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LIQUID ROCKET ENGINES

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INTRODUCTION

This paper presents a discussion on liquid propellant rocket engines. The first part contains a discussion on liquid propellants, including a description of various propellant types such as cryogenic, storable, bipropellant, and monopropellant. This part also points out desirable physical properties and includes a section on performance outlining the methods by which performance is calculated and shows performance for various liquid rocket propellant combinations.

The second part of this paper contains a discussion on various engine components and methods for controlling the rocket engine. The components to be described include the thrust chamber and parts thereof including the injector, chamber, nozzle, cooling methods, etc. A description is given of a pressure-fed liquid rocket engine and a pump-fed liquid rocket engine; the various types of feed systems, including turbomachinery and pressurization systems are also discussed. Control systems, including methods of start, stop, throttle, etc. are described.

The third section of this paper includes applications and examples of various liquid propulsion systems including space engines, pre-packaged liquid missile systems, and space launch vehicle systems.

The final section of the paper includes a discussion on future trends of liquid propellant rocket engines including advanced nozzles, cycles, combustors, etc.

LIQUID PROPELLANTS

PROPELLANT PROPERTIES AND SPECIFICATIONS

It is important to distinguish between the characteristics and properties of the liquid propellants--that is, the fuel and oxidizer liquids in their unreacted condition, and the properties of the hot gas mixture that results from the reaction in the combustion chamber. The chemical mixture of the liquid propellants determines the properties and characteristics of both of these types.

A high content of chemical energy per unit of propellant mixture is desirable because it permits a high chamber temperature. Because of incomplete combustion, friction, and dissociation, the full value of the propellant energy is, however, never realized. This heat content differs from that used in conventional combustion engines, because it refers to a unit weight of mixture and not a weight of fuel.

TYPES OF PROPELLANTS

The following definitions (Fig. 1) of various types of propellants will help classify the propellants for future reference. The mission and operating environment of a specified vehicle will often dictate the type of propellant required for the vehicle.

DEFINITIONS

MONOPROPELLANTS

BIPROPELLANT

CRYOGENIC PROPELLANT

STORABLE PROPELLANT

Figure 1. Types of Liquid Propellants

Monopropellant

A monopropellant contains an oxidizing agent and combustible matter in a single substance. It may be a mixture of several compounds such as hydrogen peroxide mixed with alcohol; or it may be a homogenous chemical agent, such as nitromethane. Monopropellants are stable at ordinary atmospheric conditions but decompose and yield hot combustion gases when heated and pressurized. The feed system of monopropellant units is usually simple because only one liquid needs to be supplied.

Bipropellant

A bipropellant rocket unit has two separate propellants, usually an oxidizer and a fuel. They are stored separately and are not mixed outside the combustion chamber. The majority of successful liquid propellant rocket units have used bipropellants.

Cryogenic Propellant

A cryogenic propellant is liquefied gas at low temperature, such as liquid oxygen (-297°F) or liquid hydrogen (-423°F). Provisions for venting the storage tank and minimizing vaporization losses are necessary with this type.

Storable Propellant

Storable propellants (for example, nitric acid or gasoline) are liquid at ambient temperatures and they can be stored for long periods in sealed tanks.

DESIRABLE PHYSICAL PROPERTIES (Fig. 2)

Low Freezing Point

This permits operation of rockets in cold weather. The addition of small amounts of special chemicals has been found to help depress the freezing point of liquid propellants which otherwise solidify readily at relatively high temperatures.

High Specific Gravity

To accommodate a large weight of propellants in a given vehicle tank space, a dense propellant is required. It permits a small vehicle construction and, consequently, a relatively low structural vehicle weight and low aerodynamic drag. Specific gravity, therefore, has an important effect on the maximum flight velocity and range of any rocket-powered vehicle or missile.

DESIRABLE PHYSICAL PROPERTIES

LOW FREEZING POINT

HIGH SPECIFIC GRAVITY

GOOD CHEMICAL STABILITY

DESIRABLE HEAT TRANSFER PROPERTIES

DESIRABLE PUMPING PROPERTIES

DESIRABLE TEMPERATURE VARIATION

Figure 2. Desirable Physical Properties

Stability

No deterioration with storage and no reaction with the atmosphere is desirable. Good chemical stability means no decomposition of the liquid propellant during storage, even at elevated temperatures.

A good liquid propellant should also have a negligible chemical reaction with piping, tank walls, valve seats, and gasket materials, even at relatively high ambient temperatures.

No appreciable absorption of moisture and no adverse effects of small amounts of impurities are desirable properties.

Heat Transfer Properties

High specific heat, high thermal conductivity, and a high boiling or decomposition temperature are desirable for propellants which are used for thrust chamber cooling.

Pumping Properties

A low vapor pressure will permit not only easier handling of propellants, but also a more effective pump design in applications where the propellant is pumped. If the viscosity of the propellant is too high, then pumping and engine-system calibration become difficult. Propellants with high vapor pressure, such as liquid oxygen, liquid methane, or other liquefied gases, require special design provisions, unusual handling techniques, and often special low-temperature materials.

Temperature Variation

The temperature variation of the physical properties of the liquid propellant should be small. For example, a wide temperature variation in vapor pressure and density (thermal coefficient of expansion) or an unduly high change in viscosity with temperature will make it very difficult to accurately calibrate a rocket engine flow system or predict its performance over any reasonable range of operating temperatures.

PERFORMANCE

The performance of a rocket propellant is characterized primarily by the specific impulse, I_s.

Specific impulse is defined as the thrust delivered per unit weight rate of propellant consumed; $I_s = F/\dot{W}$. Specific impulse is dependent both on the inherent thermo-chemical properties of the propellant and on the efficiency of the expansion process. The latter depends, in turn on the nozzle entrance, nozzle exit, and ambient pressures. To separate approximately the combustion and expansion processes, use is made of two quantities which can conveniently be measured experimentally, the characteristic velocity, c*, and the nozzle thrust coefficient, C_F (Fig. 3). These quantities are defined by the equations: $c^* = \frac{P_c\ A_t\ g_c}{\dot{W}}$ and

CHARACTERISTIC VELOCITY

$$C^{\star} = \frac{P_c \, A_t \, g_c}{\dot{W}}$$

NOZZLE THRUST COEFFICIENT

$$C_F = \frac{F}{P_c \, A_t}$$

Figure 3. Characteristic Velocity and Nozzle Thrust Coefficient

$C_f = \dfrac{F}{P_c\, A_t}$ where P_c is the nozzle entrance stagnation pressure and A_t is the nozzle throat area. For systems in which no reaction occurs in the nozzle, the nozzle entrance and throat stagnation pressures are equal. However, if chemical reaction (combustion, recombination, or condensation) occurs in the convergent section of the nozzle, these pressures are not equal and conditions at the throat must be calculated. Therefore, the frequently made statement that characteristic velocity is a function solely of combustion chamber conditions is correct only in the absence of chemical reaction in the nozzle. More generally, the characteristic velocity is a function of conditions upstream of the nozzle throat; the thrust coefficient is a function of the nozzle geometry downstream of the throat and of the exit and ambient static pressures; and both are functions of the thermodynamic properties of the combustion products.

Theoretical performance of various liquid rocket propellant combinations are shown in Fig. 4. Based on a combustion chamber pressure of 1000 psia, a nozzle exit pressure of 14.7 psia, and optimum nozzle expansion ratio, the specific impulse ranges from approximately 289 seconds for a storable combination to 460 seconds for a high-energy cryogenic system.

PROPELLANT COMBINATION	THEORETICAL SPECIFIC IMPULSE *	MEAN DENSITY	STATUS
CRYOGENIC			
$LO_2 - RP\text{-}1$	300	1.02	OPERATIONAL
$LO_2 - H_2$	391	0.28	OPERATIONAL
$OF_2 - H_2$	401	0.39	LABORATORY
$F_2 - H_2 - Li$	432	0.17	THEORETICAL
$O_2 - H_2 - Be$	458	0.23	THEORETICAL
STORABLES			
$N_2O_4 - N_2H_4$-/UDMH(50-50)	289	1.21	OPERATIONAL
$N_2H_4 - B_5H_9$	299	1.12	EXP. ENGINE TESTING
$ClF_3 - B_5H_9$	289	1.47	EXP. THRUST CHAMBER TESTING
$N_2O_4 - N_2H_4 - Be$	326	1.21	LABORATORY

*1000 PSIA AT SEA LEVEL

Figure 4. Liquid Rocket Propellants

THRUST CHAMBER

The thrust chamber is the combustion device in which the liquid propellants are metered, injected, atomized, mixed, and burned to form hot, gaseous reaction products which, in turn, are accelerated and ejected at high velocity, producing thrust.

A typical rocket thrust chamber assembly (Fig. 5) consists of the following principal parts: nozzle, combustion chamber, injector, mounting provision, and an ignition system, if non-spontaneously ignitable propellants are used. In some cases the thrust chamber assembly also includes integrally mounted propellant valves and controls. Figure 6 shows an F-1 thrust chamber that produces 1,500,000 lb of thrust with LOX/RP-1 propellants.

NOZZLE

The nozzle is that part of the rocket thrust chamber assembly in which the gases are accelerated to high velocities. For prolonged duration firing, nozzles have to be cooled. The nozzle size, shape, and proportions determine the chamber pressure, thrust, propellant flow, and exhaust velocity of thrust chambers and the variations of these performance parameters. Figure 7 shows a comparison of various nozzle shapes and sizes.

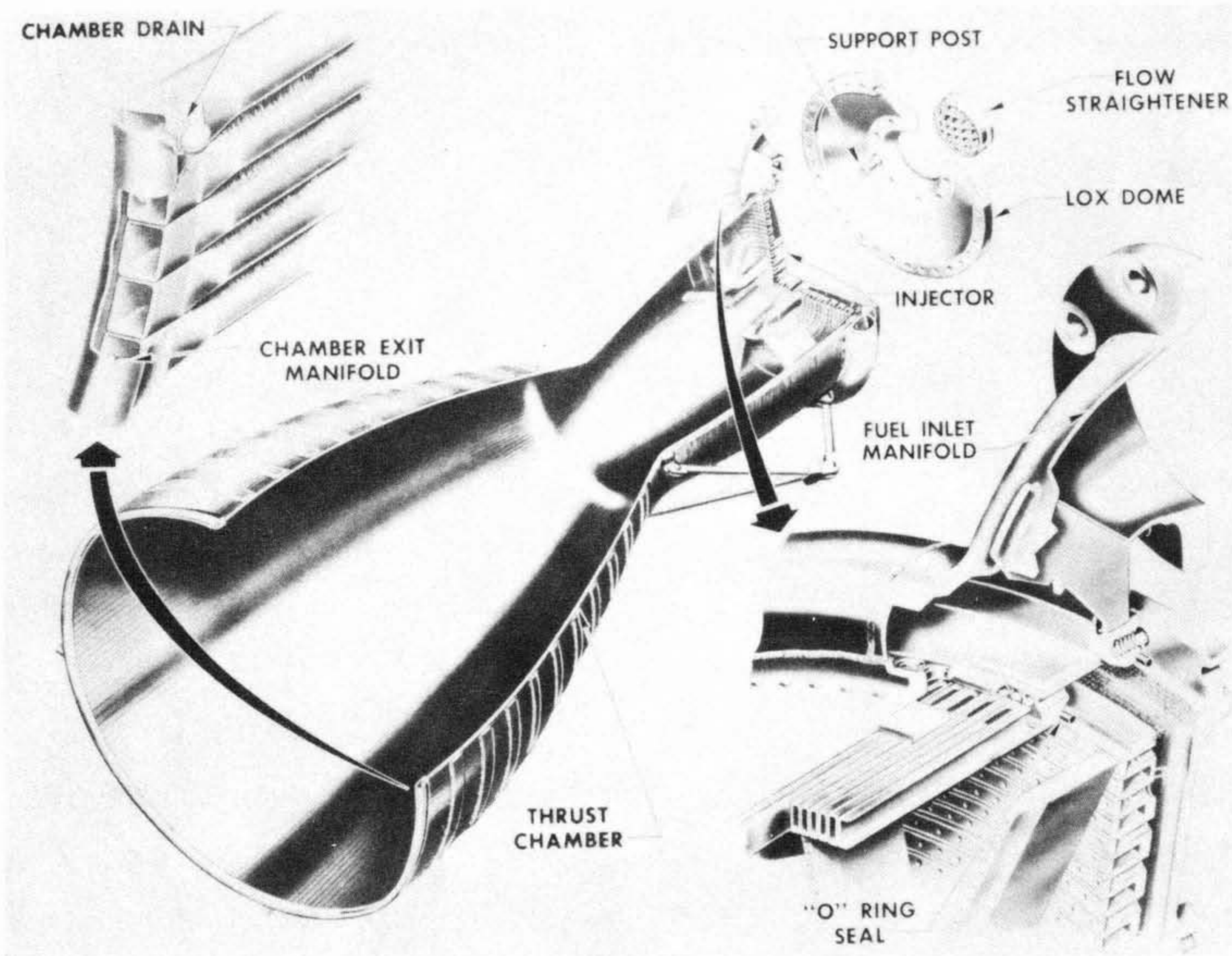


Figure 5. Thrust Chamber Assembly

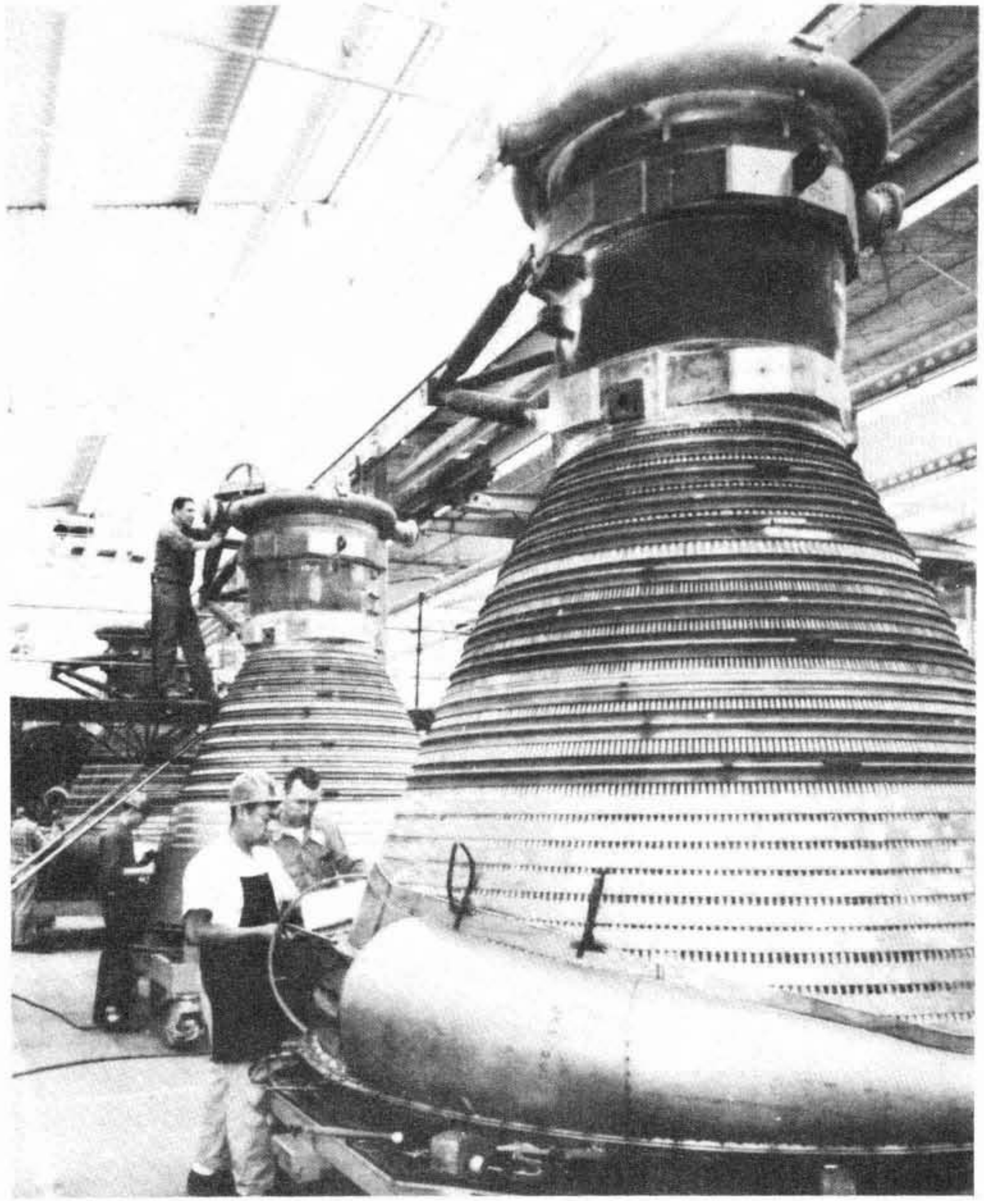


Figure 6. Large Thrust Chamber

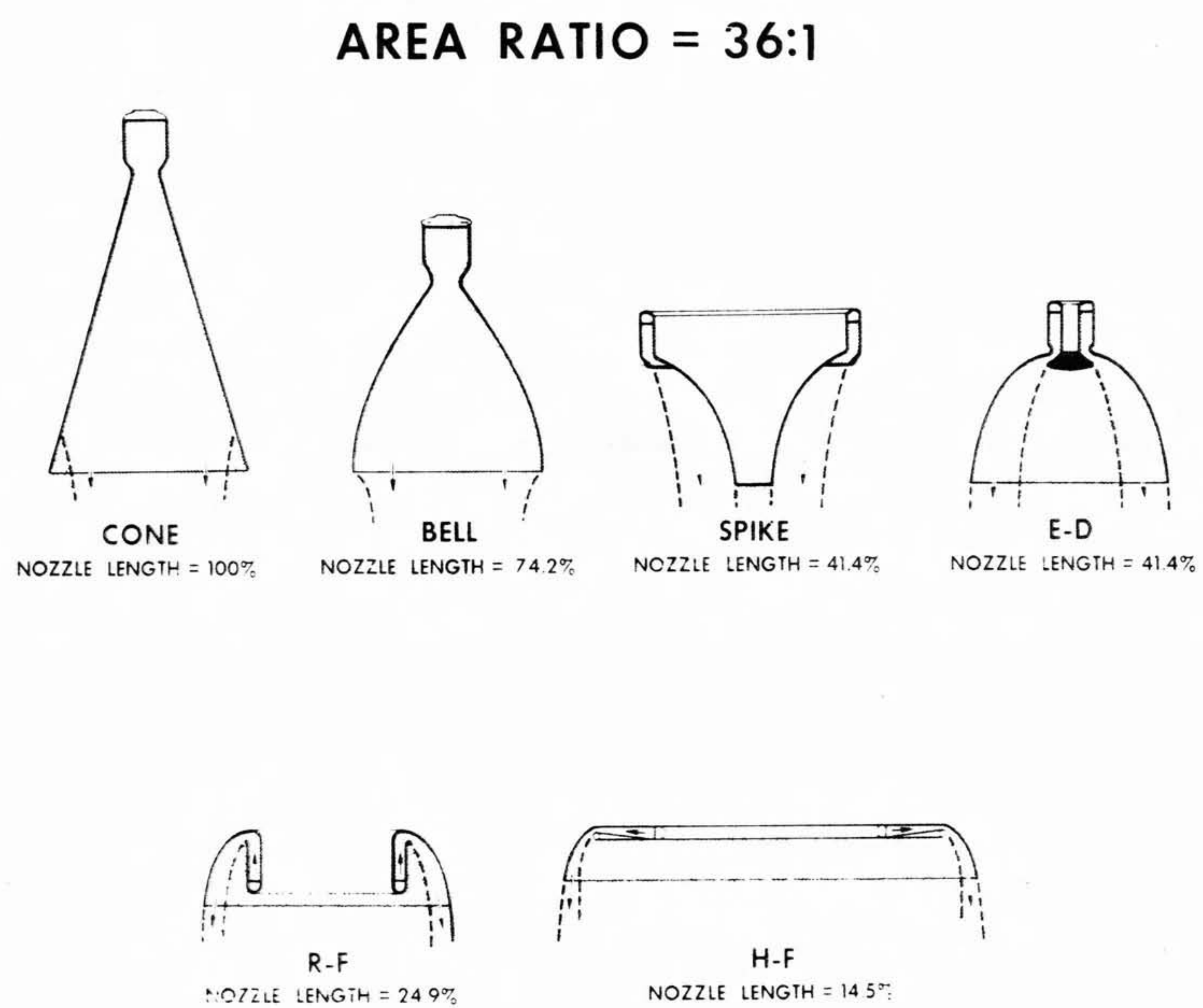


Figure 7. Comparison of Nozzle Shapes

The first rocket nozzles were built with the shape of a cone. It was later found by using the methods of characteristics, a contour could be designed and built which would produce the same or higher nozzle efficiency and yet be shorter than a cone nozzle. This was designated a bell nozzle. More recent analytical and experimental work has produced the spike, E-D, R-F and H-F nozzles, all progressively having shorter over-all lengths than the cone nozzle.

