft control and stability about the pitch and roll axes. Two elevons are located on each side of the fuselage, one inboard of the nacelle the other outboard of the nacelle. The outboard elevons are slaved to the inboard elevons. Dual, central-pivoted all-moving rudders provide directional and yaw axis control.

The flight control surfaces are positioned by hydraulic actuators which respond to electro-mechanically-controlled, hydraulically-powered servos. Hydraulic power is supplied by the A and B hydraulic systems, each of which powers one half of the actuators and servos. Either hydraulic system is capable of providing sufficient power for control surface operation. Pilot control is accomplished using a control stick for elevon control and foot pedals for rudder control. Control stick and rudder pedal forces are mechanically transmitted, via control cables and linkage, to the elevon and rudder servos. The servos, in turn, control operation of the actuators which move the control surfaces. Electronic control of the elevon and rudder servos is from the Automatic Flight Control System (AFCS) in the DAFICS. The AFCS includes an autopilot and a Stability Augmentation System (SAS).

The control surface servos do not transmit aerodynamic loads back to the cockpit controls. Artificial feel springs, which provide resistance proportional to control movement, are incorporated in each axis to provide the pilot with the sense of control surface operation. Mechanical stops are incorporated to limit elevon movement for pitch control so that sufficient differential travel remains for roll control. Total elevon travel is limited to 35 degrees up, 20 degrees down for the inboard elevons and 35 degrees down for the outboard elevons. Rudder travel is also mechanically limited to 20 degrees each side of neutral. Additionally, the pilot can further limit elevon roll and rudder travel at speeds above Mach 0.5. A caution light advises the pilot if the selectable limiter stops are improperly selected for a given speed.

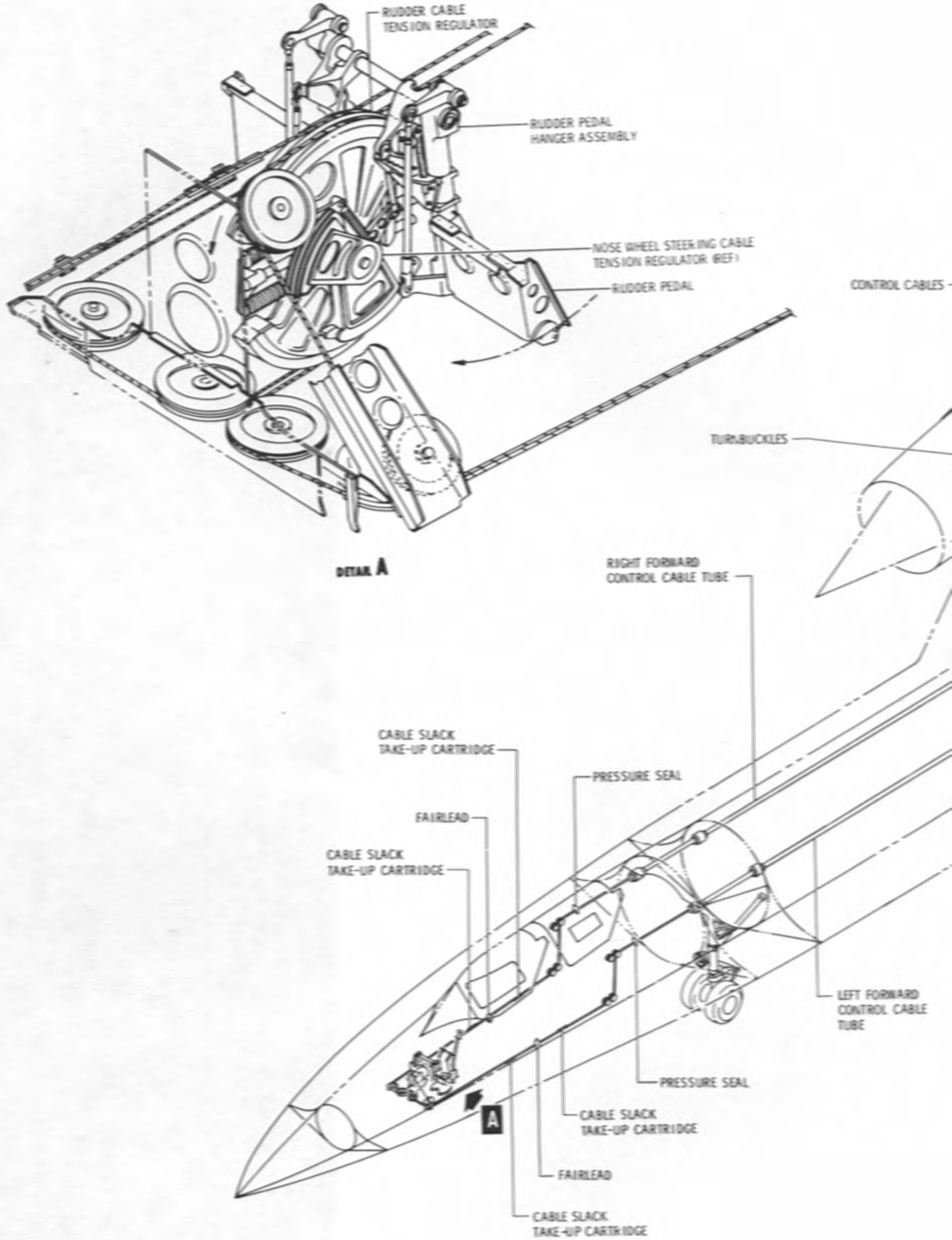
2.8.1 Rudder Control System. Directional control and yaw axis stability is provided by tetrahedral-shaped vertical fins mounted on the upper-aft part of each nacelle. The vertical fins consist of two sections; a lower fixed section (stub fin) and an upper movable section (rudder), which is pivot-mounted on a tapered post that projects vertically out of the stub fin. Each rudder and stub fin are canted inboard 15 degrees. The rudders move left or right, in a synchronized manner, for yaw control. Each rudder is positioned by movement of a gudgeon arm, positioned by four double-acting hydraulic actuators, two powered by the A hydraulic system, two by the B system. The actuators respond to the metered hydraulic output of the rudder servo, also dual powered by the A and B hydraulic systems.

Mechanical inputs to the rudder servos consist of rudder pedal displacement inputs and yaw trim actuator inputs. The rudder control system is shown in figure 2-19. The pilot controls the left and right rudder trim actuators by setting the TRIM POWER switch, on the center stand, to ON and operating the YAW-PITCH TRIM switch, on the control stick, to trim the rudders up to 10 degrees left or right. Rudder trim is indicated by displacement of the L and R pointers on the yaw trim indicator on the instrument panel. If rudders are not aligned, as shown by nonalignment of L and R pointers, the right rudder is aligned with the left by operating the RUDDER SYNC switch, on the left console, until alignment is achieved. Electrical control of the rudder servos is through operation of electro-hydraulic transfer valves which control the servos in response to signals from the yaw SAS circuits of the DAFICS. Yaw SAS reliability is ensured by the use of dual yaw transfer valves.

To prevent excessive aerodynamic loads on the rudders and airframe at speeds above Mach 0.5, the pilot restricts rudder movement to approximately one-half normal travel by use of a SURF LIMIT RELEASE handle on the control stand. The handle limits rudder travel when pushed in and allows normal travel when pulled out. A SURFACE LIMITER caution light, on the annunciator panel, alerts the pilot if the handle is in the wrong position for Mach numbers below and above 0.5. The surface release handle and caution light are common to both rudder and elevon roll limiter operation.

2.8.2 Elevon Control System. Control and stability about the pitch and roll axes is provided by four elevon surfaces, mounted on the wing trailing edge inboard and outboard of the nacelles. Changes in aircraft pitch attitude is produced by equal up or down displacement of all elevons. Roll is accomplished by equal but opposite displacement of elevon pairs on each side of the aircraft. Combined roll and pitch cause unequal and opposite elevon pair displacement. The elevons are moved by operation of double-acting hydraulic actuators; six actuators for each of the inboard elevons and fourteen for each of the outboard elevons. Half of the actuators are powered by the A hydraulic system, half by the B hydraulic system. The actuators respond to the metered hydraulic output of separate inboard and outboard elevon servos, also dual powered by the A and B hydraulic systems. The outboard servos are mechanically slaved to the inboard units, with the elevons on each side of the aircraft functioning as a pair.

Mechanical inputs to the inboard elevon servos represent control stick displacement and pitch and roll trim actuator inputs. Since the elevons function as combined elevators and ailerons, providing both pitch and roll control, inputs to the elevon servos represent combined control stick pitch and roll commands. Summation of the pitch and roll commands is accomplished by a mechanical mixer, located in the fuselage tail. The elevon control system is shown in figure 2-20. Elevon trim is provided by separate pitch and roll actuators located in the mixer assembly.



RUDDER CABLE TENSION REGULATOR
RUDDER PEDAL HANGER ASSEMBLY
NOSE WHEEL STEERING CABLE TENSION REGULATOR (REF)
RUDDER PEDAL
DETAIL A
CONTROL CABLES
TURNBUCKLES
RIGHT FORWARD CONTROL CABLE TUBE
CABLE SLACK TAKE-UP CARTRIDGE
PRESSURE SEAL
FAIRLEAD
CABLE SLACK TAKE-UP CARTRIDGE
LEFT FORWARD CONTROL CABLE TUBE
PRESSURE SEAL
A
CABLE SLACK TAKE-UP CARTRIDGE
FAIRLEAD
CABLE SLACK TAKE-UP CARTRIDGE

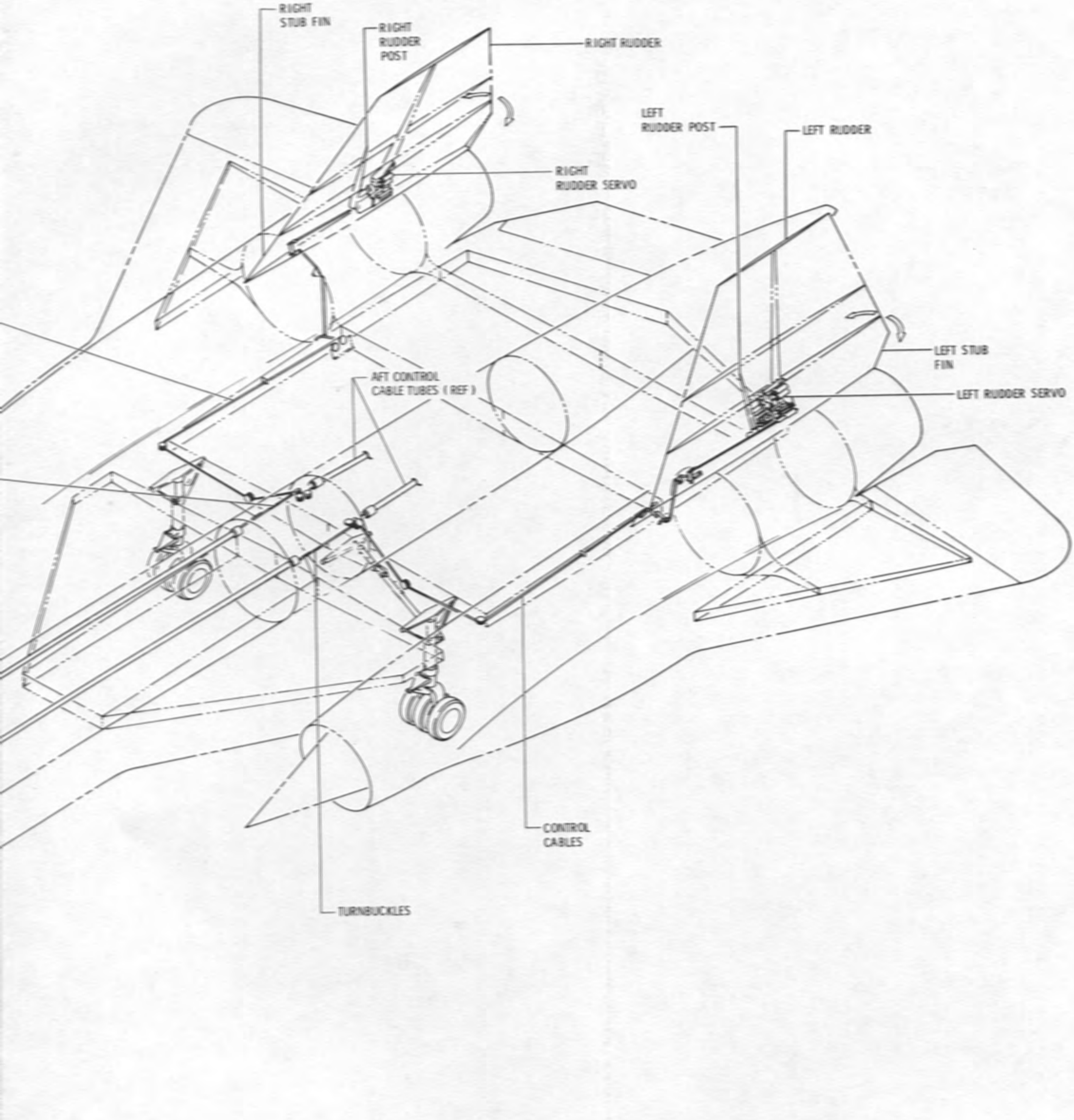
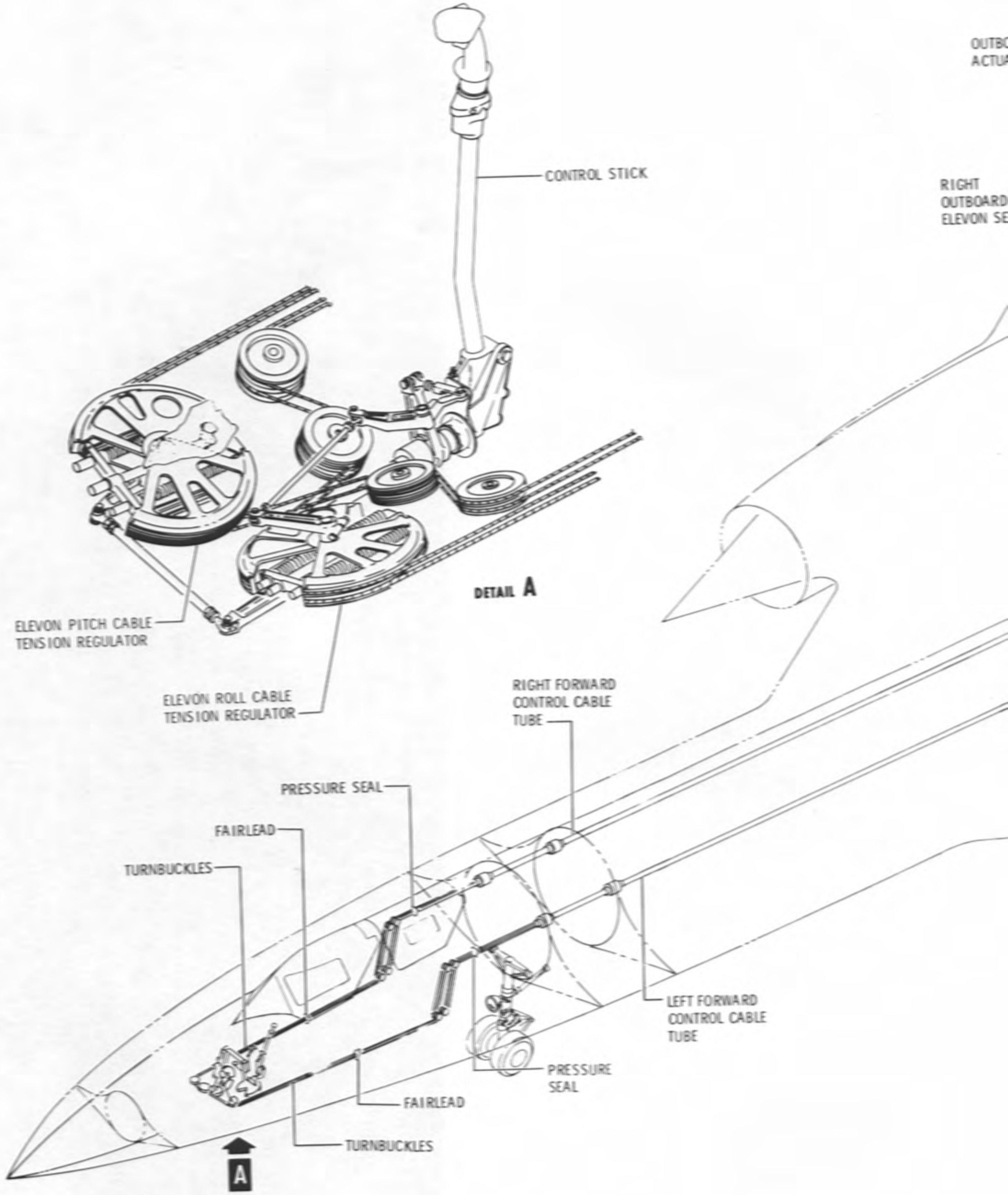


Figure 2-19. Rudder Control System



CONTROL STICK
DETAIL A
ELEVON PITCH CABLE TENSION REGULATOR
ELEVON ROLL CABLE TENSION REGULATOR
RIGHT FORWARD CONTROL CABLE TUBE
PRESSURE SEAL
FAIRLEAD
TURNBUCKLES
LEFT FORWARD CONTROL CABLE TUBE
PRESSURE SEAL
FAIRLEAD
TURNBUCKLES
A

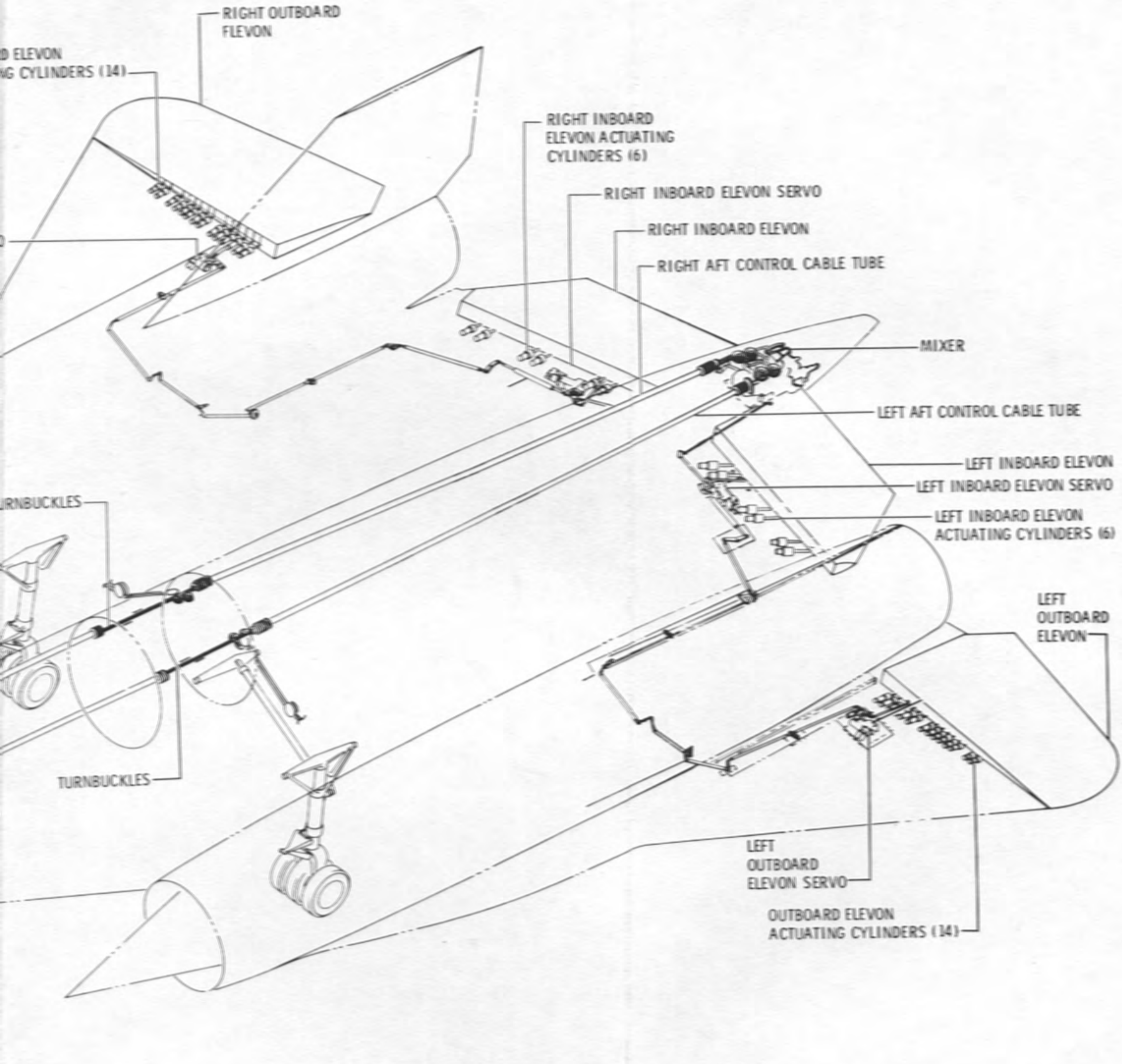


Figure 2-20. Elevon Control System

Electrical input to either actuator changes the geometrical relationship of mixer linkage in the affected axis, altering the inputs to the elevon servos and trim of the elevons. Pitch trim affects all elevons in the same direction, roll trim is applied equally but in the opposite direction to elevons on opposite sides of the aircraft. The pilot controls the pitch and roll trim actuators by setting the TRIM POWER switch, on the center stand, to ON and operating the YAW-PITCH TRIM switch, on the control stick, for pitch trim and the ROLL TRIM switch, on the left console, for roll trim. The pitch trim actuator has two motors, a high-speed motor which responds to pilot trim inputs and a low-speed motor which responds to the autopilot trim and Mach trim inputs from the AFCS.

In addition to control stick and manual/AFCS trim inputs to the pitch axis, the Automatic Pitch Warning (APW) System, part of DAFICS, applies pitch inputs under certain angle-of-attack and pitch-rate conditions. Separate pitch and roll trim indicators are provided on the pilot's instrument panel. Electrical control of the inboard elevon servos is through operation of electro-hydraulic transfer valves which control the servos in response to pitch and roll signals from the autopilot and SAS systems of the DAFICS. Pitch SAS reliability is ensured by use of dual pitch transfer valves. Roll SAS uses a single transfer valve, with either inboard elevon servo capable of providing SAS roll control if the other is inoperative, but with an aircraft response of one-half the normal rate.

To prevent excessive aerodynamic loads on the elevons and airframe, at speeds above Mach 0.5, the pilot restricts differential (roll) elevon travel to approximately 7 degrees. The SURF LIMIT RELEASE handle, also used to limit rudder travel, limits elevon differential travel when pushed in and allows normal travel when pulled out. The SURFACE LIMITER caution light alerts the pilot if the handle is in the wrong position.

2.9 DIGITAL AUTOMATIC FLIGHT AND INLET CONTROL SYSTEM (DAFICS).

The DAFICS consists of four digital-computer-controlled subsystems plus a computer-analyzer subsystem. The subsystems include an Air Data System (ADS), an Automatic Flight Control System (AFCS), an Automatic Pitch Warning (APW) System, and an Air Inlet Control System (AICS). Although subsystems of the DAFICS, they are functionally separate systems. The computer-analyzer subsystem assists with overall DAFICS maintenance. Redundant sensors, signal channels, computers, and, where required, servo channels provide extremely high system reliability.

2.9.1 General. DAFICS primary components include a computer assembly, for computational operations, a preflight BIT panel, to initiate a comprehensive built-in-test to determine system flight readiness, and a DAFICS analyzer, which permits rapid post-flight and maintenance analysis of system performance and insertion of AICS gain schedule constants. Primary system components are shown in figure 2-21. Additional DAFICS components are those related to the functioning of each of the DAFICS subsystems, as described and shown in the following text and illustrations. The primary functions of the four subsystems are shown in figure 2-22.

The DAFICS computer assembly includes three separate digital computer units (DCUs), which are line-replaceable units (LRUs). The computers, identified as A, B, and M, provide all the DAFICS computational operations and functions. Computer fail caution lights, on the pilot's annunciator panel, advise if a computer is not operating. Guarded reset switches permit an attempted reset of an inoperative computer.

DAFICS operates in two modes, a ground test (maintenance) mode and a flight (operational) mode. The ground test mode is in effect during initial warmup, during a preflight BIT test, and when the DAFICS analyzer is being used. When not in the ground test mode, the system operates in the flight mode, which is functionally the same whether the aircraft is on the ground or in flight. In the flight mode, continuous computer self tests are performed, failure of which causes illumination of the appropriate CMPTR OUT caution light and automatic switching of computer computations to the remaining operable computer(s). Redundant AFCS sensor and servo channels and ADS pressure sensor channels are continuously monitored and managed to eliminate faulty signals or channels.

2.9.2 Air Data System (ADS). The ADS uses pneumatic pressures (pitot, static, alpha, and beta), developed by the pitot static system, to produce signals representing KEAS, altitude, Mach, TAS, alpha (angle of attack), and beta (angle of sideslip). These signals and their derivatives are supplied to cockpit indicators and advisory lights as shown in figure 2-22. Additionally, the ADS supplies gain schedules and functions of KEAS and Mach to the AFCS, functions of alpha and Mach to the APW, and functions of alpha, beta, and Mach to the AICS.

As shown in figure 2-23, pitot and static pressures, from the pitot static mast, are applied to the pilot's pressure instruments. Also, pitot static pressures, together with alpha beta pressures from the alpha beta probe on the pitot static mast, are applied to a pressure transducer assembly (PTA) located in the B1-bay in the left chine (refer to Section III, figure 3-2). The PTA, with triple-redundant channels for pitot, static, and alpha signals, converts input pressures into digital equivalents which are applied to the DAFICS computer.

Computer outputs are used to operate advisory lights and indicators and as inputs to other systems. The KEAS warning light comes on if abnormally high or low airspeed conditions exist. The compressor inlet pressure (CIP) indicator reflects CIP computed from DAFICS-generated KEAS and Mach, as well as CIP monitored in the inlets. The

