CHAPTER 1

CLASSIFICATION

Propulsion in a broad sense is the act of changing the motion of a body. Propulsion mechanisms provide a force that moves bodies that are initially at rest, changes a velocity, or overcomes retarding forces when a body is propelled through a medium. *Jet propulsion* is a means of locomotion whereby a reaction force is imparted to a device by the momentum of ejected matter.

Rocket propulsion is a class of jet propulsion that produces thrust by ejecting stored matter, called the *propellant*. *Duct propulsion* is a class of jet propulsion and includes turbojets and ramjets; these engines are also commonly called air-breathing engines. Duct propulsion devices utilize mostly the surrounding medium as the "working fluid", together with some stored fuel. Combinations of rockets and duct propulsion devices are attractive for some applications and are described in this chapter.

The *energy source* most useful to rocket propulsion is *chemical combustion*. Energy can also be supplied by *solar radiation* and, in the past, also by *nuclear reaction*. Accordingly, the various propulsion devices can be divided into *chemical propulsion, nuclear propulsion*, and *solar propulsion*. Table 1–1 lists many of the important propulsion concepts according to their energy source and type of propellant or working fluid. Radiation energy can originate from sources other than the sun, and theoretically can cover the transmission of energy by microwave and laser beams, electromagnetic waves, and electrons, protons, and other particle beams from a transmitter to a flying receiver. Nuclear energy is associated with the transformations of atomic particles within the nucleus of atoms and can be of several types, namely, fission, fusion, and decay of radioactive species. Other energy sources, both internal (in the vehicle) and external, can be considered. The energy form

TABLE 1–1. Energy Sources and Propellants for Various Propulsion Concepts

Propulsion Device	Energy Source[a] Chemical	Nuclear	Solar	Propellant or Working Fluid
Turbojet	D/P	TFD		Fuel + air
Turbo–ramjet	TFD			Fuel + air
Ramjet (hydrocarbon fuel)	D/P	TFD		Fuel + air
Ramjet (H_2 cooled)	TFD			Hydrogen + air
Rocket (chemical)	D/P	TFD		Stored propellant
Ducted rocket	TFD			Stored solid fuel + surrounding air
Electric rocket	D/P	TFD	D/P	Stored propellant
Nuclear fission rocket		TFD		Stored H_2
Nuclear fusion rocket		TFND		Stored H_2
Solar heated rocket			TFD	Stored H_2
Photon rocket (big light bulb)		TFND		Photon ejection (no stored propellant)
Solar sail			TFD	Photon reflection (no stored propellant)

[a]D/P, developed and/or considered practical; TFD, technical feasibility has been demonstrated, but development is incomplete; TFND, technical feasibility has not yet been demonstrated.

found in the output of a rocket is largely the kinetic energy of the ejected matter; thus the rocket converts the input from the energy source into this form. The ejected mass can be in a solid, liquid, or gaseous state. Often a combination of two or more of these is ejected. At very high temperatures it can also be a plasma, which is an electrically activated gas.

1.1. DUCT JET PROPULSION

This class, also called air-breathing engines, comprises devices which have a duct to confine the flow of air. They use oxygen from the air to burn fuel stored in the flight vehicle. The class includes turbojets, turbofans, ramjets, and pulse-jets. This class of propulsion is mentioned primarily to provide a comparison with rocket propulsion and a background for combination rocket–duct engines, which are mentioned later. Several textbooks, such as Refs. 1–1 and 1–2, contain a discussion of duct jet propulsion fundamentals. Table 1–2 compares several performance characteristics of specific chemical rockets with those of typical turbojets and ramjets. A high specific impulse is directly related to a long flight range and thus indicates the superior range capability of air breather engines over chemical rockets at relatively low altitude. The uniqueness of the rocket, for example, high thrust to weight, high thrust to frontal

TABLE 1–2. Comparison of Several Characteristics of a Typical Chemical Rocket and Two Duct Propulsion Systems

Feature	Rocket Engine or Rocket Motor	Turbojet Engine	Ramjet Engine
Thrust-to-weight ratio, typical	75:1	5:1, turbojet and afterburner	7:1 at Mach 3 at 30,000 ft
Specific fuel consumption (pounds of propellant or fuel per hour per pound of thrust)[a]	8–14	0.5–1.5	2.3–3.5
Specific thrust (pounds of thrust per square foot frontal area)[b]	5000 to 25,000	2500 (Low Mach at sea level)	2700 (Mach 2 at sea level)
Thrust change with altitude	Slight increase	Decreases	Decreases
Thrust vs. flight speed	Nearly constant	Increases with speed	Increases with speed
Thrust vs. air temperature	Constant	Decreases with temperature	Decreases with temperature
Flight speed vs. exhaust velocity	Unrelated, flight speed can be greater	Flight speed always less than exhaust velocity	Flight speed always less than exhaust velocity
Altitude limitation	None; suited to space travel	14,000–17,000 m	20,000 m at Mach 3 30,000 m at Mach 5 45,000 m at Mach 12
Specific impulse typical[c] (thrust force per unit propellant or fuel weight flow per second)	270 sec	1600 sec	1400 sec

[a] Multiply by 0.102 to convert to kg/hr-N.
[b] Multiply by 47.9 to convert to N/m^2.
[c] *Specific impulse* is a performance parameter and is defined in Chapter 2.

area, and thrust independence of altitude, enables extremely long flight ranges to be obtained in rarefied air and in space.

The *turbojet engine* is the most common of ducted engines. Figure 1–1 shows the basic elements.

At supersonic flight speeds above Mach 2, the *ramjet engine* (a pure duct engine) becomes attractive for flight within the atmosphere. Thrust is produced by increasing the momentum of the air as it passes through the ramjet, basically as is accomplished in the turbojet and turbofan engines but without compressors or turbines, Figure 1–2 shows the basic components of one type of ramjet. Ramjets with subsonic combustion and hydrocarbon fuel have an upper speed limit of approximately Mach 5; hydrogen fuel, with hydrogen cooling, raises this to at least Mach 16. Ramjets depend on rocket boosters, or some other method (such as being launched from an aircraft) for being accelerated to near their design flight speed to become functional. The primary applications have been in shipboard and ground-launched antiaircraft missiles. Studies of a hydrogen-fueled ramjet for hypersonic aircraft look promising. The supersonic flight vehicle is a combination of a ramjet-driven high-speed airplane and a one- or two-stage rocket booster. It can travel at speeds up to a Mach number of 25 at altitudes of up to 50,000 m.

1.2. ROCKET PROPULSION

Rocket propulsion systems can be classified according to the type of energy source (chemical, nuclear, or solar), the basic function (booster stage, sustainer, attitude control, orbit station keeping, etc.), the type of vehicle (aircraft, missile, assisted take-off, space vehicle, etc.), size, type of propellant, type of construction, or number of rocket propulsion units used in a given vehicle. Each is treated in more detail in subsequent chapters.

Another way is to classify by the method of producing thrust. A thermodynamic expansion of a gas is used in the majority of practical rocket propulsion concepts. The internal energy of the gas is converted into the kinetic energy of the exhaust flow and the thrust is produced by the gas pressure on the surfaces exposed to the gas, as will be explained later. This same thermo-

FIGURE 1–1. Simplified schematic diagram of a turbojet engine.

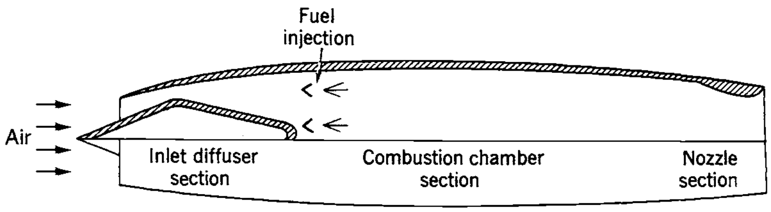

FIGURE 1–2. Simplified diagram of a ramjet with a supersonic inlet (converging and diverging flow passage).

dynamic theory and the same generic equipment (nozzle) is used for jet propulsion, rocket propulsion, nuclear propulsion, laser propulsion, solar-thermal propulsion, and some types of electrical propulsion. Totally different methods of producing thrust are used in other types of electric propulsion or by using a pendulum in a gravity gradient. As described below, these electric systems use magnetic and/or electric fields to accelerate electrically charged molecules or atoms at very low densities. It is also possible to obtain a very small acceleration by taking advantage of the difference in gravitational attraction as a function of altitude, but this method is not explained in this book.

The Chinese developed and used solid propellant in rocket missiles over 800 years ago and military bombardment rockets were used frequently in the eighteenth and nineteenth centuries. However, the significant developments of rocket propulsion took place in the twentieth century. Early pioneers included the Russian Konstantin E. Ziolkowsky, who is credited with the fundamental rocket flight equation and his 1903 proposals to build rocket vehicles. The German Hermann Oberth developed a more detailed mathematical theory; he proposed multistage vehicles for space flight and fuel-cooled thrust chambers. The American Robert H. Goddard is credited with the first flight using a liquid propellant rocket engine in 1926. An early book on the subject was written by the Viennese engineer Eugen Sänger. For rocket history see Refs. 1–3 to 1–7.

Chemical Rocket Propulsion

The energy from a high-pressure combustion reaction of propellant chemicals, usually a fuel and an oxidizing chemical, permits the heating of reaction product gases to very high temperatures (2500 to 4100°C or 4500 to 7400°F). These gases subsequently are expanded in a nozzle and accelerated to high velocities (1800 to 4300 m/sec or 5900 to 14,100 ft/sec). Since these gas temperatures are about twice the melting point of steel, it is necessary to cool or insulate all the surfaces that are exposed to the hot gases. According to the physical state of the propellant, there are several different classes of chemical rocket propulsion devices.

Liquid propellant rocket engines use liquid propellants that are fed under pressure from tanks into a thrust chamber.* A typical pressure-fed liquid propellant rocket engine system is schematically shown in Fig. 1–3. The liquid *bipropellant* consists of a liquid oxidizer (e.g., liquid oxygen) and a liquid fuel (e.g., kerosene). A *monopropellant* is a single liquid that contains both oxidizing and fuel species; it decomposes into hot gas when properly catalyzed. A large turbopump-fed liquid propellant rocket engine is shown in Fig. 1–4. Gas pressure feed systems are used mostly on low thrust, low total energy propulsion systems, such as those used for attitude control of flying vehicles, often with more than one thrust chamber per engine. Pump-fed liquid rocket systems are used typically in applications with larger amounts of propellants and higher thrusts, such as in space launch vehicles.

In the *thrust chamber* the propellants react to form hot gases, which in turn are accelerated and ejected at a high velocity through a supersonic nozzle, thereby imparting momentum to the vehicle. A nozzle has a converging section, a constriction or throat, and a conical or bell-shaped diverging section as further described in the next two chapters.

Some liquid rocket engines permit repetitive operation and can be started and shut off at will. If the thrust chamber is provided with adequate cooling capacity, it is possible to run liquid rockets for periods exceeding 1 hour, dependent only on the propellant supply. A liquid rocket propulsion system requires several precision valves and a complex feed mechanism which includes propellant pumps, turbines, or a propellant-pressurizing device, and a relatively intricate combustion or thrust chamber.

In *solid propellant rocket motors*† the propellant to be burned is contained within the combustion chamber or *case*. The solid propellant charge is called the *grain* and it contains all the chemical elements for complete burning. Once ignited, it usually burns smoothly at a predetermined rate on all the exposed internal surfaces of the grain. Initial burning takes place at the internal surfaces of the cylinder perforation and the four slots. The internal cavity grows as propellant is burned and consumed. The resulting hot gas flows through the supersonic nozzle to impart thrust. Once ignited, the motor combustion proceeds in an orderly manner until essentially all the propellant has been consumed. There are no feed systems or valves (see Fig. 1–5).

Liquid and solid propellants, and the propulsion systems that use them, are discussed in Chapters 6 to 10 and 11 to 14, respectively. Liquid and solid propellant rocket propulsion systems are compared in Chapter 17.

*The term *thrust chamber*, used for the assembly of the injector, nozzle, and chamber, is preferred by several official agencies and therefore has been used in this book. However, other terms, such as *thrust cylinder* and *combustor*, are still used in the literature. For small spacecraft control rockets the term *thruster* is commonly used and this term will be used in some sections of this book.

†Historically the word *engine* is used for a liquid propellant rocket propulsion system and the word *motor* is used for solid propellant rocket propulsion. They were developed originally by different groups.