CHAMBER

The chamber is that part of the rocket thrust chamber assembly in which the combustion or burning of propellants takes place at a high pressure. Selection of the combustion chamber volume in a liquid propellant rocket thrust chamber is determined by the following considerations.

1. The volume has to be adequate to permit adequate mixing, evaporation, and complete combustion of propellants. Chamber volumes vary for different propellants with the time delay necessary to vaporize and activate the propellants and with the speed of reaction of the propellant combination.

2. For prolonged firing durations, the wall surface area has to be cooled. To reduce the cooling requirements it is desirable to decrease the exposed wall surface area and the local heat transfer intensity.

3. Weight is a premium.

4. Manufacturing and design considerations indicate a preference for a simple chamber geometry, such as a cylinder or a sphere.

5. The maximum chamber diameter often determines various vehicle dimensions. A small unit, therefore, often permits a lower aerodynamic drag and a smaller structure.

6. The chamber volume and geometry determine certain acoustic vibration frequencies. Sometimes the chamber volume can be selected to avoid the excitation of natural frequencies which may be critical to other components in the vehicle.

7. The gas pressure drop for accelerating the combustion products with the chamber should be a minimum; any reduction in the nozzle inlet pressure will reduce the exhaust velocity and the performance of the vehicle.

COOLING OF THRUST CHAMBER ASSEMBLIES

Uncooled Liquid Propellant Thrust Chambers

In uncooled thrust chambers the operating duration is usually limited by the ability of an ingenious design to absorb heat. In cooled thrust chambers one of the propellants is used in forced-convection heat transfer and the duration is limited only by the availability of the propellant from its feed system. In designs where walls (made of special high temperature materials) can take loads and operate at the actual flame temperature without forced convection cooling, there is no limit on operating durations.

Radiation-Cooled Chambers. A high-temperature refractory metal is used for the wall and/or nozzle material. It glows white hot, loses its heat by radiation, and can operate for indefinite durations.

Figure 8 shows a radiation-cooled thrust chamber which has operated for 3100 seconds on N_2O_4/N_2H_4-UDMH (50-50) propellants at a chamber pressure of 25 psia.

Ablative Thrust Chambers. An ablative material, such as certain types of plastic, glass, or glass-reinforced plastics is used to form the chamber. The absorption of energy takes place by melting or vaporizing the material. The melted matter flows as a viscous fluid boundary layer and acts as a protection to the wall. Figure 9 shows an ablative thrust chamber.

Thermal Insulators. Two types of thermal insulated chambers have been developed: those with thin layers of special material painted or otherwise applied to a surface, and those with heavy ceramic wall linings. Both require a basic thrust chamber pressure shell structure, usually of metal.

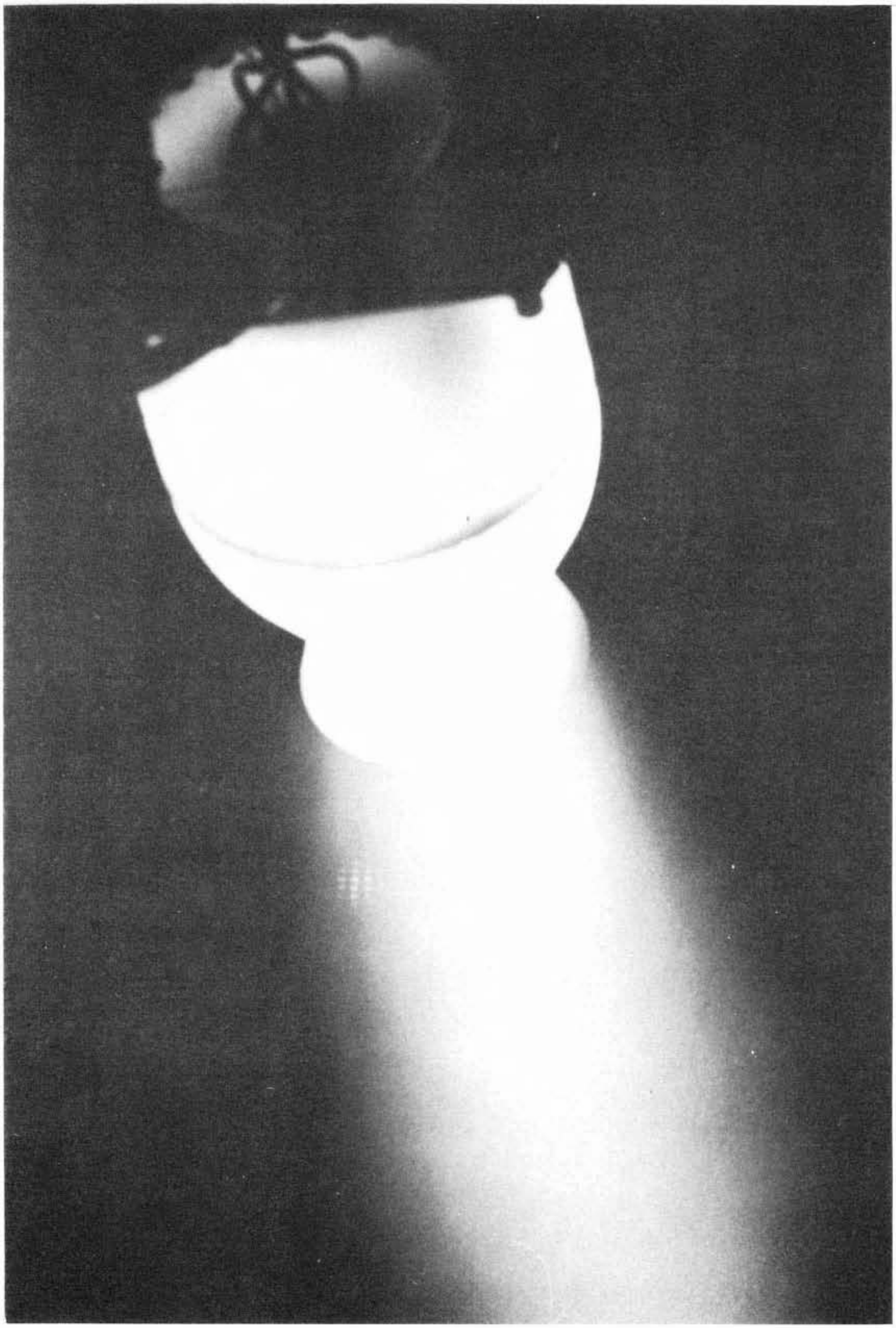


Figure 8. Radiation-Cooled Thrust Chamber

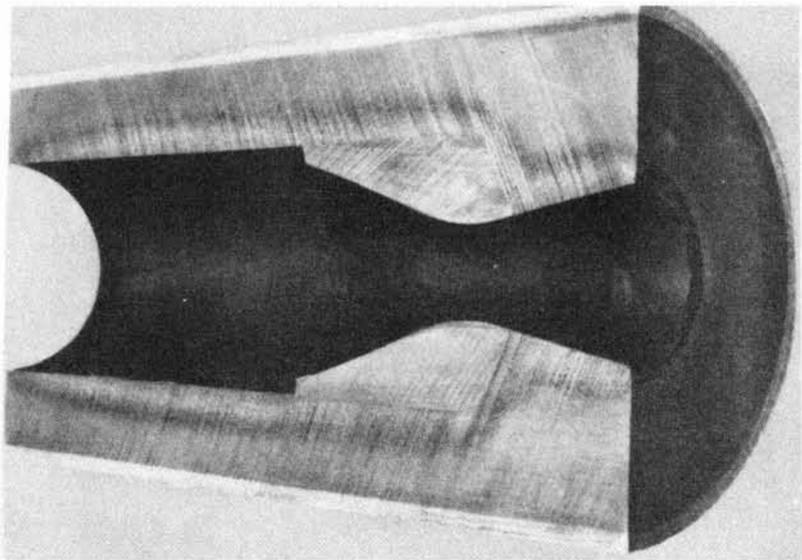


Figure 9. Ablative Thrust Chamber

Cooled Liquid Propellant Thrust Chambers

Cooled thrust chambers have provisions for cooling some or all metal parts coming into contact with hot gases, such as chamber walls, nozzle walls, and injector faces. A cooling jacket or cooling coil permits the circulation of a coolant. Jackets often consist of separate liners and outer walls or of an assembly of contoured, adjacent tubes. The inner wall confines the gases, and the space between the walls serves as the coolant passage.

The nozzle throat region is usually the location which has the highest heat-transfer intensity and is, therefore, the most difficult to cool. For this reason the cooling jacket is often designed so that the coolant velocity is highest at the critical regions by restricting the coolant passage cross section, and so that the fresh cold coolant enters the jacket at or near the nozzle.

While the selection of the coolant velocity and its variation along the wall for any given thrust chamber design depends on heat-transfer considerations; the selection of the coolant passage geometry often depends on pressure loss and manufacturing considerations.

In regenerative cooling the thrust chamber walls are cooled by a built-in jacket or cooling coil in which the oxidizer or the fuel is used as the coolant fluid. The heat absorbed by the coolant is, therefore, not wasted; it augments the initial energy content of the propellant prior to injection, increasing the exhaust velocity slightly.

The tubular chamber and nozzle construction is favored for larger rockets. This design combines the advantages of a thin wall (good for reducing thermal stresses and high wall temperatures) and a cool, good, lightweight structure. Tubes are formed to special shapes and contours usually by hydraulic means, and then brazed, welded or soldered together. In order to take the gas pressure loads in hoop tension, the tubes are reinforced on the outside by high-strength bands or wires.

Figure 10 shows the heat flux vs. combustion chamber pressure for several propellant combinations. An analysis is shown, for example, of a very high-thrust (six million pounds), high-pressure (3000 psia) liquid-propellant booster rocket with estimated heat transfer rates of 40 to 60 Btu/in.^{2}-sec. This system will probably use regenerative cooling augmented in the areas of high heat flux by film cooling or transpiration cooling.

Film cooling is a method of cooling whereby a thin fluid film covers and protects exposed wall surfaces from excessive heat transfer. The film is introduced by injecting small quantities of fuel, oxidizer, or an inert fluid at very low velocities in a large number of orifices along the

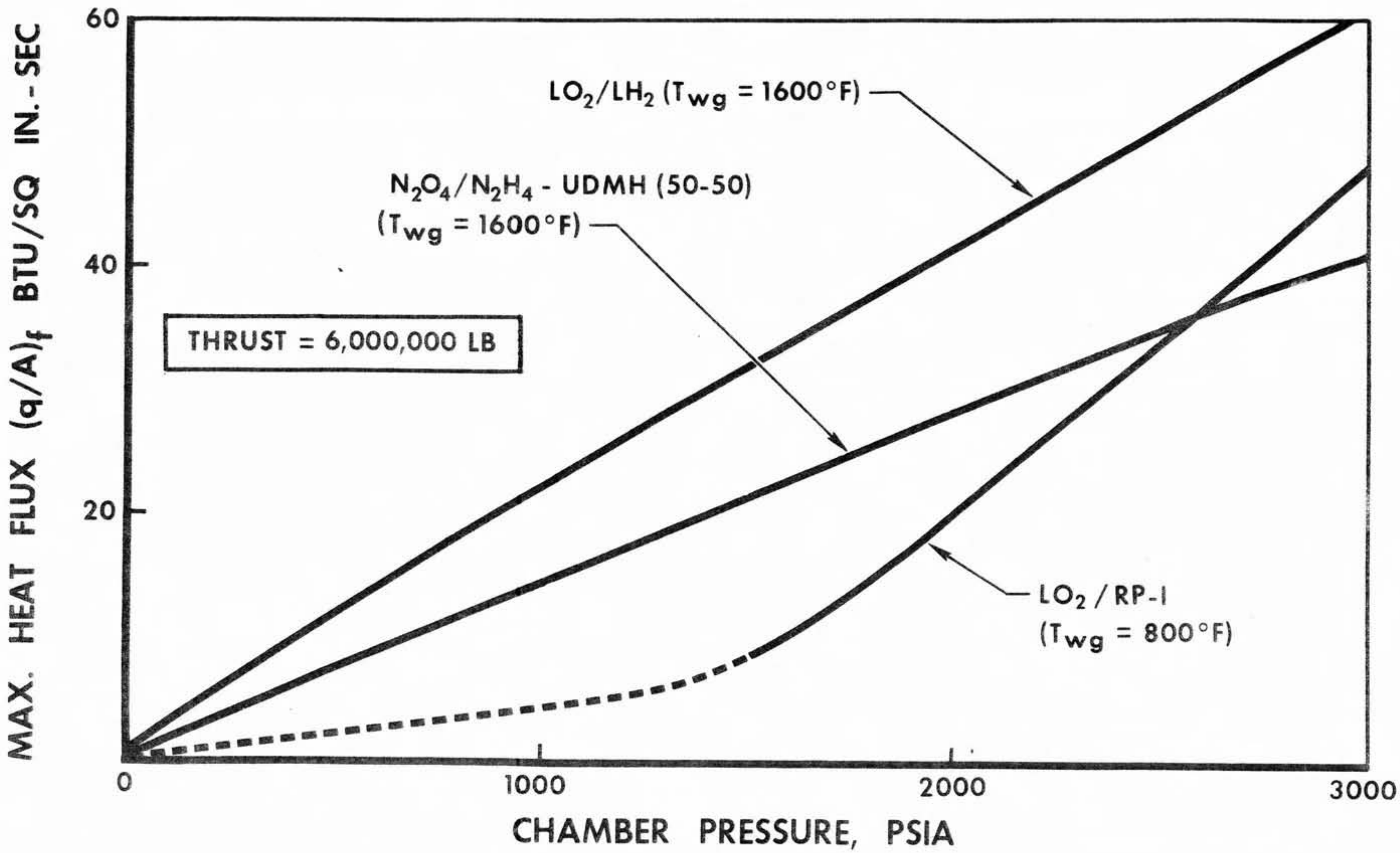


Figure 10. Heat Flux vs Chamber Pressure

exposed surfaces in such a manner that a protective film will be formed. This method is quite effective because it forms a relatively cool boundary layer and because the coolant is able to absorb considerable heat by evaporation. A coolant with a high heat of vaporization and a high boiling point is particularly desirable. Film cooling is often used to augment regenerative cooling in those regions of the chamber and nozzle wall where regenerative cooling alone is insufficient. It does decrease the performance of the rocket slightly because the fluid used in the coolant film is not completely burned and is, therefore, not used efficiently.

A special type of film cooling, sweat cooling or transpiration cooling, uses a porous wall material which admits a coolant through porous material uniformly distributed over the surface. Considerable difficulty has been encountered in making the coolant distribution uniform along a porous surface because the pressure drop across the inner thrust chamber wall varies along the axis of the chamber, particularly in the nozzle region. The problem of manufacturing large chamber pieces of uniform porosity, variable thickness, and complex shape requires considerable ingenuity.

INJECTOR

The functions of the injector are similar to those of a carburetor in an internal combustion engine. The injector has to introduce and meter the flow to the combustion chamber, and atomize and mix the propellants in such a manner that a correctly proportional, homogeneous fuel/oxidizer mixture will result, i.e., one that can readily be vaporized and burned. Several general types of injectors are shown in Fig. 11.

In the impinging stream-type, multiple hole injectors, the propellants are injected through a number of separate small holes so that the fuel and oxidizer streams impinge upon each other. Impingement will aid atomization of liquids into droplets and will also aid in distribution. The majority of rocket engines have used an impinging stream-type injector. The differences of the various types shown in the above figure are reflected in different starting characteristics, atomization, heat transfer, resistance to self-induced vibration, and other design characteristics.

The non-impinging or showerhead injector employs non-impinging streams of propellants usually emerging normal to the face of the injector. It relies on turbulence and diffusion to achieve mixing.

Splash plate-type injectors are intended to promote propellant mixing in the liquid state and use the principle of impinging the propellant streams against a surface. This injection method has been successfully used with certain nitric acid propellant combinations.

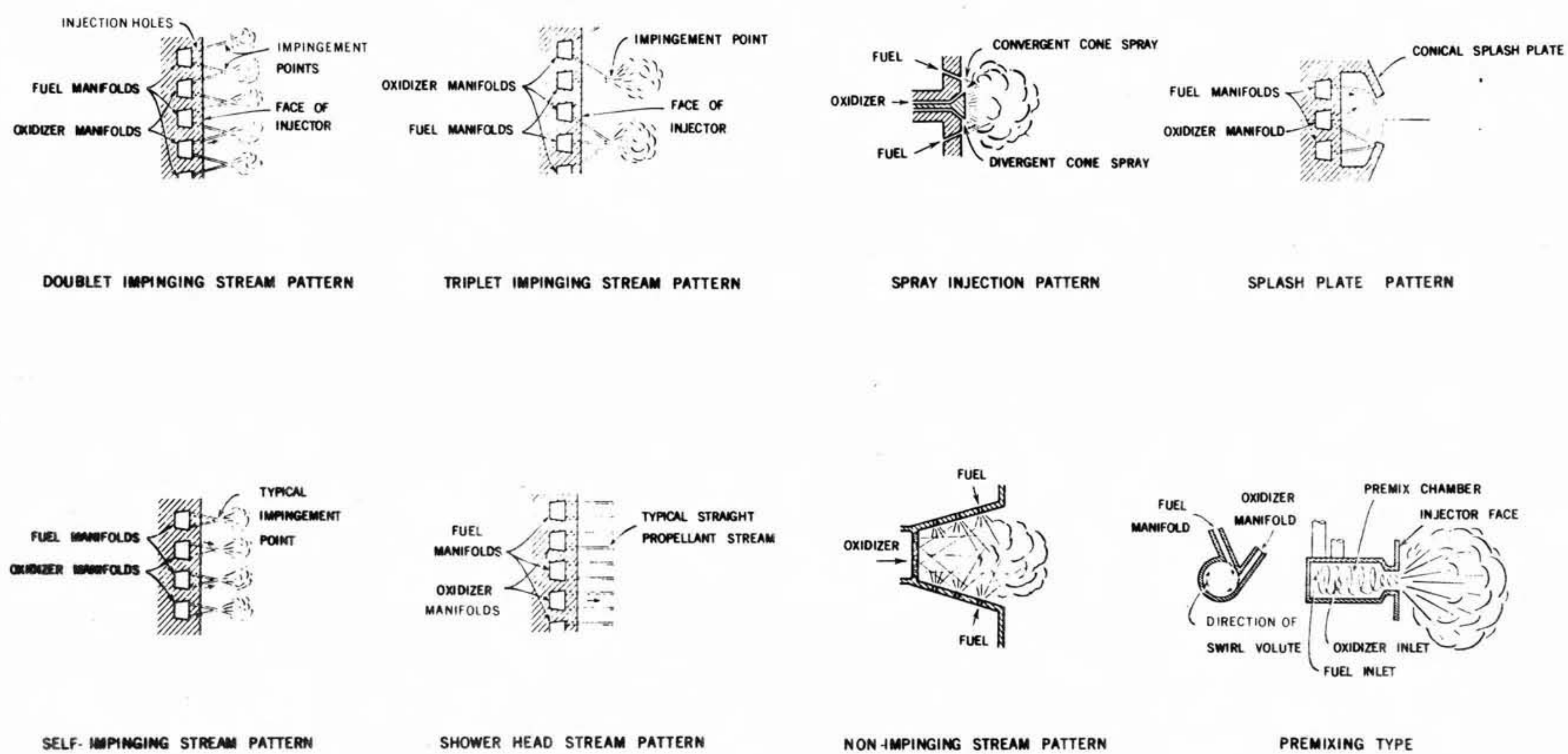


Figure 11. Schematic Diagrams of Several Injector Types

Sheet or spray-type injectors give cylindrical, conical, or other types of spray sheets of liquid rocket propellants; these sprays generally intersect and thereby promote mixing. A special type is the ring slot injector.