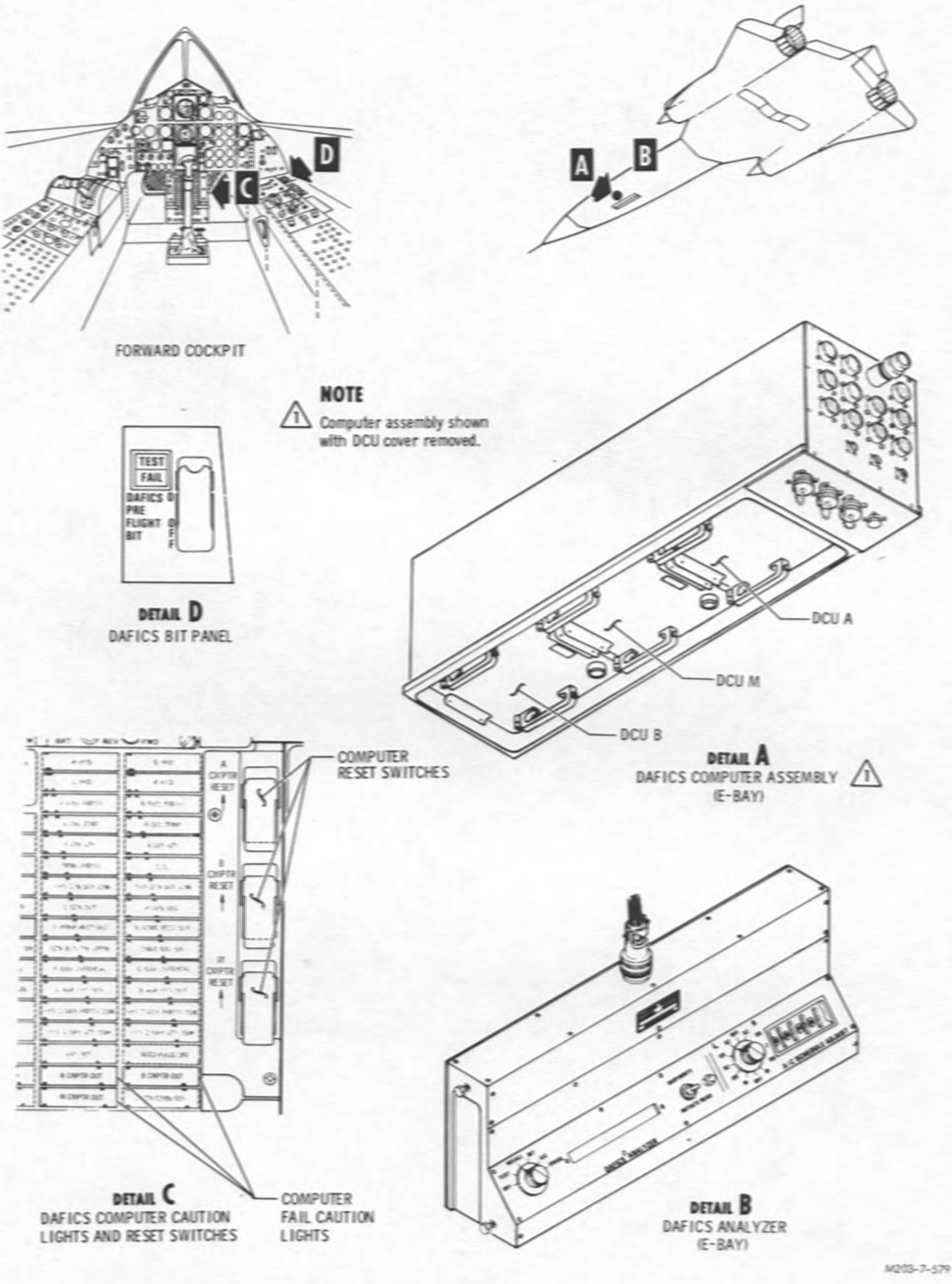


Figure 2-21. Primary DAFICS Components

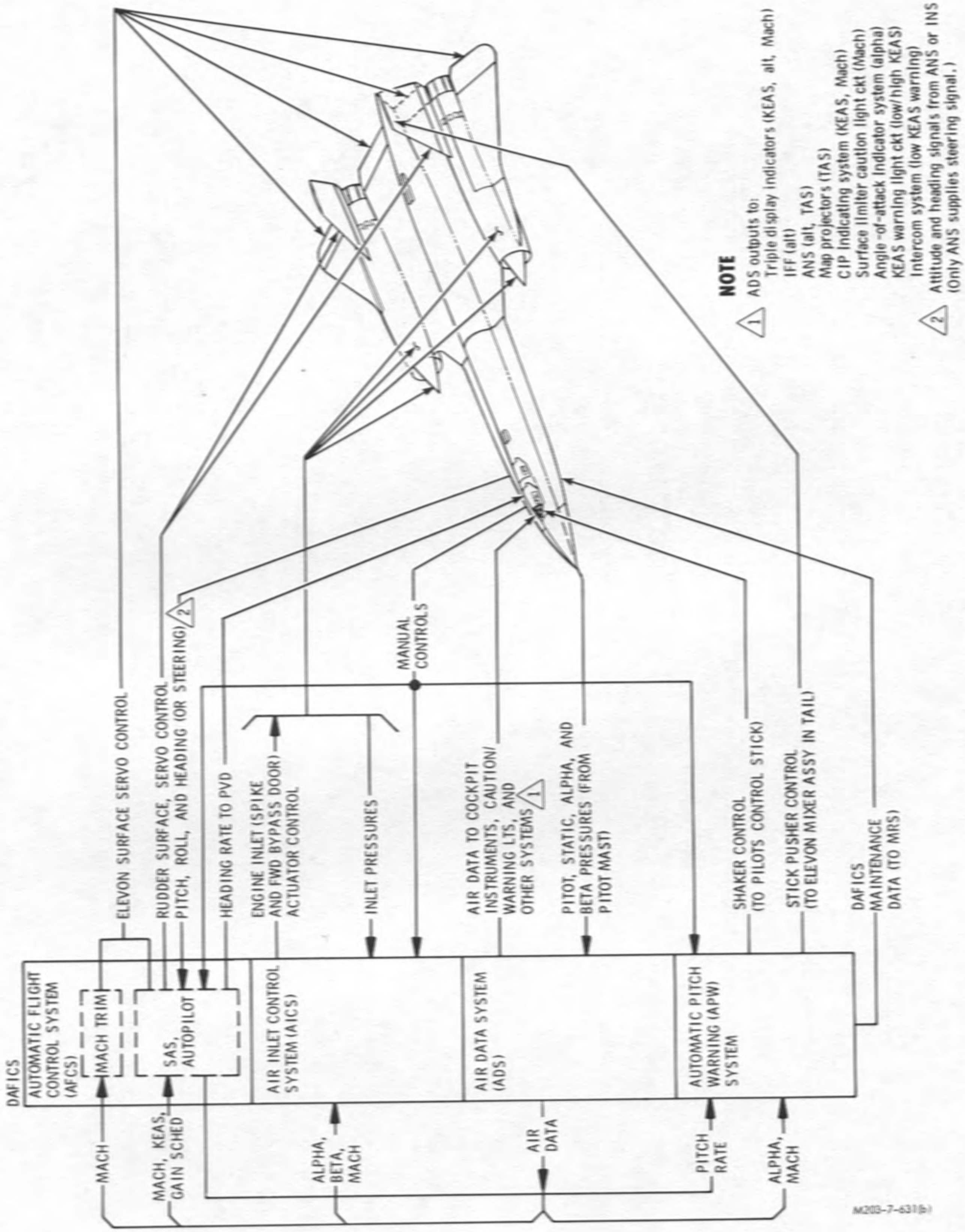


Figure 2-22. DAFICS Block Diagram

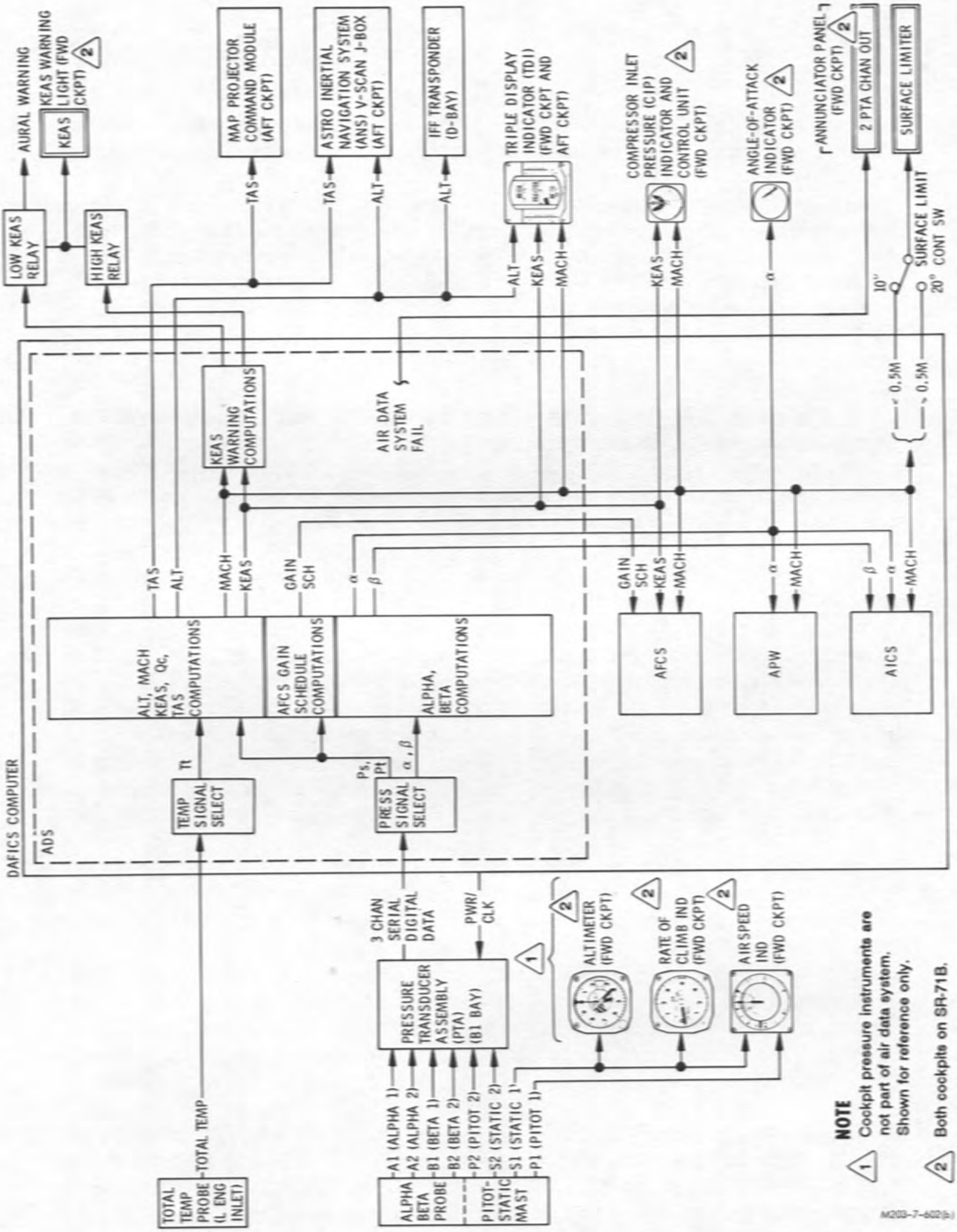


Figure 2-23. Air Data System Block Diagram

angle-of-attack indicator uses the applied alpha signal for angle-of-attack indication. Triple display indicators in both cockpits provide a digital readout of aircraft altitude, KEAS, and Mach. True airspeed (TAS) is applied to a map projector module which, in turn, develops drive signals to control the motion of film in the map projectors in both cockpits. Altitude is applied to the IFF transponder for altitude encoding. Altitude and TAS signals are applied to the astroinertial navigation system (ANS) as reference signals. The SURFACE LIMITER caution light is controlled by signals representing speeds under and over Mach 0.5 and the position of the surface limit handle. The 2 PTA CHAN OUT caution light comes on if two of the three PTA channels fail or if data in two of the channels is invalid.

2.9.3 Automatic Flight Control System (AFCS). The AFCS automatically controls the aerodynamic condition of the aircraft through operation of the rudder and elevon control surfaces. The AFCS consists of three subsystems: a Stability Augmentation System (SAS), an autopilot, and a Mach trim system. Figure 2-24 shows AFCS components and their location. Figure 2-25 is a block diagram of the AFCS.

2.9.3.1 Stability Augmentation System (SAS). The SAS provides three-axis (pitch, roll, and yaw) stabilization of the aircraft through control of rudder and elevon surfaces. The system uses rate gyros and lateral accelerometers as motion-sensing elements. Three yaw-rate gyros, three pitch-rate gyros, two roll-rate gyros, and three lateral accelerometers provide required redundancy. As shown in figure 2-25, the outputs of the gyros and accelerometers, related to motion of the aircraft about each axis and in a lateral direction, are applied to multiple-redundant SAS circuits in the DAFICS. In addition to the rate-gyro signals, analytical rate signals, derived from pilot-selected ANS or INS pitch, roll, and heading inputs, are used as references to isolate and eliminate faulty rate-gyro signals if multiple gyro failures occur.

Yaw-rate gyro and lateral accelerometer signals are summed and amplified in the DAFICS yaw channels and applied as yaw commands to the left and right rudder servos as shown in figure 2-25. YAW SAS engage switches, on the AFCS function selector (control) panel, enable separate A and B channels in each servo. The A channel is powered by hydraulic system A, the B channel by hydraulic system B. Each servo channel controls the application of hydraulic pressure to two of the four actuating cylinders that move each rudder. In addition to yaw SAS signals from DAFICS, the servos respond to mechanical inputs from the rudder pedals. Rudder pedal inputs control both servo channels and all four actuators for each rudder.

As also shown in figure 2-25, pitch-rate and roll-rate gyro signals are amplified in DAFICS pitch and roll channels and applied as commands to the inboard elevon servos. PITCH SAS and ROLL SAS engage switches, on the AFCS control panel, permit engagement of roll and pitch channels in the inboard elevon servos. Elevon control is similar to that described for rudder control; with three of the six inboard

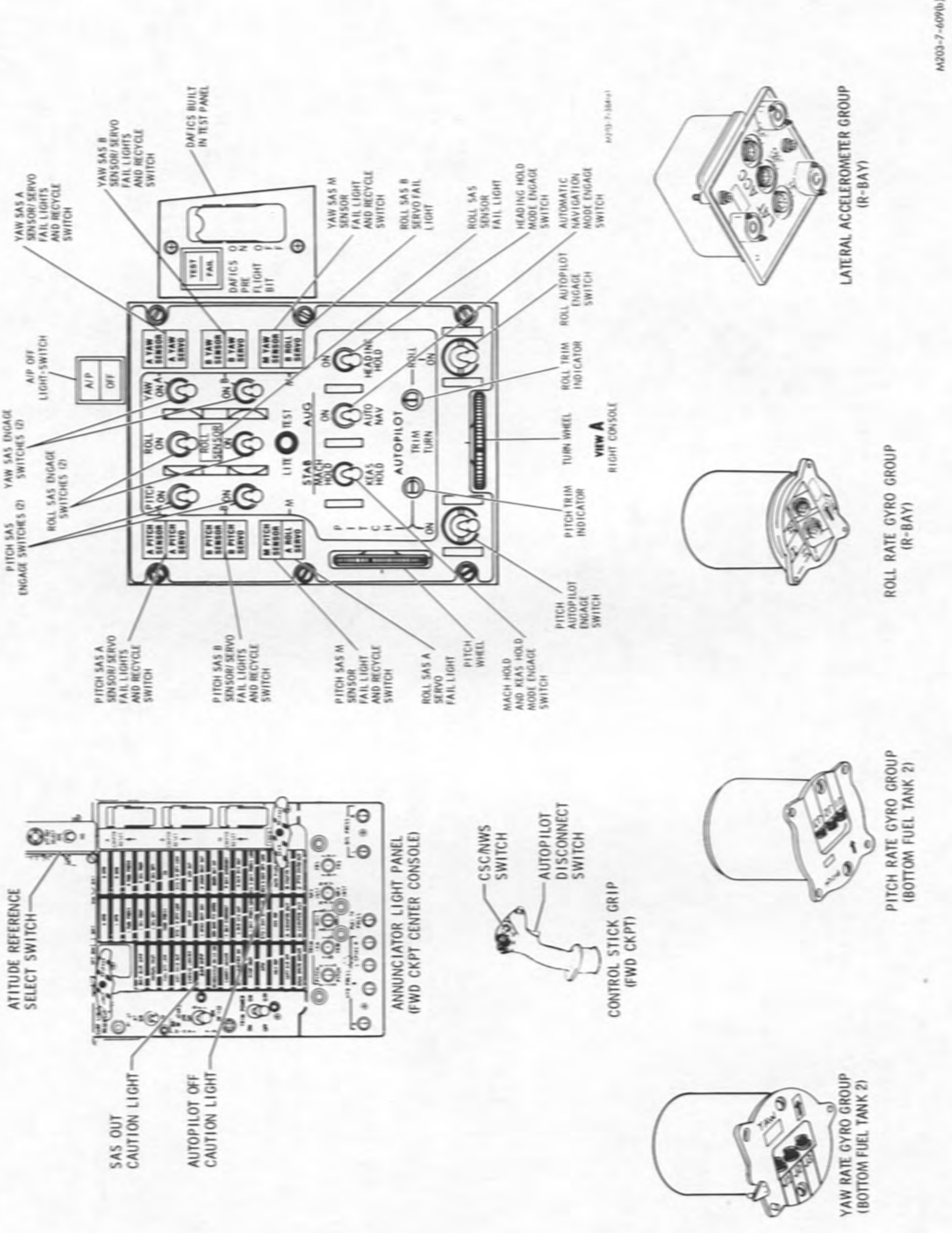
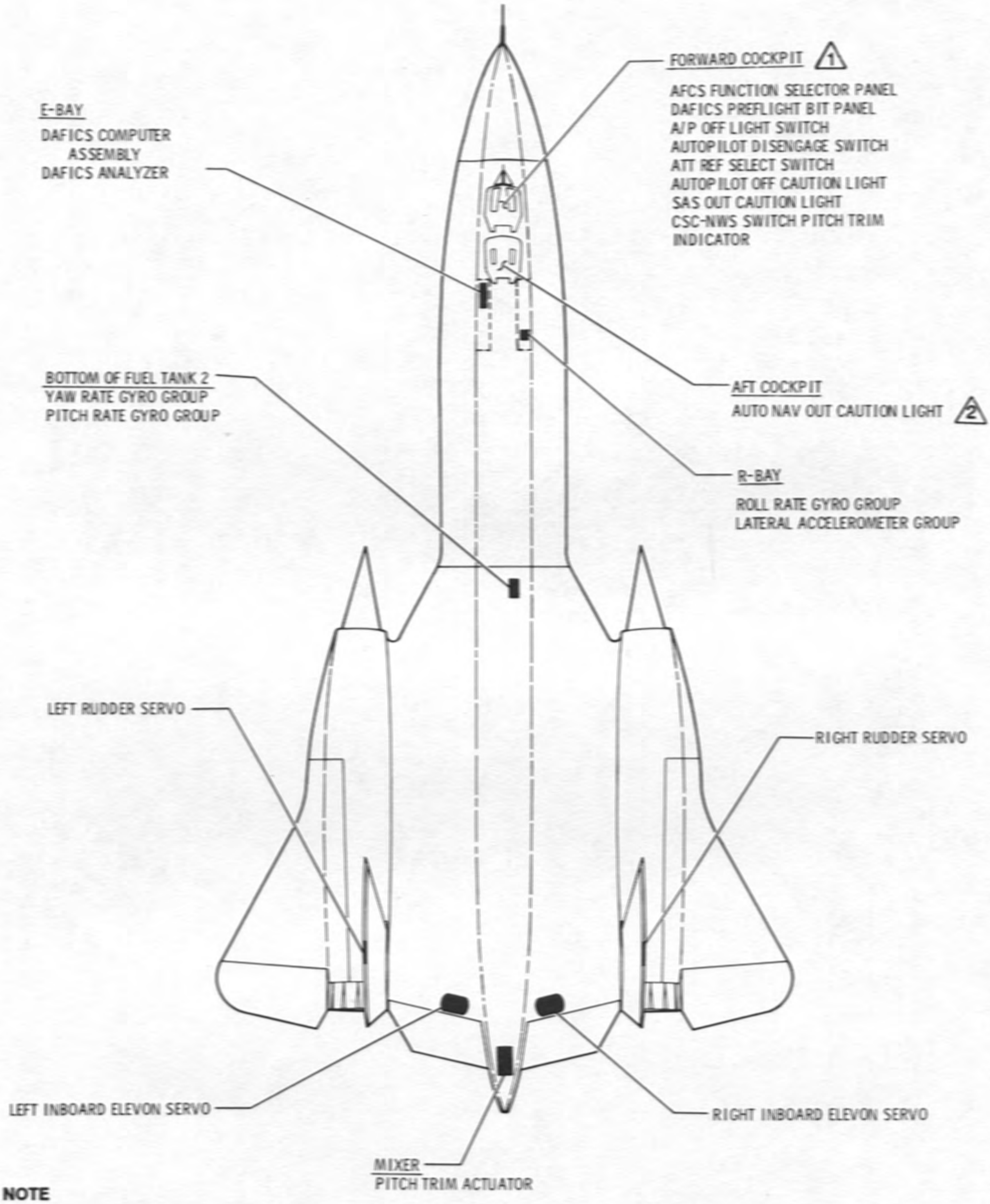


Figure 2-24. AFCS Components (Sheet 1 of 2)



NOTE

⚠1 Controls, indicator, and lights are duplicated in aft cockpit on SR-71B.

⚠2 On SR-71A only.

Figure 2-24. AFCS Components (Sheet 2 of 2)

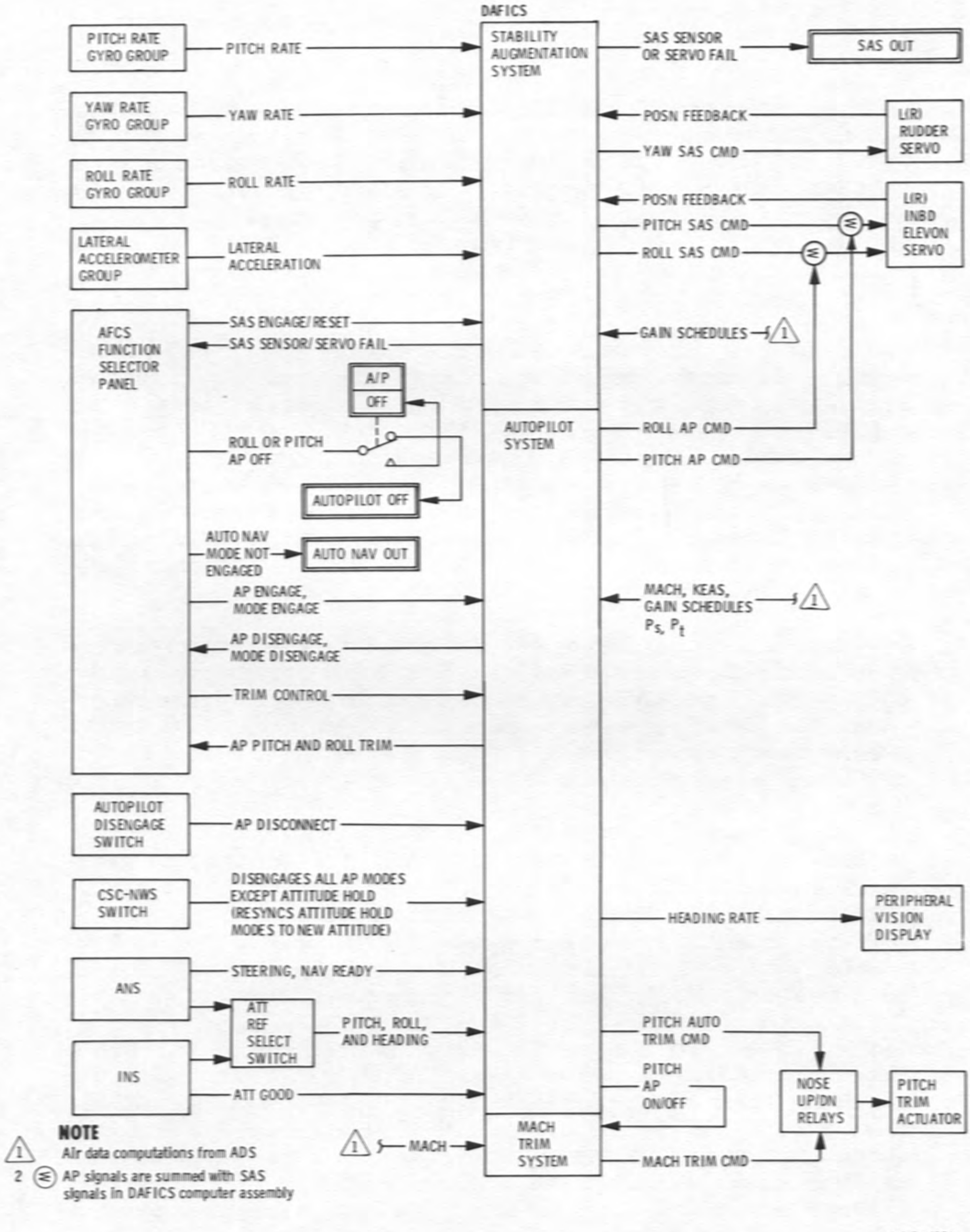


Figure 2-25. AFCS Block Diagram

elevon actuators on each side of the aircraft driven by pressure from hydraulic system A and the other three by pressure from hydraulic system B. Hydraulic pressures are ported to the actuators from the inboard elevon servos by the combined effects of DAFICS roll and pitch SAS input signals and control stick mechanical inputs from the mixer assembly. The outboard elevon servos are mechanically linked to the inboard servos and, in turn, power the fourteen actuators on each side of the aircraft that position the outboard elevons. Half of the outboard actuators are powered by hydraulic system A, the other half by hydraulic system B.

2.9.3.2 Autopilot. The autopilot (AP) system includes separate pitch and roll autopilots, which together provide control of aircraft attitude about the pitch and roll axes. The pitch autopilot operates in a pitch attitude-hold mode or in one of two selectable speed control modes, KEAS hold or Mach hold, while also providing automatic pitch trim. The roll autopilot operates in a roll attitude-hold mode or in a selectable heading-hold mode or an auto-steering (AUTO NAV) mode. Autopilot output signal strength is scheduled as a function of altitude and airspeed by inputs from the ADS. Pitch and roll autopilots are enabled by operation of PITCH and ROLL engage switches on the AFCS control panel. Both autopilots affect operation of only the elevon control surfaces.

Engagement of an autopilot causes the related autopilot command signal (pitch or roll) to be summed with the related pitch SAS or roll SAS command signal, as shown on figure 2-25. Combined SAS and autopilot commands, applied to the inboard elevon servos, cause the elevons to move and be positioned so as to counter undesired aircraft motion about the pitch and roll axes and to respond to pitch autopilot and roll autopilot commands.

When the pitch autopilot is engaged, the aircraft is retained in the pitch attitude established at the time of engagement. With MACH HOLD selected, pitch hold is released and the autopilot controls pitch attitude to maintain the Mach number established at the time of Mach-hold engagement. With KEAS HOLD selected, pitch hold is released and the pitch autopilot maintains the existing airspeed until a KEAS bleed line is intercepted. The autopilot then follows a schedule that bleeds off KEAS, from 500 to 380, as Mach increases from 2.1 to 3 plus.

When the roll autopilot is engaged, the aircraft is retained in the roll attitude established at the time of engagement. With HEADING HOLD selected, roll hold is released and the autopilot controls roll to maintain the aircraft heading established at time of heading-hold engagement. Heading hold can be engaged with the aircraft in any bank angle up to 50 degrees and, when engaged, causes the aircraft to roll out and capture the engaged heading. With AUTO NAV selected, roll hold is released and the autopilot controls roll to maintain aircraft flight along a great circle course in response to bank angle steering commands from the ANS.

The pilot's ATT REF SELECT switch is used to select either the ANS or backup INS as the source of pitch, roll, and heading reference inputs for the autopilot. Only the ANS, when selected, provides a steering command signal for AUTO NAV operation. Manually operated PITCH and TURN (roll) wheels, on the AFCS control panel, permit changes in pitch and roll attitude during autopilot engagement. Control stick inputs cause only momentary departure from engaged pitch or roll attitude unless the CSC/NWS (control stick command/nose wheel steering) switch is pressed on the control stick grip. Pressing the switch, see figure 2-24, temporarily disengages the pitch and roll autopilots, allowing the pilot to make attitude changes with the control stick without having to disengage then reengage the autopilots. Releasing the switch reengages the autopilots at the new attitude; however, engaged submodes (KEAS HOLD, MACH HOLD, HEADING HOLD, or AUTO NAV) must be reengaged. On the ground, operation of the switch allows the rudder pedals to be used for nose wheel steering. Autopilot disengagement occurs if the related autopilot switch or SAS engage switch is turned off, the autopilot disengage switch on the control stick grip is pressed, or the ATT REF SELECT switch is set to the opposite position.

2.9.3.3 Mach Trim. The Mach trim system, operational when the pitch autopilot is disengaged, provides speed stability for the aircraft while accelerating or decelerating through the Mach 0.2 to 1.5 range. The system compensates, through operation of the pitch trim actuator in the mixer assembly, for the tendency of the aircraft to nose down while accelerating and to nose up while decelerating through this Mach range. The system also restores conventional stick forces and trim requirements under those Mach conditions.

2.9.4 <u>Automatic Pitch Warning (APW) System.</u> The APW system is controlled by a switch on the pilot's instrument panel. The system has two functions. The first function, enabled with the APW switch set to SHAKER ONLY, is to warn the pilot, by shaking of the control stick and illumination of a warning light, when the aircraft is approaching a critical pitch-up boundary. The boundary, which can be selected for display on the pilot's attitude director indicator (ADI), is a function of alpha (angle of attack) and pitch-rate summation versus Mach number. The stick shaker and warning light are also activated, exclusive of APW system operation, whenever a high angle of attack occurs; at 14 degrees for low speed flight (less than approximately 1.5 Mach) and at 8 degrees for high speed flight. The second function of the APW system, enabled with the APW switch set to PUSHER/SHAKER, is to assist the pilot in recovery of the aircraft when a critical pitch-up boundary; also related to excessive angle of attack, pitch rate, and Mach number; is exceeded during subsonic flight with the gear up and locked. When the boundary is exceeded, the pitch autopilot is disengaged and the stick pusher is activated, causing the elevon control surfaces and the control stick to move in a pitch down direction. The pilot must complete the recovery initiated by the stick pusher. If required, the pusher can be overpowered by the pilot through application of a force of approximately 30 pounds. Figure 2-26 is a diagram of the APW system.

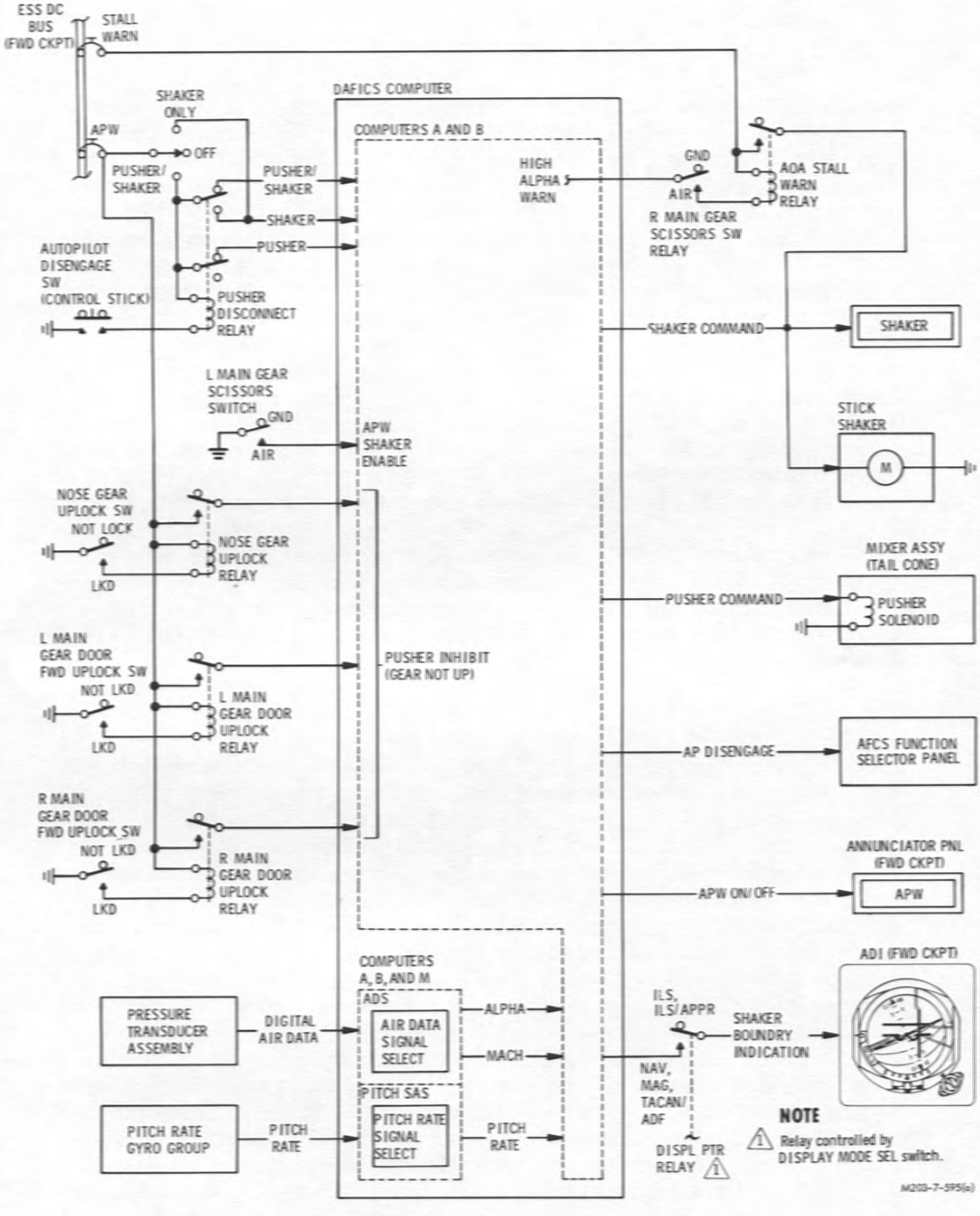


Figure 2-26. Shaker/Pusher Control Diagram

2.9.5 Air Inlet Control System (AICS). The breadth of the speed and altitude envelopes of the aircraft necessitates a variable-geometry air inlet system. The AICS operates to supply a flow of air, at correct pressure and velocity, to the engines throughout the flight envelope. At cruise speeds, the AICS not only provides a flow of subsonic air to the engines but also contributes to overall powerplant thrust. The system includes inlet air ducts in which spikes (pointed conical bodies) are translated forward and aft to capture and retain the shock wave that translates supersonic air to subsonic velocity. Forward bypass doors operate to assist the spikes in shock wave retention. The system is normally operated in the automatic mode using DAFICS computer control; however, it can also be manually controlled by the pilot. Additionally, a manually-controlled aft bypass door system is used to reduce aerodynamic drag and improve the efficiency of the forward bypass doors at higher Mach conditions. Figure 2-27 illustrates AICS components, in addition to the primary DAFICS components shown in figure 2-21. Figure 2-28 is a schematic of the AICS.

In operation, the spikes are moved forward and aft within the inlet duct as a function of Mach number, varying the size of the inlet throat areas and position of the oblique shock wave (wave extending backward from spike tip) and normal shock wave (wave at right angle to air flow). The forward bypass doors are modulated (towards open or closed) to control inlet duct air pressure and, therefore, fine tune the position of the normal shock wave relative to the inlet throat. The doors also operate to prevent excessive duct air pressure. The spikes, forward bypass doors, and aft bypass doors are moved by hydraulic actuators powered by the aircraft L and R hydraulic systems. The L system provides hydraulic pressure for the left inlet system, the R system provides hydraulic pressure for the right system. There is no crossover between the L and R hydraulic systems, and the systems have proven to be extremely reliable.

