CHAPTER 6

LIQUID PROPELLANT ROCKET ENGINE FUNDAMENTALS

This is the first of five chapters devoted to liquid propellant rocket engines. It gives an overview of the engines (a definition of various propellants, engine performance, propellant budget), and of the smaller reaction control engines. It also presents several of their principal subsystems, such as two types of feed systems (including engine cycles), propellant tanks and their pressurization subsystems, valves and piping systems, and engine structures. Chapter 7 covers liquid propellants in more detail, Chapter 8 deals with thrust chambers (and nozzles), Chapter 9 with combustion, and Chapter 10 discusses turbopumps, engine design, engine controls, propellant budgets, engine balance and calibration, overall engine systems.

A liquid propellant rocket propulsion system is commonly called a *rocket engine*. It has all the hardware components and propellants necessary for its operation, that is, for producing thrust. It consists of one or more *thrust chambers*, one or more *tanks** to store the propellants, a *feed mechanism* to force the propellants from the tanks into the thrust chamber(s), a *power source* to furnish the energy for the feed mechanism, suitable *plumbing* or *piping* to transfer the liquids, a *structure** to transmit the thrust force, and *control* devices to initiate and regulate the propellant flow and thus the thrust. In some applications an engine may also include a *thrust vector control system*, various *instrumentation* and *residual propellant* (trapped in pipes, valves, or wetting tank walls). It does not include hardware for non-propulsive purposes, such

*The *tanks* and some or all of the *engine structure* and *piping* are sometimes considered to be part of the *vehicle* or the *test facility* and not the engine, depending on the preference of the organizations working on the project.

as aerodynamic surfaces, guidance, or navigation equipment, or the useful payload, such as a scientific space exploration package or a missile warhead.

Figures 1–3 and 1–4 show the basic flow diagrams for simple rocket engines with a *pressurized* and a *turbopump feed system*. Figure 6–1 shows a complex, sophisticated, high-performance liquid propellant rocket engine. References 6–1 and 6–2 give general liquid propellant rocket engine information. Additional data and figures on other rocket engines can be found in Chapter 10.

The design of any propulsion system is tailored to fit a specific application or *mission requirement*. These requirements are usually stated in terms of the application (anti-aircraft rocket, upper stage launch vehicle propulsion, or projectile assist), mission velocity, the desired flight trajectories (surface launch, orbit transfer, altitude–performance profile), vulnerability, attitude control torques and duty cycle, minimum life (during storage or in orbit), or number of units to be built and delivered. They include constraints on cost, schedule, operating conditions (such as temperature limits), storage conditions, or safety rules. Additional criteria, constraints, and the selection process are explained in Chapter 17.

The *mission requirements* can be translated into *rocket engine requirements* in terms of thrust–time profile, propellants, number of thrust chambers, total impulse, number of restarts, minimum reliability, likely propellant, and engine masses and their sizes or envelopes. We can do this only if we select several of the key engine features, such as the feed system, chamber pressure, the method of cooling the thrust chambers, thrust modulation (restart, throttle, thrust vector control), engine cycle (if using turbopump feed), and other key design features. We can arrive at one or more engine concepts and their preliminary or conceptual designs. Tables 1–3 to 1–5 give typical data. Many different types of rocket engines have been built and flown, ranging in thrust size from less than 0.01 lbf to over 1.75 million pounds, with one-time operation or multiple starts (some have over 150,000 restarts), with or without thrust modulation (called throttling), single use or reusable, arranged as single engines or in clusters of multiple units.

One way to categorize liquid propellant rocket engines is described in Table 6–1. There are two categories, namely those used for *boosting* a payload and imparting a significant velocity increase to a payload, and *auxiliary propulsion* for *trajectory adjustments* and *attitude control*. Liquid propellant rocket engine systems can be *classified* in several other ways. They can be *reusable* (like the Space Shuttle main engine or a booster rocket engine for quick ascent or maneuvers of fighter aircraft) or suitable for a *single flight* only (as the engines in the Atlas or Titan launch vehicles) and they can be *restartable*, like a reaction control engine, or *single firing*, as in a space launch vehicle. They can also be categorized by their *propellants, application*, or *stage*, such as an upper stage or booster stage, their *thrust level*, and by the *feed system type* (*pressurized* or *turbopump*).

The *thrust chamber* or *thruster* is the combustion device where the liquid propellants are metered, injected, atomized, mixed, and burned to form hot

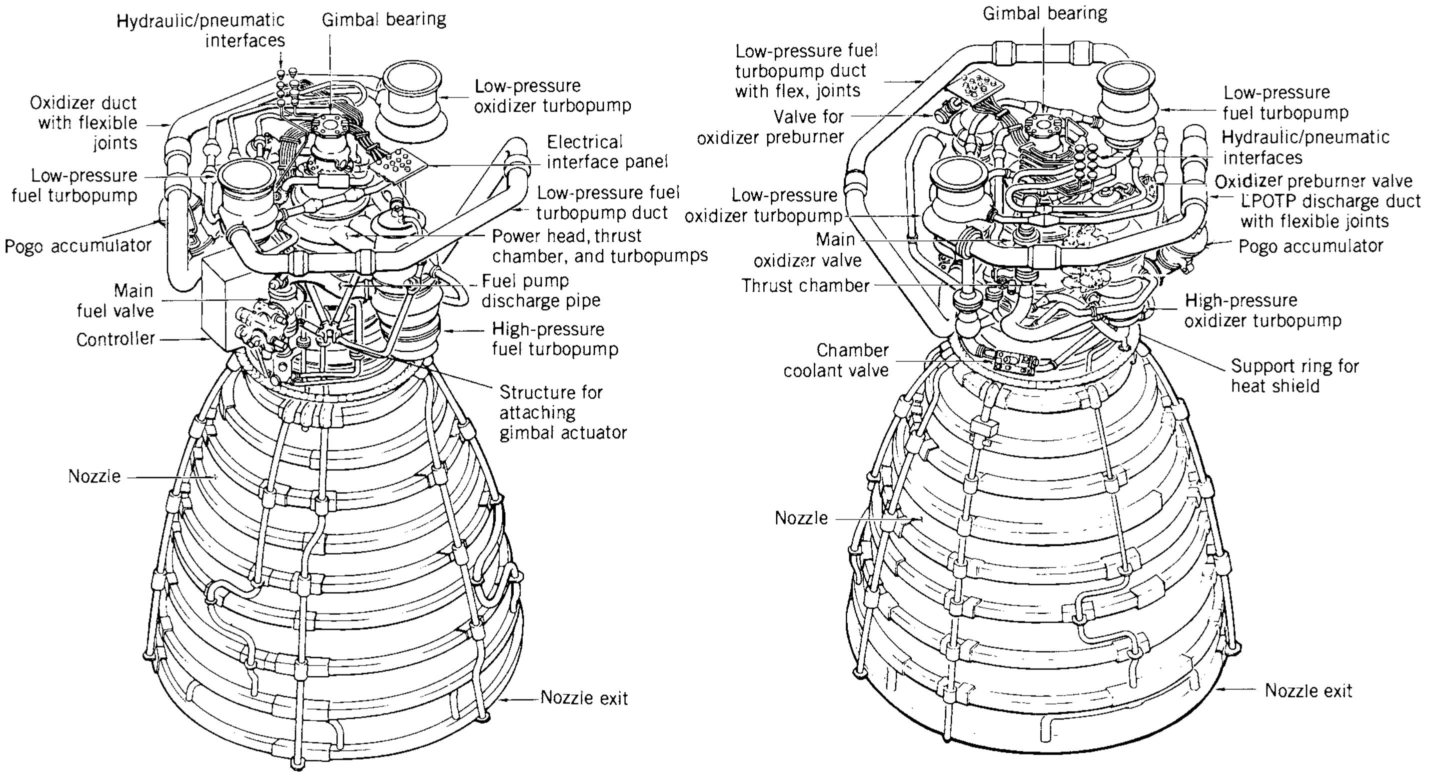


FIGURE 6–1. Two views of the Space Shuttle Main Engine (SSME). Its flowsheet is in Figure 6–12 and some component data are in Chapter 10. (Courtesy of The Boeing Company, Rocketdyne Propulsion and Power.)

TABLE 6–1. Characteristics of Two Categories of Liquid Propellant Rocket Engines

Purpose	Boost Propulsion	Auxiliary Propulsion
Mission	Impart significant velocity to propel a vehicle along its flight path	Attitude control, minor space maneuvers, trajectory corrections, orbit maintenance
Applications	Booster stage and upper stages of launch vehicles, large missiles	Spacecraft, satellites, top stage of anti-ballistic missile, space rendezvous
Total impulse	High	Low
Number of thrust chambers per engine	Usually 1; sometimes 4, 3, or 2	Between 4 and 24
Thrust level	High; 4500 N up to 7,900,000 N or 1000–1,770,000 lbf	Small; 0.001 up to 4500 N, a few go up to 1000 lbf
Feed system	Mostly turbopump type; occasionally pressurized feed system for smaller thrusts	Pressurized feed system with high-pressure gas supply
Propellants	Cryogenic and storable liquids (see next section)	Storable liquids, monopropellants, and/or stored cold gas
Chamber pressure	2.4–21 MPa or 350–3600 psi	0.14–2.1 MPa or 20–300 psi
Number of starts during a single mission	Usually no restart; sometimes one, but up to four in some cases	Several thousand starts are typical for small thrusters; fewer for larger thrust chambers, perhaps up to 10 starts
Cumulative duration of firing	Up to a few minutes	Up to several hours
Shortest firing duration	Typically 5–40 sec	0.02 sec typical for small thrusters
Time elapsed to reach full thrust	Up to several seconds	Usually very fast, 0.004–0.080 sec
Life in space	Hours, days, or months	10 years or more in space

gaseous reaction products, which in turn are accelerated and ejected at a high velocity to impart a thrust force. A thrust chamber has three major parts: an *injector*, a *combustion chamber*, and a *nozzle*. In a *cooled thrust chamber*, one of the propellants (usually the fuel) is circulated through cooling jackets or a special cooling passage to absorb the heat that is transferred from the hot reaction gases to the thrust chamber walls (see Figs 8–2 and 8–3). A *radiation-cooled* thrust chamber uses a special high-temperature material, such as niobium metal, which can radiate away its excess heat. There are *uncooled* or *heat-absorbing* thrust chambers, such as those using *ablative* materials. Thrust chambers are discussed in Chapter 8.

There are two types of *feed systems* used for liquid propellant rocket engines: those that use pumps for moving the propellants from their flight vehicle tanks to the thrust chamber, and those that use high-pressure gas for expelling or displacing their propellants from their tanks. They are discussed further in Chapter 10 and in Section 6.2 of this chapter.

Tables 17–1 to 17–4 compare the advantages and disadvantages of liquid propellant rocket engines and solid propellant rocket motors.

6.1. PROPELLANTS

The propellants, which are the working substance of rocket engines, constitute the fluid that undergoes chemical and thermodynamic changes. The term *liquid propellant* embraces all the various liquids used and may be one of the following:

1. Oxidizer (liquid oxygen, nitric acid, etc.)
2. Fuel (gasoline, alcohol, liquid hydrogen, etc.).
3. Chemical compound or mixture of oxidizer and fuel ingredients, capable of self-decomposition.
4. Any of the above, but with a gelling agent.

All are described in Chapter 7.

A *bipropellant* rocket unit has two separate liquid propellants, an oxidizer and a fuel. They are stored separately and are not mixed outside the combustion chamber. The majority of liquid propellant rockets have been manufactured for bipropellant applications.

A *monopropellant* contains an oxidizing agent and combustible matter in a single substance. It may be a mixture of several compounds or it may be a homogeneous material, such as hydrogen peroxide or hydrazine. Monopropellants are stable at ordinary atmospheric conditions but decompose and yield hot combustion gases when heated or catalyzed.

A *cold gas propellant* (e.g., nitrogen) is stored at very high pressure, gives a low performance, allows a simple system and is usually very reliable. It has been used for roll control and attitude control.

A *cryogenic propellant* is liquified gas at low temperature, such as liquid oxygen (−183°C) or liquid hydrogen (−253°C). Provisions for venting the storage tank and minimizing vaporization losses are necessary with this type.

Storable propellants (e.g., nitric acid or gasoline) are liquid at ambient temperature and can be stored for long periods in sealed tanks. *Space storable propellants* are liquid in the environment of space; this storability depends on the specific tank design, thermal conditions, and tank pressure. An example is ammonia.

A *gelled propellant* is a thixotropic liquid with a gelling additive. It behaves like a jelly or thick paint. It will not spill or leak readily, can flow under pressure, will burn, and is safer in some respects. It is described in a separate section of Chapter 7.

The propellant *mixture ratio* for a bipropellant is the ratio at which the oxidizer and fuel are mixed and react to give hot gases. The mixture ratio r is defined as the ratio of the oxidizer mass flow rate $\dot{m}_o$ and the fuel mass flow rate $\dot{m}_f$ or

$$r = \dot{m}_o/\dot{m}_f \tag{6–1}$$

The mixture ratio defines the composition of the reaction products. It is usually chosen to give a maximum value of specific impulse or $T_1/\mathfrak{M}$, where T_1 is the combustion temperature and $\mathfrak{M}$ is the average molecular mass of the reaction gases (see Eq. 3–16 or Fig. 3–2). For a given thrust F and a given effective exhaust velocity c, the total propellant flow is given by Eq. 2–6; namely, $\dot{m} = \dot{w}/g_0 = F/c$. The relationships between $r, \dot{m}, \dot{m}_o$, and $\dot{m}_f$ are

$$\dot{m}_o + \dot{m}_f = \dot{m} \tag{6–2}$$

$$\dot{m}_o = r\dot{m}/(r+1) \tag{6–3}$$

$$\dot{m}_f = \dot{m}/(r+1) \tag{6–4}$$

These same four equations are valid when w and $\dot{w}$ (weight) are substituted for m and $\dot{m}$. Calculated performance values for a number of different propellant combinations are given for specific mixture ratios in Table 5–5. Physical properties and a discussion of several common liquid propellants and their safety concerns are described in Chapter 7.

Example 6–1. A liquid oxygen–liquid hydrogen rocket thrust chamber of 10,000-lbf thrust operates at a chamber pressure of 1000 psia, a mixture ratio of 3.40, has exhaust products with a mean molecular mass of 8.9 lbm/lb-mol, a combustion temperature of 4380°F, and a specific heat ratio of 1.26. Determine the nozzle area, exit area for optimum operation at an altitude where $p_3 = p_2 = 1.58$ psia, the propellant weight and volume flow rates, and the total propellant requirements for 2 min of operation. Assume that the actual specific impulse is 97% of the theoretical value.

