CHAPTER 10

TURBOPUMPS, ENGINE DESIGN, ENGINE CONTROLS, CALIBRATION, INTEGRATION, AND OPTIMIZATION

In this chapter we first discuss a complex high-precision, high-speed, rotating subsystem, namely the turbopump. Only some high-thrust engines have turbopumps. This chapter contains an overall engine discussion which applies to all engines. This includes the liquid propellant rocket engine's design, performance, controls, calibration, propellant budget, integration, and optimization.

10.1. TURBOPUMPS

The assembly of a turbine with one or more pumps is called a *turbopump*. Its purpose is to raise the pressure of the flowing propellant. Its principal subsystems are a hot gas powered *turbine* and one or two *propellant pumps*. It is a high precision rotating machine, operating at high shaft speed with severe thermal gradients and large pressure changes, it usually is located next to a thrust chamber, which is a potent source of noise and vibration.

This turbopump feed system and its several cycles have been discussed in Section 6.6 and Fig. 6–2 categorizes the various common turbopump configurations. Turbopumps or installation of turbopumps in rocket engines are shown in Figs. 1–4, 6–1, 6–12, 8–19, 10–1, 10–2, 10–3, and 10–11; they are discussed in Refs. 6–1 and 10–1. A schematic diagram of different design arrangements of pumps and turbines for common turbopump types can be seen in Fig. 10–4. Table 10–1 shows lists parameters of pumps and turbines of two large rocket engines.

Specific nomenclature and terminology used in the next few paragraphs will be explained later in this Chapter. In Fig. 10–1 a simple turbopump with a

TABLE 10–1. Turbopump Characteristics

Engine:	Space Shuttle Main Engine[a]					LE-7[b]		
Feed System Cycle:	Modified Staged Combustion Cycle					Modified Staged Combustion Cycle		
Propellants:	Liquid Oxygen and Liquid Hydrogen					Liquid Oxygen and Liquid Hydrogen		
Pumps								
Designation[c]	LPOTP	LPFTP	HPOTP		HPFTP	HPFTP	HPOTP	
Type	Axial flow	Axial flow	Dual inlet		Radial flow	Radial flow	Radial flow	
No. of impeller stages	—	—	1 + 1[d]		3	2	1 + 1[d]	
No. of aux. or inducers	1	1	—		1	1	1	
Flow rate (kg/sec)	425	70.4	509	50.9	70.4	35.7	211.5	46.7[d]
Inlet pressure (MPa)	0.6	0.9	2.70	NA	1.63	0.343	0.736	18.2[d]
Discharge pressure (MPa)	2.89	2.09	27.8	47.8[d]	41.0	26.5	18.2	26.7[d]
Pump efficiency (%)	68	75	72	75[d]	75	69.9	76.5	78.4[d]
Turbines								
No. of stages	6	2	3		2			
Type	Hydraulic LOX driven	Reaction-impulse	Reaction-impulse		Reaction-impulse	Reaction-impulse	Reaction-impulse	
Flow rate (kg/sec)			27.7		66.8	33.1	15.4	
Inlet temperature (K)	105.5	264	756		1000	871	863	
Inlet pressure (MPa)	26.2	29.0	32.9		32.9	20.5	19.6	
Pressure ratio	NA	1.29	1.54		1.50	143	1.37	
Turbine efficiency (%)	69	60	74		79	73.2	48.1	
Turbine speed (rpm)	5020	15,670	22,300		34,270	41600	18300	
Turbine power (kW)	1120	2290	15,650		40,300	25,350	7012	
Mixture ratio, O/F	LOX only	H_2 only	~0.62		~0.88	~ 0.7	~ 0.7	

[a]Data courtesy of The Boeing Company, Rocketdyne Propulsion and Power, at flight power level of 104.5% of design thrust.
[b]Data courtesy of Mitsubishi Heavy Industries, Ltd.
[c]LPOTP, low-pressure oxidizer turbopump; HPFTP, high-pressure fuel turbopump.
[d]Boost impeller stage for oxygen flow to preburners or gas generator.

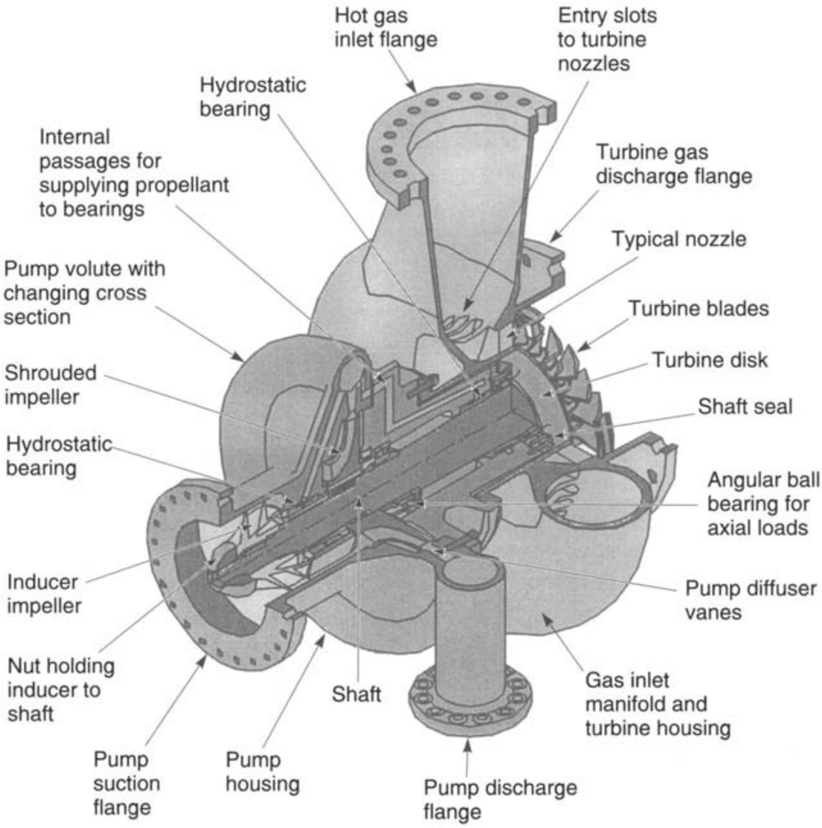


FIGURE 10–1. Cut-away view of an experimental turbopump demonstrator with a single-stage liquid oxygen pump impeller, an inducer impeller, and a single-stage turbine (one row of blades) on the same shaft. (Courtesy of The Boeing Company, Rocketdyne Propulsion and Power.)

single-stage propellant pump (with an inducer impeller ahead of the main impeller) is driven by a single-stage axial-flow turbine. The hot combustion gases, which drive this turbine, are burned in a separate gas generator (or a precombustion chamber or preburner) at a mixture ratio that gives gases between 900 and 1200 K; this is sufficiently cool, so that the hot turbine hardware (blades, nozzles, manifolds, or disks) still have sufficient strength without needing forced cooling. The gases are expanded (accelerated) in an annular set of converging–diverging supersonic turbine nozzles, which are cast into the cast turbine inlet housing. The gases then enter a set of rotating blades, which are mounted on a rotating wheel or turbine disk. The blades remove the tangential energy of the gas flow. The exhaust gas velocity exiting from the blades is relatively low and its direction is essentially parallel to the shaft. The pump is driven by the turbine through an interconnecting solid shaft. The propellant enters the pump through an *inducer*, a special impeller where the pressure of the propellant is raised only slightly (perhaps 5 to 10% of the total pressure rise). This is just enough pressure so that there will be no cavitation as the flow enters the main pump impeller. Most of the kinetic energy given to the flow by the pump impeller is converted into hydrostatic pressure in the diffusers (the diffuser vanes are not clearly visible, since they are inclined) and/or volutes of the pump. The two hydrostatic bearings support the shaft radially. All bearings and shaft seals create heat as they run. They are cooled and lubricated by a small flow of propellant, which is supplied from the pump discharge through drilled passages. One bearing (near the pump) is very cold and the other is hot, since it is close to the hot turbine. The angular ball bearing accepts the axial net loads from the unbalanced hydrodynamic pressures around the shrouded impeller, the inducer, and also the turbine blades or the turbine disk.

A novel, high speed, compact, and light weight liquid hydrogen turbopump is shown in Fig. 10–2 and in Ref 10.2. It is intended to be used with a new upper stage hydrogen/oxygen rocket engine with a thrust of about 50,000 lbf (22.4 kN), a delivered engine specific impulse of 450.6 sec at an engine mixture ratio of 6.0. This engine will run on an expander cycle, with a chamber pressure of 1375 psia (96.7 kg/m^2) and a maximum internal fuel pressure of 4500 psi (323.4 kg/m^2) at the fuel pump discharge. The unique single-piece titanium rotor turns nominally at 166,700 rpm, has two machined sets of pump vanes, a machined inducer impeller, a set of machined radial inflow turbine

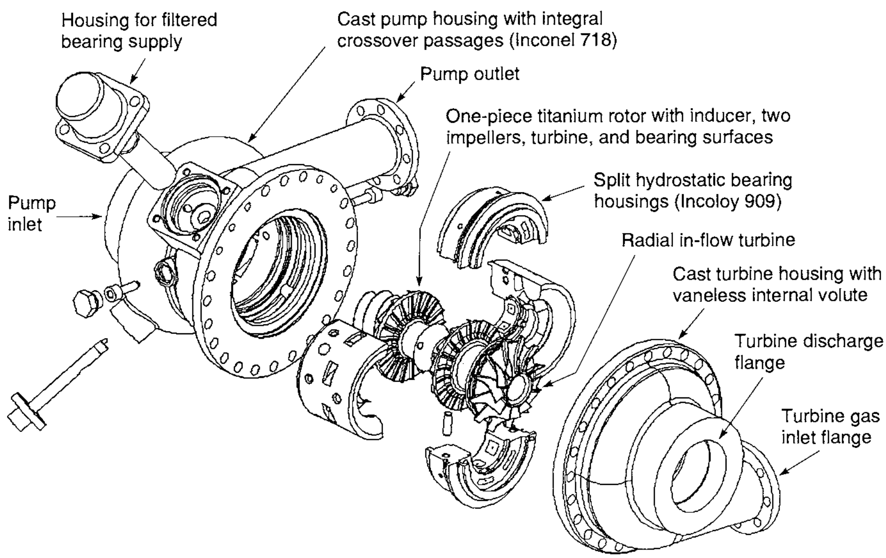


FIGURE 10–2. Exploded view of an advanced high-speed, two-stage liquid hydrogen fuel pump driven by a radial flow turbine. (Copied with permission of Pratt & Whitney, a division of United Technologies; adapted from Ref. 10–2.)

blades, and radial as well as axial bearing surfaces. A small filtered flow of hydrogen lubricates the hydrostatic bearing surfaces. The cast pump housing has internal crossover passages between stages. The unique radial in-flow turbine (3.2 in. dia.) produces about 5900 hp at an efficiency of 78%. The hydrogen pump impellers are only 3.0 in. diameter and produce a pump discharge pressure of about 4500 psi at a fuel flow of 16 lbm/sec and an efficiency of 67%. A high pump inlet pressure of about 100 psi is needed to assure cavitation-free operation. The turbopump can operate at about 50% flow (at 36% discharge pressure and 58% of rated speed). The number of pieces to be assembled is greatly reduced, compared to a more conventional turbopump, thus enhancing its inherent reliability.

The geared turbopump in Fig. 10–3 has a higher turbine and pump efficiencies, because the speed of the two-stage turbine is higher than the pump shaft speeds and the turbine is smaller. The auxiliary power package (e.g., hydraulic pump) was used only in an early application. The precision ball bearings and

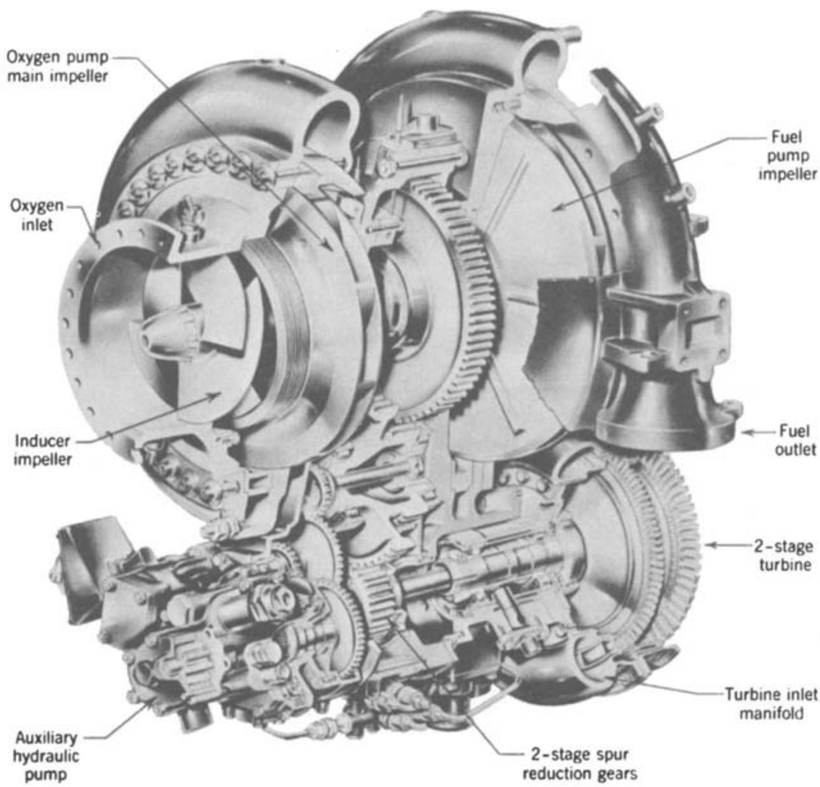


FIGURE 10–3. Typical geared turbopump assembly used on the RS-27 engine (Delta I and II Launch Vehicles) with liquid oxygen and RP-1 propellants. (Courtesy of The Boeing Company, Rocketdyne Propulsion and Power.)

seals on the turbine shaft can be seen, but the pump bearings and seals are not visible in this figure.

Approach to Turbopump Preliminary Design

With all major rocket engine components the principal criteria (high performance or efficiency, minimum mass, high reliability, and low cost) have to be weighted and prioritized for each vehicle mission. For example, high efficiency and low mass usually mean low design margins, and thus lower reliability. A higher shaft speed will allow a lower mass turbopump, but it cavitates more readily and requires a higher tank pressure and heavier vehicle tanks (which often outweigh the mass savings in the turbopump) in order to have acceptable life and reliability.

The engine requirements give the initial basic design goals for the turbopump, namely propellant flow, the pump outlet or discharge pressure (which has to be equal to the chamber pressure plus the pressure drops in the piping, valves, cooling jacket, and injector), the desired best engine cycle (gas generator or staged combustion, as shown in Fig. 6–9), the start delay, and the need for restart or throttling, if any. Also, the propellant properties (density, vapor pressure, viscosity, or boiling point) must be known. Some of the design criteria are explained in Refs. 6–1 and 10–3, and basic texts on turbines and pumps are listed as Refs. 10–4 to 10–8.

There are several design variations or geometrical arrangements for transmitting turbine power to one or more propellant pumps; some are shown schematically in Fig. 10–4. If the engine has propellants of similar density (such as liquid oxygen and RP-1), the fuel and oxidizer pumps will have similar shaft speeds and can usually be placed on a common shaft driven by a single turbine (F-1, RS-27/Delta Fig. 10–3, Atlas, or Redstone engines). If there is a mismatch between the optimum pump speed and the optimum turbine speed (which is usually higher), it may save inert mass and turbine drive gas mass to interpose a gear reduction between their shafts. See Fig. 6–11. For the last two decades designers have preferred to use direct drive, which avoids the complication of a gear case but at a penalty in efficiency and the amount of turbine drive propellant gas required. See Figs. 6–12, 10–1, or 10–2.

If the densities are very different (e.g., liquid hydrogen and liquid oxygen), the pump head rise* (head $= \Delta p/\rho$) is much higher for the lower-density propellant, and the hydrogen pump usually has to have more than one impeller or one stage and will typically operate at a higher shaft speed; in this case separate

*_Pump head_ means the difference between pump discharge and pump suction head. Its units are meters or feet. The head is the height of a column of liquid with equivalent pressure at its bottom. The conversion from pounds per square inch into feet of head is: (X) psi $= 144(X)/\text{density (lb/ft}^3)$. To convert pascals (N/m^2) of pressure into column height (m), divide by the density (kg/m^3) and g_0 (9.806 m/sec^2).

