Rocketdyne
North American Rockwell

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ROCKETDYNE
A DIVISION OF NORTH AMERICAN AVIATION, INC

F-1 ENGINE FAMILIARIZATION
TRAINING MANUAL

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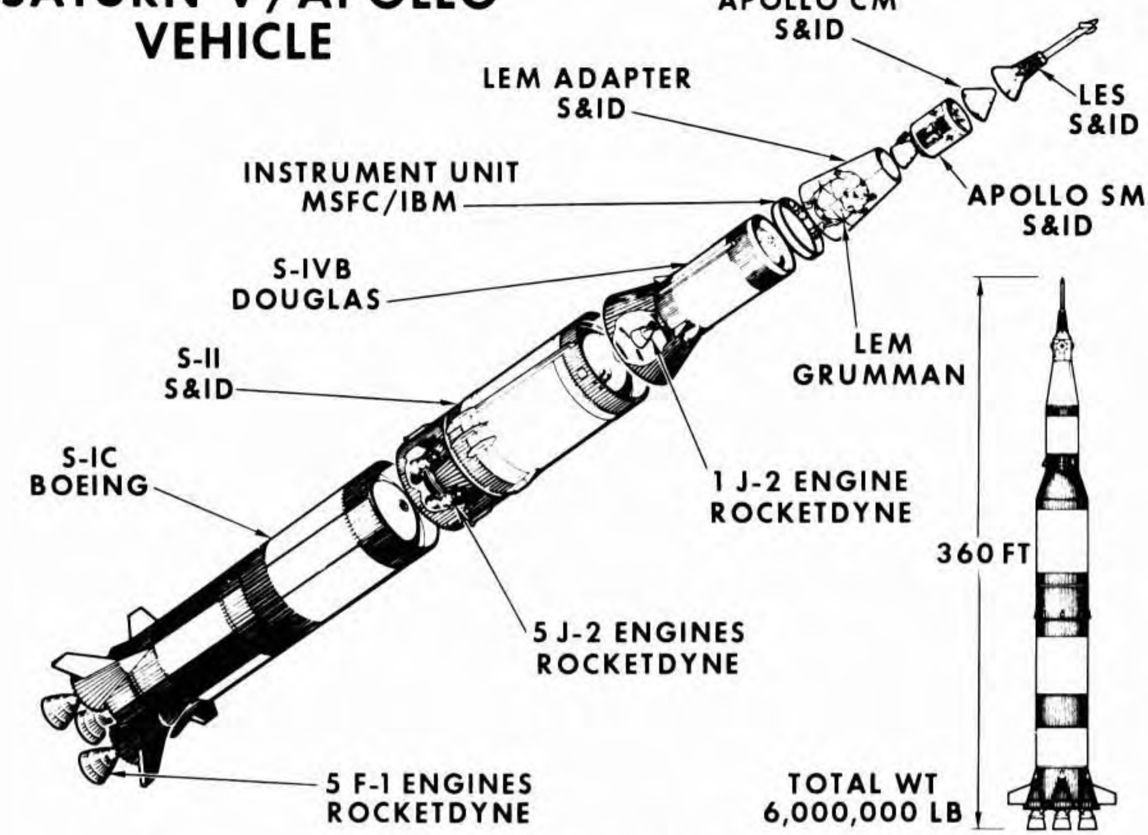
USE THIS DATA FOR
TRAINING PURPOSES ONLY