SOLUTION. The exhaust velocity for an optimum nozzle is determined from Eq. 3–16, but with a correction factor of g_0 for the foot-pound system.

$$v_2 = \sqrt{\frac{2g_0k}{k-1}\frac{R'T_1}{\mathfrak{M}}\left[1-\left(\frac{p_2}{p_1}\right)^{(k-1)/k}\right]}$$

$$= \sqrt{\frac{2 \times 32.2 \times 1.26}{0.26}\frac{1544 \times 4840}{8.9}(1 - 0.00158^{0.205})} = 13{,}900 \text{ ft/sec}$$

The theoretical specific impulse is c/g_0, or in this case v_2/g_0 or 13,900/32.2 = 431 sec. The actual specific impulse is $0.97 \times 431 = 418$ sec. The theoretical or ideal thrust coefficient can be found from Eq. 3–30 or from Fig. 3–6 ($p_2 = p_3$) to be $C_F = 1.76$. The actual thrust coefficient is slightly less, say 98% or $C_F = 1.72$. The throat area required is found from Eq. 3–31.

$$A_t = F/(C_F p_1) = 10{,}000/(1.72 \times 1000) = 5.80 \text{ in.}^2 \text{ (2.71 in. diameter)}$$

The optimum area ratio can be found from Eq. 3–25 or Fig. 3–5 to be 42. The exit area is $5.80 \times 42 = 244$ in.^{2} (17.6 in. diameter). The weight density of oxygen is 71.1 lbf/ft^3 and of hydrogen is 4.4 lbf/ft^3. The propellant weight flow rate is (Equation 2–5)

$$\dot{w} = F/I_s = 10{,}000/418 = 24.0 \text{ lbf/sec}$$

The oxygen and fuel weight flow rates are, from Eqs. 6–3 and 6–4,

$$\dot{w}_o = \dot{w}r/(r+1) = 24.0 \times 3.40/4.40 = 18.55 \text{ lbf/sec}$$
$$\dot{w}_f = \dot{w}/(r+1) = 24/4.40 = 5.45 \text{ lbf/sec}$$

The volume flow rates are determined from the densities and the weight flow rates.

$$\dot{V}_o = \dot{w}_o/\rho_o = 18.55/71.1 = 0.261 \text{ ft}^3\text{/sec}$$
$$\dot{V}_f = \dot{w}_f/\rho_f = 5.45/4.4 = 1.24 \text{ ft}^3\text{/sec}$$

For 120 sec of operations (arbitrarily allow the equivalent of two additional seconds for start and stop transients and unavailable propellant), the weight and volume of required propellant are

$$w_o = 18.55 \times 122 = 2260 \text{ lbf of oxygen}$$
$$w_f = 5.45 \times 122 = 665 \text{ lbf of hydrogen}$$
$$V_o = 0.261 \times 122 = 31.8 \text{ ft}^3 \text{ of oxygen}$$
$$V_f = 1.24 \times 122 = 151 \text{ ft}^3 \text{ of hydrogen}$$

Note that, with the low-density fuel, the volume flow rate and therefore the tank volume of hydrogen are large compared to that of the oxidizer.

6.2. PROPELLANT FEED SYSTEMS

The propellant feed system has two principal functions: to raise the pressure of the propellants and to feed them to one or more thrust chambers. The energy for these functions comes either from a high-pressure gas, centrifugal pumps, or a combination of the two. The selection of a particular feed system and its components is governed primarily by the application of the rocket, the requirements mentioned at the beginning of this chapter, duration, number or type of thrust chambers, past experience, mission, and by general requirements of simplicity of design, ease of manufacture, low cost, and minimum inert mass. A classification of several of the more important types of feed system is shown in Fig. 6–2 and some are discussed in more detail below. All feed systems have piping, a series of valves, provisions for filling and removing (draining and flushing) the liquid propellants, and control devices to initiate, stop, and regulate their flow and operation.

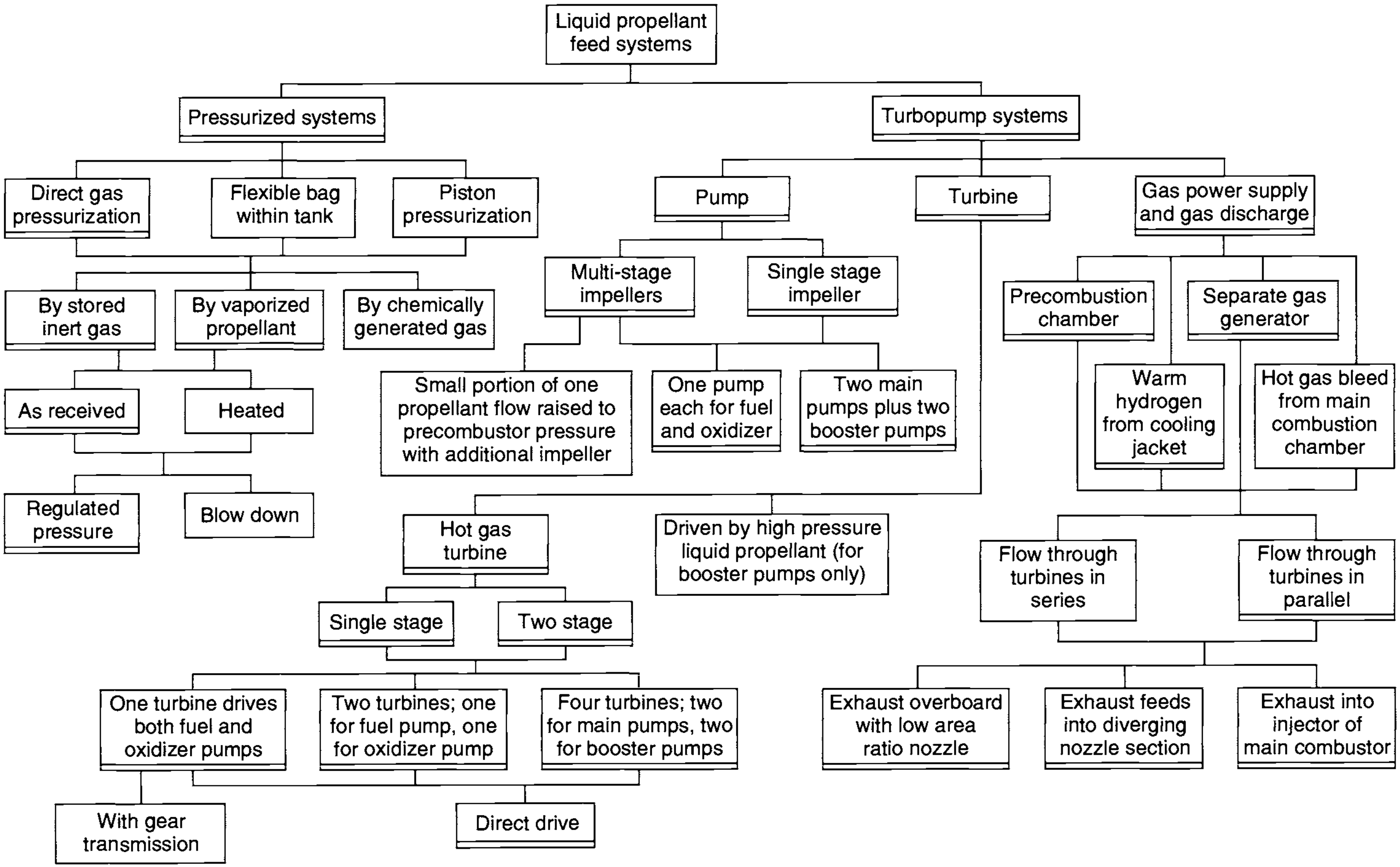


FIGURE 6–2. Design options of fed systems for liquid propellant rocket engines. The more common types are designated with a double line at the bottom of the box.

In general, a pressure feed system gives a vehicle performance superior to a turbopump system when the total impulse or the mass of propellant is relatively low, the chamber pressure is low, the engine thrust-to-weight ratio is low (usually less than 0.6), and when there are repeated short-duration thrust pulses; the heavy-walled tanks for the propellant and the pressurizing gas usually constitute the major inert mass of the engine system. In a turbopump feed systems the propellant tank pressures are much lower (by a factor of 10 to 40) and thus the tank masses are much lower (again by a factor of 10 to 40). Turbopump systems usually give a superior vehicle performance when the total impulse is large (higher Δu) and the chamber pressure is higher.

The pressurized feed system can be relatively simple, such as for a single-operation, factory-preloaded, simple unit (with burst diaphragms instead of some of the valves), or quite complex, as with multiple restartable thrusters or reusable systems. Table 6–2 shows typical features that have been designed into pressurized feed systems in order to satisfy particular design goals. Figures 1–3, 6–3, 6–4, and 6–13 show some of these features. If the propulsion system is to be reusable or is part of a manned vehicle (where the reliability requirements are very high and the vehicle's crew can monitor and override automatic commands), the feed system becomes more complex (with more safety features and redundancies) and more expensive.

The pneumatic (pressurizing gas) and hydraulic (propellant) flows in a liquid propellant engine can be simulated in a computer analysis that provides for a flow and pressure balance in the oxidizer and the fuel flow paths through the system. One approach is shown in Ref. 6–3. Some of these analyses can provide information on transient conditions (filling up of passages) during start, flow decays at cutoff, possible water hammer, or flow instabilities. The details of such analyses are not described in this book, but the basic mathematical simulation is relatively straightforward.

6.3. GAS PRESSURE FEED SYSTEMS

One of the simplest and most common means of pressurizing the propellants is to force them out of their respective tanks by displacing them with high-pressure gas. This gas is fed into the propellant tanks at a controlled pressure, thereby giving a controlled propellant discharge. Because of their relative simplicity, the rocket engines with pressurized feed systems can be very reliable. Reference 6–3 includes a design guide for pressurized gas systems.

A simple pressurized feed system is shown schematically in Fig. 1–3. It consists of a high-pressure gas tank, a gas starting valve, a pressure regulator, propellant tanks, propellant valves, and feed lines. Additional components, such as filling and draining provisions, check valves, filters, flexible elastic bladders for separating the liquid from the pressurizing gas, and pressure sensors or gauges, are also often incorporated. After all tanks are filled, the high-pressure gas valve in Fig. 1–3 is remotely actuated and admits gas through

TABLE 6–2. Typical Features of Liquid Propellant Feed Systems

Enhance Safety

Sniff devices to detect leak of hazardous vapor; used on Space Shuttle orbiter
Check valves to prevent backflow of propellant into the gas tank and inadvertent mixing of propllants inside flow passages
Features that prevent an unsafe condition to occur or persist and shut down engine safely, such as relief valves or relief burst diaphragms to prevent tank overpressurization), or a vibration monitor to shut off operation in the case of combustion instability
Isolation valves to shut off a section of a system that has a leak or malfunction
Burst diaphragms or isolation valves to isolate the propellants in their tanks and positively prevent leakage into the thrust chamber or into the other propellant tank during storage
Inert pressurizing gas

Provide Control

Valves to control pressurization and flow to the thrust chambers (start/stop/throttle)
Sensors to measure temperatures, pressures, valve positions, thrust, etc., and computers to monitor/analyze system status, issue command signals, and correct if sensed condition is outside predetermined limits
Manned vehicle can require system status display and command signal override
Fault detection, identification, and automatic remedy, such as shut-off isolation valves in compartment in case of fire, leak, or disabled thruster
Control thrust (throttle valve) to fit a desired thrust–time profile

Enhance Reliability

Fewest practical number of components/subassemblies
Ability to provide emergency mode engine operation, such as return of Space Shuttle vehicle to landing
Filters to catch dirt in propellant lines, which could prevent valve from closing or small injector holes from being plugged up or bearings from galling.
Duplication of unreliable key components, such as redundant small thrusters, regulators, check valves, or isolation valves
Heaters to prevent freezing of moisture or low-melting-point propellant
Long storage life—use propellants with little or no chemical deterioration and no reaction with wall materials

Provide for Reusability

Provisions to drain remaining propellants or pressurants
Provision for cleaning, purging, flushing, and drying the feed system and refilling propellants and pressurizing gas in field
Devices to check functioning of key components prior to next operation
Features to allow checking of engine calibration and leak testing after operation
Features for access of inspection devices for visual inspection at internal surfaces or components

Enable Effective Propellant Utilization

High tank expulsion efficiency with minimum residual, unavailable propellant
Lowest possible ambient temperature variation or matched propellant property variation with temperature so as to minimize mixture ratio change and residual propellant
Alternatively, measure remaining propellant in tanks (using a special gauge) and automatically adjust mixture ratio (throttling) to minimize residual propellant
Minimize pockets in the piping and valves that cannot be readily drained

the pressure regulator at a constant pressure to the propellant tanks. The check valves prevent mixing of the oxidizer with the fuel when the unit is not in an upright position. The propellants are fed to the thrust chamber by opening valves. When the propellants are completely consumed, the pressurizing gas can also scavenge and clean lines and valves of much of the liquid propellant residue. The variations in this system, such as the combination of several valves into one or the elimination and addition of certain components, depend to a large extent on the application. If a unit is to be used over and over, such as space-maneuver rocket, it will include several additional features such as, possibly, a thrust-regulating device and a tank level gauge; they will not be found in an expendable, single-shot unit, which may not even have a tank-drainage provision. Different bipropellant pressurization concepts are evaluated in Refs. 6–3, 6–4, and 6–5. Table 6–2 lists various optional features. Many of these features also apply to pump-fed systems, which are discussed in Section 6..6. With monopropellants the gas pressure feed system becomes simpler, since there is only one propellant and not two, reducing the number of pipes, valves, and tanks.

A complex man-rated pressurized feed system, the combined *Space Shuttle Orbital Maneuver System* (OMS) *and the Reaction Control System* (RCS), is described in Figs 6–3 and 6–4, Ref. 6–6, and Table 6–3. There are three locations for the RCS, as shown in Fig. 1–13: a forward pod and a right and left aft pod. Figures 6–3 and 6–4 refer to one of the aft pods only and show a combined OMS and RCS arrangement. The OMS provides thrust for orbit insertion, orbit circularization, orbit transfer, rendezvous, deorbit, and abort. The RCS provides thrust for attitude control (in pitch, yaw, and roll) and for small-vehicle velocity corrections or changes in almost any direction (translation maneuvers), such as are needed for rendezvous and docking; it can operate simultaneously with or separate from the OMS.

The systems feature various redundancies, an automatic RCS thruster selection system, various safety devices, automatic controls, sensors to allow a display to the Shuttle's crew of the system's status and health, and manual command overrides. The reliability requirements are severe. Several key components, such as all the helium pressure regulators, propellant tanks, some valves, and about half the thrusters are duplicated and redundant; if one fails, another can still complete the mission. It is possible to feed up to 1000 lbm of the liquid from the large OMS propellant tanks to the small RCS ones, in case it is necessary to run one or more of the small reaction control thrusters for a longer period and use more propellant than the smaller tanks allow; it is also possible to feed propellant from the left aft system to the one on the vehicle's right side, and vice versa. These features allow for more than nominal total impulse in a portion of the thrusters, in case it is needed for a particular mission mode or an emergency mode.