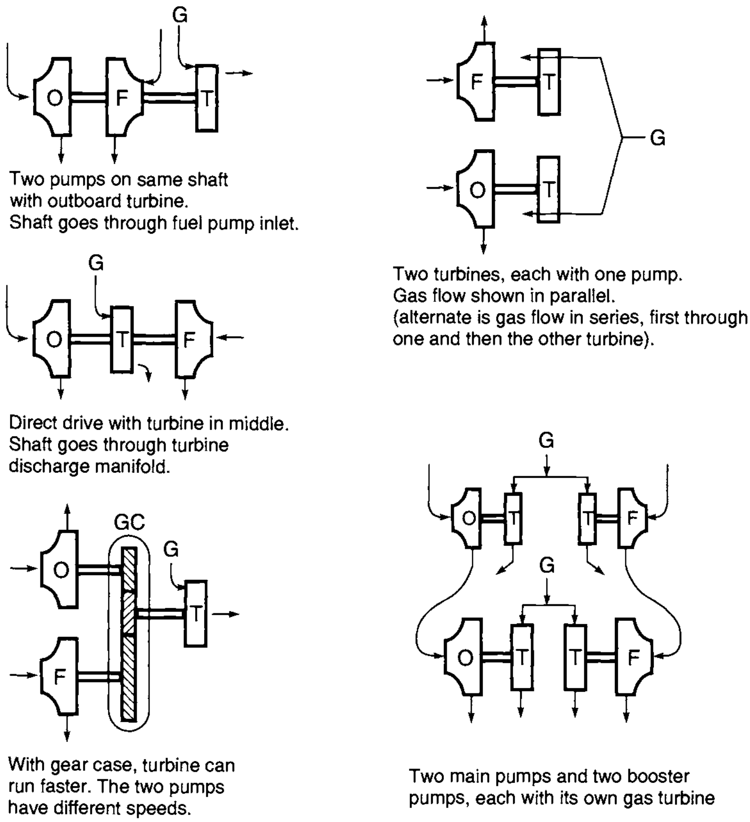


FIGURE 10–4. Simplified diagrams of different design arrangements of turbopumps. F is fuel pump, O is oxidizer pump, T is turbine, G is hot gas, and GC is gear case.

turbopumps for the fuel and the oxidizer can give the lowest energy and overall mass (J-2, SSME, LE-7, Vulcain 60).

Usually, the preliminary analysis for the pump is done first. Avoiding excessive cavitation sets a key pump parameter, namely the maximum shaft speed. This is the highest possible shaft speed, which in turn allows the lightest turbopump mass, without excessive cavitation in the pump. If excessive cavitation occurs at the leading edge of the first impeller (inducer or main impeller), then the flow will become unsteady and variable, leading to lower thrust and possible combustion instability. The amount of pressure in the vehicle (gas pressure in propellant tank plus the static elevation pressure) that can be made available to the engine (at the pump inlet) for suppressing cavitation has to be larger than the impeller vanes' own pressure limit to cavitate. This allows us then to determine the shaft speed, which in turn can establish the approximate pump efficiencies, impeller tip speed (usually limited by the material strength of the impeller), number of pump stages, key dimensions of the impeller, and the pump power requirements. All this will be discussed further (including key equations) in the pump section of this chapter.

The key turbine parameter can be estimated, because the power output of the turbine essentially has to equal the power demand of the pump. If the pump is driven directly, that is without a gear case, then the pump speed and the turbine speed are equal. From the properties of the turbine drive gas (temperature, specific heat, etc.), the strength limits of the turbine materials, and the likely pressure drop, it is possible to determine the basic dimensions of the blades (pitch line velocity, turbine nozzle outlet velocity, number of rows (stages) of blades, turbine type, or turbine efficiency). The particular arrangement or geometry of the major turbopump components is related to their selection process. Most propellant pumps have a single-stage main impeller. For liquid hydrogen with its low density, a two- or three-stage pump is normally needed. Usually some design limit is reached which requires one or more iterations, each with a new changed approach or parameter. The arrangement of the major turbopump components (Fig. 10–4) is also influenced by the position of the bearings on the shaft. For example, we do not want to place a bearing in front of an impeller inlet because it will cause turbulence, distort the flow distribution, raise the suction pressure requirement, and make cavitation more likely to occur. Also, bearings positioned close to a turbine will experience high temperatures, which influences the lubrication by propellant and may demand more cooling of the bearings.

The use of booster pumps allows lower tank pressure, and thus lower inert vehicle mass, and provides adequate suction pressures to the main pump inlet. Booster pumps are used in the Space Shuttle main engine and the Russian RD-170, as seen in Figs. 6–12 and 10–11. Some booster pumps have been driven by a liquid booster turbine using a small flow of high-pressure liquid propellant that has been tapped off the discharge side of the main pump. The discharged turbine liquid then mixes with the main propellant flow at the discharge of the booster pump.

Later in this section a few of the equations that apply to the steady-state (full thrust) operating condition will be described. However, no detailed discussion will be given of the *transient starting conditions*, such as the filling of pipes, pumps, or manifolds with liquid propellants, or the filling of turbines and their manifolds with high-pressure gas. These dynamic conditions can be complex, are related to the combustion reactions, and are sometimes difficult to analyze, yet they are very significant in the proper and safe operation of the engine. Each major rocket engine manufacturer has developed some methodology, usually analysis and hydraulic models, for these system dynamics that are often peculiar to specific engines and hardware (see Refs. 10–3 and 10–4).

Mass is at a premium in all flying installations, and the feed system is selected to have a minimum combined mass of tubines, pumps, gas generator, valves, tanks, and gas generator propellants. Some of the considerations in the design of turbopumps are the thermal stresses, warpage due to thermal expansion or contraction, axial loads, adequate clearances to prevent rubbing yet minimize leakage, alignment of bearings, provisions for dynamic balancing of rotating parts, mounting on an elastic vehicle frame without inducing external forces, and avoiding undue pressure loads in the liquid and gas pipes.

Vibrations of turbopumps have caused problems during development. The analyses of the various vibrations (shaft, turbine blades, liquid oscillations, gas flow oscillations, or bearing vibrations) are not given here. At the *critical speed* the natural structural resonance frequency of the rotating assembly (shaft, impellers, turbine disk, etc.) coincides with the rotation operating speed. A slight unbalance can be amplified to cause significant shaft deflections (in bending), bearing failure, and other damage. The operating speed therefore is usually lower and sometimes higher than the critical speed. A large diameter stiff shaft, rigid bearings, and stiff bearing supports will increase this critical speed, and damping (such as the liquid lubricant film in the bearing) will reduce the vibration amplitude. Also, this critical shaft frequency or the operating speed should not coincide with and excite other natural vibration frequencies, such as those of various parts (piping, bellows, manifolds, or injector dome). The solving of various internal vibrations problems, such as whirl in bearings and blade vibrations, is reported in Ref. 10–5.

Bearings in most existing turbopumps are high precision, special alloy ball or roller bearings. Some ball bearings can take both radial and axial loads. Ball and roller bearings are limited in the loads and speeds at which they can operate reliably. In some turbopump designs this maximum bearing speed determines the minimum size of turbopump, rather than the cavitation limit of the pump. More recently, we use hydrostatic bearings where the shaft rides on a high-pressure fluid film; they have good radial load capacity, can provide some damping of oscillations and a stiff support. Axial loads (due to pressure unbalance on impellers and turbine blades) can be taken by special hydrostatic bearings. Since there is no direct contact between rolling and stationary assemblies, there is little or no wear and the life expectancy of these hydrostatic bearings is long. However, there is rubbing contact and wear at low speeds, namely during start or shutdown (see Ref. 10–6).

Cooling and lubricating the bearings and seals is essential for preventing bearing problems. A small flow of one of the propellants is used. Hydrocarbon fuels are usually good lubricants and hydrogen is a good coolant, but a marginal lubricant. If an oxidizer is used as the coolant and lubricant, then the materials used for bearings and seals have to be resistant to oxidation when heated during operation.

If the turbopump is part of a reusable rocket engine, it becomes more complex. For example, it can include provision to allow for inspection and automatic condition evaluation after each mission or flight. This can include an inspection of bearings through access holes for boroscope instruments, checking for cracks in highly stressed parts (turbine blade roots or hot-gas high-pressure manifolds), or the measurement of shaft torques (to detect possible binding or warpage).

Pumps

Classification and Description. The *centrifugal pump* is generally considered the most suitable for pumping propellant in large rocket units. For the large flows and high pressures involved, they are efficient as well as economical in terms of mass and space requirement.

Figure 10–5 is a schematic drawing of a centrifugal pump. Fluid entering the *impeller*, which is essentially a wheel with spiral curved vanes rotating within a *casing*, is accelerated within the impeller channels and leaves the impeller periphery with a high velocity to enter the *volute*, or collector, and thereafter the *diffuser*, where conversion from kinetic energy (velocity) to potential energy (pressure) takes place. In some pumps the curved diffuser vanes are upstream of the collector. The three-dimensional hydraulic design of impeller vanes, diffuser vanes, and volute passages can be accomplished by computer programs to give high efficiency and adequate strength. Internal leakage, or circulation between the high-pressure (discharge) side and the low-pressure (suction) side of an impeller, is held to a minimum by maintaining close clearances between the rotating and stationary parts at the *seals* or *wear ring surfaces*. External leakage along the shaft is minimized or prevented by the use of a *shaft seal*. Single-stage pumps (one impeller only) are stress-limited in the pressure rise they can impart to the liquid, and multiple-stage pumps are therefore needed for high pump head,* such as with liquid hydrogen. References 10–5 to 10–7 give information on different pumps. There is a free passage of flow through the pump at all times, and no positive means for shutoff are provided. The pump characteristics, that is, the pressure rise, flow, and efficiency, are functions of the pump speed, the impeller, the vane shape, and the casing configuration. Figure 10–6 shows a typical set of curves for centrifugal

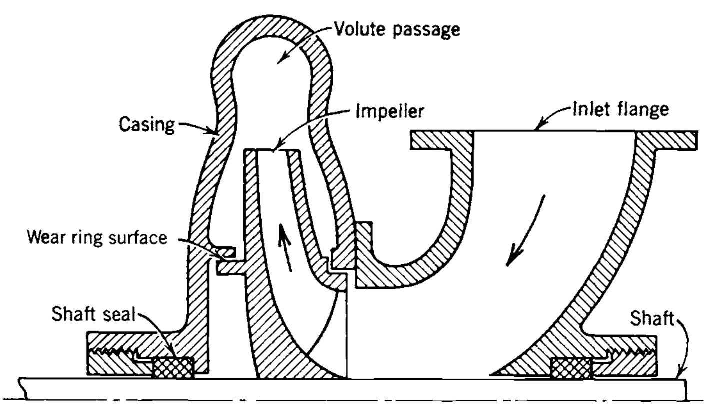


FIGURE 10–5. Simplified schematic half cross section of a typical centrifugal pump.

*See footnote on page 367.

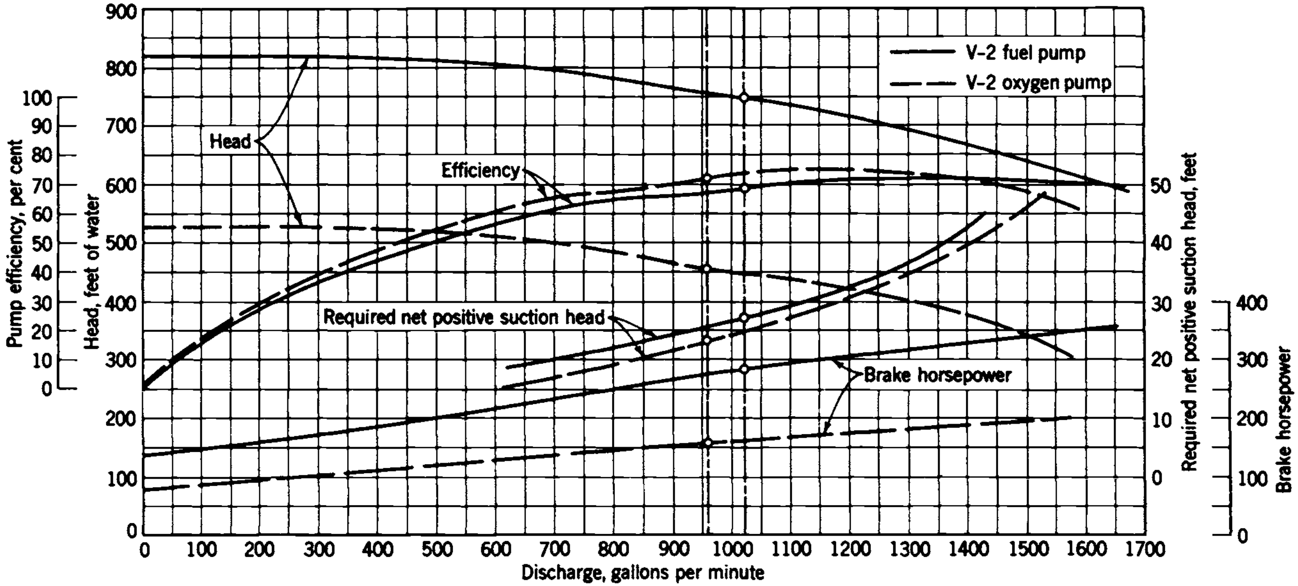


FIGURE 10–6. Water test performance curves of the centrifugal pumps of the German V-2 rocket engine. The propellants are diluted 75% ethyl alcohol and liquid oxygen.

pumps. The negative slope on the head versus flow curve indicates a stable pump behavior. References 10–7 and 10–8 describe the development of a smaller turbopump and the testing of a spiral high-speed first-stage impeller, called an inducer.

A *shrouded impeller* has a shroud or cover (in the shape of a surface of revolution) on top of the vanes as shown in Figs. 10–1, 10–3, and 10–5. This type usually has higher stresses and lower leakage around the impeller. In an unshrouded impeller or turbine the vanes are not covered as seen in the turbine vanes in Fig. 10.2.

Pump Parameters. This section outlines some of the important parameters and features that have to be considered in the design of rocket propellant centrifugal pumps under steady flow conditions.

The *required pump flow* is established by the rocket design for a given thrust, effective exhaust velocity, propellant densities, and mixture ratio. In addition to the flow required by the thrust chamber, the propellant consumption of the gas generator, and in some designs also a bypass around the turbine and auxiliaries have to be considered in determining the pump flows. The required *pump discharge pressure* is determined from the chamber pressure and the hydraulic losses in valves, lines, cooling jacket, and injectors (see Eq. 6–15). To obtain the rated flow at the rated pressure, an additional adjustable pressure drop for a control valve or orifice is usually included which permits a calibration adjustment or change in the required feed pressure. A regulation of the pump speed can also change the required adjustable pressure drop. As described in Section 10.6, this adjustment of head and flow is necessary to allow for hydraulic and performance tolerances on pumps, valves, injectors, propellant density, and so on.

It is possible to predict the *pump performance at various speeds* if the performance is known at any given speed. Because the fluid velocity in a given pump is proportional to the pump speed N, the flow quantity or discharge Q is also proportional to the speed and the head H is proportional to the square of the speed. This gives the following relations:

$$\begin{aligned} Q \text{ (flow)} &\sim N \text{ (rpm or rad/sec)} \\ H \text{ (pump head)} &\sim N^2 \\ P \text{ (pump power)} &\sim N^3 \end{aligned} \tag{10–1}$$

From these relations it is possible to derive a parameter called the *specific speed* N_s. It is a dimensionless number derived from a dimensional analysis of pump parameters as shown in Ref. 10–9.

$$N_s = N\sqrt{Q_e}/(g_0 \Delta H_e)^{3/4} \tag{10–2}$$

Any set of consistent units will satisfy the equation: for example, N in radians per second, Q in m^3/s, g_0 as 9.8 m/sec^2, and H in meters. The subscript e refers to the maximum efficiency condition. In U.S. pump practice it has become the custom to delete g_0, express N in rpm, and Q in gallons per minute or ft^3/sec. Much of the existing U.S. pump data is in these units. This leads to a modified form of Eq. 10–2, where N_s is not dimensionless, namely

$$N_s = 21.2N\sqrt{Q_e}/(\Delta H_e)^{3/4} \tag{10–3}$$

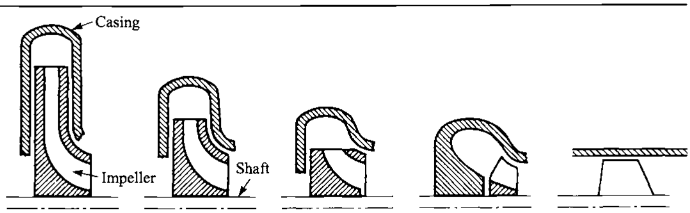
The factor 21.2 applies when N is in rpm, Q is in ft^3/sec, and H is in feet. For each range of specific speed, a certain shape and impeller geometry has proved most efficient, as shown in Table 10–2. Because of the low density, hydrogen can be pumped effectively by axial flow devices.

The *impeller tip speed* in centrifugal pumps is limited by design and material strength considerations to about 60 to 450 m/sec or roughly 200 to 1475 ft/sec. With titanium (lower density than steel) and machined unshrouded impellers a tip speed of over 2150 ft/sec is now possible and used on the pumps shown in Fig. 10–2. For cast impellers this limiting value is lower than for machined impellers. This maximum impeller tip speed determines the maximum head that can be obtained from a single stage. The impeller vane tip speed u is the product of the shaft speed, expressed in radians per second, and the impeller radius and is related to the pump head by

$$u = \psi\sqrt{2g_0\Delta H} \tag{10–4}$$

where ψ has values between 0.90 and 1.10 for different designs. For many pumps, $\psi = 1.0$.