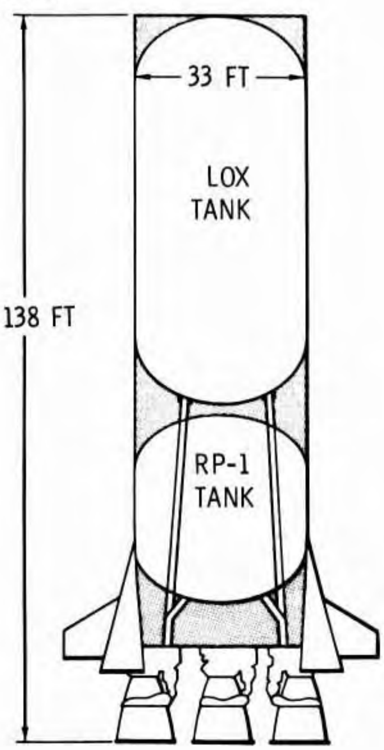
SATURN V/APOLLO VEHICLE

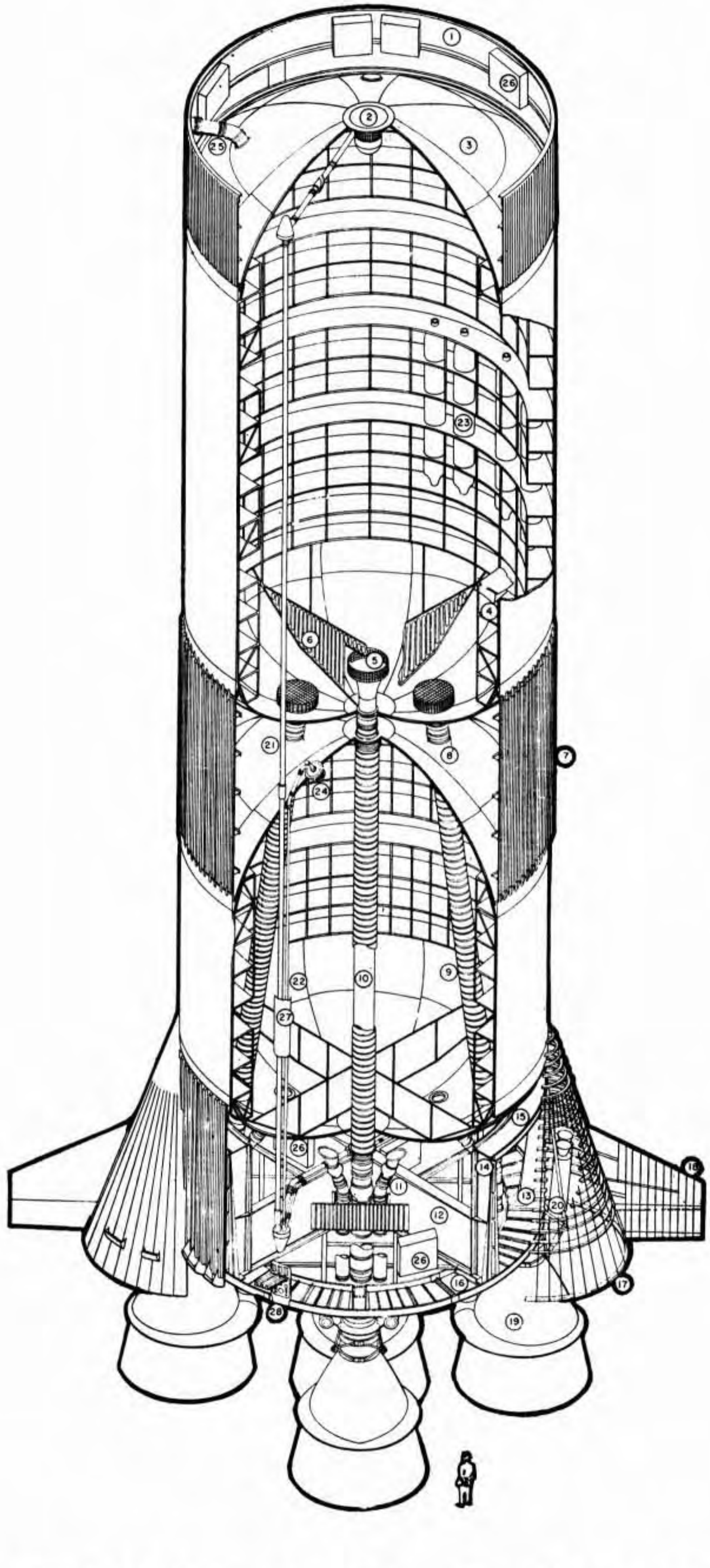


S-IC STAGE

SATURN V, 1ST STAGE

- PRIME CONTRACTOR: BOEING
  ASSEMBLED AT MICHOUD, LA.
- ENGINES: 5 ROCKETDYNE F-1
- THRUST: 7,500,000 LB
  EACH ENGINE: 1.5 MILLION LBS
- BURNING TIME: 150 SEC (2.5 MIN)
- PROPELLANT: 4,400,000 LB
  RP-1 - 206,000 GALS
  LOX - 340,000 GALS
- GROSS WT: 4,800,000 LB
- BURNOUT WT: 425,000 LB
- ALTITUDE: 0-200,000 FT (38 MILES)
- VELOCITY: 0-7,700 FT/SEC (5,460 MPH)





MAJOR COMPONENTS

1. FORWARD SKIRT STRUCTURE
2. GOX DISTRIBUTOR
3. LOX TANK
4. ANTI-SLOSH BAFFLES
5. ANTI-VORTEX DEVICE
6. CRUCIFORM BAFFLE
7. INTERTANK STRUCTURE
8. FUEL TANK
9. SUCTION LINE TUNNELS
10. LOX SUCTION LINES
11. FUEL SUCTION LINES
12. CENTER ENGINE SUPPORT
13. THRUST COLUMN
14. HOLD DOWN POST
15. UPPER THRUST RING
16. LOWER THRUST RING
17. ENGINE FAIRING
18. FIN
19. F-1 ENGINE
20. RETRO ROCKETS
21. GOX LINE
22. HELIUM LINE
23. HELIUM BOTTLES
24. HELIUM DISTRIBUTOR
25. LOX VENT LINE
26. INSTRUMENTATION PANELS
27. CABLE TUNNEL
28. UMBILICAL PANEL

SATURN C-5, S-IC STAGE

SECTION I

DESCRIPTION AND OPERATION

1-1. SCOPE. This section contains a general description of the F-1 propulsion system and a detailed description of each subsystem and component. Engine operation from the preparation phase through and including the engine cutoff phase is defined. Also included, are external inputs necessary for engine operation, typical engine operating parameters, and a description of the flow the engine follows from the time it is accepted by the Customer through Apollo/Saturn V launch.

1-2. F-1 ROCKET ENGINE.

1-3. The F-1 propulsion system was developed to provide the power for the booster flight phase of the Saturn V vehicle. Five engines are clustered in the S-IC stage of the Saturn V to obtain the necessary 7,610,000 pounds thrust.

1-4. The engine features a two-piece thrust chamber that is tubular-walled and regeneratively cooled to the 10:1 expansion ratio plane, and double-walled and turbine gas cooled to the 16:1 expansion ratio plane; a thrust chamber mounted turbopump that has two centrifugal pumps spline-connected on a single shaft driven by a two-stage, direct-driven turbine; one-piece rigid propellant ducts that are used in pairs to direct the fuel and oxidizer to the thrust chamber; and a hypergolic fluid cartridge that is used for thrust chamber ignition.