In premixing injectors the propellants are mixed before they are introduced into the combustion chamber. The time delay of the reaction dictates the dimensions and the flow rate of the pre-mixing chamber; generally the control of explosions of the premixed propellants inside the injector is critical.

Figure 12 shows an injector used on a current engine system.

IGNITION

The starting of a thrust chamber has to be controlled so that a timely and even ignition of propellants is achieved and the flow and thrust are built up smoothly and quickly to their rated value.

Non-spontaneously ignitable propellants need to be activated by absorbing energy prior to combustion initiation. This energy is supplied by the ignition system. The igniter has to be located near the injector in such a manner that a satisfactory mixture is present at the time of igniter activation, yet it should not hinder or obstruct the steady-state combustion process.

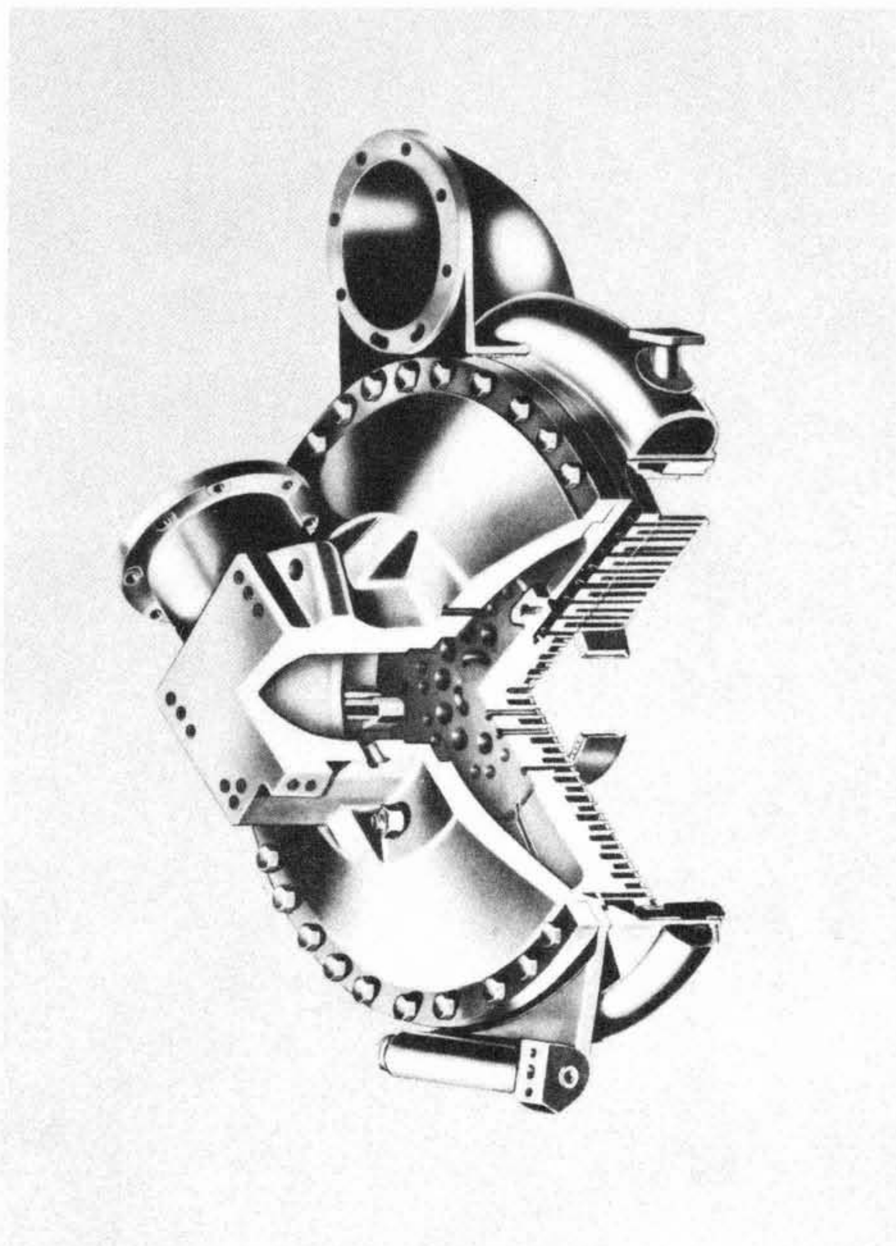


Figure 12. Injector

Spark plug ignition has been used successfully on liquid oxygen - RP-1 and on nitromethane-oxygen thrust chambers. The spark plug is usually located in a region where initial fuel and oxidizer vapors will form an ignitable mixture. This method seems to be particularly applicable to small rocket engines which must be restarted in flight. The spark plug is often built into the injector.

Pyrotechnic ignition uses a solid propellant squib or grain of a few seconds burning duration. The solid propellant charge is electrically ignited by a hot wire or other means and burns with a hot flame within the combustion chamber. Almost all solid propellant rockets and many liquid rockets are ignited in this fashion. The igniter container may be designed to fit directly onto the injector, or the chamber, or may be held in the chamber from outside through the nozzle. This ignition method can only be used once; thereafter the charge has to be replaced.

In precombustion chamber ignition a small chamber is built next to the main combustion chamber and convected through an orifice. A small amount of fuel and oxidizer is injected into the precombustion chamber and ignited by a spark plug, catalyst, or other means. The burning mixture enters the main combustion chamber in a torch-like fashion and ignites the larger main propellant flow injected into the main chamber. This ignition procedure permits repeated starting of variable thrust engines and has proved successful with liquid oxygen-RP-1 and oxygen-alcohol thrust chambers.

Auxiliary fluid ignition is a method whereby some liquid or gas, in addition to the regular fuel and oxidizer, is injected into the combustion chamber for very short periods during the starting operations. This fluid is hypergolic, which means it produces spontaneous combustion with either the fuel or the oxidizer. The combustion of nitric acid and some organic fuels can, for instance, be initiated by the introduction of a small quantity of aniline at the beginning of rocket operation.

PROPELLANT FEED SYSTEM

The selection of a particular feed system and its components is governed primarily by the application of the rocket, its size, propellant, thrust, flight program duration, past experience, and by general requirements of simplicity of design, ease of manufacture, reliability of operation and minimum weight. A classification of several of the more important types of feed systems is shown in Fig. 13.

GAS-PRESSURE FEED SYSTEM

These simple systems expel liquid propellants from high-pressure tanks by displacing them with a high-pressure gas. The first part of gas leaving the high-pressure gas storage tank is at a slightly below ambient temperature. The gas remaining in the tank undergoes essentially an isentropic expansion, causing the temperature of the gas to decrease steadily; the last portions of the pressurizing gas leaving the tank will be very much colder than the ambient temperature and will readily absorb heat from the piping and the tank walls. The Joule-Thomson effect will cause a further small temperature change. In an actual propulsion system installation, the pressurized gas is required to perform other functions such as the operation of valves and controls, according to the particular system requirements. Figure 14 shows a schematic of a typical space engine system.

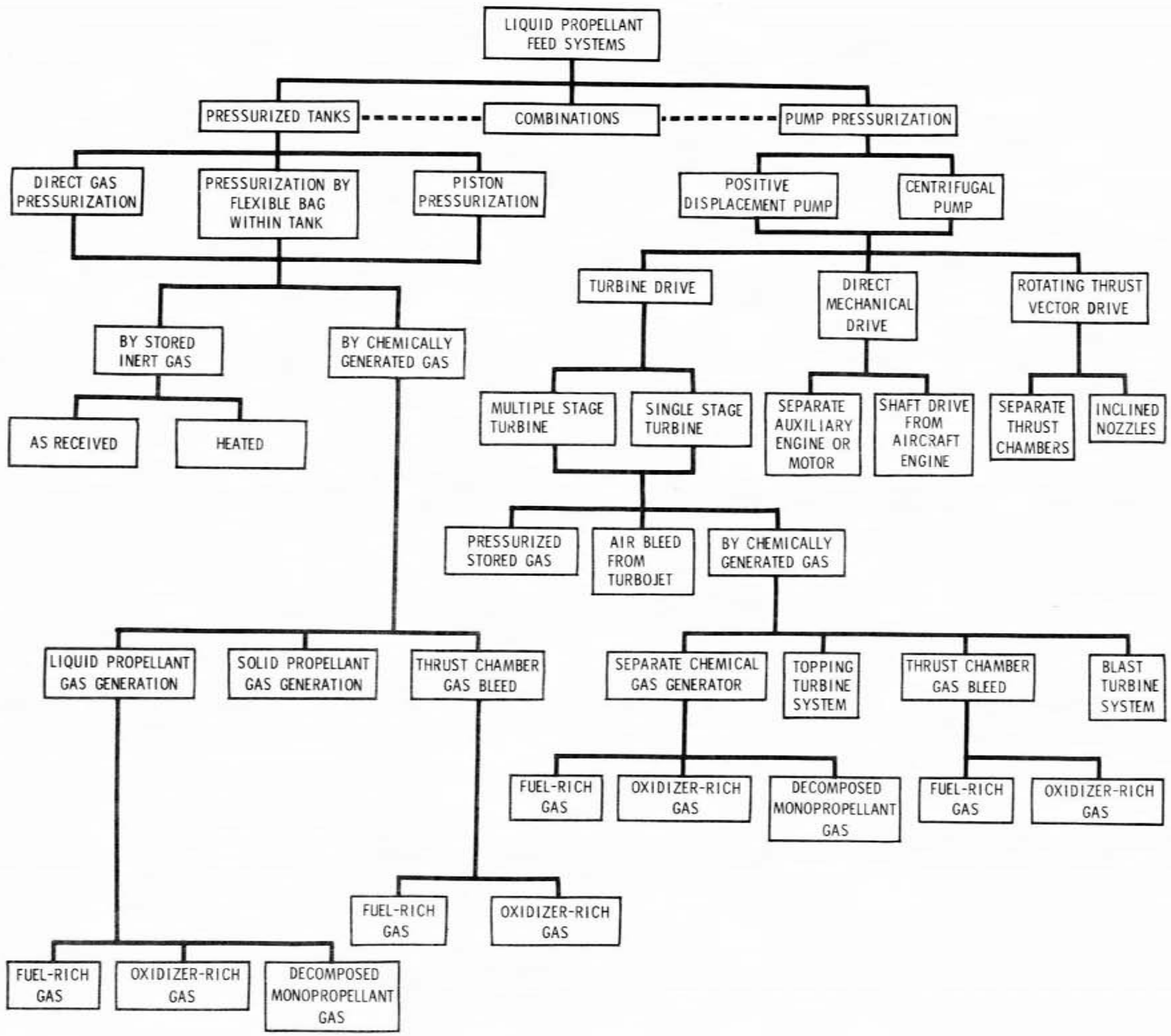


Figure 13. Classification of Liquid Propellant Feed Systems

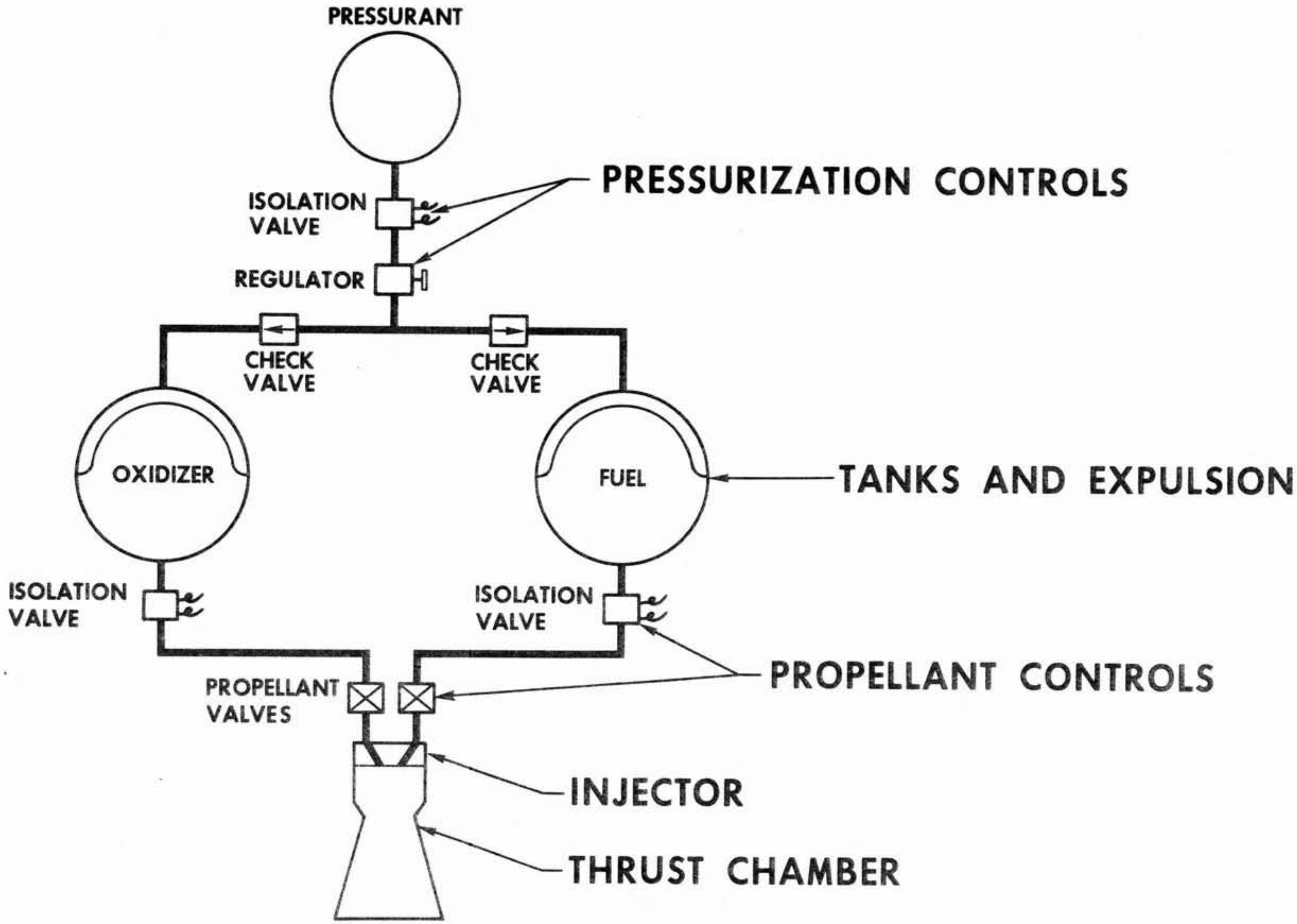


Figure 14. Typical Pressure-Fed System

Many different applications will require pressurized systems and many will have a thrust several magnitudes above that of a space engine.

A pressure-fed system is inherently rugged because the main propellant tanks must be rated at a pressure greater than chamber pressure; this same condition restrains optimum chamber pressure to relatively low levels. Extreme simplicity, high reliability, and minimum weight are easily designed into a pressure-fed engine system.

The pressure-fed engine consists of a thrust chamber assembly, pressurization system, propellant feed and control system, and the subsystems which may be required for the functions of thrust vector control, roll control, velocity trim, and/or propellant utilization, and main propellant tanks. A typical pressure-fed system will have the following features:

1. Propellants are completely isolated in main tanks, and in the pressurization system tanks by isolation valves.
2. Pressurization is by pressurant tank.
3. Starting and shut-off functions are each controlled by a single electrical signal to the valves. Start and cut-off sequencing is automatic and positive, once these valves are signalled.
4. Normally open main valves, with integral burst diaphragms, provide positive starting upon pressurization of main propellant tanks.

TURBOPUMP FEED SYSTEM

The turbopump rocket feed system pressurizes the propellants by means of pumps, which, in turn, are driven by turbines. The turbines derive their power from the expansion of hot gases. A separate gas generator ordinarily produces these gases in the required quantities and at the desired turbine inlet temperatures by means of a chemical reaction in the combustion chamber. A turbopump feed system is shown schematically in Figure 15.

The pump-fed engine offers high performance and low engine weight. Low-pressure propellant tanks augment the performance of a pump-fed engine, provided ruggedness is not a missile requirement. Continuing development of pump-fed engines has resulted in gross reduction of complexity, and increased reliability and performance.

Turbopump-fed engines are characterized by high chamber pressure operation, a performance advantage afforded by the use of high-pressure pumps.

The pump-fed engine consists of a thrust chamber assembly, turbopump assembly, propellant feed and control system, turbine drive system (either tap-off or a separate gas generator assembly) and those controls that may be required for thrust vector orientation, roll torque, velocity trim, and propellant utilization.

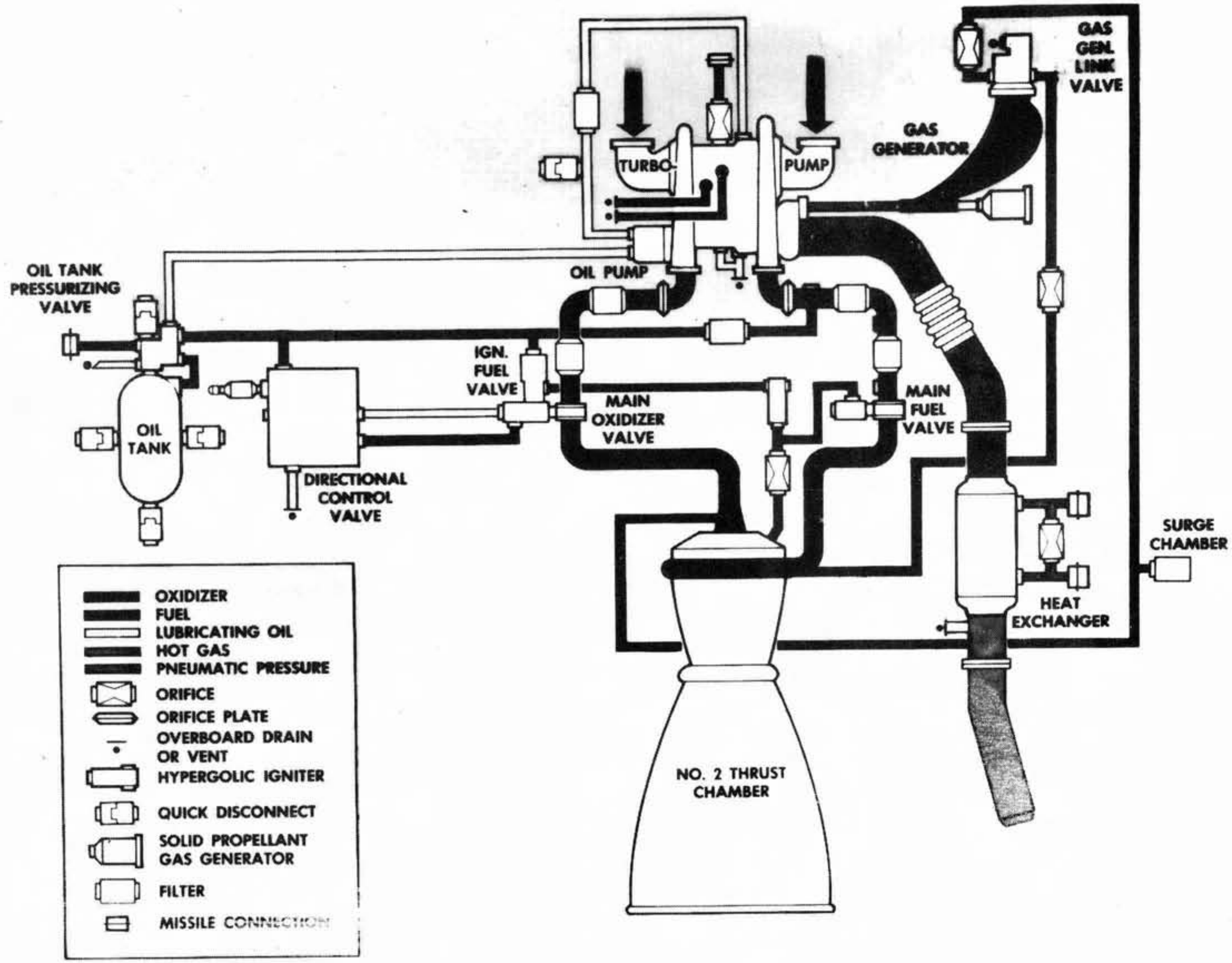


Figure 15. Schematic Diagram of a Liquid Propellant Rocket Engine With Turbopump Feed System

TURBOPUMPS

Function - The function of the turbopump in an engine installation is to receive propellants from the tanks and deliver them to the thrust chamber at design pressure levels and flow rates so that the engine can develop design thrust at the required chamber pressure. The turbopump is a rotating machinery assembly including a pump (or pumps) for increasing the pressure level of the propellant(s). The power to drive the pump(s) in this assembly is supplied by a turbine which utilizes working fluids supplied by the engine system gas generator. Turbopump design configurations can vary depending on the engine combustion process, the installation requirements, and the propellants being pumped.