Operation of a spike and bypass doors, and resulting airflow patterns, is depicted in figure 2-29. At altitudes below 30,000 feet and speeds of less than Mach 1.4, the spike is locked fully forward. At altitudes above 30,000 feet, with DAFICS control, the spike begins to move aft as Mach increases above 1.6. The spike is automatically scheduled aft as a function of Mach number biased by alpha (angle of attack), beta (angle of side slip), and vertical acceleration. Aft movement of the spike properly positions the oblique and normal shock waves relative to the inlet and increases the inlet contraction ratio (inlet area versus throat area). At Mach 3.2, the spike has moved 26 inches aft of its full forward position and the captured airstream tube area has increased 112 percent, from 8.7 square feet to 18.5 square feet. Meanwhile, the throat has closed down 54 percent, from 7.7 to 4.16 square feet. A peripheral "shock trap" bleed slot, around the inside surface of the duct just forward of the throat, removes duct boundary layer air which is ducted aft and exhausted through the ejector nozzle. Spike boundary layer air is removed at a porous bleed section in the spike surface at its maximum diameter. This air is ducted through the spike body and supporting struts then exhausted overboard through nacelle louvers.

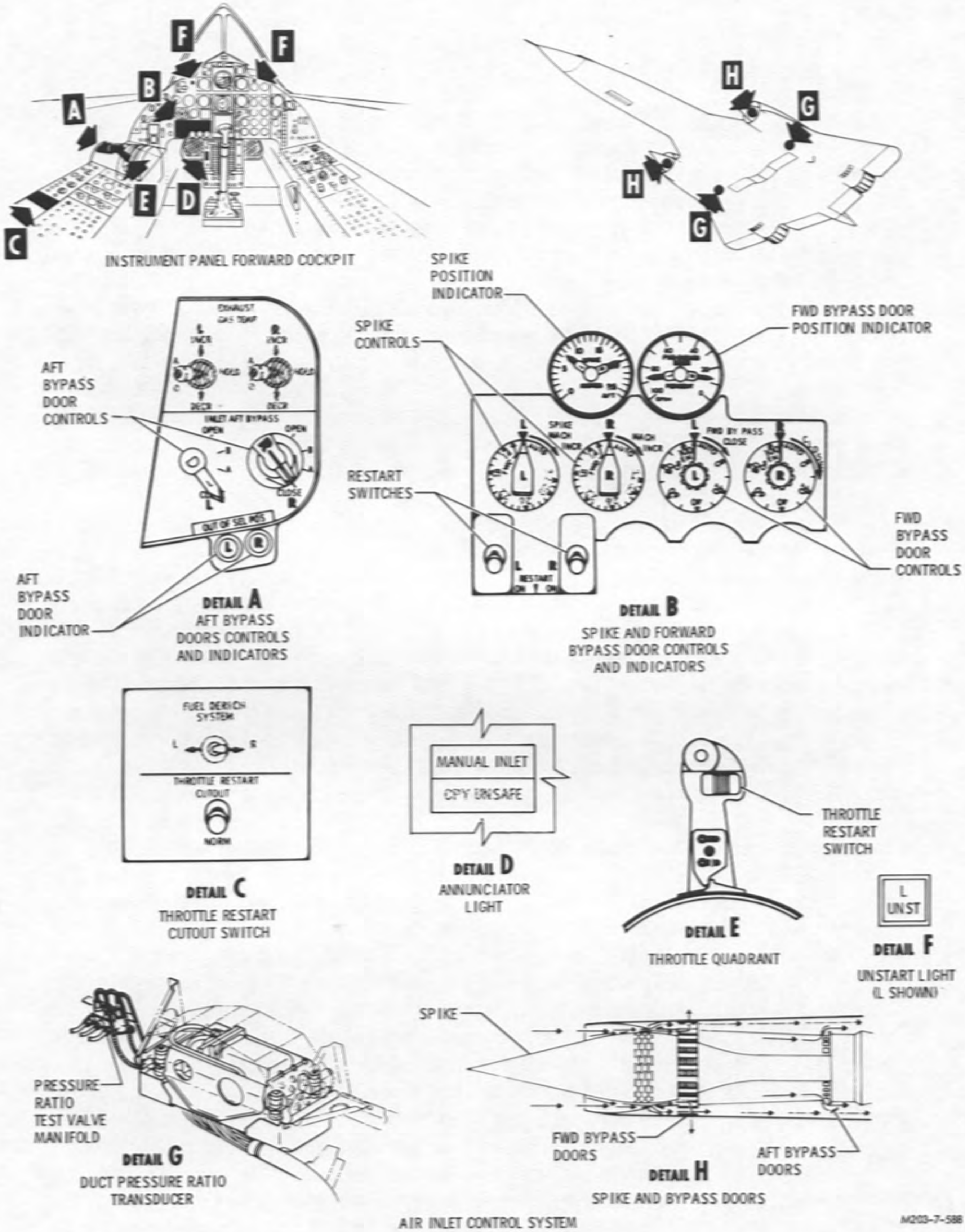


Figure 2-27. AICS Components

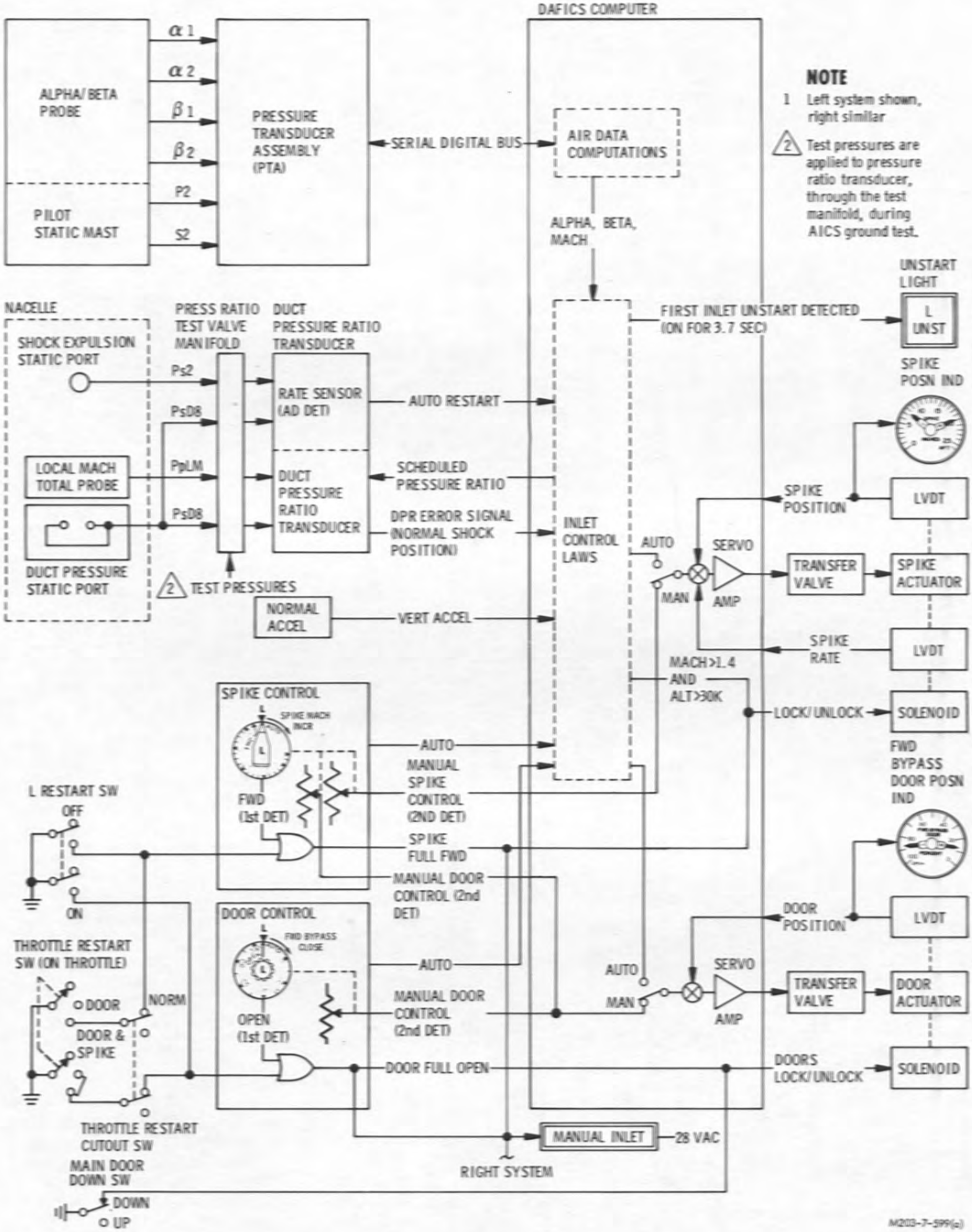


Figure 2-28. Inlet Control Diagram

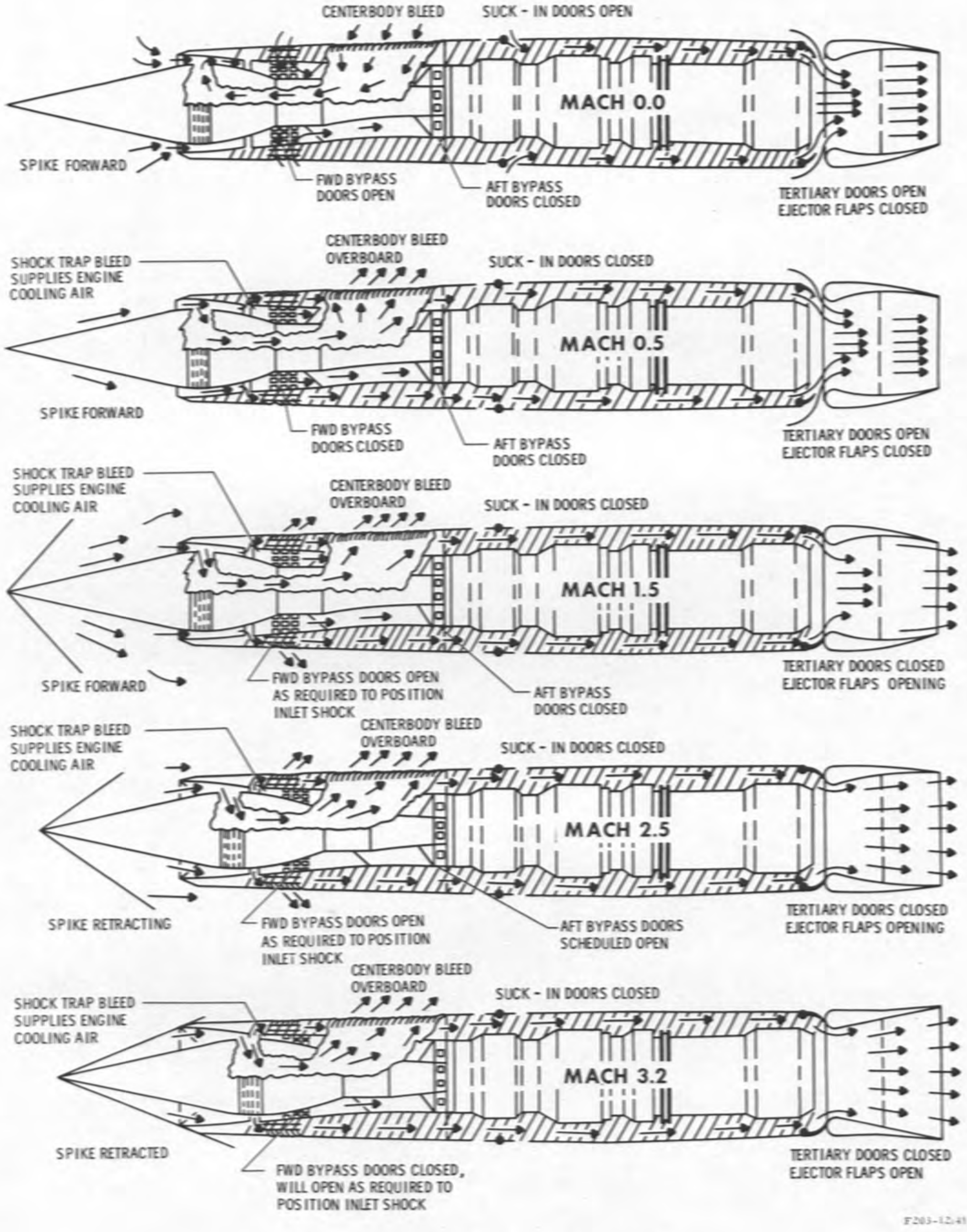


Figure 2-29. Inlet Airflow Diagram

The forward bypass doors consist of two concentric, annular bands, located just aft of the inlet throat. The outer band is rotated slightly about the stationary inner band so that rectangular openings in the two bands are shifted from full correspondence (doors fully open) to where the openings are covered (doors fully closed). The doors are fully open when the landing gear are extended and fully closed when the gear are retracted. The doors remain closed until Mach 1.5 is reached above 30,000 feet, at which point, with DAFICS control, they modulate open or closed, as a function of sensed inlet pressures and a pressure ratio schedule, to keep the normal shock wave near the inlet throat. The schedule is followed by comparing the ratio of sensed internal duct pressures (identified as PsD8 pressures) with external pressures sensed by pitot probes on the nacelle exterior surface (identified as PpLM pressures). As shown in figure 2-28, the sensed pressures and the pressure ratio schedule are compared within a duct pressure ratio transducer (DPRT). Any difference between the sensed pressure ratio and the pressure ratio schedule is used as a signal to drive the doors more open or more closed.

The aft bypass doors, roller-cage-type doors similar to the forward bypass doors, are pilot controlled. The aft bypass doors are opened at higher Mach to reduce aerodynamic drag created by excessive dumping of air overboard through the forward bypass doors. Opening the aft doors relieves duct internal pressures permitting the forward doors to operate closer to a closed condition, with reduced dumping of air and less drag.

2.9.5.1 DAFICS (Automatic) Control. When selected with the spike and forward bypass door controls, as described in the following paragraphs, DAFICS automatically controls operation of the spikes and the forward bypass doors. In developing control commands, DAFICS computers use Mach, alpha, and beta signals generated by the ADS from redundant input signals. Nonredundant duct pressure ratio error signals, from the left and right inlet DPRTs, are cross-connected to DAFICS A and B DCUs, providing computer redundancy for these signals. The left and right systems share a common vertical accelerometer signal.

There are two SPIKE controls on the pilot's instrument panel, one for each inlet. With the controls set to AUTO, the spikes are controlled by DAFICS, as previously described. A dual-needle SPIKE indicator, on the pilot's instrument panel, displays the positions of the spikes in inches aft of their full forward position.

There are two FWD BYPASS (door) controls on the pilot's instrument panel. With the controls set to AUTO, the forward bypass doors are controlled by DAFICS as previously described. A dual-needle DOOR indicator, located on the pilot's instrument panel, displays the positions of the forward doors as a percentage of their full open position.

2.9.5.2 Inlet Aerodynamic Disturbance (AD), or Unstart. An AD is a sudden pressure change in the inlet that results in the loss of normal shock wave control. The shock wave moves forward from its controlled position in the inlet throat and may even be expelled from the duct. The inlet is then in an unstarted condition. An AD detection circuit senses such a event and initiates inlet restart.

Static duct pressure (identified as Ps2) is sensed near the face of the engine. When an AD occurs there is a sudden decrease in Ps2 pressure. The pressure change is sensed in the DPRT, which develops an auto restart signal. This signal causes, for 3.75 seconds, the spike to move forward and the forward bypass doors to open. Forward movement of the spike reduces inlet throat contraction and opening of the doors reduces back pressure, thereby accelerating duct airflow and returning the normal shock to its desired position in the inlet throat. After the 3.75-second period, the spike and forward bypass doors, over a period of 10 seconds, slowly return to their scheduled positions. At speeds above Mach 2.3, the restart signal is applied to both inlets even though a problem is sensed on only one side. This reduces yaw due to asymmetric thrust and also prevents a sympathetic unstart of the other inlet, which may occur during a severe yaw condition. Restart crosstie is not in effect below Mach 2.3, allowing independent inlet restart operation at slower speeds.

2.9.5.3 Manual Control. The forward bypass doors can be manually controlled while the spikes are automatically controlled; however, if the spikes are manually controlled, the forward doors are also.

Manual control of the forward bypass doors is obtained when the FWD BYPASS (doors) controls are positioned at or between settings of 0 (doors closed) and 100 (doors open). Manual control of the spikes is obtained when the SPIKE controls are positioned at settings from FWD to 3.2 (Mach). As the controls are repositioned by the pilot to the correct Mach number, the spikes are moved to the corresponding position within the inlets and the forward bypass doors automatically operate in accordance with a schedule that prevents inadvertent door closure and an inlet unstart. The pilot can take manual control of an inlet unstart by using RESTART switches on the instrument panel or by operating a restart switch on the right engine throttle lever.

2.9.5.4. Aft Bypass Door Control. The aft bypass doors are manually controlled from two, four-position rotary switches, one for each set of doors, located outboard of the throttle quadrant. L and R OUT OF SEL POS lights advise the pilot when the doors are in transit or differ in position from that selected.

2.10 ELECTRICAL SYSTEM.

The electrical system includes an ac power generation and distribution system and a dc power generation and distribution system. Figure 2-30 is a simplified block diagram of the electrical system showing the principal busses and interconnection of the major components, all of which, with exception of the generators, control switches and caution lights, and external power connector, are located in the E-bay. Figure 2-31 shows the manual controls and caution lights associated with the system. The ac and dc systems have an abundance of excess capacity, as specified in Section III. The ac system is little affected by sudden changes in electrical loading and has excellent frequency stability due to its capacity and the parallel operation of the generators.

2.10.1 AC System. The ac system is a 115/200 volt, 3-phase, 400 Hz system. Primary power is supplied by two 60 KVA generators, mounted in the nacelles and engine driven through constant speed drive (CSD) units, part of the Accessory Drive Systems (ADS). A backup source of emergency power is provided by a 1 KVA solid-state inverter. The generators normally operate in parallel and supply power to five ac busses: a left generator bus, a right generator bus, an essential ac bus, an emergency ac bus, and a hot ac bus. Additionally, there are 26 volt essential and emergency ac busses. A connector in the nose wheel well permits the application of 115/220 volt, 3-phase, 400 Hz external ground power to the ac busses in lieu of generator power for aircraft maintenance and during preflight before engine start. Bus tie contactors close during application of ground electrical power to the aircraft and remain closed to provide parallel generator operation when the generators are brought on line. Electrical power phase relationships are used to control ac power source paralleling; first, the right generator with ground power, then the left generator with the right.

As shown in figure 2-30, the left and right generator busses supply power to the fuel boost pumps and, through relays, to the hot bus, the emergency ac bus, and the essential ac bus. The hot bus supplies power for fuel transfer and crossfeed, pitch and yaw trim, and for forward cockpit instrument lights. The emergency ac bus supplies power for fire warning, DAFICS, and other systems and instruments essential to flight. The essential bus supplies power to the balance of systems and equipment requiring ac power. The 26 volt emergency and essential ac busses, respectively, provide power for instruments essential to and not essential to flight.

2.10.2 DC System. The dc system is nominally 28 volts, with actual line voltage closer to 29 volts, depending on load. DC power is normally supplied by two 200-ampere transformer-rectifiers, separately powered from the left and right generator ac busses. The transformer-rectifiers supply dc power to a monitored dc bus and to the No. 1 and No. 2 essential dc busses. The essential dc busses provide

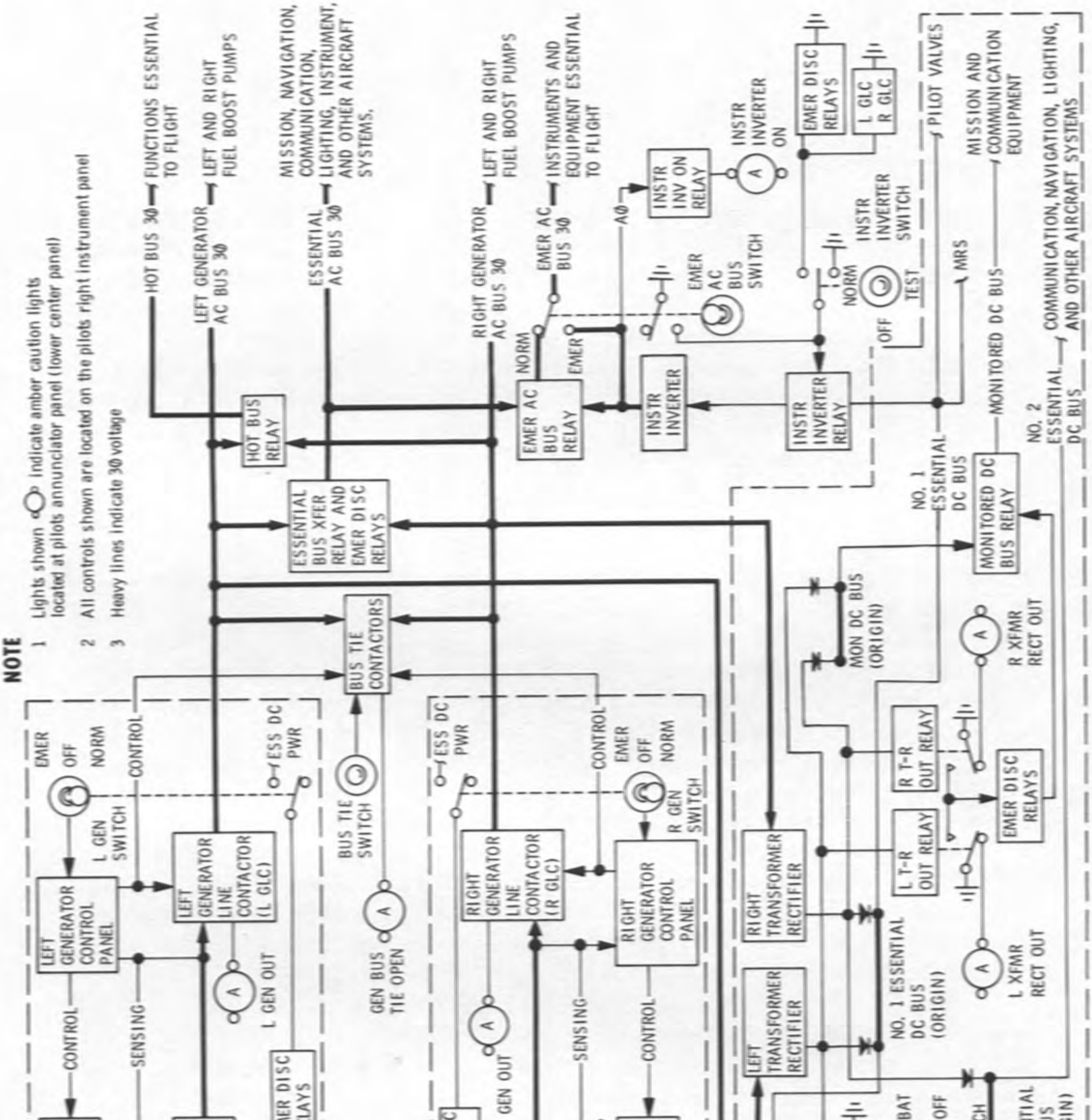


Figure 2-30. Electrical System Block Diagram

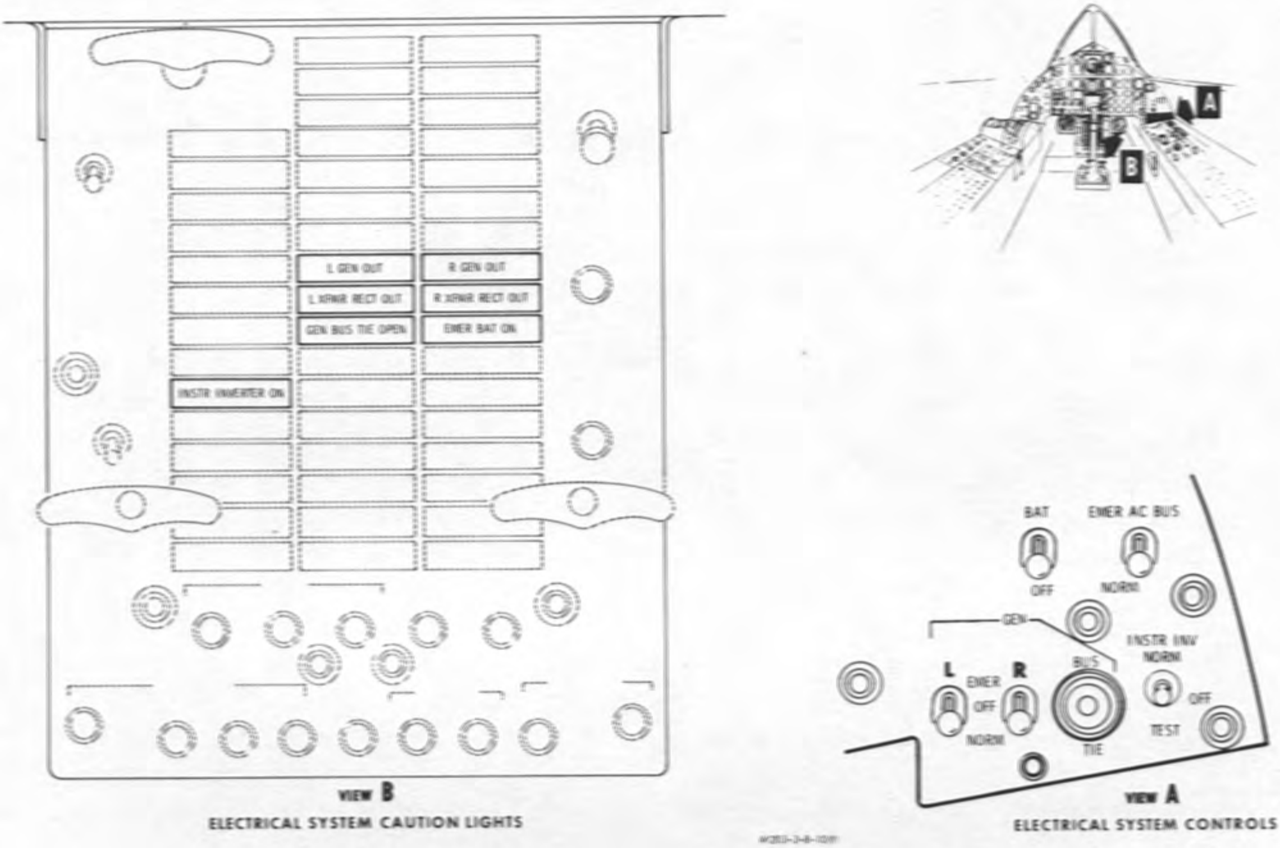


Figure 2-31. Electrical System Controls and Caution Lights

power for flight-essential systems and equipment, including the instrument inverter, if turned on. The monitored dc bus provides power for payloads and non-essential communication systems. A dual battery is normally connected to and floats on the essential dc busses.

2.10.3 System Operation. Electrical system operation is automatic. Under normal conditions, the generators operate in parallel to supply power to all the ac busses and, through the transformer-rectifiers, power to the dc busses. Caution lights, on the pilot's annunciator panel, advise of significant problems or conditions.

If a generator or ac bus fault occurs, it can lead to either single generator operation or to independent (non-parallel) generator operation. Single generator operation occurs when the operable generator supplies power to all the ac busses. This mode is initiated, following removal of a faulted generator from the line, by automatic reclosure of the bus tie contactors or by closure through pilot operation of the BUS TIE switch. Independent generator operation results in the generators independently supplying power to their respective ac busses or, because the bus tie cannot be closed because it would tie an operating generator to a faulted bus, causes the operable generator to supply power to all the ac busses except the generator bus on the faulted side. In all cases there is more than adequate reserve ac power to handle the load requirements of the aircraft; however, in the last case, with a generator bus deenergized and half of the fuel boost pumps inoperable, there will be reduced afterburner thrust at lower altitudes. The pilot can attempt to reset a faulted generator by operation of the related generator switch.

In the event both generators fail and cannot be reset, the generators may be operated in an emergency mode in which generator voltage and frequency are unregulated, the bus tie contactors are open, and the essential ac and monitored dc busses are deenergized. The principal purpose of emergency generator operation is to supply power to the fuel boost pumps and, from the hot bus, to supply power for fuel crossfeed, fuel transfer, pitch and yaw trim, and forward cockpit instrument lights. If either generator fails or is placed in emergency operation, the instrument inverter is automatically turned on to supply power to the emergency ac bus and to DAFICS and other systems required for flight. In the event of a dual generator failure or dual transformer-rectifier failure, the monitored dc bus is deenergized and the dual battery supplies power to the essential dc busses.

2.10.4 Lighting Systems. The aircraft lighting systems include interior lights and exterior lights. Cockpit instruments and control panels are illuminated by integral lighting. In addition, there are flood lights, for side console lighting, and hand-controllable and variable spot/utility lights in each cockpit. Flood lights are also located on the pilot's left and right glareshields and above his right console, as emergency lights. Thunderstorm lights, facing forward on each side of the pilot, provide steady cockpit illumination during thunderstorms and lightning. The pilot's instrument lights remain operable as long as either generator is in operation (in normal or emergency mode). The pilot's emergency flood lights and thunderstorm lights, and the spot/utility lights in both cockpits, remain operable even when only battery power is available.

Separate landing and taxi lights are located on the nose landing gear. Navigation lights consist of red-flashing strobe lights (beacons) and white tail lights. The beacons are located on the top and bottom of the aircraft fuselage, halfway back from the aircraft nose. The tail lights are mounted on the top and bottom of the tail cone. Lights in the air refueling receptacle come on when the pilot initiates aerial refueling.

2.10.5 Fire Warning System. A fire warning system is provided to advise the pilot of a fire condition in the nacelles. The system includes dual sensing elements, to prevent false warning indications, mounted in each nacelle as shown in figure 2-32. If the temperature at the elements exceeds their trip level, a red warning light illuminates on the pilot's instrument panel.

