CHAPTER 14

SOLID ROCKET COMPONENTS AND MOTOR DESIGN

This is the last of four chapters on solid propellant rockets. We describe the key inert components of solid propellant rocket motors, namely the motor case, nozzle, and igniter case, and then discuss the design of motors. Although the thrust vector control mechanism is also a component of many rocket motors, it is described separately in Chapter 16. The key to the success of many of these components is new materials which have been developed in recent years.

14.1. MOTOR CASE

The case not only contains the propellant grain, but also serves as a highly loaded pressure vessel. Case design and fabrication technology has progressed to where efficient and reliable motor cases can be produced consistently for any solid rocket application. Most problems arise when established technology is used improperly or from improper design analysis, understating the requirements, or improper material and process control, including the omission of nondestructive tests at critical points in the fabrication process. Case design is usually governed by a combination of motor and vehicle requirements. Besides constituting the structural body of the rocket motor with its nozzle, propellant grain, and so on, the case frequently serves also as the primary structure of the missile or launch vehicle. Thus the optimization of a case design frequently entails trade-offs between case design parameters and vehicle design parameters. Often, case design is influenced by assembly and fabrication requirements.

Table 14–1 lists many of the types of loads and their sources; they must be considered at the beginning of a case design. Only some of them apply to any one rocket motor application. In addition, the environmental conditions peculiar to a specific motor and its usage must be carefully considered. Typically, these conditions include the following: (1) temperature (internal heating, aerodynamic heating, temperature cycling during storage, or thermal stresses and strains); (2) corrosion (moisture/chemical, galvanic, stress corrosion, or hydrogen embrittlement); (3) space conditions: vacuum or radiation.

Three classes of materials have been used: high-strength metals (such as steel, aluminum, or titanium alloys), wound-filament reinforced plastics, and a combination of these in which a metal case has externally wound filaments for extra strength. Table 14–2 gives a comparison of several typical materials. For filament-reinforced materials it gives the data not only for the composite material, but also for several strong filaments and a typical binder. The strength-to-density ratio is higher for composite materials, which means that they have less inert mass. Even though there are some important disadvantages, the filament-wound cases with a plastic binder are usually superior on a vehicle performance basis. Metal cases combined with an external filament-wound reinforcement and spiral-wound metal ribbons glued together with plastic have also been successful.

The shape of the case is usually determined from the grain configuration or from geometric vehicle constraints on length or diameter. The case configurations range from long and thin cylinders (L/D of 10) to spherical or near-

TABLE 14–1. Rocket Motor Case Loads

Origin of Load	Type of Load/Stress
Internal pressure	Tension biaxial, vibration
Axial thrust	Axial, vibration
Motor nozzle	Axial, bending, shear
Thrust vector control actuators	Axial, bending, shear
Thrust termination equipment	Biaxial, bending
Aerodynamic control surfaces or wings mounted to case	Tension, compression, bending, shear, torsion
Staging	Bending, shear
Flight maneuvering	Axial, bending, shear, torsion
Vehicle mass and wind forces on launch pad	Axial, bending, shear
Dynamic loads from vehicle oscillations	Axial, bending, shear
Start pressure surge	Biaxial
Ground handling, including lifting	Tension, compression, bending, shear, torsion
Ground transport	Tension, compression, shear, vibration
Earthquakes (large motors)	Axial, bending, shear

TABLE 14–2. Physical Properties of Selected Solid Propellant Motor Case Materials at 20°C

Material	Tensile Strength, N/mm^2 (10^3 psi)	Modulus of Elasticity, N/mm^2 (10^6 psi)	Density, g/cm^3 (lbm/in.^{3})	Strength to Density Ratio (1000)
		Filaments		
E-glass	1930–3100 (280–450)	72,000 (10.4)	2.5 (0.090)	1040
Aramid (Kevlar 49)	3050–3760 (370–540)	124,000 (18.0)	1.44 (0.052)	2300
Carbon fiber or graphite fibers	3500–6900 (500–1000)	230,000–300,000 (33–43)	1.53–1.80 (0.055–0.065)	2800
		Binder (by itself)		
Epoxy	83 (12)	2800 (0.4)	1.19 (0.043)	70
		Filament-Reinforced Composite Material		
E Glass	1030 (150–170)	35,000 (4.6–5.0)	1.94 (0.070)	500
Kevlar 49	1310 (190)	58,000 (8.4)	1.38 (0.050)	950
Graphite IM	2300 (250–340)	102,000 (14.8)	1.55 (0.056)	1400
		Metals		
Titanium alloy	1240 (180)	110,000 (16)	4.60 (0.166)	270
Alloy steel (heat treated)	1400–2000 (200–290)	207,000 (30)	7.84 (0.289)	205
Aluminum alloy 2024 (heat treated)	455 (66)	72,000 (10.4)	2.79 (0.101)	165