TABLE 10-2. Pump Types

	Impeller type				
	Radial	Francis	Mixed flow	Near axial	Axial
Basic shape (half section)					
Specific speed N_s					
U.S. nomenclature	500–1000	1000–2000	2000–3000	3000–6000	Above 8000
SI consistent units	0.2–0.3	0.4	0.6–0.8	1.0–2.0	Above 2.5
Efficiency %	50–80	60–90	70–92	76–88	75–82

The flow quantity defines the impeller inlet and outlet areas according to the equation of continuity. The diameters obtained from this equation should be in the proportion indicated by the diagrams for a given specific speed in Table 10–2. The continuity equation for an incompressible liquid is

$$Q = A_1 v_1 = A_2 v_2 \tag{10–5}$$

where the subscripts refer to the impeller inlet and outlet sections, all areas being measured normal to their respective flow velocity. The inlet velocity v_1 ranges usually between 2 and 6 m/sec or 6.5 to 20 ft/sec and the outlet velocity v_2 between 3 and 15 m/sec or 10 to 70 ft/sec. For a compressible liquid, such as liquid hydrogen, the density will change with pressure. The continuity equation then is:

$$\dot{m} = A_1 v_1 \rho_1 = A_2 v_2 \rho_2 \tag{10–6}$$

The head developed by the pump will then also depend on the change in density.

The pump performance is limited by *cavitation*, a phenomenon that occurs when the static pressure at any point in a fluid flow passage becomes less than the fluid's vapor pressure. The formation of vapor bubbles causes cavitation. These bubbles collapse when they reach a region of higher pressure, that is, when the static pressure in the fluid is above the vapor pressure. In centrifugal pumps cavitation is most likely to occur behind the leading edge of the pump impeller vane at the inlet because this is the point at which the lowest absolute pressure is encountered. The excessive formation of vapor causes the pump discharge mass flow to diminish and fluctuate and can reduce the thrust and make the combustion erratic and dangerous (see Ref. 10–10).

When the bubbles travel along the pump impeller surface from the low-pressure region (where they are formed) to the downstream higher-pressure region, the bubbles collapse. The sudden collapses create local high-pressure pulses that have caused excessive stresses in the metal at the impeller surface. In most rocket applications this cavitation erosion is not as serious as in water or chemical pumps, because the cumulative duration is relatively short and the erosion of metal on the impeller is not usually extensive. It has been a concern with test facility transfer pumps.

The *required suction head* $(H_s)_R$ is the limit value of the head at the pump inlet (above the local vapor pressure); below this value cavitation in the impeller will not occur. It is a function of the pump and impeller design and its value increases with flow as can be seen in Fig. 10–6. To avoid cavitation the *suction head* above vapor pressure *required* by the pump $(H_s)_R$ must always be *less* than the *available* or *net positive suction head* furnished by the line up to the pump $(H_s)_A$, that is, $(H_s)_R \le (H_s)_A$. The required suction head above vapor pressure can be determined from the suction specific speed S:

$$S = 21.2 N \sqrt{Q_e} / (H_s)_R^{3/4} \tag{10–7}$$

The suction specific speed S depends on the quality of design and the specific speed N_s, as shown in Table 10–2. The suction specific speed S has a value between 5000 and 60,000 when using ft-lbf units. For pumps with poor suction characteristics it has values near 5000, for the best pump designs without cavitation it has values near 10,000 and 25,000, and for pumps with limited and controllable local cavitation it has values above 40,000. In Eq. 10–7 the required suction head $(H_s)_R$ is usually defined as the critical suction head at which the developed pump discharge head has been diminished arbitrarily by 2% in a pump test with increasing throttling in the suction side. Turbopump development has, over the last several decades, led to impeller designs which can operate successfully with considerably more cavitation than the arbitrary and commonly accepted 2% head loss limit. Inducers are now designed to run stably with extensive vapor bubbles near the leading edge of their vanes, but these bubbles collapse at the trailing end of these vanes. Inducers now can have S values above 80,000. A discussion of the design of impeller blades can be found in Ref. 10–9.

The head that is available at the pump suction flange is called the *net positive suction head* or *available suction head above vapor pressure* $(H_s)_A$. It is an absolute head value determined from the tank pressure (the absolute gas pressure in the tank above the liquid level), the elevation of the propellant level above the pump inlet, the friction losses in the line between tank and pump, and the vapor pressure of the fluid. When the flying vehicle is undergoing accelerations, the head due to elevation must be corrected accordingly. These various heads are defined in Fig. 10–7. The net positive suction head $(H_s)_A$ is the maximum head available for suppressing cavitation at the inlet to the pumps:

$$(H_s)_A = H_{\text{tank}} + H_{\text{elevation}} - H_{\text{friction}} - H_{\text{vapor}} \qquad (10\text{–}8)$$

To avoid pump cavitation, $(H_s)_A$ has to be higher than $(H_s)_R$. If additional head is required by the pump, the propellant may have to be pressurized by external means, such as by the addition of another pump in series (called a booster pump) or by gas pressurization of the propellant tanks. This latter method requires thicker tank walls and, therefore, heavier tanks, and a bigger gas-pressurizing system. For example, the oxygen tank of the German V-2 was pressurized to 2.3 atm, partly to avoid pump cavitation. For a given value of $(H_s)_A$, propellants with high vapor pressure require correspondingly higher tank pressures and heavier inert tank masses. For a given available suction head $(H_s)_A$, a pump with a low required suction pressure usually permits designs with high shaft speeds, small diameter, and low pump inert mass. A small value of $(H_s)_R$ is desirable because it may permit a reduction of the requirements for tank pressurization and, therefore, a lower inert tank mass. The value of $(H_s)_R$ will be small if the impeller and fluid passages are well designed and if the shaft speed N is low. A very low shaft speed, however, requires a large diameter pump, which will be excessively heavy. The trend in

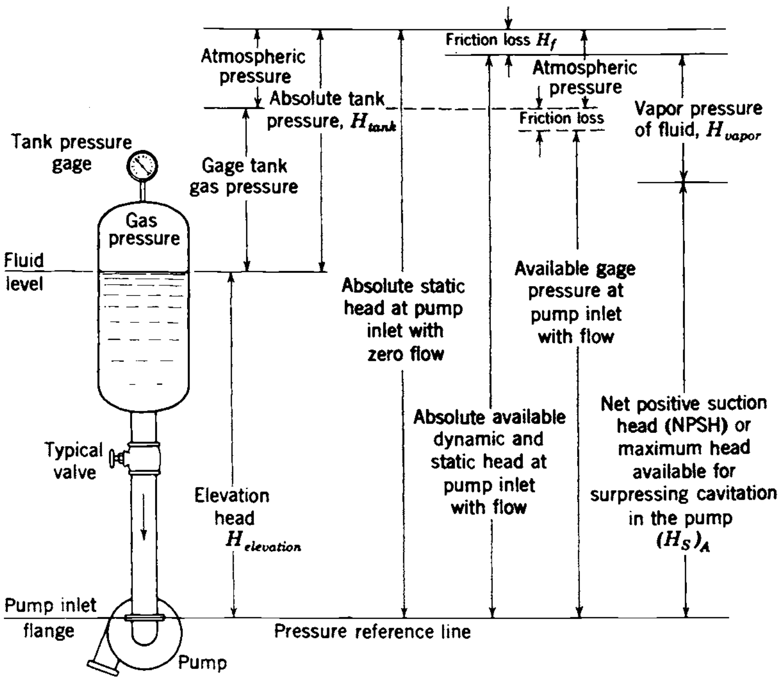


FIGURE 10–7. Definition of pump suction head.

selecting centrifugal pumps for rocket application has been to select the highest shaft speed that gives a pump with a low value of $(H_S)_R$, does not require excessive tank pressurization or other design complications, and thereby permits relatively lightweight pump design. This places a premium on pumps with good suction characteristics.

There have been some low-thrust, low-flow, experimental engines that have used positive displacement pumps, such as diaphragm pumps, piston pumps, or rotary displacement pumps (gear and vane pumps). For low values of N_S these pumps have much better efficiencies, but their discharge pressures fluctuate with each stroke and they are noisy.

One method to provide a lightweight turbopump with minimal tank pressure is to use an *inducer*, which is a special pump impeller usually on the same shaft and rotating at the same speed as the main impeller. It has a low head rise and therefore a relatively high specific speed. Inducer impellers are immediately upstream of the main impeller. They are basically axial flow pumps with a spiral impeller, and many will operate under slightly cavitating conditions. The inducer stage's head rise (typically, 2 to 10% of the total pump head) has to be just large enough to suppress cavitation in the main pump impeller; this allows a smaller, lighter, higher-speed main pump. Figures 10–3 and 10–8 show an inducer and Ref. 10–8 describes the testing of one of them.

FIGURE 10–8. Fuel pump inducer impeller of the Space Shuttle main engine low-pressure fuel turbopump. It has a diameter about 10 in., a nominal hydrogen flow of 148.6 lbm/sec, a suction pressure of 30 psi, a discharge pressure of 280 psi at 15,765 rpm, an efficiency of 77%, and a suction specific speed of 39,000 when tested with water. (Courtesy of The Boeing Company, Rocketdyne Propulsion and Power.)

In some rockets the inert mass of the turbopump and tank system can be further reduced by putting the inducer impeller into a separate low-power, low-speed booster turbopump, driven by its own separate turbine. In the Space Shuttle main engine there are two such low-pressure-rise turbopumps, as shown in the flow diagram of Fig. 6–4 and the engine view of Fig. 6–1. This allows the inducer impeller to be operated at an optimum (lower) shaft speed.

Influence of Propellants. For the same power and mass flow, the pump head is inversely proportional to the propellant density. Since pumps are basically constant-volume flow machines, the propellant with the highest density requires less head, less power and thus allows a smaller pump assembly.

Because many of the propellants are dangerous to handle, special provision has to be made to prevent any leakage through the shaft seals. With spontaneously ignitable propellants the leakages can lead to fires in the pump compartment and may cause explosions. Multiple seals are often used with a drainage provision that safely removes or disposes of any propellants that flow past the first seal. Inert-gas purges of seals have also been used to remove hazardous propellant vapors. The sealing of corrosive propellants puts very severe requirements on the sealing materials and design. With cryogenic propellants the pump bearings are usually lubricated by the propellant, since lubricating oil would freeze at the low pump hardware temperature.

Centrifugal pumps should operate at the highest possible *pump efficiency*. This efficiency increases with the volume flow rate and reaches a maximum value of about 90% for very large flows (above 0.05 m^3/sec) and specific speeds above about 2500 (see Refs. 6–1 and 10–9). Most propellant pump efficiencies are between 30 and 70%. The pump efficiency is reduced by surface roughness of casing and impellers, the power consumed by seals, bearings, and stuffing boxes, and by excessive wear ring leakage and poor hydraulic design. The pump efficiency η_P is defined as the fluid power divided by the pump shaft power P_P:

$$\eta_P = \rho Q \, \Delta H / P_P \tag{10–9}$$

A correction factor of 550 ft-lbf/hp has to be added if P_P is given in horsepower, H in feet, and Q in ft^3/sec. When using propellants, the pump power has to be multiplied by the density ratio if the required power for water tests is to be determined.

Example 10–1. Determine the shaft speed and the overall impeller dimensions for a liquid oxygen pump which delivers 500 lb/sec of propellant at a discharge pressure of 1000 psia and a suction pressure of 14.7 psia. The oxygen tank is pressurized to 35 psia. Neglect the friction in the suction pipe and the suction head changes due to acceleration and propellant consumption. The initial tank level is 15 ft above the pump suction inlet.

SOLUTION. The density of liquid oxygen is 71.2 lbm/ft^3 at its boiling point. The volume flow will be $500/71.2 = 7.022$ ft^3/sec. The vapor pressure of the oxygen is 1 atm = 14.7 psi= 29.8 ft. The suction head is $35 \times 144/71.2 = 70.8$ ft. From Eq. 10–8 the available suction head is $70.8 + 14.7 = 85.5$ ft. The available suction head above vapor pressure is $(H_s)_A = 70.8 + 14.7 - 0 - 29.8 = 55.7$ ft. The discharge head is $1000 \times 144/71.2 = 2022$ ft. The head delivered by the pump is then $2022 - 85.5 = 1937$ ft.

The required suction head will be taken as 80% of the available suction head in order to provide a margin of safety for cavitation $(H_s)_R = 0.80 \times 85.5 = 68.4$ ft. Assume a suction specific speed of 15,000, a reasonable value if no test data are available. From Eq. 10–7 solve for the shaft speed N:

$$S = 21.2N\sqrt{Q}/(H_s)_R^{3/4} = 21.2N\sqrt{7.022}/68.4^{0.75} = 15{,}000$$

Solve for $N = 6350$ rpm or 664.7 rad/sec.

The specific speed, from Eq. 10–3, is

$$N_s = 21.2N\sqrt{Q}/H^{3/4} = 21.2 \times 6350\sqrt{7.022}/1937^{0.75} = 1222$$

According to Table 10–2, the impeller shape for this value of N_s will be a Francis type. The impeller discharge diameter D_2 can be evaluated from the tip speed by Eq. 10–4:

$$u = \psi\sqrt{2g_0\Delta H} = 1.0\sqrt{2 \times 32.2 \times 1937} = 353 \text{ ft/sec}$$
$$D_2 = 353 \times 2/664.7 = 1.062 \text{ ft} = 12.75 \text{ in.}$$

The impeller inlet diameter D_1 can be found from Eq. 10–5 by assuming a typical inlet velocity of 15 ft/sec and a shaft cross section 5.10 in.^{2} (2.548 in. diameter).

$$A = Q/v_1 = 7.022/15 = 0.468 \text{ ft}^2 = 67.41 \text{ in.}^2$$
$$A = \tfrac{1}{4}\pi D_1^2 + 5.10 = 67.41 + 5.10 = 72.51 \text{ in.}^2$$
$$D_1 = 9.61 \text{ in. (internal flow passage diameter)}$$

This is enough data to draw a preliminary sketch of the impeller.

Turbines

The turbine must provide adequate shaft power for driving the propellant pumps (and sometimes also auxiliaries) at the desired speed and torque. The turbine derives its energy from the expansion of a gaseous working fluid through fixed nozzles and rotating blades. The blades are mounted on disks to the shaft. The gas is expanded to a high, nearly tangential, velocity and through inclined nozzles and then flows through specially shaped *blades*, where the gas energy is converted into tangential forces on each blade. These forces cause the turbine wheel to rotate (see Refs.10–1 and 10–11).

Classification and Description. The majority of turbines have blades at the periphery of a turbine disk and the gas flow is axial, similarly in concept to the axial flow pattern shown for pumps in Table 10–2 and the single-stage turbine of Fig. 10–1. However, there are a few turbines with radial flow (particularly at high shaft speeds), such as the one shown in Fig. 10–2. Ideally there are two types of axial flow turbines of interest to rocket pump drives: impulse turbines and reaction turbines, as sketched in Fig. 10–9. In an *impulse turbine* the enthalpy of the working fluid is converted into kinetic energy within the first set of stationary turbine nozzles and not in the rotating blade elements. High-velocity gases are delivered (in essentially a tangential direction) to the rotating blades, and blade rotation takes place as a result of the impulse imparted by the momentum of the fluid stream of high kinetic energy to the rotating blades which are mounted on the turbine disk. The *velocity-staged impulse turbine* has a stationary set of blades which changes the flow direction after the gas leaves the first set of rototating blades and directs the gas to enter a second set of rotating blades in which the working fluid gives up further energy to the turbine wheel. In a *pressure-staged impulse turbine*, the expansion of the gas takes place in all the stationary rows of blades. In a *reaction turbine* the expansion of the gas is roughly evenly split between the rotating and stationary blade elements. The high pressure drop available for the expansion of the turbine working fluid in a gas generator cycles favors simple, lightweight one- or two-stage impulse turbines for high thrust engines. Many rocket turbines are neither pure impulse nor reaction turbines, but often are fairly close to an impulse turbine with a small reaction in the rotating vanes.