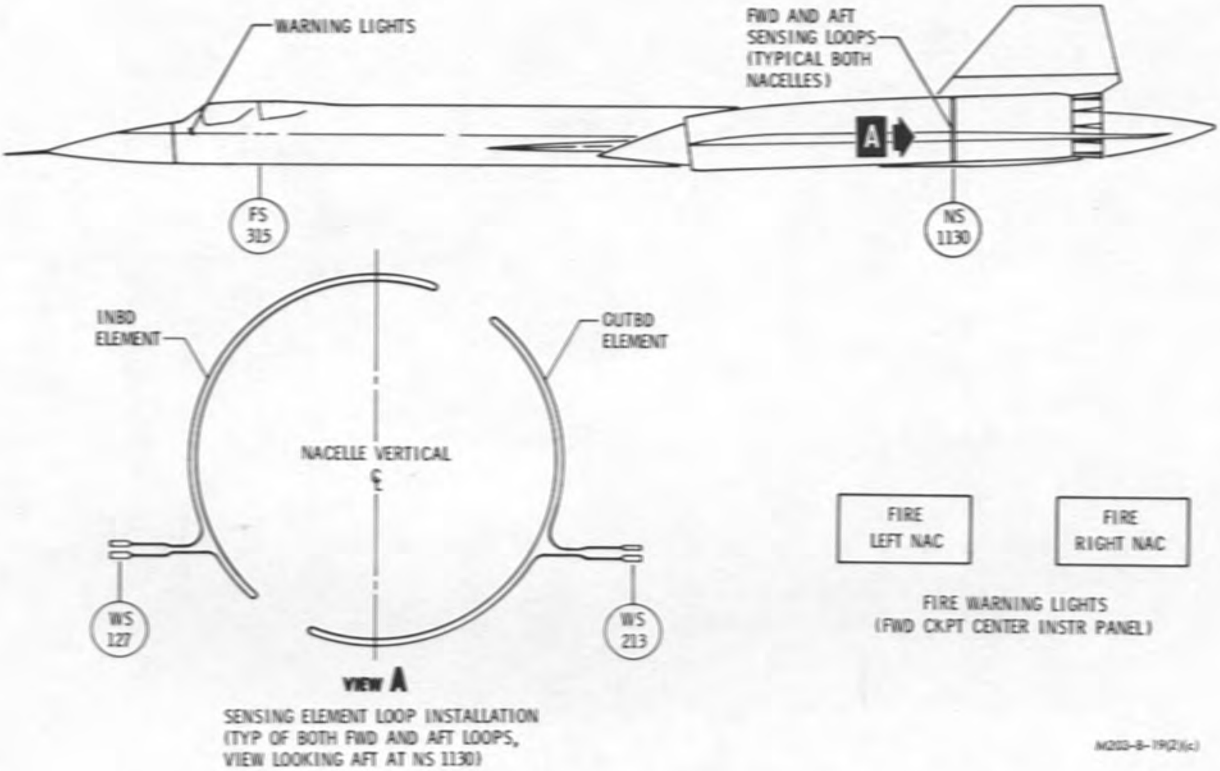


Figure 2-32. Fire Warning Sensing Loops and Warning Lights

2.11 INSTRUMENTS.

Forward and aft cockpit instrument panels and side consoles are shown in figures 2-33 and 2-34 for the SR-71A aircraft, in figures 2-35 and 2-36 for the SR-71B aircraft. In addition to the instruments associated with the systems described in this section, there is a pitot static and alpha beta system, and specialized instrument systems. These include the: rear-view periscope, peripheral vision display, map projectors, viewsight, V/H system, sensor event/frame count system, and exposure control system.

2.11.1 Pitot Static and Alpha Beta Systems. The aircraft is equipped with a dual pitot static system and an alpha beta system. The pitot static system provides air pressures reflecting static and dynamic (ram) air pressure conditions sensed at the pitot mast. (See figure 2-37.) As shown in figure 2-23, one set of pitot static pressures (P1 and S1) are supplied to the pilot's pressure instruments: altimeter, airspeed indicator, and rate-of-climb indicator. The other set of pitot static pressures (P2 and S2) are applied to the pressure transducer assembly (PTA) in the B2 bay. Alpha and beta pressures, representing angle of attack and angle of sideslip, are sensed at a probe attached to the pitot mast. The alpha and beta pressures (A1, A2, B1, and B2) are also applied to the PTA. The PTA converts the pitot static and alpha beta signals into digital equivalents which are supplied to the air data subsystem of the DAFICS.

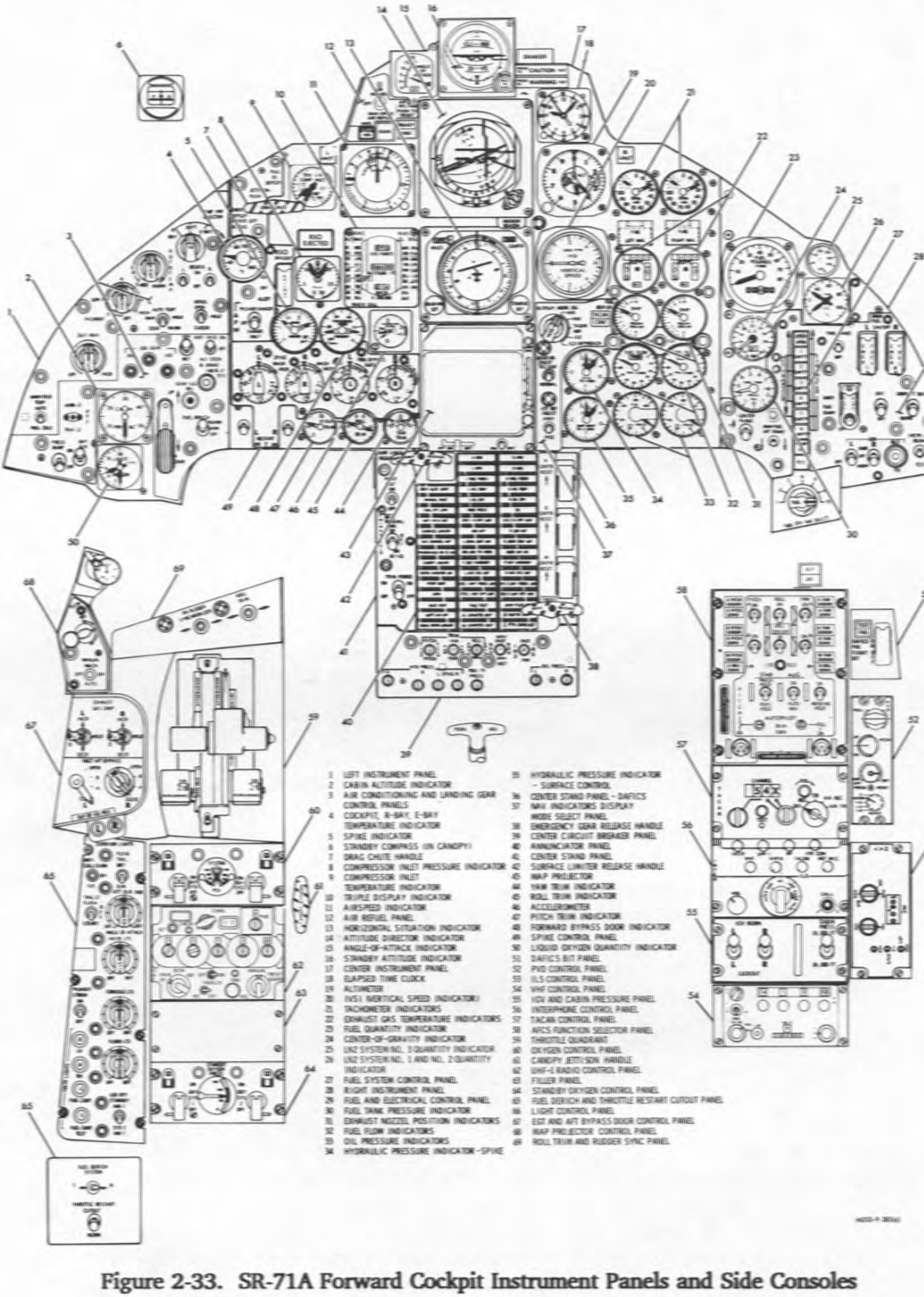
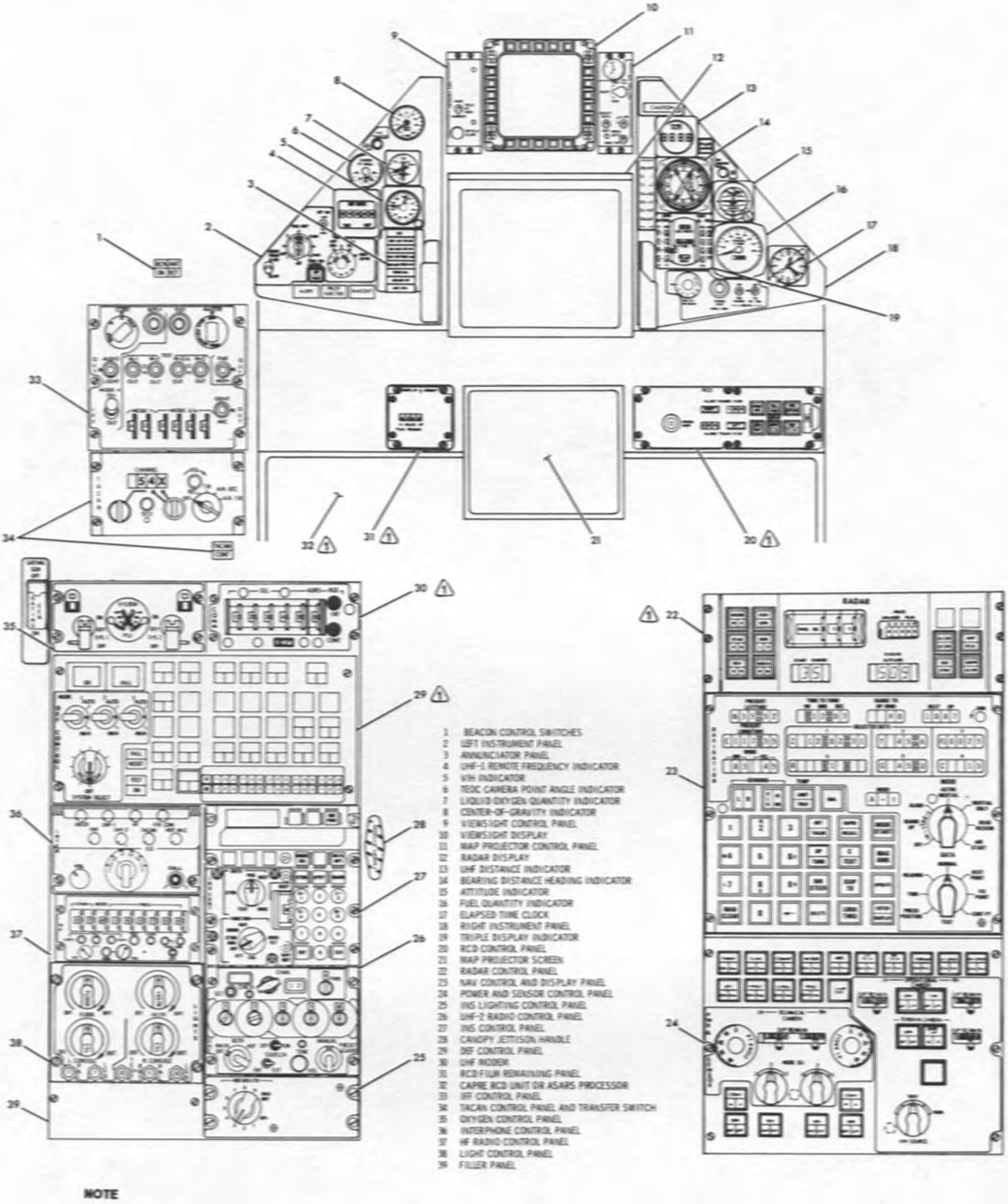


Figure 2-33. SR-71A Forward Cockpit Instrument Panels and Side Consoles



NOTE

⚠ Not available

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Figure 2-34. SR-71A Aft Cockpit Instrument Panels and Side Consoles

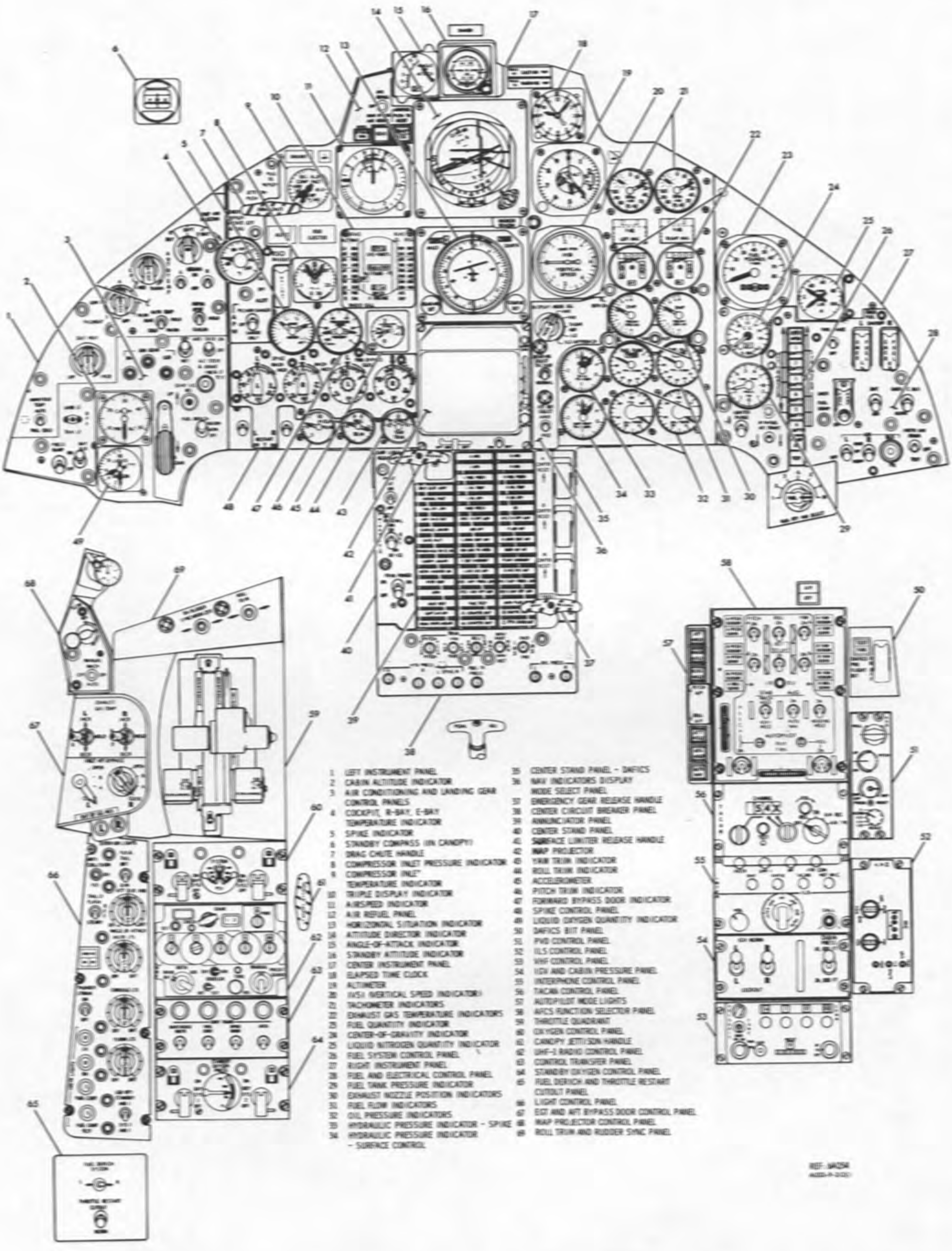
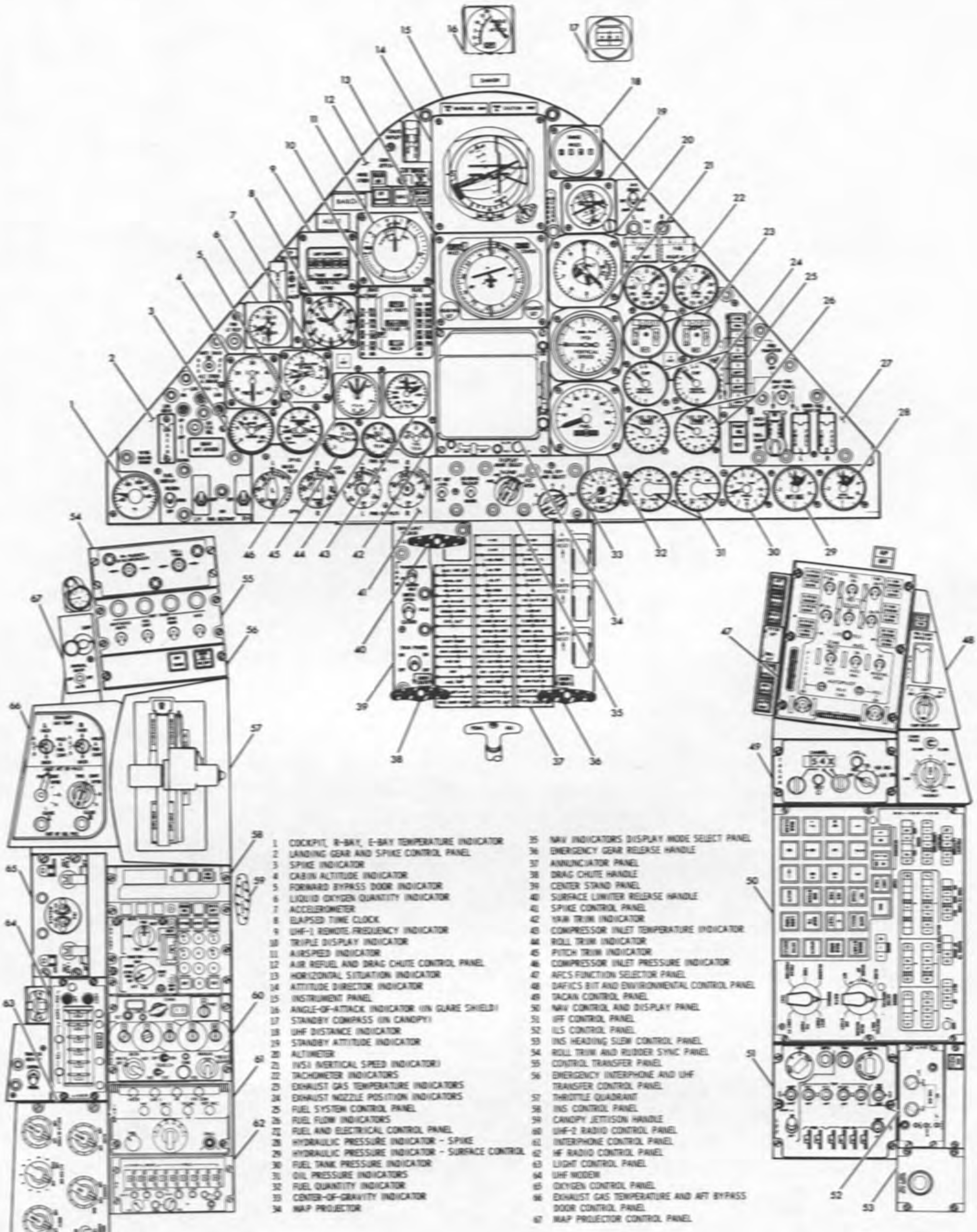


Figure 2-35. SR-71B Forward Cockpit Instrument Panels and Side Consoles



1 COCKPIT, R-BAY, E-BAY TEMPERATURE INDICATOR
2 LANDING GEAR AND SPIKE CONTROL PANEL
3 SPIKE INDICATOR
4 CABIN ALTITUDE INDICATOR
5 FORWARD BYPASS DOOR INDICATOR
6 LIQUID OXYGEN QUANTITY INDICATOR
7 ACCELEROMETER
8 ELAPSED TIME CLOCK
9 UHF-1 REMOTE FREQUENCY INDICATOR
10 TRIPLE DISPLAY INDICATOR
11 AIRSPEED INDICATOR
12 AIR REFUEL AND DRAG CHUTE CONTROL PANEL
13 HORIZONTAL SITUATION INDICATOR
14 ATTITUDE DIRECTOR INDICATOR
15 INSTRUMENT PANEL
16 ANGLE-OF-ATTACK INDICATOR (IN GLARE SHIELD)
17 STANDBY COMPASS (IN CANOPY)
18 UHF DISTANCE INDICATOR
19 STANDBY ATTITUDE INDICATOR
20 ALTIMETER
21 (VSI) VERTICAL SPEED INDICATOR)
22 TACHOMETER INDICATORS
23 EXHAUST GAS TEMPERATURE INDICATORS
24 EXHAUST NOZZLE POSITION INDICATORS
25 FUEL SYSTEM CONTROL PANEL
26 FUEL FLOW INDICATORS
27 FUEL AND ELECTRICAL CONTROL PANEL
28 HYDRAULIC PRESSURE INDICATOR - SPIKE
29 HYDRAULIC PRESSURE INDICATOR - SURFACE CONTROL
30 FUEL TANK PRESSURE INDICATOR
31 OIL PRESSURE INDICATORS
32 FUEL QUANTITY INDICATOR
33 CENTER-OF-GRAVITY INDICATOR
34 MAP PROJECTOR
35 NAV INDICATORS DISPLAY MODE SELECT PANEL
36 EMERGENCY GEAR RELEASE HANDLE
37 ANNUNCIATOR PANEL
38 DRAG CHUTE HANDLE
39 CENTER STAND PANEL
40 SURFACE LIMITER RELEASE HANDLE
41 SPIKE CONTROL PANEL
42 YAW TRIM INDICATOR
43 COMPRESSOR INLET TEMPERATURE INDICATOR
44 ROLL TRIM INDICATOR
45 PITCH TRIM INDICATOR
46 COMPRESSOR INLET PRESSURE INDICATOR
47 AFCS FUNCTION SELECTOR PANEL
48 DAFICS BIT AND ENVIRONMENTAL CONTROL PANEL
49 TACAN CONTROL PANEL
50 NAV CONTROL AND DISPLAY PANEL
51 IFF CONTROL PANEL
52 ILS CONTROL PANEL
53 INS HEADING SLEW CONTROL PANEL
54 ROLL TRIM AND RUDDER SYNC PANEL
55 CONTROL TRANSFER PANEL
56 EMERGENCY INTERPHONE AND UHF TRANSFER CONTROL PANEL
57 THROTTLE QUADRANT
58 INS CONTROL PANEL
59 CANOPY JETTISON HANDLE
60 UHF-2 RADIO CONTROL PANEL
61 INTERPHONE CONTROL PANEL
62 HF RADIO CONTROL PANEL
63 LIGHT CONTROL PANEL
64 UHF MODEM
65 OXYGEN CONTROL PANEL
66 EXHAUST GAS TEMPERATURE AND AFT BYPASS DOOR CONTROL PANEL
67 MAP PROJECTOR CONTROL PANEL

Figure 2-36. SR-71B Aft Cockpit Instrument Panels and Side Consoles

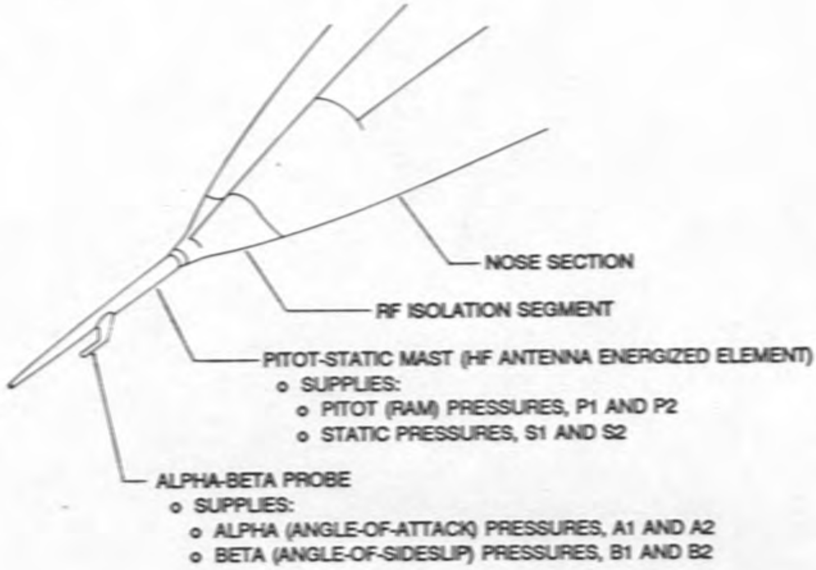


Figure 2-37. Pitot Mast

2.11.2 Rear-View Periscope. A manually-extendable periscope is mounted in the top of the pilot's canopy in the SR-71A, both canopies on the SR-71B. Cockpit pressure assists in extension and retention of the periscope in the up position. The field of view of the instrument is an aft-looking cone, approximately 10 degrees across. Head movement extends the viewing angle to approximately 30 degrees. Additionally, the periscope can be rotated horizontally up to 10 degrees each side of the aircraft centerline. Accordingly, the periscope total viewing angle is 30 degrees in the vertical plane and 50 degrees in the horizontal plane as shown in figure 2-38.

2.11.3 Peripheral Vision Display (PVD). The PVD is a twilight and night attitude orientation device that projects a laser-generated, thin red line on the pilot's instrument panel. The display, pitch and roll attitude stabilized, provides the pilot with peripheral-vision subconscious support for spatial orientation. Pitch and roll control signals are from the ANS or the INS, as selected with the pilot's ATT REF SELECT switch. (In the SR-71B, only the forward (student pilot's) cockpit is equipped with a PVD.)

2.11.4 Map Projectors. Each cockpit has a map projector which can display a moving strip map of the route to be flown plus ancillary and emergency data. The pilot's projector holds a single film. On SR-71A aircraft, the aft cockpit projector can hold two separate films, one of which, as with the pilot's projector, moves so as to reflect motion of the aircraft over the ground. (In the SR-71B, both projectors are pilot projectors.) Film motion is normally automatically controlled by signals supplied from the V/H system; however, film motion can also be manually controlled.

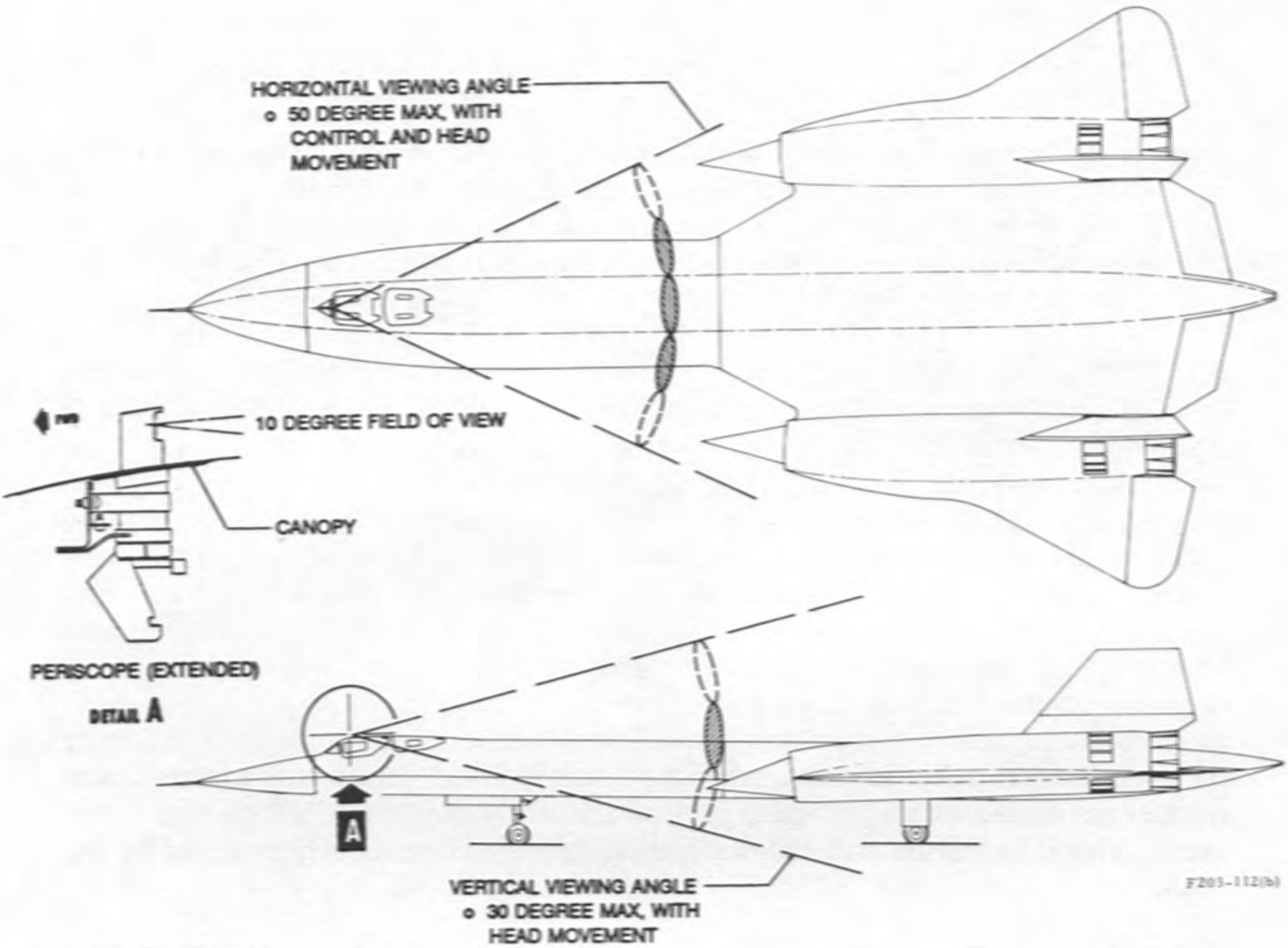


Figure 2-38. Rear-View Periscope Fields of View

2.11.5 Viewsight. A video viewsight system is provided in the aft cockpit of SR-71A aircraft. (There is no viewsight in the SR-71B.) The viewsight display is a black and white video presentation of terrain below and forward of the aircraft. The viewsight, primarily used during daylight hours, can be used to provide position update information to the ANS. The instrument has two selectable fields of view, a wide-angle field of view, 114 degrees wide, and a narrow-angle field of view, 44 degrees wide. The instrument functions as a demagnification device with a power of one-fifth and one-half, respectively, for the wide and narrow fields of view.

2.11.6 V/H System. A V/H system is provided in SR-71A aircraft. (There is no V/H system in the SR-71B.) The system produces dc voltages scaled to represent the angular rate of motion between the aircraft and the terrain over which it is moving. Signal sources for the system are the Astroinertial Navigation System (ANS) and a V/H indicator, see figure 2-34, on the aft cockpit instrument panel. The ANS V/H signal, accurately computed and continuously updated by the ANS, is displayed by one of two needles of the V/H indicator. The signal generated by the V/H indicator represents the manual setting of the other needle. Manual V/H, a backup for ANS V/H, is as accurate as the needle setting on the indicator. ANS or manual V/H signal selection is made using a switch on the Power and Sensor Control Panel, located on the aft cockpit right console. System output signals are scaled at 0.20 volts dc per milliradian per second.

2.11.7 Sensor Event/Frame Count System. The sensor event/ frame count system was designed to correlate images of LED frame counters, exposed on film in payload equipment, against ANS time and navigation data recorded in the MRS. The system, however, can also be used without LED frame counters for similar correlation of specific and unique events occurring within a payload system. The correlation process permits accurate establishment of the exact time, location, altitude, aircraft heading and attitude, etc., at which an event occurred. In addition to the ANS and MRS, the system uses a signal sensor processor, located in the R-bay. The aircraft system is fully automatic, with no manual controls. Event input signals to the signal sensor processor can vary in duration from 0.1 to 13 milliseconds. The processor converts the signals to 6-millisecond wide event marker pulses. The ANS monitors the event marker pulse lines and, when a pulse is received, increments a frame count register associated with a particular payload and stores the time (GMT) related to the event. This information and ANS navigation data are continuously recorded by the MRS.