Source: Data adapted in part from Chapter 4A by Evans and Chapter 7 by Scippa of Ref. 11–1.

spherical geometries (see Figs. 1–5, 11–1 to 11–4, and 11–17). The spherical shape gives the lowest case mass per unit of enclosed volume. The case is often a key structural element of the vehicle and it sometimes has to provide for mounting of other components, such as fins, skirts, electric conduits, or thrust vector control actuators. The propellant mass fractions of the motor are usually strongly influenced by the case mass and typically range from 0.70 to 0.94. The higher values apply to upper stage motors. For small-diameter motors the mass fraction is lower, because of practical wall thicknesses and the fact that the wall surface area (which varies roughly as the square of the diameter) to chamber volume (which varies roughly as the cube of diameter) is less favorable in small sizes. The minimum thickness is higher than would be determined from simple stress analysis; for a a fiber composite case it is two layers of filament strands and the minimum metal thickness is dictated by manufacturing and handling considerations.

Simple membrane theory can be used to predict the approximate stress in solid propellant rocket chamber cases; this assumes no bending in the case walls and that all the loads are taken in tension. For a simple cylinder of radius R and thickness d, with a chamber pressure p, the longitudinal stress σ_l is one-half of the tangential or hoop stress σ_θ:

$$\sigma_\theta = 2\sigma_l = pR/d \tag{14–1}$$

For a cylindrical case with hemispherical ends, the cylinder wall has to be twice as thick as the walls of the end closures.

The combined stress should not exceed the working stress of the wall material. As the rocket engine begins to operate, the internal pressure p causes a growth of the chamber in the longitudinal as well as in the circumferential direction, and these deformations must be considered in designing the support of the motor or propellant grain. Let E be Young's modulus of elasticity, ν be Poisson's ratio (0.3 for steel), and d be the wall thickness; then the growth in length L and in diameter D due to pressure can be expressed as

$$\Delta L = \frac{pLD}{4Ed}(1 - 2\nu) = \frac{\sigma_l L}{E}(1 - 2\nu) \tag{14–2}$$

$$\Delta D = \frac{pD^2}{4Ed}\left(1 - \frac{\nu}{2}\right) = \frac{\sigma_\theta D}{2E}\left(1 - \frac{\nu}{2}\right) \tag{14–3}$$

Details can be found in a text on thin shells or membranes. For a hemispherical chamber end, the stress in each of two directions at right angles to each other is equal to the longitudinal stress of a cylinder of identical radius. For ellipsoidal end-chamber closures, the local stress varies with the position along the surface, and the maximum stress is larger than that of a hemisphere. The radial displacement of a cylinder end is not the same as that of a hemispherical or ellipsoidal closure if computed by thin-shell theory. Thus a discontinuity exists which causes some shearing and bending stresses. Similarly, a boss for the attachment of an igniter, a pressure gauge, or a nozzle can make it necessary for bending and shear stresses to be superimposed on the simple tension stresses of the case. In these locations it is necessary to reinforce or thicken the chamber wall locally.

Finite element computerized stress analysis programs exist and are used in motor design companies today to determine the case design configuration with reasonable stress values. This analysis must be done simultaneously with the stress analysis on the grain (since it imposes loads on the case), and with a finite element thermal analysis to determine thermal stresses and deformations, since these analyses are interdependent on each other.

The fast heating of the inner wall surface produces a temperature gradient and therefore thermal stresses across the wall. The theory of transient heat transfer has been treated by a number of authors, and, by means of a relaxation method, a reasonable approximation of the temperature–time history at any location may be obtained. The inner wall of the case, which is exposed to hot gas, is usually protected by thermal insulation, as described in Section 12.6. Therefore the heat transfer to the case is very low. In fact, for a single operation (not two thrust periods) it is the designer's aim to keep the case temperatures near ambient or at the most 100°C above ambient.