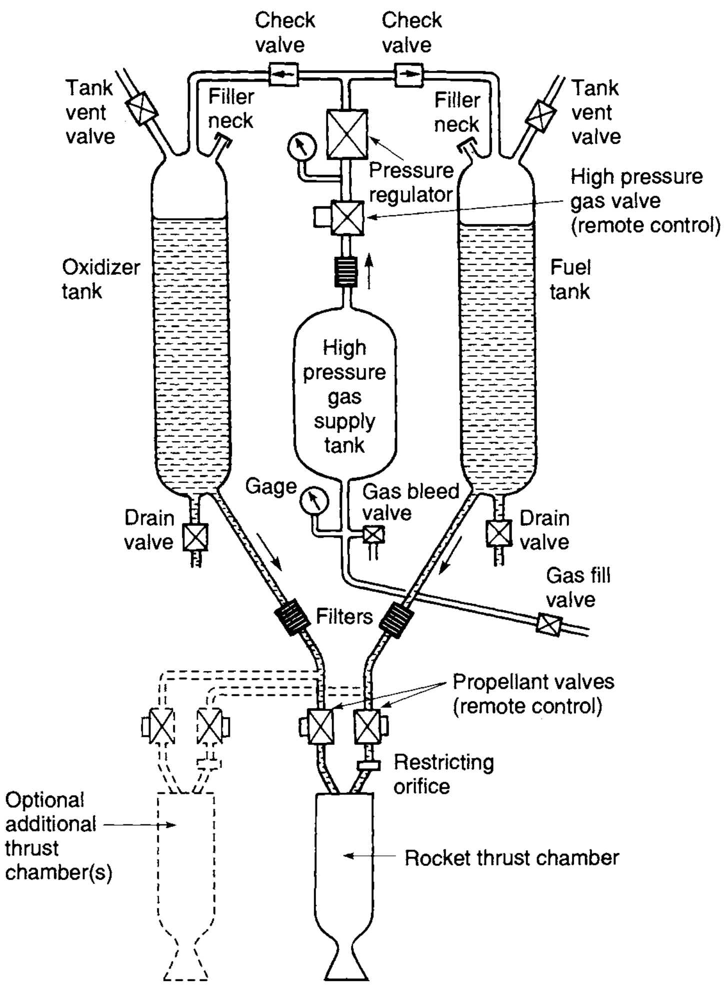

FIGURE 1–3. Schematic flow diagram of a liquid propellant rocket engine with a gas pressure feed system. The dashed lines show a second thrust chamber, but some engines have more than a dozen thrust chambers supplied by the same feed system. Also shown are components needed for start and stop, controlling tank pressure, filling propellants and pressurizing gas, draining or flushing out remaining propellants, tank pressure relief or venting, and several sensors.

Gaseous propellant rocket engines use a stored high-pressure gas, such as air, nitrogen, or helium, as their working fluid or propellant. The stored gas requires relatively heavy tanks. These cold gas engines have been used on many early space vehicles as attitude control systems and some are still used today. Heating the gas by electrical energy or by combustion of certain monopropellants improves the performance and this has often been called *warm gas propellant rocket propulsion.*

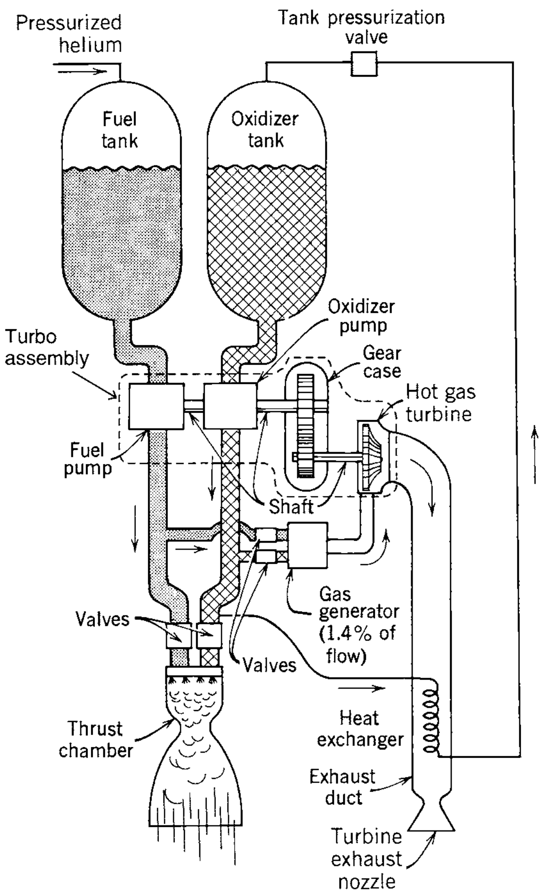

FIGURE 1–4. Simplified schematic diagram of one type of liquid propellant rocket engine with a turbopump feed system and a separate gas generator, which generates warm gas for driving the turbine. Not shown are components necessary for controlling the operation, filling, venting, draining, or flushing out propellants, filters or sensors. The turbopump assembly consists of two propellant pumps, a gear case, and a high speed turbine.

Hybrid propellant rocket propulsion systems use both a liquid and a solid propellant. For example, if a liquid oxidizing agent is injected into a combustion chamber filled with solid carbonaceous fuel grain, the chemical reaction produces hot combustion gases (see Fig. 1–6). They are described further in Chapter 15.

There are also chemical rocket propulsion *combination systems* that have both solid and liquid propellants. One example is a pressurized liquid propellant system that uses a solid propellant to generate hot gases for tank pressurization; flexible diaphragms are necessary to separate the hot gas and the reactive liquid propellant in the tank.

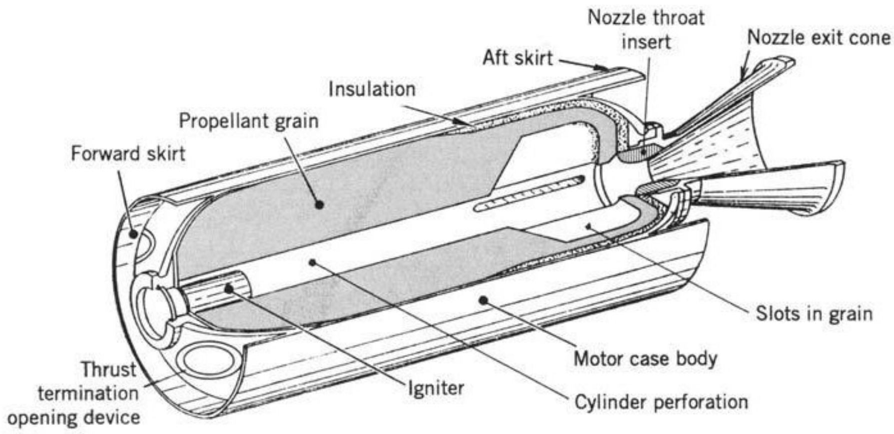

FIGURE 1–5. Simplified perspective three-quarter section of a typical solid propellant rocket motor with the propellant grain bonded to the case and the insulation layer and with a conical exhaust nozzle. The cylindrical case with its forward and aft hemispherical domes form a pressure vessel to contain the combustion chamber pressure. Adapted with permission from Reference 11–1.

Combinations of Ducted Jet Engines and Rocket Engines

The Tomahawk surface-to-surface missile uses two stages of propulsion in sequence. The solid propellant rocket booster lifts the missile away from its launch platform and is discarded after its operation. A small turbojet engine sustains the low level flight at nearly constant speed toward the target.

A *ducted rocket*, sometimes called an *air-augmented rocket*, combines the principles of rocket and ramjet engines; it gives higher performance (specific impulse) than a chemical rocket engine, while operating within the earth's atmosphere. Usually the term *air-augmented rocket* denotes mixing of air with the rocket exhaust (fuel-rich for afterburning) in proportions that enable the propulsion device to retain the characteristics typifying a rocket engine, for example, high static thrust and high thrust-to-weight ratio. In contrast, the ducted rocket often is like a ramjet in that it must be boosted to operating speed and uses the rocket components more as a fuel-rich gas generator (liquid, solid, or hybrid), igniter, and air ejecter pump.

The principles of the rocket and ramjet can be combined so that the two propulsion systems operate in sequence and in tandem and yet utilize a common combustion chamber volume as shown in Fig. 1–7. The low-volume configuration, known as an *integral rocket–ramjet*, can be attractive in air-launched missiles using ramjet propulsion (see Ref. 1–8). The transition from the rocket to the ramjet requires enlarging the exhaust nozzle throat (usually by ejecting rocket nozzle parts), opening the ramjet air inlet–combustion chamber interface, and following these two events with the normal ramjet starting sequence.

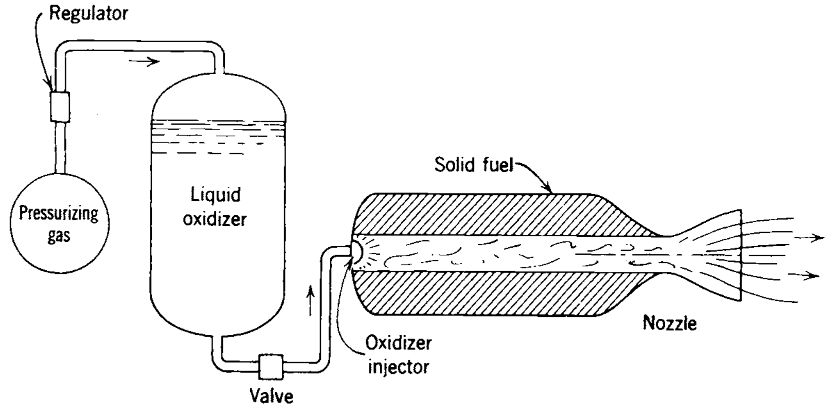

FIGURE 1–6. Simplified schematic diagram of a typical hybrid rocket engine. The relative positions of the oxidizer tank, high pressure gas tank, and the fuel chamber with its nozzle depend on the particular vehicle design.

A *solid fuel ramjet* uses a grain of solid fuel that gasifies or ablates and reacts with air. Good combustion efficiencies have been achieved with a patented boron-containing solid fuel fabricated into a grain similar to a solid propellant and burning in a manner similar to a hybrid rocket propulsion system.

Nuclear Rocket Engines

Three different types of nuclear energy sources have been investigated for delivering heat to a working fluid, usually liquid hydrogen, which subsequently can be expanded in a nozzle and thus accelerated to high ejection velocities (6000 to 10,000 m/sec). However, none can be considered fully developed today and none have flown. They are the *fission reactor*, the

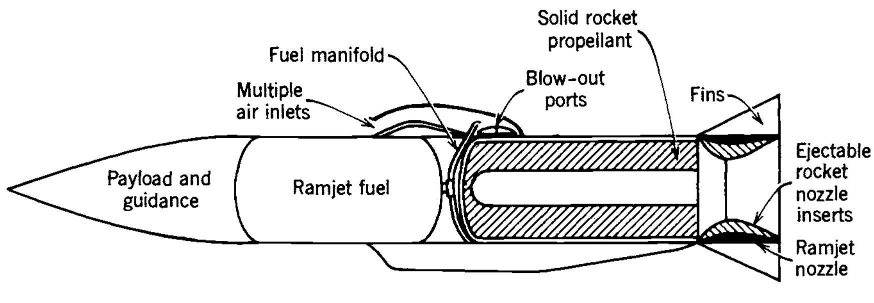

FIGURE 1–7. Elements of an air-launched missile with integral rocket–ramjet propulsion. After the solid propellant has been consumed in boosting the vehicle to flight speed, the rocket combustion chamber becomes the ramjet combustion chamber with air burning the ramjet liquid fuel.

radioactive isotope decay source, and the *fusion reactor*. All three types are basically extensions of liquid propellant rocket engines. The heating of the gas is accomplished by energy derived from transformations within the nuclei of atoms. In chemical rockets the energy is obtained from within the propellants, but in nuclear rockets the power source is usually separate from the propellant.