Fluids and Fluid Properties - The major factor influencing the type of turbopump design chosen for any application is the density of the propellants to be pumped. Variations in oxidizer and fuel density requires the individual pumps to be operated at speeds capable of obtaining respective pump head and flow volumes. For turbopumps pumping propellant combinations that have similar densities, both pumps can be run at the same speed. In cases where a great variation in propellant density exists between the oxidizer and fuel, as in the liquid oxygen/liquid hydrogen combination, each pump is driven at its best design speed to most efficiently meet individual head requirements.

Operating Range - Figure 16 is a plot showing the range of operation for typical propellant pumps in terms of pump head and flow. This curve demonstrates how the head requirements for the less dense propellant, liquid hydrogen, are much greater than those required by either liquid oxygen or RP-1. The plot shown in Fig. 17 is the operation envelope of current turbopump turbine designs based on power and speed requirements. Turbine working fluid mass flowrate depends on the properties of these fluids, the power development requirements, amount of energy from these fluids made available to the turbine to convert into work, and the turbine design and operating parameters. As power requirements for a specific design operating point increase, the ratio of turbine mass flowrate to engine flow increases. If chamber pressure is increased for a fixed thrust condition, the turbine power requirements to develop the needed pump heads become greater.

Turbopump Configurations - Turbopumps can be designed in a number of different configurations and arrangements. The final selection depends on the desired speed ratio between pumps, the arrangement of components, and the energy source of the turbine working fluid. There are three basic turbopump designs: (1) geared turbopump, (2) single-shaft turbopump, and (3) dual-shaft turbopump. Figure 18 shows these various configurations.

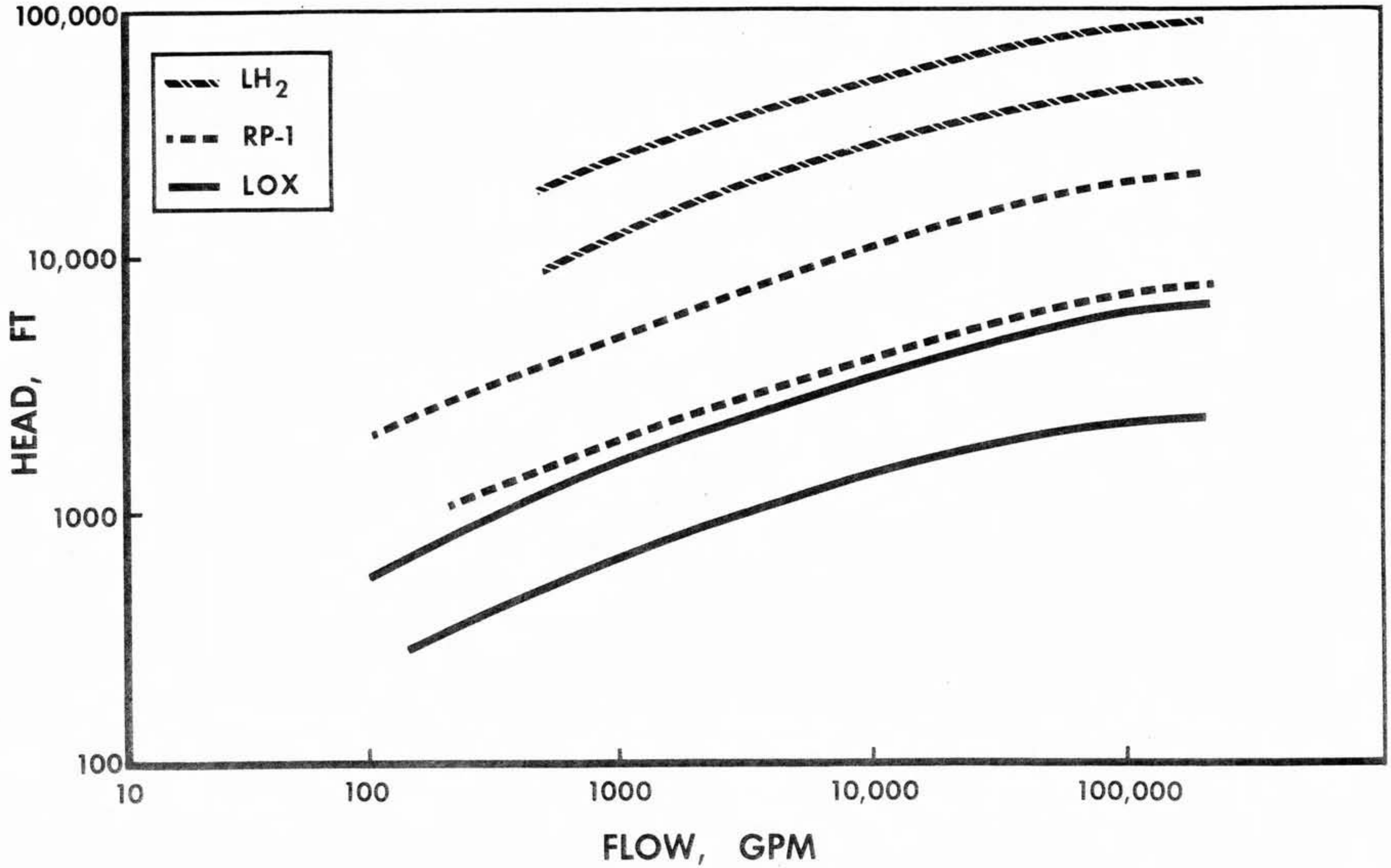


Figure 16. Range of Operation for a Typical Propellant Pump

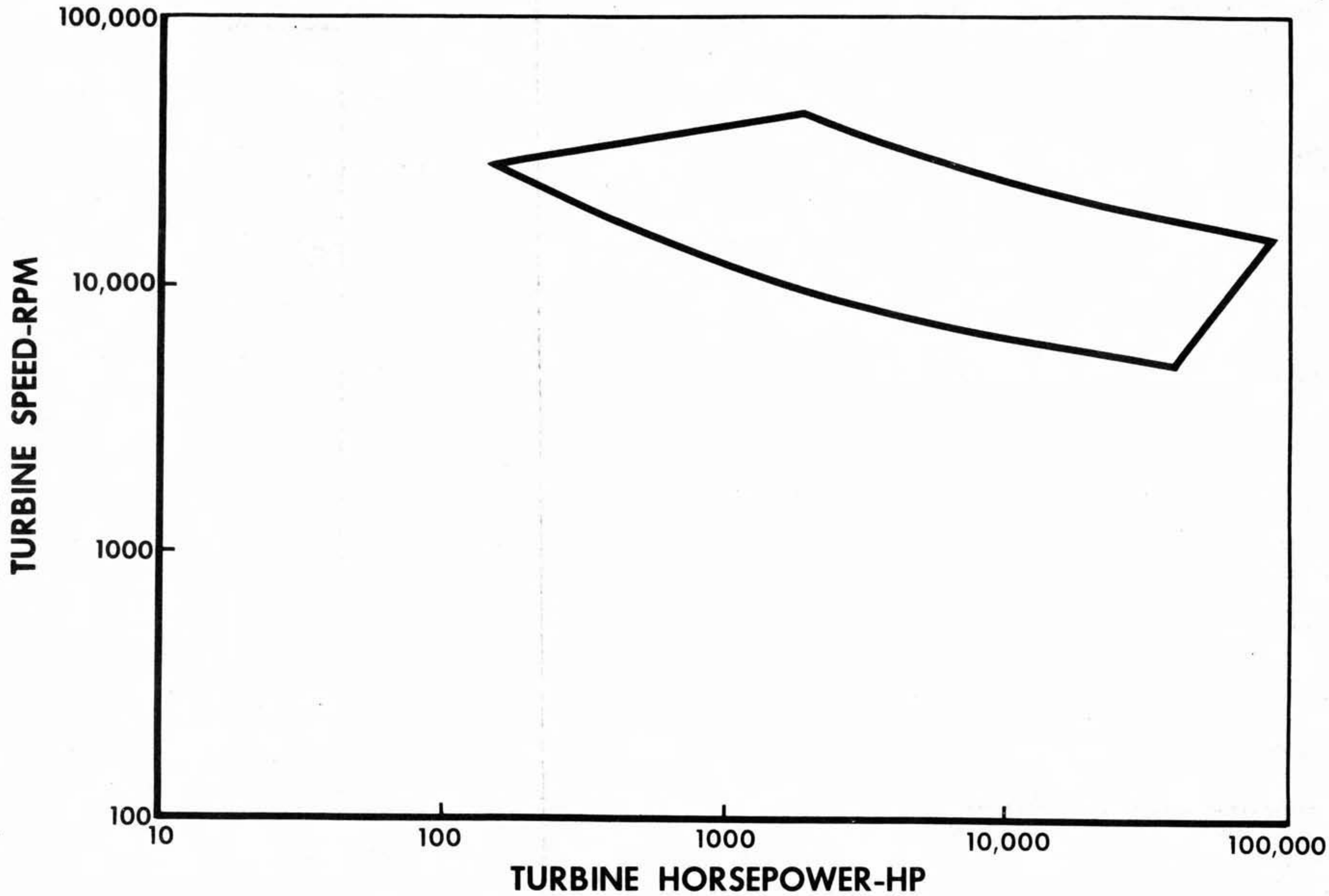


Figure 17. Operation Envelope of Current Turbines

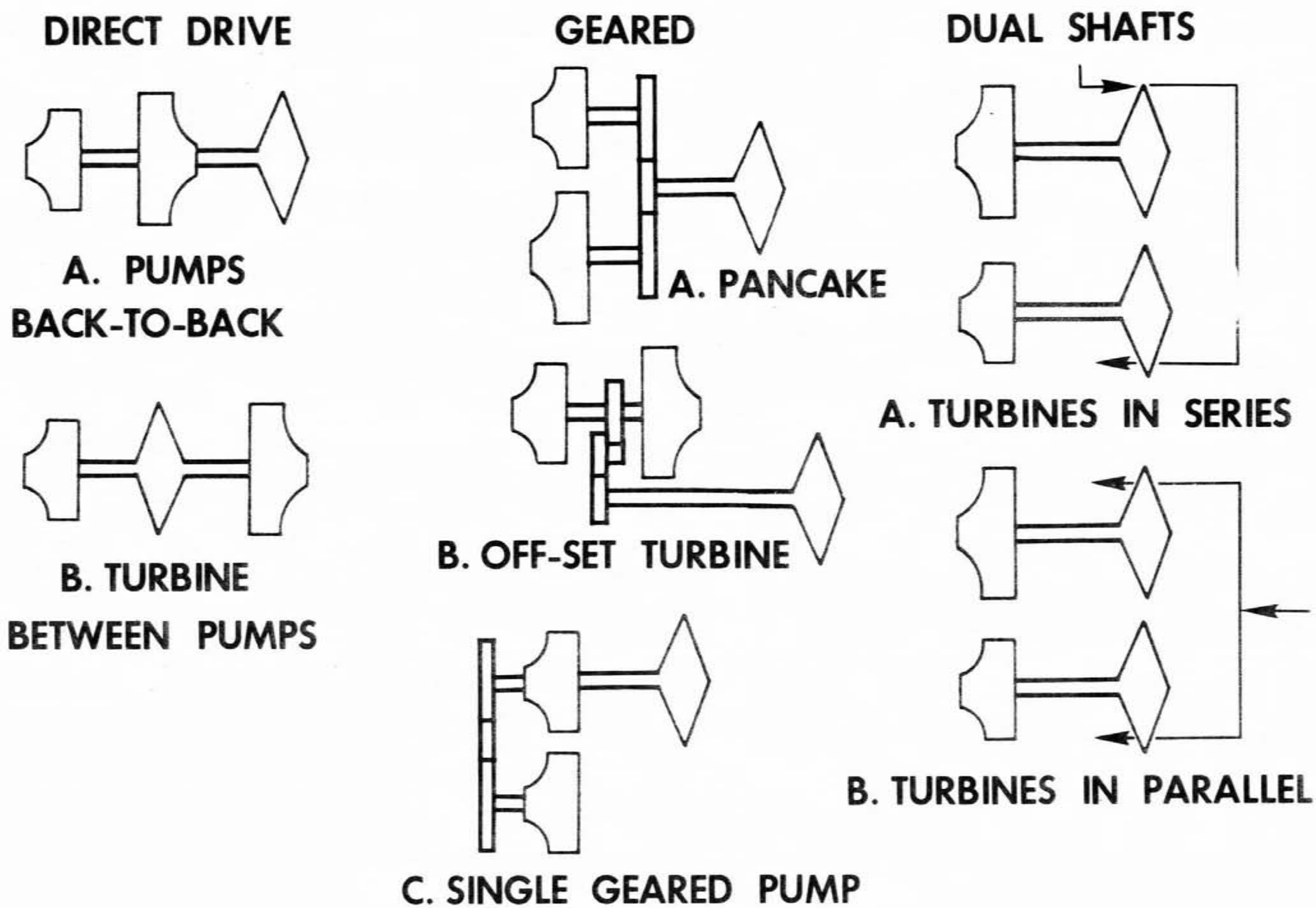


Figure 18. Turbopump Configurations

Geared Turbopump - The geared turbopump design utilizes a gear box with which to drive the fuel and oxidizer pump at different speeds with a single turbine drive assembly. Figure 19 is a cut-away photograph of a LOX RP-1 gear turbopump configuration currently in service in a 150,000-lb thrust booster engine.

Single-Shaft Turbopump - The turbopump shown in Figure 20 is a single-shaft configuration where both the oxidizer and fuel pumps are driven on one shaft by a single turbine. This single-shaft turbopump is being used in the LOX/RP-1 1,500,000-lb-thrust engine.

Dual-Shaft Turbopump - The dual-shaft turbopump utilizes separate shafts to drive the oxidizer and fuel pumps at the best speed to meet the head and flow requirements of the propellants being pumped. Each pump is driven by its own turbine; pump speeds, heads, and flows can be adjusted independently with this pump. Dual-shaft configurations are used for pumping propellant combinations that have large differences in densities; one such propellant combination is LOX/LH_2. The two pumps shown in Figure 21 are dual-shaft, LOX and LH_2 turbopumps respectively, for a 200,000-pound-thrust engine application. In dual-shaft installation, the turbines can be installed either in series or in parallel to one another. In Figure 21 the oxygen pump is a one-stage centrifugal type; the hydrogen pump is a multi-stage axial flow.