2.11.8 Exposure Control System. An exposure control system is provided in SR-71A aircraft. (There is no exposure control system in the SR-71B.) The exposure control system was designed to provide a means of manually controlling the exposure settings of onboard cameras for different conditions of sun angle and terrain brightness. The exposure control, located on the aft cockpit instrument panel, is graduated in degrees of sun angle, with reference indices for low, normal, high, and very high terrain reflectivity. The exposure control system produces regulated dc voltages scaled between approximately 10 and 38 volts for sun angle settings between 5 degrees and 90 degrees, high reflectivity.

2.12 COMMUNICATION AND RADIO NAVIGATION SYSTEMS.

Aircraft communication systems include an intercom system, an HF transceiver system, a VHF transceiver system, a dual UHF and ADF system, and an IFF system. Radio navigation systems include conventional TACAN, ILS, and marker beacon equipment. All communication and radio navigation transceiver and receiver equipment, except for the IFF, is located in the R-bay. The IFF transceiver is located in the D-bay in SR-71A aircraft and in the N-bay in the SR-71B. (Refer to Section III, figure 3-2, for D, N, and R-bay locations.) See figures 2-33 through 2-36 for control panels and indicators associated with the com/radio nav systems.) Figure 2-39 is a block diagram of the systems.

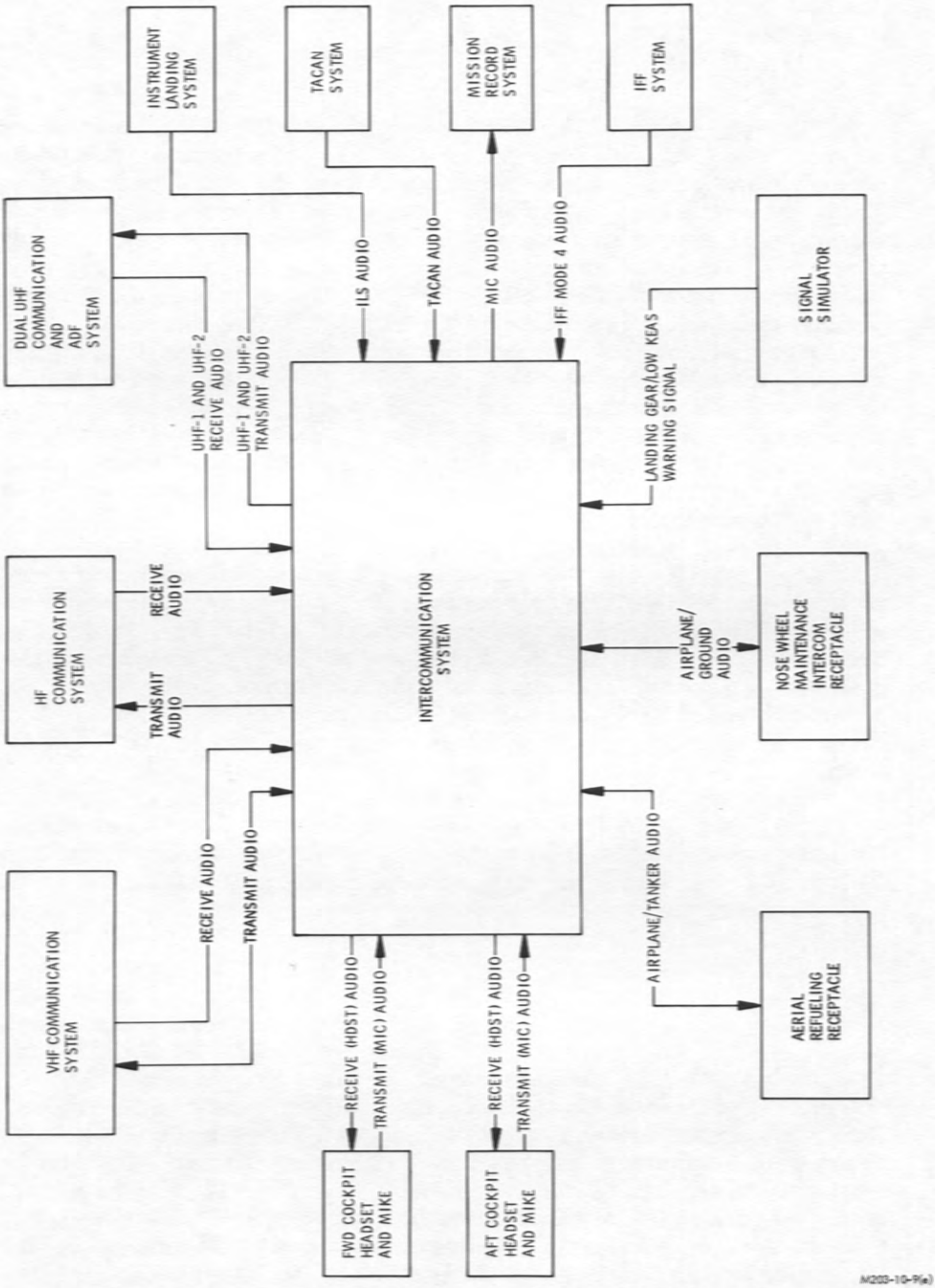


Figure 2-39. Communication and Radio Navigation Systems Block Diagram

2.12.1 Intercom System. The intercom system provides for voice communications between the cockpits, between the cockpits and the ground maintenance crew (at the nose wheel well), and, during aerial refueling, between the aircraft and the tanker via the refueling boom. The primary components of the system are the cockpit control panels and the intercom facilities in the nose wheel well and refueling receptacle. The control panels provide for monitoring selected radio and radio navigation receiver audios (HF, VHF, UHF-1, UHF-2, TACAN, and ILS) and for keying and modulating selected radio transmitters (HF, VHF, UHF-1, and UHF-2). Both cockpits have selectable hot microphone capability. (The SR-71B also has emergency intercom capability.) The aft cockpit, on the SR-71A only, has a selectable quiet listen feature which disconnects most incoming audio except intercom. Warning signals for KEAS below 250, or for landing gear not down and locked below 9,000 feet with throttles set below minimum cruise, are provided as headset audio to both cockpits. All intercom and radio transmissions by the flight crew are recorded by the Mission Recorder System (MRS).

2.12.2 VHF System. The VHF transceiver system, controlled from the forward cockpit, provides VHF AM radio communications, with receiving capability from 108.0 to 151.975 MHz and transmitting capability from 115.975 to 151.975 MHz. The system uses a blade antenna located on the lower left chine. (Antenna configuration may cause some degradation in performance at frequencies below 118.0 MHz and above 136.0 MHz.) In addition to its use for normal VHF communications, the dc-powered system provides for emergency radio communications in the event of a complete ac power failure in the aircraft.

2.12.3 HF System. The HF transceiver system, controlled from the aft cockpit, provides selectable sideband or equivalent AM radio communications in the 2 to 29.9999 Mhz range. The system energizes the RF-isolated pitot mast and the aircraft functions as an antenna.

2.12.4 Dual UHF and ADF System. The dual UHF transceiver system provides UHF AM communications in the 225.0 to 339.95 Mhz range. The system includes two identical UHF radios, identified as UHF-1 AND UHF-2, controlled from the forward and aft cockpits respectively. These radios are switchable between two blade antennas located on the lower forward left and lower aft right chines. For ADF operation, an ADF antenna replaces the forward blade antenna for radio reception. On SR-71A aircraft, bearing information to the received station is selectable for display on the Horizontal Situation Indicator (HSI) in the forward cockpit and on the Bearing, Distance, Heading Indicator (BDHI) in the aft cockpit. (On the SR-71B, bearing information is separately selectable for display on the HSIs in both cockpits.) With the addition of a classified line-replaceable unit (LRU), continuous ranging information to a cooperating and similarly equipped UHF station is displayed on a UHF distance indicator in the aft cockpit.

2.12.5 IFF System. An IFF system provides responses in modes 1, 2, 3/A, 4, and C. Mode C code data, encoded barometric altitude, is supplied from the air data subsystem of the DAFICS. The IFF system is controlled from the aft cockpit.

2.12.6 TACAN, ILS, and Marker Beacon Equipment. The ILS and marker beacon receivers are controlled from the forward cockpit in SR-71A aircraft. (On the SR-71B, the ILS receiver control is transferable between cockpits and the marker beacon receiver is always on.) There are TACAN control panels in both cockpits and system control is transferable between the cockpits. On SR-71A aircraft, ILS and marker beacon information is selectable for display on the HSI in the forward cockpit. TACAN range and bearing data is available to the HSI in the forward cockpit and to the BDHI in the aft cockpit. (On the SR-71B, ILS and marker beacon information, and TACAN range and bearing data, are selectable for display on the HSIs in both cockpits.)

2.13 NAVIGATION SYSTEMS.

Navigation systems include an Astroinertial Navigation System (ANS) and an Inertial Navigation system (INS). Both systems are controlled from the aft cockpit. Either the ANS or the INS can be selected as inputs to the autopilot, a subsystem of DAFICS, and cockpit instruments.

2.13.1 Astroinertial Navigation System (ANS). In its primary mode of operation (Astroinertial) the ANS utilizes a gyro-stabilized inertial platform to sense aircraft motion and employs star tracking to limit the buildup of navigational errors. In the Astroinertial mode of navigation, the circular error of probability (CEP), or probable radial error, is historically 0.5 nautical miles for periods of up to 10 hours and longer. The system can also be operated in backup modes, with reduced navigational accuracy; these modes, in descending order of accuracy, are: Inertial Only, Airstart, and Dead Reckon. The system is operated from a Control and Display Panel in the aft cockpit. This panel provides a continuous display of aircraft latitude, longitude, and other relevant navigational data.

One of the principal purposes of the ANS is that of an automatic navigator which, when the roll autopilot is in the AUTO NAV mode, causes the aircraft to accurately fly a predetermined flight path made up of a series of great circle legs. The flight plan is loaded into the system prior to flight. The plan, which can be modified through operation of the Control and Display Panel, prior to or during flight, defines the start, end, and intermediate points of a flight. These points are referred to as destination points (DPs). Each point represents a specific geographic location which is stored in the ANS as a unique four-digit code together with the location's exact latitude, longitude, elevation, and other relevant data. Aside from the start and end points, which are normally the same geographical point, all in-route destination points represent end points of the legs of the mission; points towards which the aircraft is directed but never reaches because it begins a turn to the next leg. Turn radius is normally based on aircraft groundspeed. In addition to destination points, the ANS utilizes fixpoints (FPs) and control points (CPs). If required for a specific mission, the exact location of these points can also be stored in the ANS as part of the flight plan. Fixpoints are used in conjunction with the viewsight to monitor and, if determined necessary, update ANS navigational data. Control points are used to define geographic locations at which the ANS, when the aircraft is abreast of the point, supplies on/off, operate, mode control, and other commands to payload equipment. Figure 2-40 is an example of a mission plan, showing destination and control points. Figure 2-41 shows the relationship between the principal units of the system: the Control and Display Panel, guidance group, and chronometer; and principal system interface signals.

When selected as a signal source, the ANS supplies aircraft attitude, heading, and course data to the following equipment:

- o DAFICS (autopilot): aircraft pitch, roll, heading, and steering (bank angle) command.

- o DAFICS (SAS): aircraft roll, pitch, and heading

- o Horizontal Situation Indicator (HSI), forward cockpit on SR-71A, both cockpits on SR-71B: aircraft true heading, command course, course deviation, and range to next destination point (DP) - marks end point of each mission leg

- o Attitude Director Indicator (ADI), forward cockpit on SR-71A, both cockpits on SR-71B: aircraft pitch, roll, vertical velocity, and, via a flight director computer, pitch steering and bank steering

- o Peripheral Vision Display (PVD), forward cockpit only on SR-71A and SR-71B: aircraft pitch and roll

- o Attitude indicator, aft cockpit on SR-71A only: aircraft pitch and roll.

Additionally, the ANS interfaces with the following equipment/systems:

- o Mission Recorder System (MRS): The ANS constantly supplies operational and flight profile data to the MRS. The operational data, analog and digital, is for ANS maintenance analysis. The digital flight profile data consists of continuously updated navigation and related information (GMT, latitude and longitude, altitude, pitch, roll, true heading, ground track and ground speed, range to and data regarding next DP, next control point (CP) data - data for payload control, as described in the following paragraph).

- o Payloads: At preselected geographic locations, control points (CPs), the ANS can supply on/off and other commands, in addition to pitch, roll, and yaw signals. Also, the ANS can provide the same digital ANS navigation data, described above, as recorded by the MRS.

- o Sensor Event/Frame Count System: The ANS can accept event signals supplied by payload systems and include the occurrence and time of the events with digital ANS navigation data recorded by the MRS, permitting later correlation of the events with the location, attitude and heading, etc. of the aircraft. If applicable, the ANS can also provide signals to update frame counters located in payload equipment.

- o Viewsight: The ANS can accept position (navigation) correction updates from the viewsight at predetermined geographic locations (fixpoints).

- o V/H System: The ANS supplies analog V/H (aircraft angular velocity relative to the ground) signals.

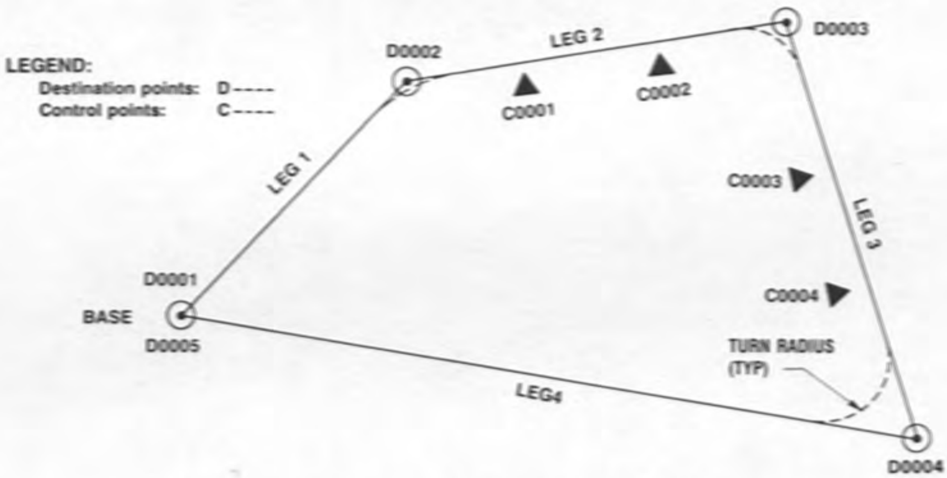


Figure 2-40. Typical ANS Mission Plan

2.13.2 <u>Inertial Navigation System (INS).</u> The INS, a backup navigation system, is a pure inertial system with a historical CEP of 1 nautical mile per hour. The primary components of the system include a control panel in the aft cockpit, which provides a readout of navigational data, and an inertial navigation unit, located in the R-bay, which contains the inertial platform. In addition to providing a continuous display of navigational data in the aft cockpit, the INS, when selected, supplies essentially the same signals to the autopilot and cockpit instruments as described for the ANS; however, the INS does not supply a steering (bank angle) command to the autopilot for Auto Nav navigation. INS pitch, roll, and heading signals are monitored by the Mission Recorder System (MRS).

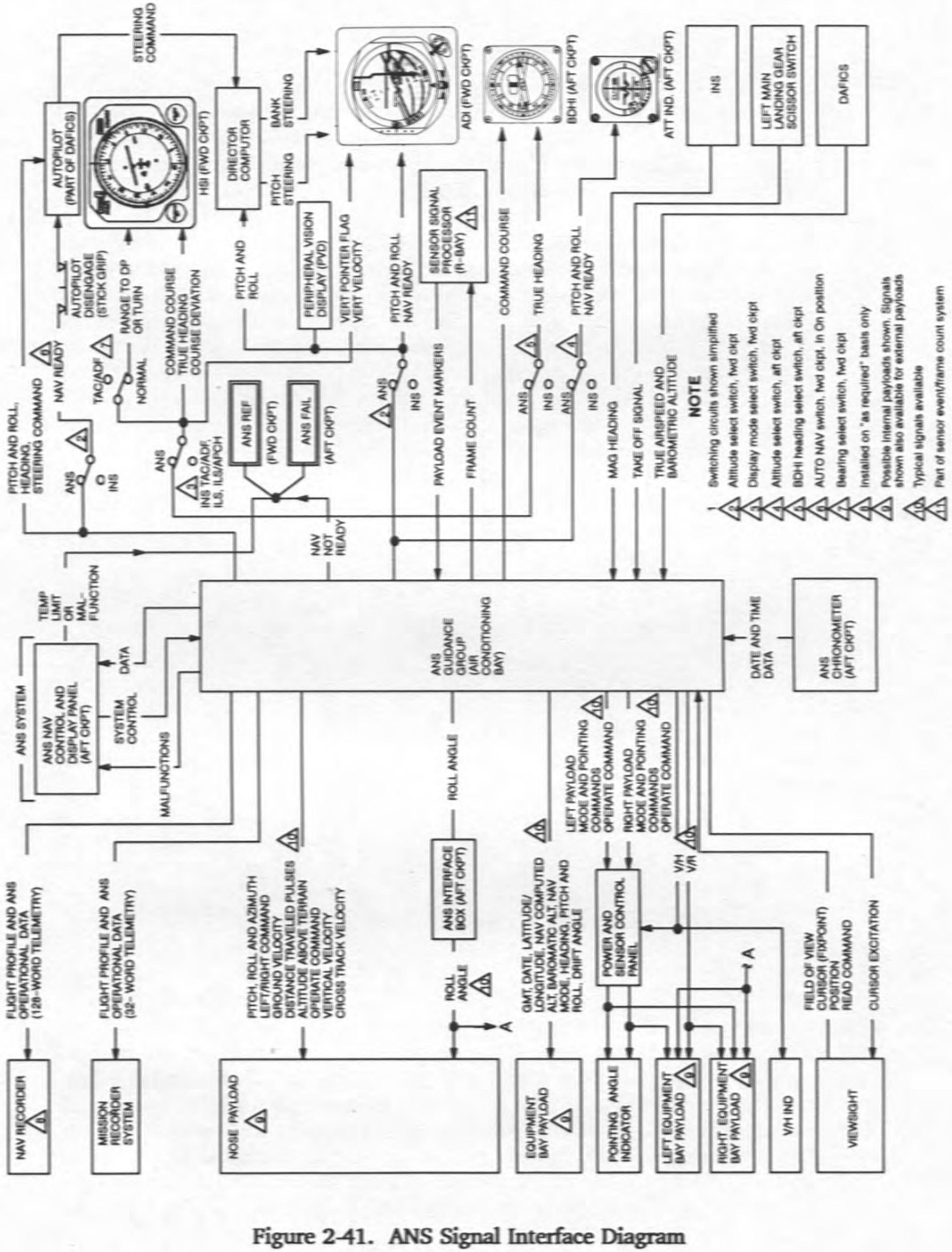


Figure 2-41. ANS Signal Interface Diagram

2-14 MISSION RECORDER SYSTEM (MRS).

The MRS is an airborne, integrated, mission and maintenance data recording system that continuously monitors and records the performance of aircraft and payload systems, and crew voice audio. The system has the capacity to record a total of 12 continuous hours of data. The MRS monitors the operation of systems and equipment under preflight, flight, and postflight conditions, permitting identification of failures, impending failures, and low performance.

Multiplex switches within the MRS collect, on a sequential time-share basis, analog signals representing the control, performance, and health of various aircraft systems. A multiplexing circuit in the DAFICS collects analog signals representing the functioning of that system, providing a stream of premultiplexed data to the MRS. The ANS supplies the MRS with data in digital and analog form. The MRS can accept premultiplexed as well as non-multiplexed inputs from payload equipment. Most multiplexed signals are sampled at a rate of once every 3 seconds; however, some signals that require greater time resolution are sampled at a higher rate. Before being multiplexed, analog signals are conditioned within a range from zero to two volts dc. The conditioned and multiplexed analog signals are digitized and then recorded on tape together with ANS digital navigation data and crew voice audio from the intercom system. Following flight, the airborne tape is copied then computer processed. Processed MRS data is used for performance analysis and fault isolation of monitored systems and for reconstruction of a flight. The tape in the recorder assembly is not removed during its service life. The recorder assembly is crash survivable.

The airborne system, consisting of three units: a multiplex switch and signal conditioner (unit 501/507), a recorder electronic assembly (unit 502), and a recorder assembly (unit 500), is shown in figure 2-42. The system is turned on and monitored from the aft cockpit. System operation is automatic.

Following a flight, recorded data on the unit 500 tape is copied using a mobile ground formatter unit (MGFU) for reproduction of non-voice data and a tape copy unit (TCU) for crew voice audio reproduction. As shown in figure 2-43, the TCU tape is played back in a voice playback unit which provides crew voice audio on a tape or cassette.

Also as shown in figure 2-43, the MGFU creates a computer compatible tape (CCT) of MRS recorded digital data. The CCT is processed in a specially programmed general purpose computer which produces printouts on the performance and condition of MRS-monitored systems and equipment. The computer also drives a plotter which produces analog graphs (traces) of MRS monitored signals.

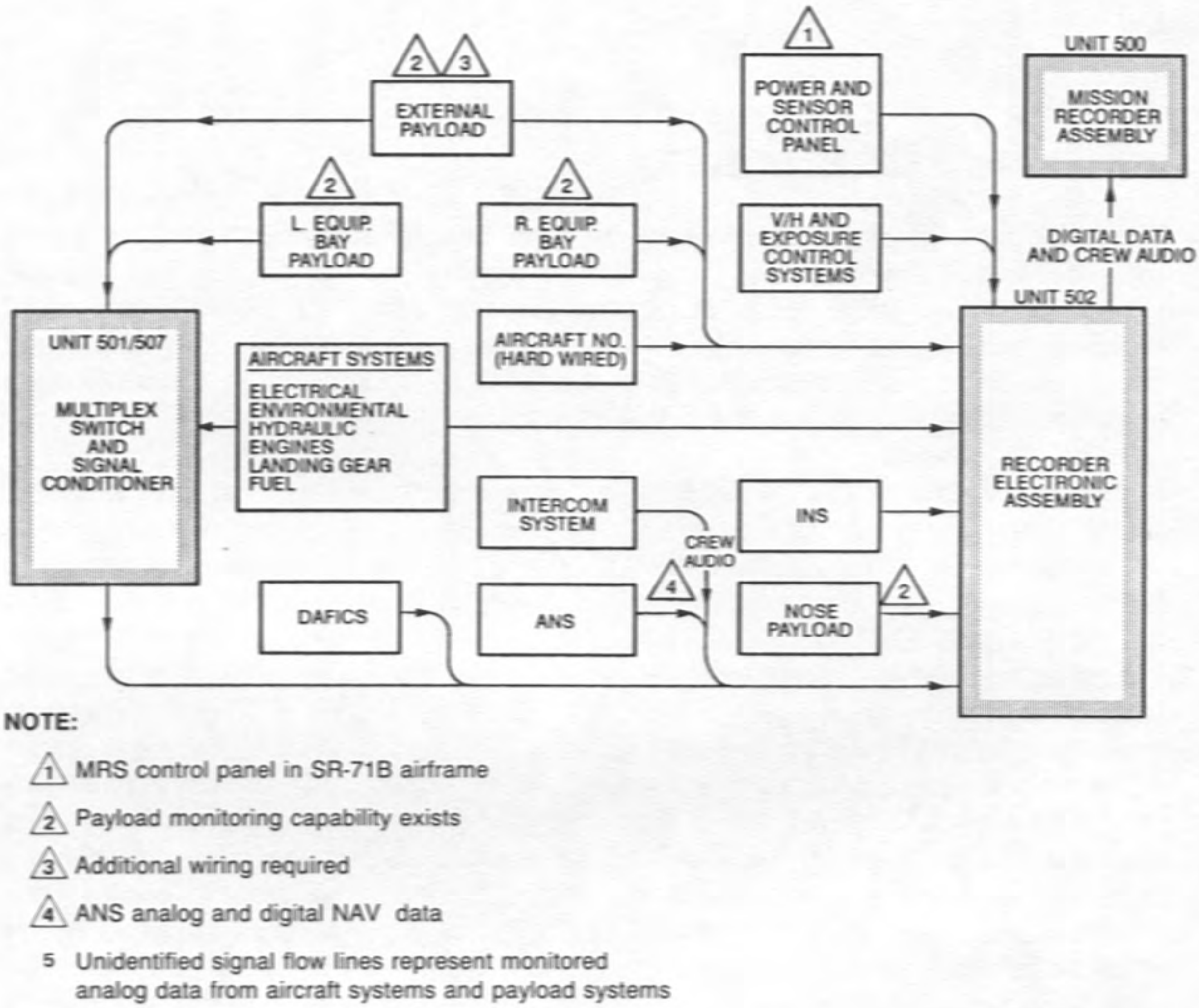


Figure 2-42. MRS Data Monitoring

The following is a summary of aircraft systems monitored by the MRS and the number of signals monitored within each system. Some of these signals are multiplexed (sampled) at a rate greater than once every three seconds. For payload purposes, there is additional system capacity for recording hundreds of more signals, as described in Section III.

- o Engine: 16
- o Hydraulic System: 13
- o Fuel System: 29
- o Environmental Control System: 16
- o DAFICS: 58
- o Electrical System: 24
- o MRS: 18
- o Miscellaneous: 20
- o ANS: Digital data representing approximately 100 parameters

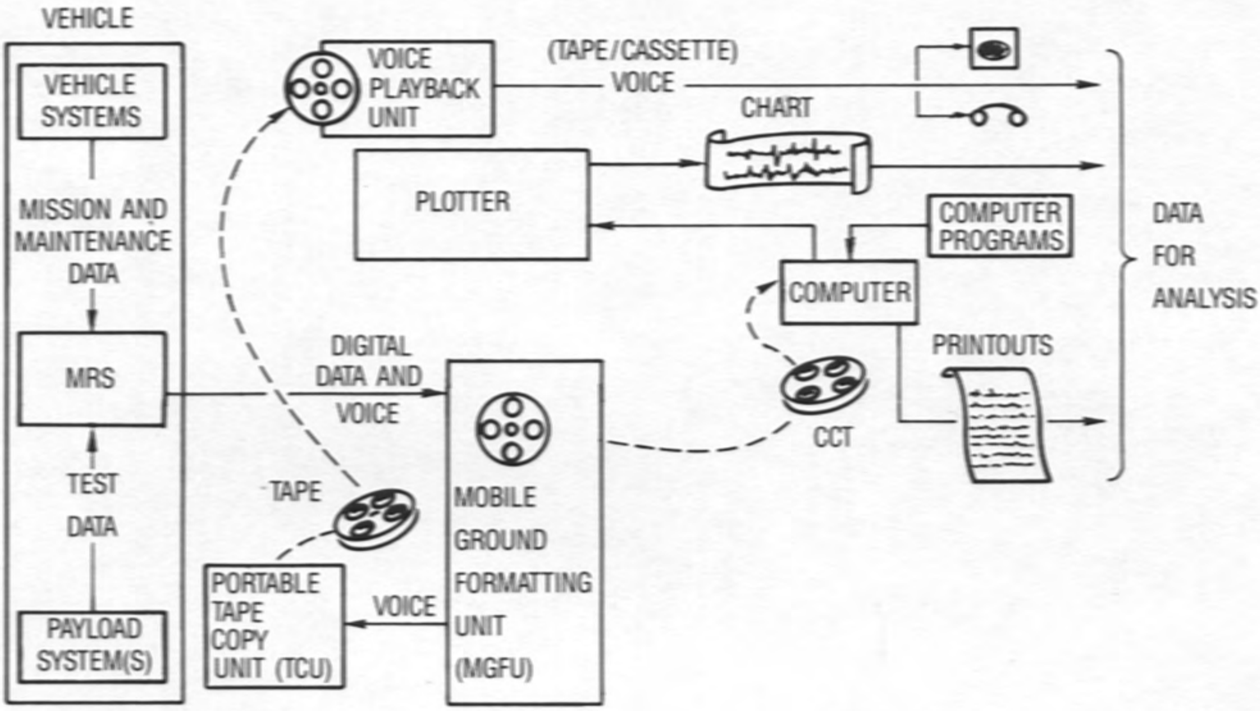


Figure 2-43. MRS Data Processing

SECTION III

PAYLOAD CAPABILITIES

3.1 GENERAL.

The SR-71 aircraft has the structural capabilities and the space, electrical power, cooling air, and in-flight recording provisions and capacities that permit carrying and supporting a variety of internally or externally-mounted research or test payload equipment. Hard points exist in some of the equipment bays for mounting internal payloads. There are no existent hard points for mounting external payloads. Internal strongback modifications, to support external payloads, have been designed and flown on A-12 and YF-12A aircraft. Similar modifications, using these or comparable designs, can be made to NASA SR-71 aircraft. These modifications will, in most cases, be permanent and available for future use. External payload provisions (pylons, etc.), utilizing the internal modifications, will be reversible.

Lockheed and NASA are prepared to assist the researcher with integration of his experiment or payload into or onto the aircraft, and to design and implement required modifications to the aircraft and aircraft systems to support the experiment or payload. Work will be carried out in accordance with negotiated contracts and within the framework of agreed upon budgets and schedules.

3.2 EXTERNAL PAYLOAD CAPABILITY.

External payloads, with weights in excess of 20,000 pounds, can be carried either above or below the aft section of the fuselage as shown in figure 3-1. In general, top-mounted payloads are carried on a pylon assembly, while bottom-mounted payloads are slung from racks. Section IV provides examples of previous payload and experiment installations. Excess hydraulic power, electrical power, and cooling air are available and can be provided by modifying the aircraft in accordance with the requirements of a particular payload.

3.3 INTERNAL PAYLOAD CAPABILITY.

Payload equipment can be installed within the aircraft in two general areas: (1), the payload equipment bays (2), the detachable nose section. Figure 3-2 shows the location of these areas and figure 3-3 indicates their approximate capacity. The specific location of installation depends on payload weight, space, electrical, cooling air, and operational requirements. Installation of a payload in these areas requires engineering evaluation of the payload and its requirements, and determination of any required changes to aircraft electrical wiring, cooling air, and equipment mounting provisions.