In the *nuclear fission reactor rocket*, heat can be generated by the fission of uranium in the solid reactor material and subsequently transferred to the working fluid (see Refs. 1–9 to 1–11). The *nuclear fission rocket* is primarily a high-thrust engine (above 40,000 N) with specific impulse values up to 900 sec. Fission rockets were designed and tested in the 1960s. Ground tests with hydrogen as a working fluid culminated in a thrust of 980,000 N (210,000 lb force) at a graphite core nuclear reactor level of 4100 MW with an equivalent altitude-specific impulse of 848 sec and a hydrogen temperature of about 2500 K. There were concerns with the endurance of the materials at the high temperature (above 2600 K) and intense radiations, power level control, cooling a reactor after operation, moderating the high-energy neutrons, and designing lightweight radiation shields for a manned space vehicle.

In recent years there have been renewed interest in nuclear fission rocket propulsion primarily for a potential manned planetary exploration mission. Studies have shown that the high specific impulse (estimated in some studies at 1100 sec) allows shorter interplanetary trip transfer times, smaller vehicles, and more flexibility in the launch time when planets are not in their optimum relative position.

In the *isotope decay engine* a radioactive material gives off radiation, which is readily converted into heat. Isotope decay sources have been used successfully for generating electrical power in space vehicles and some have been flown as a power supply for satellites and deep space probes. The released energy can be used to raise the temperature of a propulsive working fluid such as hydrogen or perhaps drive an electric propulsion system. It provides usually a lower thrust and lower temperature than the other types of nuclear rocket. As yet, isotope decay rocket engines have not been developed or flown.

Fusion is the third nuclear method of creating nuclear energy that can heat a working fluid. A number of different concepts have been studied. To date none have been tested and many concepts are not yet feasible or practical. Concerns about an accident with the inadvertent spreading of radioactive materials in the earth environment and the high cost of development programs have to date prevented a renewed experimental development of a large nuclear rocket engine. Unless there are some new findings and a change in world attitude, it is unlikely that a nuclear rocket engine will be developed or flown in the next few decades, therefore no further discussion of it is given in this book.

Electric Rocket Propulsion

In all electric propulsion the source of the electric power (nuclear, solar radiation receivers, or batteries) is physically separate from the mechanism that produces the thrust. This type of propulsion has been handicapped by heavy and inefficient power sources. The thrust usually is low, typically 0.005 to 1 N. In order to allow a significant increase in the vehicle velocity, it is necessary to apply the low thrust and thus a small acceleration for a long time (weeks or months) (see Chapter 19 and Refs. 1–12 and 1–13).

Of the three basic types, electrothermal rocket propulsion most resembles the previously mentioned chemical rocket units; propellant is heated electrically (by heated resistors or electric arcs) and the hot gas is then thermodynamically expanded and accelerated to supersonic velocity through an exhaust nozzle (see Fig. 1–8). These electrothermal units typically have thrust ranges of 0.01 to 0.5 N, with exhaust velocities of 1000 to 5000 m/sec, and ammonium, hydrogen, nitrogen, or hydrazine decomposition product gases have been used as propellants.

The two other types—the electrostatic or ion propulsion engine and the electromagnetic or magnetoplasma engine—accomplish propulsion by different principles and the thermodynamic expansion of gas in a nozzle, as such, does not apply. Both will work only in a vacuum. In an ion rocket (see Fig. 1–9) a working fluid (typically, xenon) is ionized (by stripping off electrons) and then the electrically charged heavy ions are accelerated to very high velocities (2000 to 60,000 m/sec) by means of electrostatic fields. The ions are subsequently electrically neutralized; they are combined with electrons to prevent the buildup of a space charge on the vehicle.

In the magnetoplasma rocket an electrical plasma (an energized hot gas containing ions, electrons, and neutral particles) is accelerated by the interaction between electric currents and magnetic fields and ejected at high velocity

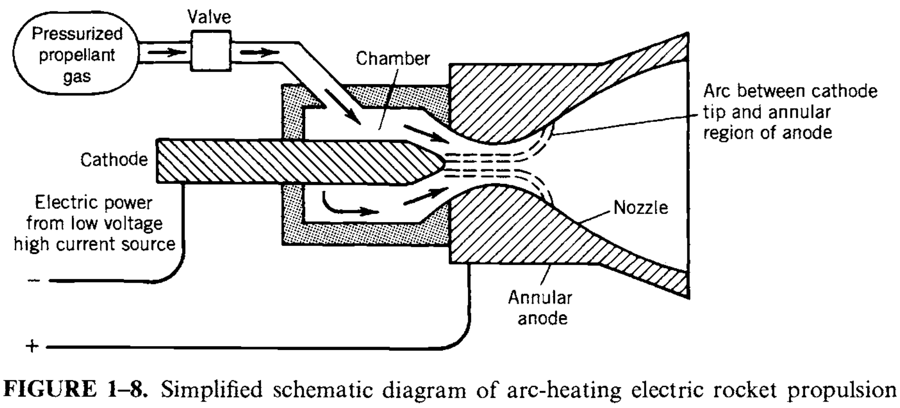

FIGURE 1–8. Simplified schematic diagram of arc-heating electric rocket propulsion system. The arc plasma temperature is very high (perhaps 15,000 K) and the anode, cathode, and chamber will get hot (1000 K) due to heat transfer.

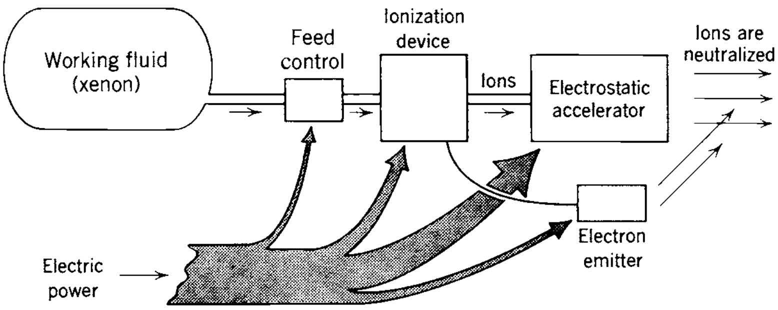

FIGURE 1–9. Simplified schematic diagram of a typical ion rocket, showing the approximate distribution of the electric power.

(1000 to 50,000 m/sec). There are many different types and geometries. A simple pulsed (not continuously operating) unit with a solid propellant is shown in Fig. 1–10. This type has had a good flight record as a spacecraft attitude control engine.

Other Rocket Propulsion Concepts

Several technologies exist for harnessing solar energy to provide the power for spacecraft and also to propel spacecraft using electrical propulsion. Solar cells generate electric power from the sun's radiation. They are well developed and have been successful for several decades. Most electric propulsion systems have used solar cells for their power supply.

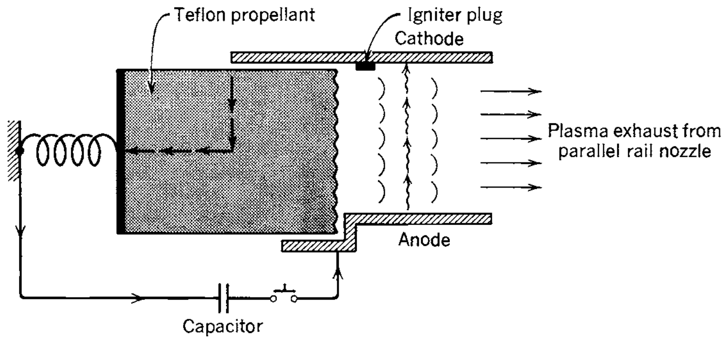

FIGURE 1–10. Simplified diagram of a rail accelerator for self-induced magnetic acceleration of a current-carrying plasma. When the capacitor is discharged, an arc is struck at the left side of the rails. The high current in the plasma arc induces a magnetic field. The action of the current and the magnetic field causes the plasma to be accelerated at right angles to both the magnetic field and the current, namely in the direction of the rails. Each time the arc is created a small amount of solid propellant (Teflon) is vaporized and converted to a small plasma cloud, which (when ejected) gives a small pulse of thrust. Actual units can operate with many pulses per second.

An attractive concept, the *solar thermal rocket*, has large diameter optics to concentrate the sun's radiation (e.g., by lightweight precise parabolic mirrors or Fresnel lenses) onto a receiver or optical cavity. Figure 1–11 shows one concept and some data is given in Table 2–1. The receiver is made of high temperature metal (such as tungsten or rhenium) and has a cooling jacket or heat exchanger. It heats a working fluid, usually liquid hydrogen, up to perhaps 2500°C and the hot gas is controlled by hot gas valves and exhausted through one or more nozzles. The large mirror has to be pointed toward the sun and this requires the mirror to be adjustable in its orientation. Performance can be two to three times higher than that of a chemical rocket and thrust levels in most studies are low (1 to 10 N). Since large lightweight optical elements cannot withstand drag forces without deformation, the optical systems are deployed outside the atmosphere. Contamination is negigible, but storage or refueling of liquid hydrogen is a challenge. Problems being investigated include rigid, lightweight mirror or lens structures, operational life, minimizing hydrogen evaporation, and heat losses to other spacecraft components. To date the solar thermal rocket has not yet provided the principal thrust of a flying spacecraft.

The *solar sail* is another concept. It is basically a big photon reflector surface. The power source for the solar sail is the sun and it is external to the vehicle (see Ref. 1–14). Approaches using nuclear explosions and pulsed nuclear fusion have been analyzed (Refs. 1–15 and 1–16), but are not yet feasible. Concepts for transmitting radiation energy (by lasers or microwaves) from earth stations to satellites have been proposed, but are not yet developed.

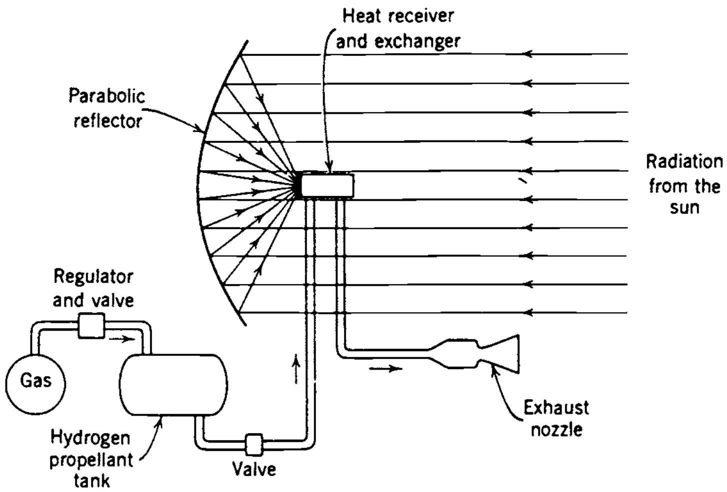

FIGURE 1-11. Simplified schematic diagram of a solar thermal rocket concept.

International Rocket Propulsion Effort

Active development or production of rocket propulsion systems is currently under way in more than 30 different countries. Some of them have made significant and original contributions to the state of the art of the technologies. There is mention in this book of a few foreign rocket units and their accomplishments and references to international rocket literature. Although most of the data in this book are taken from U.S. rocket experience, this is not intended to minimize foreign achievements.

At the time of this writing the major international program was the *International Space Station* (ISS), a multi-year cooperative effort with major contributions from the USA and Russia and active participation by several other nations. This manned orbital space station is used for conducting experiments and observations on a number of research projects.

1.3. APPLICATIONS OF ROCKET PROPULSION

Because the rocket can reach a performance unequaled by other prime movers, it has its own fields of application and does not usually compete with other propulsion devices. Examples of important applications are given below and discussed further in Chapter 4.

Space Launch Vehicles

Between the first space launch in 1957 and the end of 1998 approximately 4102 space launch attempts have taken place in the world and all but about 129 were successful (see Ref. 1–17). *Space launch vehicles* or *space boosters* can be classified broadly as expendable or recoverable/reusable. Other bases of classification are the type of propellant (storable or cryogenic liquid or solid propellants), number of stages (single-stage, two-stage, etc.), size/mass of payloads or vehicles, and manned or unmanned. Figure 1–12 shows the Titan III-C space launch vehicle, one member of the Titan family of storable propellant space launch vehicles, which is used extensively for boosting satellites into synchronous earth orbit or into escape trajectories for planetary travel. This heavy-duty launch vehicle consists of the basic 2-stage Titan III standard launch vehicle (liquid propellant rockets) supplemented by two solid propellant "strap-on motors." A fourth stage, known as the transtage, permits a wide variety of maneuvers, orbit changes, and trajectory transfers to be accomplished with the payload, which can be one or more satellites or spacecraft.