The case design has to provide means for attaching a nozzle (rarely more than one nozzle), for attaching it to the vehicle, igniters, and provisions for loading the grain. Sometimes there are also attached aerodynamic surfaces (fins), sensing instruments, a raceway (external conduit for electrical wires), handling hooks, and thrust vector control actuators with their power supply. For upper stages of ballistic missiles the case can also include blow-out ports or thrust termination devices, as described in Chapter 13. Typical methods for attaching these items include tapered or straight multiple pins, snap rings, or bolts. Gaskets and/or O-ring seals prevent gas leaks.

Metal Cases

Metal cases have several advantages compared to filament-reinforced plastic cases: they are rugged and will take considerable rough handling (required in many tactical missile applications), are usually reasonably ductile and can yield before failure, can be heated to a relatively high temperature (700 to 1000°C or 1292 to 1832°F and higher with some special materials), and thus require less insulation. They will not deteriorate significantly with time or weather exposure and are easily adapted to take concentrated loads, if made thicker at a flange or boss. Since the metal case has much higher density and less insulation, it occupies less volume than does a fiber-reinforced plastic case; therefore, for the same external envelope it can contain somewhat more propellant.

Figure 14–1 shows the various sections of a typical large solid rocket case made of welded steel. The shape of the case, particularly the length-to-diameter ratio for cylindrical cases, influences not only the stresses to be withstood by the case but the amount of case material required to encase a given amount of propellant. For very large and long motors both the propellant grain and the motor case are made in sections; the *case segments* are mechanically attached and sealed to each other at the launch site. The segmented solid rocket booster for the Space Shuttle is shown in Fig. 14–2 and discussed in Ref. 14–1. For the critical seal between the segments a multiple-O-ring joint is often used, as shown in Fig. 14–3 and discussed in Ref. 14–2. Segments are used when an unsegmented motor would be too large and too heavy to be transported over

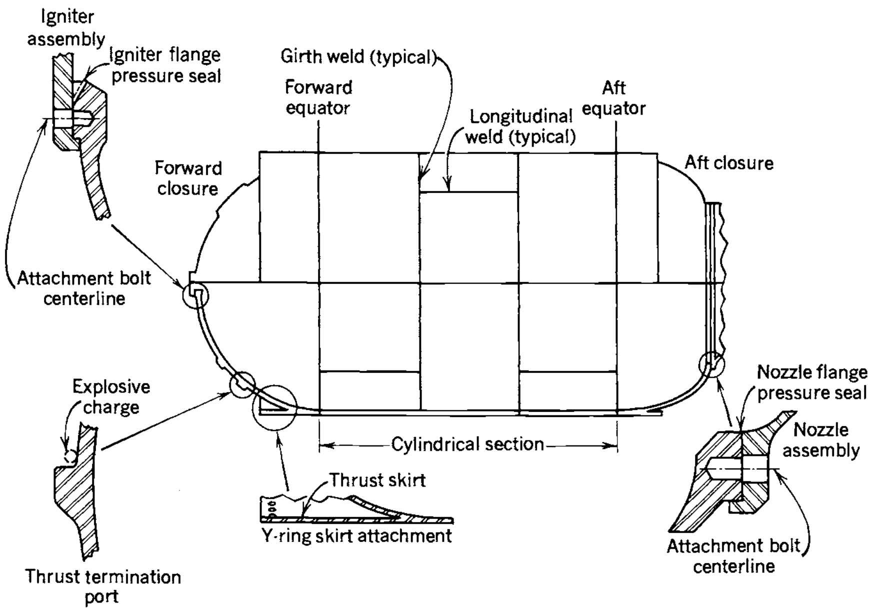


FIGURE 14–1. Typical large solid rocket motor case made of welded alloy steel.