The compartmented steel propellant tanks with antislosh and antivortex baffles, sumps, and a surface tension propellant retention device allow propellant to be delivered independent of the propellant load, the orientation, or the

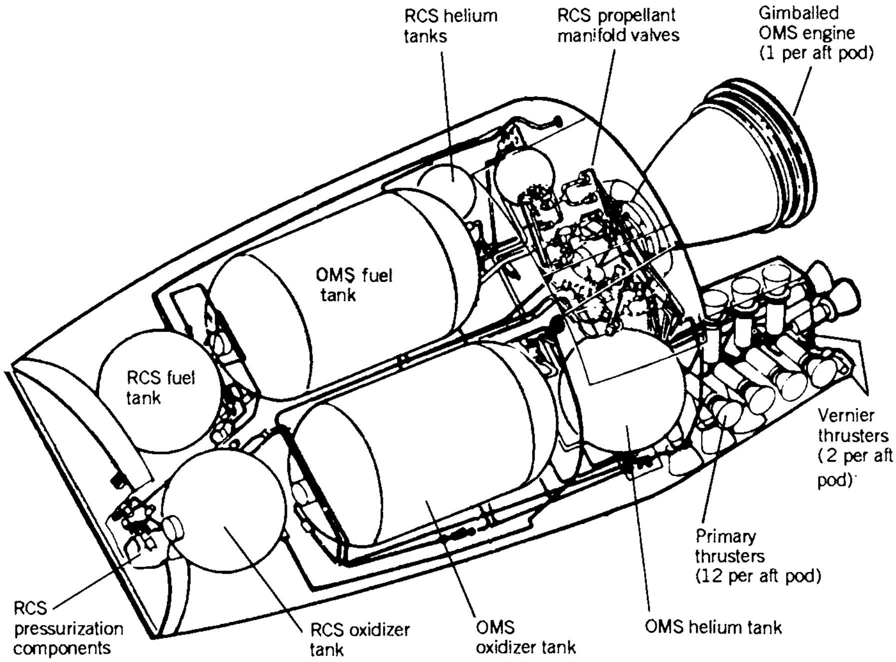


FIGURE 6–3. Simplified sketch at the left aft pod of the Space Shuttle's Orbiting Maneuvering System (OMS) and the Reaction Control System (RCS). (Source: NASA.)

acceleration environment (some of the time in zero-g). Gauges in each tank allow a determination of the amount of propellant remaining, and they also indicate a leak. Safety features include sniff lines at each propellant valve actuator to sense leakage. Electrical heaters are provided at propellant valves, certain lines, and injectors to prevent fuel freezing or moisture forming into ice.

A typical RCS feature that enhances safety and reliability is a self-shutoff device is small thrusters that will cause a shutdown in case they should experience instability and burn through the walls. Electrical lead wires to the propellant valves are wrapped around the chamber and nozzle; a burnout will quickly melt the wire and cut the power to the valve, which will return to the spring-loaded closed position and shut off the propellant flow.

The majority of pressurized feed systems use a pressure regulator to maintain the propellant tank pressure and thus also the thrust at constant values. The required mass of pressurizing gas can be significantly reduced by a *blowdown system* with a "tail-off" pressure decay. The propellants are expelled by the expansion of the gas already in the enlarged propellant tanks. The tank pressure and the chamber pressure decrease or progressively decay during this adiabatic expansion period. The alternatives of either regulating the inert gas pressure or using a blowdown system are compared in Table 6–4; both types

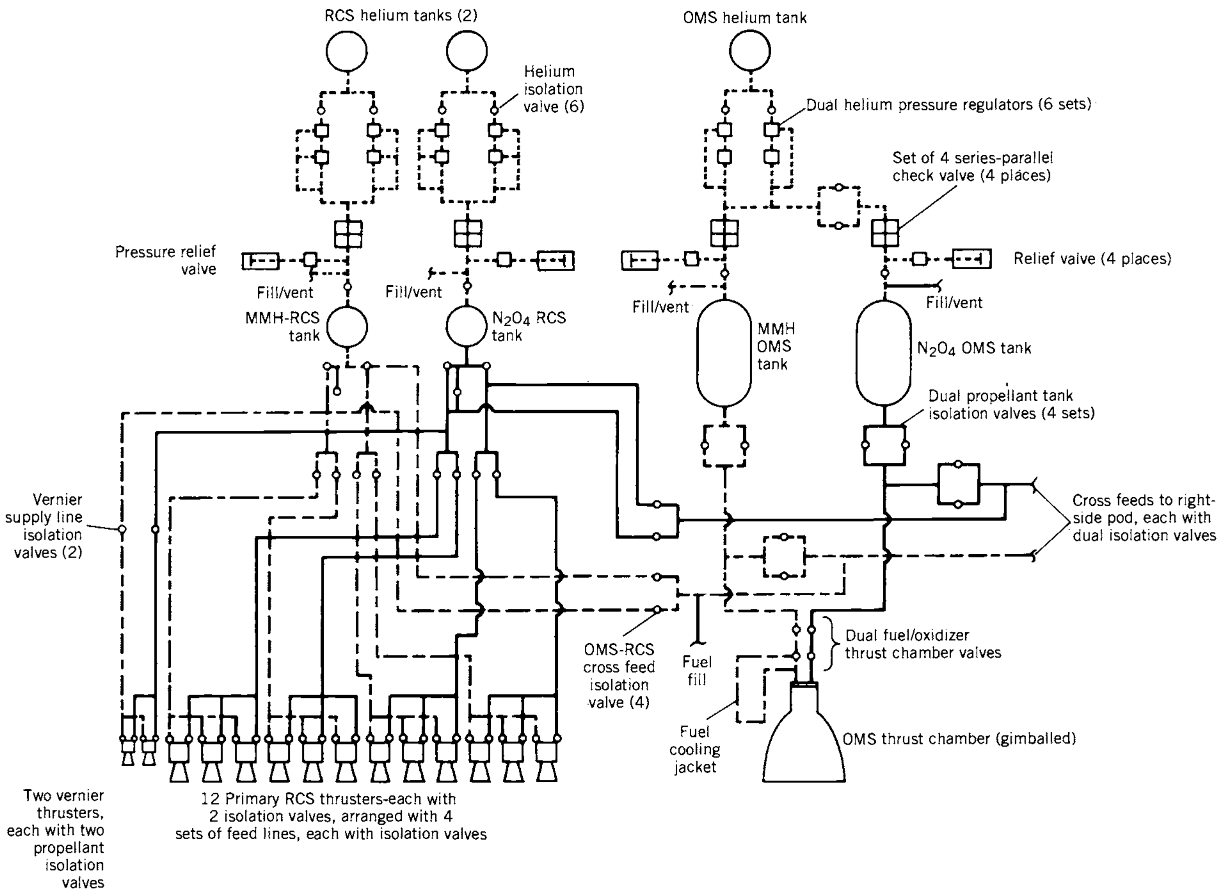


FIGURE 6–4. Simplified flow diagram of the propellant feed system flow for the left aft pod of the Orbital Maneuvering System (OMS) and Reaction Control System (RCS) of the Space Shuttle Orbiter Vehicle. Solid lines: nitrogen tetroxide (N_2O_4); dash-dot lines: monomethylhydrazine (MMH); short dashed lines: high-pressure helium. (Source: NASA.)

TABLE 6–3. Characteristics of the Orbital Maneuver System (OMS) and the Reaction Control System (RCS) of the Space Shuttle in One of the Aft Side Podes

Item	OMS	Primary RCS	Vernier RCS
Thrust (per nozzle) (lbf)	6000	870	25
Number of thrusters per pod	1	12	2
Thrust chamber cooling	Regenerative and radiation	Radiation cooling	
Chamber pressure, nominal (psi)	125	152	110
Specific impulse (vacuum nominal) (sec)	313	280[a]	265[a]
Nozzle area ratio	55	22–30[a]	20–50[a]
Mixture ratio (oxide/fuel mass flow)	1.65	1.6	1.6
Burn time, minimum (sec)	2	0.08	0.08
Burn time, maximum (sec)	160	150	125
Burn time, cumulative (sec)	54,000	12,800	125,000
Number of starts, cumulative (sec)	1000	20,000	330,000
Oxidizer (N_2O_4) weight in tank (lb)	14,866	1464	
Fuel (MMH) weight in tank (lb)	9010	923	
Number of oxidizer/fuel tanks	1/1	1/1	
Propellant tank volume, each tank (ft^3)	90	17.9	
Ullage volume, nominal (full tank) (ft^3)	7.8	1.2–1.5	
Tank pressure, nominal (psi)	250	280	
Helium storage tank pressure (psi)	4700	3600	
Number of helium tanks	1	2	
Volume of helium tanks (ft^3)	17	1.76	

[a]Depends on specific vehicle location and scarfing of nozzle.
Sources: NASA, Aerojet Propulsion Company and Kaiser Marquardt Company.

are currently being used. The selection depends on specific application requirements, cost, inert mass, reliability, and safety considerations (see Refs. 6–4 and 6–5).

Some pressure feed systems can be prefilled with propellant and pressurizing agent at the factory and stored in readiness for operation. Compared to a solid propellant rocket unit, these storable prepackaged liquid propellant pressurized feed systems offer advantages in long-term storability and resistance to transportation vibration or shock.

The *thrust* level of a rocket propulsion system with a pressurized gas feed system is determined by the magnitude of the propellant flow which, in turn, is determined by the gas pressure regulator setting. The propellant *mixture ratio* in this type of feed system is controlled by the hydraulic resistance of the liquid propellant lines, cooling jacket, and injector, and can usually be adjusted by means of variable or interchangeable restrictors. Further discussion of the adjusting of thrust and mixture ratio can be found in Section 10.6 and in Example 10–3.

TABLE 6-4. Comparison of Two Types of Gas Pressurization Systems

Type	Regulated Pressure	Blowdown
Pressure/thrust	Stays essentially constant	Decreases as propellant is consumed
Gas storage	In separate high-pressure tanks	Gas is stored inside propellant tank with large ullage volume (30 to 60%)
Required components	Needs regulator, filter, gas valve, and gas tank	Larger, heavier propellant tanks
Advantages	Constant-pressure feed gives essentially constant propellant flow and approximately constant thrust, constant I_s and r Better control of mixture ratio	Simpler system Less gas required Can be less inert mass
Disadvantages	Slightly more complex Regulator introduces a small pressure drop Gas stored under high pressure Shorter burning time	Thrust decreases with burn duration Somewhat higher residue propellant due to less accurate mixture ratio control Thruster must operate and be stable over wide range of thrust values and modest range of mixture ratio Propellants stored under pressure; slightly lower I_s toward end of burning time

6.4. PROPELLANT TANKS

In liquid bipropellant rocket engine systems propellants are stored in one or more oxidizer tanks and one or more fuel tanks; monopropellant rocket engine systems have, of course, only one set of propellant tanks. There are also one or more high-pressure gas tanks, the gas being used to pressurize the propellant tanks. Tanks can be arranged in a variety of ways, and the tank design can be used to exercise some control over the change in the location of the vehicle's center of gravity. Typical arrangements are shown in Fig. 6-5. Because the propellant tank has to fly, its mass is at a premium and the tank material is therefore highly stressed. Common tank materials are aluminum, stainless steel, titanium, alloy steel, and fiber-reinforced plastics with an impervious thin inner liner of metal to prevent leakage through the pores of the fiber-reinforced walls.

The extra volume of gas above the propellant in sealed tanks is called *ullage*. It is necessary space that allows for thermal expansion of the propellant liquids, for the accumulation of gases that were originally dissolved in the propellant, or for gaseous products from slow reactions within the propellant during storage. Depending on the storage temperature range, the propellants' coefficient of thermal expansion, and the particular application, the ullage volume is usually between 3 and 10% of the tank volume. Once propellant is loaded into

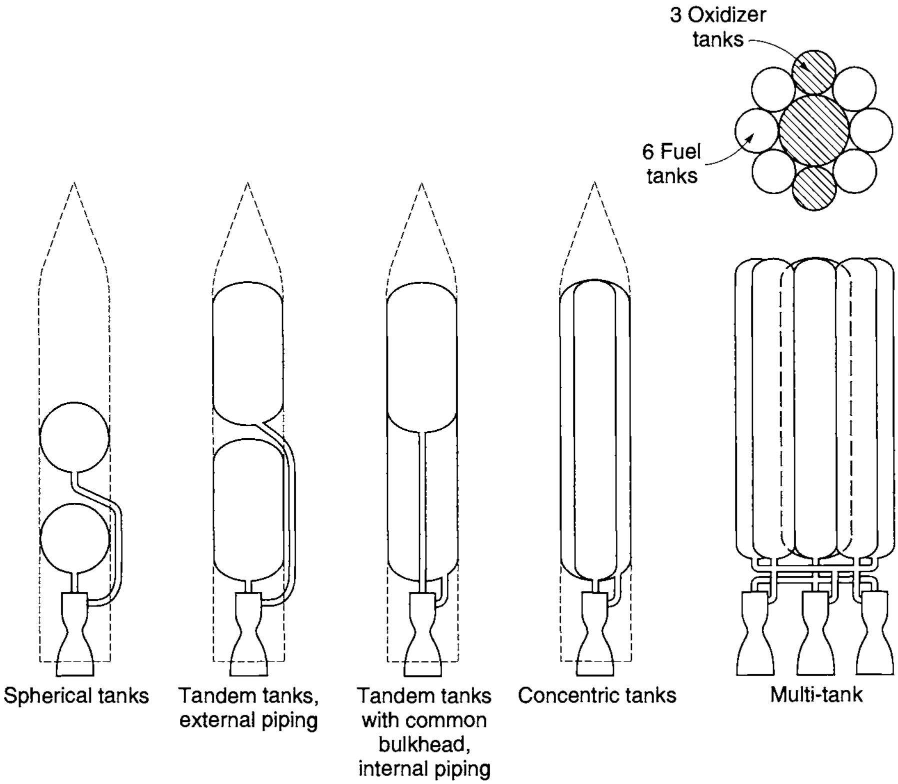


FIGURE 6–5. Typical tank arrangements for large turbopump-fed liquid propellant rocket engines.

a tank, the ullage volume (and, if it is sealed, also its pressure) will change as the bulk temperature of the propellant varies.