Each space launch vehicle has a specific space flight objective, such as an earth orbit or a moon landing. It uses between two and five stages, each with its own propulsion system, and each is usually fired sequentially after the lower stage is expended. The number of stages depends on the specific space trajectory, the number and types of maneuvers, the energy content of a unit mass of

FIGURE 1–12. Titan III launch vehicle shortly after lift-off, with bright radiant exhaust gas. Two solid propellant rocket motors, each providing about 2.4 million pounds of thrust, boost the first stage, which also gets a sustained thrust of 470,000 pounds from two liquid rocket engines. The second stage has 100,000 pounds of thrust from a single liquid rocket engine, and one version of the third stage has two liquid rocket engines, each at 16,000 pounds of thrust.

the propellant, and other factors. The initial stage, usually called the booster stage, is the largest and it is operated first; this stage is then separated from the ascending vehicle before the second-stage rocket propulsion system is ignited and operated. As will be explained in Chapter 4, adding an extra stage permits a significant increase in the payload (such as more scientific instruments or more communications gear).

Each stage of a multistage launch vehicle is essentially a complete vehicle in itself and carries its own propellant, its own rocket propulsion system or systems, and its own control system. Once the propellant of a given stage is expended, the dead mass of that stage (including empty tanks, cases, instruments, etc.) is no longer useful in providing additional kinetic energy to the succeeding stages. By dropping off this useless mass it is possible to accelerate the final stage with its useful payload to a higher terminal velocity than would be attained if multiple staging were not used. Both solid propellant and liquid propellant rocket propulsion systems have been used for low earth orbits.

A single stage to orbit vehicle, attractive because it avoids the costs and complexities of staging, is expected to have improved reliability (simple structures, fewer components), and some versions may be recoverable and reusable. However, its payload is relatively very small. A low earth orbit (say 100 miles altitude) can only be achieved with such a vehicle if the propellant performance is very high and the structure is efficient and low in mass. Liquid propellants such as liquid hydrogen with liquid oxygen are usually chosen.

The missions and payloads for space launch vehicles are many, such as military (reconnaissance satellites, command and control satellites), non-military government (weather observation satellites, GPS or geopositioning satellites), space exploration (space environment, planetary missions), or commercial (communication satellites). Forecasts indicate that a large number of future commercial communications satellites will be needed.

Table 1–3 lists several important U.S. launch vehicles and their capabilities and Table 1–4 gives data on the Space Shuttle, which is really a combination of launch vehicle, spacecraft, and a glider. It can be seen that the thrust levels are highest for booster or first stages and are relatively high for upper stages (thousands of pounds). Only for the attitude control system of the vehicle (also called reaction control in Table 1–4) are the thrust levels low (from a fraction of a pound for small spacecraft to as high as about 1000 pounds thrust in the space shuttle vehicle). Frequent propulsion starts and stops are usually required in these applications.

Spacecraft

Depending on their missions, *spacecraft* can be categorized as earth satellites, lunar, interplanetary, and trans-solar types, and as manned and unmanned spacecraft. Rocket propulsion is used for both primary propulsion (i.e.,

TABLE 1–3. Selected United States Space Launch Vehicles

Name	Stage	Number of Engines or Motors per Stage	Thrust kN	Thrust lbf	Propellants	Launch Mass (metric tons)	Two-stage Payload Weight 100 n·mi (185 km) Orbit kg	Two-stage Payload Weight 100 n·mi (185 km) Orbit lbf	Three-stage Payload Weight Geosynchronous Orbit kg	Three-stage Payload Weight Geosynchronous Orbit lbf
Titan 34D						1091	13,600	30,000	1820	4000
	0	2	10,750	2,400,000 vac	Solid composite					
	1	2	2370	529,000	N_2O_4/N_2H_4					
	2	1	452	101,000	+ UDMH					
	3	1	107	23,800	Solid composite					
Delta II 6925						132	2545	5600	1454	3200
	0	6 + 3	443.5	Each 97,000 SL	Solid composite					
	1	1	927	207,000 SL	LO_2/RP-1					
			1037	231,700 vac						
	2	1	43.2	9645	N_2O_4/N_2H_4 – UDMH					
	3	1	67.6	15,100 vac	Solid composite					
Atlas Centaur						141	2772	6100	1545	3400
	$\frac{1}{2}$	2	Each 829 SL	Each 185,000 SL	LO_2/RP-1					
	1	1	269	60,000	LO_2/RP-1					
	2	2	Each 74 vac	Each 16,500 vac	LO_2/LH_2					
Pegasus (air-launched)						23.1	490 (Three stages)	1078	NA	NA
	1	1	726	163,000	Solid					
	2	1	196	44,200	Solid					
	3	1	36	8060	Solid					

"SL" refers to sea level and "vac" refers to altitude or vacuum conditions.

TABLE 1–4. Propulsion Systems for the Space Shuttle

Vehicle Section	Propulsion System (No. of Units)	Number of Starts and Typical Burn Time	Propellant and Specific Impulse	Thrust	Mission
Shuttle orbiter	Space Shuttle main engine (3)	Start at launch 8.4 min duration Life: 55 starts and 7.5 hr	Liquid hydrogen–liquid oxygen 4464 N-sec/kg (455 sec)	1670 kN each (375,000 lb) at sea level 2100 kN each (470,000 lbf) at space vacuum Throttled 109 to 65% of rated power	Lift orbiter off ground and accelerate to orbit velocity. Individual engines can be shut down to reduce thrust level.
	Orbital maneuver systems (2)	3 to 10 starts/mission; designed for 1000 starts, 100 flights, 15 hours of cumulative time	See Note 1; $I_s = 313$ sec	27 kN each (6000 lbf) in vacuum	Insert orbiter vehicle into earth orbit, correct orbit, abort, and deorbit maneuver.
	Reaction control system, 38 primary thrusters, 6 vernier thrusters	Multiple operations; thousands of starts; duration from a few milliseconds to seconds	See Note 1; $I_s =$ 280–304 sec, depending on nozzle area ratio	Primary thruster 3870 N each (870 lbf), vernier thruster 106.8 N each (25 lbf)	Small vehicle velocity adjustments and attitude control during orbit insertion, on orbit corrections, rendezvous, and reentry.
Solid rocket boosters (SRBs)	Attached to external tank; multisection, 2 units	Single start at launch 2 min	See Note 2	14,700 kN each, or 3.3×10^6 lbf each	Boost Shuttle vehicle to about 5500 km/hr
	Separation rocket motors; 16 units	4 each at forward frustum and aft skirt; 0.66 sec, nominal	Solid propellant; $I_s = 250$ sec	97,840 N each or 22,000 lbf	Move SRB away from vehicle after cut-off

Notes:
1. MMH, monomethylhydrazine and NTO, nitrogen tetroxide.
2. 70% Ammonium perchlorate; 16% aluminum; 12% polybutadiene acrylic acid binder; 2% epoxy curing agent.

along the flight path, such as for orbit insertion or orbit change maneuvers) and secondary propulsion functions in these vehicles. Some of the *secondary propulsion* functions are attitude control, spin control, momentum wheel and gyro unloading, stage separation, and the settling of liquids in tanks. A spacecraft usually has a series of different rocket propulsion systems, some often very small. For spacecraft attitude control about three perpendicular axes, each in two rotational directions, the system must allow the application of pure torque for six modes of angular freedom, thus requiring a minimum of 12 thrust chambers. Some missions require as few as four to six rocket units whereas the more complex manned spacecraft have 40 to 80 rocket units in all of its stages. Often the small *attitude control rockets* must give pulses or short bursts of thrust, necessitating thousands of restarts.

Table 1–5 presents a variety of spacecraft along with their weights, missions, and propulsion. Although only U.S. launch vehicles are listed in this table, there are also launch vehicles developed by France, the European Space Agency, Russia, Japan, China, India, and Israel that have successfully launched payloads into satellite orbits. They use rocket propulsion systems that were developed in their own countries.

The U.S. Space Shuttle program, using technology and experience from the X-15 rocket-powered research airplane, the Mercury and Gemini orbital flights, the Apollo lunar flight program, and Skylab, provided the first *reusable spacecraft* that lands on a runway. Figure 1–13 shows the basic configuration of the Space Shuttle, which consists of two stages, the booster and the orbiter. It shows all the 67 rocket propulsion systems of the shuttle. The orbiter is really a reusable combination vehicle, namely a spacecraft combined with a glider. The two solid propellant rocket motors are the largest in existence; they are equipped with parachutes for sea recovery of the burned-out motors. The large liquid oxygen/liquid hydrogen (LO_2/LH_2) external tank is jettisoned and expended just before orbit insertion (see Ref. 1–18). Details of several of these Space Shuttle rocket propulsion systems are given elsewhere in this book. The Space Shuttle accomplishes both civilian and military missions of placing satellites in orbit, undertaking scientific exploration, and repairing, servicing, and retrieving satellites.

A reusable single stage to orbit, experimental vehicle with a novel rocket engine is currently (1997) under development in the USA. It is a combination launch vehicle and spacecraft. The design takes advantage of advances in lightweight structures, a clever lifting aerodynamic body concept, and a tailored novel rocket engine that requires little space and fits well into the flight vehicle. This engine, known as a linear aerospike, has a novel configuration and is described further in Chapter 8.

The majority of spacecraft have used liquid propellant engines, with solid propellant boosters. Several spacecraft have operated successfully with electrical propulsion for attitude control. Electrical propulsion systems will probably also be used for some primary and secondary propulsion missions on long-duration space flights, as described in Chapter 19.

TABLE 1–5. Selected United States Spacecraft

Name	Space Maneuver Propulsion		Weight	Remarks
	Thrust (lbf)	Propellants[a]	(lbf)	
Mariner 69	50 (primary)	Hydrazine monopropellant	1100	Flyby of Venus/Mercury
	1.0 (secondary)	Hydrazine monopropellant		
Pioneer 10, 11	50 (primary)	Hydrazine monopropellant	570	Fly to Jupiter and beyond
Viking	600 (primary)	Hydrazine monopropellant	7500	Mars orbiter with soft lander
	5.0 (secondary)	Hydrazine monopropellant		
Nimbus 5	0.5 (secondary)	Stored nitrogen	1700	Weather satellite
Apollo command and service module	20,500 (primary)	N_2O_4/50:50 UDMH $-N_2H_4$	64,500	Manned lunar landing
	100 lbf 16 units			
	93 lbf 6 units (secondary)	N_2O_4/MMH		
Space Shuttle orbiter	Two 6000-lbf units (primary)	N_2O_4/MMH	150,000	Reusable spacecraft with runway landing
	38 units @ 900 lbf (secondary)	N_2O_4/MMH		
	Six 25-lbf units (secondary)	N_2O_4/MMH		
Fleet Communications Satellite	0.1 (secondary)	Hydrazine monopropellant	1854	UHF communications
Photo Recon	4.0 (secondary)	Hydrazine monopropellant	25,000	Radio/photo communications
Intelsat V communication satellite	0.10	Hydrazine	4180	Resistojet, electric propulsion for N–S station keeping
Deep Space I (DS1)	0.02 (primary)	Xenon	1070	Ion propulsion engine for asteroid fly-by

[a] N_2O_4, nitrogen tetroxide (oxidizer); MMH, monomethylhydrazine (fuel); 50:50 UDMH–N_2H_4 is a 50% mixture of unsymmetrical dimethylhydrazine and hydrazine.