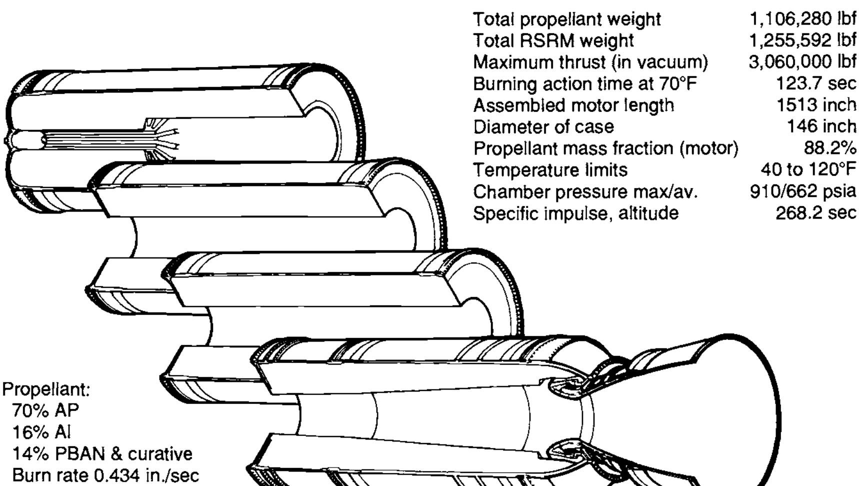


FIGURE 14–2. Simplified diagram of the four segments of the Space Shuttle solid rocket motor. Details of the thrust vector actuating mechanism or the ignition system are not shown. (Courtesy of NASA and Thiokol Propulsion, a Division of Cordant Technologies, Inc.)

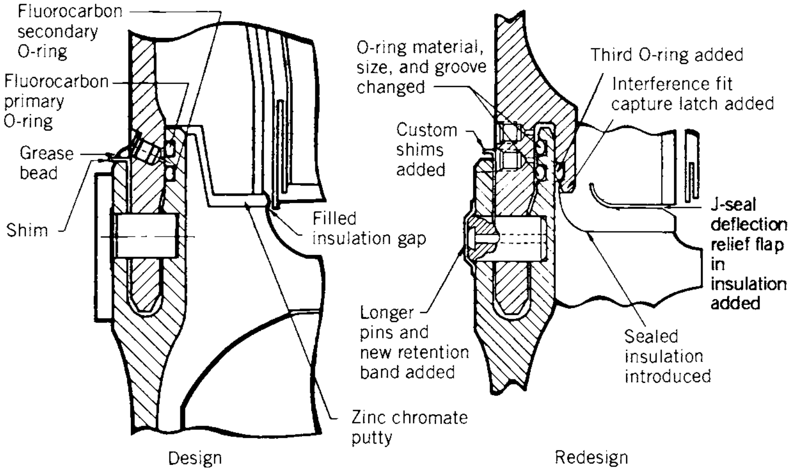


FIGURE 14–3. The joints between segments of the Shuttle solid rocket booster (SRB) were redesigned after a dramatic failure. The improvements were not only in a third O-ring, the mechanical joint, and its locking mechanism, but also featured a redesign of the insulation between propellant segments. (Courtesy of NASA.)

ordinary roads (cannot make turns) or railways (will not go through some tunnels or under some bridges) and are often too difficult to fabricate.

Small metal *cases for tactical missile motors* can be extruded or forged (and subsequently machined), or made in three pieces as shown in Fig. 11–4. This case is designed for loading a free-standing grain and the case, nozzle, and blast tube are sealed by O-rings (see Chapter 6 of Ref. 14–3 and Chapter 7 of Ref. 14–4). Since the mission velocities for most tactical missiles are relatively low (100 to 1500 m/sec), their propellant mass fractions are also relatively low (0.5 to 0.8) and the percentage of inert motor mass is high. Safety factors for tactical missile cases are often higher to allow for rough handling and cumulative damage. The emphasis in selecting motor cases (and other hardware components) for tactical missiles is therefore not on highest performance (lowest inert motor mass), but on reliability, long life, low cost, ruggedness, or survivability.

High-strength alloy steels have been the most common case metals, but others, like aluminum, titanium, and nickel alloys, have also been used. Table 14–2 gives a comparison of motor case material properties. Extensive knowledge exists for designing and fabricating motor cases with low-alloy steels with strength levels to 240,000 psi.

The *maraging steels* have strengths up to approximately 300,000 psi in combination with high fracture toughness. The term *maraging* is derived from the fact that these alloys exist as relative soft low-carbon martensites in the annealed condition and gain high strength from aging at relatively low temperatures.