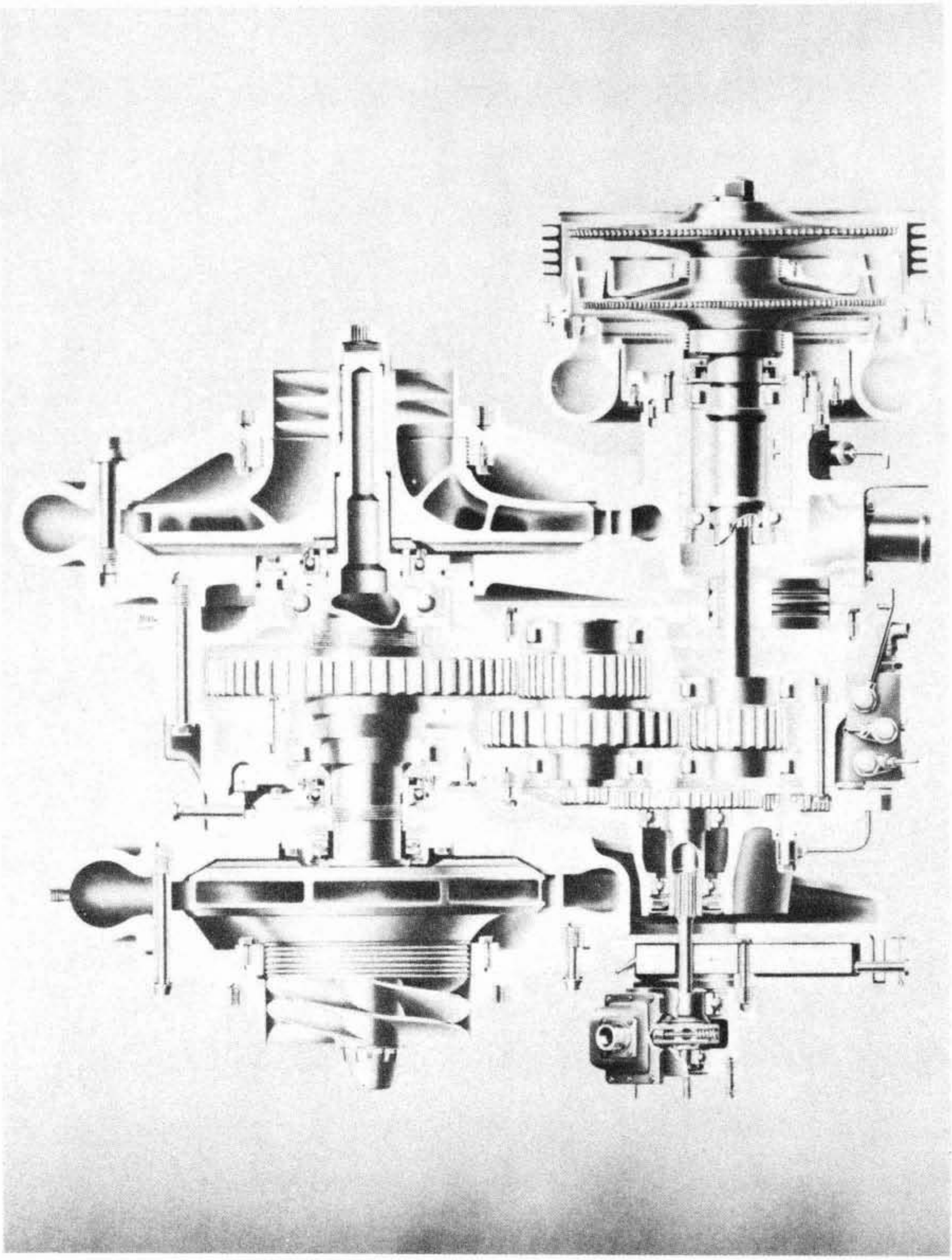


Figure 19. Geared Turbopump

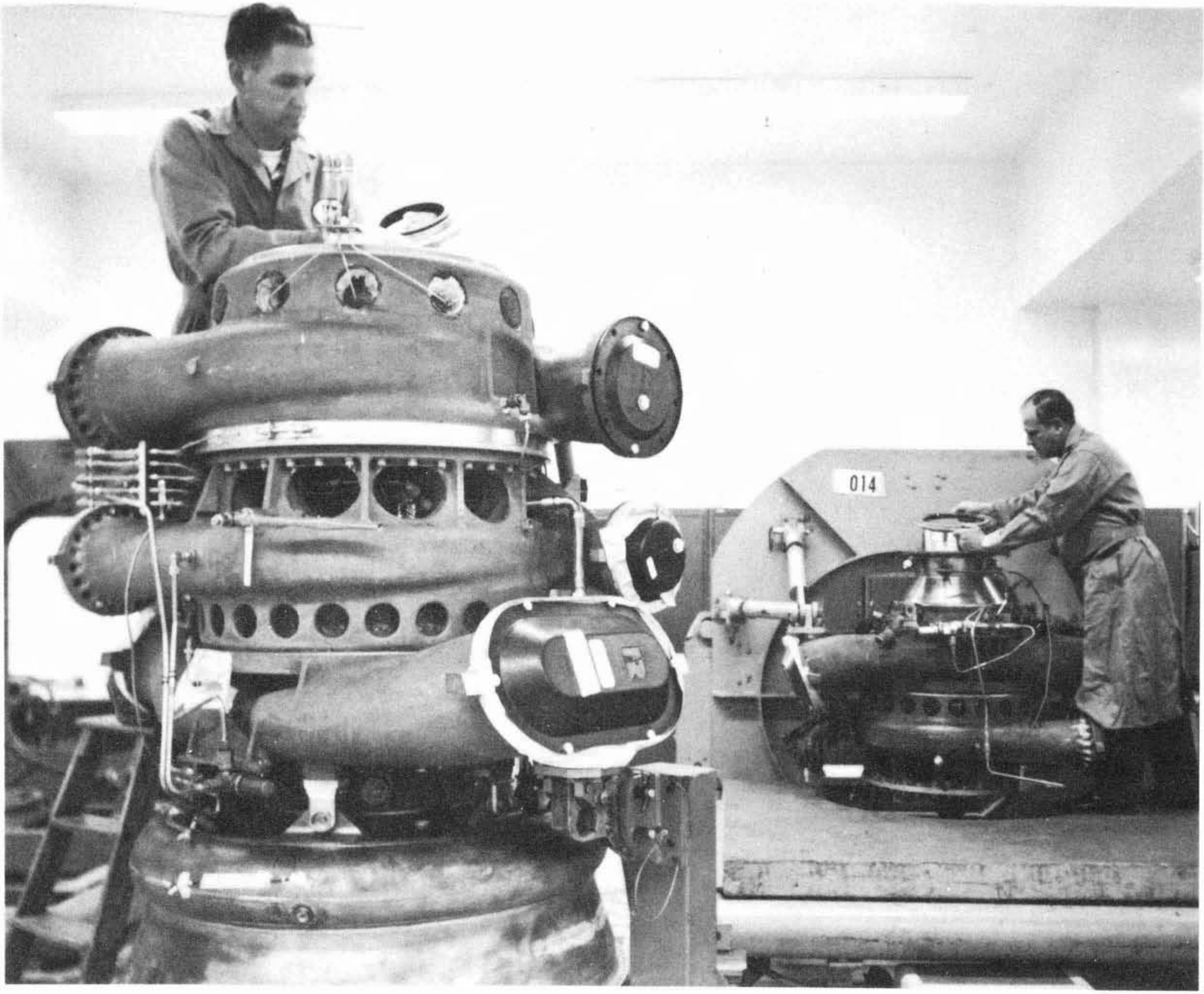


Figure 20. Single Shaft Turbopump

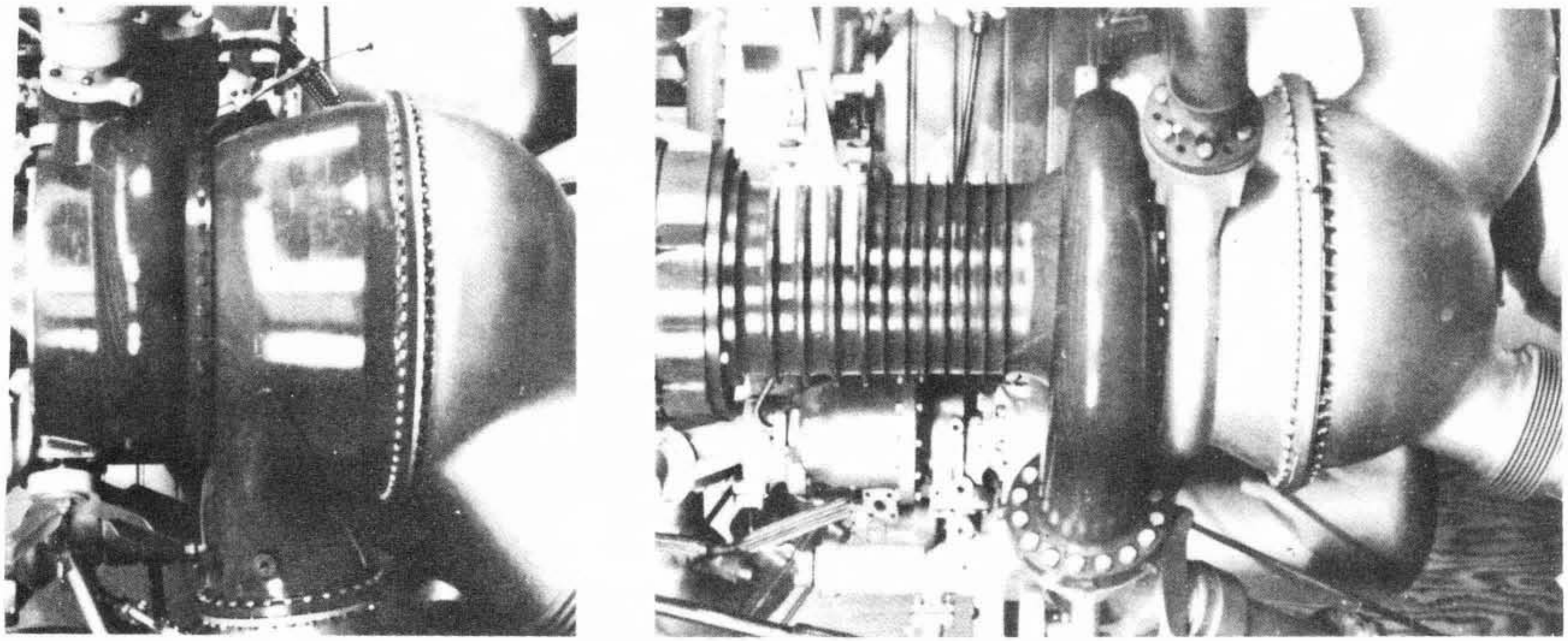


Figure 21. Dual-Shaft Turbopump

GAS GENERATORS

The power necessary to drive the turbine is usually obtained from the gas generator, which generates gases by a chemical reaction of propellants similar to those in thrust chambers. The generated gases have to be relatively cool compared to thrust chamber reaction gases, because excessive temperatures will cause a failure of turbine buckets, turbine nozzles, and sometimes the turbine wheel. The propellants used for this purpose are intentionally mixed in such proportions that the resultant gas temperature will be relatively low.

There are many methods for generating gases. The more common types (Figure 22) are: (1) gas-pressurized generator, having its own propellant supply; (2) feed-system pressurized generator using the same propellant as a rocket from the same feed system, but with a separate combustion chamber; (3) gas bleeding (tap-off) directly from the main combustion chamber; (4) solid propellant charges burning at a slow rate and giving off gases; and (5) vaporized propellants.

Figure 23 is a feed-system pressurized gas generator using the same propellants as a rocket from the same feed system, but with a separate combustion chamber; it is used on a current 200,000-lb-thrust LOX/RP-1 engine system. Figure 24 is a schematic showing how combustion gases are tapped off from the main combustion chamber to drive the turbine.

GAS GENERATOR TYPES

GAS-PRESSURIZED GENERATOR

FEED-SYSTEM PRESSURIZED GENERATOR

TAP-OFF

SOLID PROPELLANT

VAPORIZED PROPELLANTS

Figure 22. Gas Generator Types

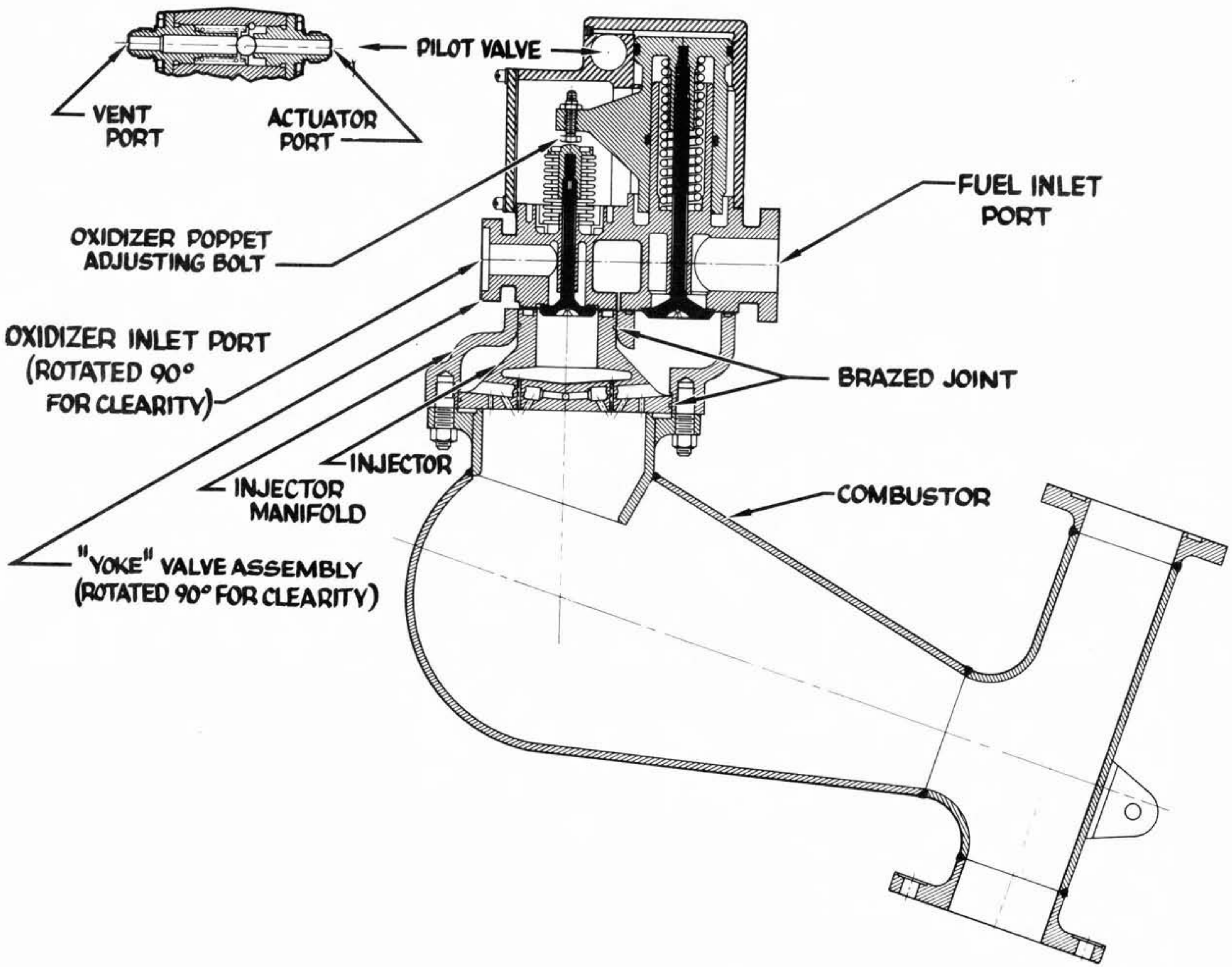


Figure 23. H-1 Gas Generator Assembly

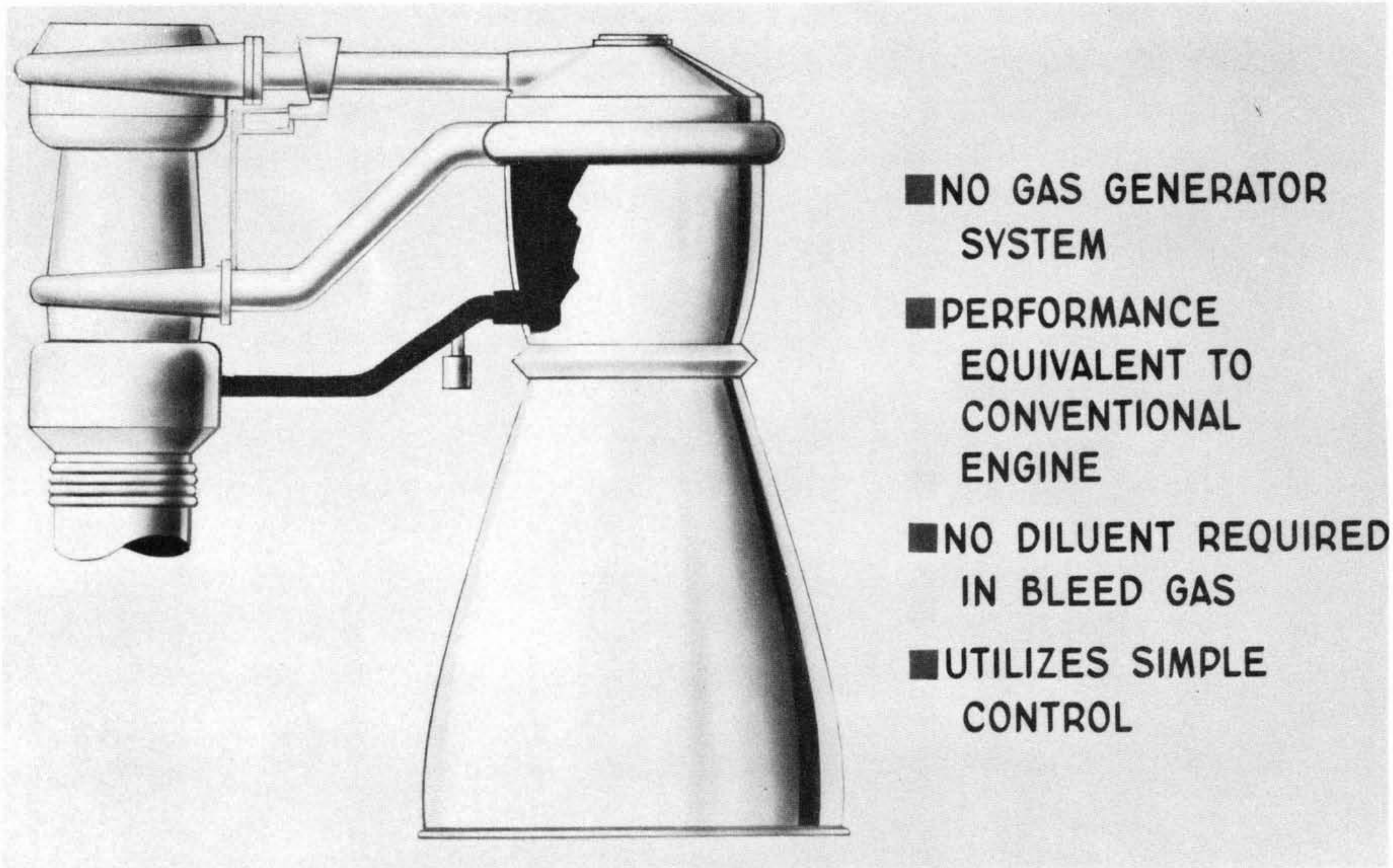


Figure 24. Tap-Off Engine

PRESSURIZATION SYSTEMS

The application of the particular liquid propulsion system will often dictate the type of pressurization system required for that particular mission. Figure 25 shows various pressurization system advancements made throughout the past several years. For propellants which react with this pressurization gas, an inert gas (for example nitrogen to pressurize liquid oxygen or helium to pressurize liquid hydrogen) must be used unless provisions are made to separate the chemically active gas from the propellant. This separation may be accomplished by means of a collapsible, flexible bag within the propellant tank or by means of a piston cylinder arrangement as shown in Figure 26. The required weight of pressurizing gas, and thus the size and weight of the gas storage container can be significantly reduced by a "tail-off" pressure decay. Here the pressurizing gas supply is shut off while a major portion of the liquid propellants are still in the tanks. These remaining propellants are then expelled by the adiabatic expansion of the gas already in the propellant tanks. The tank pressure and chamber pressure decrease or progressively decay during this adiabatic expansion period.

Figure 25 depicts several types of pressurization systems. The heated helium system is used to pressurize both the oxidizer and the fuel. In the dual gas generator system, one gas generator is used to pressurize the fuel tank. In the case of the single gas generator the exhaust

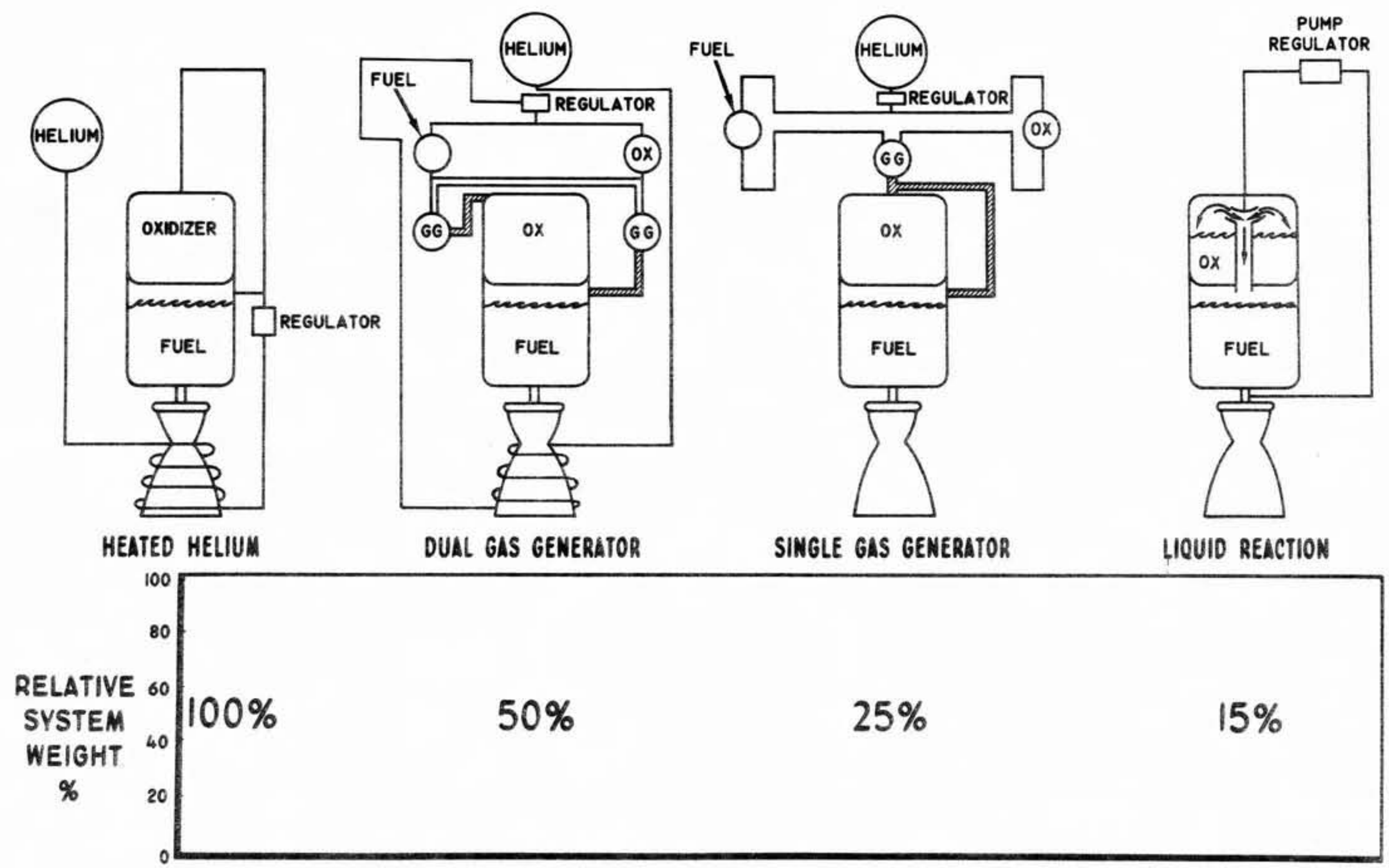


Figure 25. Pressurization System Advancements

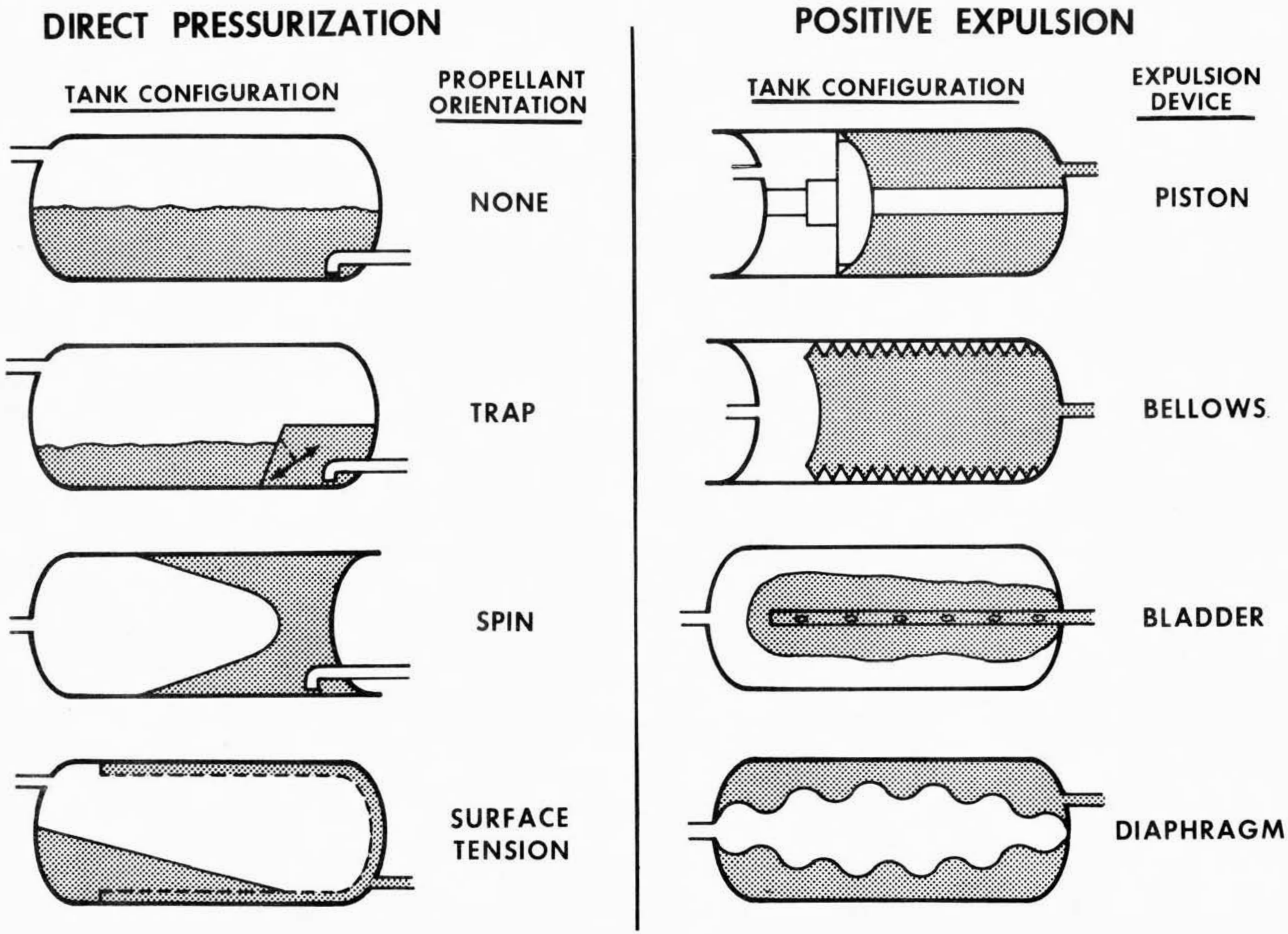


Figure 26. Expulsion Techniques

products of the gas generator are used to pressurize both the oxidizer and the fuel tanks. In the liquid reaction pressurization system a small amount of fuel is injected into the oxidizer tank, allowing a slight chemical reaction to take place which in turn pressurizes the oxidizer and the fuel tank.