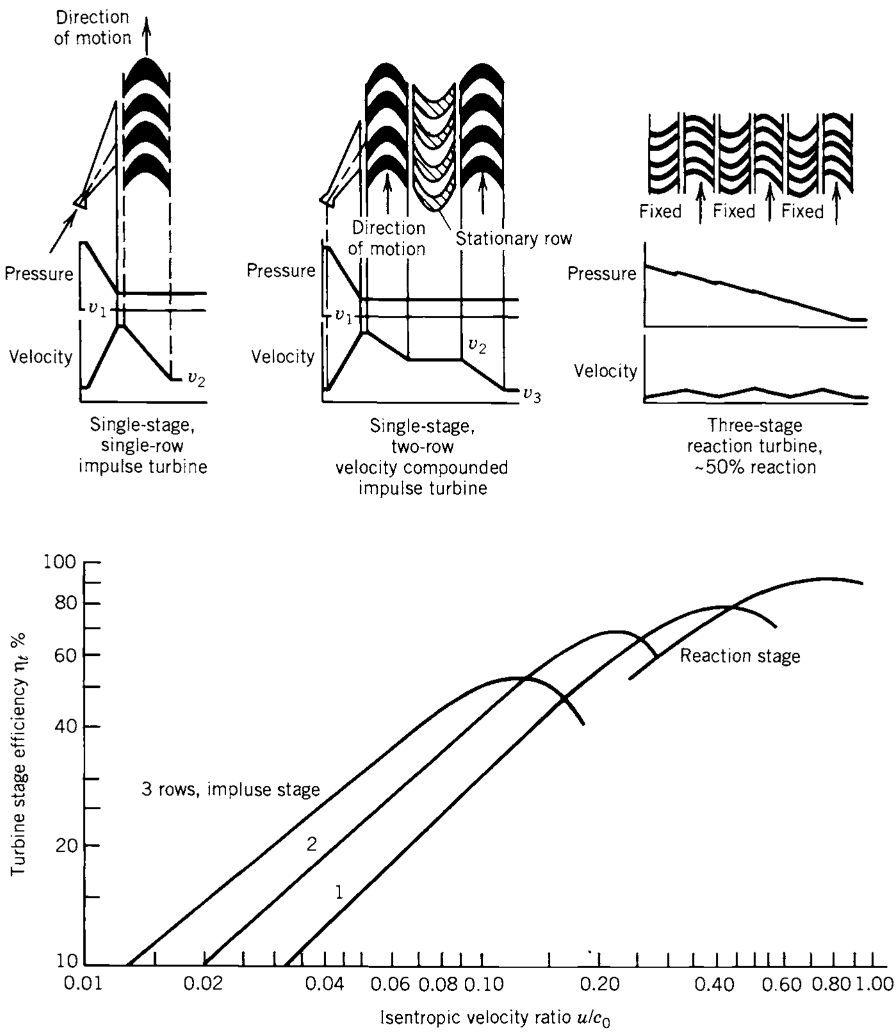


FIGURE 10–9. Top view diagram, pressure and velocity profiles, and efficiency curves for impulse and reaction type turbines. The velocity ratio is the pitch line velocity of the rotating blades u divided by the theoretical gas spouting velocity c_0 derived from the enthalpy drop. Adapted with permission from Refs. 10–1 and 10–12.

With some cycles the turbine exhaust gases pass through a *De Laval nozzle* at the exit of the exhaust pipe (see Fig. 1–4). The high turbine outlet pressure gives critical flow conditions at the venturi throat (particularly at high altitudes) and thereby assures a constant turbine outlet pressure and a constant turbine power which will not vary with altitude. Furthermore, it provides a small additional thrust to the engine.

Turbine Performance and Design Considerations. The power supplied by the turbine is given by a combined version of Eqs. 3–1 and 3–7:

$$P_T = \eta_T \dot{m}_T \Delta h \tag{10–10}$$

$$P_T = \eta_T \dot{m}_T c_p T_1 [1 - (p_2/p_1)^{(k-1)/k}] \tag{10–11}$$

The power delivered by the turbine P_T is proportional to the turbine efficiency η_T, the flow through the turbine $\dot{m}_T$, and the available enthalpy drop per unit of flow Δh. The units in this equation have to be consistent (1 Btu = 778 ft-lbf= 1055 J). This enthalpy is a function of the specific heat c_p, the nozzle inlet temperature T_1, the pressure ratio across the turbine, and the ratio of the specific heats k of the turbine gases. For gas generator cycles the pressure drop between the turbine inlet and outlet is relatively high, but the turbine flow is small (typically 2 to 5% of full propellant flow). For staged combustion cycles this pressure drop is very much lower, but the turbine flow is much larger.

For very large liquid propellant engines with high chamber pressure the turbine power can reach over 250,000 hp, and for small engines this could be perhaps around 35 kW or 50 hp.

According to Eq. 6–12, the power delivered by the turbine P_T has to be equal to the power required by the propellant pumps, the auxiliaries mounted on the turbopump (such as hydraulic pumps, electric generators, tachometers, etc.), and power losses in bearings, gears, seals, and wear rings. Usually these losses are small and can often be neglected. The effect of the turbine gas flow on the specific impulse of the rocket engine system is discussed in Sections 6.2 and 10.2. For gas generator engine cycles, the rocket designer is interested in obtaining a high turbine efficiency and a high turbine inlet temperature T_1 in order to reduce the flow of turbine working fluid, and for gas generator cycles also to raise the overall effective specific impulse, and, therefore, reduce the propellant mass required for driving the turbine. Three-dimensional computer analyses of the gas flow behavior and turbine blade geometry have resulted in efficient blade designs.

Better turbine blade materials (such as single crystals which have been unidirectionally solidified) and specialty alloys can allow turbine inlet temperatures between 1400 K (or about 2050°F) and perhaps 1600 K (or 2420°F); these higher temperatures or higher gas enthalpies reduce the required turbine flow. Reliability and cost considerations have kept actual turbine inlet temperatures at conservative values, such as 1150 to 1250°F or about 900 to 950 K, using lower cost steel alloy as the material. The *efficiency* of turbines for rocket turbopumps is shown in Fig. 10–9. Maximum *blade speeds* with good design and strong high-temperature materials are typically 400 to 700 m/sec or about 1300 to 2300 ft/sec. Higher blade speeds generally allow an improvement in efficiency. For the efficiency to be high the turbine blade and nozzle profiles have to have smooth surfaces. Small clearances at the turbine blade tips are also needed, to minimize leakage.

The low efficiency in many rocket turbines is dictated by centrifugal *pump design considerations*, which *limit the shaft speed* for turbopumps in which the pump and turbine are mounted on a common shaft, as discussed in the next section. A low shaft speed together with minimum mass requirements, which prohibit a very large turbine wheel diameter, give a low blade speed, which in turn reduces the efficiency.

The advantage of increased turbine efficiency (less gas generator propellant requirement) can be realized only if the turbopump design allows high blade speeds. This can be achieved in rockets of medium and low thrust by gearing the turbine to the pumpshaft or by using pumps that permit high shaft speeds; in rockets of very high thrust the pumps have diameters and shaft speeds close to those of the turbines and can be mounted on the same shaft as the turbine. The power input to the turbine can be regulated by controlling the flow to the turbine inlet. This can be accomplished by throttling or by-passing some of the flow of the working fluid to the turbine and varying the turbine inlet pressure.

There is *no warm-up time* available in rocket turbines. The sudden admission of hot gas at full flow causes severe thermal shock and thermal distortion and increases the chances for rubbing between moving metal parts. The most severe stresses of a turbine blade often are thermal stresses; they come during the engine start when the leading edge is very hot but other parts of the blade are still cold. This and other loading conditions can be more severe in rocket turbines than in air-burning gas turbines.

For low-thrust engines the shaft speeds can become very high, such as over 100,000 rpm. Also, the turbine blade height becomes very short and friction losses can become prohibitive. In order to obtain a reasonable blade height we go to partial admission turbine designs. Here a portion of the turbine nozzles are effectively plugged or eliminated.

Gas Generators and Preburners

A gas generator is used as the source of hot gas (from combustion of propellants) for driving many of the turbines of turbopumps in a liquid rocket engine. Depending on the engine cycle, other sources of turbine drive gases are sometimes employed, as described in Section 6.3.

Gas generators can be classified as monopropellant, bipropellant, or solid propellant. Actually the basic design parameters for gas generators are similar to those for engine thrust chambers or solid rocket motors. The combustion temperature is usually kept below 1400 to 1600 K (or 2000 to 2400°F) by intentionally regulating or mixing the propellants in proportions substantially different from stoichiometric mixture, usually fuel rich. These lower gas temperatures allow uncooled chamber construction and prevent melting or limit the erosion or turbine blades. With monopropellants, such as hydrogen peroxide (H_2O_2) or hydrazine (N_2H_4), the flow is easily controlled and the gases are generated at predictable temperatures depending on the details of the catalyst and the gas generator design. In principle a gas generator looks like an uncooled rocket thrust chamber except that the nozzle is replaced by a pipe leading to the turbine nozzles.

Propellants supplied to the liquid propellant gas generators can come from separate pressurized tanks or can be tapped off from the engine propellant pumps. When starting pump-fed gas generators, the turbomachinery needs to be brought up to operating speed. This can be done by a solid propellant gas generator starter, an auxiliary pressurized propellant supply, or by letting the engine "bootstrap" itself into a start using the liquid column head existing in the vehicle tankage and feed system lines—usually called a "tank-head" start.

Gas generators have been used for *other applications* besides supplying power to rocket feed systems. They have a use wherever there is a need for a large amount of power for a relatively short time, because they are simpler and lighter than conventional short-duration power equipment. Typical applications are gas generators for driving torpedo turbines and gas generators for actuating airplane catapults.

In a staged combustion cycle all of one propellant and a small portion of the other propellant (either fuel-rich or oxidizer-rich mixture) are burned to create the turbine drive gases. This combustion device is called a *preburner* and it is usually uncooled. It has a much larger flow than the gas generators mentioned above, its turbines have a much smaller pressure drop, and the maximum pressure of the propellants is higher.

10.2. PERFORMANCE OF COMPLETE OR MULTIPLE ROCKET PROPULSION SYSTEMS

The simplified relations that follow give the basic method for determining the overall specific impulse, the total propellant flow, and the overall mixture ratio as a function of the corresponding component performance terms for complete rocket engine systems. This applies to engine systems consisting of one or more thrust chambers, auxiliaries, gas generators, turbines, and evaporative propellant pressurization systems all operating at the same time.

Refer to Eqs. 2–5 and 6–1 for the specific impulse I_s, propellant flow rate $\dot{w}$ or $\dot{m}$ and mixture ratio r. The overall thrust F_{oa} is the sum of all the thrusts from thrust chambers and turbine exhausts and the overall flow $\dot{m}$ is the sum of their flows. The subscripts oa, o, and f designate the overall engine system, the oxidizer, and the fuel, respectively. Then

$$(I_s)_{oa} = \frac{\sum F}{\sum \dot{w}} = \frac{\sum F}{g_0 \sum \dot{m}} \tag{10–12}$$

$$\dot{w}_{oa} = \sum \dot{w} \qquad \text{or} \qquad \dot{m}_{oa} = \sum \dot{m} \tag{10–13}$$

$$r_{oa} \sim \frac{\sum \dot{w}_o}{\sum \dot{w}_f} = \frac{\sum \dot{m}_o}{\sum \dot{m}_f} \tag{10–14}$$

These same equations should be used for determining the overall performance when more than one rocket engine is contained in a vehicle propulsion system and they are operating simultaneously. They also apply to multiple solid propellant rocket motors and combinations of liquid propellant rocket engines and solid propellant rocket booster motors, as in the Space Shuttle (see Fig. 1–13).

Example 10–2. For an engine system with a gas generator similar to the one shown in Fig. 1–4, determine a set of equations that will express (1) the overall engine performance and (2) the overall mixture ratio of the propellant flows from the tanks. Let the following subscripts be used: c, thrust chamber; gg, gas generator; and tp, tank pressurization. For a nominal burning time t, a 1% residual propellant, and a 6% overall reserve factor, give a formula for the amount of fuel and oxidizer propellant required with constant propellant flow. Ignore stop and start transients, thrust vector control, and evaporation losses.

SOLUTION. Only the oxidizer tank is pressurized by vaporized propellant. Although this pressurizing propellant must be considered in determining the overall mixture ratio, it should not be considered in determining the overall specific impulse since it stays with the vehicle and is not exhausted overboard.

$$(I_s)_{oa} \sim \frac{F_c + F_{gg}}{(\dot{m}_c + \dot{m}_{gg})g_0} \tag{10–15}$$

$$r_{oa} \sim \frac{(\dot{m}_o)_c + (\dot{m}_o)_{gg} + (\dot{m}_o)_{tp}}{(\dot{m}_f)_c + (\dot{m}_f)_{gg}} \tag{10–16}$$

$$m_f = [(\dot{m}_f)_c + (\dot{m}_f)_{gg}]\ t\ (1.00 + 0.01 + 0.06)$$
$$m_o = [(\dot{m}_o)_c + (\dot{m}_o)_{gg} + (\dot{m}_o)_{tp}]\ t\ (1.00 + 0.01 + 0.06)$$

For this gas generator cycle the engine mixture ratio or r_{oa} is different from the thrust chamber mixture ratio $r_c = (m_o)_c/(m_f)_c$. Similarly, the overall engine specific impulse is slightly lower than the thrust chamber specific impulse. However, for an expander cycle or a staged combustion cycle these two mixture ratios and two specific impulses are the same, provided that there are no gasified propellant used for tank pressurization.

The overall engine specific impulse is influenced by the nozzle area ratio and the chamber pressure, and to a lesser extent by the engine cycle, and the mixture ratio. Table 10–3 describes 11 different rocket engines using liquid oxygen and liquid hydrogen propellants designed by different companies in different countries, and shows the sensitivity of the specific impulse to these parameters. References 10–13 to 10–15 give additional data on several of these engines.

TABLE 10–3. Comparison of Rocket Engines Using Liquid Oxygen and Liquid Hydrogen Propellants

Engine Designation Engine Cycle, Manuf. or Country (Year Qualified)	Vehicle	Thrust in Vacuum, kN (lbf)	Specific Impulse in Vacuum (sec)	Chamber Pressure, bar (psia)	Mixture Ratio	Nozzle Area Ratio	Engine Mass (dry), kg
SSME, staged combustion, Rocketdyne (1998)	Space Shuttle (3 required)	2183 (490,850)	452.5	196 (2870)	6.0	68.8	3400
RS-68, gas generator, Rocketdyne (2000)	Delta	3313 (745,000)	415	97.2 (1410)	6.0	21.5	6800
LE-5A, Expander bleed, MHI, Japan, (1991)	HII	121.5 (27,320)	452	37.2 (540)	5.0	130	255
LE-7, staged combustion, MHI, Japan (1992)	HII	1080 (242,800)	445.6	122 (1769)	6.0	52	1720
Vulcain, gas generator, SEP and other European Co.'s	Ariane 5	1120 (251,840)	433	112 (1624)	5.35	45	1585
HM7, gas generator, SEP France	Ariane 1,2,3,4	62.7 (14,100)	444.2	36.2 (525)	5.1	45	155
RL 10-A3, Expander, Pratt & Whitney (1965)	Various upper stages	73.4 (16,500)	444.4	32.75 (475)	5.0	61	132
RL 10-B2, (1998), same as above	—	110 (24,750)	466.5	44.12 (640)	6.0	375	275
YF 73, China	Long March	44,147 (10,000)	420	26.28 (381)	5.0	40	236
YF 75 (2 required), China		78.45 (17,600)	440	36.7 (532)	5.0	80	550

10.3. PROPELLANT BUDGET

In all liquid propellant rocket engines some of the propellants are used for purposes other than producing thrust or increasing the velocity of the vehicle. This propellant must also be included in the propellant tanks. A propellant budget can include the eleven items listed below, but very few engines have allowances for all these items. Table 10–4 shows an example of a budget for a spacecraft pressure-fed engine system with several small thrusters and one larger thrust chamber.

1. Enough propellant has to be a available for achieving the *required vehicle velocity increase* of the particular application and the particular flight vehicle or stage. The nominal velocity increment is usually defined by systems analysis and mission optimization using an iterative calculation based on Eqs. 4–19 or 4–35. If there are alternative flight paths or missions for the same vehicle, the mission with an unfavorable flight path and the highest total impulse should be selected. This mission-required propellant is the largest portion of the total propellants loaded into the vehicle tanks.
2. In a turbopump system using a gas generator cycle, a small portion of the overall propellant is burned in a separate *gas generator*. It has a lower flame temperature than the thrust chamber and operates at a different mixture ratio; this causes a slight change in the overall mixture ratio of propellants flowing from the tanks, as shown by Eqs. 10–14 and 10–16.

TABLE 10–4. Example of a Propellant Budget for a Spacecraft Propulsion System with a Pressurized Monopropellant Feed System

Budget Element	Typical Value
1. Main thrust chamber (increasing the velocity of stage or vehicle)	70–90% (determined from mission analysis and system engineering)
2. Flight control function (for reaction control thrusters and flight stability)	5–15% (determined by control requirements)
3. Residual propellant (trapped in valves, lines, tanks, etc.)	0.5–2% of total load[a]
4. Loading uncertainty	0.5% of total load[a]
5. Allowance for off-nominal performance	0.5–1.0% of total load[a]
6. Allowance for off-nominal operations	0.25–1.0% of total load[a]
7. Mission margin (reserve for first two items above)	3–10% of items 1 and 2
8. Contingency	2–10% of total load[a]

Source: Adapted from data supplied by TRW, Inc.
[a]Total load is sum of items 1, 2, and 7.