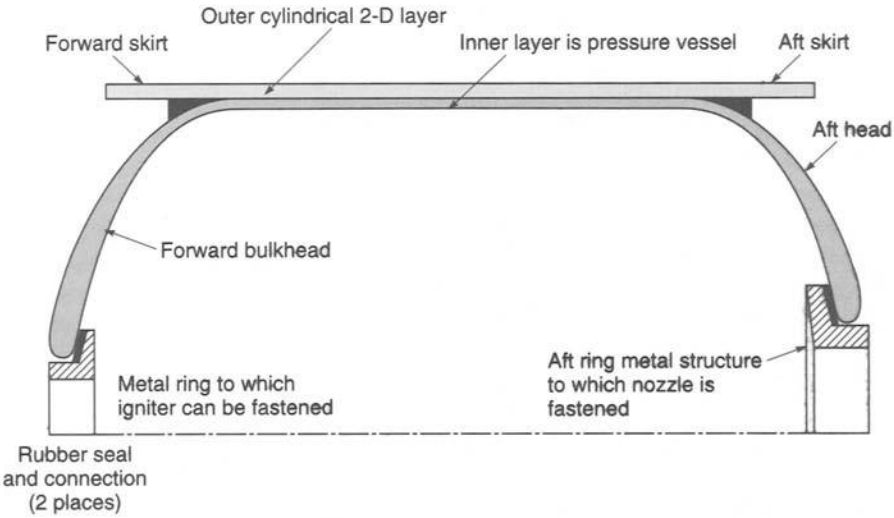


FIGURE 14–4. Simplified half-section of a typical design of a filament-wound composite material case. Elastomeric adhesives are shown in black. The outer layer reinforces the cylinder portion and provides attachment skirts. The thickness of the inner case increases at smaller diameter.

The *HY steels* (newer than the maraging steels) are attractive because of their toughness and resistance to tearing, a property important to motor cases and other pressure vessels because failures are less catastrophic. This toughness characteristic enables a "leak before failure" to occur, at least during hydrostatic proof testing. The HY steels have strengths between 180,000 and 300,000 psi (depending on heat treatment and additives).

Stress-corrosion cracking of certain metals presents a unique problem which can result in spontaneous failure without any visual evidence of impending catastrophe. Emphasis given to lightweight thin metal cases aggravates stress corrosion and crack propagation, often starting from a flaw in the metal, with failure occurring at a stress level below the yield strength of the metal.

Wound-Filament-Reinforced Plastic Cases

Filament-reinforced cases use continuous filaments of strong fibers wound in precise patterns and bonded together with a plastic, usually an epoxy resin. Their principal advantage is their lower weight. Most plastics soften when they are heated above about 180°C or 355°F; they need inserts or reinforcements to allow fastening or assembly of other components and to accept concentrated loads. The thermal expansion of reinforced plastics is often higher than that of metal and the thermal conductivity is much lower, causing a higher temperature gradient. References 14–3 and 14–4 explain the design and winding of these composite cases, and Ref. 14–5 discusses their damage tolerance.

Typical fiber materials are, in the order of increasing strength, glass, aramids (Kevlar), and carbon, as listed in Table 14–2. Typically, the inert mass of a case made of carbon fiber is about 50% of a case made with glass fibers and around 67% of a case mass made with Kevlar fibers.

Individual fibers are very strong in tension (2400 to 6800 MPa or 350,000 to 1,000,000 psi). The fibers are held in place by a plastic binder of relatively low density; it prevents fibers slipping and thus weakening in shear or bending. In a filament-wound composite (with tension, hoop, and bending stresses) the filaments are not always oriented along the direction of maximum stress and the material includes a low-strength plastic; therefore, the composite strength is reduced by a factor of 3 to 5 compared to the strength of the filament itself. The plastic binder is usually a thermosetting epoxy material, which limits the maximum temperature to between 100 to 180°C or about 212 to 355°F. Although resins with higher temperature limits are available (295°C or 563°F), their adhesion to the fibers has not been as strong. The safety factors used (in deterministic structural analysis) are typically for failure to occur at 1.4 to 1.6 times the maximum operating stress, and proof testing is done to 1.15 to 1.25 times the operating pressure.

A typical case design is shown schematically in Fig. 14–4. The forward end, aft end, and cylindrical portion are wound on a preform or mold which already contains the forward and aft rings. The direction in which the bands are laid onto the mold and the tension that is applied to the bands is critical in obtaining a good case. The curing is done in an oven and may be done under pressure to assure high density and minimum voids of the composite material. The preform is then removed. One way is to use sand with a water-soluble binder for the preform; after curing the case, the preform is washed out with water. Since filament-wound case walls can be porous, they must be sealed. The liner between the case and the grain can be the seal that prevents hot gases from seeping through the case walls. Scratches, dents, and moisture absorption can degrade the strength of the case.