Figure 26 shows various expulsion techniques. In the direct pressurization expulsion technique, propellant orientations shown include no propellant orientation required, trap, spin, and surface tension propellant orientation. The positive expulsion devices include pistons, bellows, bladders, and diaphragms.

CONTROLS

All liquid propellant rockets have controls to accomplish some or all of the following tasks (Fig. 27):

1. Start rocket operation
2. Shut down rocket operation
3. Restart
4. Maintain programmed operation (predetermined constant or varied thrust, preset propellant mixture ratio or flow) by calibration of feed system or by automatic controls

CONTROL FUNCTIONS

START ROCKET OPERATION

SHUT DOWN ROCKET OPERATION

RESTART

MAINTAIN PROGRAMMED OPERATION

THROTTLING

THRUST VECTOR CONTROL

Figure 27. Control Functions

5. Make emergency shutdown when safety device senses a malfunction or a critical condition of the vehicle or the engine

6. Fill with propellants

7. Drain excess propellant after operation

8. Pre-flight check-out of proper functioning of critical components without actual operation

9. Throttling

10. Thrust vector control

The complexity of these control elements and the complexity of the engine systems depend very much on the nature of the mission of the vehicle. Rockets which are used only once, and are filled with propellant at the factory, and which have to operate over a narrow range of environmental conditions generally tend to be simpler than rocket systems intended for repeated use, for applications where satisfactory operation must be demonstrated prior to use, and for manned vehicles. Because of the nature of the liquid propellants, most of the control functions are achieved by valves, regulators, pressure switches, and flow controls.

THROTTLING

The throttling of rocket propulsion systems represents an effective means for obtaining precise control of ballistic missile and space vehicle flight characteristics, and for enabling a single rocket engine to accommodate the propulsion requirements of each of several widely different maneuvers.

The basic advantage of throttling capability is that it permits a single rocket unit to satisfy mission objectives that might otherwise require a multi-unit system; the basic disadvantage is that providing such a capability means that the unit must be more complex than a constant-thrust engine. Overall, the diverse propulsion requirements of various weapon and space vehicle systems clearly warrant the use of throttling so long as attended performance and reliability penalties can be kept within reasonable bounds. The conception and development of efficient, reliable throttling techniques are therefore a primary objective in the advancement of propulsion technology.

The main concern in the area of throttling is to accomplish a smooth, continuous transition from one thrust level to another accompanied by a predictable, if not constant, performance efficiency. This latter requirement necessitates precise mixture ratio control during throttling. A major problem area is that of maintaining high specific impulse efficiency at the lower propellant flow rates associated with the low-thrust-level end of the range.

In addition to continuous throttling, stepped or discontinuous throttling also has certain applications for space propulsion systems. One such application is the use of a single engine to efficiently perform a dual function, as in a combination of mid-course correction (low thrust) and orbit injection (high thrust) maneuvers, thus eliminating the need for two separate engines.

Throttling Techniques

The techniques explored for continuous thrust modulation of liquid bipropellant rocket engines have been many and varied. The simplest and perhaps the best technique for limited throttling ranges (up to about 5:1) is to use variable-orifice propellant valves. However, as the required throttling range is increased, the propellant injection velocities at the lower end of the range decrease to a point where smooth combustion efficiency and stability problems can be incurred. Therefore, throttling techniques must be employed which maintain a minimum allowable propellant injection velocity as the propellant flow rate, and hence thrust, is decreased. Two technical approaches are possible. First, the injector orifice areas can be decreased with increased propellant flow, and second, the flow can be decreased by the injection of an inert gas. Both techniques have been developed. The variable orifice area injector is the more complex mechanically, while the gas injection approach requires a small amount of stored gas, usually available from the system pressure tank. Both systems are theoretically capable of large throttling ranges as high as several hundred to one (the variable area injector has been demonstrated to 300:1), and both are currently being applied to spacecraft propulsion.

THRUST VECTOR CONTROL (Fig. 28)

Gimballed Engine

The gimballed engine method of thrust vector control is a proven concept, and has been used on many vehicle systems with no known engine failures attributable to it. This system involves swiveling of the thrust chamber and turbopump combination. The performance loss with this system is essentially zero. Comparing the weight and performance losses of various systems shows that the gimballed engine is one of the best methods.

Fixed Injector/Gimballed Chamber

This system consists of a fixed injector assembly and a fixed turbopump. Attitude control is provided by a movable chamber assembly which pivots about the injector. This arrangement eliminates the need for flexible propellant lines and reduces the mass to be swiveled. However, dynamic seals are required for the coolant flowing to and from the thrust chamber walls. The performance is identical to that of the gimballed engine, but the weight is less.

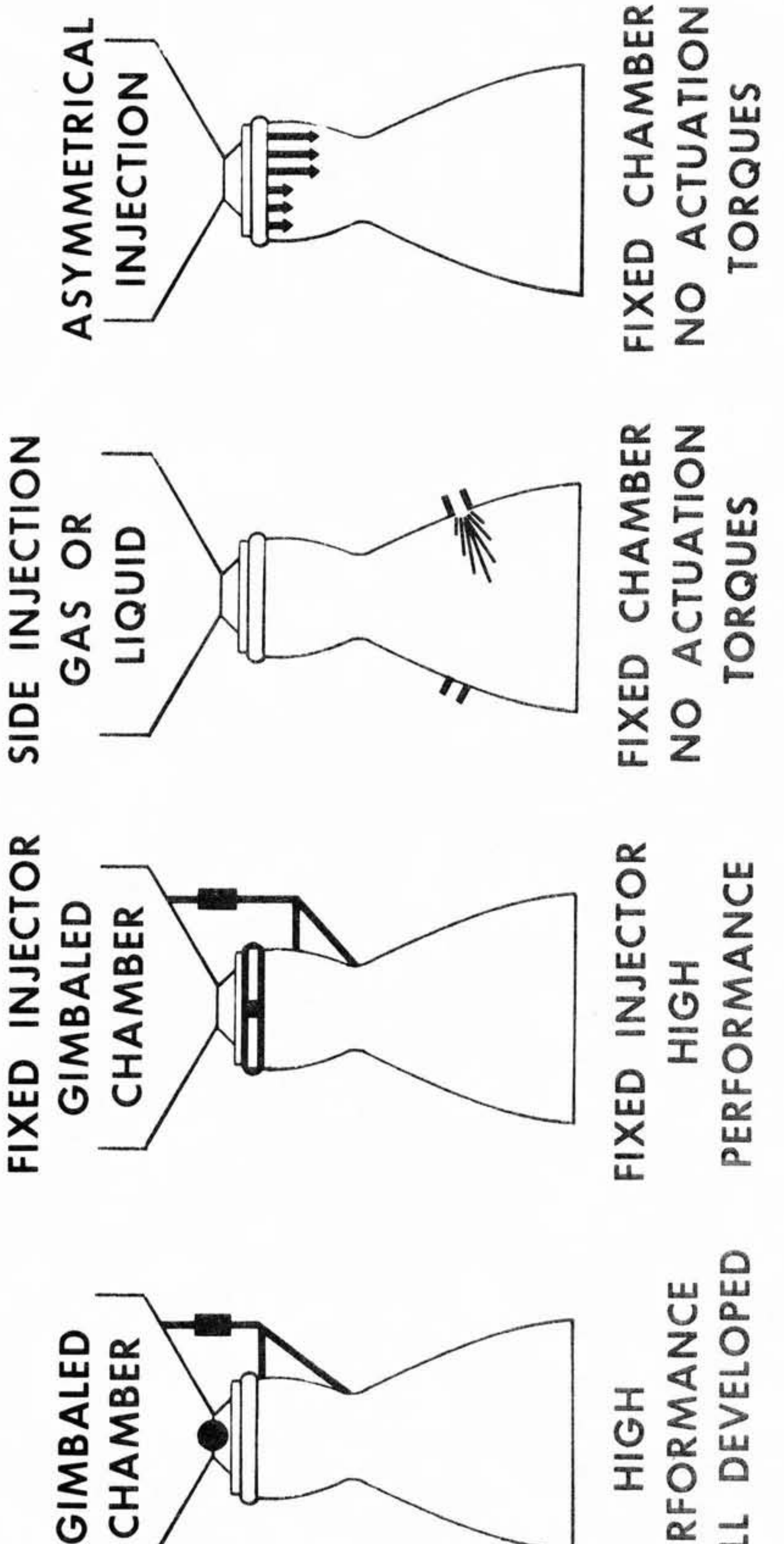


Figure 28. Thrust Vector Control Methods

Side Injection of Fluids

Three types of systems using side injection of fluids may be mentioned: side gas injection, side liquid injection, and assymetrical throat injection. The side gas injection system produces better performance in some cases than the side liquid system, and in many cases assymetrical throat injection will be ineffective. In the side gas injection system gases are injected into the divergent portion of the nozzle. The interaction of these gases with the main stream changes the direction of the thrust vector. This system, though eliminating all movement of the thrust chamber assembly, requires an increased flow of propellant to produce the required vector displacement. This increase is large at times, especially if the injected gas temperature is considerably lower than that of the mainstream gas.

Assymetrical Propellant Injection

The injector of a rocket combustion chamber could be throttled in sections to cause assymetrical flow through the combustion chamber and nozzle, thus generating side forces which could be used for thrust vector control. It can be found that the flow through one quadrant of an injector must be completely stopped and the flow in the opposite quadrant must be doubled to get a maximum effective gimbal angle of only 3 degrees. Therefore, assymetrical propellant injection is not particularly a practical method of vector control.

APPLICATIONS

SPACE ENGINE DESCRIPTION

A space engine is defined as a liquid-propellant, pressure-fed system in the lower-thrust range, designed primarily for operation in a space environment. The propellant tankage and feed subsystem are included in this engine system definition, although, in many instances, these components and sub-systems are provided by airframe contractors rather than by propulsion companies, and often comprise much of the vehicle airframe itself. Figure 29 shows a schematic representation of a typical system. Functionally, a pressurizing gas, usually gaseous nitrogen or helium, is released through a system of valves and pressure regulators into tanks containing liquid propellants, either cryogenic or storable, and usually hypergolic. The propellants, under pressure, are delivered through lines to the one or more thrust chamber assemblies; then they are metered, on demand, through propellant valves into the thrust chamber where combustion takes place. A monopropellant system is even simpler, as one leg of the propellant feed system is eliminated.

SPACE ENGINE APPLICATION

Considering the space engine category, in a wide sense, as these propulsion systems have primary functions, other than freeing a vehicle from the pull of the earth's gravity, thrusts to something in excess of 20,000 lb

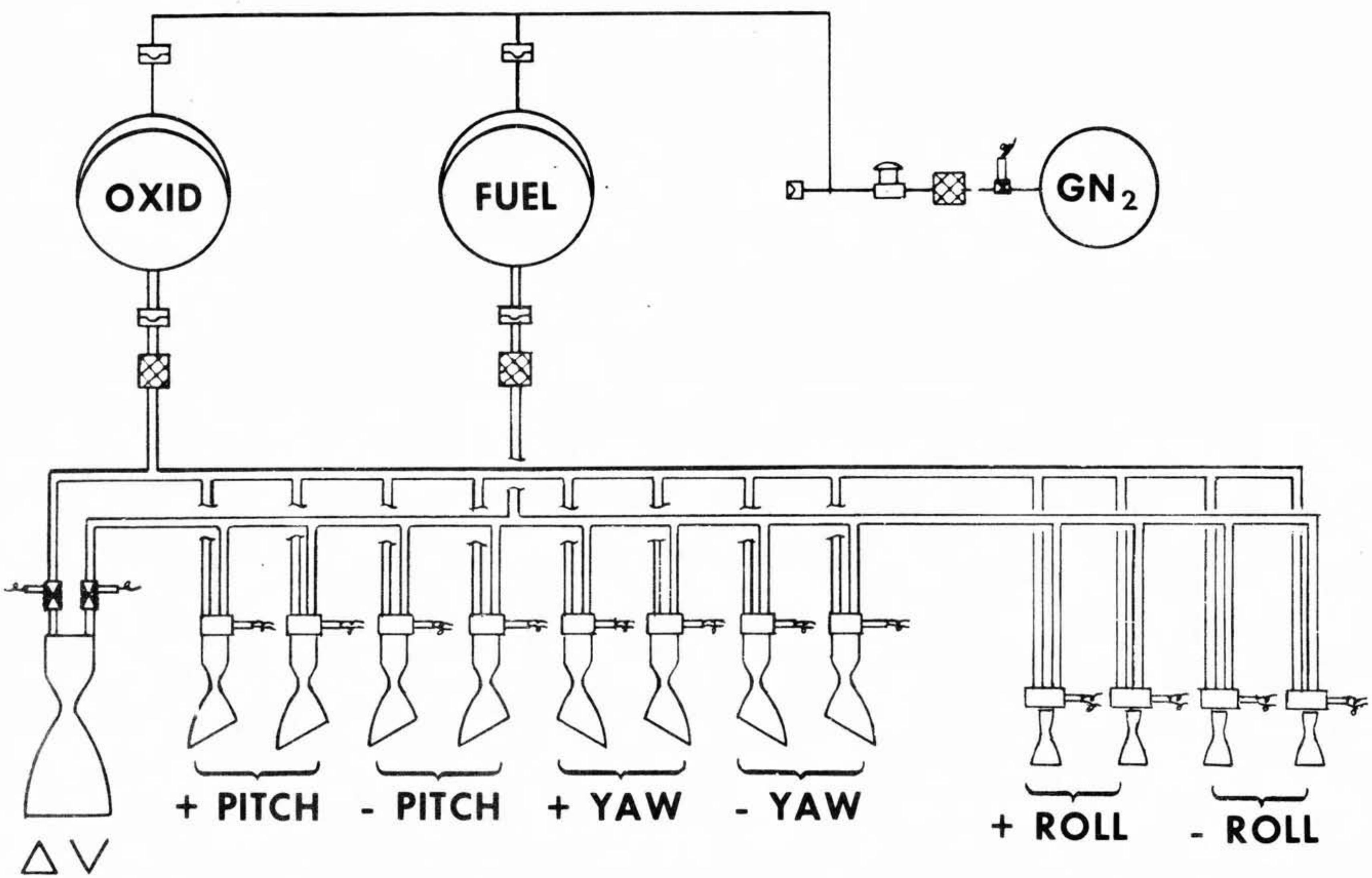


Figure 29. Typical Space Engine System Schematic

must be included to encompass the Apollo Service module primary engine. The next steps down the thrust ladder are the Agena and RL-10 propulsion systems at the 15,000-lb-thrust level; the 10,500-lb thrust level LEM descent engines; the Trnastage, Able-Star, and Delta engines, all at 7,000 to 8,000 lb thrust; and the LEM Ascent engine at 3,400 lb thrust. The RL-10 and Agena systems are in the pump-fed category, and, along with the Able-Star and Delta systems, fall into a "transition" thrust class between the booster stages and the payload stages incorporating the smaller space engines. Thrust classes under 4,000 lb and particularly under 1,000 lb, include a host of relatively low-thrust, liquid-propellant, pressure-fed propulsion systems.

The Gemini propulsion systems, including the orbit attitude and the maneuver systems, and re-entry control system, are shown in Figure 30. The orbit attitude and maneuver system is a complete propulsion system including eight 25-lb ablative thrust chambers, six 100-lb ablative thrust chambers, two 85-lb ablative thrust chambers, and a Teflon bladder positive expulsion system. The propellants for this system are NTO/MMH. In addition to steady state operation this system has a pulsing requirement of 0.25 lb per second minimum impulse bit for the 25 lb thrust chambers and 25 lb per second minimum impulse bit for the 85 and 100 lb thrust chambers. The Gemini re-entry control system consists of two complete propulsion systems consisting of eight 25 lb ablative thrust chambers per system, and a Teflon bladder positive expulsion system; NTO/MMH propellants are used. This system also has a pulsing requirement of 0.25 lb second minimum impulse bit.