The *expulsion efficiency* of a tank and/or propellant piping system is the amount of propellant expelled or available divided by the total amount of propellant initially present. Typical values are 97 to 99.7%. The losses are unavailable propellants that are trapped in grooves or corners of pipes, fittings, and valves, are wetting the walls, retained by surface tension, or caught in instrument taps. This *residual propellant* is not available for combustion and must be treated as inert mass, causing the vehicle mass ratio to decrease slightly. In the design of tanks and piping systems, an effort is made to minimize the residual propellant.

The optimum shape of a propellant tank (and also a gas pressurizing tank) is spherical, because for a given volume it results in a tank with the least weight. Small spherical tanks are often used with reaction control engine systems, where they can be packaged with other vehicle equipment. Unfortunately, the larger spheres, which are needed for the principal propulsion systems, are not very efficient for using the space in a vehicle. These larger tanks are often made integral with the vehicle fuselage or wing. Most are cylindrical with half ellipses at the ends, but they can be irregular in shape. A more detailed discussion of tank pressurization is given in the next section.

Cryogenic propellants cool the tank wall temperature far below the ambient air temperature. This causes condensation of moisture on the outside of the tank and usually also formation of ice during the period prior to launch. The ice is undesirable, because it increases the vehicle inert mass and can cause valves to malfunction. Also, as pieces of ice are shaken off or break off during the initial flight, these pieces can damage the vehicle; for example, the ice from the Shuttle's cryogenic tank can hit the orbiter vehicle.

For an extended storage period, cryogenic tanks are usually thermally insulated; porous external insulation layers have to be sealed to prevent moisture from being condensed inside the insulation layer. With liquid hydrogen it is possible to liquify or solidify the ambient air on the outside of the fuel tank. Even with heavy insulation and low-conductivity structural tank supports, it is not possible to prevent the continuous evaporation of the cryogenic fluid. Even with good thermal insulation, all cryogenic propellants evaporate slowly during storage and therefore cannot be kept in a vehicle for more than perhaps a week without refilling of the tanks. For vehicles that need to be stored or to operate for longer periods, a storable propellant combination must be used.

Prior to loading very cold cryogenic propellant into a flight tank, it is necessary to remove or evacuate the air to avoid forming solid air particles or condensing any moisture as ice. These frozen particles would plug up injection holes, cause valves to freeze shut, or prevent valves from being fully closed. Tanks, piping, and valves need to be chilled or cooled down before they can contain cryogenic liquid without excessive bubbling. This is usually done by letting the initial amount of cryogenic liquid absorb the heat from the relatively warm hardware. This initial propellant is vaporized and vented through appropriate vent valves.

If the tank or any segment of piping containing low-temperature cryogenic liquid is sealed for an extended period of time, heat from ambient-temperature hardware will result in evaporation and this will greatly raise the pressure until it exceeds the strength of the container (see Ref. 6–7). This self-pressurization will cause a failure, usually a major leak or even an explosion. All cryogenic tanks and piping systems are therefore vented during storage on the launch pad, equipped with pressure safety devices (such as burst diaphragms or relief valves), and the evaporated propellant is allowed to escape from its container. For long-term storage of cryogenic propellants in space vacuum (or on the ground) some form of a powered refrigeration system is needed to recondense the vapors and minimize evaporation losses. The tanks are refilled or topped off just before launch to replace the evaporated vented propellant. When the tank is pressurized, just before launch, the boiling point is usually raised slightly and the cryogenic liquid can usually absorb the heat transferred to it during the several minutes of rocket firing.

There are several categories of tanks in liquid propellant propulsion systems:

1. For pressurized feed systems the propellant tanks typically operate at an average pressure between 1.3 and 9 MPa or about 200 to 1800 lbf/in.^{2}. These tanks have thick walls and are heavy.
2. For high-pressure gas (used to expel the propellants) the tank pressures are much higher, typically between 6.9 and 69 MPa or 1000 to 10,000 lbf/in.^{2}. These tanks are usually spherical for minimum inert mass. Several small spherical tanks can be connected together and then they are relatively easy to place within the confined space of a vehicle.
3. For turbopump feed systems it is necessary to pressurize the propellant tanks slightly (to suppress pump cavitation as explained in Section 10.1) to average values of between 0.07 and 0.34 MPa or 10 to 50 lbf/in^2. These low pressures allow thin tank walls, and therefore turbopump feed systems have relatively low tank weights.

Liquid propellant tanks can be difficult to empty under side accelerations, zero-*g*, or negative-*g* conditions during flight. Special devices and special types of tanks are needed to operate under these conditions. Some of the effects that have to be overcome are described below.

The oscillations and side accelerations of vehicles in flight can cause *sloshing* of the liquid in the tank, very similar to a glass of water that is being jiggled. In an antiaircraft missile, for example, the side accelerations can be large and can initiate sloshing. Typical analysis of sloshing can be found in Refs. 6–8 and 6–9. When the tank is partly empty, sloshing can uncover the tank outlet and allow gas bubbles to enter into the propellant discharge line. These bubbles can cause major combustion problems in the thrust chambers; the aspirating of bubbles or the uncovering of tank outlets by liquids therefore needs to be avoided. Sloshing also causes shifts in the vehicle's center of gravity and makes flight control difficult.

Vortexing can also allow gas to enter the tank outlet pipe; this phenomenon is similar to the Coriolis force effects in bath tubs being emptied and can be augmented if the vehicle spins or rotates in fight. Typically, a series of internal baffles is often used to reduce the magnitude of sloshing and vortexing in tanks with modest side accelerations. A positive expulsion mechanism can prevent gas from entering the propellant piping under multidirectional major accelerations or spinning (centrifugal) acceleration. Both the vortexing and sloshing can greatly increase the unavailable or residual propellant, and thus cause a reduction in vehicle performance.

In the gravity-free environment of space, the stored liquid will float around in a partly emptied tank and may not always cover the tank outlet, thus allowing gas to enter the tank outlet or discharge pipe. Figure 6–6 shows that gas bubbles have no orientation. Various devices have been developed to solve this problem: namely, *positive expulsion devices* and *surface tension devices*. The positive expulsion tank design include movable pistons, inflatable

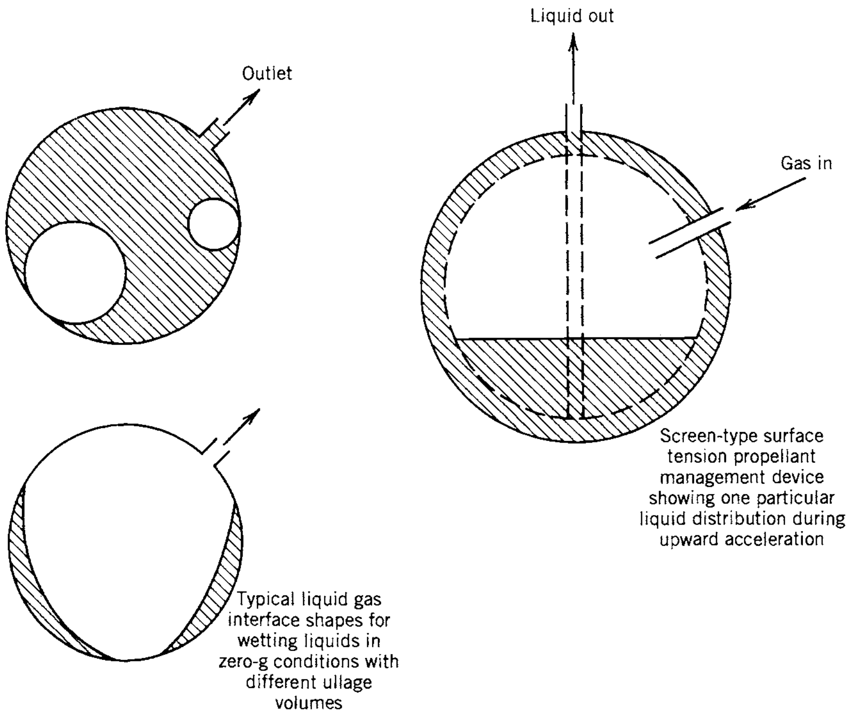


FIGURE 6–6. Ullage bubbles can float around in a zero-gravity environment; surface tension device can keep tank outlet covered with liquid.

flexible bladders, or thin movable, flexible metal diaphragms. Surface tension devices rely on surface tension forces to keep the outlet covered with liquid.

Several basic types of *positive expulsion devices* have been used successfully in propellant tanks of pressurized feed systems. They are compared in Table 6–5 and shown in Fig. 6–7 for simple tanks. These devices mechanically separate the pressurizing gas from the liquid propellant in the propellant tank. Separation is needed for these reasons:

1. It prevents pressurizing gas from dissolving in the propellant. Dissolved pressurizing gas dilutes the propellant, reduces its density as well as its specific impulse, and makes the pressurization inefficient.
2. It allows hot and reactive gases (generated by gas generators) to be used for pressurization, and this permits a reduction in pressurizing system mass and volume. The mechanical separation prevents a chemical reaction between the hot gas and the propellant, prevents gas from being dissolved in the propellant, and reduces the heat transfer to the liquid.
3. In some cases tanks containing toxic propellant must be vented without spilling any toxic liquid propellant or its vapor. For example, in servicing

TABLE 6–5. Comparison of Propellant Expulsion Methods for Spacecraft Hydrazine Tanks

Selection Criteria	Positive Expulsion Devices: Single Elastomeric Diaphragm (Hemispherical)	Inflatable Dual Elastomeric Bladder (Spherical)	Foldable Metallic Diaphragm (Hemispherical)	Piston or Bellows	Rolling Diaphragm	Surface Tension Screens
Application history	Extensive	Extensive	Limited	Extensive in high acceleration vehicles	Limited	Extensive
Weight (normalized)	1.0	1.1	1.25	1.2	1.0	0.9
Expulsion efficiency	Excellent	Good	Good	Excellent	Very good	Good or fair
Maximum side acceleration	Low	Low	Medium	High	Medium	Lowest
Control of center of gravity	Poor	Limited	Good	Excellent	Good	Poor
Long service life	Excellent	Excellent	Excellent	Very good	Unproven	Excellent
Preflight check	Leak test	Leak test	Leak test	Leak test	Leak test	None
Disadvantages	Chemical deterioration	Chemical deterioration; fits only into a few tank geometries	High-pressure drop; fits only certain tank geometries; high weight	Potential seal failure; critical tolerances on piston seal; heavy	Weld inspection is difficult; adhesive (for bonding to wall) can deteriorate)	Limited to low accelerations

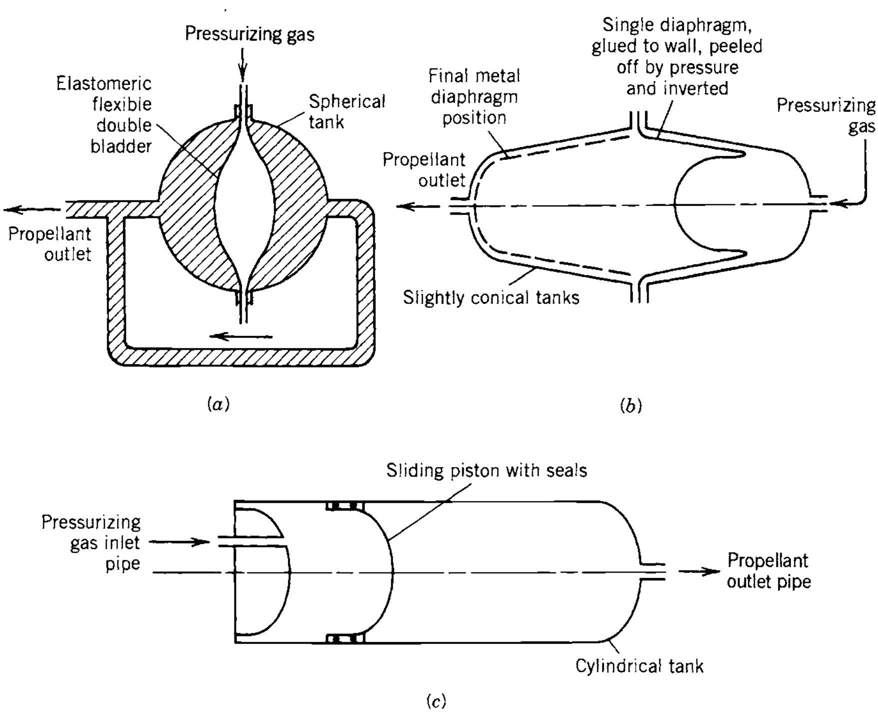


FIGURE 6–7. Three concepts of propellant tanks with positive expulsion: (a) inflatable dual bladder; (b) rolling, peeling diaphragm; (c) sliding piston. As the propellant volume expands or contracts with changes in ambient temperature, the piston or diaphragm will also move slightly and the ullage volume will change during storage.

a reusable rocket, the tank pressure needs to be relieved without venting or spilling potentially hazardous material.

A *piston expulsion* device permits the center of gravity (CG) to be accurately controlled and its location to be known. This is important in rockets with high side accelerations such as antiaircraft missiles or space defense missiles, where the thrust vector needs to go through the CG; if the CG is not well known, unpredictable turning moments may be imposed on the vehicle. A piston also prevents sloshing or vortexing.

Surface tension devices use capillary attraction for supplying liquid propellant to the tank outlet pipe. These devices (see Fig. 6–6) are often made of very fine (300 mesh) stainless steel wire woven into a screen and formed into tunnels or other shapes (see Refs. 6–10 and 6–11). These screens are located near the tank outlet and, in some tanks, the tubular galleries are designed to connect various parts of the tank volume to the outlet pipe sump. These devices work best in a relatively low-acceleration environment, when surface tension forces can overcome the inertia forces.

The combination of surface tension screens, baffles, sumps, and traps is called a *propellant management device*. Although not shown in any detail, they are included inside the propellant tanks of Figs. 6–6 and 6–13.

High forces can be imposed on the tanks and thus on the vehicle by strong sloshing motions of the liquid and also by sudden changes in position of liquid mass in a partly empty tank during a gravity-free flight when suddenly accelerated by a relatively large thrust. These forces can be large and can cause tank failure. The forces will depend on the tank geometry, baffles, ullage volume, and its initial location and the acceleration magnitude and direction.

6.5. TANK PRESSURIZATION

Subsystems for pressurizing tanks are needed for both of the two types of feed systems, namely pressure feed systems and pump feed systems. The tank pressures for the first type are usually between 200 and 1800 psi and for the second between 10 and 50 psig. Refs. 6–1, 6–3 to 6-5 give further descriptions. Inert gases such as helium or nitrogen are the most common method of pressurization. In pump feed systems a small positive pressure in the tank is needed to suppress pump cavitation. For cryogenic propellants this has been accomplished by heating and vaporizing a small portion of the propellant taken from the high-pressure discharge of the pump and feeding it into the propellant tank, as shown in Fig. 1–4. This is a type of low-pressure gas feed system.