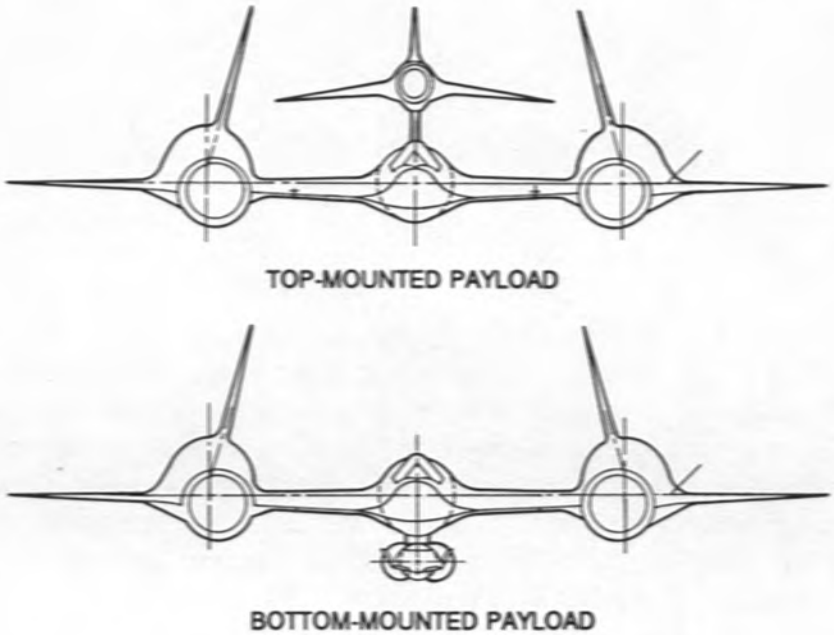
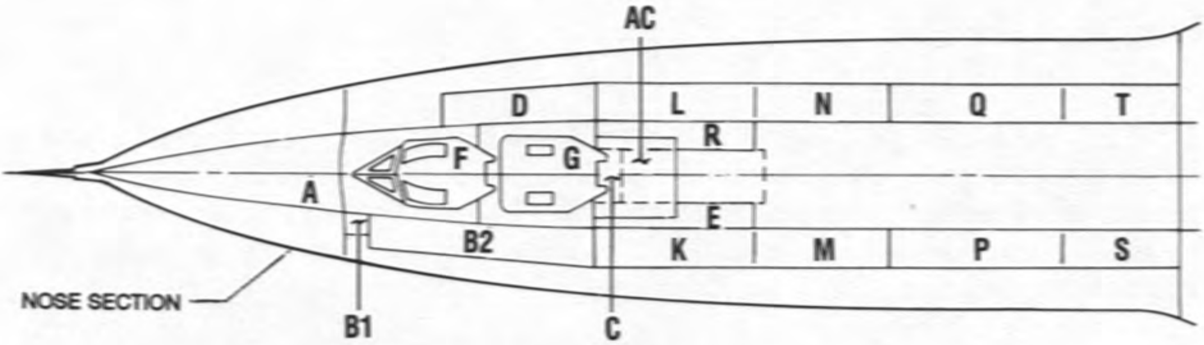


Figure 3-1. Examples of Exterior Payload Mountings

The E (electrical)-bay and the R (radio)-bay contain electrical and radio equipment and components required for operation of the aircraft. The E-bay does not have space provisions for payload equipment. The R-bay has space provisions for a UHF translator, a classified line-replaceable-unit (LRU). If this LRU is not installed, payload equipment of acceptable weight and size can be installed in its place, provided that electrical wiring, cooling air, and mounting provisions are modified as required.

3.3.1 Payload Equipment Bays. Payload equipment bays are located in the chine areas, along each side of the forebody of the aircraft, and in a compartment (C-bay) just forward of the nose wheel well. Except for the C-bay, payload equipment is mounted in the bays in one of two ways. The equipment can be secured (hard mounted) to specially designed fittings attached to the sides (webs) of the bays, with the bay door secured in place after the equipment has been uploaded. Payload equipment can also be mounted directly on a bay door, with the door and equipment uploaded together and the door then secured in place. Payload equipment can be hard mounted in all of the mission equipment bays. Payload equipment can be mounted on the bay doors of all the equipment bays except the M-bay. To mount equipment on the M-bay door, the door must be modified and new hinge parts fabricated. Equipment is mounted in the C-bay by rolling it into the compartment on rails secured to the outboard webs of the bay.



COMPARTMENT DESIGNATION	FUSELAGE COMPARTMENT	ALTERNATE COMPARTMENT NOMENCLATURE
AC	AC-BAY	AIR CONDITIONING BAY
B1	B1-BAY	PRESSURE TRANSDUCER ASSEMBLY (PTA) BAY
B2	B2-BAY	LOX AND 3RD LN_2 SYSTEM 1 COMPARTMENT
C	C-BAY	CENTER BAY
D	D-BAY	RIGHT CHINE BAY 2
E	E-BAY	ELECTRICAL EQUIPMENT BAY
K	K-BAY	FWD LEFT PAYLOAD EQUIPMENT BAY, FWD END
L	L-BAY	FWD RIGHT PAYLOAD EQUIPMENT BAY, FWD END
M	M-BAY	FWD LEFT PAYLOAD EQUIPMENT BAY, AFT END
N	N-BAY	FWD RIGHT PAYLOAD EQUIPMENT BAY, AFT END
P	P-BAY	AFT LEFT PAYLOAD EQUIPMENT BAY, FWD END
Q	Q-BAY	AFT RIGHT PAYLOAD EQUIPMENT BAY, FWD END
R	R-BAY	RADIO EQUIPMENT BAY
S	S-BAY	AFT LEFT PAYLOAD EQUIPMENT BAY, AFT END
T	T-BAY	AFT RIGHT PAYLOAD EQUIPMENT BAY, AFT END

NOTE

1 Third LN_2 system installed on SR-71A only.

2 On SR-71 aircraft only.

3. All identified areas, except the AC, B1, B2, and E-bays, and for the most part the R-bay, can be used for internal payload installations.

Figure 3-2. Internal Payload Locations

3.3.2 Nose Section Compartments.

3.3.2.1 OBC Nose. The OBC nose has a single equipment compartment, accessible through a hatch at the bottom of the nose. Payload equipment can be mounted on the hatch and uploaded into the nose, or hardmounted to specially designed internal fittings within the nose, with the hatch then secured in place.

3.3.2.2 Lightweight Nose. The lightweight nose has a single equipment compartment, accessible from the aft end of the nose assembly, when detached from the aircraft. Payload equipment can be mounted on and beneath a pallet which is supported by rails installed on each side of the interior of the nose. Payload installation is accomplished by rolling the pallet into the nose. A hatch opening in the top of the nose permits servicing and maintenance of an installed payload.

3.3.2.3 ASARS Nose. The ASARS nose has a single compartment on top and a larger compartment on the bottom for payload equipment. The nose is particularly adaptable to research and development of synthetic aperture radar or comparable transmitting/receiving equipment, as the radome is full monocoque polyimide/astroquartz, optimized for X-band transmissivity.

BAY/ COMPARTMENT	VOLUME (Cu Ft)	TYP PAYLOAD WEIGHT (Lbs)	MAXIMUM DIMENSIONS/ METHOD OF MOUNTING
Nose	23.0	550	30 x 30 x 75; Hatch, pallet, or hard mount
C	7.2	150	24 x 24 x 16; Hard mount
D	12.0	230	11 x 17 x 80; Door or hard mount
K & L	29.2	900	16 x 17 x 92; Door or hard mount
M & N	21.7	200	18 x 18 x 49; M-bay: Hard mount △1 N-bay: Door or hard mount
Q & P	32.0	340	18 x 18 x 90; Door or hard mount
T & S	22.7	400	18 x 18 x 62; Door or hard mount

NOTE

△1 Door mount possible with modification to M-bay door and the addition of new hinge parts.

Figure 3-3. Payload Bays/Compartments Capacities

3.4 COOLING AIR PROVISIONS.

As described in Section II, cooling air is provided to the equipment bay and detachable nose areas. (See figure 2-15.) Each of the equipment bays has regulated (orificed) air outlets which provide air for indirect cooling of equipment located in the bays. These outlets can also be modified to supply cooling air directly to equipment. Cooling air data for bay and nose areas, under cruise flight conditions (Mach 3.2; 80,000 feet), is provided in figure 3-4. Cooling air delivered to the bays varies from approximately -20 degrees F, at the beginning of cruise flight, to about +10 degrees, at the end of cruise. Additionally, on SR-71A aircraft, cooling air in this temperature range is also supplied to the nose. The nose section and the C-bay (on all SR-71 aircraft) also receive cockpit exhaust air in the temperature range of + 50 to + 80 degrees F. By modification of the aircraft, cooling air can also be provided to externally-mounted equipment. Air, at approximately 10 pounds per minute and at the same temperature range as provided in the equipment bays, can be made available for exterior payload cooling.

EQUIPMENT BAY	BAY PRESSURE ALTITUDE	BAY AIR TEMPERATURE (1)		AVAILABLE AIR FLOW
Nose	60K	<120 F	(2)	12 lb/min
			(3)	2 lb/min
C	50K	< 80 F	(2)	18 lb/min
D	53K	<130 F		5 lb/min
K & L	53K	<130 F		4 lb/min
M & N	53K	<130 F		1.5 lb/min
Q & P	57K	<140 F		6 lb/min
T & S	57K	<140 F		7 lb/min

NOTE

(1) Highest temperature with heat-producing equipment installed in the bay.

(2) Cockpit exhaust air.

(3) Cooling air from D-bay (SR-71 only).

Figure 3-4. Payload Equipment Bays Cooling Air Data

3.5 ELECTRICAL POWER PROVISIONS.

Primary electrical power for the aircraft is provided by two 60 KVA generators, derated to 55 KVA for cruise flight due to the high temperature environment in which they are located. Therefore, during cruise flight, a total of 110 KVA, 115/200 volts, three-phase, 400 Hz ac power is available for use. Aircraft systems do not normally use more than 19 KVA of power during cruise conditions; however, during peak loading lasting up to 30 seconds, aircraft systems may use up to 30 KVA. Therefore, under these conditions, total payload electrical loads are limited to a maximum of 80 KVA. The aircraft dc system is powered by two transformer-rectifiers, rated at 200 amps each. Normal aircraft systems loads do not exceed 100 amps, leaving a surplus of 300 amps available for payload use. Since dc power is derived from generator power, additional dc loading above 100 amps reduces available ac power to slightly less than 80 KVA. The instrument inverter and the dual battery are not considered to be sources of power for payload equipment as they are emergency power sources, for flight purposes only.

3.6 FLIGHT INSTRUMENTATION AND RECORDING CAPABILITY.

3.6.1 Flight Instrumentation. NASA has available, for use by the researcher, a versatile flight test instrumentation package, adaptable for installation in most of the payload equipment bays. This equipment features numerous combinations of pulse code modulation (PCM) and frequency modulation (FM) conditioning, with outputs available for the recorders identified in paragraph 3.6.3. Specialized instrumentation will be provided by the researcher.

3.6.2 Mission Recorder System (MRS). In addition to monitoring several hundred parameters related to the aircraft systems, the MRS has additional (excess) capacity to monitor hundreds of additional parameters from payload equipment. Of the 945 possible samples collected at 3-second intervals, approximately 230 are assigned to aircraft systems, leaving approximately 700 for payload use. If the rate of signal sampling is greater than once every 3 seconds, as with some aircraft systems parameters, the total number of payload signals that can be monitored will be reduced accordingly. Sampling rates of up to 15 per second are possible without modification to the system. With minor modification, the sampling rate can be increased to as high as 135 samples per second, with commensurate reduction in the total number of signals that can be monitored.

3.6.3 Additional Recording Capability. NASA has fourteen-track wide-band tape recorders available for use by the researcher, using the instrumentation package described in paragraph 3.6.1 or specialized instrumentation provided by the researcher.

3.7 POSSIBLE PAYLOADS AND EXPERIMENTS.

The SR-71, modified with a top-mounted pylon, can be used for testing supersonic/transonic inlets, mixed compression engines, airfoil/fuselage designs, and various missiles (Pegasus-type missiles, Phoenix-type missiles, target missiles, etc.) Additionally, a Lockheed/Raytheon study showed that AIM-7 Sparrow-type missiles could be launched from an SR-71/YF-12A type aircraft.[1] In addition to these payloads, the aircraft is an excellent platform for the purposes described below. Refer to Section IV for information regarding previous payloads and experiments.

3.7.1 Air Sampling. The aircraft can be used as a platform for sampling air particulates and/or radiation over a considerable area, at altitudes of 85,000 feet. An air particulate sampler, with a 100 square inch capture area, has been designed to function at Mach 3.2 without adversely affecting the sampling process.

3.7.2 Drone or Model Testing. The SR-71 can be used as a platform for testing various types of drones. The aircraft is also suited for testing models of different supersonic vehicles, in advance of building protype or production vehicles. Such testing will reveal aerodynamic performance under actual subsonic, transonic, and supersonic conditions.

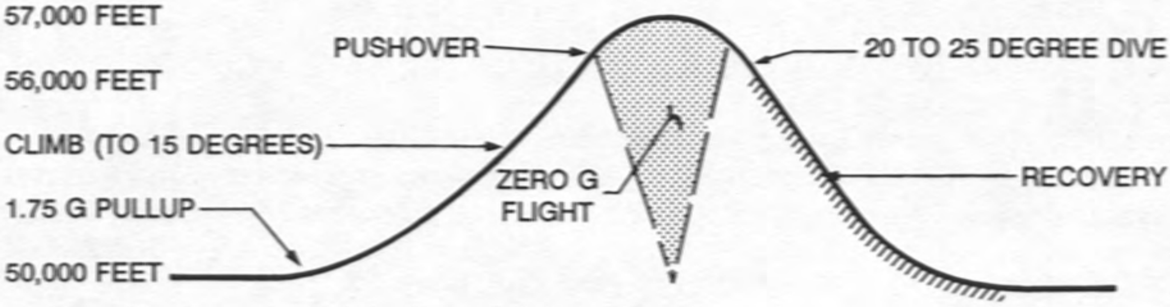
3.7.3 Material Testing. Test materials can be mounted on the aircraft and evaluated under conditions representative of their intended use.

3.7.4 New Technology Evaluation. New technologies associated with National Aero-Space Plane (NASP), the HL-20, or other supersonic/hypersonic airborne/space vehicles can be mounted on the aircraft and evaluated under actual conditions of flight.

3.7.4.1 External Burn Experiment. This is an experiment related to drag reduction for the NASP. The frontispiece is an artist's conception of this experiment mounted on the SR-71A.

3.7.4.2 Testing of New Engine, Inlet, Fuel, and Other Concepts.

3.7.4.3 Technology/Experiments Requiring Short Period Zero G Conditions. A computer study, not tested in flight, indicates that a pitch over maneuver will produce a zero g condition of approximately 30 seconds in duration. Figure 3-5 shows the flight profile and the principal parameters of the maneuver. The maneuver starts from level flight at an altitude of about 50,000 feet, at 450 KEAS, Mach 2.0. Pull up is made to a climb of about 15 degrees and, as KEAS bleeds through approximately



FLIGHT PROFILE

TIME (IN SECONDS) FROM START OF PULLUP	5	10	20	30 PUSHOVER STARTS	40	50	55	60
"G"	1.75	1.50	1.2	0	0	0	0	0.2
PITCH	4	12	13	15	-3	-12	-18	-22
PITCH RATE (DEG/SEC)	2.0	0.3	0.2	+1.0 to -4.0	-1.0	-1.0	-0.9	-0.8
STICK FORCE (LBS)	21	21	21	-23	-24	-30	-32	-33
MACH	2.00	1.97	1.90	1.93	1.86	1.90	1.95	2.00
KEAS	448	438	410	373	370	400	436	480

NOTE

1. Tabulated data is computer generated.
2. Indicated flight profile is approximate and the vertical scale is exaggerated. The exact maneuver and its specifics will depend on pilot technique.
3. Maneuver must be flight tested.

Figure 3-5. Zero G Capability

370, after an altitude gain of 6,000 to 7,000 feet, push over is made to initiate zero g flight. Recovery occurs near an altitude of about 50,000 feet. The maneuver is made in smooth air, within the flight limitations defined in Section I, figure 1-13.

3.7.5 Vehicle Launch. The aircraft is particularly suited for use as a high-speed launch platform for a variety of missiles and other vehicles.

3.7.6 New Sensors Evaluation. New optical, electro-optical, I.R., radar, ELINT, and other sensor technologies can be carried in the aircraft for evaluation under conditions of high altitude and high Mach flight.

SECTION IV

PREVIOUS PAYLOADS, EXPERIMENTS, AND STUDIES

4.1 GENERAL.

Various payloads and experiments have been carried on the SR-71 and its predecessors, the A-12 and YF-12A aircraft. Also, various studies have been made using the SR-71 or its predecessors as the intended test platforms. The following paragraphs summarize these programs. The mission-related payloads and equipment for the SR-71 and A-12 aircraft are not covered.

4.2 PREVIOUS PAYLOADS

4.2.1 D-21 Drone. The D-21 was designed as a high-altitude reconnaissance drone to be launched from a modified A-12 aircraft, designated M (mother) -12. The D (daughter) -21 has a delta wing, an internally-mounted ramjet engine, and uses 5900 pounds of JP-7 fuel, the same fuel as used by the SR-71. Construction is of titanium and the inlet consists of a fixed spike and air bypass. The drone is over 42 feet long, has a wing span of over 19 feet, and a total (fueled) weight of 11,200 pounds.

The D-21 was pylon-mounted on the aft fuselage of the M-12. (See figure 4-1). It was launched at altitudes of about 80,000 feet, at speeds of about Mach 3.2. This speed/altitude environment was required to minimize lift loads and drag on the vehicle and to "start" the drone engine. The D-21 cruises above Mach 3.2 and has a considerable range. Operational mission applications are still classified. Approximately 17 D-21s are stored for possible reuse at Davis-Monthan AFB in Arizona.

4.2.2 GAR-9 (AIM-47A) Missile. The GAR-9 missile was essentially a large scale AIM-4 (Phoenix-type) missile, with a weight of 815 pounds, a length of over 12-1/2 feet, a diameter of 13-1/2 inches, and a span of 33 inches. The missiles, mounted on launchers, one to a bay, were carried in three internal bays in the YF-12A aircraft. Figure 4-2 shows a missile on a dolly, beneath the aircraft, prior to upload. During launch, the missiles were ejected downward by pyrotechnic charges. There were seven supersonic launches of the missile at aircraft speeds from Mach 2.19 to 3.2, at altitudes from 65,000 to 76,000 feet.

Figure 4-1. M-12 Aircraft and D-21 Drone

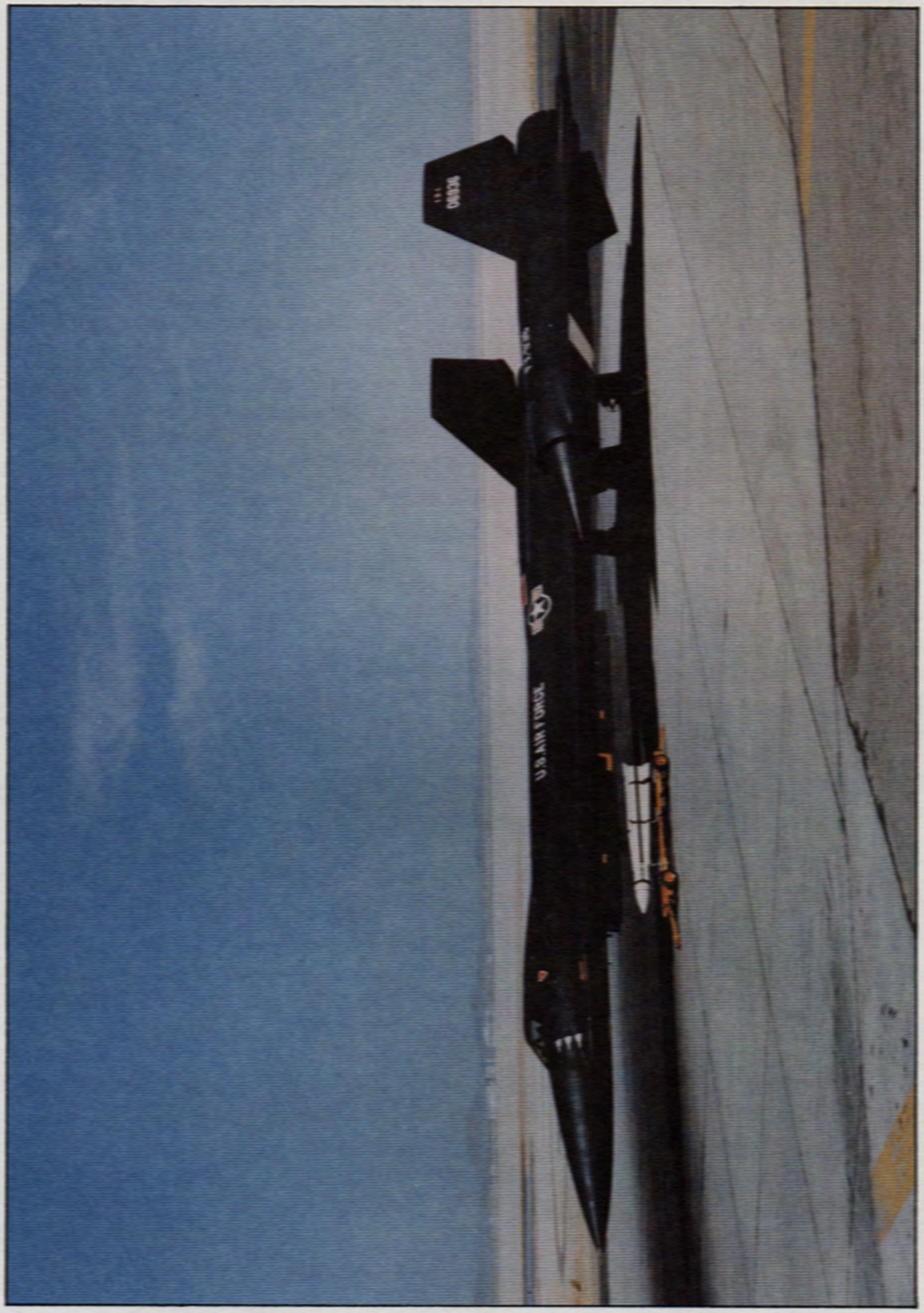


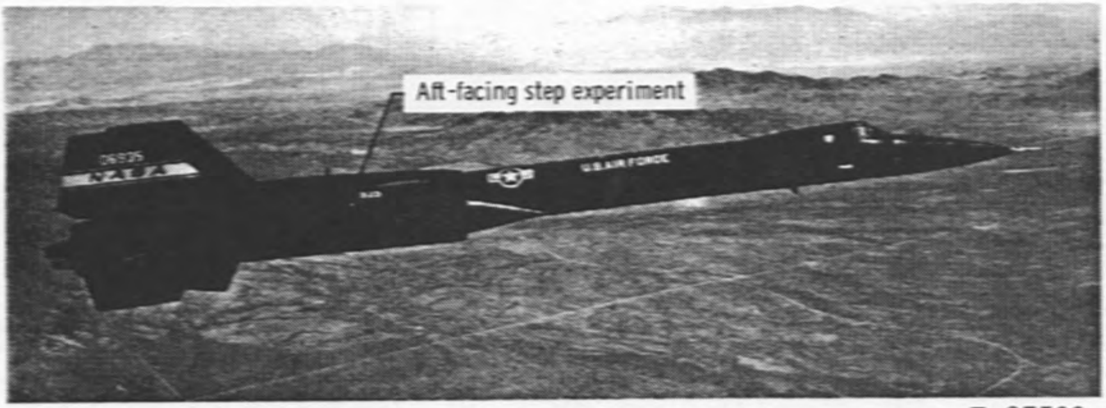
Figure 4-2. YF-12A Aircraft and GAR-9 Missile

4.3 PREVIOUS EXPERIMENTS.

4.3.1 Aft-Facing Step in High-Reynolds Number/High-Mach Environment. Surface discontinuities formed by the trailing edges of wings, panels, and other aft-facing discontinuous surfaces are known to be sources of drag. This experiment was initiated to provide more information about aft-facing discontinuities in thick boundary layer flow.[2] The study involved different Reynolds number conditions than previous studies, providing more insight into the effect of Reynolds number on drag levels associated with aft-facing discontinuities in a thick boundary layer. The YF-12 aircraft was selected as the facility for the experiment because of its ability to maintain flight condition at speeds from subsonic to Mach 3.

The experiment involved installing a stepped section on the upper aft fuselage of a YF-12A aircraft to measure and evaluate pressure characteristics of the step. As shown in figure 4-3, the test section, an area 0.9 meters wide by 3.2 meters long, consisted of a ramp region, a reference region, and a recovery region, together with a boundary layer rake complex. Base pressure data were obtained for step heights of 0.33, 0.63, and 1.19 centimeters, at Mach numbers from 0.6 to approximately 3.0, with a range in altitude for many of the Mach numbers. Surface static pressures were measured ahead of and behind the step. The boundary layer rakes were used to determine velocity profiles and local surface and boundary layer edge flow conditions. Figure 4-4 shows velocity profiles at the forward and aft rakes under no step conditions and, at the aft rake, for various step heights.

4.3.2 Composite Material Testing. Composite-honeycomb panels are used along the edges and upper and lower surfaces of the SR-71 fuselage and nacelle chines and the wing leading and trailing edges. The inlet spikes are also made from composite-honeycomb. (Figure 4-5 shows the composite-honeycomb areas of the aircraft.) During cruise flight, the panels reach temperatures ranging up to and above 600 degrees F. (See figure 4-6 for panel temperatures during cruise flight.) The original panels were made from silicone-asbestos and, in some cases, phenyl silane. An asbestos-free replacement was required for the silicone-asbestos material. Under an engineering study proposal, following extensive studies and tests, a replacement material was developed.[3] This material, designated F174, consisted of a polyimide resin and an E-glass fiber. Wing leading edge panels (subjected to temperatures of 640 degrees F) and trailing edge panels (subjected to temperatures of 520 to 530 degrees F) were manufactured using the F174 material and were flight qualified. Successful flight testing of the new panels resulted in a program to replace the original panels with panels fabricated of the F174 material. The program was not implemented. The spikes were not included in the study and the replacement program.



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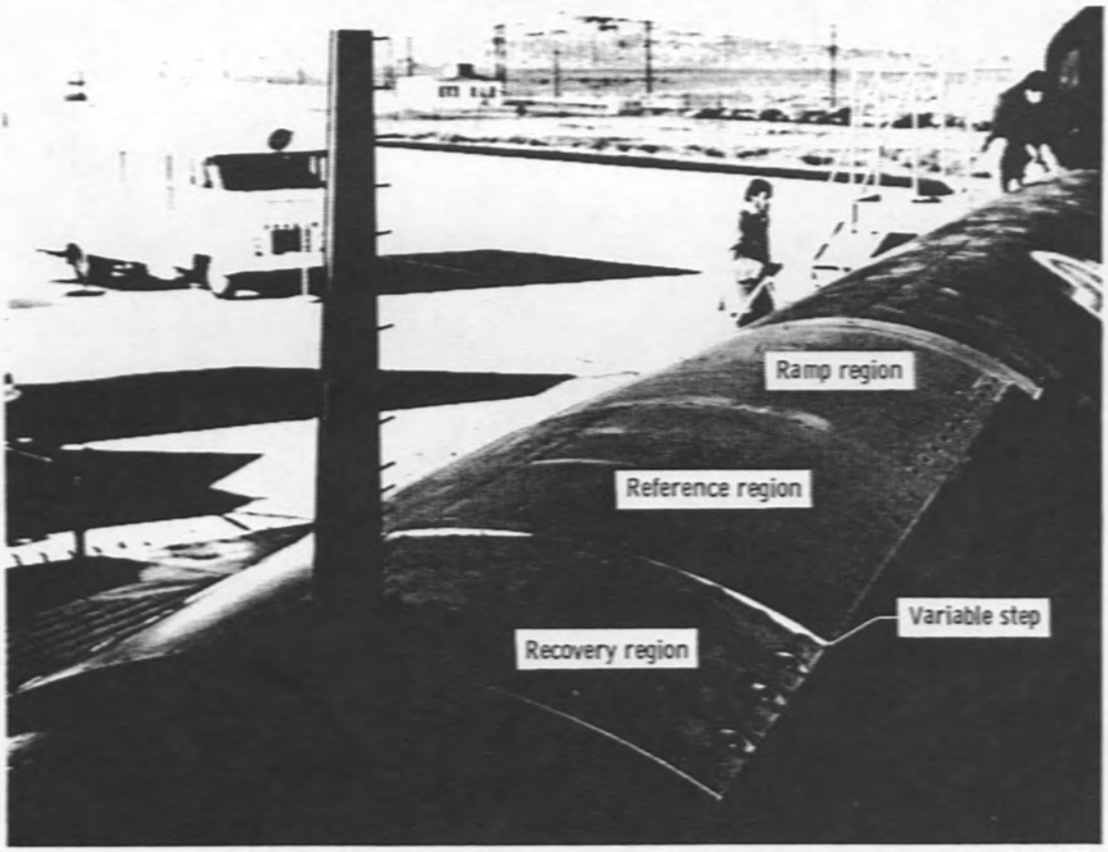
YF-12 airplane in flight, showing location of aft-facing step experiment.



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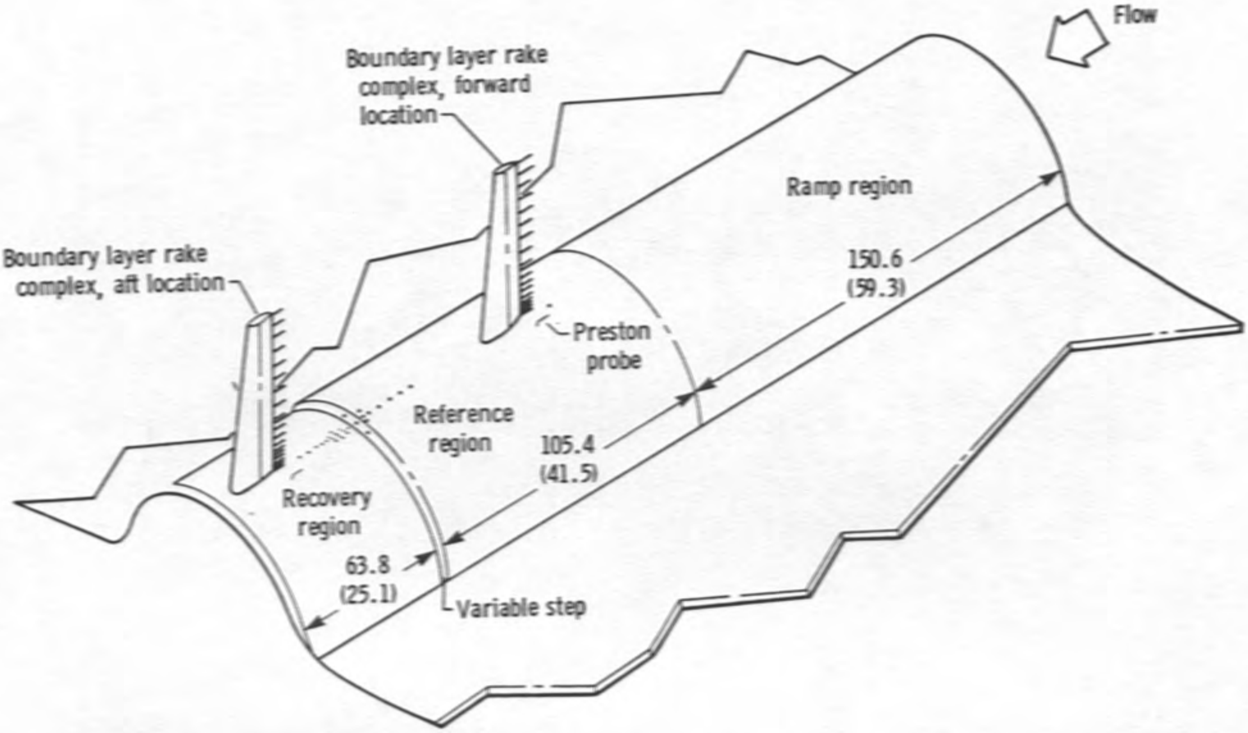
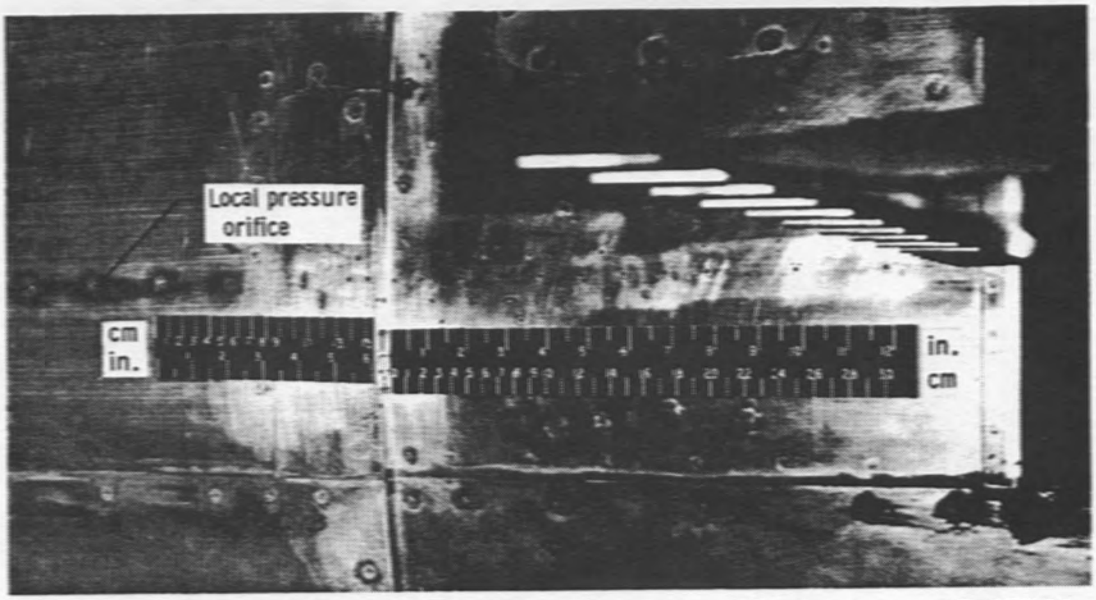
(a) Test section viewed from rear. Boundary layer rake complex is in aft location; step height is 0.63 centimeters (0.25 inch).