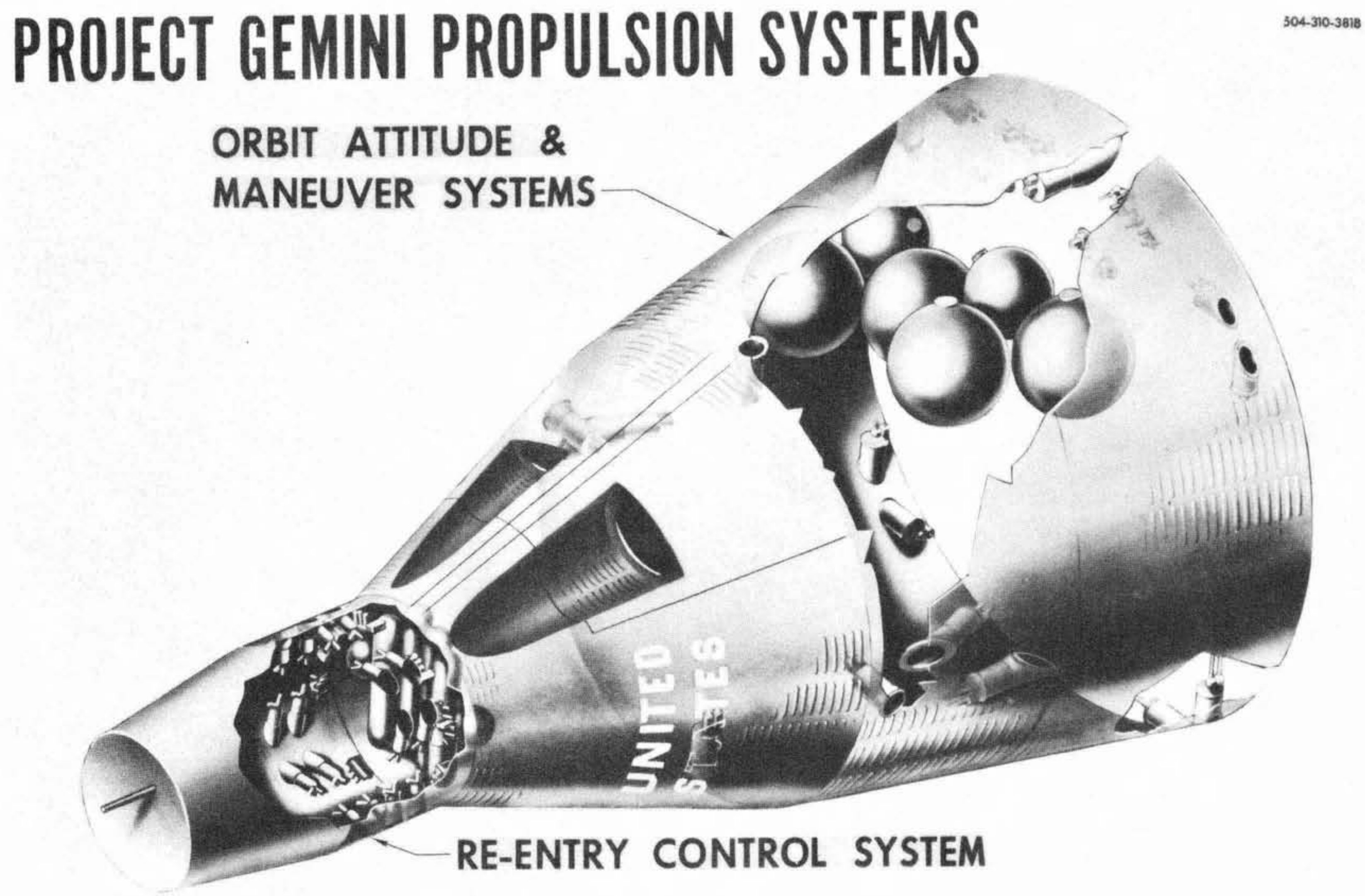


Figure 30. Project Gemini Propulsion Systems

PRE-PACKAGED LIQUID MISSILE SYSTEMS

A pre-packaged liquid propulsion system is normally equipped with a liquid propellant, pressure-fed propulsion system comprising a tankage assembly, pressurization system, thrust chamber assembly, and control system. Normally the propellants are hypergolic and are loaded into the system at the factory prior to delivery of the missile system. These missiles are designed for a simple operation, long-term storage under a wide temperature range, and require a minimum amount of maintenance. Figure 31 shows the Lance missile with a pre-packaged liquid propulsion system being launched at the White Sands Missile Range.

SPACE LAUNCH VEHICLE SYSTEMS

Saturn I Launch Vehicle - The Saturn vehicle has a booster thrust of 1,500,000 lb. The first stage of the vehicle is powered by eight Rocketdyne H-1 engines as shown on Figure 32. This vehicle will stand about seventeen stories high and will put three men into a 300 nautical mile orbit where they will be able to stay for a period of up to two weeks. Nine Saturn I vehicles have been launched successfully (as of this date), using a total of 72 engines in the first stage with 100 percent reliability. The later flights of the current series of vehicles also had an S-IV upper stage each using six engines of 15,000 lb thrust each, operating on oxygen-hydrogen propellant systems.

Figure 31. Prepackaged Storable Propulsion System

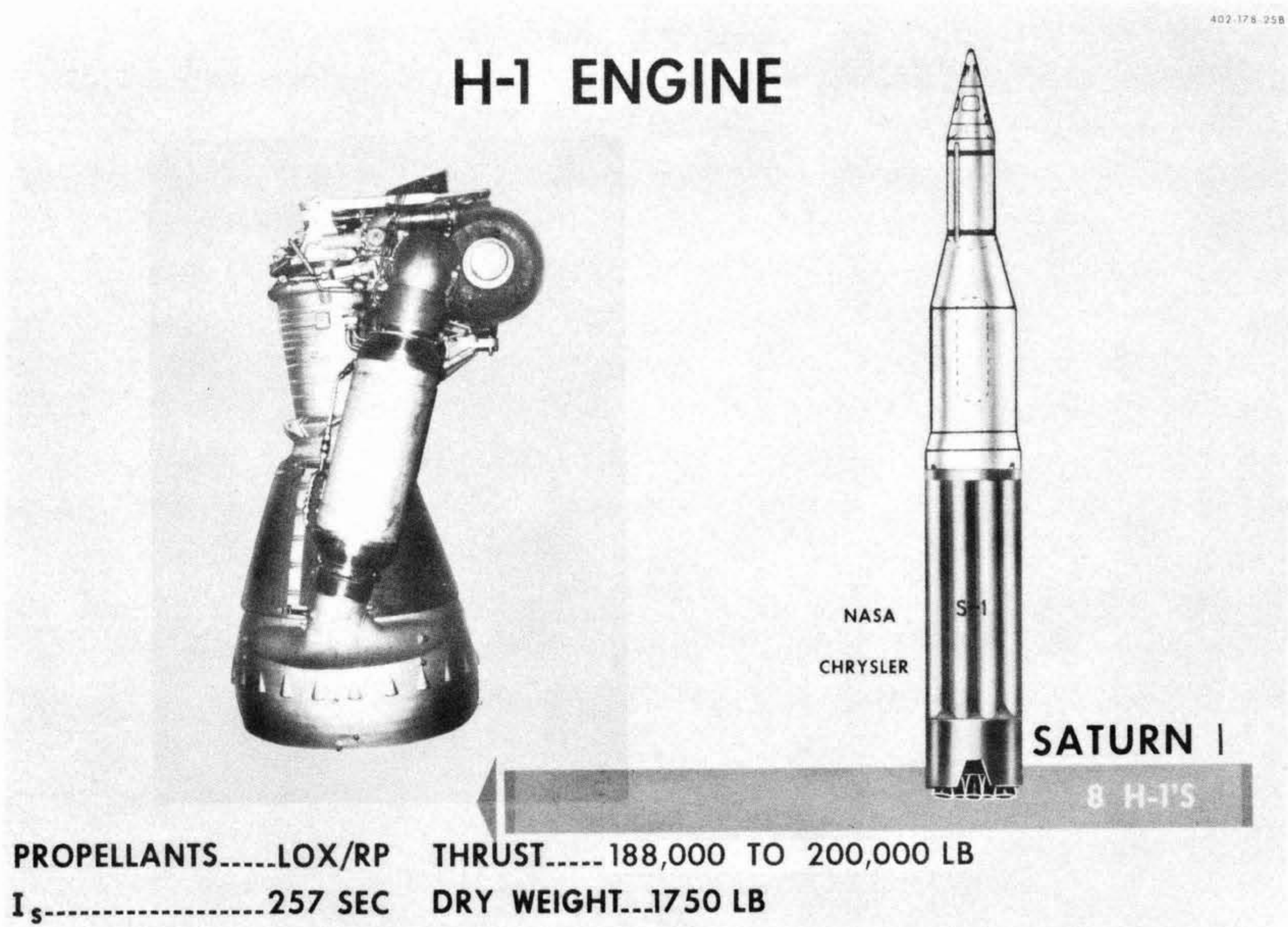


Figure 32. Saturn I Propulsion System

After the current series of the Saturn I flights, the vehicle will be uprated to the S-IB configuration by substituting the S-IV B with one Rocketdyne J-2, 200,000-lb-thrust, oxygen-hydrogen engine system for the second stage. This new stage will have over twice the thrust of the earlier stage. This launch vehicle will be used to place the Apollo "boilerplate" spacecraft and the Lunar excursion module, or LEM, in earth orbit for crew training in weightless environment, and rendezvous and docking techniques. Figure 33 shows a launch of one of the Saturn I vehicles.

Saturn V Launch Vehicle - The Saturn V vehicle as shown in Fig. 34 will have a booster thrust of 7,500,000 lb. The booster stage is powered by five F-1 engines each developing 1,500,000 lb of thrust operating on liquid oxygen/RP-1. The pad weight of the vehicle is 6,000,000 lb. Saturn V will launch 45 tons into lunar orbit trajectory. This vehicle is almost as tall as the Los Angeles City Hall.

Figure 35 shows a size comparison between the 188,000-lb-thrust H-1 engine, eight of which are used in the Saturn I vehicle, and also the 1,500,000-lb-thrust F-1 engine, five of which are used in the Saturn V first stage.

SATURN I LAUNCH

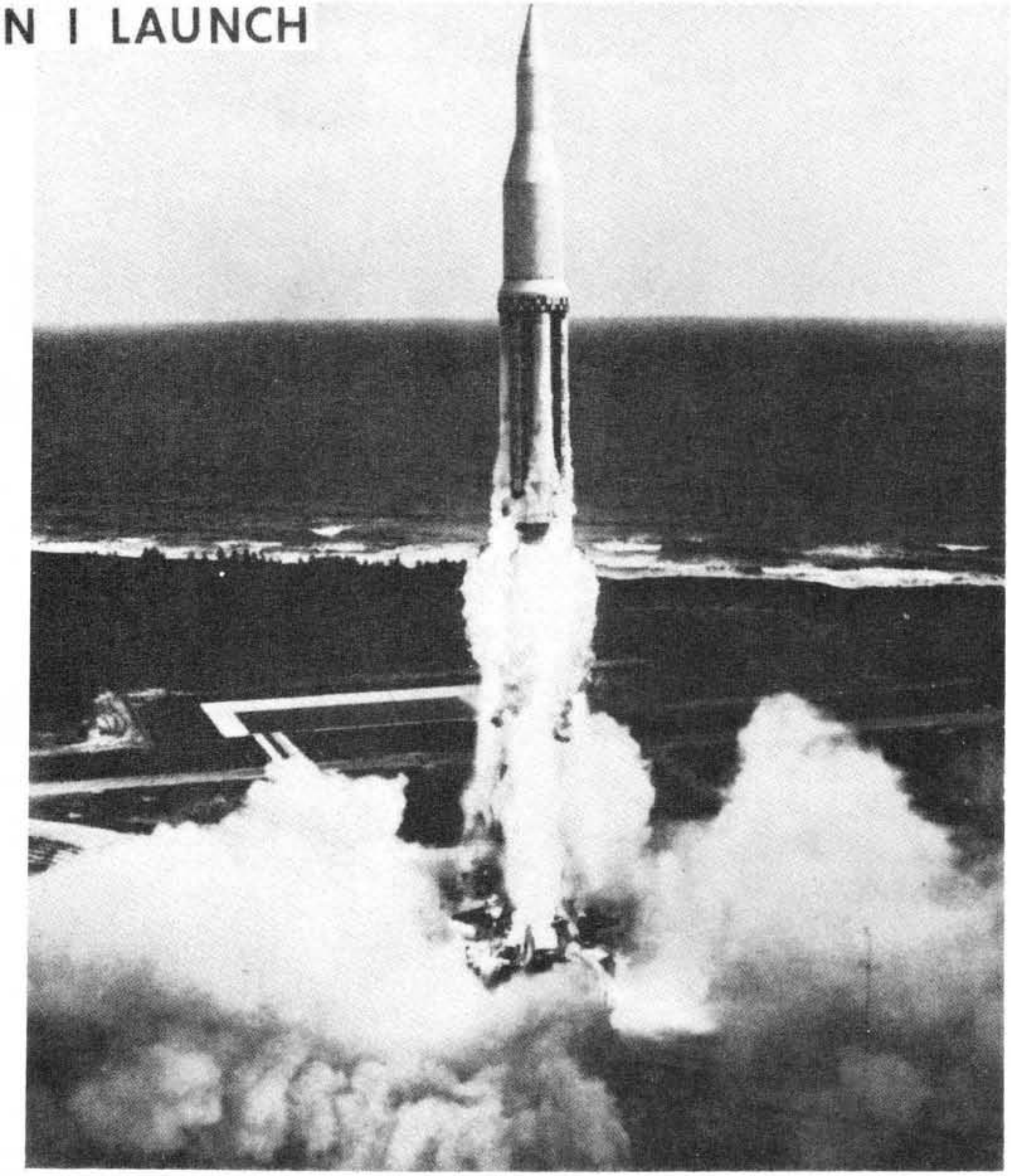


Figure 33. Saturn 1 Launch

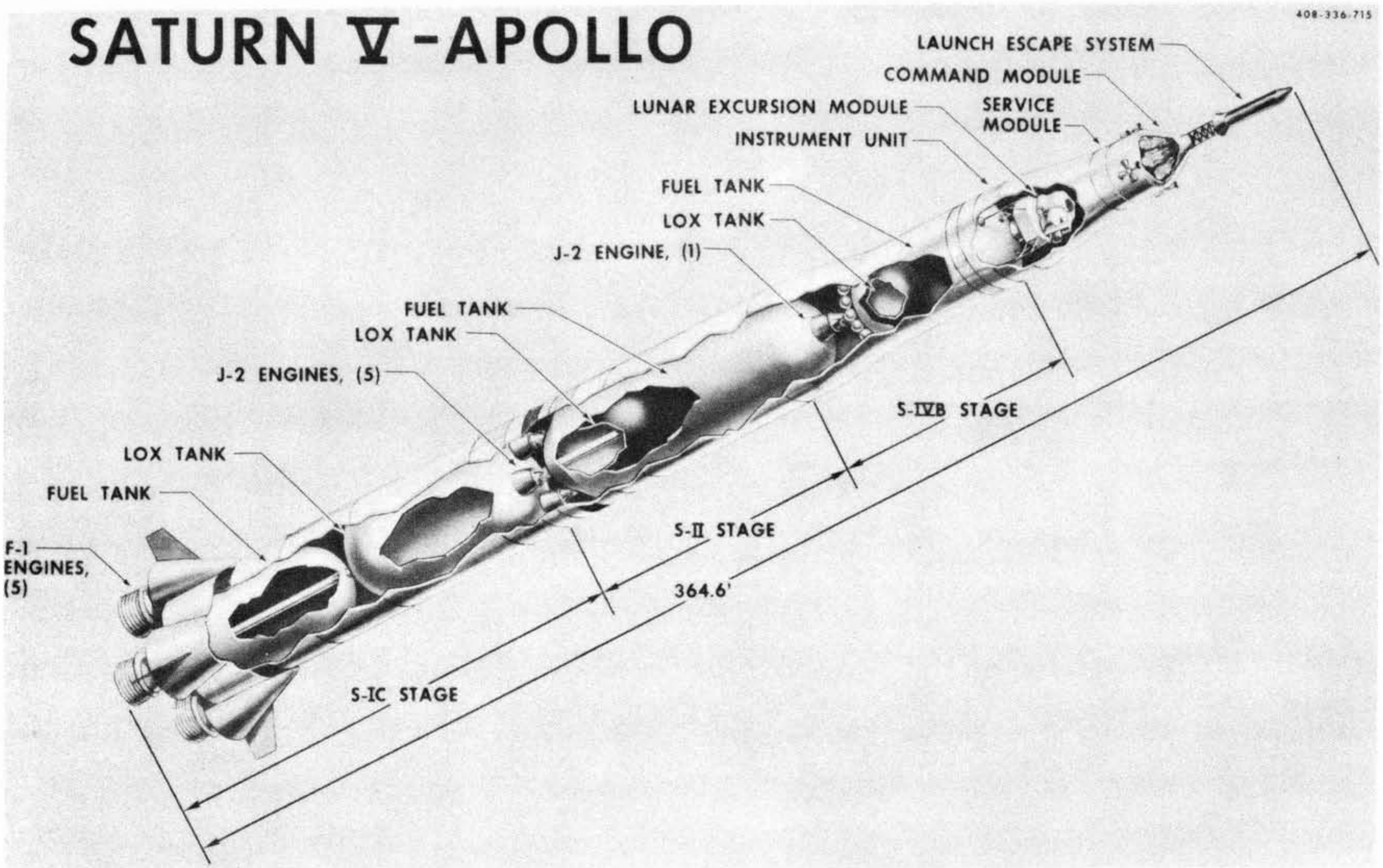


Figure 34. Saturn V Vehicle

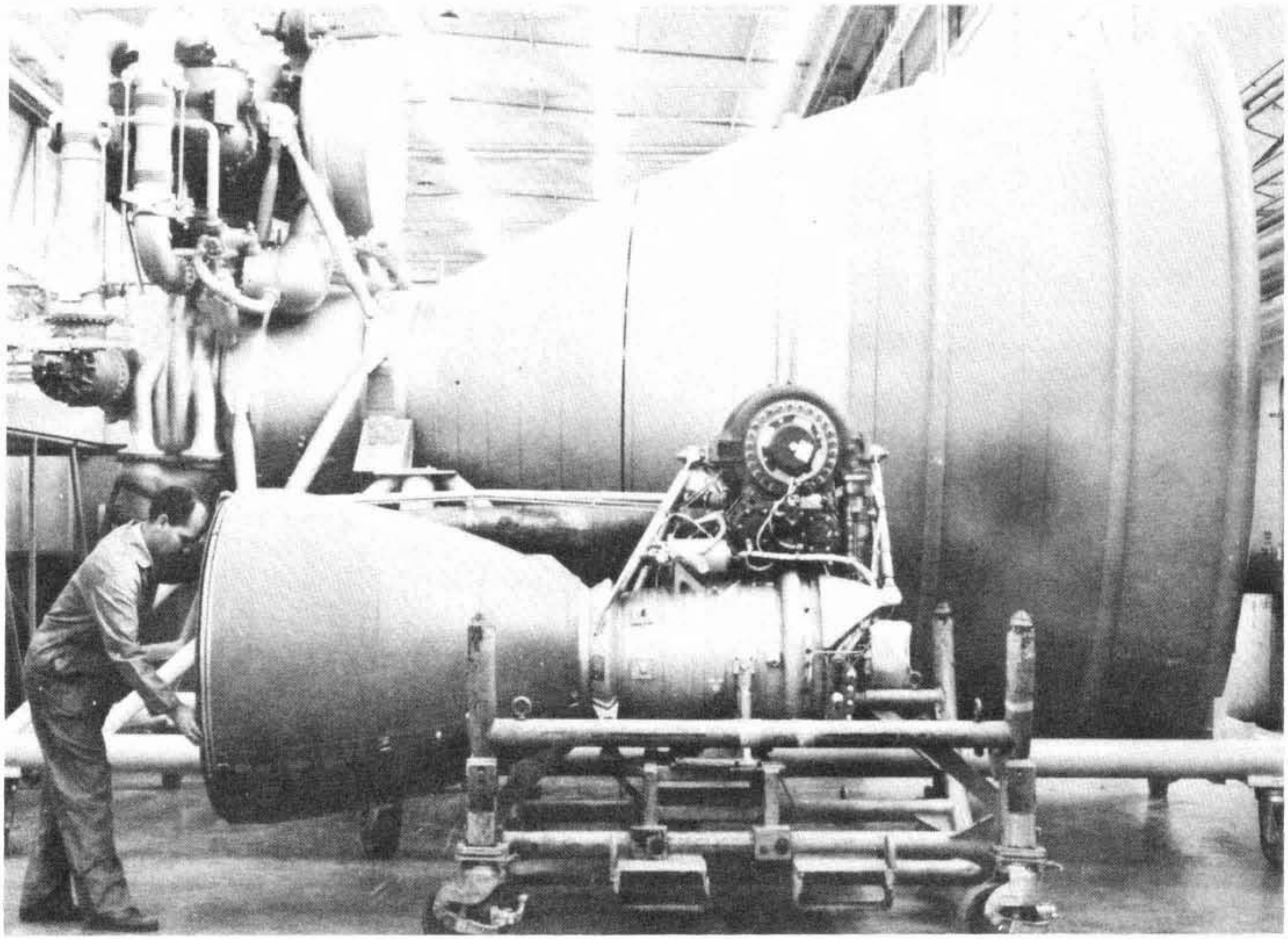


Figure 35. F-1 and H-1 Engines

The second stage of the Saturn vehicle is powered by five J-2 200,000-lb-thrust oxygen/hydrogen engines. One of these 200,000-lb-thrust oxygen-hydrogen engine systems is used in the third stage. This is the same stage used in the Saturn IB vehicle. The J-2 engine system is shown in Figure 36.