The pressurizing gas must not condense, or be soluble in the liquid propellant, for this can greatly increase the mass of required pressurant and the inert mass of its pressurization system hardware. For example, nitrogen pressurizing gas will dissolve in nitrogen tetroxide or in liquid oxygen and reduce the concentration and density of the oxidizer. In general, about $2\frac{1}{2}$ times as much nitrogen mass is needed for pressurizing liquid oxygen if compared to the nitrogen needed for displacing an equivalent volume of water at the same pressure. Oxygen and nitrogen tetroxide are therefore usually pressurized with helium gas, which dissolves only slightly. The pressurizing gas must not react chemically with the liquid propellant. Also, the gas must be dry, since moisture can react with some propellants or dilute them.

The pressurizing gas above a cryogenic liquid is usually warmer than the liquid. The heat transfer to the liquid cools the gas and that increases the density; therefore a larger mass of gas is needed for pressurization even if none of the gas dissolves in the liquid propellant. If there is major sloshing and splashing in the tank during flight, the gas temperature can drop quickly, causing irregularities in the tank pressure.

Chemical pressurization permits the injection of a small amount of fuel or other suitable spontaneously ignitable chemical into the oxidizer tank (or vice versa) which creates the pressurizing gas by combustion inside the propellant tank. While ideally this type of pressurization system is very small and light, in practice it has not usually given reproducible tank pressures, because of irregular combustion the sloshing of propellant in the tank during vehicle maneuvers has caused sudden cooling of the hot pressurizing gas and thus some erratic tank pressure changes. This problem can be avoided by physically separating the hot reactive gas from the liquid propellant by a piston or a flexible bladder. If hot gas from a solid propellant gas generator of from the decomposition of a monopropellant is used (instead of a high-pressure gas supply), a substantial reduction in the gas and inert mass of the pressurizing system can be achieved. For example, the pressurizing of hydrazine monopropellant by warm gas (from the catalytic decomposition of hydrazine) has been successful for moderate durations.

The prepackaged compact experimental liquid propellant rocket engine shown in Fig. 6–8 is unique. It uses a gelling agent to improve propellant safety and density (see Section 7.5 and Ref. 7–11), a solid propellant for pressurization of propellant tanks, two concentric annular pistons (positive expulsion), and a throttling and multiple restart capability. It allows missiles to lock on to targets before or after launch, slow down and search for targets, loiter, maneuver, or speed up to a high terminal velocity. This particular experimental engine, developed by TRW, has been launched from a regular Army mobile launcher.

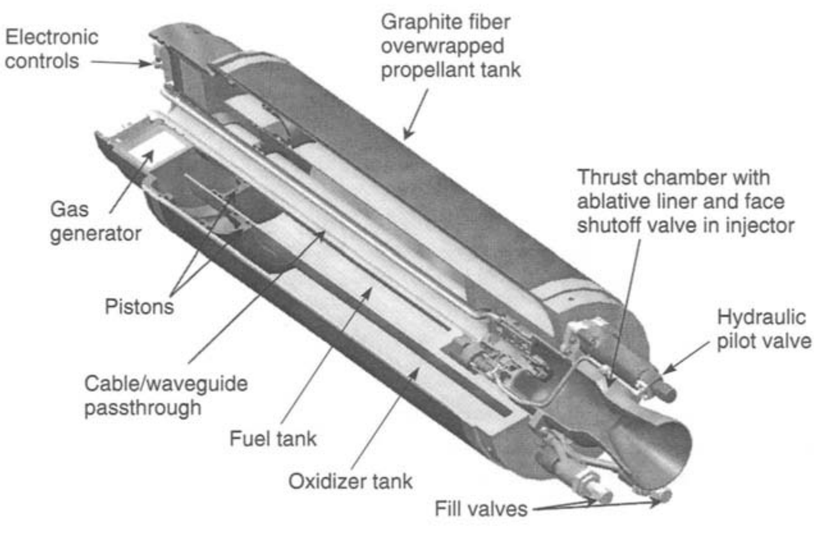


FIGURE 6–8. Simplified diagram of a compact pre-loaded, pressure-fed, bipropellant experimental rocket engine aimed at propelling smart maneuvering ground-to-ground missiles. It uses gelled red fuming nitric acid and gelled monomethylhydrazine as propellants. A solid propellant gas generator provides the gas for tank pressurization and the hot gases are isolated from the propellants by pistons. The concentric spray injector allows restart, throttling, and flow shut-off at the injector face. The rocket engine is 6 in. diameter and 23.5 in. long. (Courtesy of Space and Electronics Group, TRW, Inc.)

Estimating the Mass of the Pressurizing Gas

The major function of the pressurizing gas is to expel the propellants from their tanks. In some propulsion system installations, a small amount of the pressurized gas also performs other functions such as the operation of valves and controls. The first part of the gas leaving the high-pressure-gas storage tank is at ambient temperature. If the high-pressure gas expands rapidly, then 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 are very much colder than the ambient temperature and readily absorb heat from the piping and the tank walls. The Joule–Thomson effect causes a further small temperature change.

A *simplified analysis* of the pressurization of a propellant tank can be made on the basis of the conservation of energy principle by assuming an adiabatic process (no heat transfer to or from the walls), an ideal gas, and a negligibly small initial mass of gas in the piping and the propellant tank. Let the initial condition in the gas tank be given by subscript 0 and the instantaneous conditions in the gas tank by subscript g and in the propellant tank by subscript p. The gas energy after and before propellant expulsion is

$$m_g c_v T_g + m_p c_v T_p + p_p V_p = m_0 c_v T_0 \tag{6–5}$$

The work done by the gas in displacing the propellants is given by $p_p V_p$. Using Eqs. 3–3 to 3–5, the initial storage gas mass m_0 may be found.

$$\begin{gathered} c_v p_g V_0/R + c_v p_p V_p/R + p_p V_p = m_0 c_v T_0 \\ m_0 = (p_g V_0 + p_p V_p k)/(RT_0) \end{gathered} \tag{6–6}$$

This may be expressed as

$$m_0 = \frac{p_g m_0}{p_0} + \frac{p_p V_p}{RT_0} k = \frac{p_p V_p}{RT_0} \left(\frac{k}{1 - p_g/p_0} \right) \tag{6–7}$$

The first term in this equation expresses the mass of gas required to empty a completely filled propellant tank if the gas temperature is maintained at the initial storage temperature T_0. The second term expresses the availability of the storage gas as a function of the pressure ratio through which the gas expands.

Heating of the pressurizing gas reduces the storage gas and tank mass requirements and can be accomplished by putting a heat exchanger into the gas line. Heat from the rocket thrust chamber, the exhaust gases, or from other devices can be used as the energy source. The reduction of storage gas mass depends largely on the type and design of the heat exchanger and the duration.

If the expansion of the high-pressure gas proceeds slowly (e.g., with an attitude control propulsion system with many short pulses over a long period of time), then the gas expansion comes close to an isothermal process; heat is absorbed from the vehicle and the gas temperature does not decrease appreciably. Here $T_0 = T_g = T_p$. The actual process is between an adiabatic and an isothermal process and may vary from flight to flight.

The heating and cooling effects of the tank and pipe walls, the liquid propellants, and the values on the pressurizing gas require an iterative analysis. The effects of heat transfer from sources in the vehicle, changes in the mission profile, vaporization of the propellant in the tanks, and heat losses from the tank to the atmosphere or space have to be included and the analyses can become quite complex. The design of storage tanks therefore allows a reasonable excess of pressurizing gas to account for these effects, for ambient temperature variations, and for the absorption of gas by the propellant. Equation 6–7 is therefore valid only under ideal conditions.

Example 6–2. What air tank volume is required to pressurize the propellant tanks of a 9000-N thrust rocket thrust chamber using 90% hydrogen peroxide as a monopropellant at a chamber pressure of 2.00 MPa for 30 sec in conjunction with a solid catalyst? The air tank pressure is 14 MPa and the propellant tank pressure is 3.0 MPa. Allow for 1.20% residual propellant.

SOLUTION. The exhaust velocity is 1300 m/sec and the required propellant flow can be found from Eq. 3–42 ($\zeta_d = 1.06$):

$$\dot{m} = \zeta_d F/c = 1.06 \times 9000/1300 - 7.34 \text{ kg/sec}$$

The total propellant required is $m = 7.34$ kg/sec $\times$ 30 sec $\times 1.012 = 222.6$ kg. The density of 90% hydrogen peroxide is 1388 kg/m^3. The propellant volume is $222.6/1388 = 0.160$ m^3. With 5% allowed for ullage and excess propellants, Eq. 6–7 gives the required weight of air ($R = 289$ J/kg-K; $T_0 = 298$ K; $k = 1.40$) for displacing the liquid.

$$\begin{aligned} m_0 &= \frac{p_p V_p}{R T_0} \frac{k}{[1 - (p_g/p_0)]} = \frac{3.0 \times 10^6 \times 0.16 \times 1.05 \times 1.4}{289 \times 298 \times [1 - (3/14)]} \\ &= 10.4 \text{ kg of compressed air} \end{aligned}$$

With an additional 5% allowed for excess gas, the high-pressure tank volume will be

$$\begin{aligned} V_0 &= m_0 R T_0/p_0 = 1.05 \times 10.4 \times 289 \times 298/(14 \times 10^6) \\ &= 0.067 \text{ m}^3. \end{aligned}$$

6.6. TURBOPUMP FEED SYSTEMS AND ENGINE CYCLES

The principal components of a rocket engine with one type of turbopump system are shown in the simplified diagram of Fig. 1–4. Here the propellants are pressurized by means of *pumps*, which in turn are driven by *turbines*. These turbines derive their power from the expansion of hot gases. Engines with turbopumps are preferred for booster and sustainer stages of space launch vehicles, long-range missiles, and in the past also for aircraft performance augmentation. They are usually lighter than other types for these high thrust, long duration applications. The inert hardware mass of the rocket engine (without tanks) is essentially independent of duration. Examples can be seen In Figs. 6–1 and 6–9 and also in Refs. 6–1, 6–2, and 6–6. For aircraft performance augmentation the rocket pump can be driven directly by the jet engine, as in Ref. 6–12. From the turbopump feed system options depicted in Fig. 6–2, the designer can select the most suitable concept for a particular application.

An *engine cycle* for turbopump-fed engines describes the specific propellant flow paths through the major engine components, the method of providing the hot gas to one or more turbines, and the method of handling the turbine exhaust gases. There are *open cycles* and *closed cycles*. *Open* denotes that the working fluid exhausting from the turbine is discharged overboard, after having been expanded in a nozzle of its own, or discharged into the nozzle of the thrust chamber at a point in the expanding section far downstream of the nozle throat. In *closed cycles* or *topping cycles* all the working fluid from the turbine is injected into the engine combustion chamber to make the most efficient use of its remaining energy. In closed cycles the turbine exhaust gas is expanded through the full pressure ratio of the main thrust chamber nozzle, thus giving a little more performance than the open cycles, where these exhaust gases expand only through a relatively small pressure ratio. The overall engine performance difference is typically between 1 and 8% of specific impulse and this is reflected in even larger differences in vehicle performance.

Figure 6–9 shows the three most common cycles in schematic form. Reference 6–13 shows variations of these cycles and also other cycles. The gas generator cycle and the staged combustion cycle can use most of the common liquid propellants. The expander cycle works best with vaporized cryogenic hydrogen as the coolant for the thrust chamber, because it is an excellent heat absorber and does not decompose. The schematic diagrams of Fig. 6–9 show each cycle with a separate turbopump for fuel and for oxidier. However, an arrangement with the fuel and oxdizer pump driven by the same turbine is also feasible and sometimes reduces the hardware mass, volume, and cost. The "best" cycle has to be selected on the basis of the mission, the suitability of existing engines, and the criteria established for the particular vehicle. There is an optimum chamber pressure and an optimum mixture ratio for each application, engine cycle, or optimization criterion, such as maximum range, lowest cost, or highest payload.

In the *gas generator cycle* the turbine inlet gas comes from a separate gas generator. Its propellants can be supplied from separate propellant tanks or can be bled off the main propellant feed system. This cycle is relatively simple; the pressures in the liquid pipes and pumps are relatively low (which reduces inert engine mass). It has less engine-specific impulse than an expander cycle or a staged combustion cycle. The pressure ratio across the turbine is relatively