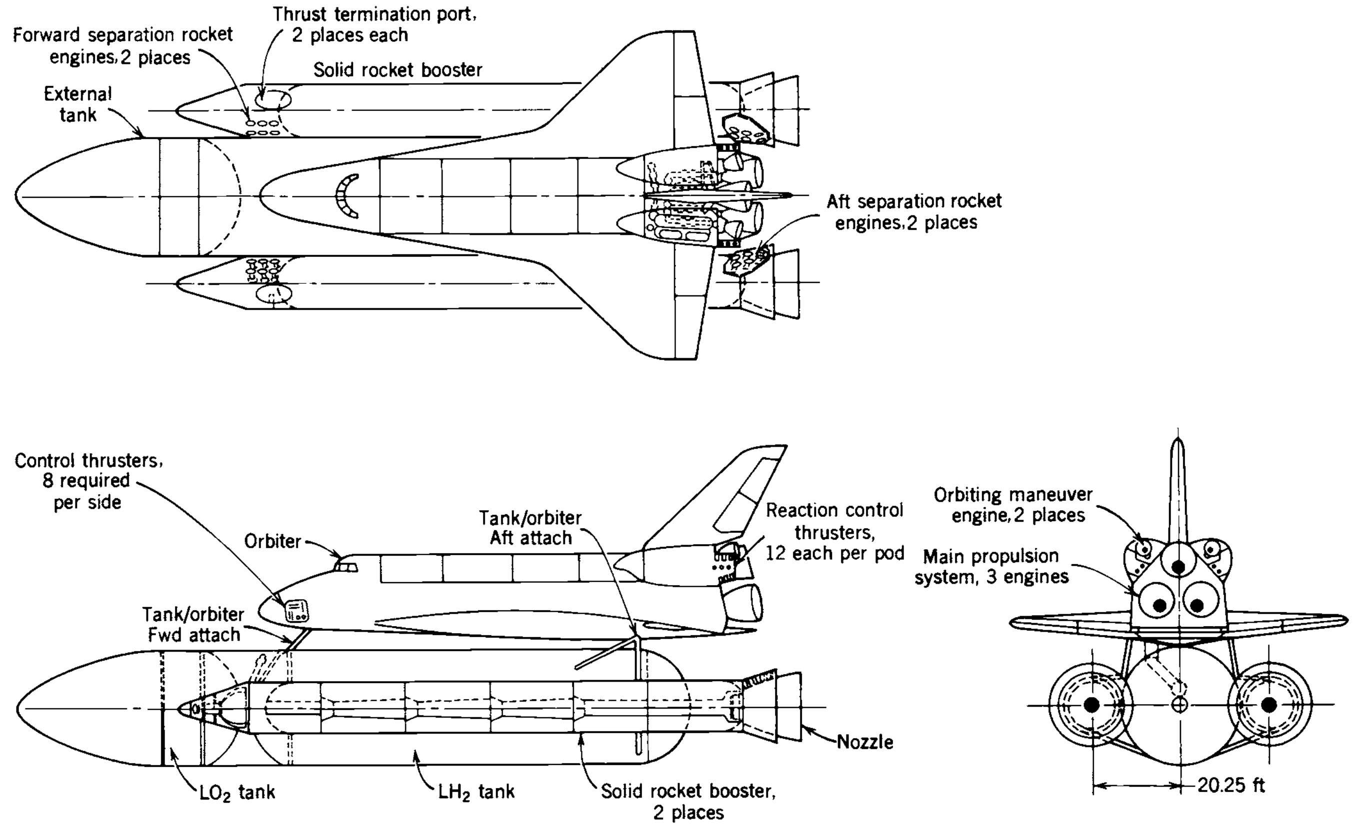

FIGURE 1–13. Simplified sketch of the Space Shuttle vehicle. The Shuttle Orbiter—the delta-winged vehicle about the size of a medium-range jet liner—is a reusable, cargo carrying, spacecraft–airplane combination that takes off vertically and lands horizontally like a glider. Each shuttle orbiter was designed for a minimum of 100 missions and can carry as much as 65,000 lb of payload to a low Earth orbit, and a crew of up to four members and 10 passengers. It can return up to 25,000 lb of payload back to Earth.

TABLE 1–6. Selected United States Missiles

Mission Category	Name	Diameter (ft)	Length (ft)	Propulsion	Launch Weight (lb)
Surface-to-surface (long range)	Minuteman III	6.2	59.8	3 stages, solid	78,000
	Poseidon	6.2	34	2 stages, solid	65,000
	Titan II	10	103	2 stages, liquid	330,000
Surface-to-air (or to missile)	Chaparral	0.42	9.5	1 stage, solid	185
	Improved Hawk	1.2	16.5	1 stage, solid	1398
	Standard Missile	1.13	15 or 27	2 stage, solid	1350/2996
	Redeye	0.24	4	1 stage, solid	18
	Patriot	1.34	1.74	1 stage, solid	1850
Air-to-surface	Maverick	1.00	8.2	1 stage, solid	475
	Shrike	0.67	10	1 stage, solid	400
	SRAM	1.46	14	2 staged grains	2230
Air-to-air	Falcon	0.6	6.5	1 stage, solid	152
	Phoenix	1.25	13	1 stage, solid	980
	Sidewinder	0.42	9.5	1 stage, solid	191
	Sparrow	0.67	12	1 stage, solid	515
Antisubmarine	Subroc	1.75	22	1 stage, solid	4000
Battlefield Support (surface-to-surface, short range)	Lance	1.8	20	2 stages, liquid	2424
	Hellfire (antitank)	0.58	5.67	1 stage, solid	95
	Pershing II	3.3	34.5	2 stages, solid	10,000
	Tow (antitank)	0.58	3.84	1 stage, solid	40
cruise missile (subsonic)	Tomahawk	1.74	21	solid booster + turbofan	3900

TABLE 1–7. Typical Propulsion Characteristics of Some Rocket Applications

Application	Type of Propellant	Thrust Profile	Typical Duration	Maximum Acceleration[a]
Large space launch vehicle booster	Solid or cryogenic liquid	Nearly constant thrust	2–8 min	2–6 g_0
Antiaircraft or antimissile-missile	Solid, some with liquid terminal divert stage	High thrust boost, decreasing thrust sustain phase	2–75 sec each	5 to 20 g_0, but can be up to 100 g_0
Spacecraft orbit maneuvers	Storable liquid or cryogenic liquid	Restartable	Up to 10 min cumulative duration	0.2–6 g_0
Air launched guided missile	Solid	High thrust boost phase with low thrust or decreasing thrust for sustain phase; sometimes 2 pulses	Boost: 2–5 sec Sustain: 10–30 sec	Up to 25 g_0
Battlefield support—surface launched	Solid	Same as above	Up to 2 min each stage	Up to 10 g_0
Rocket assisted projectile, gun launched	Solid	Increase and then decrease in thrust	A few sec	Up to 20,000 g_0
Spacecraft attitude control–large vehicles	Storable liquid (monopropellant or bipropellant); electric propulsion; xenon	Many restarts (up to 60,000); pulsing	Up to 1 hr cumulative duratiaon	Less than 0.1 g_0
Spacecraft attitude control—small vehicle	Cold or warm gas or storable liquid, electric propulsion	Same	Up to 40 min cumulative	Same
Reusable main engines for space shuttle	Cryogenic liquid (O_2/H_2)	Variable thrust, many flights with same engine	8 min, over 7 hr cumulative in several missions	
Single stage to orbit (has not yet flown)	Cryogenic liquid (O_2/H_2)	Throttled to lower thrust	6–10 min	4–7 g_0
Lunar landing	Storable bipropellant	10:1 thrust variation	4 min	Several g_0
Weather sounding rocket	Solid	Single burn period—often decreasing thrust	5–50 sec	Up to 15 g_0
Antitank	Solid	Single burn period	0.2–3 sec	Up to 20 g_0

[a] g_0 is acceleration of gravity at the Earth's surface = 9.8066 m/sec^2 or 32.17 ft/sec^2

Missiles and Other Applications

Military missiles can be classified as shown in Table 1–6. Rocket propulsion for new U.S. missiles uses now almost exclusively solid propellant rocket motors. They can be *strategic missiles*, such as long-range ballistic missiles (800 to 9000 km range) which are aimed at military targets within an enemy country, or *tactical missiles*, which are intended to support or defend military ground forces, aircraft, or navy ships.

The term *surface launch* can mean a launch from the ground, the ocean surface (from a ship), or from underneath the sea (submarine launch). Some tactical missiles, such as the air-to-surface SRAM missile, have a two-pulse solid propellant motor, where two separate, insulated grains are in the same motor case; the time interval before starting the second pulse can be timed to control the flight path or speed profile. Most countries now have tactical missiles in their military inventories, and many of these countries have a capability to produce their own rocket propulsion systems that are used to propel them.

Other applications of rockets include primary engines for research airplanes, assist-take-off rockets for airplanes, ejection of crew escape capsules and stores, personnel "propulsion belts,"and propulsion for target drones, weather sounding rockets, signal rockets, decoy rockets, spin rockets, vernier rockets, underwater rockets for torpedoes and missiles, the throwing of lifelines to ships, and "Fourth of July" rockets.

Tables 1–6 and 1–7 show some parameters of rocket propulsion devices for different applications. The selection of the best rocket propulsion system type and design for any given application is a complex process involving many factors, including system performance, reliability, propulsion system size, and compatibility, as described in Chapter 17. Comparisons and evaluations of many of these criteria are discussed in this book. Many factors, such as development, production or operating costs, available technology, and service life, though beyond the scope of this book, enter strongly into such a selection.

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CHAPTER 2

DEFINITIONS AND FUNDAMENTALS

Rocket propulsion is an exact but not a fundamental subject, and there are no basic scientific laws of nature peculiar to propulsion. The basic principles are essentially those of mechanics, thermodynamics, and chemistry.

Propulsion is achieved by applying a force to a vehicle, that is, accelerating the vehicle or, alternatively, maintaining a given velocity against a resisting force. This propulsive force is obtained by ejecting propellant at high velocity. This chapter deals with the definitions and the basic relations of this propulsive force, the exhaust velocity, and the efficiencies of creating and converting the energy and other basic parameters. The symbols used in the equations are defined at the end of the chapter. Wherever possible the American Standard letter symbols for rocket propulsion (as given in Ref. 2–1) are used.

2.1. DEFINITIONS

The *total impulse* I_t is the thrust force F (which can vary with time) integrated over the burning time t.

$$I_t = \int_0^t F \, dt \tag{2–1}$$

For constant thrust and negligible start and stop transients this reduces to

$$I_t = Ft \tag{2–2}$$

I_t is proportional to the total energy released by all the propellant in a propulsion system.

The *specific impulse* I_s is the total impulse per unit weight of propellant. It is an important figure of merit of the performance of a rocket propulsion system, similar in concept to the miles per gallon parameter used with automobiles. A higher number means better performance. Values of I_s are given in many chapters of this book and the concept of an optimum specific impulse for a particular mission is introduced later. If the total mass flow rate of propellant is $\dot{m}$ and the standard acceleration of gravity at sealevel g_0 is 9.8066 m/sec^2 or 32.174 ft/sec^2, then

$$I_s = \frac{\int_0^t F\,dt}{g_0 \int \dot{m}\,dt} \tag{2–3}$$

This equation will give a time-averaged specific impulse value for any rocket propulsion system, particularly where the thrust varies with time. During transient conditions (during start or the thrust buildup period, the shutdown period, or during a change of flow or thrust levels) values of I_s can be obtained by integration or by determining average values for F and $\dot{m}$ for short time intervals. For constant thrust and propellant flow this equation can be simplified; below, m_p is the total effective propellant mass.

$$I_s = I_t/(m_p g_0) \tag{2–4}$$

In Chapter 3 there is further discussion of the specific impulse. For constant propellant mass flow $\dot{m}$, constant thrust F, and negligibly short start or stop transients:

$$\begin{aligned} I_s &= F/(\dot{m} g_0) = F/\dot{w} \\ I_t/(m_p g_0) &= I_t/w \end{aligned} \tag{2–5}$$

The product $m_p g_0$ is the total effective propellant weight w and the weight flow rate is $\dot{w}$. The concept of weight relates to the gravitational attraction at or near sea level, but in space or outer satellite orbits, "weight" signifies the mass multiplied by an arbitrary constant, namely g_0. In the *Système International* (SI) or metric system of units I_s can be expressed simply in "seconds," because of the use of the constant g_0. In the USA today we still use the English Engineering (EE) system of units (foot, pound, second) in many of the chemical propulsion engineering, manufacturing, and test operations. In many past and current US publications, data and contracts, the specific impulse has units of thrust (lbf) divided by weight flow rate of propellants (lbf/sec), simplified as seconds. The numerical value of I_s is the same in the EE and the SI system of units. However, the units of I_s do not represent a measure of elapsed time, but a thrust force per unit "weight"-flow-rate. In this book the symbol I_s is used for the specific impulse, as listed in Ref. 2–1. For solid propellant systems the symbol I_{sp} is sometimes used, as listed in Ref. 2–2.

In a rocket nozzle the actual exhaust velocity is not uniform over the entire exit cross-section and does not represent the entire thrust magnitude. The velocity profile is difficult to measure accurately. For convenience a uniform axial velocity c is assumed which allows a one-dimensional description of the problem. This *effective exhaust velocity* c is the average equivalent velocity at which propellant is ejected from the vehicle. It is defined as

$$c = I_s g_0 = F/\dot{m} \tag{2–6}$$

It is given either in meters per second or feet per second. Since c and I_s differ only by an arbitrary constant, either one can be used as a measure of rocket performance. In the Russian literature c is generally used.