1-5. The engine is within an envelope 12 feet in diameter and 16 feet long and weighs approximately 18,500 pounds dry. Thrust vector changes are achieved by gimbaling the entire engine. The gimbal block is located on the thrust chamber dome, and actuator attach points are provided by two outriggers on the thrust chamber body.

1-6. Component locations on the engine in the horizontal position are basically referenced to No. 1 (left) (figure 1-1) or No. 2 (right) (figure 1-2) sides of the engine as viewed from the exit end of the thrust chamber with the turbopump at 12 o'clock (top) and the hypergol manifold assembly at 6 o'clock (bottom). Component locations on the engine in the vertical position are referenced to the principal component on the four sides of the engine (eg, gas generator side (No. 1), engine control valve side (No. 2), turbopump side, and hypergol manifold side). A view of the forward end of the engine is shown in figure 1-3.

1-7. ENGINE PHYSICAL DESCRIPTION.

1-8. The F-1 engine is a single-start, fixed-thrust, liquid bipropellant engine, calibrated to develop a sea-level-rated thrust of 1,522,000 pounds with a specific impulse (I_{sp}) of 265.3 seconds. Engine propellants are liquid oxygen and RP-1 fuel at a mixture ratio of 2.27:1. The RP-1 fuel is used as the working fluid for the gimbal actuators and for the engine control system and is also used as the turbopump bearing lubricant. The F-1 engine is comprised of seven operational systems:

(1) A propellant feed system, which supplies pressurized propellants for combustion and hydraulic pressure for the engine control system.

(2) An ignition system, which initiates combustion in the gas generator and the thrust chamber.

(3) A gas generating system, which produces the energy to drive the turbopump and condition propellant tank pressurants.

(4) An engine control system, which regulates the start, operating level, and shutdown of the engine.

(5) A flight instrumentation system, which measures selected engine parameters for monitoring and evaluating the operational characteristics of the engine.

(6) An environmental conditioning system, which protects the engine from extreme temperature environment caused by plume radiation and backflow during flight.

(7) A purge and drain system, which inhibits contamination and facilitates the overboard disposition of expended fluids. Detailed informa- of the engine system and its components are in the following paragraphs. An engine schematic (figure 1-4) and engine parameters (figure 1-5) are included to support the text. Detailed information on engine operation is presented in paragraphs 1-121 through 1-133.