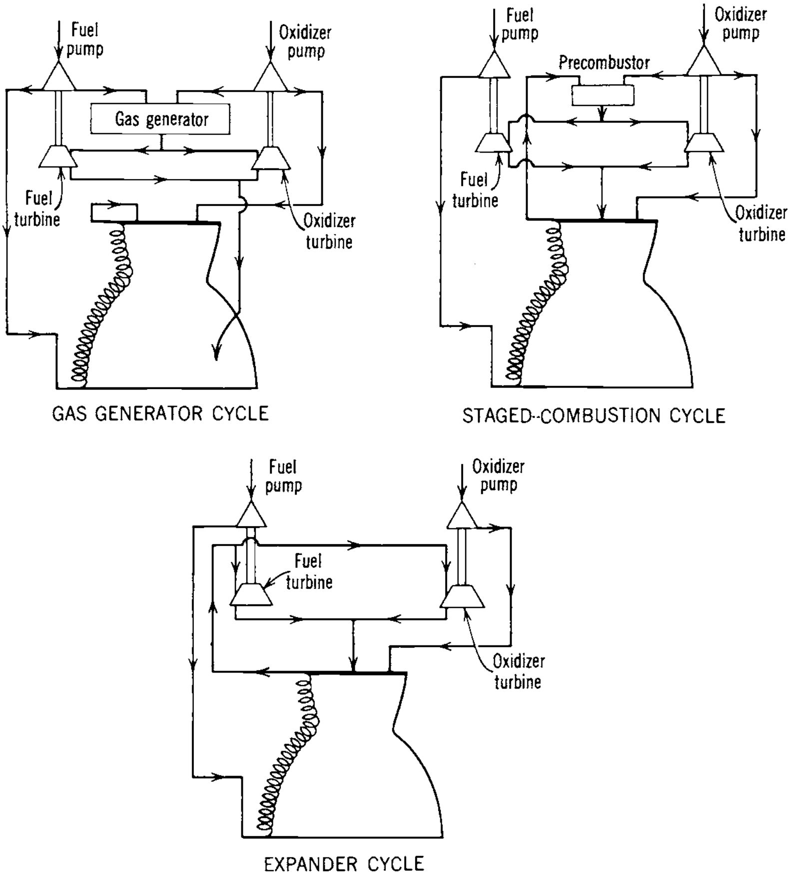


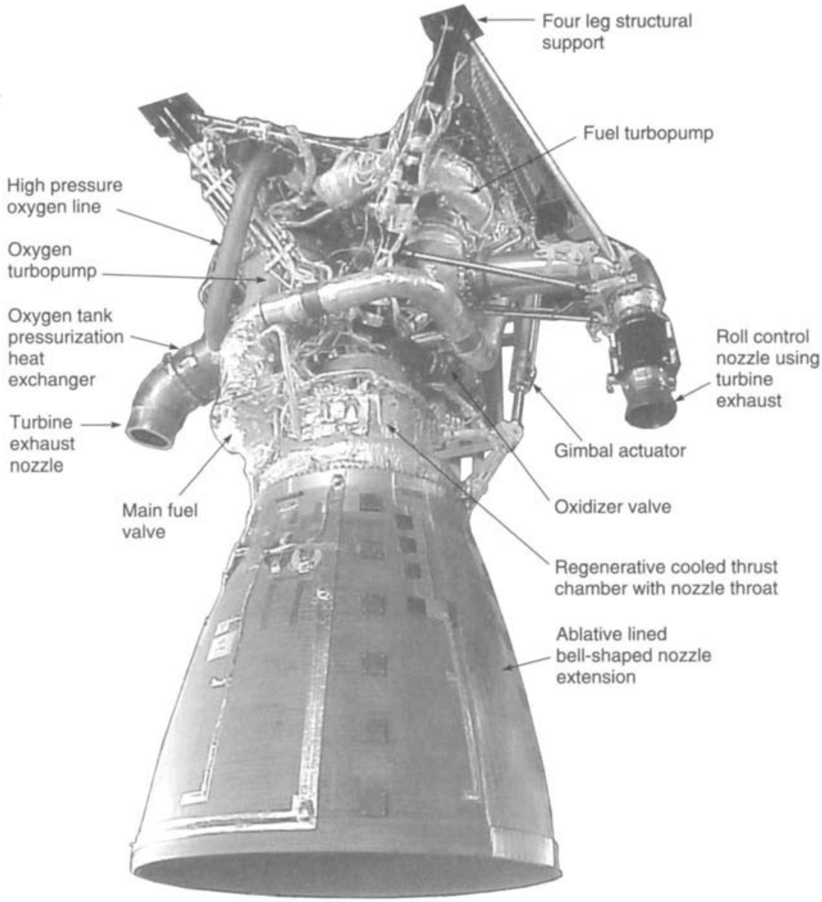
FIGURE 6–9. Simplified diagrams of three engine cycles for liquid propellant rocket engines. The spirals are a symbol for an axisymmetric cooling jacket where heat is absorbed.

high, but the turbine or gas generator flow is small (1 to 4% of total propellant flow) if compared to closed cycles. Some early engines used a separate monopropellant for creating the generator gas. The German V-2 missile engine used hydrogen peroxide, which was decomposed by a catalyst. Typically, the turbine exhaust gas is discharged overboard through one or two separate small low-area-ratio nozzles (at relatively low specific impulse), as shown schematically in Fig. 1–4 and in the Vulcain engine or RS-68 engine listed in Table 10-3. Alternatively, this turbine exhaust can be aspirated into the main flow through openings in the diverging nozzle section, as shown schematically in Fig. 6–9. This gas then protects the walls near the nozzle exit from high temperatures. Both methods can provide a small amount of additional thrust. The gas generator mixture ratio is usually fuel rich (in some engine it is oxidizer rich) so that the gas temperatures are low enough (typically 900 to 1350 K) to allow the use of uncooled turbine blades and uncooled nozzle exit segments. The RS-68 rocket engine, shown in Fig. 6–10, has a simple gas generator cycle. This engine is the largest liquid hydrogen/liquid oxygen rocket engine built to date. As can be seen from the data in the figure, with a gas generator cycle the specific impulse of the thrust chamber by itself is always a little higher than that of the engine and the thrust of the thrust chamber is always slightly lower than that of the engine.

In the *expander cycle* most of the engine coolant (usually hydrogen fuel) is fed to low-pressure-ratio turbines after having passed through the cooling jacket where it picked up energy. Part of the coolant, perhaps 5 to 15%, bypasses the turbine (not shown in Fig. 6-9) and rejoins the turbine exhaust flow before the entire coolant flow is injected into the engine combustion chamber where it mixes and burns with the oxidizer (see Refs. 6–2 and 6–14). The primary advantages of the expander cycle are good specific impulse, engine simplicity, and relatively low engine mass. In the expander cycle all the propellants are fully burned in the engine combustion chamber and expanded efficiently in the engine exhaust nozzle.

This cycle is used in the RL10 hydrogen/oxygen rocket engine, and different versions of this engine have flown successfully in the upper stages of several space launch vehicles. Data on the RL10-A3-3A are given in Table 10-3. A recent modification of this engine, the RL10B-2 with an extendible nozzle skirt, can be seen in Fig. 8–19 and data on this engine are contained in Table 8–1. It delivers the highest specific impulse of any chemical rocket engine to date. The RL10B-2 flow diagram in Fig. 6–11 shows its expander cycle. Heat absorbed by the thrust chamber cooling jacket gasifies and raises the gas temperature of the hydrogen so that it can be used to drive the turbine, which in turn drives a single-stage liquid oxygen pump (through a gear case) and a two-stage liquid hydrogen pump. The cooling down of the hardware to cryogenic temperatures is accomplished by flowing (prior to engine start) cold propellant through cooldown valves. The pipes for discharging the cooling propellants overboard are not shown here, but can be seen in Fig. 8–19. Thrust is regulated by controlling the flow of hydrogen gas to the turbine, using a bypass to maintain constant chamber pressure. Helium is used as a means of power boost by actuating several of the larger valves through solenoid-operated pilot valves.

In the *staged combustion cycle*, the coolant flow path through the cooling jacket is the same as that of the expander cycle. Here a high-pressure precombustor (gas generator) burns all the fuel with part of the oxidizer to provide high-energy gas to the turbines. The total turbine exhaust gas flow is injected into the main combustion chamber where it burns with the remaining oxidizer. This cycle lends itself to high-chamber-pressure operation, which allows a small thrust chamber size. The extra pressure drop in the precombustor and turbines causes the pump discharge pressures of both the fuel and the oxidizer to be higher than with open cycles, requiring heavier and more



Parameter	Thrust chamber	Engine
Specific impulse at sea level (max.), sec	368	362
Specific impulse in vacuum (max.), sec	421	415
Thrust, at sea level, lbf	640,700	650,000
Thrust in vacuum lbf	732,400	745,000
Mixture ratio	6.74	6.0

FIGURE 6–10. Simplified view of the RS–68 rocket engine with a gas generator cycle. For engine data see Table 10–3. (Courtesy of The Boeing Company, Rocketdyne Propulsion and Power.)

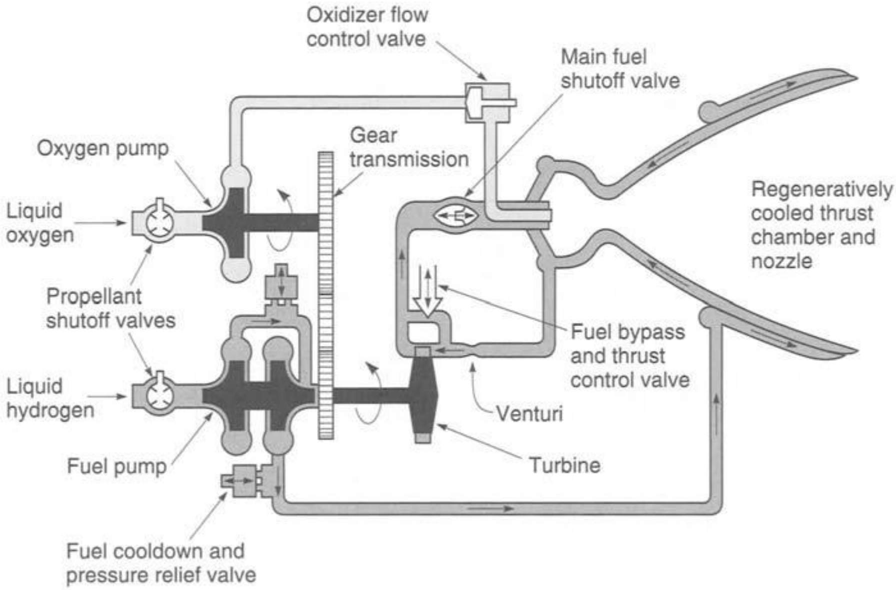


FIGURE 6–11. Schematic flow diagram of the RL10B-2 upper stage rocket engine. For data see Table 8–1. (Courtesy of Pratt & Whitney, a division of United Technologies.)

complex pumps, turbines, and piping. The turbine flow is relatively high and the turbine pressure drop is low, when compared to an open cycle. The staged combustion cycle gives the highest specific impulse, but it is more complex and heavy. In contrast, an open cycle can allow a relatively simple engine, lower pressures, and can have a lower production cost. A variation of the staged combustion cycle is used in the Space Shuttle main engine, as shown in Figs. 6–1 and 6–12. This engine actually uses two separate precombustion chambers, each mounted directly on a separate main turbopump. In addition, there are two more turbopumps for providing a boost pressure to the main pumps, but their turbines are not driven by combustion gases; instead, high-pressure liquid oxygen drives one booster pump and evaporated hydrogen drives the other. The injector of this reusable liquid propellant high-pressure engine is shown in Fig. 9–6 and performance data are given in Tables 10–1 and 10–3. While the space shuttle main engine (burning hydrogen with oxygen) has fuel-rich preburners, oxidizer-rich preburners are used in the RD120 engine (kerosene/oxygen) and other Russian rocket engines. See Table 10–5. Another example of a staged combustion cycle is the Russian engine RD253; all of the nitrogen tetroxide oxidizer and some of the unsymmetrical dimethyl hydrazine fuel are burned in the precombustor, and the remaining fuel is injected directly into the main combustion chamber, as shown in Table 10–5.

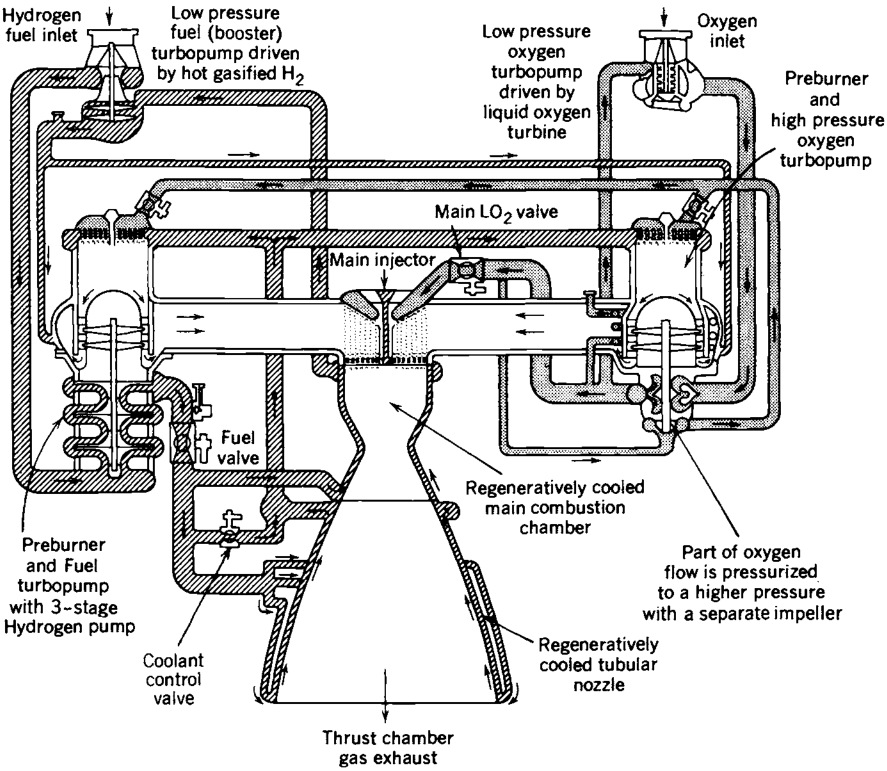


FIGURE 6–12. Flow diagram for the staged combustion cycle of the Space Shuttle Main Engine (SSME) using liquid oxygen and a liquid hydrogen fuel. (Courtesy of The Boeing Company, Rocketdyne Propulsion and Power.)

6.7. FLOW AND PRESSURE BALANCE

From an inspection of the schematic flow diagram of an engine with a gas generator in Fig. 1–4, the following basic feed system relationships are readily deduced. The flow through both pumps $\dot{m}_f$ and $\dot{m}_o$ must equal the respective propellant flow through the gas generator $\dot{m}_{gg}$ and one or more thrust chambers $\dot{m}_c$. With some cycles $\dot{m}_{gg}$ is zero. See equation on Section 10–2.

$$\dot{m}_o = (\dot{m}_o)_{gg} + (\dot{m}_o)_c \tag{6–8}$$

$$\dot{m}_f = (\dot{m}_f)_{gg} + (\dot{m}_f)_c$$

$$\dot{m}_c = (\dot{m}_o)_c + (\dot{m}_f)_c \tag{6–9}$$

$$\dot{m}_{gg} = (\dot{m}_o)_{gg} + (\dot{m}_f)_{gg} \tag{6–10}$$

In the turbopump the torques, powers, and shaft speeds must match. The balance of shaft speeds N can be simply written as

$$N_t = a_o N_o = a_f N_f \tag{6–11}$$

where a_o and a_f are gear ratios. If no gears are used, $a_o = a_f = 1$. The power balance implies that the power of turbine P_T equals the power consumed by pumps and auxiliaries. The power is expressed as the product of torque L and shaft speed N:

$$P_T = L_T N_T = L_o N_o + L_f N_f + P_b \tag{6–12}$$

where P_b represents the bearing, seal, friction, and transmission power losses. If there are no gears in a particular turbopump, then

$$N_T = N_o = N_f \tag{6–13}$$

$$L_T = L_o + L_f + L_b \tag{6–14}$$

The pressure balance equations for the fuel line at a point downstream of the fuel pump can be written as

$$\begin{aligned}(p_f)_d &= (p_f)_s + (\Delta p)_{\text{pump}} \\ &= (\Delta p)_{\text{main fuel system}} + p_1 \\ &= (\Delta)_{\text{generator fuel system}} + p_{gg}\end{aligned} \tag{6–15}$$

Here the fuel pump discharge pressure $(p_f)_d$ equals the fuel pump suction pressure $(p_f)_s$ plus the pressure rise across the pump $(\Delta p)_{\text{pump}}$; this in turn equals the chamber pressures p_1 plus all the pressure drops in the main fuel system downstream of the pump, and this is further equal to the chamber pressure in the gas generator combustion chamber p_{gg} augmented by all the pressure losses in the fuel piping between the generator and the downstream side of the fuel pump. The pressure drop in the main fuel system usually includes the losses in the cooling jacket and the pressure decrease in the injector. Equations 6–8 to 6–15 relate to a steady-state condition. A similar pressure balance is needed for the oxidizer flow. The transients and the dynamic change conditions are rather complex but have been analyzed using iterative procedures and digital computers.