3. In a rocket propulsion system with a *thrust vector control* (TVC) system, such as a swiveling thrust chamber or nozzle, the thrust vector will be rotated by a few degrees. Thrust vector control systems are described in Chapter 16. There is a slight decrease in the axial thrust and that reduces the vehicle velocity increment in item 1. The extra propellant needed to compensate for the small velocity reduction can be determined from the mission requirements and TVC duty cycle. It could be between 0.1 and 4% of the total propellant.
4. In some engines a small portion of cryogenic propellants is heated, vaporized, and used to *pressurize cryogenic propellant tanks*. A heat exchanger is used to heat liquid oxygen from the pump discharge and pressurize the oxygen tank, as shown schematically in Fig. 1–4. This method is used in the hydrogen and oxygen tanks of the Space Shuttle external tank (see Ref. 6–6).
5. Auxiliary rocket engines that provide for *trajectory corrections*, *station keeping*, *maneuvers*, or *attitude control* usually have a series of small restartable thrusters (see Chapter 4). The propellants for these auxiliary thrusters have to be included in the propellant budget if they are supplied from the same feed system and tanks as the larger rocket engine. Depending on the mission and the propulsion system concept, this auxiliary propulsion system can consume a significant portion of the available propellants.
6. The *residual propellant* that clings to tank walls or remains trapped in valves, pipes, injector passages, or cooling passages is unavailable for producing thrust. It is typically 0.5 to 2% of the total propellant load. It increases the final vehicle mass at thrust termination and reduces the final vehicle velocity slightly.
7. A *loading uncertainty* exists due to variations in tank volume or changes in propellant density or liquid level in the tank. This is typically 0.25 to 0.75% of the total propellant. It depends, in part, on the accuracy of the method of measuring the propellant mass during loading (weighing the vehicle, flow meters, level gages, etc.).
8. The *off-nominal rocket performance* is due to variations in the manufacture of hardware from one engine to another (such as slightly different pressure losses in a cooling jacket, in injectors and valves, or somewhat different pump characteristics); these cause slight changes in combustion behavior, mixture ratio, or specific impulse. If there are slight variations in *mixture ratio*, one of the two liquid propellants will be consumed fully and an unusable residue will remain in the other propellant's tank. If a minimum total impulse requirement has to be met, extra propellant has to be tanked to allow for these mixture ratio variations. This can amount up to perhaps 2.0% of one of the propellants.
9. *Operational factors* can result in additional propellant requirements, such as filling more propellant than needed into a tank or incorrectly,

adjusting regulators or control valves. It can also include the effect of changes in flight acceleration from the nominal value. For an engine that has been carefully calibrated and tested, this factor can be small, usually betwen 0.1 and 1.0%.

10. When using cryogenic propellants an *allowance for evaporation and cooling down* has to be included. It is the mass of extra propellant that is allowed to evaporate (and be vented overboard while the vehicle is waiting to be launched) or that is fed through the engine to cool it down, before the remaining propellant in the tank becomes less than the minimum needed for the flight mission. Its quantity depends on the amount of time between topping off (partial refilling) of the tank.
11. Finally, an *overall contingency* or ignorance factor is needed to allow for unforeseen propellant needs or inadequate or uncertain estimates of any of the items above. This can also include allowances for vehicle drag uncertainties, variations in the guidance and control system, wind, or leaks.

Only some of the items above provide axial thrust (items 1, 2, and sometimes also 3 and 5), but all the items need to be considered in determining the total propellant mass and volume.

10.4. ENGINE DESIGN

The approach, methods, and resources used for rocket engine preliminary design and final design are usually different for each design organization and for each major type of engine. They also differ by the degree of novelty.

1. *A totally new engine with new major components* and some *novel design concepts* will result in an optimum engine design for a given application, but it is usually the most expensive and longest development approach. One of the major development costs is usually in sufficient testing of components and several engines (under various environmental and performance limit conditions), in order to establish credible reliability data with enough confidence to allow the initial flights and initial production. Since the state of the art is relatively mature today, the design and development of a truly novel engine does not happen very often.
2. *New engine using major components or somewhat modified key components from proven existing engines.* This is a common approach today. The design of such an engine requires working within the capability and limits of existing or slightly modified components. It requires much less testing for proving relability.
3. *Uprated or improved version of an existing, proven engine.* This approach is quite similar to the second. It is needed when an installed engine for a given mission requires more payload (which really means higher thrust)

and/or longer burning duration (more total impulse). Uprating often means more propellant (larger tanks), higher propellant flows and higher chamber and feed pressures, and more feed system power. The engine usually has an increased inert engine mass (thicker walls).

In a simplified way, we describe here a typical process for designing an engine. At first the basic function and requirements of the new engine must be established. These engine requirements are derived from the vehicle mission and vehicle requirements, usually determined by the customer and/or the vehicle designers, often in cooperation with one or more engine designers. The engine requirements can include key parameters such as thrust level, the desired thrust–time variation, restart or pulsing, altitude flight profile, environmental conditions, engine locations within the vehicle, and limitations or restraints on cost, engine envelope, test location, or schedule. It also includes some of the factors listed later in Table 17–5. If an existing proven engine can be adapted to these requirements, the subsequent design process will be simpler and quite different than the design of a truly new engine.

Usually some early tentative decisions about the engine are made, such as the selection of the propellants, their mixture ratio, or the cooling approach for the hot components. They are based on mission requirements, customer preferences, past experiences, some analysis, and the judgement of the key decision makers. Some additional selection decisions include the engine cycle, having one, two, or more thrust chambers fed from the same feed system, redundancy of auxiliary thrusters, or type of ignition system. Trade-off studies between several options are appropriate at this time. With a modified existing engine these parameters are well established, and require few trade-off studies or analyses. Initial analyses of the pressure balances, power distribution between pumps and turbines, gas generator flow, propellant flows and reserves, or the maximum cooling capacity are appropriate. Sketches and preliminary estimates of inert mass of key components need to be made, such as tanks, thrust chambers, turbopumps, feed and pressurization systems, thrust vector control, or support structure. Alternate arrangements of components (layouts) are usually examined, often to get the most compact configuration. An initial evaluation of combustion stability, stress analysis of critical components, water hammer, engine performance at some off-design conditions, safety features, testing requirements, cost, and schedule are often performed at this time. Participation of appropriate experts from the field of manufacturing, field service, materials, stress analysis, or safety can be critical for selecting the proper engine and the key design features. A design review is usually conducted on the selected engine design and the rationale for new or key features.

Test results of subscale or full-scale components, or related or experimental engines, will have a strong influence on this design process. The key engine selection decisions need to be validated later in the development process by testing new components and new engines.

The *inert mass of the engine and other mass properties (center of gravity or moment of inertia)* are key parameters of interest to the vehicle designer or customer. They are needed during preliminary design and again, in more detail, in the final design. The engine mass is usually determined by summing up the component or subsystem masses, each of which is either weighed or estimated by calculating their volumes and knowing or assuming their densities. Sometimes early estimates are based on known similar parts or subassemblies.

Preliminary engine performance estimates are often based on data from prior similar engines. If these are not available, then theoretical performance values can be calculated (see Chapter 2, 3, and 5) for F, I_s, k, or $\mathfrak{M}$, using appropriate correction factors. Measured static test data are, of course, better than estimates. The final performance values are obtained from flight tests or simulated altitude tests, where airflow and altitude effects can interact with the vehicle or the plume.

If the preliminary design does not meet the engine requirements, then changes need to be made to the initial engine decisions and, if that is not sufficient, sometimes also to the mission requirements themselves. Components, pressure balances, and so forth will be reanalyzed and the results will be a modified version of the engine configuration, its inert mass, and performance. This process is iterated until the requirements are met and a suitable engine has been found. The initial design effort culminates in preliminary layouts of the engine, a preliminary inert mass estimate, an estimated engine performance, a cost estimate, and a tentative schedule. These preliminary design data form the basis for a written proposal to the customer for undertaking the final or detail design, development, testing, and for delivering engines.

Optimization studies are made to select the best engine parameters for meeting the requirements; some of them are done before a suitable engine has been identified, some afterwards. They are described further in Section 10.7. We optimize parameters such as chamber pressure, nozzle area ratio, thrust, mixture ratio, or number of large thrust chambers supplied by the same turbopump. The results of optimization studies indicate the best parameter, which will give a further, usually small, improvement in vehicle performance, propellant fraction, engine volume, or cost.

Once the engine proposal has been favorably evaluated by the vehicle designers, and after the customer has provided authorization and funding to proceed, then the final design can begin. Some of the analyses, layouts, and estimates will be repeated, but in more detail, specifications and manufacturing documents will be written, vendors will be selected, and tooling will be built. The selection of some of the key parameters (particularly those associated with some technical risk) will need to be validated. After another design review, key components and prototype engines are built and ground tested as part of a planned development effort. If proven reliable, one or two sets of engines will be installed in a vehicle and operated during flight. In those programs where a fair number of vehicles are to be built, the engine will then be produced in the required quantity.

Table 10–5 shows some of the characteristics of three different Russian designs staged combustion cycle engine designs, each at a different thrust and with different propellants (see Ref. 10–17). It shows primary engine para-

TABLE 10–5. Data on Three Russian Large Liquid Propellant Rocket Engines Using a Staged Combustion Cycle

Engine Designation	RD-120	RD-170	RD-253
Application (number of engines)	Zenit second stage (1)	Energia launch vehicle booster (4) and Zenit first stage (1)	Proton vehicle booster (1)
Oxidizer	Liquid oxygen	Liquid oxygen	N_2O_4
Fuel	Kerosene	Kerosene	UDMH
Number and types of turbopumps (TP)	One main TP and two boost TPs	One main TP and two boost TPs	Single TP
Thrust control, %	Yes	Yes	±5
Mixture ratio control, %	±10	±7	±12
Throttling (full flow is 100%), %	85	40	None
Engine thrust (vacuum), kg	85,000	806,000	167,000
Engine thrust (SL), kg	—	740,000	150,000
Specific impulse (vacuum), sec	350	337	316
Specific impulse (SL), sec	—	309	285
Propellant flow, kg/sec	242.9	2393	528
Mixture ratio, O/F	2.6	2.63	2.67
Length, mm	3872	4000	2720
Diameter, mm	1954	3780	1500
Dry engine mass, kg	1125	9500	1080
Wet engine mass, kg	1285	10500	1260
	Thrust Chamber Characteristics		
Chamber diameter, mm	320	380	430
Characteristic chamber length, mm	1274	1079.6	999.7
Chamber area contraction ratio	1.74	1.61	1.54
Nozzle throat diameter, mm	183.5	235.5	279.7
Nozzle exit diameter, mm	1895	1430	1431
Nozzle area ratio,	106.7	36.9	26.2
Thrust chamber length, mm	2992	2261	2235
Nominal combustion temperature, K	3670	3676	3010
Rated chamber pressure, kg/cm^2	166	250	150
Nozzle exit pressure, kg/cm^2	0.13	0.73	0.7
Thrust coefficient, vacuum	1.95	1.86	1.83
Thrust coefficient, SL	—	1.71	1.65
Gimbal angle, degree	Fixed	8	Fixed
Injector type	Hot, oxidizer-rich precombustor gas plus fuel		

With a staged combustion cycle the thrust, propellant flow, and mixture ratio for the thrust chamber have the same values as for the entire engine.

TABLE 10–5. (*Continued*)

Engine Designation	RD-120		RD-170		RD-253	
	Turbopump Characteristics					
Pumped liquid	*Oxidizer*	*Fuel*	*Oxidizer*	*Fuel*	*Oxidizer*	*Fuel*
Pump discharge pressure, kg/cm^2	347	358	614	516	282	251
Flow rate, kg/sec	173	73	1792	732	384	144
Impeller diameter, mm	216	235	409	405	229	288
Number of stages	1	1	1	1 + 1[a]	1	1 + 1[a]
Pump efficiency, %	66	65	74	74	68	69
Pump shaft power, hp	11,210	6145	175,600	77,760	16,150	8850
Required pump NPSH, m	37	23	260	118	45	38
Shaft speed, rpm	19,230		13,850		13,855	
Pump impeller type	Radial flow		Radial flow		Radial flow	
Turbine power, hp	17,588		257,360		25,490	
Turbine inlet pressure, main turbine, kg/cm^2	324		519		239	
Pressure ratio	1.76		1.94		1.42	
Turbine inlet temperature, K	735		772		783	
Turbine efficiency, %	72		79		74	
Number of turbine stages	1		1		1	
	Preburner Characteristics					
Flow rate, kg/sec	177		836		403.5	
Mixture ratio, O/F	53.8		54.3		21.5	
Chamber pressure, kg/cm^2	325		546		243	
Number of preburners	1		2		1	

[a]Fuel flow to precombustor goes through a small second-stage pump. (Courtesy of NPO Energomash, Moscow.)

meters (chamber pressure, thrust, specific impulse, weight, propellant combination, nozzle area ratio, dimensions, etc.) which influence the vehicle performance and configuration. It also shows secondary parameters, which are internal to the engine but important in component design and engine optimization. The Space Shuttle main engine (see Figs. 6–1 and 6–12) has two fuel-rich preburners, but the Russian engines use oxidizer-rich preburners. Figure 10–10 shows the RD-170 engine with four thrust chambers (and their thrust vector actuators) supplied by a centrally located single large turbopump (257,000 hp; not visible in the photo) and one of the two oxidizer-rich preburners. The flow diagram of Fig. 10–11 shows this turbopump and the two booster turbopumps; one is driven by a turbine using a bleed of oxygen-rich gas from the turbine exhaust (the gas is condensed when it mixes with the liquid oxygen flow) and the other by a liquid turbine using high-pressure liquid fuel.

Much of today's engine design, preliminary design and design optimization can be performed on computer programs. These include infinite element analyses, codes for stress and heat transfer, weight and mass properties, stress and strain analysis of a variety of structures, water hammer, engine performance analyses, feed system analyses (for balance of flow, pressures, and power), gas

FIGURE 10–10. The RD-170 rocket engine, shown here on a transfer cart, can be used as an expendable or reusable engine (up to 10 flights). It has been used on the Zenith, Soyuz booster, and Energiya launch vehicles. The tubular structure supports the four hinged thrust chambers and its control actuators. It is the highest thrust liquid rocket engine in use today. (Courtesy of NPO Energomash, Moscow.)

pressurization, combustion vibrations, and various exhaust plume effects (see Ref. 10–16). Some customers require that certain analyses (e.g., safety, static test performance) be delivered to them prior to engine deliveries.

Many computer programs are specific to a particular design organization, a certain category of application (e.g., interplanetary flight, air-to-air combat, long-range ballistic missile, or ascent to earth orbit), and many are specific to a particular engine cycle. One is called *engine balance program* and it balances the pressure drops in the fuel, oxidizer, and pressurizing gas flow systems; similar programs balance the pump and turbine power, speeds, and torques (see

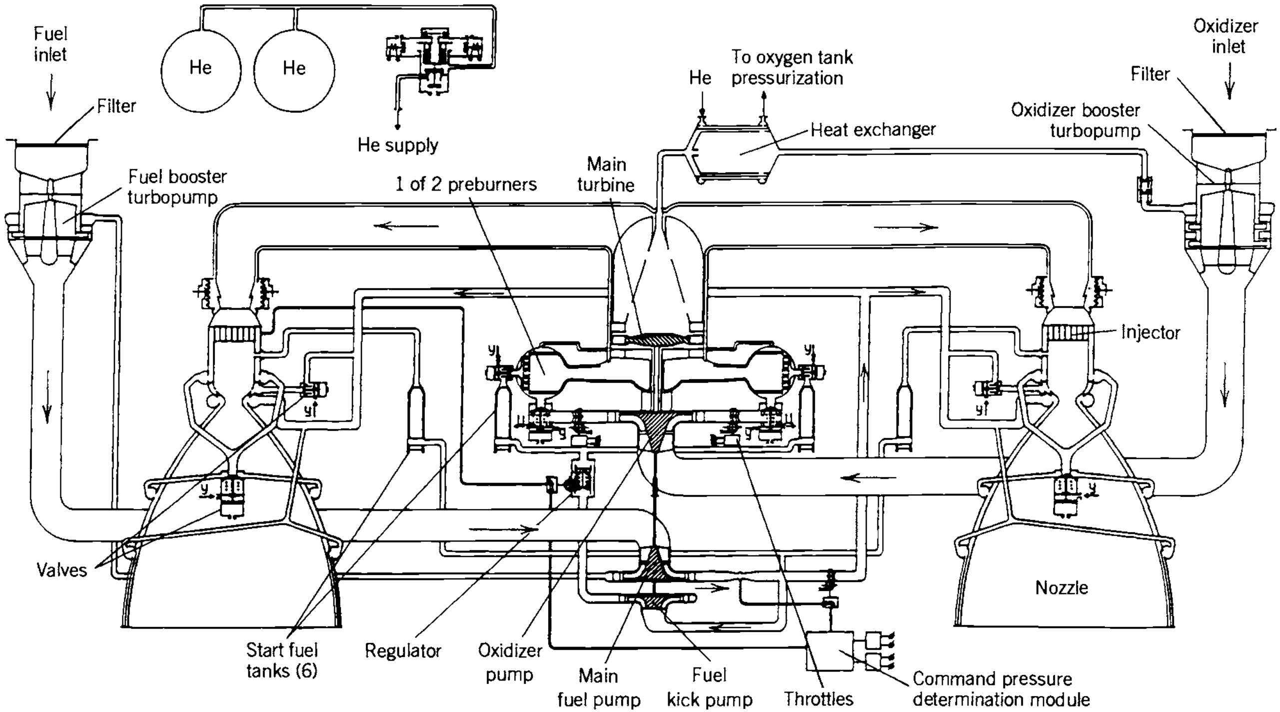


FIGURE 10–11. Simplified flow diagram of the RD-170 high-pressure rocket engine. The single-shaft large turbopump has a single-stage reaction turbine, two fuel pumps, and a single-stage oxygen pump with an inducer impeller. All of the oxygen and a small portion of the fuel flow supply two oxidizer-rich preburners. Only two of the four thrust chambers are shown in this diagram. The two booster pumps prevent cavitation in the main pumps. The pressurized helium subsystem (only shown partially) supplies various actuators and control valves; it is indicated by the symbol y. Ignition is accomplished by injecting a hypergolic fuel into the two preburners and the four thrust chambers. (Courtesy of NPO Energomash, Moscow.)