In solid propellant rockets it is difficult to measure the propellant flow rate accurately. Therefore, the specific impulse is often calculated from total impulse and the propellant weight (using the difference between initial and final motor weights and Eq. 2–5). In turn the total impulse is obtained from the integral of the measured thrust with time, using Eq. 2–1. In liquid propellant units it is possible to measure thrust and instantaneous propellant flow rate and thus to use Eq. 2–3 for calculation of specific impulse. Eq. 2–4 allows another definition for specific impulse, namely, the amount of impulse imparted to a vehicle per unit sea-level weight of propellant expended.

The term *specific propellant consumption* refers to the reciprocal of the specific impulse and is not commonly used in rocket propulsion. It is used in automotive and duct propulsion systems. Typical values are listed in Table 1–2.

The mass ratio **MR** of a vehicle or a particular vehicle stage is defined to be the final mass m_f (after rocket operation has consumed all usable propellant) divided by m_0 (before rocket operation). The various terms are depicted in Fig. 4–1.

$$\mathbf{MR} = m_f/m_0 \tag{2–7}$$

This applies to a single or a multi-stage vehicle; for the latter, the overall mass ratio is the product of the individual vehicle stage mass ratios. The final mass m_f is the mass of the vehicle after the rocket has ceased to operate when all the useful propellant mass m_p has been consumed and ejected. The final vehicle mass m_f includes all those components that are not useful propellant and may include guidance devices, navigation gear, payload (e.g., scientific instruments or a military warhead), flight control systems, communication devices, power supplies, tank structure, residual or unusable propellant, and all the propulsion hardware. In some vehicles it can also include wings, fins, a crew, life support systems, reentry shields, landing gears, etc. Typical values of **MR** can range from 60% for some tactical missiles to less than 10% for some unmanned launch vehicle stages. This mass ratio is an important parameter in analyzing flight performance, as explained in Chapter 4. When **MR** is applied to a single stage, then its upper stages become the "payload."

The *propellant mass fraction* ζ indicates the fraction of propellant mass m_p in an initial mass m_0. It can be applied to a vehicle, a stage of a vehicle or to a rocket propulsion system.

$$\zeta = m_p/m_0 \tag{2–8}$$

$$\zeta = (m_0 - m_f)/m_0 = m_p/(m_p + m_f) \tag{2–9}$$

$$m_0 = m_f + m_p \tag{2–10}$$

When applied to a rocket propulsion system, the mass ratio **MR** and propellant fraction ζ are different from those that apply to a vehicle as described above. Here the initial or loaded mass m_0 consists of the inert propulsion mass (the hardware necessary to burn and store the propellant) and the effective propellant mass. It would exclude masses of nonpropulsive components, such as payload or guidance devices. For example, in a liquid propellant rocket engine the final or inert propulsion mass m_f would include the propellant feed tanks, the pressurization system (with turbopump and/or gas pressure system), one or more thrust chambers, various piping, fittings and valves, an engine mount or engine structure, filters and some sensors. The *residual or unusable remaining propellant* is usually considered to be part of the final inert mass m_f, as it will be in this book. However, some rocket propulsion manufacturers and some literature assign residuals to be part of the propellant mass m_p. When applied to a rocket propulsion system, the value of the propellant mass fraction ζ indicates the quality of the design; a value of, say, 0.91 means that only 9% of the mass is inert rocket hardware and this small fraction contains, feeds, and burns a substantially larger mass of propellant. A high value of ζ is desirable.

The *impulse-to-weight* ratio of a complete propulsion system is defined as the total impulse I_t divided by the initial or propellant-loaded vehicle weight w_0. A high value indicates an efficient design. Under our assumptions of constant thrust and negligible start and stop transients, it can be expressed as

$$\frac{I_t}{w_0} = \frac{I_t}{(m_f + m_p)g_0} \tag{2–11}$$

$$= \frac{I_s}{m_f/m_p + 1} \tag{2–12}$$

The *thrust to weight* ratio F/w_0 expresses the acceleration (in multiples of the earth's surface acceleration of gravity) that the engine is capable of giving to its own loaded propulsion system mass. For constant thrust the maximum value of the thrust to weight ratio, or maximum acceleration, occurs just before termination or burnout because the vehicle mass has been diminished by the mass of useful propellant. Values of F/w are given in Table 2–1. The *thrust to weight ratio* is useful to compare different types of rocket systems.

Example 2–1. A rocket projectile has the following characteristics:

Initial mass	200 kg
Mass after rocket operation	130 kg
Payload, nonpropulsive structure, etc.	110 kg
Rocket operating duration	3.0 sec
Average specific impulse of propellant	240 sec

Determine the vehicle's mass ratio, propellant mass fraction, propellant flow rate, thrust, thrust-to-weight ratio, acceleration of vehicle, effective exhaust velocity, total impulse, and the impulse-to-weight ratio.

SOLUTION. Mass ratio of vehicle (Eq. 2–8) $\mathbf{MR} = m_f/m_0 = 130/200 = 0.65$; mass ratio of rocket system $\mathbf{MR} = m_f/m_0 = (130 - 110)/(200 - 110) = 0.222$. Note that the empty and initial masses of the propulsion system are 20 and 90 kg, respectively.

The propellant mass fraction (Eq. 2–9) is

$$\zeta = (m_0 - m_f)/m_0 = (90 - 20)/90 = 0.778$$

The propellant mass is $200 - 130 = 70$ kg. The propellant mass flow rate is $\dot{m} = 70/3 = 23.3$ kg/sec,

The thrust (Eq. 2–5) is

$$F = I_s\dot{w} = 240 \times 23.3 \times 9.81 = 54{,}857 \text{ N}$$

The thrust-to-weight ratio of the vehicle is

$$\begin{aligned}&\text{initial value } F/w_0 = 54{,}857/(200 \times 9.81) = 28\\&\text{final value } 54{,}857/(130 \times 9.81) = 43\end{aligned}$$

The maximum acceleration of the vehicle is $43 \times 9.81 = 421$ m/sec^2. The effective exhaust velocity (Eq. 2–6) is

$$c = I_s g_0 = 240 \times 9.81 = 2354 \text{ m/sec}$$

The total impulse (Eqs. 2–2 and 2–5) is

$$I_t = I_s w = 240 \times 70 \times 9.81 = 164{,}808 \text{ N-sec}$$

This result can also be obtained by multiplying the thrust by the duration. The impulse-to-weight ratio of the propulsion system (Eq. 2–11) is

$$I_t/w_0 = 164{,}808/[(200 - 110)9.81] = 187$$

2.2. THRUST

The thrust is the force produced by a rocket propulsion system acting upon a vehicle. In a simplified way, it is the reaction experienced by its structure due to the ejection of matter at high velocity. It represents the same phenomenon that pushes a garden hose backwards or makes a gun recoil. In the latter case, the forward momentum of the bullet and the powder charge is equal to the recoil or rearward momentum of the gun barrel. Momentum is a vector quantity and is defined as the product of mass times velocity. All ship propellers and oars generate their forward push at the expense of the momentum of the water or air masses, which are accelerated towards the rear. Rocket propulsion differs from these devices primarily in the relative magnitude of the accelerated masses and velocities. In rocket propulsion relatively small masses are involved which are *carried within* the vehicle and ejected at high velocities.

The thrust, due to a change in momentum, is given below. A derivation can be found in earlier editions of this book. The thrust and the mass flow are constant and the gas exit velocity is uniform and axial.

$$F = \frac{dm}{dt} v_2 = \dot{m} v_2 = \frac{\dot{w}}{g_0} v_2 \tag{2–13}$$

This force represents the total propulsion force when the nozzle exit pressure equals the ambient pressure.

The pressure of the surrounding fluid (i..e, the local atmosphere) gives rise to the second contribution that influences the thrust. Figure 2–1 shows schematically the external pressure acting uniformly on the outer surface of a rocket chamber and the gas pressures on the inside of a typical thermal rocket engine. The size of the arrows indicates the relative magnitude of the pressure forces. The axial thrust can be determined by integrating all the pressures acting on areas that can be projected on a plane normal to the nozzle axis. The forces acting radially outward are appreciable, but do not contribute to the axial thrust because a rocket is typically an axially symmetric chamber. The conditions prior to entering the nozzle are essentially stagnation conditions.

Because of a fixed nozzle geometry and changes in ambient pressure due to variations in altitude, there can be an imbalance of the external environment or atmospheric pressure p_3 and the local pressure p_2 of the hot gas jet at the exit plane of the nozzle. Thus, for a steadily operating rocket propulsion system moving through a homogeneous atmosphere, the total thrust is equal to

$$F = \dot{m} v_2 + (p_2 - p_3) A_2 \tag{2–14}$$

The first term is the *momentum thrust* represented by the product of the propellant mass flow rate and its exhaust velocity relative to the vehicle. The second term represents the *pressure thrust* consisting of the product of the cross-sectional area at the nozzle exit A_2 (where the exhaust jet leaves the

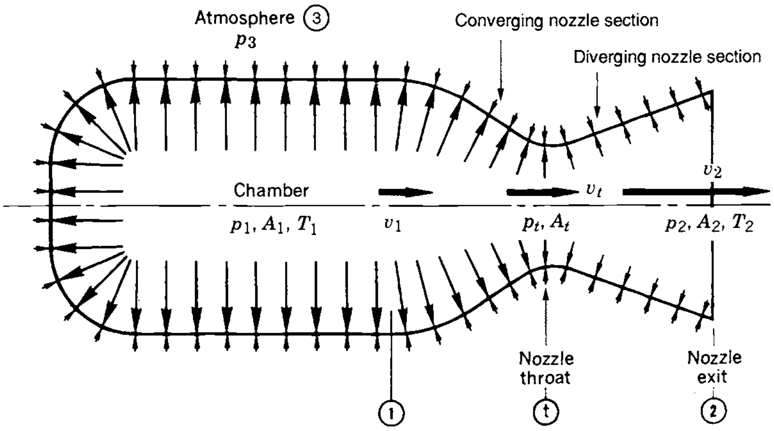


FIGURE 2–1. Pressure balance on chamber and nozzle interior walls is not uniform. The internal gas pressure (indicated by length of arrows) is highest in the chamber (p_1) and decreases steadily in the nozzle until it reaches the nozzle exit pressure p_2. The external or atmospheric pressure p_3 is uniform. At the throat the pressure is p_t. The four subscripts (shown inside circles) refer to the quantities A, v, T, and p at specific locations.

vehicle) and the difference between the exhaust gas pressure at the exit and the ambient fluid pressure. If the exhaust pressure is less than the surrounding fluid pressure, the pressure thrust is negative. Because this condition gives a low thrust and is undesirable, the rocket nozzle is usually so designed that the exhaust pressure is equal or slightly higher than the ambient fluid pressure.

When the ambient atmosphere pressure is equal to the exhaust pressure, the pressure term is zero and the thrust is the same as in Eq. 2–13. In the vacuum of space $p_3 = 0$ and the thrust becomes

$$F = \dot{m}v_2 + p_2A_2 \tag{2–15}$$

The pressure condition in which the exhaust pressure is exactly matched to the surrounding fluid pressure ($p_2 = p_3$) is referred to as the rocket nozzle with *optimum expansion ratio*. This is further elaborated upon in Chapter 3.

Equation 2–14 shows that the thrust of a rocket unit is independent of the flight velocity. Because changes in ambient pressure affect the pressure thrust, there is a variation of the rocket thrust with altitude. Because atmospheric pressure decreases with increasing altitude, the thrust and the specific impulse will increase as the vehicle is propelled to higher altitudes. This change in pressure thrust due to altitude changes can amount to between 10 and 30% of the overall thrust, as is shown for a typical rocket engine in Fig. 2–2. Table 8–1 shows the sea level and high altitude thrust for several rocket engines. Appendix 2 gives the properties of the Standard Atmosphere (ambient pressure).