Figure 4-3. Aft-Facing Step Experiment, Location and Design
(Sheet 1 of 3)



(b) Details of component locations. Dimensions are in centimeters (inches).



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(c) View of step region showing location of pressure orifices. Step height is 1.19 centimeters (0.47 inch).

Figure 4-3. Aft-Facing Step Experiment, Location and Design (Sheet 2 of 3)

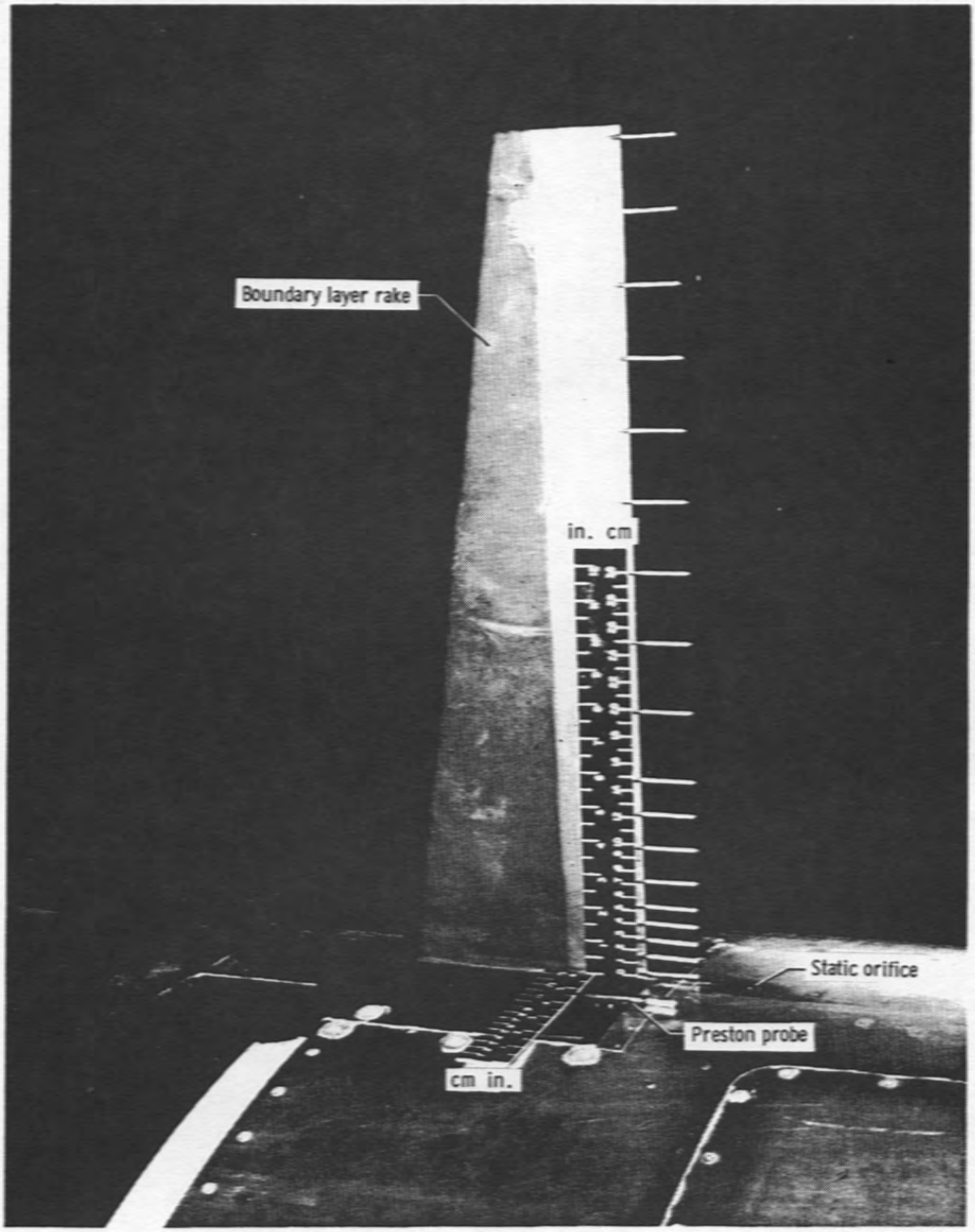

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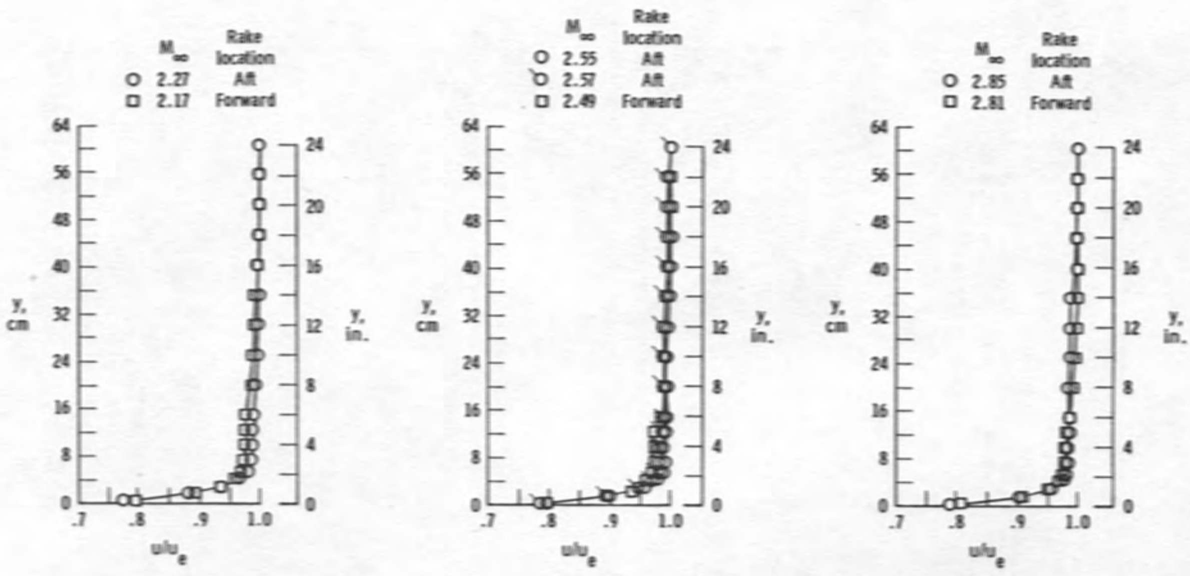
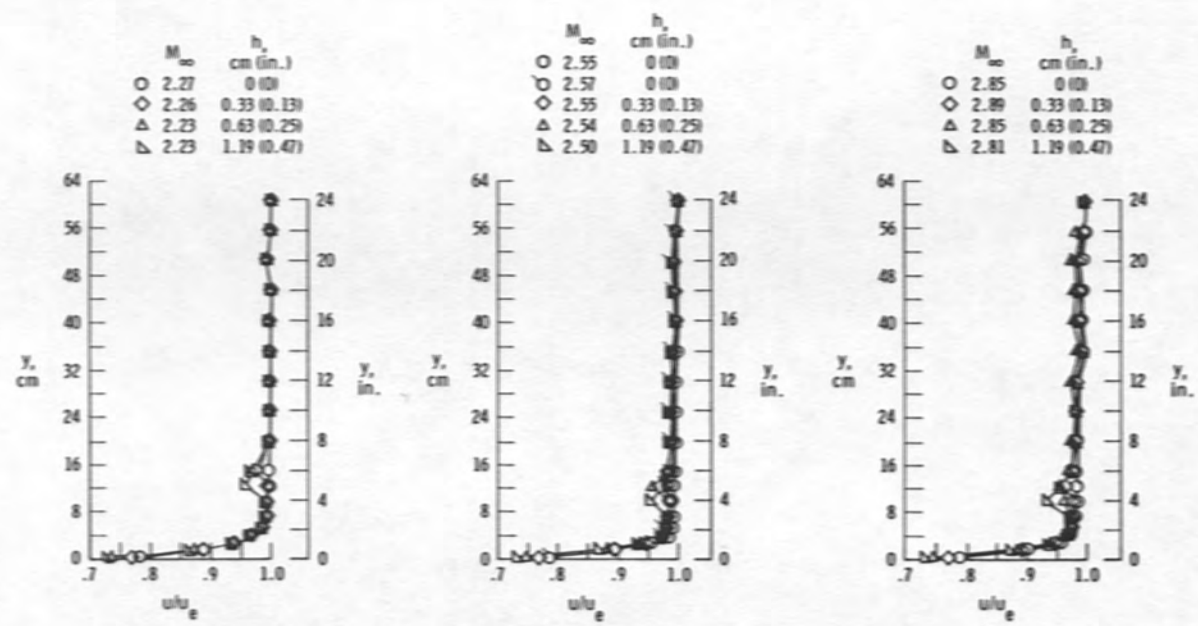
Figure 4-3. Aft-Facing Step Experiment, Location and Design
(Sheet 3 of 3)



(a) $M_\infty \approx 2.20$. *(b) $M_\infty \approx 2.50$.* *(c) $M_\infty \approx 2.80$.*

Velocity profiles for forward and aft (no step) rake locations.
u_e obtained from probe most distant from aircraft surface.



(a) $M_\infty \approx 2.20$. *(b) $M_\infty \approx 2.50$.* *(c) $M_\infty \approx 2.80$.*

Velocity profiles for aft rake location with various step heights.
u_e obtained from probe most distant from aircraft surface.

Figure 4-4. Velocity Profiles at Forward and Aft Rakes

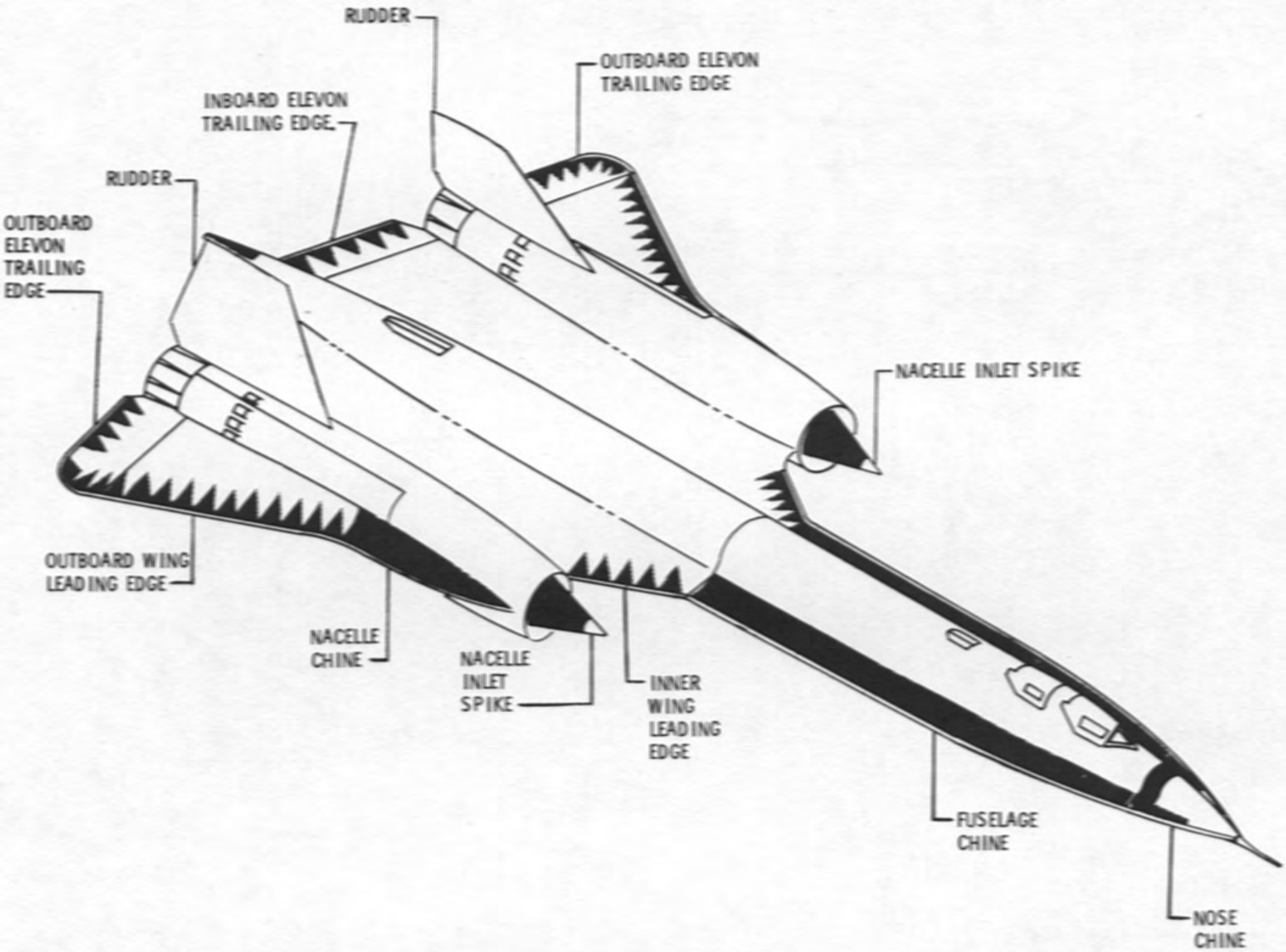


Figure 4-5. Composite-Honeycomb Areas of the SR-71

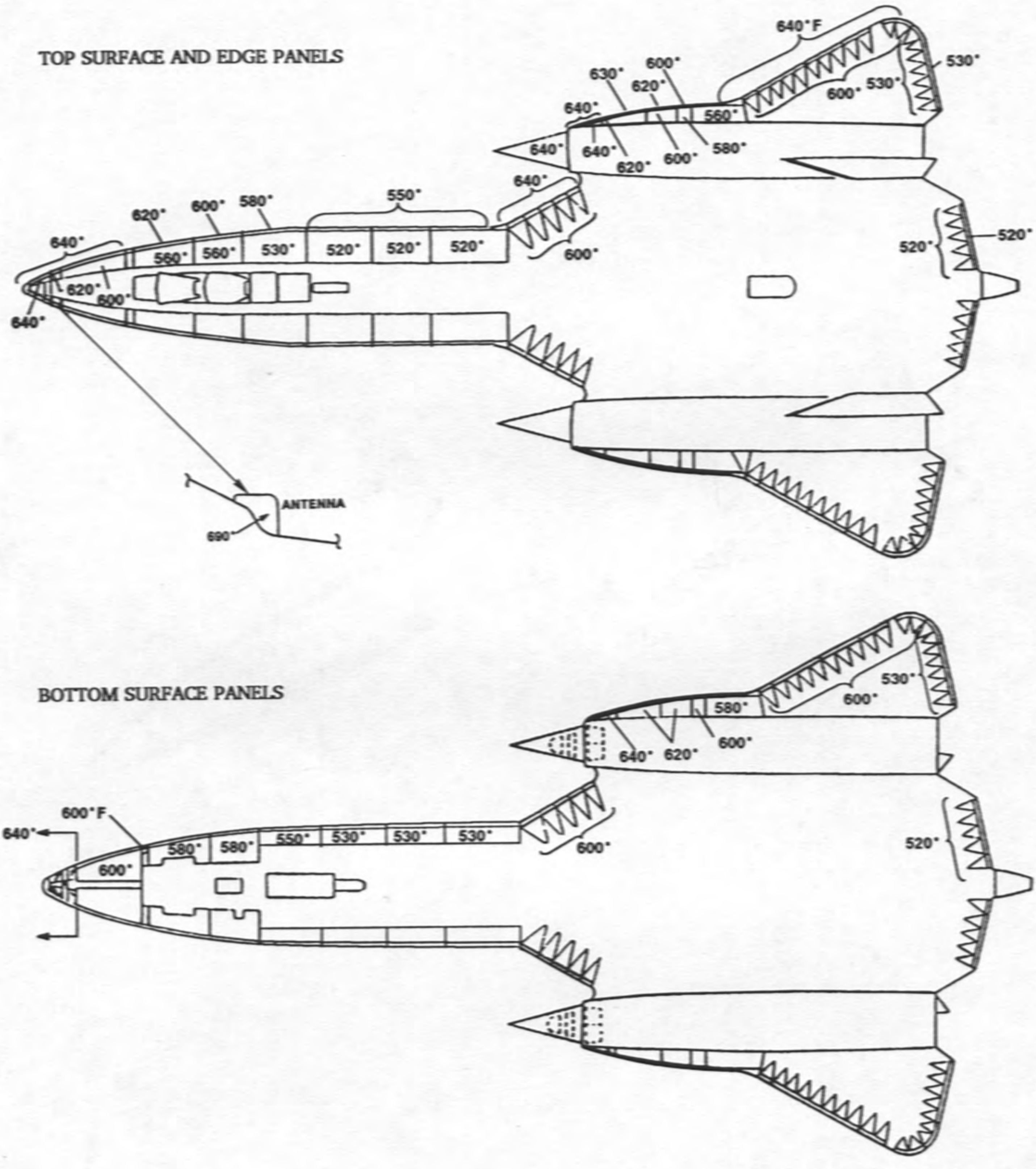


Figure 4-6. SR-71 Composite-Honeycomb Panel Temperatures

4.3.3 In-Flight Boundary-Layer Measurements (Cold Wall Experiment). This experiment was performed to resolve the validity of various theories used for prediction of skin friction and heat transfer in supersonic turbulent boundary-layer environments; parameters important in the design of high speed aircraft.[4] This resolution was important, not only because the predicted values derived from the various theories differed substantially, but also because the results from wind tunnel and flight test experiments, performed to verify the theories, also did not agree.

The experiment included an instrumented hollow cylinder, approximately 9 feet long by 18 inches in diameter. Instrumentation included thermocouples, static pressure orifices, a skin friction gage, a pitot pressure rake, and a total temperature boundary-layer rake. The cylinder, secured to a pylon, was suspended about 18 inches below a YF-12A aircraft. (Figure 4-7 shows a YF-12A, with the cold wall experiment suspended beneath it, followed by a YF-12C chase plane). The cylinder was designed to be small enough to permit testing in a wind tunnel so that flight and wind tunnel data could be directly compared.

Skin friction, heat transfer, and boundary-layer profile data were acquired at the lower external surface of the cylinder. Two different cylinder configurations were flown. One configuration, uncooled and uninsulated, was not used for collection of heat transfer data as the skin (wall) temperature remained at or near the radiation equilibrium temperature. The other configuration was insulated and cooled to 211 degrees K, before flight, using liquid nitrogen. When, in flight, the aircraft reached desired test conditions, the insulation was explosively removed just before heat transfer test data was obtained.

The theory of van Driest predicted skin friction and heat transfer coefficients that were in excellent agreement with the results of this experiment. Also, the measured velocity profiles, transformed by the theory of van Driest and Eckert's reference enthalpy method, were in good agreement with Cole's incompressible law-of-the-wall velocity profile.

4.4 PREVIOUS STUDIES.

4.4.1 Air Particulate Sampling. A study was conducted to investigate the installation of air particulate samplers on SR-71 aircraft.[5] The purpose of the sampler was to collect radioactive debris at high altitude (80,000 feet), at high speed (Mach 3.2). The air particulate sampler consisted of a long rectangular assembly with an inlet designed for no air spillage at speeds over Mach 2.4, an E-type filter with a capture area of 600 square inches, and a radiation detector sensitive to gamma radiation. (See figure 4-8.) Two sampler capacities were studied, one with an air flow rate of 708 standard cubic feet per minute (SCFM), and one with a flow rate of 1,000 SCFM. The larger capacity sampler was selected because of its greater capture capacity, with still-acceptable air velocities at the filter.

Figure 4-7. YF-12A Aircraft with Cold Wall Experiment

FLOW RATE SCFM	W_1 INLET WIDTH	W_2 EXIT WIDTH	FILTER APPROACH VELOCITY – MAXIMUM FPS
708	5.43	1.00	177
1000	7.68	1.71	258

(SIZED FOR DOUBLE THICKNESS PBI FILTER MEDIA)

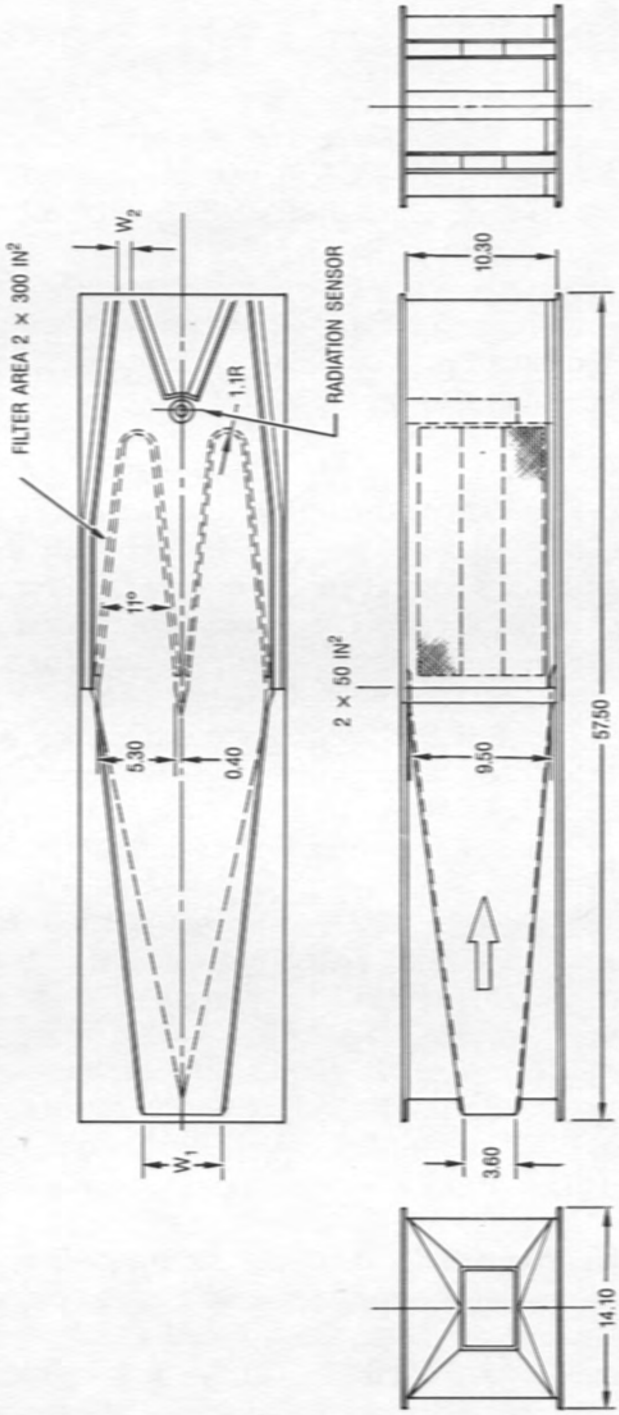


Figure 4-8. Air Particulate Sampler

The sampler was to be maintained within the equipment bay until sampling was required; at which time, upon command from the aft cockpit, the sampler would be mechanically lowered into the free airstream. The study assumed that four samplers would be installed for flight. There were six possible locations: the K, L, P, Q, S, and T-bays (refer to Section III, figure 3-2), with the last four bays being preferable. The samplers were to remain retracted within the equipment bays until the area of their intended use was reached, at which point one sampler would be lowered into the airstream until activity was detected, as indicated in the aft cockpit. At this time, one or more of the other samplers could be lowered for further collection. The capability of extending the samplers only when needed would minimize the collection of unrelated particles and drag on the aircraft.

4.4.2 Exploration of Mach Growth. The aircraft was designed for extended cruise flight at Mach 3.2. Various studies have been conducted regarding extending the speed of the SR-71 beyond 3.2.[6, 7, 8] The results of these studies conclude that extension to Mach 3.5, for short periods of time (of 10 minutes or more), is feasible, with approach to that speed to be accomplished incrementally. Extension above Mach 3.5 will be more involved and difficult, requiring substantially greater effort. (Refer to Section V, paragraph 5.2.4, Factors Limiting Speed Above Mach 3.2 and Sustained Flight Above 85,000 Feet.) Mach extension to 3.4 can be achieved without any significant flight test program or change to the aircraft. Extension from 3.4 to 3.5 may require the addition of instrumentation; insulation of inlet hydraulic lines and, possibly, inlet actuators; and a flight extension program, which is planned.

4.4.3 Hypersonic Drone. NASA requested a study to investigate mounting a hypersonic drone on a YF-12C aircraft for launch at cruise altitude and Mach.[9] (See figure 4-9.) For the study, a NASA HT-4 drone was to be scaled for a fuselage length of 50 feet, with a total (fueled) weight of 14,800 pounds. The drone was enlarged to accommodate a Pratt and Whitney RL-10 liquid rocket engine and a 115-inch tail cone fairing was added to decrease drag during mated flight. A canoe, with a 5 by 7-inch cross-section, was added along the bottom centerline of the drone to accommodate launch attachments and landing provisions.

Various mounting provisions, locations, and methods of launch were investigated. In its final configuration, the mounting consisted of a single, thin pylon, with a length approximately that of the drone, located on the top centerline of the aft fuselage of the aircraft. This location required blocking the wing sections of the aircraft aft fuel tanks (to maintain c.g. during mated flight) and modification of the aircraft fuel system to forward transfer fuel when the drone was elevated for launch (to compensate for aft movement of combined drone/aircraft c.g.) Although this location required modification of the aircraft fuel system it provided clearance for crew ejection and minimized drone engine plume impingement on the aircraft.

After considering other methods (flyaway and impulse), aerodynamic lift was selected as the most appropriate means of drone separation from the aircraft. The drone was stowed at a low angle of attack (low drag) position until just before launch. At launch, a launch beam was raised from a horizontal to a vertical position, elevating the drone 8 degrees in pitch and moving it somewhat aft on rollers. (See figure 4-10.) The elevated position of the drone gave it a positive lift of at least twice its weight, imposing on it a vertical acceleration away from the aircraft of at least 1 G. (See figure 4-11 for drone trajectory at launch.) The acceleration of the drone could be modified by application of aircraft elevon control. The study concluded that installation and launch of the specified drone was feasible.

4.4.4 Supersonic Inlet. High performance inlet systems are sensitive to many factors that are difficult to simulate during wind tunnel testing of subscale models. In-flight testing permits evaluation of inlet performance under actual Mach, temperature, and pressure conditions. A feasibility study was conducted at the NASA Ames Research Laboratory to determine if a test inlet could be carried, for supersonic testing, using a YF-12A aircraft as a flight test platform.[10] (See model shown in figure 4-12.)

Wind tunnel tests were performed on two nacelle models representing capture areas of 5 and 10 square feet. The models, mounted on a 1/12 scale model of the YF-12A, were tested at Mach 1.4 and below in the transonic tunnel and at Mach 2.5 and above in the supersonic tunnel. The model nacelles were tested at two different pylon heights and at two different fuselage station locations on the top centerline of the YF-12A aft fuselage. Testing included simulated supersonic unstarts of the test inlets. A rotating four-armed, pressure-sensing rake, mounted on the inlet spike, measured local Mach number and flow conditions. (Figure 4-13 shows typical flow and Mach conditions at the forward end of the test inlet.) After consideration of various factors, including loads, drag, and flow interference effects, the lower forward pylon location was considered preferable. (Figure 4-14 shows the test inlet mounted at this location.)

The results of the study indicate that inlets of up to 10 square feet capture area could be carried on the aircraft and that the test inlet would be located in a flow field which has a degree of distortion similar to that which might be expected on many aircraft installations.