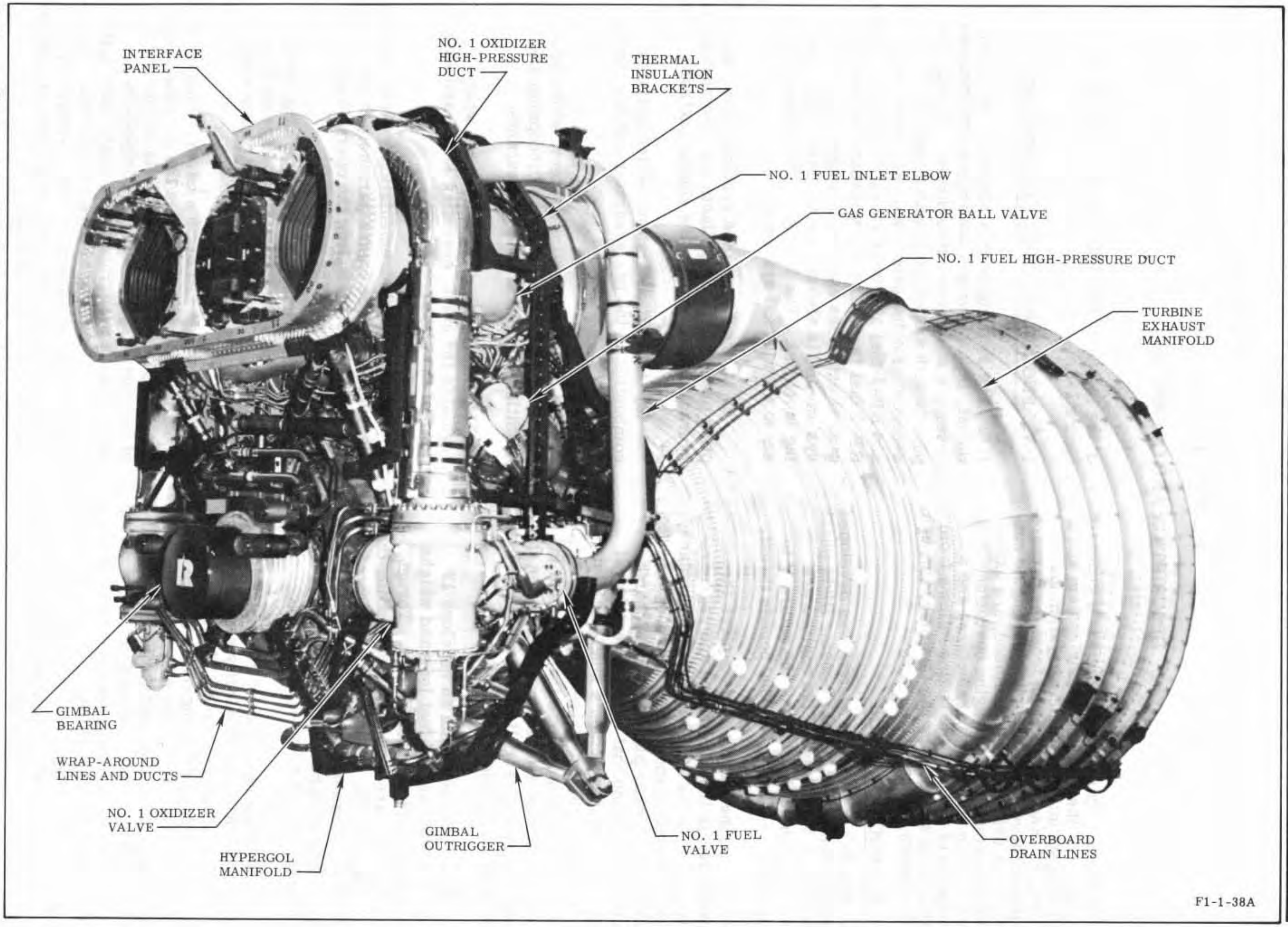


Figure 1-1. F-1 Rocket Engine, Number One Side

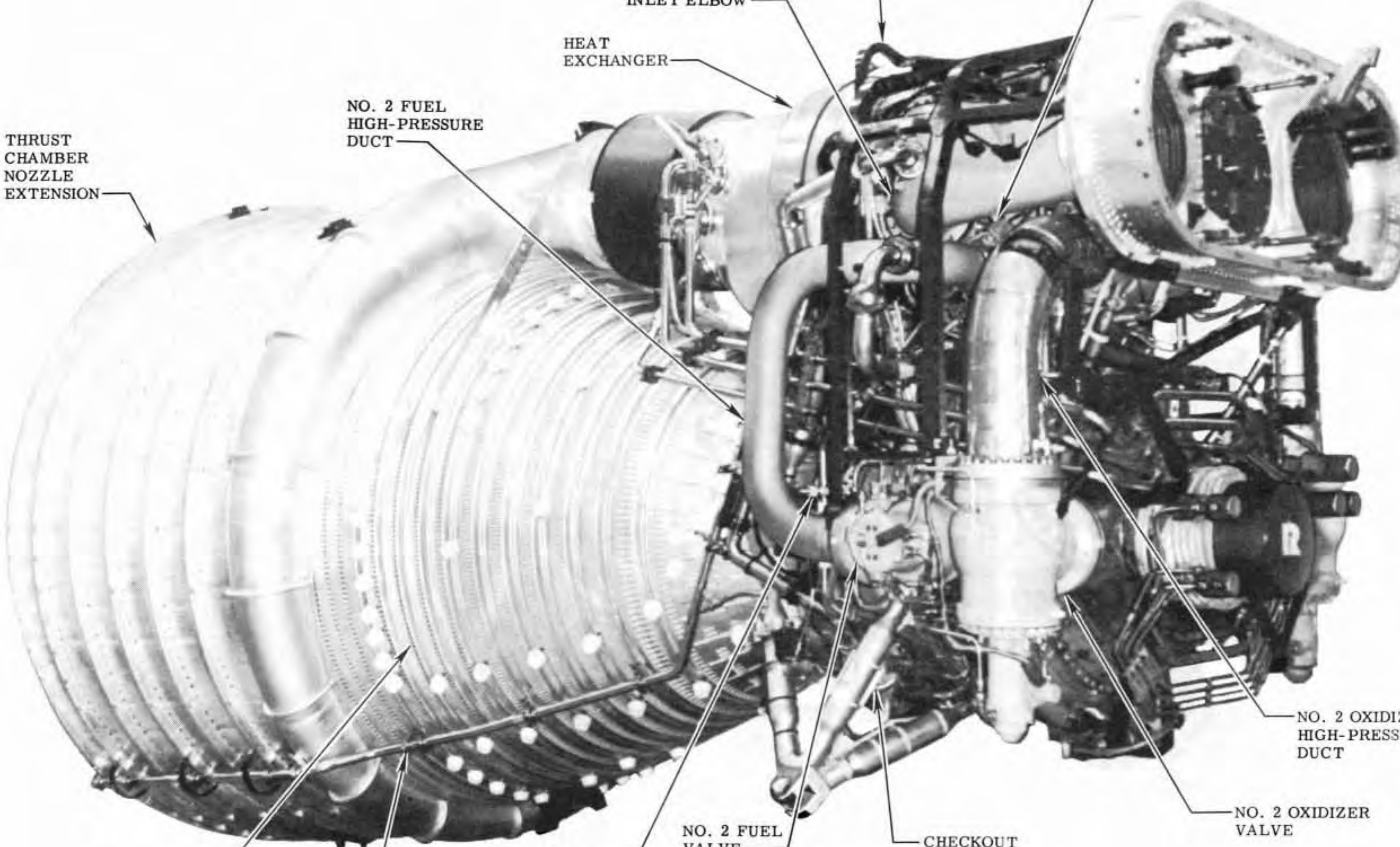


Figure 1-2. F-1 Rocket Engine, Number Two Side

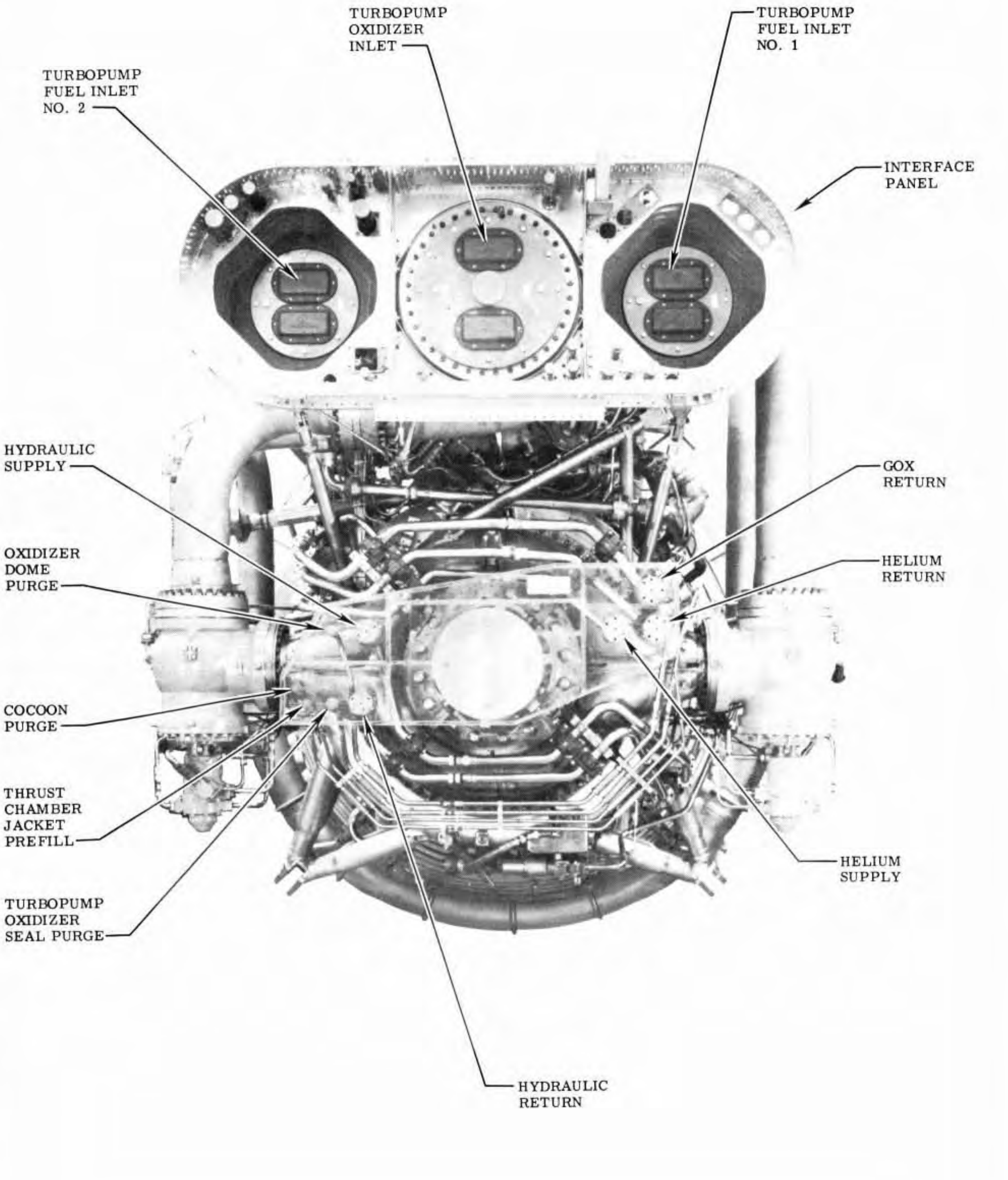


Figure 1-3. F-1 Rocket Engine, Forward End

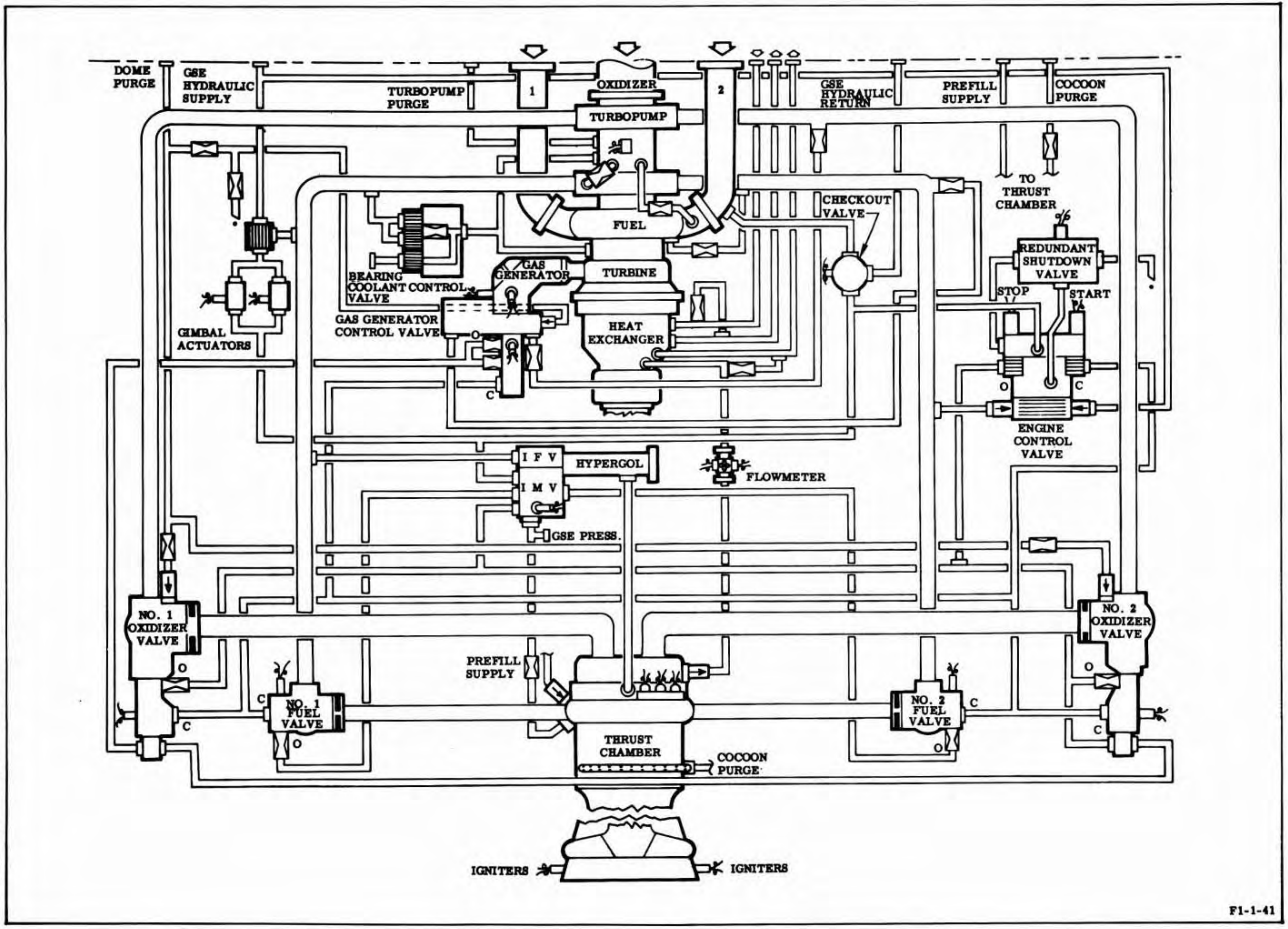


Figure 1-4. F-1 Engine Schematic

Thrust level (sea level)	1,522,000 pounds
Specific impulse (sea level)	265.3 seconds
Total propellant flowrate	5,736 lb/sec (40,644 gpm)
a. Fuel	1,754 lb/sec (15,606 gpm)
b. Oxidizer	3,982 lb/sec (25,038 gpm)
Mixture ratio	2.27:1
Expansion ratio	16:1
Thrust chamber pressure	1,125 psia
Thrust chamber temperature	5,970° F
Thrust chamber exit pressure (16:1)	9.6 psia
Fuel pump discharge pressure	1,870 psia
Oxidizer pump discharge pressure	1,602 psia
Gas generator flowrate (included in total)	167 lb/sec
a. Fuel	118 lb/sec
b. Oxidizer	49 lb/sec
Gas generator mixture ratio	0.416:1
Gas generator combustor pressure	980 psia
Gas generator temperature	1,453° F
Turbine speed	5,492 rpm
a. Time from turbopump initiation to rated speed	5.2 seconds
b. Time from cutoff to zero rpm	3.5 seconds
Turbine brake horsepower	53,146 hp