Section 10.7), compare different turbopump configurations (see Section 10.1); some balance programs also calculate approximate masses for engine, tanks, turbine drive fluids. The program allows iterations of various pressures and pressure drops, mixture ratios, thrust levels, number of thrust chambers, distribution of total velocity increment between different vehicle stages, trades between constant thrust (or propellant flow) and decreasing thrust (throttling) or pulsed (intermittent) thrust.

10.5. ENGINE CONTROLS

All liquid propellant rocket engines have controls to accomplish some or all of the following tasks:

1. Start rocket operation
2. Shut down rocket operation.
3. Restart, if desired.
4. Maintain programmed operation (predetermined constant or randomly varied thrust, preset propellant mixture ratio and flow).
5. When safety devices sense an impending malfunction or a critical condition of the vehicle or the engine, the control system will automatically change the engine operating conditions to remedy the defected defect, or cause a safe emergency engine shutdown. Only some of the likely failure modes can be remedied by sensing a potential problem and initiating a remedial action. Some failures occur so rapidly that there is not enough time to counteract them. Others are too difficult to identify reliably as a failure and others are not well understood.
6. Fill with propellants.
7. Drain excess propellant after operation.
8. With cryogenic propellants the pipes, pumps, cooling jackets, injectors, and valves have to be cooled to the cryogenic fluid temperature prior to start, by bleeding cold propellant through them; this cooling propellant is not used to produce thrust. Its periodic flow has to be controlled.
9. Check out proper functioning of critical components or a group of components without actual hot operation before and/or after flight.
10. For recoverable and reusable rocket engines, also provide built-in self-test features to perform continuous checks in flight and on the ground and recycle the engine to a ready condition within a few minutes after a launch abort without any ground servicing.

The *complexity* of these control elements and the complexity of the engine systems depend very much on the mission of the vehicle. In general, rockets that are used only once (single-shot devices), that are filled with propellants at the factory, and that have to operate over a narrow range of environmental conditions tend to be simpler than rocket systems intended for repeated use, for applications where satisfactory operation must be demonstrated prior to use, and for manned vehicles. Because of the nature of the liquid propellants, most of the control actuation functions are achieved by valves, regulators, pressure switches, and flow controls. The use of special computers for automatic control in large engines is now common. The flow control devices, namely the valves, were discussed in Section 6.9.

Safety controls are intended to protect personnel and equipment in case of malfunction. For example, the control system is usually so designed that a failure of the electrical power supply to the rocket causes a nonhazardous shutdown (all electrical valves automatically returning to their normal position) and no mixing or explosion of unreacted propellant can occur. Another example is an electrical interlock device which prevents the opening of the main propellant valves until the igniter has functioned properly.

Check-out controls permit a simulation of the operation of critical control components without actual hot operation of the rocket unit. For example, many rockets have provisions for permitting actuation of the principal valves without having propellant or pressure in the system.

Control of Engine Starting and Thrust Buildup

In the *starting* and *stopping* process of a rocket engine, it is possible for the mixture ratio to vary considerably from the rated design mixture ratio because of a lead of one of the propellants and because the hydraulic resistances to propellant flow are not the same for the fuel and the oxidizer passages. During this transition period it is possible for the rocket engine to pass through regions of chamber pressure and mixture ratio which can permit combustion instability. The starting and stopping of a rocket engine is very critical in timing, valve sequencing, and transient characteristics. A good control system must be designed to avoid undesirable transient operation. Close *control* of the *flow* of propellant of the *pressure*, and of the *mixture ratio* is necessary to obtain reliable and repeatable rocket performance. The starting and ignition of thrust chambers has been discussed in Section 8.4.

Fortunately, most rocket units operate with a nearly constant propellant consumption and a constant mixture ratio, which simplifies the operating control problem. Stable operation of liquid propellant flows can be accomplished without automatic control devices because the liquid flow system in general tends to be inherently stable. This means that the hydraulic system reacts to any disturbance in the flow of propellant (a sudden flow increase or decrease) in such a manner as to reduce the effect of the disturbance. The system, therefore, usually has a natural tendency to control itself. However, in some cases the natural resonances of the system and its components can have frequency values that tend to destabilize the system.

The *start delay time* for a pressure feed system is usually small. Prior to start, the pressurization system has to be activated and the ullage volume has to be pressurized. This start delay is the time to purge the system if needed, open valves, initiate combustion, and raise the flow and chamber pressure to rated values. A turbopump system usually requires more time to start. In addition to the foregoing starting steps for a pressurized system, it has to allow a time period for starting a gas generator or preburner and for bringing the turbopumps up to a speed at which combustion can be sustained and thereafter up to full flow. If the propellant is nonhypergolic, additional time has to be allowed for the igniter to function and for feedback to confirm that it is working properly. All these events need to be controlled. Table 10–6 describes many of these steps.

Starting of small thrusters with a pressurized feed system can be very fast, as short as 3 to 15 millisec, enough time for a small valve to open, the propellant to flow to the chamber and to ignite, and the small chamber volume to be filled with high-pressure combustion gas. For turbopump-fed systems and larger thrust engines, the time from start signal to full chamber pressure is longer, about 1 to 5 sec, because the pump rotors have inertia, the igniter flame has to heat a relatively large mass of propellants, the propellant line volumes to be filled are large, and the number of events or steps that need to take place is larger.

Large turbopump-fed rocket engines have been started in at least four ways:

1. A *solid propellant start grain* or start cartridge is used to pressurize the gas generator or preburner, and this starts turbine operations. This method is used on Titan III hypergolic propellant rocket engines (first and second stages) and on the H-1 (nonhypergolic), where the start grain flame also ignites the liquid propellants in the gas generator. This is usually the fastest start method, but it does not provide for a restart.
2. This method, known as *tank head start*, is used on the SSME, is slower, does not require a start cartridge, and permits engine restart. The head of liquid from the vehicle tanks (usually in vertically launched large vehicles) plus the tank pressure cause a small initial flow of propellants; then slowly more pressure is built up as the turbine begins to operate and in a couple of seconds the engine "bootstraps" its flows and the pressures then rise to their rated values.
3. A small *auxiliary pressurized propellant feed system* is used to feed the initial quantity of fuel and oxidizer (at essentially full pressure) to the thrust chamber and gas generator. This method was used on the RS-27 engine in the first stage of a Delta II space launch vehicle.
4. The *spinner start* method uses clean high-pressure gas from a separate tank to spin the turbine (usually at less than full speed) until the engine provides enough hot gas to drive the turbine. The high-pressure tank is heavy, the connections add complexity, and this method is seldom used today.

TABLE 10–6. Major Steps in the Starting and Stopping of a Typical Large Liquid Propellant Rocket Engine with a Turbopump Feed System

1. *Prior to Start*

Check out functioning of certain components (without propellant flow), such as the thrust vector control or some valve actuators.

Fill tanks with propellants.

Bleed liquid propellants to eliminate pockets of air or gas.

When using propellants that can react with air (e.g., hydrogen can freeze air, small solid air crystals can plug injection holes, and solid air with liquid hydrogen can form an explosive mixture), it is necessary to purge the piping system (including injector, valves and cooling jacket) with an inert, dry gas (e.g., helium) to remove air and moisture. In many cases several successive purges are undertaken.

With cryogenic propellants the piping system needs to be cooled to cryogenic temperatures to prevent vapor pockets. This is done by repeated bleeding of cold propellant through the engine system (valves, pumps, pipes, injectors, etc.) just prior to start. The vented cold gas condenses moisture droplets in the air and this looks like heavy billowing clouds escaping from the engine.

Refill or "top off" tank to replace cryogenic propellant that has evaporated or been used for cooling the engine.

Pressurize vehicle's propellant tanks just before start.

2. *Start: Preliminary Operation*

Provide start electric signal, usually from vehicle control unit or test operator.

With nonhypergolic propellants, start the ignition systems in gas generator or preburner and main chambers; for nonhypergolic propellants a signal has to be received that the igniter is burning before propellants are allowed to flow into the chambers.

Initial operation: opening of valves (in some cases only partial opening or a bypass) to admit fuel and oxidizer at low initial flows to the high pressure piping, cooling jacket, injector manifold, and combustion chamber(s). Valve opening rate and sequencing may be critical to achieve proper propellant lead. Propellants start to burn and turbine shaft begins to rotate.

Using an automated engine control, make checks (e.g., shaft speed, igniter function, feed pressures) to assure proper operation before initiating next step.

In systems with gearboxes the gear lubricant and coolant fluid start to flow.

For safety reasons, one of the propellants must reach the chamber first.

3. *Start: Transition to Full Flow/Full Thrust*

Turbopump power and shaft speed increase.

Propellant flows and thrust levels increase until they reach full-rated values. May use controls to prevent exceeding limits of mixture ratio or rates of increase during transient.

Principal valves are fully opened. Attain full chamber pressure and thrust.

In systems where vaporized propellant is fed into the propellant tanks for tank pressurization, the flow of this heated propellant is initiated.

Systems for controlling thrust or mixture ratio or other parameter are activated.

4. *Stop*

Signal to stop deactivates the critlcal valve(s).

Key valves close in a predetermined sequence. For example, the valve controlling the gas generator or preburner will be closed first. Pressurization of propellant tanks is stopped.

As soon as turbine drive gas supply diminishes the pumps will slow down. Pressure and flow of each propellant will diminish quickly until it stops. The main valves are closed, often by spring forces, as the fluid pressures diminish. Tank pressurization may also be stopped. In some engines the propellant trapped in the lines or cooling jacket may be blown out by vaporization or gas purge.

SSME Start and Stop Sequences. This is an example of the transient start and stop behavior of a complex staged combustion cycle engine with a tank head start. It illustrates the rapid functions of an electronic controller. The SSME flow sheet in Fig. 6–12 identifies the location of the key components mentioned below and Fig. 10–12 shows the sequence and events of these transients. The remainder of this subsection is based on information provided by The Boeing Company, Rockerdyne Propulsion and Power.

For a tank head start, initial energy to start the turbines spinning is all derived from initial propellant tank pressures (fuel and oxidizer) and gravity (head of liquid column). Combining the tank head start with a staged combustion cycle consisting of five pumps, two preburners, and a main combustion chamber (MCC) results in a complicated and sophisticated start sequence, which is very robust and reliable. Prior to test, the SSME turbopumps and ducting (down to the main propellant valves) are chilled with liquid hydrogen and liquid oxygen (LOX) to cryogenic temperature to ensure liquid propellants for proper pump operation. At engine start command, the main fuel valve (MFV) is opened first to provide chilling below the MFV and a fuel lead to the engine. The three oxidizer valves sequence the main events during the crucial first two seconds of start. The fuel preburner oxidizer valve (FPOV) is ramped to 56% to provide LOX for ignition in the fuel preburner (FPB) in order to provide initial turbine torque of the high-pressure fuel turbopump (HPFTP). Fuel system oscillations (FSO), which occur due to heat transfer downstream of the initially chilled system can result in flowrate dips. These fuel flow dips can lead to damaging temperature spikes in the FPB as well as the oxidizer preburner (OFB) at ignition and 2 Hz cycles thereafter until the hydrogen is above critical pressure. The oxidizer preburner oxidizer valve (OPOV) and main oxidizer valve (MOV) are ramped open next to provide LOX for OPB and MCC ignition.

The next key event is FPB prime. Priming is filling of the OX system upstream of the injectors with liquid propellant. This results in increased combustion and higher power. This event occurs around 1.4 sec into start. The high-pressure fuel turbopump (HPFTP) speed is automatically checked at 1.24 sec into start to ensure it will be at a high enough level before the next key event, MCC prime, which is controlled by the MOV. Priming and valve timing are critical. We explain some of the events that could go wrong. At MCC prime, an abrupt rise in backpressure on the fuel pump/turbine occurs. If flowrate through the fuel pump at this time is not high enough (high speed), then the heat imparted to the fluid as it is being pumped can vaporize it, leading to unsatisfactory flow in the engine, and subsequent high mixture ratio with high gas temperature and possible burnout in the hot gas system. This occurs if the MCC primes too early or HPFTP speed is abnormally low. If the MCC primes too late, the HPFTP may accelerate too fast due to low backpressure after FPB prime and exceed its safe speed. The MCC prime normally occurs at 1.5 sec. The OPB is primed last since it controls LOX flow and a strong fuel lead and healthy fuel pump flow are desirable to prevent engine burnout due to

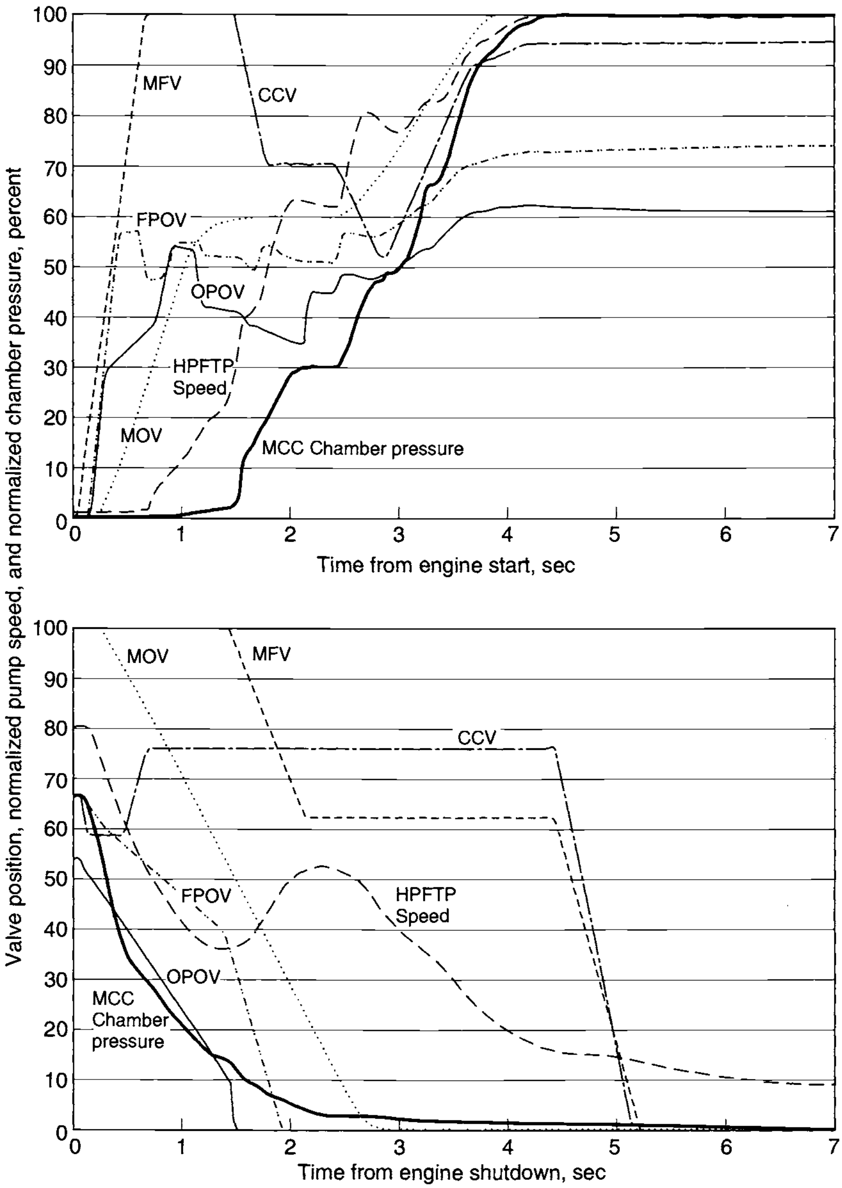


FIGURE 10–12. The sequence and events for starting and shutdown of the SSME (Space Shuttle main engine). This particular start sequence leads to a chamber pressure of 2760 psia (normalized here to 100%), a high-pressure fuel turbopump speed of 33,160 rpm (100%) , at a sea-level thrust of 380,000 lbf (shown as 100%). This shutdown occurs at altitude when the engine has been throttled to 67% of its power level or a vacuum thrust of 312,559 lbf, which is shown as 67% of the MCC chamber pressure. (Courtesy of The Boeing Company, Rocketdyne Propulsion and Power.)

a high mixture ratio. The OPOV provides minimal flowrate during the early part of the start to force the oxidizer to prime last at 1.6 sec into start. Again, the FSO influences temperature spikes in the OPB and must be sequenced around, prior to the MCC prime which raises the fuel pressure above critical in the fuel system. At two seconds into start, the propellant valves are sequenced to provide 25% of rated power level (RPL). During the first 2.4 sec of start, the engine is in an open-loop mode, but proportional control of the OPOV is used, based on MCC pressure. At this point, additional checks are carried out to ensure engine health, and a subsequent ramp to mainstage at 2.4 sec is done using closed-loop MCC–chamber-pressure/OPOV control. At 3.6 sec, closed-loop mixture ratio/FPOV control is activated.