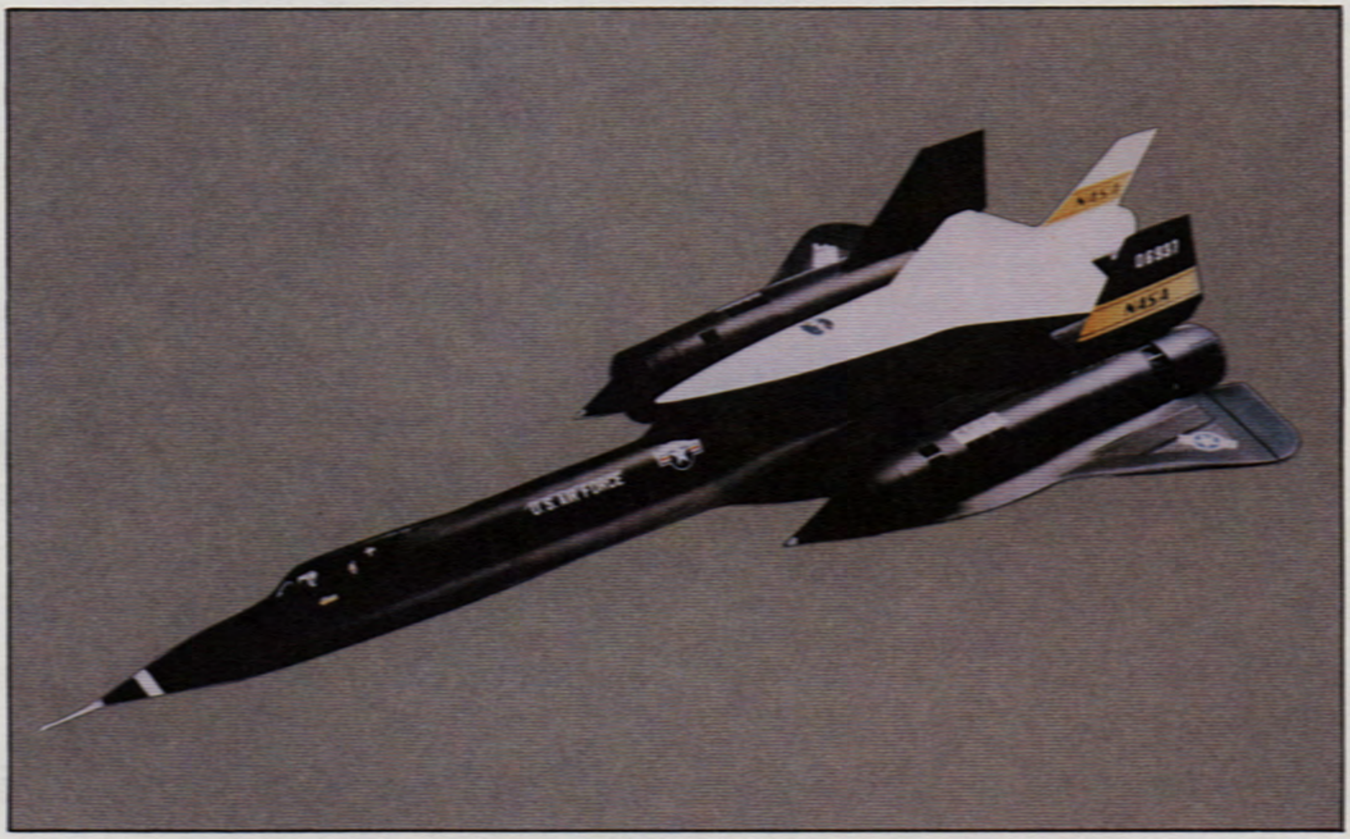


Figure 4-9. Models of YF-12C Aircraft and Hypersonic Drone

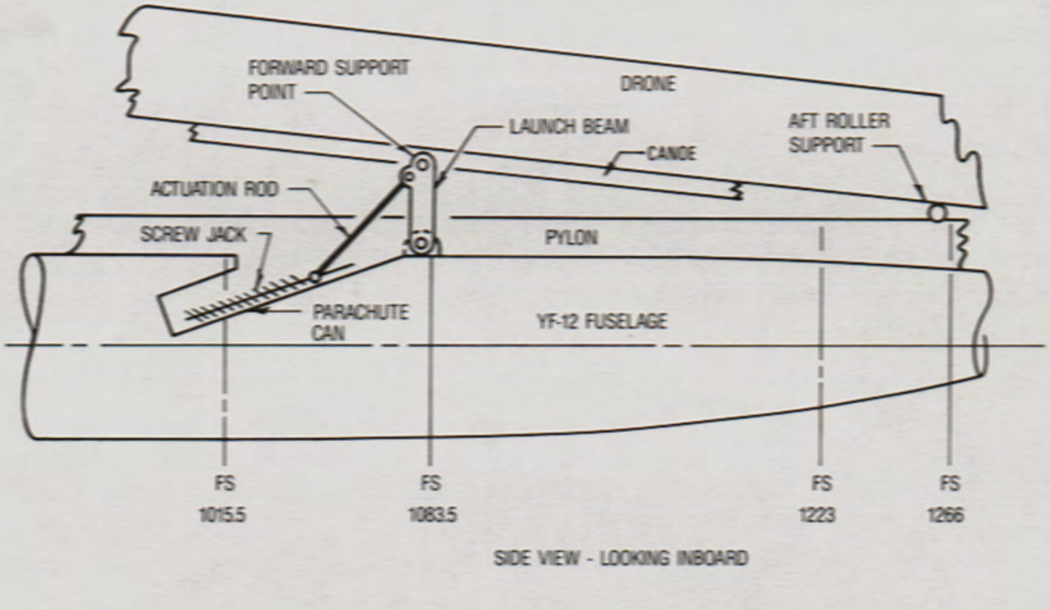


Figure 4-10. Hypersonic Drone Launch System

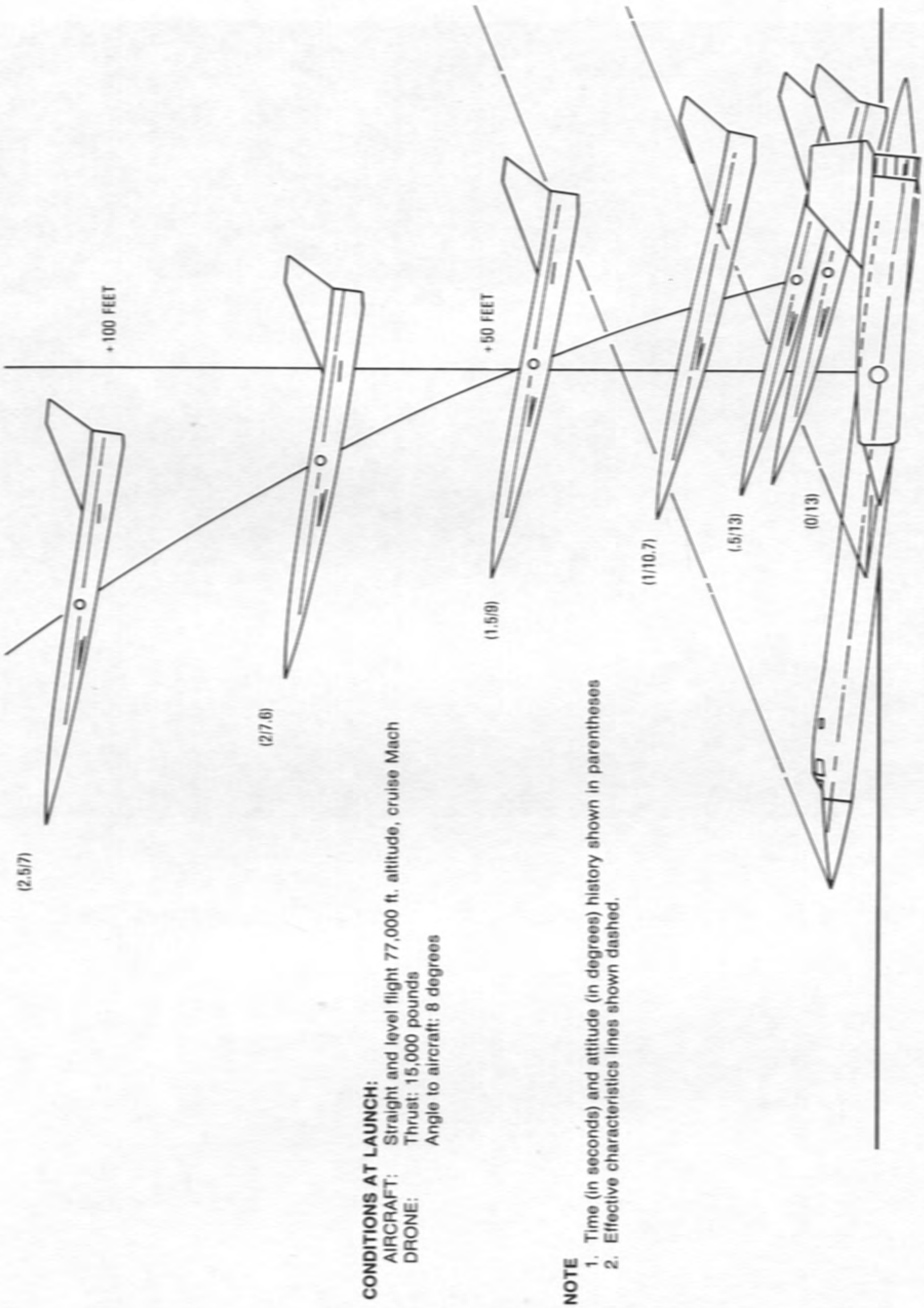


Figure 4-11. Drone Trajectory at Launch

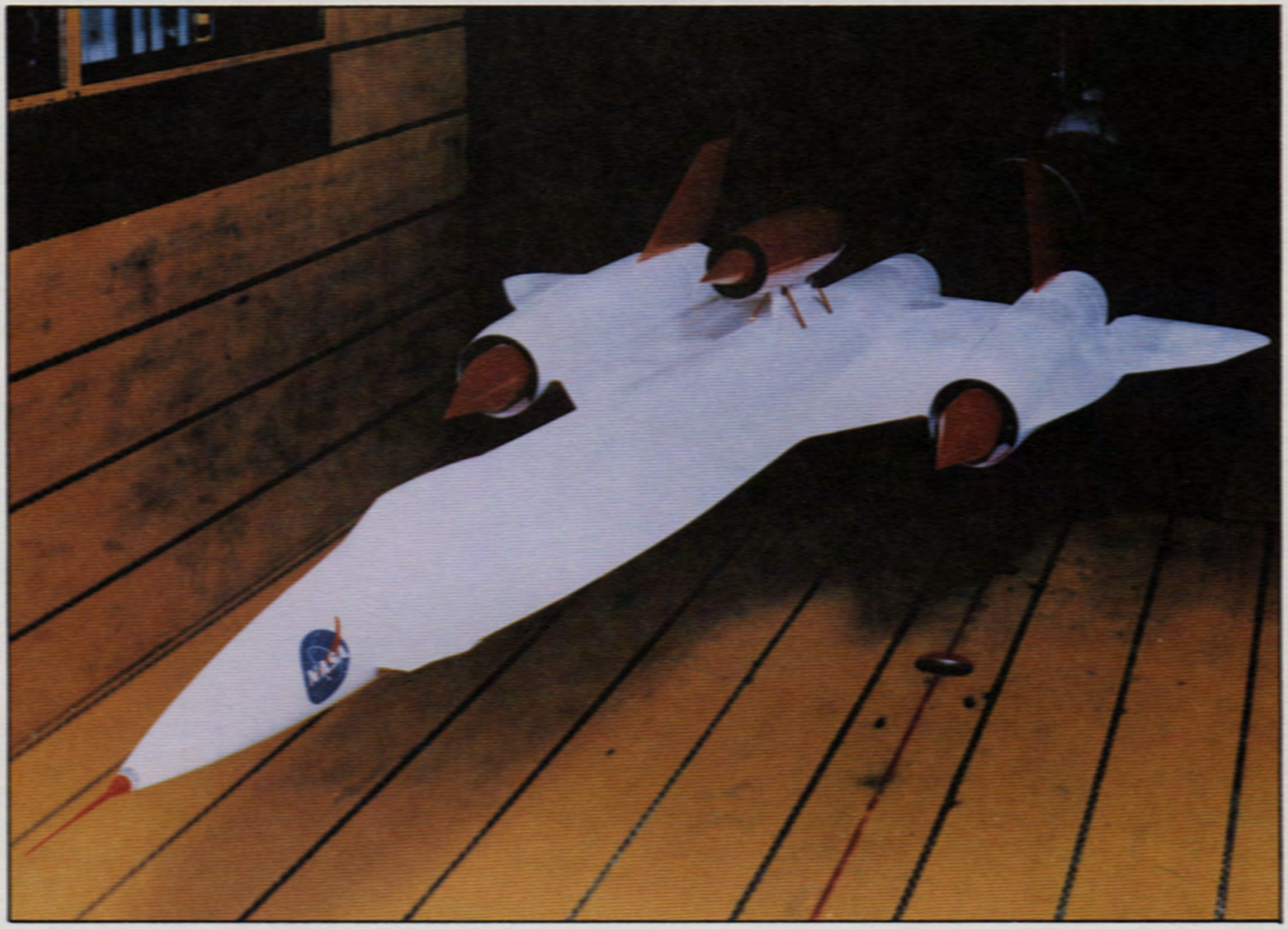


Figure 4-12. Wind Tunnel Models of YF-12A Aircraft and Supersonic Inlet

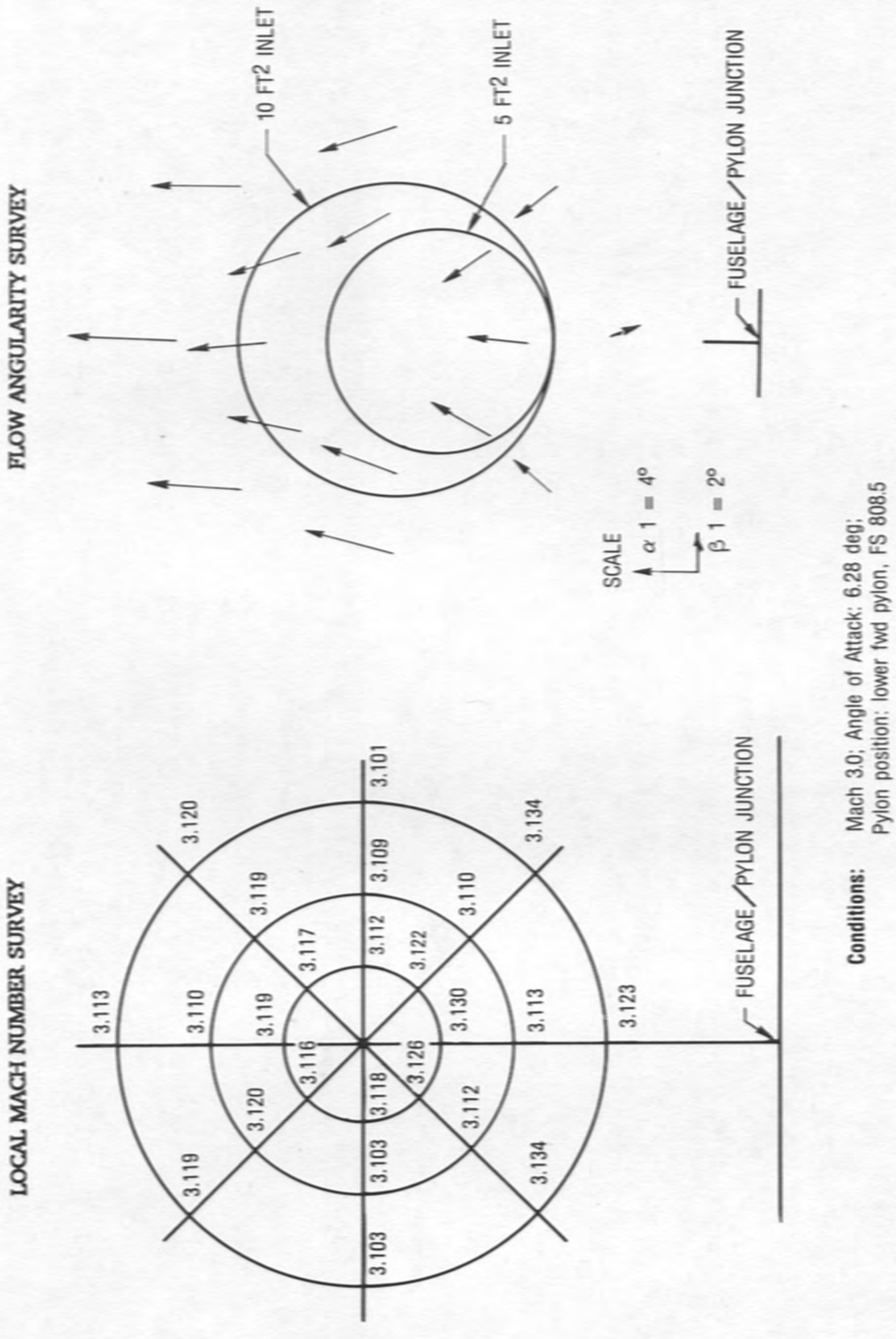


Figure 4-13. Typical Flow and Mach Conditions for Supersonic Inlet

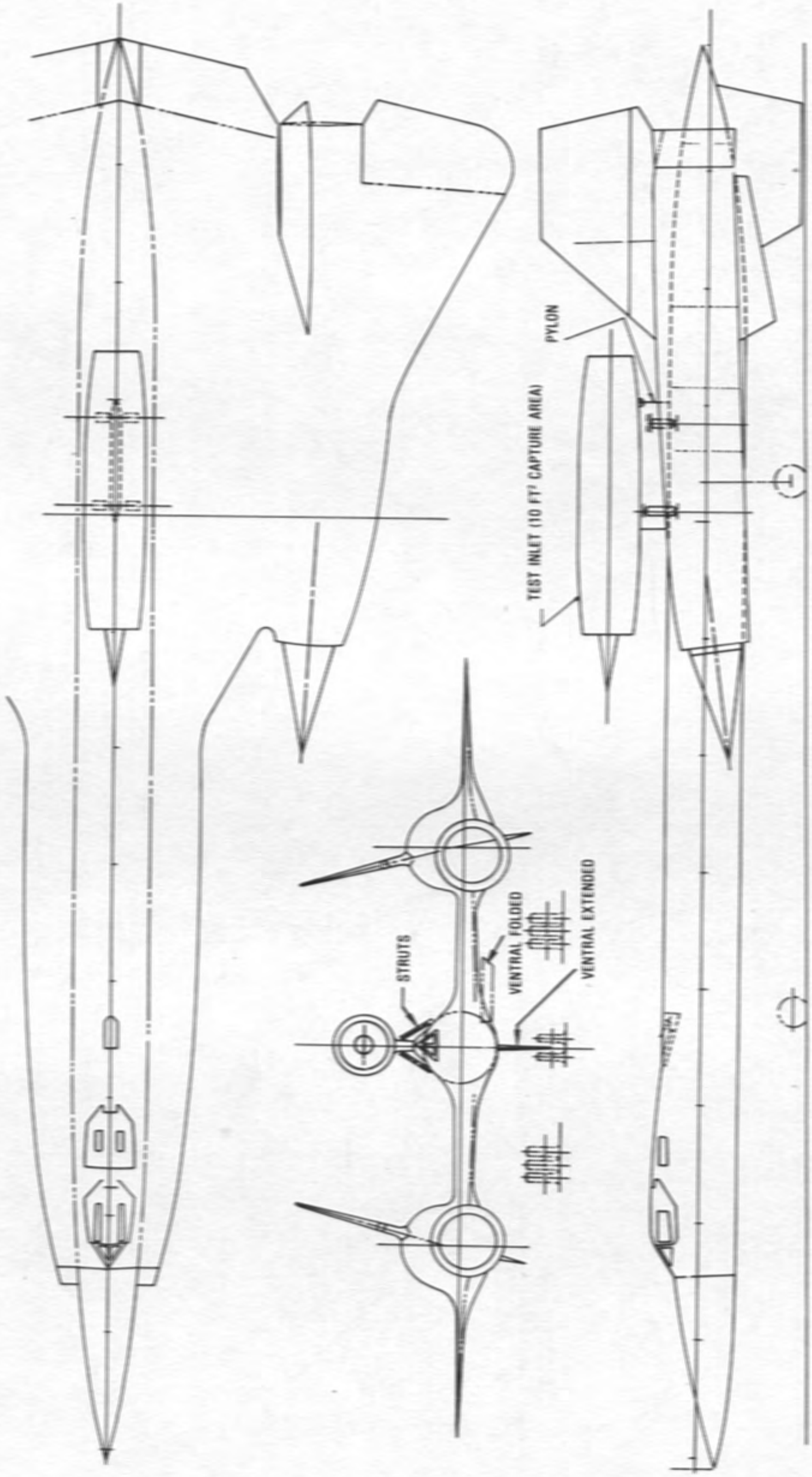


Figure 4-14. Final YF-12A/Supersonic Inlet Configuration

SECTION V

EXTENDED FLIGHT CAPABILITIES

5.1 GENERAL.

The SR-71 was intended for maximum range flight at altitudes of or approaching 85,000 feet, with speeds at or approaching Mach 3.2. Figure 1-10, in Section I, describes the previous speed and altitude envelope of the aircraft. During its military career, the SR-71 rarely and only slightly exceeded its design speed and altitude limits. Studies have been made to determine if these limits could be extended. Study results show that a Mach number of 3.5 is readily attainable, with sufficient thrust at that speed to carry a large external payload.

5.2 EXTENSIONS OF THE SPEED/ALTITUDE ENVELOPE.

5.2.1 Envelope Expansion Flight Program. An envelope expansion flight program is planned to qualify the aircraft to the extended limits. Figure 5-1 shows the planned speed and altitude extensions to the old flight envelope.

5.2.2 Speeds Above Mach 3.2. As described in Section IV, paragraph 4.4.2, Exploration of Mach Growth, various studies have concluded that it is feasible to extend the aircraft's speed out to Mach 3.5, at least for short periods of time. This extension in speed is shown in figure 5-1 by the shaded area to the right of the current design speed limit of Mach 3.2.

5.2.3 Altitudes Above 85,000 Feet. One of the studies conducted regarding Mach 3.5 flight also addressed achieving higher altitude flight.[6] The study indicated that the aircraft could be zoomed to an altitude of about 95,000 feet. This would be accomplished with an aircraft gross weight of about 85,000 pounds. The aircraft would be accelerated from Mach 3.2 to Mach 3.5, at an altitude of 80,000 feet, then zoomed to 95,000 feet, with Mach decaying back to approximately 3.2. The aircraft would subsequently settle back to an altitude of about 84,000 feet. High altitude zoom is shown in figure 5-1.

5.2.4 Factors Limiting Speed above Mach 3.5 and Sustained Altitudes Above 85,000 Feet. The only structural limitation related to speed above 3.5 is a KEAS limit of 420, set by inlet duct pressures and temperatures which exceed acceptable values. Other factors which limit speed above Mach 3.5 are inlet capture area and excessive engine compressor inlet temperature (CIT). Factors which limit sustained flight at

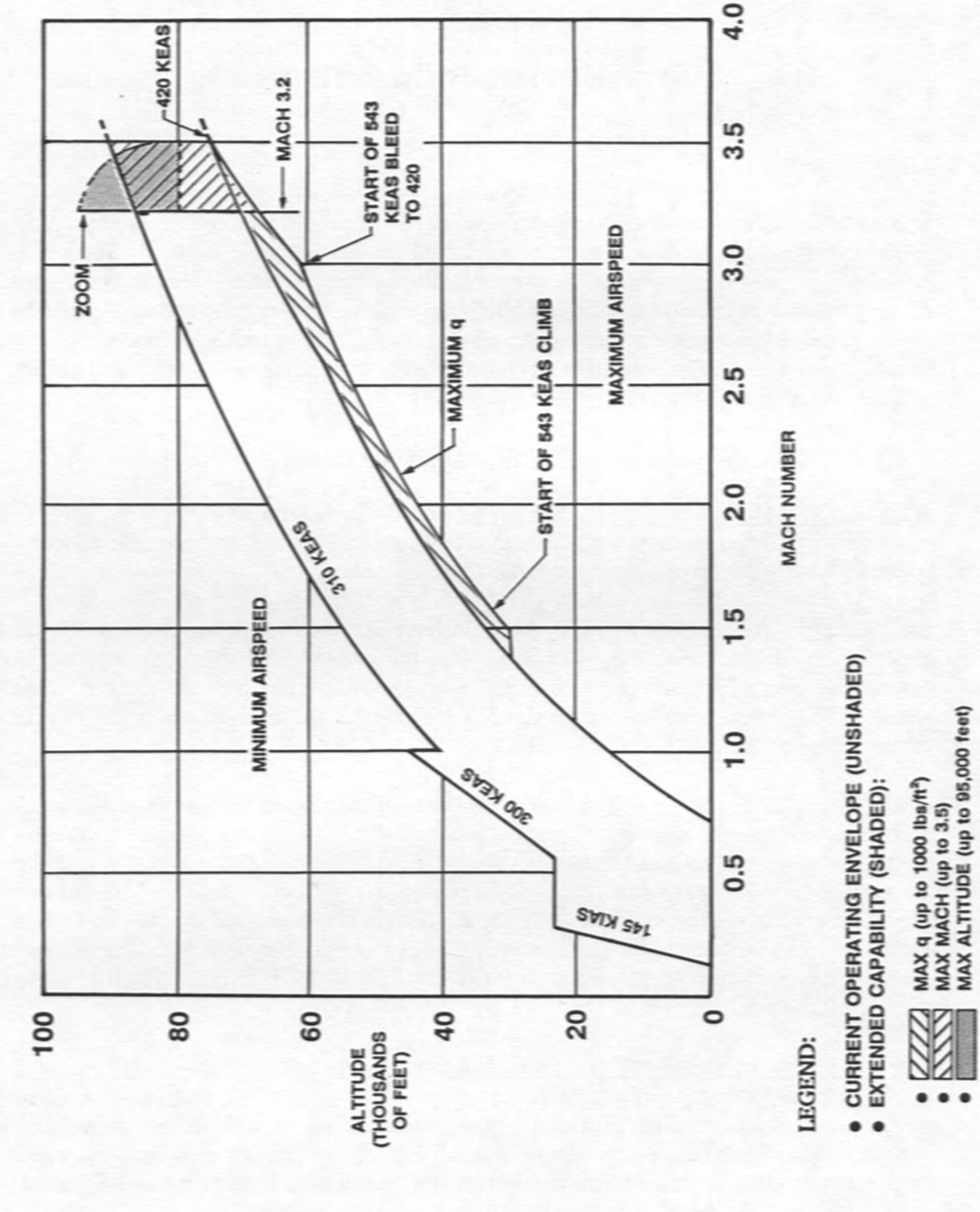


Figure 5-1. Extended Flight Envelope

altitudes above 85,000 feet are wing area and/or propulsion system thrust. Replacement of existing outboard wing panels with larger ones will provide increased wing surface area. Increased thrust will require installation of a new powerplant and inlet redesign.

5.3 INCREASED DYNAMIC PRESSURE "*q*" FLIGHT.

Flight at 543 KEAS will achieve a *q* of 1,000 pounds per square foot. High *q* straight and level flight appears possible, for undefined periods of time. Also, as shown in figure 5-1, a climb of 543 KEAS is possible, with the high *q* climb beginning at 30,000 feet and continuing up to 55,000 feet. At 55,000 feet, KEAS will begin to bleed off to 420 and *q* will diminish as the climb proceeds.

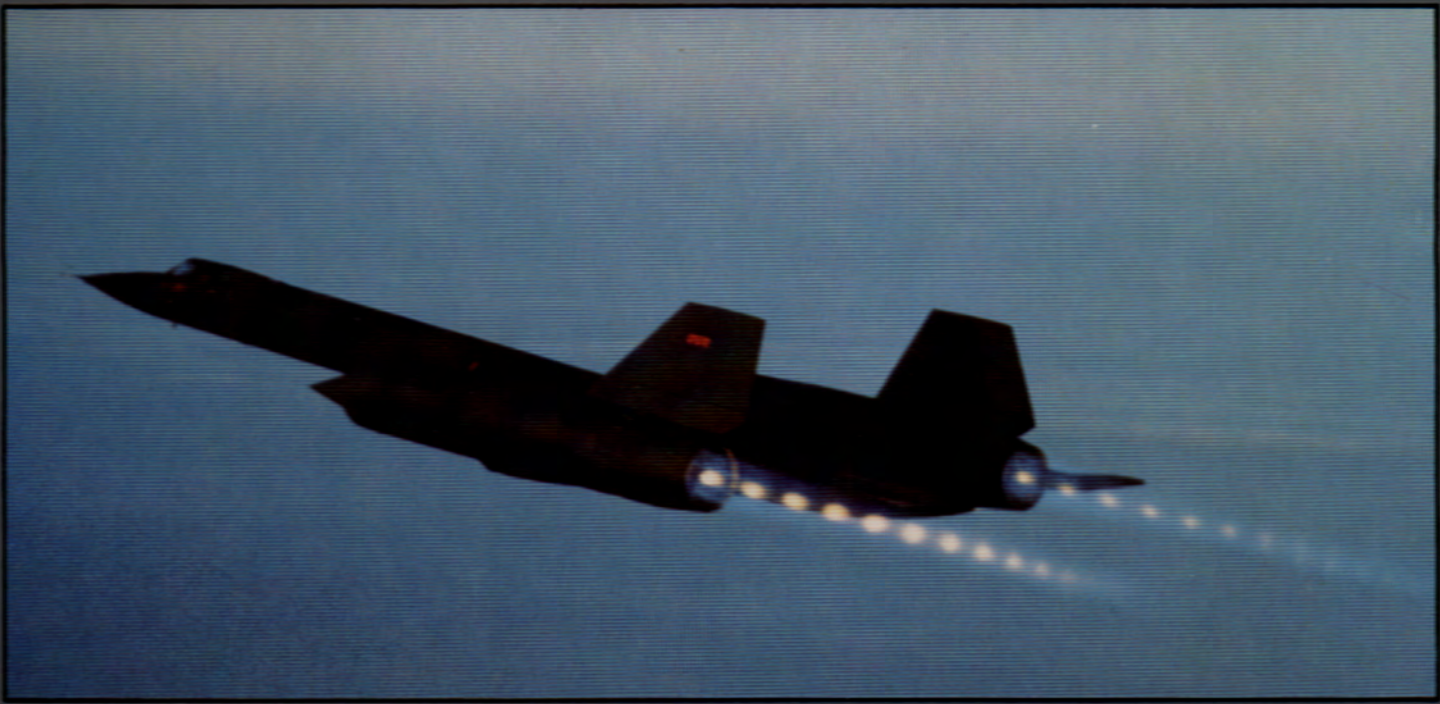
SECTION VI

NOTES

6.1 PUBLICATIONS RELATED TO DESCRIBED STUDIES AND EXPERIMENTS.

The following is a listing of publications related to studies and experiments referenced in sections III, IV, and V.

1	Page 3-7:	Interceptor Aircraft, *Lockheed Proposal No. RD 2190*, November 1982
2	Page 4-4:	Flight-Measured Pressure Characteristics of Aft-Facing Steps in High-Reynolds Number Flow at Mach Numbers 2.20, 2.50, and 2.80 and Comparison With Other Data; *NASA Technical Memorandum No. 72855*, May 1978
3	Page 4-4:	Asbestos-Free Plastic Panels, *Lockheed Engineering Study Proposal No. SR-71-949*, May 1987
4	Page 4-11:	In-flight Boundary-Layer Measurements on a Hollow Cylinder at a Mach Number of 3.0; *NASA Technical Paper No. 1764*, November 1980
5	Page 4-11:	Study - Air Particulate Sampler for the SR-71 Aircraft; *Lockheed Report No. SP-1406*, December 1968
6	Page 4-14: Page 5-1:	Documentation of Mach 3.5 Growth Capabilities, *Lockheed Report No. SP-4322*, January 1975
7	Page 4-14:	Determination of Extended Speed Aerodynamic Effects, *Lockheed Report No. SP-4399*, August 1975
8	Page 4-14:	Determination of Extended Speed Thermodynamic Effects, *Lockheed Report No. SP-4347*, March 1975
9	Page 4-14:	Feasibility Study on Launching a NASA HT-4 Hypersonic Drone from a Lockheed YF-12C Airplane, *Lockheed Report No. SP-1779*, July 1971
10	Page 4-15:	The YF-12A as a Prototype Inlet Flight Test Bed, *Lockheed Report No. SP-4137*, March 1974



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