Nozzle extension coolant gas temperature	1,138° F
Hydraulic recirculation flowrate	11.6 ±1.1 gpm at 1,500 psig
Engine dry weight (average)	18,619 pounds

Figure 1-5. Nominal F-1 Engine Parameters

1-9. PROPELLANT FEED SYSTEM DESCRIPTION.

1-10. The propellant feed system transfers oxidizer and fuel, under pressure, from the propellant tanks to the thrust chamber and gas generator. The system consists of the following major components: A thrust chamber, a turbopump, two oxidizer valves, two fuel valves, two high-pressure oxidizer ducts, two high-pressure fuel ducts, and two fuel inlet elbows.

1-11. THRUST CHAMBER ASSEMBLY DESCRIPTION.

1-12. The thrust chamber assembly (figure 1-6) is the engine section within which the engine thrust is developed and by which this thrust is transmitted to the thrust structure of the booster stage or test stand. The thrust is developed through the process of burning propellants in the combustion chamber and accelerating, to supersonic velocity, the gaseous products of this combustion through an expansion nozzle. The thrust is transmitted through a gimbal bearing and two gimbal actuator outrigger assemblies.

1-13. The thrust chamber assembly consists of a two-piece thrust chamber, an injector, an oxidizer dome and manifold, and a gimbal assembly. The gimbal assembly attaches to the oxidizer dome by eight bolts. The oxidizer dome is bolted to the injector by 16 inner-dome support bolts, and both the oxidizer dome and injector are bolted to the thrust chamber body by 64 outer-dome attach bolts. The dome, injector, and thrust chamber body are indexed to each other by one diamond-shaped and one round, noninterchangeable index pin, spaced 180 degrees apart at the interface flanges below the two oxidizer dome inlets. The mating flanges of the dome and injector are sealed by a Teflon-filled Flexitallic gasket. The mating flanges of the injector and thrust chamber body are sealed at the outer diameter by a Viton-A O-ring and at the inner diameter by a hollow Inconel-X O-ring. The Inconel-X O-ring incorporates drilled holes in its outer diameter to permit injector manifold fuel pressure to enter the hollow section to increase its sealing capability. Thrust chamber parameters are presented in figure 1-7. Thrust chamber and nozzle extension are illustrated in figure 1-8.