6.8. ROCKET ENGINES FOR MANEUVERING, ORBIT ADJUSTMENTS, OR ATTITUDE CONTROL

These engines have usually a set of small thrusters, that are installed at various places in a vehicle, and a common pressurized feed system, similar to Figures

1–3, 4–13, or 6–13. They are called *reaction control systems* or *auxiliary rockets* as contrasted to higher-thrust *primary or boost propulsion systems* in Table 6–1. Most use storable liquid propellants, require a highly accurate repeatability of pulsing, a long life in space, and/or a long-term storage with loaded propellants in flight tanks. Figure 4–13 shows that it requires 12 thrusters for the application of pure torques about three vehicle axes. If a three-degree-of-rotation freedom is not a requrement, or if torques can be combined with some translation maneuvers, fewer thrusters will be needed. These *auxiliary rocket engines* are commonly used in spacecraft or missiles for the accurate *control of flight trajectories*, *orbit adjustments*, or *attitude control* of the vehicle. References 6–1 and 6–2 give information on several of these. Figure 6–13 shows a simplified flow diagram for a post-boost control rocket engine, with one larger rocket thrust chamber for changing the velocity vector and eight small thrusters for attitude control.

Section 4.6 describes various space trajectory correction maneuvers and satellite station-keeping maneuvers that are typically performed by these small auxiliary liquid propellant rocket engines with multiple thrusters. Attitude control can be provided both while a primary propulsion system (of a vehicle or of a stage) is operating and while its auxiliary rocket system operates by itself. For instance, this is done to point satellite's telescope into a specific orientation or to rotate a spacecraft's main thrust chamber into the desired direction for a vehicle turning maneuver.

A good method for achieving accurate velocity corrections or precise angular positions is to use pure modulation, that is, to fire some of the thrusters in a *pulsing mode* (for example, fire repeatedly for 0.020 sec, each time followed by a pause of perhaps 0.020 to 0.100 sec). The guidance system determines the maneuver to be undertaken and the vehicle control system sends command signals to specific thrusters for the number of pulses needed to accomplish this maneuver. Small liquid propellant engine systems are uniquely capable of these pulsing operations. Some thrusters have been tested for more than 300,000 pulses. For very short pulse durations the specific impulse is degraded by 5 to 25%, because the performance during the thrust build-up and thrust decay period (at lower chamber pressure) is inferior to operating only at the rated chamber pressure and the transient time becomes a major portion of the total pulse time.

Ballistic missile defense vehicles usually have highly maneuverable upper stages. These require substantial side forces (200 to 6000 N) during the final closing maneuvers just prior to reaching the target. In concept the system is similar to that of Fig. 6–13, except that the larger thrust chamber would be at right-angles to the vehicle axis. A similar system for terminal maneuvers, but using solid propellants, is shown in Fig. 11-28.

The Space Shuttle performs its reaction control with 38 different thrusters, as shown schematically in Figs. 1–13 and 6–4; this includes several duplicate (spare or redundant) thrusters. Selected thrusters are used for different maneuvers, such as space orbit corrections, station keeping, or positioning the

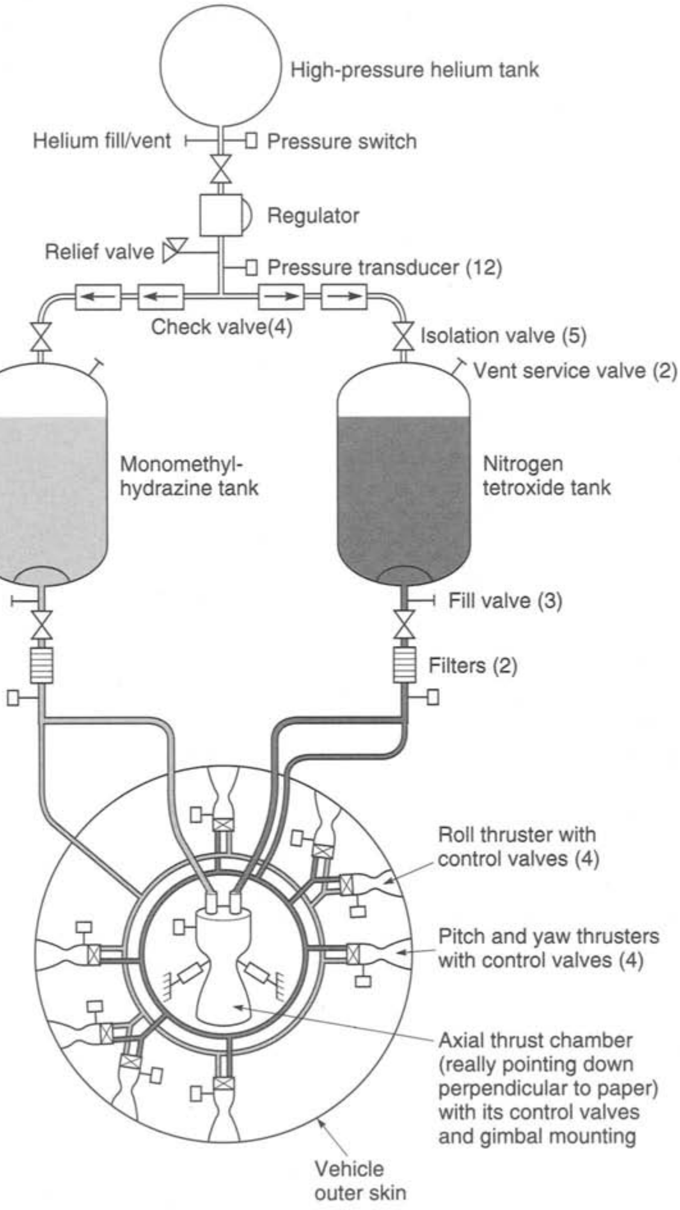


FIGURE 6–13. Schematic flow diagram of the helium-pressurized, bipropellant rocket engine system of the fourth stage of the Peacekeeper ballistic missile, which provides the terminal velocity (in direction and magnitude) to each of several warheads. It has one larger gimballed thrust chamber for trajectory translation maneuvers and eight small thrusters (with scarfed nozzles) for attitude control in pitch, yaw, and roll. (Courtesy of USAF.)

Space Shuttle for reentry or visual observations. These small restartable rocket engines are also used for space *rendezvous* or *docking maneuvers*, where one spacecraft slowly approaches another and locks itself to the other, without causing excessive impact forces during this docking manuever. This docking operation requires rotational and translational maneuvers from a series of rocket engines.

Broadly, the application of pure torque to spacecraft can be divided into two classes, *mass expulsion* types (rockets) and *nonmass expulsion* types. Nonmass expulsion types include momentum storage, gravity gradient, solar radiation, and magnetic systems. Some space satellites are equipped with both the mass and nonmass expulsion types. *Reaction wheels* or flywheels, a momentum storage device, are particularly well suited to obtaining vehicle angular position control with high accuracies of less than 0.01° deviation and low vehicle angular rates of less than 10^{-5} degrees/sec with relatively little expenditure of energy. The vehicle angular momentum is changed by accelerating (or decelerating) the wheel. Of course, when the wheel speed reaches the maximum (or minimum) permissible, no further electrical motor torquing is possible; the wheel must be decelerated (or accelerated) to have its momentum removed (or augmented), a function usually accomplished through the simultaneous use of small attitude control rockets, which apply a torque to the vehicle in the opposite direction.

The propellants for *auxiliary rockets* fall into three categories: cold gas jets (also called inert gas jets), warm or heated gas jets, and chemical combustion rockets, such as bipropellant liquid propellant rockets. The specific impulse is typically 50 to 120 sec for cold gas systems and 105 to 250 sec for warm gas systems. Warm gas systems can use inert gas with an electric heater or a monopropellant which is catalytically and/or thermally decomposed. Bipropellant attitude control thrust chambers allow an I_s of 220 to 325 sec and have varied from 5 to 4000 N thrust; the highest thrusts apply to large spacecraft. All basically use pressurized feed systems with multiple thrusters or thrust chambers equipped with fast-acting, positive-closing precision valves. Many systems use small, uncooled, metal-constructed supersonic exhaust nozzles strategically located on the periphery of the spacecraft. Gas jets are used typically for low thrust (up to 10 N) and low total impulse (up to 4000 N-sec). They have been used on smaller satellites and often only for roll control.

Small liquid monopropellant and liquid bipropellant rocket units are common in auxiliary rocket systems for thrust levels typically above 2 N and total impulse values above 3000 N-sec. Hydrazine is the most common monopropellant used in auxiliary control rockets; nitrogen tetroxide and monomethylhydrazine is a common bipropellant combination. The next chapter contains data on all three categories of these propellants, and Chapter 10 shows diagrams of small auxiliary rocket engines and their thrusters.

Combination systems are also in use. Here a bipropellant with a relatively high value of I_s, such as N_2O_4 and N_2H_4, is used in the larger thrusters, which consume most of the propellant; then several simple monopropellant thrusters (with a lower I_s), used for attitude control pulsing, usually consume a relatively small fraction of the total fuel. Another combination system is to employ bipropellant or monopropellant thrusters for adding a velocity increment to a flight vehicle or to bleed or pulse some of the pressurizing gas, such as helium, through small nozzles controlled by electromagnetic valves to provide roll control. The specific mission requirements need to be analyzed to determine which type or combination is most advantageous for a particular application.

Special thruster designs exist which can be used in a bipropellant mode at higher thrust and also in a monopropellant mode for lower thrust. This can offer an advantage in some spacecraft applications. An example is the TRW secondary combustion augmented thruster (SCAT), which uses hydrazine and nitrogen tetroxide, is restartable, vaporizes the propellants prior to injection and therefore has very efficient combustion (over 99%), can operate over a wide range of mixture ratios, and can be throttled from 5 to 15 lbf thrust.

6.9. VALVES AND PIPE LINES

Valves control the flows of liquids and gases and pipes conduct these fluids to the intended components. There are no rocket engines without them. There are many different types of valves. All have to be reliable, lightweight, leakproof, and must withstand intensive vibrations and very loud noises. Table 6–6 gives several key classification categories for rocket engine valves. Any one engine will use only some of the valves listed here.

The art of designing and making valves is based, to a large extent, on experience. A single chapter cannot do justice to it by describing valve design and operation. References 6–1 and 6–2 decribe the design of specific valves, lines, and joints. Often the design details, such as clearance, seat materials, or opening time delay present development difficulties. With many of these valves, any leakage or valve failure can cause a failure of the rocket unit itself. All valves are tested for two qualities prior to installation; they are tested for leaks—through the seat and also through the glands—and for functional soundness or performance.

The propellant valves in high thrust units handle relatively large flows at high service pressures. Therefore, the forces necessary to actuate the valves are large. Hydraulic or pneumatic pressure, controlled by pilot valves, operates the larger valves; these pilot valves are in turn actuated by a solenoid or a mechanical linkage. Essentially this is a means of power boost.

TABLE 6–6. Classification of Valves Used in Liquid Propellant Rocket Engines

1. *Fluid*: fuel; oxidizer; cold pressurized gas; hot turbine gas.
2. *Application or Use*: main propellant control; thrust chamber valve (dual or single); bleed; drain; fill; by-pass; preliminary stage flow; pilot valve; safety valve; overboard dump; regulator; gas generator control; sequence control; isolation of propellant or high-pressure gas prior to start.
3. *Mode of Actuation*: automatically operated (by solenoid, pilot valve, trip mechanism, pyrotechnic, etc.); manually operated; pressure-operated by air, gas, propellant, or hydraulic fluid (e.g., check valve, tank vent valve, pressure regulator, relief valve), with or without position feedback, rotary or linear actuator.
4. The *flow* magnitude determines the *size* of the valve.
5. *Duty cycle*: single or multiple pulse operation; reusable for other flights; long or short life.
6. *Valve Type*: normally open; normally closed; normally partly open; two-way; three-way, with/without valve position feedback; ball valve, gate valve, butterfly type, spring loaded.
7. *Temperature* and *pressure* allow classification by high, low, or cryogenic temperature fluids, or high or low pressure or vacuum capability.
8. *Accessible or not accessible* to inspection, servicing, or replacement of valve or its seal.

Two valves commonly used in pressurized feed systems are *isolation valves* (when shut, they isolate or shut off a portion of the propulsion system) and *latch valves*; they require power for brief periods during movements, such as to open or shut, but need no power when latched or fastened into position.

A very simple and very light valve is a *burst diaphragm*. It is essentially a circular disk of material which blocks a pipeline and is designed so that it will fail and burst at a predetermined pressure differential. Burst diaphragms are positive seals and prevent leakage, but they can be used only once. The German *Wasserfall* antiaircraft missile used four burst disks; two were in high pressure air lines and two were in the propellant lines.

Figure 6–14 shows a main liquid oxygen valve. It is normally closed, rotary actuated, cryogenic, high pressure, high flow, reusable ball valve, allowing continuous throttling, a controlled rate of opening through a crank and hydraulic piston (not shown), with a position feedback and anti-icing controls.

Pressure regulators are special valves which are used frequently to regulate gas pressures. Usually the discharge pressure is regulated to a predetermined standard pressure value by continuously throttling the flow, using a piston, flexible diaphragm, or electromagnet as the actuating mechanism. Regulators can be seen in Figs. 1–3 and 6–13.