The chamber cooling valve (CCV) is open at engine start and sequenced to provide optimum coolant fuel flow to the nozzle cooling jacket and the chamber and preburners during the ignition and main stage operation. It diverts flow to the cooling passages in the nozzle after MCC prime causes the heat load to increase. The description above is simplified and does not mention several other automatic checks, such as verifying ignition in the MCC or FPB or the fuel or chamber pressure buildup, which are sensed and acted upon at various times during the start sequence. The spark-activated igniters are built into the three injectors (MCC, FPB, OPB) using the same propellants. They are not mentioned above or shown in the flow sheet, but one of them can be seen in Fig. 9–6.

The shutdown sequence is initiated by closing the OPOV, which powers down the engine (reduces oxygen flow, chamber pressure, and thrust); this is followed quickly by closing the FPOV, so the burning will shut down fuel rich. Shortly thereafter the MOV is closed. The MFV stays open for a brief time and then is moved into an intermediate level to balance with the oxygen flow (from trapped oxygen downstream of the valves). The MPV and the CCV are closed after the main oxygen mass has been evaporated or expelled.

Automatic Controls

Automatically monitored controls are frequently used in liquid propellant rockets to accomplish thrust control or mixture ratio control. The automatic control of the thrust vector is discussed in Chapter 16.

Before electronic controls became common for large engines, pneumatic controls were used with helium gas. We still use helium to actuate large valves, but no longer for logic control. A pressure ladder sequence control was used, where pressures (and a few other quantities) were sensed and, if satisfactory, the next step of the start sequence was pneumatically initiated. This was used on the H-1 engine and the Russian RD-170 engine, whose flow sheet is shown in Figure 10–11.

Most automatic controls use a servomechanism. They generally consist of three basic elements: a *sensing mechanism*, which measures or senses the variable quantity to be controlled; a *computing or controlling mechanism*, which compares the output of the sensing mechanism with a reference value and gives a control signal to the third component, the *actuating device*, which manipulates the variable to be controlled. Additional discussion of computer control with automatic data recording and analysis is given in Chapter 20.

Figure 10–13 shows a typical simple thrust control system for a gas generator cycle aimed at regulating the chamber pressure (and therefore also the thrust) during the flight to a predetermined value. A pressure-measuring device with an electric output is used for the sensing element, and an automatic control device compares this gauge output signal with a signal from the reference gauge or a computer voltage and thus computes an error signal. This error signal is amplified, modulated, and fed to the actuator of the throttle valve. By controlling the propellant flow to the gas generator, the generator pressure is regulated and, therefore, also the pump speed and the main propellant flow; indirectly, the chamber pressure in the thrust chamber is regulated and, therefore, also the thrust. These quantities are varied until such time as the error signal approaches zero. This system is vastly simplified here, for the sake of

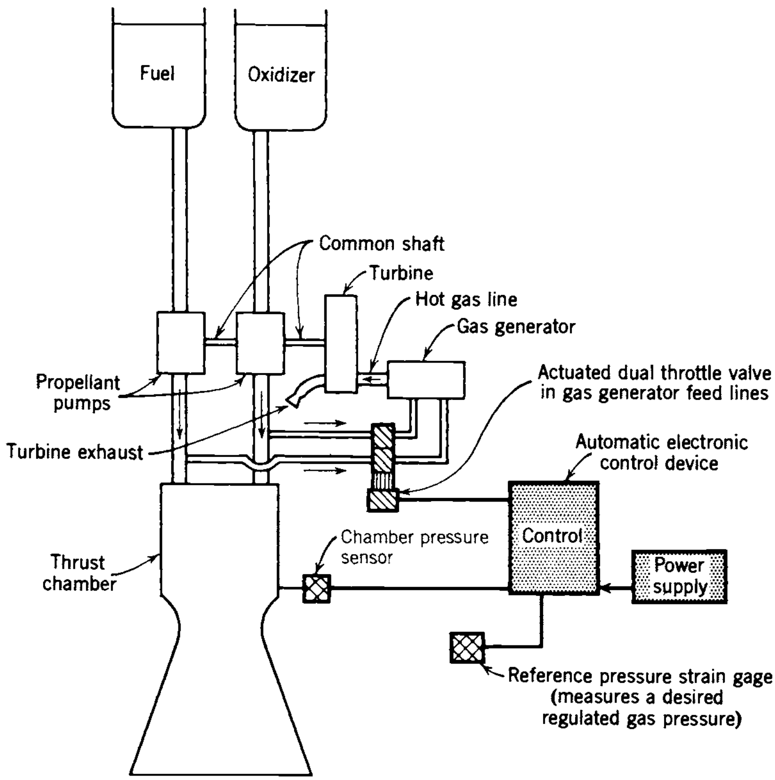


FIGURE 10–13. Simplified schematic diagram of an automatic servomechanism-type chamber pressure control of a liquid propellant rocket engine with a turbopump feed system, a gas generator, and a tank head, boot strap (self-pumping) starting system.

illustration; in actual practice the system may have to be integrated with other automatic controls. In this diagram the mixture of the gas generator is controlled by the pintle shapes of the fuel and oxidizer valves of the gas generator and by yoking these two valves together and having them moved in unison by a single actuator.

In the expander cycle shown schematically in Fig. 6–11, the thrust is regulated by maintaining a desired chamber pressure and controlling the amount of hydrogen gas flowing to the turbine by means of a variable bypass. The flow through this bypass is small (typically 5% of gas flow) and is controlled by the movement of a control valve.

In a *propellant utilization* system the mixture ratio is varied to insure that the fuel and oxidizer propellant tanks are both simultaneously and completely emptied; no undue propellant residue should remain to increase the empty mass of the vehicle, which in turn would detrimentally decrease the vehicle mass ratio and the vehicle's flight performance (see Chapter 4). For example, the oxidizer flow rate may be somewhat larger than normal due to its being slightly denser than normal or due to a lower than normal injector pressure drop; if uncontrolled, a fuel residue would remain at the time of oxidizer exhaustion; however, the control system would cause the engine to operate for a period at a propellant mixture ratio slightly more fuel-rich than normal, to compensate and assure almost simultaneous emptying of both propellant tanks. Such a control system requires accurate measurement of the amount of propellant remaining in the two propellant tanks during the flight.

Any of the three principal components of an automatic control system can have many differerent forms. Typical sensing devices include those that measure chamber pressure, propellant pressures, pump rotational speeds, tank level, or propellant flow. The actuating device can throttle propellant flow or control a bypass device or the gas generator discharge. There are many operating mechanisms for the controller, such as direct electrical devices, electronic analog or digital computers, hydraulic or pneumatic devices, and mechanical devices. The actuators can be driven by electrical motors, hydraulic, pneumatic, or mechanical power. The hydraulic actuators can provide very high forces and quick response. The exact type of component, the nature of the power supply, the control logic, the system type, and the operating mechanisms for the specific control depend on the details of the application and the requirements. Controls are discussed further in Refs. 6–1 and 10–18.

In applications where the final vehicle velocity must be accurately determined, the amount of impulse that is imparted to the vehicle during the cutoff transient may be sufficiently variable to exceed the desired velocity tolerance. Therefore, in these applications close control over the thrust decay curve is necessary and this can be accomplished by automatic control over the sequencing and closing rates of the main propellant valves and the location of the valves in relation to the injector.

Control by Computer

Early rocket engines used simple timers and, later, a pressure ladder sequence to send commands to the engine for actuating valves and other steps in the operation. Pneumatic controllers were also used in some engines for starting and stopping. For the last 20 years we have used *digital computers* in large liquid propellant rocket engines for controlling their operation (see Ref. 10–15). In addition to controlling the start and stop of engines, they can do a lot more and can contribute to making the engine more reliable. Table 10–7 gives a list of typical functions that a modern engine control computer has undertaken in one or more engines. This list covers primarily large turbopump-fed engines and does not include consideration of multiple small thruster attitude control rocket engines.

The design of control computers is beyond this text. In general it has to consider carefully all the possible engine requirements, all the functions that have to be monitored, all the likely potential failure modes and their compensating or ameliorating steps, all the sensed parameters and their scales, the method of control, such as open, closed, or multiple loops, adaptive or self-learning (expert system), the system architecture, the software approach, the interrelation and division of tasks with other computers on board the vehicle or on the ground, and the method of validating the events and operations. It is also convenient to have software that will allow some changes (which become necessary because of engine developments or failures) and allow the control of several parameters simultaneously. While the number of functions performed by the control computer seems to have increased in the last 20 years, the size and mass of the control computer has actually decreased substantially.

The control computer is usually packaged in a waterproof, shockproof black box, which is mounted on the engine. Fire-resistant and waterproof cable harnesses lead from this box to all the instrument sensors, valve position indicators, tachometers, accelerometers, actuators, and other engine components, to the power supply, the vehicle's controller, and an umbilical, severable multi-wire harness leads to the ground support equipment.

10.6. ENGINE SYSTEM CALIBRATION

Although an engine has been designed to deliver a specific performance ($F, I_s, \dot{m}, r$), a newly manufactured engine will not usually perform precisely at these nominal parameters. If the deviation from the nominal performance values is more than a few percent, the vehicle will probably not complete its intended flight course. There are several reasons for these deviations. Because of unavoidable dimensional tolerances on the hardware, the flow–pressure profile or the injector impingement (combustion efficiency) will deviate slightly from the nominal design value. Even a small change in mixture

TABLE 10–7. Typical Functions to Be Performed by Digital Computers in Monitoring and Controlling the Operation of a Liquid Propellant Rocket Engine

1. *Sample the signals from significant sensors* (e.g., chamber pressure, gas and hardware temperatures, tank pressure, valve position, etc.) at frequent intervals, say once, 10, 100, or 1000 times per second. For parameters that change slowly, e.g., the temperature of the control box, sampling every second or every five seconds may be adequate, but chamber pressure would be sampled at a high frequency.
2. *Keep a record of all the significant signals* received and all the signals generated by the computer and sent out as commands or information.
3. *Control the steps and sequence of the engine start.* Figure 10–12 and Table 10–6 list typical steps that have to be taken, but do not list the measured parameters that will confirm that the commanded step was implemented. For example, if the igniter is activated, a signal change from a properly located temperature sensor or a radiation sensor could verify that the ignition had indeed happened.
4. *Control the shutdown of the engine.* For each of the steps listed at the bottom of Table 10–6 or in Fig. 10–12 there often has to be a sensing of a pressure change or other parameter change to verify that the commanded shutdown step was taken. An *emergency shutdown* may be commanded by the controller, when it senses certain kinds of malfunctions, that allow the engine to be shut down safely before a dramatic failure occurs. This emergency shutdown procedure must be done quickly and safely and may be different from a normal shutdown, and must avoid creating a new hazardous condition.
5. *Limit the duration of full thrust operation.* For example, cutoff is to be initiated just before the vehicle attains the desired mission flight velocity.
6. *Safety monitoring and control.* Detect combustion instability, over-temperatures in precombustors, gas generators, or turbopump bearings, violent turbopump vibration, turbopump overspeed or other parameter known to cause rapid and drastic component malfunction, that can quickly lead to engine failure. Usually, more than one sensor signal will show such a malfunction. If detected by several sensors, the computer may identify it as a possible failure whose in-flight remedy is well known (and preprogrammed into the computer); then a corrective action or a safe shutdown may be automatically commanded by the control computer.
7. *Control propellant tank pressurization.* The tank pressure value has to be within an allowable range during engine operation and also during a coasting flight period prior to a restart. Sensing the activation of relief valves on the tank confirms overpressure. Automatically, the computer can then command stopping or reducing the flow of pressurant.
8. *Perform automatic closed-loop control of thrust and propellant utilization* (described before).
9. *Transmit signals to a flying vehicle's telemetering system,* which in turn can send them to a ground station, thus providing information on the engine status, particularly during experimental or initial flights.
10. *Self-test the computer and software.*
11. *Analyze key sensor signals for deviation from nominal performance* before, during, and after engine operation. Determine whether sensed quantities are outside of predicted limits. If appropriate and feasible, if more than one sensor indicates a possible out-of-limit value, and if the cause and remedy can be predicted (preprogrammed), then the computer can automatically initiate a compensating action.

ratio will cause a significant increase of residual, unused propellant. Also, minor changes in propellant composition or storage temperature (which affects density and viscosity) can cause deviations. Regulator setting tolerances or changes in flight acceleration (which affects static head) are other factors. An engine calibration is the process of adjusting some of its internal parameters so that it will deliver the intended performance within the allowed tolerance bands.

Hydraulic and pneumatic components (valves, pipes, expansion joints) can readily be water flow tested on flow benches and corrected for pressure drops and density (and sometimes also viscosity) to determine their pressure drop at rated flow. Components that operate at elevated temperatures (thrust chambers, turbines, preburners, etc.) have to be hot fired and cryogenic components (pumps, some valves) often have to be tested at the cryogenic propellant temperature. The engine characteristics can be estimated by adding together the corrected values of pressure drops at the desired mass flow. Furthermore, the ratio of the rated flows $\dot{m}_o/\dot{m}_f$ has to equal the desired mixture ratio r. This is shown in the example below. The adjustments include adding pressure drops with judiciously placed orifices, or changing valve positions or regulator setting.

In most pressurized feed systems the pressurizing gas is supplied from its high pressure tank through a regulator to pressurize both the fuel and the oxidizer in their respective tanks. The pressure drop equations for the oxidizer and the fuel (subscripts o and f) are given below for a pressurized feed system at nominal flows.

$$p_{\text{gas}} - (\Delta p_{\text{gas}})_f = p_1 + \Delta p_f + (\Delta p_{\text{inj}})_f + (\Delta p_j)_f + \tfrac{1}{2}\rho_f v_f^2 + La\rho_f \tag{10–17}$$

$$p_{\text{gas}} - (\Delta p_{\text{gas}})_o = p_1 + \Delta p_o + (\Delta p_{\text{inj}})_o + \tfrac{1}{2}\rho_o v_o^2 + La\rho_o \tag{10–18}$$

The gas pressure in the propellant tank is the regulated pressure p_{gas}, diminished by the pressure losses in the gasline Δp_{gas}. The static head of the liquid $La\rho$ (L is the distance of the liquid level above the thrust chamber, a is the flight acceleration, and ρ is the propellant density) augments the gas pressure. It has to equal the chamber pressure p_1 plus all the pressure drops in the liquid piping or valves Δp, the injector Δp_{inj}, the cooling jacket Δp_j, and the dynamic flow head $\frac{1}{2}\rho v^2$. If the required liquid pressures do not equal the gas pressure in the propellant tank at the nominal propellant flow, then an additional pressure drop (calibration orifice) has to be inserted. A good design provides an extra pressure drop margin for this purpose.

Two methods are available for precise control of the engine performance parameters. One uses an automatic system with feedback and a digital computer to control the deviations in real time, while the other relies on an initial static calibration of the engine system. The latter appoach is simpler and is sometimes preferred, and is still quite accurate.

The *pressure balance* is the process of balancing the available pressure supplied to the engine (by pumps and/or pressurized tanks) against the pressure drops plus the chamber pressure. It is necessary to do this balancing in order to calibrate the engine, so it will operate at the desired flows and mixture ratio. Figure 10–14 shows the pressure balance for one of the two branches of propellant systems in a bipropellant engine with a pressurized feed system. It plots the pressure drops (for injector, cooling passages, pressurizing gas passages, valves, propellant feed lines, etc.) and the chamber pressure against the propellant flow, using actual component pressure drop measurements (or estimated data) and correcting them for different flows. The curves are generally plotted in terms of head loss and volumetric flow to eliminate the fluid density as an explicit variable for a particular regulated pressure. The regulated pressure is the same for the fuel and oxidizer pressure balance and it also can be adjusted. This balance of head and flow must be made for both the fuel and oxidizer systems, because the ratio of their flows establishes the actual mixture ratio and the sum of their flows establishes the thrust. The pressure balance between available and required tank pressure, both at the desired flow, is achieved by adding a calibration orifice into one of the lines, as can be seen in Fig. 10–14. Not shown in the figure is the static head provided by the elevation of the liquid level, since it is small for many space launch systems. However, with high acceleration and dense propellants, it can be a significant addition to the available head.