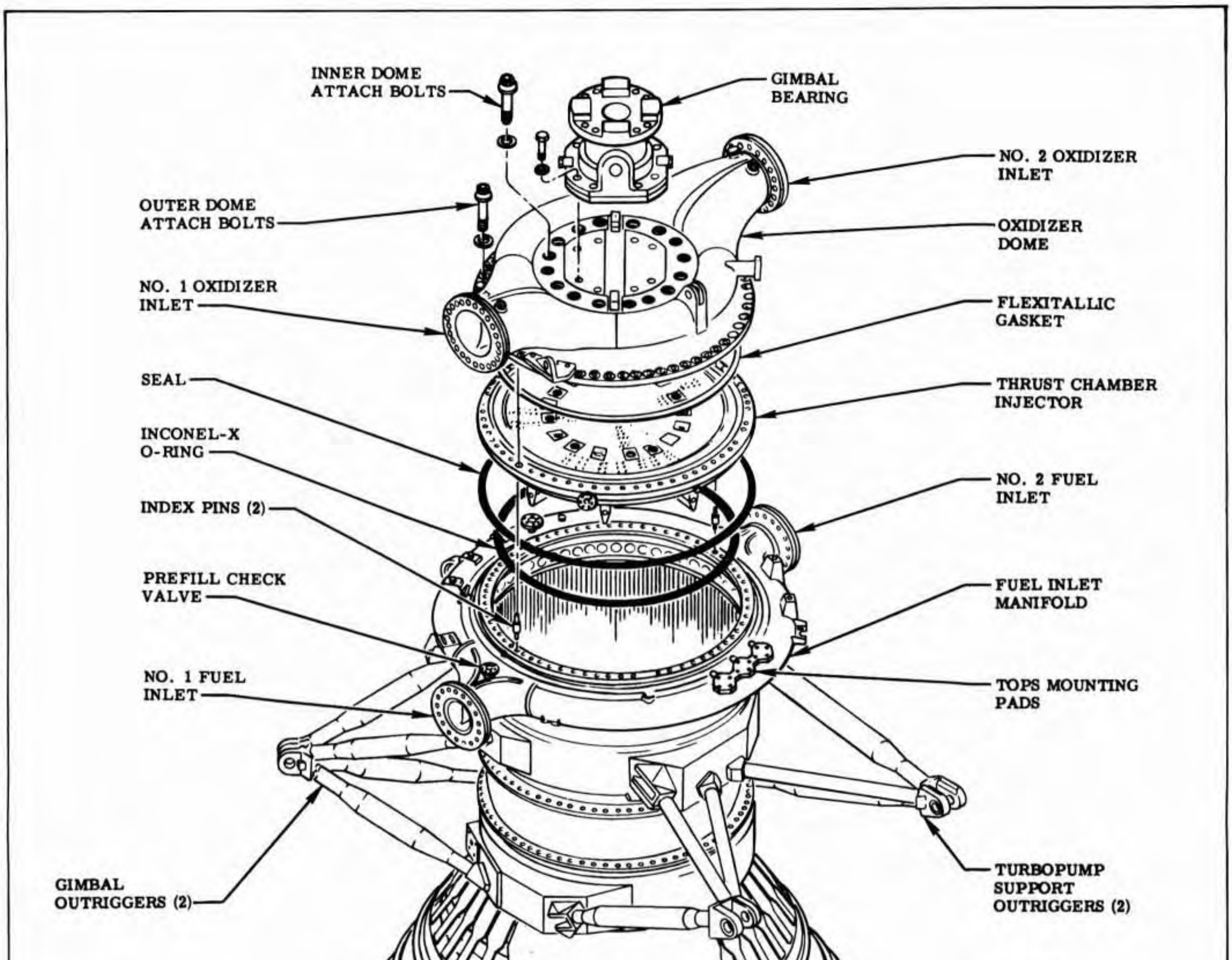


Figure 1-6. Thrust Chamber Assembly

Thrust level (sea level)	1,522,000 pounds	Oxidizer dome pressure drop	57 psia
Mixture ratio	2.40:1	Fuel jacket pressure drop	244 psia
Propellant flowrates		Valves pressure drops	
a. Oxidizer	3,933 lb/sec	a. Oxidizer	91 psia
b. Fuel	1,636 lb/sec	b. Fuel	210 psia
Injector end pressure	1,125 psia		
Fuel injector manifold pressure	1,222 psia	Expansion ratios	
		a. Thrust chamber	10:1
Exit pressure (16:1)	9.6 psia	b. Thrust chamber and nozzle extension	16:1
Combustion area temperature	5,970° F		
Nozzle extension coolant gas temperature	1,138° F		
		Fuel jacket prefill	
Fuel inlet manifold pressure	1,466 psia	a. Solution	Ethylene glycol
		b. Capacity	103-105 gallons
Injector pressure drops			
a. Oxidizer	309 psia		
b. Fuel	97 psia		

Figure 1-7. Nominal Thrust Chamber Parameters

1-14. **THRUST CHAMBER BODY DESCRIPTION.** The thrust chamber body contains a combustion chamber for the burning of the propellants, and a nozzle of the required 10:1 expansion ratio for expelling gases produced by the burned propellants at the supersonic velocity necessary to produce the desired thrust.