On top of the three stages mentioned before will be located the lunar excursion module and the Apollo Command module as shown in Figure 37. The manned flight to the moon will be accomplished by the lunar orbit concept. That is, the Apollo spacecraft, including the Lunar Excursion module or LEM, will be launched by Saturn V to a lunar orbit. The first stage, developing 7-1/2 million pounds of thrust, will burn for 150 seconds; the second stage will burn for 390 seconds. The third stage will burn for 160 seconds, shut down, coast, and then reignite to run for a total of 400 seconds to impart the velocity of 36,000 feet per second (26,600 miles per hour) required to place the Apollo spacecraft into an earth escape trajectory toward the moon. Injected into space are the third (S-IV) stage (now dry) and the Apollo spacecraft still attached to and mounted on top of the S-IV B stage. Shortly after entering the Lunar trajectory, shaped charges peel away the fairing, leaving the Apollo in free flight. The spacecraft is then turned by means of an attitude engine to dock, nose to nose, with the LEM riding in the forward portion of the third stage. Explosive bolts are then fired to separate the LEM form the third stage, which is left behind.

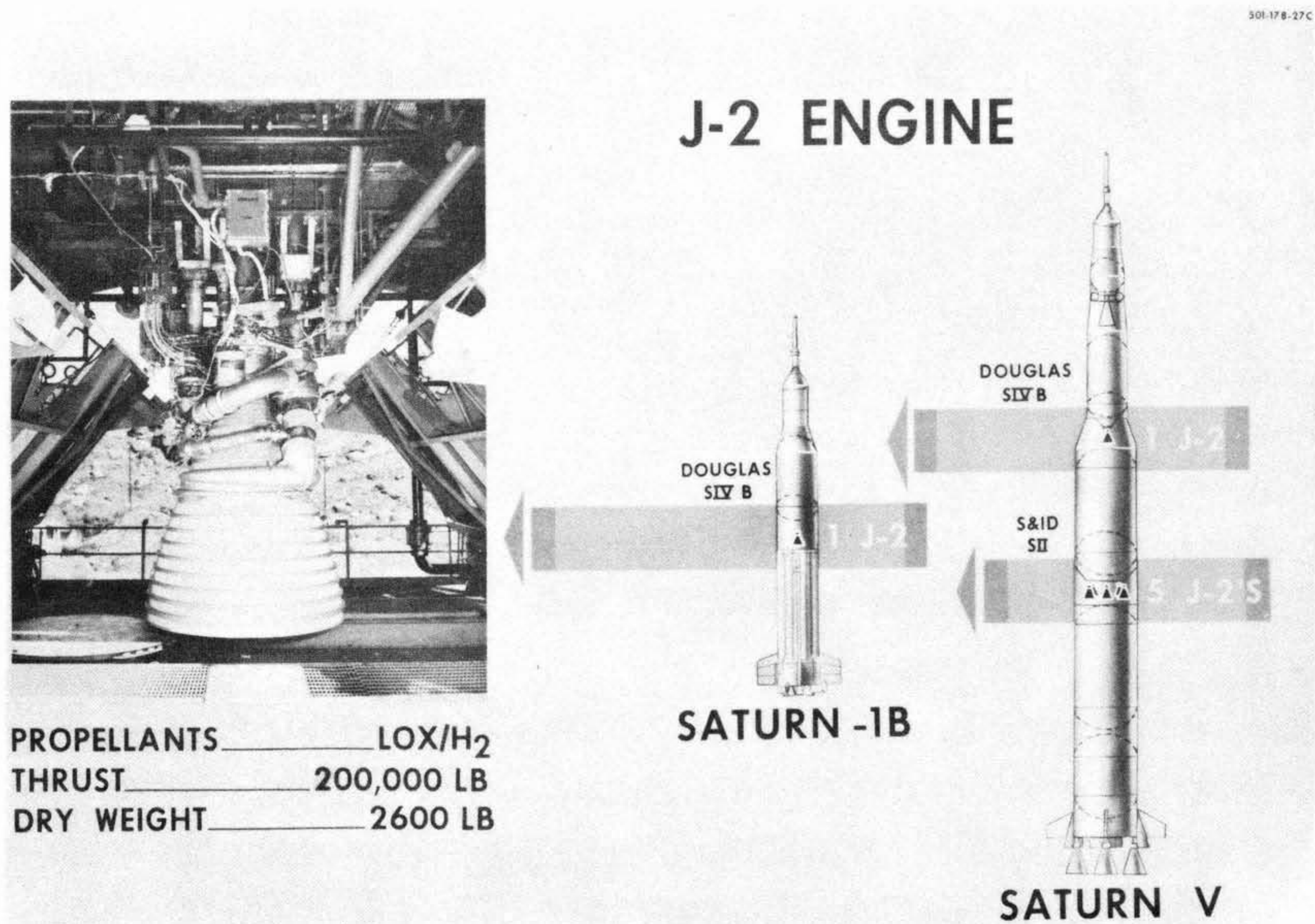


Figure 36. J-2 Engine System

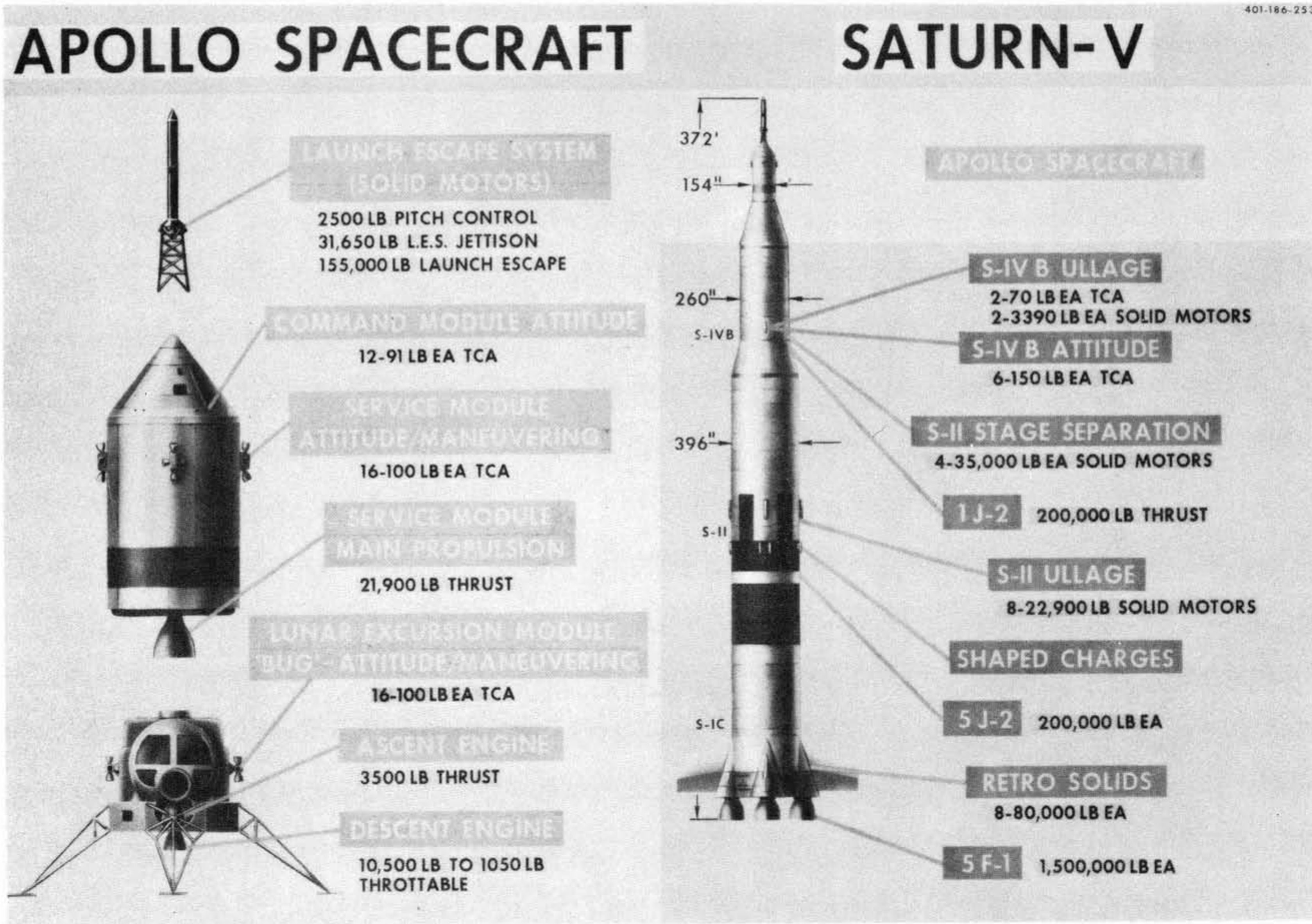


Figure 37. Apollo Spacecraft - Saturn V

It is with this vehicle that we will explore extra-terrestial areas. With this vehicle two men will be landed on the moon. They will explore it and return back to earth along with a third man who has been riding in the Apollo capsule above the moon. We have made tremendous strides with the Project Mercury and Project Gemini programs; the Apollo Launch Vehicle and Spacecraft will be one more step toward man conquering space.

FUTURE TRENDS

ADVANCED COMBUSTOR AND CHAMBER COOLING CONCEPTS

Performance increases can be achieved in bell nozzle design by higher expansion ratio nozzles. This desirability for increased area ratios has been hindered by the resulting nozzle size; this can be alleviated by going to higher chamber pressure operation. Today's chamber feasibility programs at 2500 and 3000 psia are paving the way for future increases in chamber operating pressures for new engine designs, just as the high pressure work at the 1500 psia level conducted in 1957 preceded the selection of the 1000 psia operating pressures for the current F-1 and M-1 systems. High chamber pressures place later importance on the combustor design, cooling of the combustor and nozzle throat section, and the turbomachinery. For the advanced annular nozzles (Fig. 38) a higher chamber pressure will also provide size reductions with similar problems of cooling and turbomachinery. For the bell nozzle designs, cylindrical combustion combustors have been developed and are being used on current engines. For the advanced annular nozzle, two types of combustors have evolved; the annular toroidal and the multi-chamber combustor. The annular toroidal combustor is a simple toroid with an appropriate annular opening to form the throat. The multi-chamber combustor system uses an annular arrangement of separated combustors either of cylindrical or oval shape, arranged so that combustor gas is exhausted into the common nozzle. High performance and combustion stability should be obtainable with both

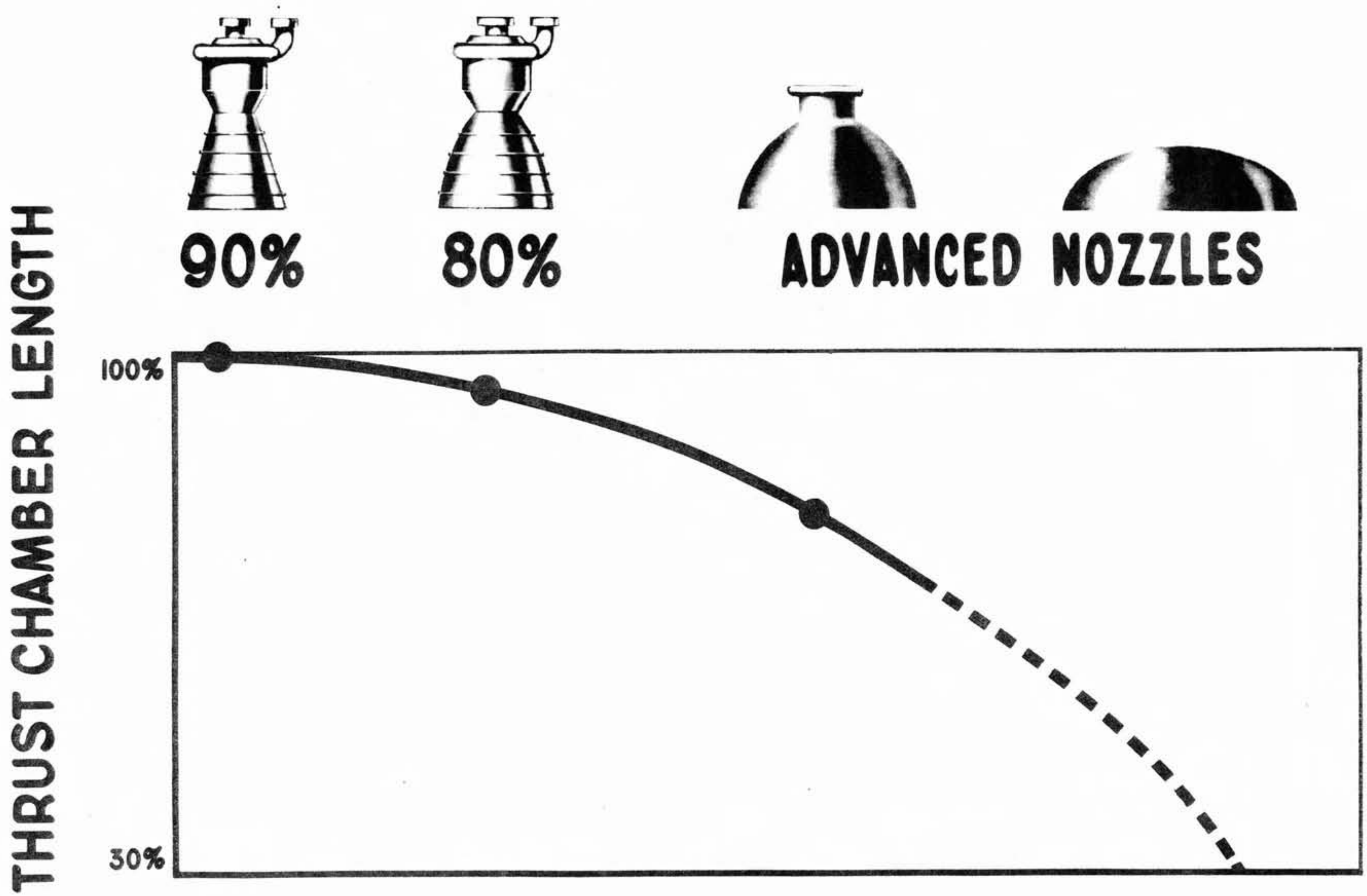


Figure 38. Nozzle Advancements

combustor concepts; with each configuration, modular or segment testing can be used to minimize development costs and prove out performance and stability. Many of the performance gains associated with high-pressure operation are jeopardized by the requirements for cooling the increasing heat fluxes. Technology programs are providing engineering data which will assist in the definition of a feasible system for high chamber pressure operation.

ADVANCED TURBOMACHINERY

The high flow rates and discharge pressure required by advanced engine designs have necessitated new high performance and low weight concepts of turbomachinery. Under several programs design studies and experimental investigations are being directed toward advanced turbomachinery for several propellant combinations and thrust levels. Evaluation of turbomachinery designs has shown that a major part of the weight in the structural members, volutes, and outer casings can be eliminated. New concepts which permit light weight designs for these items have evolved through novel arrangements of the flow passages, pump inducers, and pump impellers.

Effort is now focused on design improvements which will permit high-speed pump operation. High-speed pumps will permit small impellers and inducers to provide high flowrates and discharge pressures, and thus provide a lower weight unit. As attempts were made to increase pump design speed, subsequential technology advances were required to operate with available NPSH, overcome bearing and seal speed limitations, and design configurations which would not overstress turbine components. Advanced technology programs have indicated promising innovations in each of these areas, and pump designs with substantially increased speeds appear imminent.

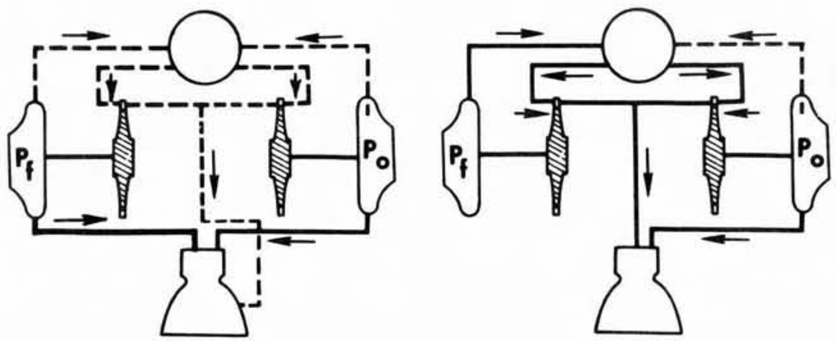
ADVANCED CYCLES

Gas generator and topping cycles (Fig. 39) are used on engine systems today. These cycles will still be used on many new engine systems. However, with higher performance being required on new engine systems and with higher chamber pressures, work is continuing to improve these cycles and new cycle concepts are being created.

PACKAGING (Fig. 40)

By combining all of the above concepts it will be possible to produce a propulsion system that will require much less space than in the past. The propulsion system will have higher performance, be more compact, have higher reliability, and will put up heavier payloads than ever before.

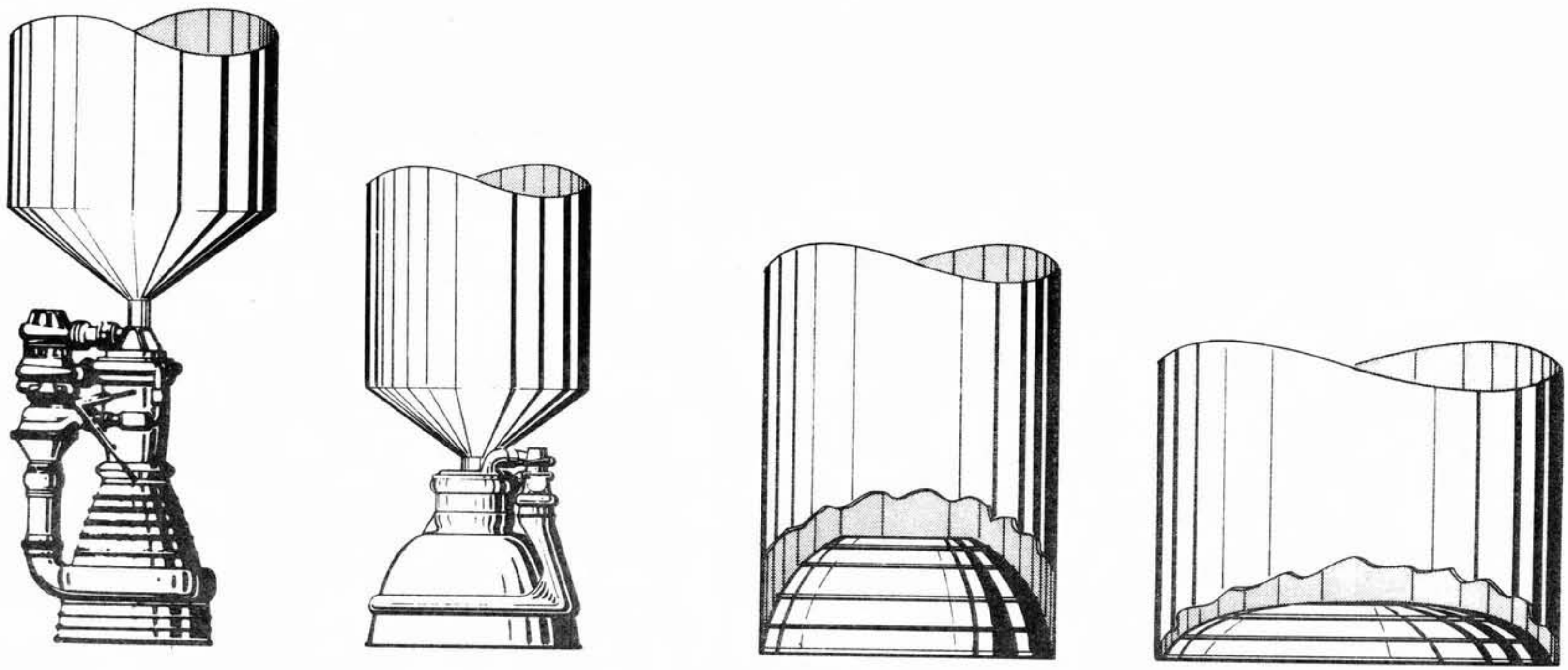
	GAS GENERATOR CYCLE	TOPPING CYCLE



RELATIVE RATING

	GAS GENERATOR CYCLE	TOPPING CYCLE
PERFORMANCE	2 (η= 0.98)	1 (η= 1.00)
SYSTEM COMPLEXITY	2	3
EXPANSION NOZZLE	2	2
PUMP DISCH. PRESS.	1	3
SYSTEM WEIGHT	2	3

Figure 39. Cycle Evolution



- COMPACT
- LIGHTWEIGHT
- ELIMINATION OF CONCENTRATED THRUST LOADS
- SHORT HIGH PRESSURE DUCTING
- REDUCED MISSILE LENGTHS
- REDUCED BENDING MOMENT

Figure 40. Engine Packaging

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