The various fluids in a rocket engine are conveyed by *pipes* or *lines*, usually made of metal and joined by fittings or welds. Their design must provide for

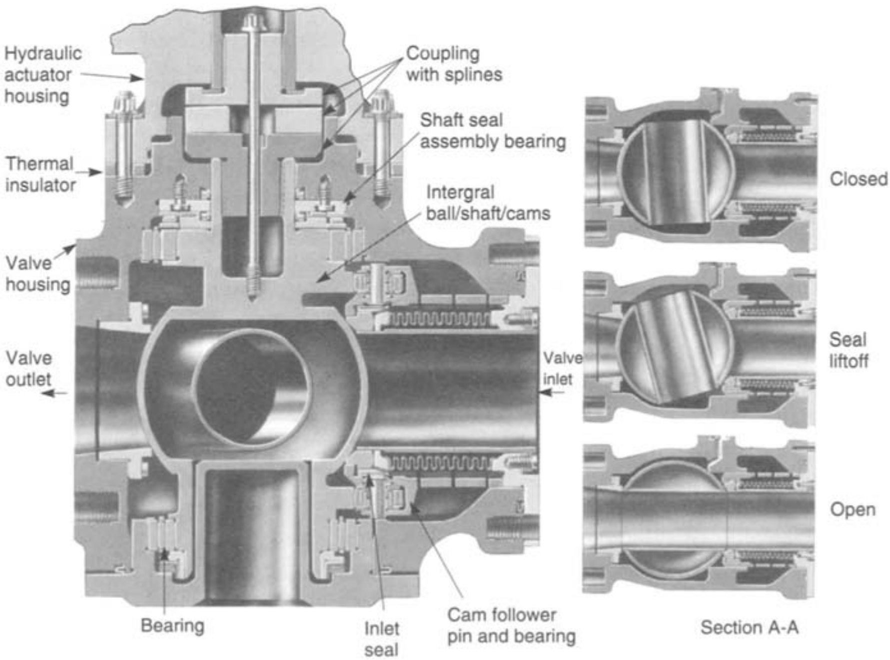


FIGURE 6–14. The SSME main oxidizer valve is a low-pressure drop ball valve representative of high-pessure large valves used in rocket engines. The ball and its integral shaft rotate in two bearings. The seal is a machined plastic ring spring-loaded by a bellows against the inlet side of the ball. Two cams on the shaft lift the seal a short distance off the ball within the first few degrees of ball rotation. The ball is rotated by a precision hydraulic actuator (not shown) through an insulating coupling. (Courtesy of The Boeing Company, Rocketdyne Propulsion and Power.)

thermal expansion and provide support to minimize vibration effects. For gimballed thrust chambers it is necessary to provide flexibility in the piping to allow the thrust axis to be rotated through a small angle, typically ± 3 to 10°. This flexibility is provided by flexible pipe joints and/or by allowing pipes to deflect when using two or more right-angle turns in the lines. The high-pressure propellant feed lines of the SSME have both flexible joints and right-angle bends, as shown in Figs 6–1 and 6–15. This joint has flexible bellows as a seal and a universal joint-type mechanical linkage with two sets of bearings for carrying the separating loads imposed by the high pressure.

Sudden closing of valves can cause water hammer in the pipelines, leading to unexpected pressure rises which can be destructive to propellant system components. An analysis of this water hammer phenomenon will allow determination of the approximate maximum pressure (Refs. 6–15 and 6–16). The friction of the pipe and the branching of pipelines reduce this maximum pressure.

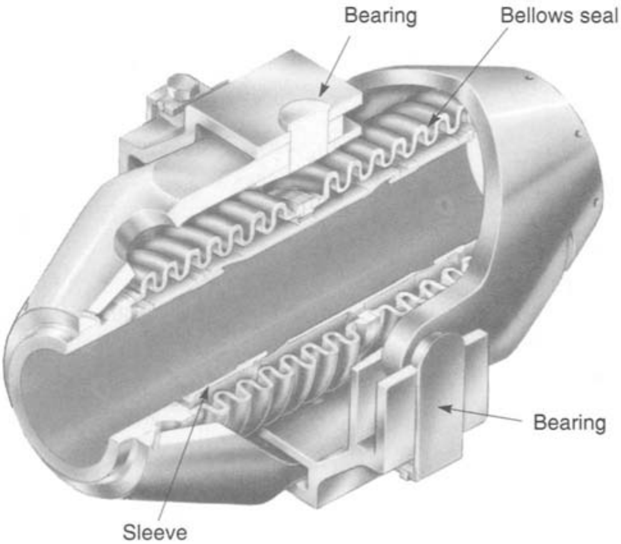


FIGURE 6–15. Flexible high-pressure joint with external gimbal rings for a high-pressure hot turbine exhaust gas. (Courtesy of The Boeing Company, Rocketdyne Propulsion and Power.)

Water hammer can also occur when admitting the initial flow of high-pressure propellant into evacuated pipes. The pipes are under vacuum to remove air and prevent the forming of gas bubbles in the propellant flow, which can cause combustion problems.

Many liquid rocket engines have *filters* in their lines. This is necessary to prevent dirt, particles, or debris, such as small pieces from burst diaphragms, from entering precision valves or regulators (where debris can cause a malfunction) or from plugging small injection holes, which could cause hot streaks in the combustion gases, in turn causing a thrust chamber failure.

Occasionally a convergent–divergent *venturi section*, with a sonic velocity at its throat, is placed into one or both of the liquid propellant lines. The merits are that it maintains constant flow and prevents pressure disturbances from traveling upstream. This can include the propagating of chamber pressure oscillations or coupling with thrust chamber combustion instabilities. The venturi section can also help in minimizing some water hammer effects in a system with multiple banks of thrust chambers.

6.10. ENGINE SUPPORT STRUCTURE

Most of the larger rocket engines have their own mounting structure or support structure. On it the major components are mounted. It also transmits the thrust force to the vehicle. Welded tube structures or metal plate/sheet metal assemblies have been used. In some large engines the thrust chamber is used as a structure and the turbopump, control boxes, or gimbal actuators are attached to it.

In addition to the thrust load, an engine structure has to withstand forces imposed by vehicle maneuvers (in some cases a side acceleration of 10 g_0), vibration forces, actuator forces for thrust vector control motions, and loads from transportation over rough roads.

In low-thrust engines with multiple thrusters there often is no separate engine mounting structure; the major components are in different locations of the vehicle, connected by tubing, wiring, or piping, and each is usually mounted directly to the vehicle or spacecraft structure.

PROBLEMS

1. Enumerate and explain the merits and disadvantages of pressurized and turbopump feed systems.

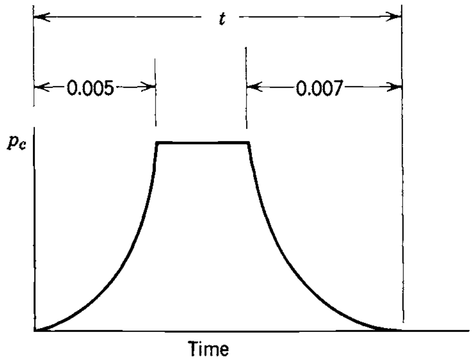
2. In a turbopump it is necessary to do more work in the pumps if the thrust chamber operating pressure is raised. This of course requires an increase in turbine gas flow which, when exhausted, adds little to the engine specific impulse. If the chamber pressure is raised too much, the decrease in performance due to an excessive portion of the total propellant flow being sent through the turbine and the increased mass of the turbopump will outweigh the gain in specific impulse that can be attained by increased chamber pressure and also by increased thrust chamber nozzle exit area. Outline in detail a method for determining the optimum chamber pressure where the sea level performance will be a maximum for a rocket engine that operates in principle like the one shown in Fig. 1–4.

3. The engine performance data for a turbopump rocket system are as follows:

Engine system specific impulse	272 sec
Engine system mixture ratio	2.52
Engine system thrust	40,000 N
Oxidizer vapor flow to pressurize oxidizer tank	0.003% of total oxidizer flow
Propellant flow through turbine	2.1% of total propellant flow
Gas generator mixture ratio	0.23
Gas generator specific impulse	85 sec

Determine performance of the thrust chamber I_s, r, F (see Sect. 10–2).

4. For a pulsing rocket engine, assume a simplified parabolic pressure rise of 0.005 sec, a steady-state short period of full chamber pressure, and a parabolic decay of 0.007 sec approximately as shown in the sketch. Plot curves of the following ratios as a function of operating time t from $t = 0.013$ to $t = 0.200$ sec; (a) average pressure to



ideal steady-state pressure (with zero rise or decay time); (b) average I_s to ideal steady-state I_s; (c) average F to ideal steady-state F.

5. For a total impulse of 100 lbf-sec compare the volume and system weights of a pulsed propulsion system using different gaseous propellants, each with a single spherical gas storage tank (at 3500 psi and 0°C). A package of small thrust nozzles with piping and controls is provided which weighs 5.2 lb. The gaseous propellants are hydrogen, nitrogen, and argon (see Table 7–3).

6. Compare several systems for a potential roll control application which requires four thrusters of 1 lbf each to operate for a cumulative duration of 2 min each. Include the following:

Pressurized helium	Cold
Pressurized nitrogen	Cold
Pressurized krypton	Cold
Pressurized helium at 500°F (electrically heated)	

The pressurized gas is stored at 5000 psi in a single spherical fiber-reinforced plastic tank; use a tensile strength of 200,000 psi and a density of 0.050 lbm/in.^{3} with a 0.012 in. thick aluminum inner liner as a seal against leaks. Neglect the gas volume in the pipes, valves, and thrusters, but assume the total hardware mass of these to be about 1.3 lbm. Use Table 7–3. Make estimates of the tank volume and total system weight. Discuss the relative merits of these systems.

7. Make tables comparing the merits and disadvantages of engines using the gas generator cycle and engines having the staged combustion cycle.

8. Prepare dimensioned rough sketches of the two propellant tanks needed for operating a single RD253 engine (Table 10–5) for 80 sec at full thrust and an auxiliary rocket system using the same propellants, with eight thrust chambers, each of 100 kg thrust, but operating on the average with only two of the eight firing at any one time, with a duty cycle of 12 percent (fires only 12% of the time), but for a total flight time of 4.00 hours. Describe any assumptions that were made with the propellant budget, the engines, or the vehicle design, as they affect the amount of propellant.

9. Table 10–5 shows that the RD 120 rocket engine can operate at 85% of full thrust and with a mixture ratio variation of ±10.0%. Assume a 1.0% unavailable residual

propellant. The allowance for operational factors, loading uncertainties, off-nominal rocket performance, and a contingency is 1.27% for the fuel and 1.15% for the oxidizer.

(a) In a particular flight the average thrust was 98.0% of nominal and the mixture ratio was off by +2.00% (oxidizer rich). What percent of the total fuel and oxidizer loaded into the vehicle will remain unused at thrust termination?

(b) If we want to run at a fuel-rich mixture in the last 20% of the flight duration (in order to use up all the intended flight propellant), what would the mixture ratio have to be for this last period?

(c) In the worst possible scenario with maximum throttling and extreme mixture ratio excursion (but operating for the nominal duration), what is the largest possible amount of unused oxidizer or unused fuel in the tanks?

SYMBOLS

a	gear ratio
F	thrust, N (lbf)
g_0	acceleration of gravity at sea level, 9.8066 m/sec^2
I_s	specific impulse, sec
k	specific heat ratio
L	shaft torque, m-N (ft-lbf)
m	propellant mass, kg (lbm)
$\dot{m}$	mass flow rate, kg/sec (lb/sec)
N	shaft speed, rpm (rad/sec)
p	pressure, N/m^2 (psi)
Δp	pressure drop, N/m^2 (psi)
P	power, W
r	mixture ratio (oxidizer to fuel mass flow rate)
t	time, sec
T	absolute temperature, K
u	vehicle velocity, m/sec (ft/sec)
V	volume flow rate, m^3/sec (ft^3/sec)
w	total propellant weight, N (lbf)
$\dot{w}$	weight flow rate, N/sec (lbf/sec)
α	nozzle divergence angle

Subscripts

b	bearings, seals
c	chamber or thrust chamber
d	discharge side
f	fuel
gg	gas generator
oa	overall

o	oxidizer
s	suction side
tp	tank pressurization
1	chamber (stagnation condition)
2	nozzle exit
3	ambient atmosphere

REFERENCES

6–1. D. K. Huzel and D. H. Huang. *Design of Liquid Propellant Rocket Engines*, Revised edition, AIAA, 1992, 437 pages.

6–2. G. G. Gakhun, V. I. Baulin, et ala., *Construction and Design of Liquid Propellant Rocket Engines* (in Russian), *Konstruksiya i Proyektirovaniye Zhidkostniyk Raketnykh Dvigateley*, Mashinostroyeniye, Moscow, 1989, 424 pages.

6–3. C. J. G. Dixon and J. G. B. Marshall, "Mathematical Modelling of Bipropellant Combined Propulsion Subsystems," *AIAA Paper 90–2303*, 26th Joint Propulsion Conference, July 1990; and *Design Guide for Pressurized Gas Systems*, Vols. I and II, prepared by IIT Research Institute, NASA Contract NAS7-388, March 1966.

6–4. H. C. Hearn, "Design and Development of a large Bipropellant Blowdown Propulsion System," *Journal of Propulsion and Power*, Vol. 11, No. 5, September–October 1995.

6–5. H. C. Hearn, "Evaluation of Bipropellant Pressurization Concepts for Spacecraft," *Journal of Spacecraft and Rockets*, Vol. 19, July 1982, pp. 320–325.

6–6. *National Space Transportation System Reference*, Vol. 1, National Aeronautics and Space Administration, Washington, DC, June 1988 (description of Space Shuttle system and operation).

6–7. J. I. Hochsten, H.-C. Ji, and J. Ayelott, "Prediction of Self-Pressurization Rate of Cryogenic Propellant Tankage," *Journal of Propulsion and Power*, Vol. 6, No. 1, January–February 1990, pp. 11–17.

6–8. B. Morton, M. Elgersma, and R. Playter, "Analysis of Booster Vehicle Slosh Stability during Ascent to Orbit," *AIAA Paper 90-1876*, July 1990, 7 pages.

6–9. J. J. Pocha, "Propellant Slosh in Spacecraft and How to Live with It," *Aerospace Dynamics*, Vol. 20, Autumn 1986, pp. 26–31.

6–10. G. P. Purohit and L. D. Loudenback, "Application of Etched Disk Stacks in Surface Tension Propellant Management Devices," *Journal of Propulsion and Power*, Vol. 7, No. 1, January–February 1991, pp. 22–30.

6–11. J. R. Rollins, R. K. Grove, and D. R. Walling, Jr. "Design and Qualification of a Surface Tension Propellant Tank for an Advanced Spacecraft," *AIAA Paper 88–2848*, 24th Joint Propulsion Conference, 1988.

6–12. H. Grosdemange and G. Schaeffer. "The SEPR 844 Reuseable Liquid Rocket Engine for Mirage Combat Aircraft", *AIAA Paper 90–1835*, July 1990.

6–13. D. Manski, C. Goertz, H. D. Sassnick, J. R. Hulka, B. D. Goracke, and D. J. H. Levack, "Cycles for Earth to Orbit Propulsion," *Journal of Propulsion and Power, AIAA, Vol. 14, No. 5, September–October 1998.*

6–14. J. R. Brown. "Expander Cycle Engines for Shuttle Cryogenic Upper Stages, *AIAA Paper 83–1311*, 1983.

6–15. R. P. Prickett, E. Mayer, and J. Hermel, "Waterhammer in Spacecraft Propellant Feed Systems," *Journal of Propulsion and Power*, Vol. 8, No. 3, May–June 1992.

6–16. Chapter 9 in: I. Karassik, W. C. Krutzsch, W. H. Fraser, and J. P. Messina (eds), *Pump Handbook*, McGraw-Hill Book Company, New York, 1976 (pumps and waterhammer).