1-15. The thrust chamber body is a furnace-brazed, tubular-walled, regeneratively fuel-cooled, bell-shaped chamber incorporating two outrigger arms to support the turbopump and two outrigger arms to which the gimbal actuators attach. A fuel inlet manifold and a turbine exhaust manifold are welded to opposite ends of the chamber. One hundred seventy-eight primary tubes, hydraulically formed from 1-3/32 inch outside diameter Inconel-X tubing, make up the chamber body above the 3:1 expansion ratio plane (approximately 30 inches below the throat centerline plane). Three hundred fifty-six one-inch-outside-diameter secondary tubes of the same material form the chamber from the 3:1 to the 10:1 expansion ratio plane. A raised weld bead with the tube number and a directional flow arrow, identify fuel-up tube No. 1 and fuel-down tubes No. 60 and 120 on the chamber internal faces of the injector end ring and fuel return manifold. External to the chamber the same tubes are similarly identified on reinforcing bands and straps below the thrust chamber throat.

1-16. Two secondary tubes are brazed to each primary tube at the 3:1 expansion ratio area plane. Every other primary tube is a fuel-down tube and is slotted on its outboard side at the fuel inlet manifold area into which fuel from the inlet manifold is directed. An orificed plug is brazed into the tube above the slot to permit 30 percent of the fuel to go directly to the fuel injector manifold. The remaining 70 percent of the fuel is used for regeneratively cooling the thrust chamber and is directed down the tube to the fuel return manifold at the end of the chamber. From the fuel return manifold, the fuel is directed by the adjacent fuel return tubes to the fuel injector manifold. The return manifold is welded to the bottom of the thrust chamber secondary tubes and incorporates four drain ports, located 90 degrees apart, to drain residual fluids. Forty lugs are welded to the inside wall of the return manifold for attaching the turbine exhaust leak-test fixture.

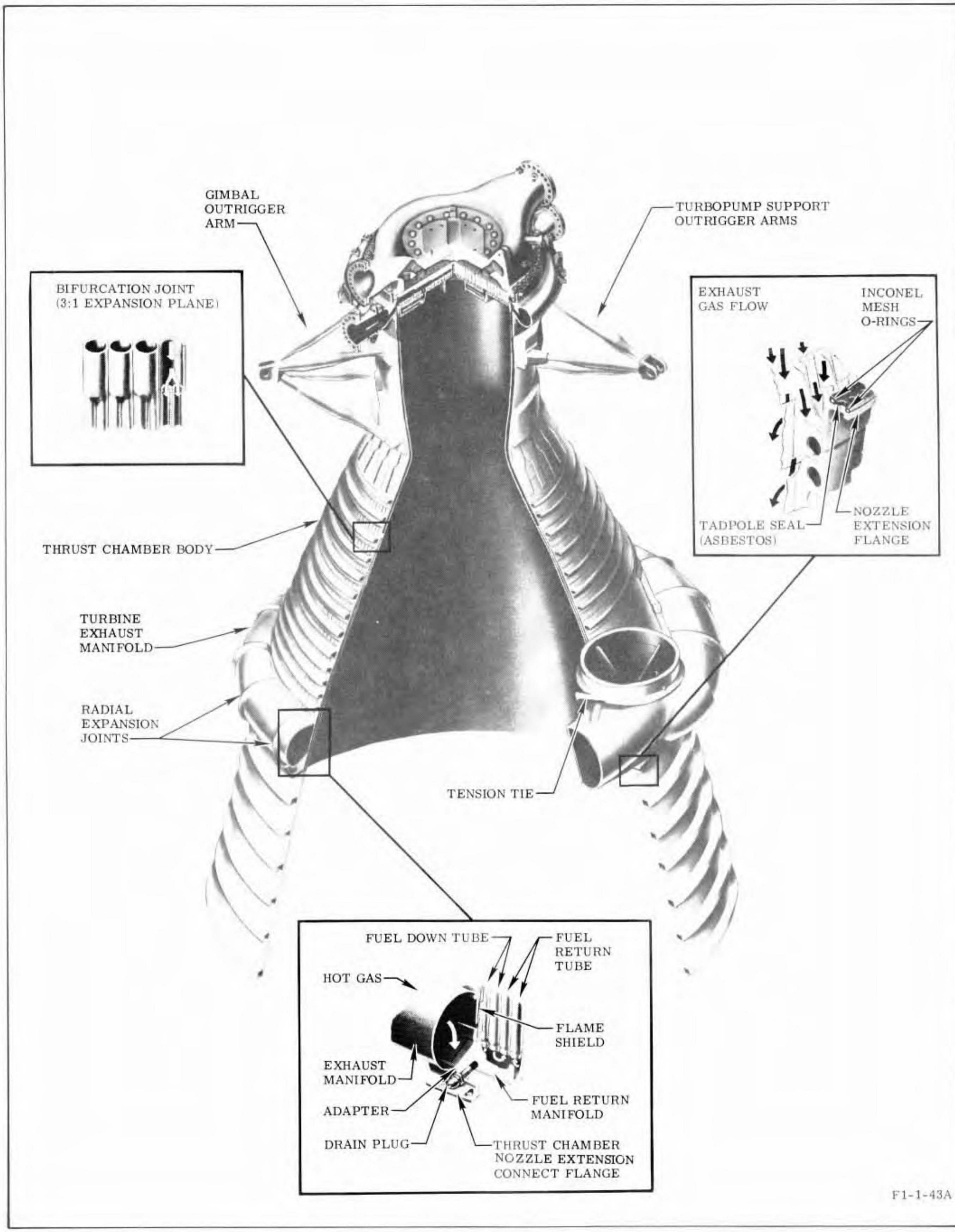


Figure 1-8. Thrust Chamber and Nozzle Extension