For a pumped feed system of a bipropellant engine, Fig. 10–15 shows a balance diagram for one branch of the two propellants systems. The pump speed is an additional variable. The calibration procedure is usually more complex for a turbopump system, because the pump calibration curves (flow–head–power relation) can not readily be estimated without good test data and cannot easily be approximated by simple analytical relations. The flow of the propellants to a gas generator or preburner also needs to be calibrated. In this case the turbine shaft torque has to equal the torque required by the pumps and the energy losses in bearings, seals or windage. Thus a power balance must be achieved in addition to the matching of pressures and the individual propellant flows. Since these parameters are interdependent, the determination of the calibration adjustments may not always be simple. Many rocket organizations have developed computer programs to carry out this balancing.

Example 10–3. The following component data and design requirements are given for a pressurized liquid propellant rocket system similar to that in Figs. 1–3 and 10–14: fuel, 75% ethyl alcohol; oxidizer, liquid oxygen; desired mixture ratio, 1.30; desired thrust, 5000 lbf at sea level. For this propellant combustion gas $k = 1.22$.

Component test data: Pressure losses in gas systems were found to be negligible. Fuel valve and line losses were 9.15 psi at a flow of 9.63 lbm/sec of water. Oxidizer valve and line losses were 14.2 psi at a flow of 12.8 lbm/sec of liquid oxygen. Fuel cooling jacket prssure loss was 52 psi at a flow of 9.61 lbm/sec of water. Oxidizer side injector pressure

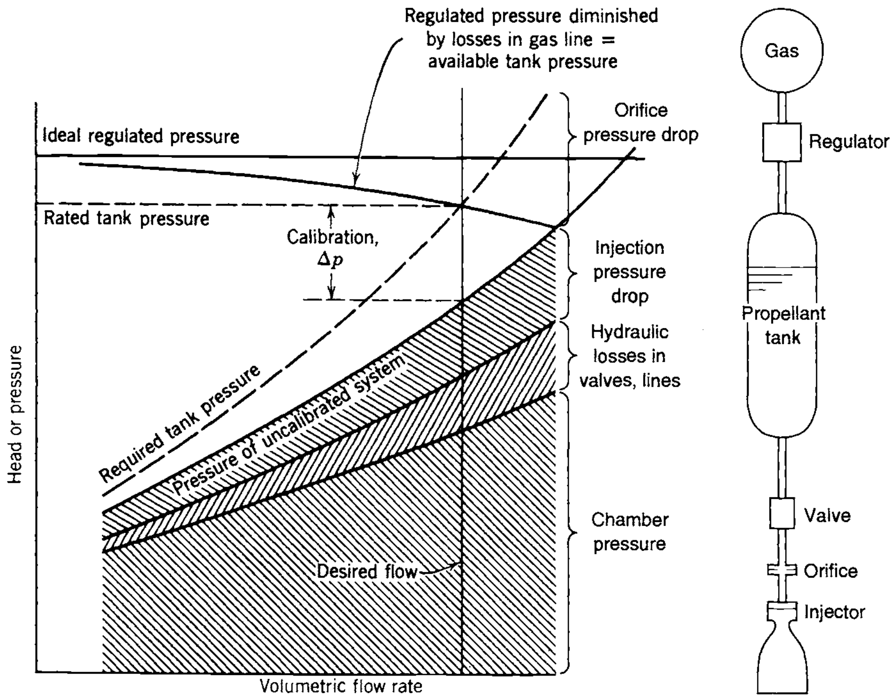


FIGURE 10–14. Simplified flow diagram and balance curves for the fuel or the oxidizer of a typical gas-pressurized bipropellant feed system. This diagram is also the same for a monopropellant feed system, except that it has no calibration orifice; it is calibrated by setting the proper regulated pressure.

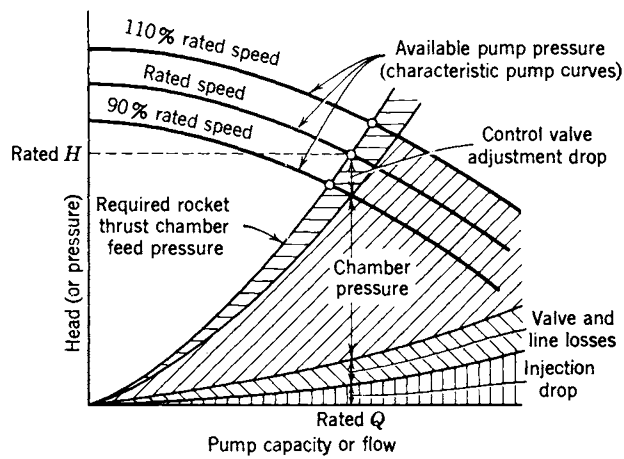


FIGURE 10–15. Simplified diagram of the balance of available and required feed pressures versus flow for one of the propellants in a rocket engine with a turbopump feed system. Chamber pressure is increased by liquid column.

drop was 90.0 psi at 10.2 lb/sec of oxygen flow under thrust chamber operating conditions. Fuel side injector pressure drop was 48.3 psi at 10.2 lb/sec of fuel flow under thrust chamber operating conditions. Average results of several sea-level thrust chamber tests were: thrust = 5410 lbf; mixture ratio = 1.29; specific impulse = 222 sec; chamber pressure = 328 psia; nozzle area ratio = 4.0. Determine regulator setting and size and location of calibration orifices.

SOLUTION. First, the corrections necessary to obtain the desired thrust chamber conditions have to be determined. The experimental thrust chamber data must be adjusted for deviations in mixture ratio, thrust, and specific impulse. The variation of specific impulse with *mixture ratio* is determined from experimental data or (on a relative basis) from theoretical calculations similar to those that are the basis of Fig. 5–1. Because the value of I_s at the desired mixture ratio of 1.30 is within 0.08% of the value of I_s under the actual test conditions ($r = 1.29$), any mixture ratio correction of I_s is neglected here.

The correction of the *specific impulse* for chamber pressure is made next. The specific impulse is essentially proportional to the thrust coefficients as determined from Eq. 3–30. For $k = 1.22$, and the pressure ratios $p_1/p_3 = 328/14.7 = 22.2$ and $300/14.7 = 20.4$, the values of C_F can be calculated as 1.420 and 1.405, respectively. In this calculation p_2 has to be determined for isentropic conditions, such as those in Figs. 3–7 or 3–8 for the given nozzle area ratio. The sea-level specific impulse is therefore corrected to $I_s = 222\ (1.405/1.420) = 220$ sec. The *chamber pressure* has to be reduced from 328 psi to a lower value in order to bring the thrust from its test value of 5410 lbf to the design value of 5000 lbf. In accordance with Eq. 3–31, $F = C_F A_t p_1$. The chamber pressure is inversely proportional to the thrust coefficient C_F and proportional to the thrust, and therefore

$$p_1/p_1' = (F_1/F_1')(C_F'/C_F)$$

The primes refer to the component test condition.

$$p_1 = 328(5000/5410)(1.420/1.405) = 306 \text{ psi}$$

The desired total *propellant flow* is, from Eq. 2–5,

$$\dot{w} = F/I_s = 5000/220 = 22.7 \text{ lbf/sec}$$

For a mixture ratio of 1.3, the desired *fuel and oxidizer flows* are obtained from Eqs. 6–3 and 6–4 as $\dot{w}_f = 9.9$ lbf/sec and $\dot{w}_o = 12.8$ lbf/sec. Next, the various component pressure drops are corrected to the desired flow values and to the corrected propellant densities in accordance with Eq. 8–2, which applies to all hydraulic devices. By neglecting variations in discharge coefficients, this equation can be rewritten into a convenient form:

$$\dot{w}/\dot{w}' = \sqrt{\rho/\rho'}\sqrt{\Delta p/\Delta p'}$$

With this equation and the specific gravity values (from Fig. 7–1) of 1.14 for oxygen, 0.85 for diluted ethyl alcohol, and 1.0 for water, the new pressure drops for the corrected flow conditions can be found, and these are tabulated below with flow values given in pounds per second and pressure values in pounds per square inch.

Component	Component Test Data			Design Conditions		
	Fluid	$\dot{w}$	Δp	Fluid	$\dot{w}$	Δp
Fuel injector	Fuel	10.2	48.3	Fuel	9.9	45.3
Oxidizer injector	Oxygen	14.0	90.0	Oxygen	12.8	75.0
Fuel cooling jacket	Water	9.61	52.0	Fuel	9.9	64.9
Fuel valve and line	Water	9.63	9.15	Fuel	9.9	11.4
Oxidizer valve and line	Oxygen	12.8	14.2	Oxygen	12.8	14.2

The total pressure drop in the fuel system is 45.3 + 64.9 + 11.4 = 121.6 psi, and in the oxidizer system it is 75.0 + 14.2 = 89.2 psi.

The tank pressures required to obtain the desired flows are calculated by adding the chamber pressure to these pressure drops; that is, $(p)_o = 306 + 89.2 = 395.2$ psi and $(p)_f = 306 + 121.6 = 427.6$ psi. To equalize the tank pressures so that a single gas pressure regulator can be used, an *additional pressure loss must be introduced into the oxygen system.* The correction to this simple pressurized liquid propellant system is accomplished by means of an orifice, which must be placed in the propellant piping between the oxidizer tank and the thrust chamber. Allowing 10 psi for regulator functioning, the pressure drop in a calibration orifice will be $\Delta p = 427.6 - 395.2 + 10 = 42.4$ psi. The *regulator setting* should be adjusted to give a regulated downstream pressure of 427.6 psi under flow conditions. The orifice area (assume $C_d = 0.60$ for a sharp-edged orifice) can be obtained from Eq. 8–2, but corrected with a g_o for English units.

$$A = \frac{\dot{m}}{C_d\sqrt{2g\rho\,\Delta p}} = \frac{12.8 \times 144}{0.60\sqrt{2 \times 32.2 \times 1.14 \times 62.4 \times 42.4 \times 144}}$$
$$= 0.581 \text{ in.}^2 \text{ (or 0.738 in. diameter)}$$

A set of balancing equations can be assembled into a computer program to assist in the calibration of engines. It can also include some of the system's dynamic analogies that enable proper calibration and adjustment of transient performance of the engine as during start. There is a trend to require tighter tolerances on rocket engine parameters (such as thrust, mixture ratio, or specific impulse), and therefore the measurements, calibrations, and adjustments are also being performed to much tighter tolerances than was customary 25 years ago.

10.7. SYSTEM INTEGRATION AND ENGINE OPTIMIZATION

Rocket engines are part of a vehicle and must interact and be integrated with other vehicle subsystems. There are *interfaces* (connections, wires, or pipelines) between the engine and the vehicle's structure, electric power system, flight control system (commands for start or thrust vector control), and ground support system (check-out or propellant supply). The engine also imposes limitations on vehicle components by its heat emissions, noise, and vibrations.

Integration means that the engine and the vehicle are compatible with each other, interfaces are properly designed, and there is no interference or unnecessary duplication of functions with other subsystems. The engine works with other subsystems to enhance the vehicle's performance and reliability, and reduce the cost. In Chapter 17 we describe the process of selecting rocket propulsion systems and it includes a discussion of interfaces and vehicle integration. This discussion in Chapter 17 is supplementary and applies to several different rocket propulsion systems. This section concerns liquid propellant rocket engines.

Since the propulsion system is usually the major mass of the vehicle, its structure (which usually includes the tanks) often becomes a key structural element of the vehicle and has to withstand not only the thrust force but also various vehicle loads, such as aerodynamic forces or vibrations. Several alternate tank geometries and locations (fuel, oxidizer, and pressurizing gas tanks), different tank pressures, and different structural connections have to be evaluated to determine the best arrangement.

The thermal behavior of the vehicle is strongly affected by the heat generation (hot plume, hot engine components, or aerodynamic heating) and the heat absorption (the liquid propellants are usually heat sinks) and by heat rejection to its surroundings. Many vehicle components must operate within narrow temperature limits, and their thermal designs can be critical when evaluated in terms of the heat balance during, after, and before the rocket engine operation.

Optimization studies are conducted to select the best values or to optimize various engine parameters such as chamber pressure (or thrust), mixture ratio (which affects average propellant density and specific impulse), number of thrust chambers, nozzle area ratio, or engine volume. By changing one or more of these parameters, it is usually possible to make some improvement to the vehicle performance (0.1 to 5.0%), its reliability, or to reduce costs. Depending on the mission or application, the studies are aimed at maximizing one or more vehicle parameter such as range, vehicle velocity increment, payload, circular orbit altitude, propellant mass fraction, or minimizing costs. For example, the mixture ratio of hydrogen–oxygen engines for maximum specific impulse is about 3.6, but most engines operate at mixture ratios between 5 and 6 because the total propellant volume is less, and this allows a reduced mass for the propellant tanks and the turbopump (resulting in a higher vehicle velocity increment) and a reduced vehicle drag (more net thrust). The selection of the best nozzle area ratio was mentioned in Chapter 3; it depends on the flight path's altitude–time history; the increase in specific impulse is offset by the extra nozzle weight and length. The best thrust–time profile can also usually be optimized, for a given application, by using trajectory analyses.

PROBLEMS

1. Estimate the mass and volume of nitrogen required to pressurize an N_2O_4–MMH feed system for a 4500 N thrust chamber of 25 sec duration ($\zeta_v = 0.92$, the ideal, $I_s =$ 285 sec at 1000 psi or 6894 N/M^2 and expansion to 1 atm). The chamber pressure is 20 atm (abs.) and the mixture ratio is 1.65. The propellant tank pressure is 30 atm, and the initial gas tank pressure is 150 atm. Allow for 3% excess propellant and 50% excess gas to allow some nitrogen to dissolve in the propellant. The nitrogen regulator requires that the gas tank pressure does not fall below 29 atm.

2. What are the specific speeds of the four SSME pumps? (See the data given in Table 10–1.)

3. Compute the turbine power output for a gas consisting of 64% by weight of H_2O and 36% by weight of O_2, if the turbine inlet is at 30 atm and 658 K with the outlet at 1.4 atm and with 1.23 kg flowing each second. The turbine efficiency is 37%.

4. Compare the pump discharge gage pressures and the required pump powers for five different pumps using water, gasoline, alcohol, liquid oxygen, and diluted nitric acid. The respective specific gravities are 1.00, 0.720, 0.810, 1.14, and 1.37. Each pump delivers 100 gal/min, a head of 1000 ft, and arbitrarily has a pump efficiency of 84%. *Answers*: 433, 312, 350, 494, and 594 psi; 30.0, 21.6, 24.3, 34.2, and 41.1 hp.

5. The following data are given on a liquid propellant rocket engine:

Thrust	40,200 lbf
Thrust chamber specific impulse	210.2 sec
Fuel	Gasoline (sp. gr. 0.74)
Oxidizer	Red fuming nitric acid (sp. gr. 1.57)
Thrust chamber mixture ratio	3.25
Turbine efficiency	58%
Required pump power	580 hp
Power to auxiliaries mounted on turbopump gear case	50 hp
Gas generator mixture ratio	0.39
Turbine exhaust pressure	37 psia
Turbine exhaust nozzle area ratio	1.4
Enthalpy available for conversion in turbine per unit of gas	180 Btu/lb
Specific heat ratio of turbine exhaust gas	1.3

Determine the engine system mixture ratio and the system specific impulse. *Answers*: 3.07 and 208.

SYMBOLS

a	acceleration, m/sec^2 (ft/sec^2)
A	area, m^2 (ft^2)
c_p	specific heat at constant pressure, J/kg-K (Btu/lbm-R)

C_F	thrust coefficient (see Eq. 3–30)
D	diameter, m (ft)
F	thrust, N (lbf)
g_0	sea-level acceleration of gravity, 9.806 m/sec^2 (32.17 ft/sec^2)
Δh	enthalpy change, J/kg (Btu/lb)
H	head, m (ft)
$(H_s)_A$	available pump suction head above vapor pressure, often called net positive suction head, m (ft)
$(H_s)_R$	required pump suction head above vapor pressure, m (ft)
I_s	specific impulse, sec (lbf-sec/lbf)
k	specific heat ratio
L	length, m (ft)
$\dot{m}$	mass flow rate, kg/sec
N	shaft speed, rpm (rad/sec)
N_s	specific speed of pump
p	pressure, N/m^2 (lbf/in.^{2})
P	power, W (hp)
Q	volume flow rate, m^3/sec (ft^3/sec)
r	flow mixture ratio (oxidizer to fuel flow)
S	suction specific speed of pump
t	time, sec
T	absolute temperature, K (R)
u	tip speed or mean blade speed, m/sec (ft/sec)
u	velocity, m/sec (ft/sec)

Greek Letters

Δ	finite differential
ζ_d	discharge correction factor
ζ_F	thrust correction factor
η	efficiency
λ	coefficient of thermal expansion, m/m-K (in./in.-R)
ρ	density, kg/m^3 (lb/ft^3)
ψ	constant

Subscripts

c	chamber
e	maximum efficiency
f	fuel
gg	gas generator
o	oxidizer
oa	overall engine system
P	pump
T	turbine

0 initial condition
1 inlet
2 outlet

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