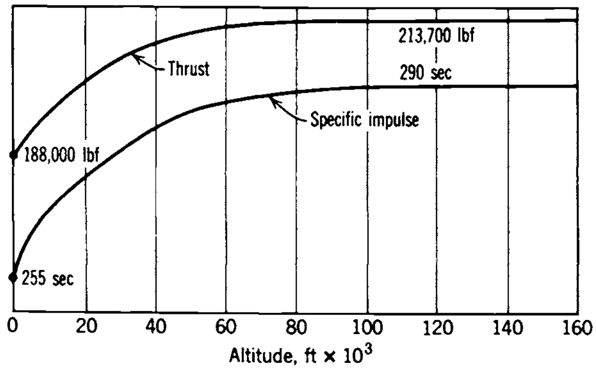


FIGURE 2–2. Altitude performance of RS 27 liquid propellant rocket engine used in early versions of the Delta launch vehicle.

2.3. EXHAUST VELOCITY

The *effective exhaust velocity* as defined by Eq. 2–6 applies to all rockets that thermodynamically expand hot gas in a nozzle and, indeed, to all mass expulsion systems. From Eq. 2–14 and for constant propellant mass flow this can be modified to

$$c = v_2 + (p_2 - p_3)A_2/\dot{m} \tag{2–16}$$

Equation 2–6 shows that c can be determined from thrust and propellant flow measurements. When $p_2 = p_3$, the effective exhaust velocity c is equal to the average actual exhaust velocity of the propellant gases v_2. When $p_2 \neq p_3$ then $c \neq v_2$. The second term of the right-hand side of Eq. 2–16 is usually small in relation to v_2; thus the effective exhaust velocity is usually close in value to the actual exhaust velocity. When $c = v_2$ the thrust (from Eq. 2–14) can be rewritten as

$$F = (\dot{w}/g_0)v_2 = \dot{m}c \tag{2–17}$$

The *characteristic velocity* has been used frequently in the rocket propulsion literature. Its symbol c^*, pronounced "cee-star," is defined as

$$c^* = p_1 A_t/\dot{m} \tag{2–18}$$

The characteristic velocity c^* is used in comparing the relative performance of different chemical rocket propulsion system designs and propellants; it is easily determined from measured data of $\dot{m}$, p_1, and A_t. It relates to the efficiency of the combustion and is essentially independent of nozzle characteristics.

However, the specific impulse I_s and the effective exhaust velocity c are functions of the nozzle geometry, such as the nozzle area ratio A_2/A_t, as shown in Chapter 3. Some values of I_s and c^* are given in Tables 5–4 and 5–5.

Example 2–2. The following measurements were made in a sea level test of a solid propellant rocket motor:

Burn duration	40 sec
Initial mass before test	1210 kg
Mass of rocket motor after test	215 kg
Average thrust	62,250 N
Chamber pressure	7.00 MPa
Nozzle exit pressure	0.070 MPa
Nozzle throat diameter	0.0855 m
Nozzle exit diameter	0.2703 m

Determine $\dot{m}$, v_2, c^*, c, and I_s at sea level, and c and I_s at 1000 and 25,000 m altitude. Assume an invariant thrust and mass flow rate and negligible short start and stop transients.

SOLUTION. The mass flow rate $\dot{m}$ is determined from the total propellant used (initial motor mass – final motor mass) and the burn time.

$$\dot{m} = (1210 - 215)/40 = 24.9 \text{ kg/sec}$$

The nozzle areas at the throat and exit are

$$A_t = \pi D^2/4 = \pi \times 0.0855^2/4 = 0.00574 \text{ m}^2$$
$$A_2 = \pi D^2/4 = \pi \times 0.2703^2/4 = 0.0574 \text{ m}^2$$

Equation 2–14 is to be solved for v_2, the actual average exhaust velocity.

$$\begin{aligned} v_2 &= F/\dot{m} - (p_2 - p_3)A_2/\dot{m} \\ &= 62{,}250/24.9 - (0.070 - 0.1013)10^6 \times 0.0574/24.9 \\ &= 2572 \text{ m/sec} \end{aligned}$$

The characteristic velocity and effective exhaust velocity are found from Eqs. 2–6 and 2–18 for sea level conditions.

$$\begin{aligned} c^* &= p_1 A_t/\dot{m} = 7.00 \times 10^6 \times 0.00574/24.9 = 1613 \text{ m/sec} \\ I_s &= F/\dot{m}g_0 = 62{,}250/(24.9 \times 9.81) = 255 \text{ sec} \\ c &= I_s g_0 = 255 \times 9.81 = 2500 \text{ m/sec} \end{aligned}$$

For altitudes of 1000 and 25,000 m the ambient pressure (see Appendix 2) is 0.0898 and 0.00255 MPa. From Eq. 2–16 the altitude values of c can be obtained.

$$c = v_2 + (p_2 - p_3)A_2/\dot{m}$$

At 1000 m altitude,

$$c = 2572 + (0.070 - 0.0898) \times 10^6 \times 0.0574/24.9 = 2527 \text{ m/sec}$$
$$I_s = 2527/9.81 = 258 \text{ sec}$$

At 25,000 m altitude,

$$c = 2572 + (0.070 - 0.00255) \times 10^6 \times 0.0574/24.9 = 2727 \text{ m/sec}$$
$$I_s = 2727/9.80 = 278 \text{ sec}$$

2.4. ENERGY AND EFFICIENCIES

Although efficiencies are not commonly used directly in designing rocket units, they permit an understanding of the energy balance of a rocket system. Their definitions are arbitrary, depending on the losses considered, and any consistent set of efficiencies, such as the one presented in this section, is satisfactory in evaluating energy losses. As stated previously, two types of energy conversion processes occur in any propulsion system, namely, the generation of energy, which is really the conversion of stored energy into available energy and, subsequently, the conversion to the form in which a reaction thrust can be obtained. The kinetic energy of ejected matter is the form of energy useful for propulsion. The *power of the jet* P_{jet} is the time rate of expenditure of this energy, and for a constant gas ejection velocity v this is a function of I_s and F

$$P_{\text{jet}} = \tfrac{1}{2}\dot{m}v^2 = \tfrac{1}{2}\dot{w}g_0 I_s^2 = \tfrac{1}{2}F g_0 I_s = \tfrac{1}{2}F v_2 \tag{2–19}$$

The term *specific power* is sometimes used as a measure of the utilization of the mass of the propulsion system including its power source; it is the jet power divided by the loaded propulsion system mass, P_{jet}/m_0. For electrical propulsion systems which carry a heavy, relatively inefficient energy source, the specific power can be much lower than that of chemical rockets. The energy input from the energy source to the rocket propulsion system has different forms in different rocket types. For chemical rockets the energy is created by combustion. The maximum energy available per unit mass of chemical propellants is the heat of the combustion reaction Q_R; the *power input to a chemical engine* is

$$P_{\text{chem}} = \dot{m} Q_R J \tag{2–20}$$

where J is a conversion constant which depends on the units used. A large portion of the energy of the exhaust gases is unavailable for conversion into kinetic energy and leaves the nozzle as residual enthalpy. This is analogous to the energy lost in the high-temperature exhaust gases of internal combustion engines.

The *combustion efficiency* for chemical rockets is the ratio of the actual and the ideal heat of reaction per unit of propellant and is a measure of the source efficiency for creating energy. Its value is high (approximately 94 to 99%), and it is defined in Chapter 5. When the power input P_{chem} is multiplied by the combustion efficiency, it becomes the power available to the propulsive device, where it is converted into the kinetic power of the exhaust jet. In electric propulsion the analogous efficiency is the power conversion efficiency. For solar cells it has a low value; it is the efficiency for converting solar radiation energy into electric power (10 to 20%).

The *power transmitted to the vehicle* at any one time is defined in terms of the thrust of the propulsion system F and the vehicle velocity u:

$$P_{\text{vehicle}} = Fu \tag{2–21}$$

The *internal efficiency* of a rocket propulsion system is an indication of the effectiveness of converting the system's energy input to the propulsion device into the kinetic energy of the ejected matter; for example, for a chemical unit it is the ratio of the kinetic power of the ejected gases expressed by Eq. 2–19 divided by the power input of the chemical reaction as given in Eq. 2–20. Internal efficiencies are used in Example 2–3. The energy balance diagram for a chemical rocket (Fig. 2–3) shows typical losses. The internal efficiency can be expressed as

$$\eta_{\text{int}} = \frac{\text{kinetic power in jet}}{\text{available chemical power}} = \frac{\frac{1}{2}\dot{m}v^2}{\eta_{\text{comb}} P_{\text{chem}}} \tag{2–22}$$

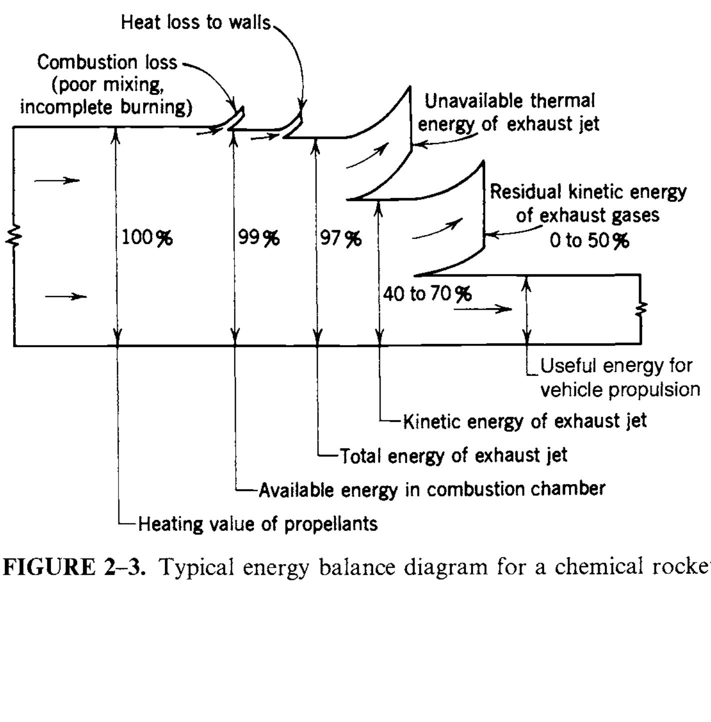


FIGURE 2–3. Typical energy balance diagram for a chemical rocket.

Typical values of η_{int} are listed later in Example 2–3.

The *propulsive efficiency* (Fig. 2–4) determines how much of the kinetic energy of the exhaust jet is useful for propelling a vehicle. It is also used often with duct jet engines and is defined as

$$\begin{aligned}\eta_P &= \frac{\text{vehicle power}}{\text{vehicle power + residual kinetic jet power}} \\ &= \frac{Fu}{Fu + \frac{1}{2}(\dot{w}/g_0)(c-u)^2} = \frac{2u/c}{1+(u/c)^2}\end{aligned} \tag{2–23}$$

where F is the thrust, u the absolute vehicle velocity, c the effective rocket exhaust velocity with respect to the vehicle, $\dot{w}$ the propellant weight flow rate, and η_p the propulsive efficiency. The propulsive efficiency is a maximum when the forward vehicle velocity is exactly equal to the exhaust velocity. Then the residual kinetic energy and the absolute velocity of the jet are zero and the exhaust gases stand still in space.

While it is desirable to use energy economically and thus have high efficiencies, there is also the problem of minimizing the expenditure of ejected mass, which in many cases is more important than minimizing the energy. In nuclear reactor energy and some solar energy sources, for example, there is an almost unlimited amount of heat energy available; yet the vehicle can only carry a limited amount of working fluid. Economy of mass expenditures of working fluid can be obtained if the exhaust velocity is high. Because the specific impulse is proportional to the exhaust velocity, it is a measure of this propellant mass economy.

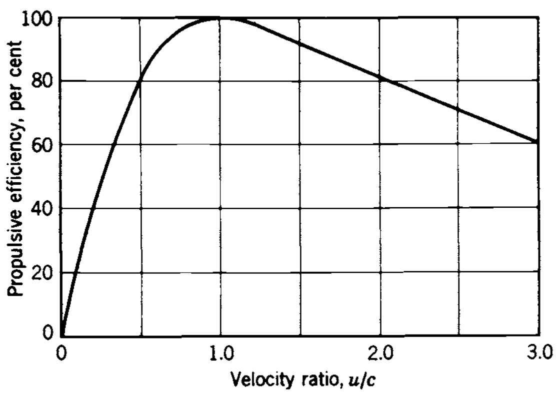


FIGURE 2–4. Propulsive efficiency at varying velocities.

2.5. TYPICAL PERFORMANCE VALUES

Typical values of representative performance parameters for different types of rocket propulsion are given in Table 2–1 and in Fig. 2–5.

Chemical rockets have relatively low values of specific impulse, relatively light machinery (i.e., low engine weight), a very high thrust capability, and therefore high acceleration and high specific power. At the other extreme, the ion propulsion devices have a very high specific impulse, but they must carry a heavy electrical power source with them to deliver the power necessary for high ejection velocities. The very low acceleration potential for the electrical propulsion units and those using solar radiation energy usually requires a long period for accelerating and thus these systems are best used for missions where the flight time is long. The low thrust values of electrical systems imply that they are not useful in fields of strong gravitational gradients (for takeoff or landing) but are best used in a true space flight mission.

The chemical systems (solid and liquid propellant rockets) are fully developed and widely used for many different vehicle applications. They are described in Chapters 5 to 15. Electrical propulsion has been in operation in many space flight applications (see Chapter 19). Some of the other types are still in their exploratory or development phase, but may become useful.

Example 2–3. As a comparison of different propulsion systems, compute the energy input and the propellant flow required for 100 N thrust with several types of propulsion systems.

SOLUTION. From Equations 2–13 and 2–19,

$$\dot{m} = F/(I_s g_0)$$
$$\text{power input} = P_{\text{jet}}/\eta_{\text{int}} = \tfrac{1}{2}\dot{m}v_2^2/\eta_{\text{int}}$$

From Table 2–1 typical values of I_s and from experience typical internal efficiencies were selected. Depending on the propellant and the design, these values may vary somewhat. The equations above were solved for $\dot{m}$ and the power input as indicated in the table below.

Engine Type	η_{int}	I_s	v_2 (m/sec)	$\dot{m}$ (kg/sec)	Power Input (kW)
Chemical rocket	0.50	300	2940	0.0340	294
Nuclear fission	0.50	800	7840	0.0128	787
Arc—electrothermal	0.50	600	5880	0.0170	588
Ion electrostatic	0.90	2000	19,600	0.0051	1959

More than half a megawatt of power is needed for the last three propulsion systems, but the propellant flows are small. The data for the last two types are illustrative, but hypothetical. To date the largest experimental units have been about 120 kW for arcjets and perhaps 10 kW with ion propulsion. Although thruster designs for megawatt-level units are feasible, it is unlikely that the needed flight-qualified electrical power generator would be available in the next decade.

TABLE 2–1. Ranges of Typical Performance Parameters for Various Rocket Propulsion Systems

Engine Type	Specific Impulse[a] (sec)	Maximum Temperature (°C)	Thrust-to-Weight Ratio[b]	Propulsion Duration	Specific Power[c] (kW/kg)	Typical Working Fluid	Status of Technology
Chemical—solid or liquid bipropellant	200–410	2500–4100	10^{-2}–100	Seconds to a few minutes	10^{-1}–10^{3}	Liquid or solid propellants	Flight proven
Liquid monopropellant	180–223	600–800	10^{-1}–10^{-2}	Seconds to minutes	0.02–200	N_2H_4	Flight proven
Nuclear fission	500–860	2700	10^{-2}–30	Seconds to minutes	10^{-1}–10^{3}	H_2	Development was stopped
Resistojet	150–300	2900	10^{-2}–10^{-4}	Days	10^{-3}–10^{-1}	H_2, N_2H_4	Flight proven
Arc heating—electrothermal	280–1200	20,000	10^{-4}–10^{-2}	Days	10^{-3}–1	N_2H_4,H_2,NH_3	Flight proven
Electromagnetic including Pulsed Plasma (PP)	700–2500	—	10^{-6}–10^{-4}	Weeks	10^{-3}–1	H_2 Solid for PP	Flight proven
Hall effect	1000–1700	—	10^{-4}	Weeks	10^{-1}–5×10^{-1}	Xe	Flight proven
Ion—electrostatic	1200–5000	—	10^{-6}–10^{--4}	Months	10^{-3}–1	Xe	Several have flown
Solar heating	400–700	1300	10^{-3}–10^{-2}	Days	10^{-2}–1	H_2	In development

[a]At $p_1 = 1000$ psia and optimum gas expansion at sea level ($Ip_2 = p_3 = 14.7$ psia).
[b]Ratio of thrust force to full propulsion system sea level weight (with propellants, but without payload).
[c]Kinetic power per unit exhaust mass flow.

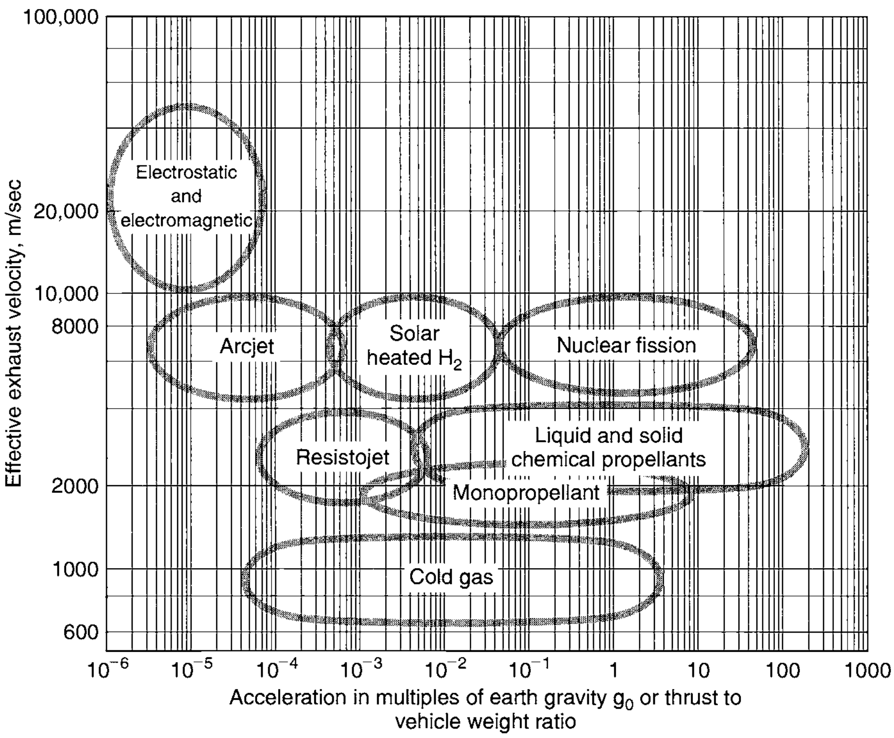


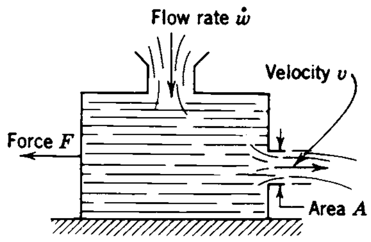
FIGURE 2–5. Exhaust velocities as a function of typical vehicle accelerations. Regions indicate approximate performance values for different types of propulsion systems. The mass of the vehicle includes the propulsion system, but the payload is assumed to be zero.

PROBLEMS

When solving problems, three appendixes (see end of book) may be helpful:

Appendix 1. Conversion Factors and Constants
Appendix 2. Properties of the Earth's Standard Atmosphere
Appendix 3. Summary of Key Equations

1. Prove that the value of the reaction thrust F equals twice the total dynamic pressure across the area A for an incompressible fluid as shown below.



2. The following data are given for a certain rocket unit: thrust, 8896 N; propellant consumption, 3.867 kg/sec; velocity of vehicle, 400 m/sec; energy content of propellant, 6.911 MJ/kg. Assume 100% combustion efficiency.

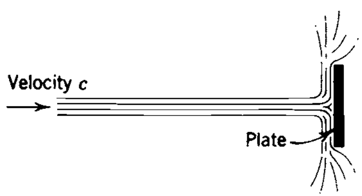
   Determine (*a*) the effective velocity; (*b*) the kinetic jet energy rate per unit flow of propellant; (c) the internal efficiency; (*d*) the propulsive efficiency; (*e*) the overall efficiency; (*f*) the specific impulse; (*g*) the specific propellant consumption.
   Answers: (*a*) 2300 m/sec; (*b*) 2.645 MJ-sec/kg; (*c*) 38.3%; (*d*) 33.7%; (*e*) 13.3%; (*f*) 234.7 sec; (*g*) 0.00426 sec^{-1}.

3. A certain rocket has an effective exhaust velocity of 7000 ft/sec; it consumes 280 lbm/sec of propellant mass, each of which liberates 2400 Btu/lbm. The unit operates for 65 sec. Construct a set of curves plotting the propulsive, internal, and overall efficiencies versus the velocity ratio u/c ($0 < u/c < 1.0$). The rated flight velocity equals 5000 ft/sec. Calculate (*a*) the specific impulse; (*b*) the total impulse; (*c*) the mass of propellants required; (*d*) the volume that the propellants occupy if their average specific gravity is 0.925.
   Answers: (*a*) 217.5 sec; (*b*) 3,960,000 lbf-sec; (*c*) 18,200 lbm; (*d*) 315 ft^3.

4. For the rocket in Problem 2, calculate the specific power, assuming a propulsion system dry mass of 80 kg and a duration of 3 min.

5. For the values given in Table 2–1 for the various propulsion systems, calculate the total impulse for a fixed propellant mass of 2000 kg.

6. A jet of fluid hits a stationary flat plate in the manner shown below.
   (a) If there is 50 kg of fluid flowing per minute at an abolute velocity of 200 m/sec, what will be the force on the plate?
   Answer: 167 N.
   (b) What will this force be when the plate moves in the direction of flow at $u = 50$ km/h?
   Answer: 144 N.



7. Plot the variation of the thrust and specific impulse against altitude, using the atmospheric pressure information given in Appendix 2, and the data for the Minuteman first-stage rocket thrust chamber in Table 11–3. Assume that $p_2 = 8.66$ psia.

8. Derive an equation relating the mass ratio **MR** and the propellant mass fraction.
   Answer: $\zeta = 1 - \mathbf{MR}$.

SYMBOLS (English engineering units are given in parentheses)

A	area, m^2 (ft^2)
A_t	nozzle throat area, m^2 (ft^2)
A_2	exist area of nozzle, m^2 (ft^2)
c	effective velocity, m/sec (ft/sec)
c^*	characteristic velocity, m/sec (ft/sec)
E	energy, J (ft-lbf)
F	thrust force, N (lbf)
g_0	standard sea level acceleration of gravity, 9.80665 m/sec^2 (32.174 ft/sec^2)
I_s	specific impulse, sec
I_t	impulse or total impulse, N-sec (lbf-sec)
J	conversion factor or mechanical equivalent of heat, 4.184 J/cal or 1055 J/Btu or 778 ft-lbf/Btu.
m	mass, kg (slugs) (1 slug = mass of 32.174 lb of weight at sea level)
$\dot{m}$	mass flow rate, kg/sec (lbm/sec)
m_f	final mass (after rocket propellant is ejected), kg (lbm or slugs)
m_p	propellant mass, kg (lbm or slugs)
m_0	initial mass (before rocket propellant is ejected), kg (lbm or slugs)
MR	mass ratio (m_f/m_0)
p	pressure, pascal [Pa] or N/m^2 (lbf/ft^2)
p_3	ambient or atmospheric pressure, Pa (lbf/ft^2)
p_2	rocket gas pressure at nozzle exit, Pa (lbf/ft^2)
p_1	chamber pressure, Pa (lbf/ft^2)
P	power, J/sec (ft-lbf/sec)
P_s	specific power, J/sec-kg (ft-lbf/sec-lbm)
Q_R	heat of reaction per unit propellant, J/kg (Btu/lbm)
t	time, sec
u	vehicle velocity, m/sec (ft/sec)
v_2	gas velocity leaving the rocket, m/sec (ft/sec)
w	weight, N or kg-m/sec^2 (lbf)
$\dot{w}$	weight flow rate, N/sec (lbf/sec)
w_0	initial weight, N or kg-m/sec^2 (lbf)

Greek Letters

ζ	propellant mass fraction
η	efficiency
η_{comb}	combustion efficiency
η_{int}	internal efficiency
η_p	propulsive efficiency

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2–2. "Solid Propulsion Nomenclature Guide," *CPIA Publication 80*, Chemical Propulsion Information Agency, Johns Hopkins University, Laurel, MD., May 1965, 18 pages.