In some designs the insulator is placed on the preform before winding and the case is cured simultaneously with the insulator, as seen in Ref. 14–6. In another design the propellant grain with its forward and aft closures is used as the preform. A liner is applied to this grain, then an insulator, and the high-strength fibers of the case are wound in layers directly over the insulated live propellant. Curing has to be done at a relatively low temperature so that the propellant will not be adversely affected. This process works well with extruded cylindrical grains. There are also cylindrical cases made of steel with an overwrap layer of filament-wound composite material, as described in Ref. 14–7.

The allowable stresses are usually determined from tensile tests of a roving or band and rupture tests on subscale composite cases made by an essentially identical filament winding process. Some companies reduce the allowable strength to account for the degradation due to moisture, manufacturing imperfections, or nonuniform density.

In a motor case the filaments must be oriented in the direction of principal stress and must be proportioned in number to the magnitude of stress. Compromise occurs around parts needed for nozzles, igniters, and so on, and then orientation is kept as close to the ideal as is practicable. Filaments are customarily clustered in *yarns*, *rovings*, or *bands*, as defined in Fig. 14–5. By using two or more winding angles (i.e., helicals and circumferentials) and calculating the proportion of filaments in each direction, a balanced stress structure is achieved. The ideal in balance is for each fiber in each direction to carry an equal load (tension only). Realistically, the filaments supported by the epoxy resin must absorb stress compression, bending loads, cross-laminar shear, and interlaminar shear. Even though the latter stresses are small compared to the tensile stress, each must be examined by analysis since each can lead to case failure before a filament fails in tension. In a proper design, failure occurs when the filaments reach their ultimate tensile strength, rather than because of stresses in other directions. Figure 16–5 shows a cross section of a Kevlar filament motor case and flexible nozzle made of ablative materials.

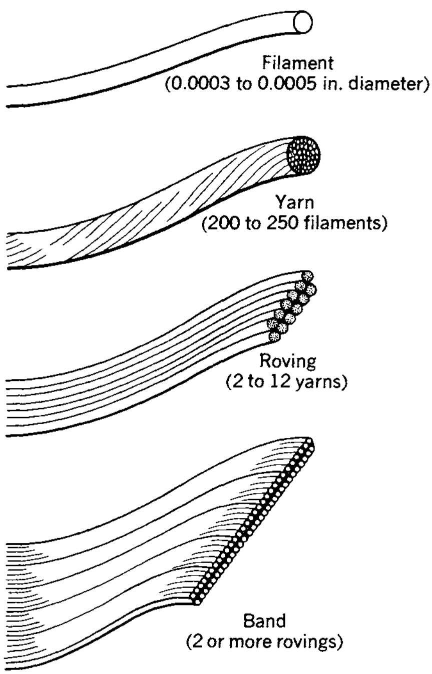


FIGURE 14–5. Filament winding terminology (each sketch is drawn to a different scale).

14.2. NOZZLES*

The supersonic nozzle provides for the expansion and acceleration of the hot gases and has to withstand the severe environment of high heat transfer and erosion. Advances in material technology have allowed substantial mass reductions and performance improvements. Nozzles range in size from 0.05 in. throat diameter to about 54 in., with operating durations of a fraction of a second to several minutes (see Chapters 2 and 3 of Ref. 14–3 and Chapter 6 in Ref. 14–4).

Classification

Nozzles for solid propellant rocket motors can be classified into five categories as listed below and shown in Fig. 14–6.

1. *Fixed Nozzle*. Simple and used frequently in tactical weapon propulsion systems for short-range air-, ground-, and sea-launched missiles, also as strap-on propulsion for space launch vehicles such as Atlas and Delta, and in spacecraft motors for orbital transfer. Typical throat diameters are between 0.25 and 5 in. for tactical missile nozzles and approximately 10 in. for strap-on motors. Fixed nozzles are generally not submerged (see below) and do not provide thrust vector control (although there are exceptions). See Fig. 14–7.
2. *Movable Nozzle*. Provides thrust vector control for the flight vehicle. As explained in Chapter 16, one movable nozzle can provide pitch and yaw control and two are needed for roll control. Movable nozzles are typically submerged and use a flexible sealed joint or bearing with two actuators 90 degrees apart to achieve omniaxial motion. Movable nozzles are primarily used in long-range strategic propulsion ground- and sea-launched systems (typical throat diameters are 7 to 15 in. for the first stage and 4 to 5 in. for the third stage) and in large space launch boosters such as the Space Shuttle reusable solid rocket motor, Titan boost rocket motor, and Ariane V solid rocket booster, with throat diameters in the 30 to 50 in. range.
3. *Submerged Nozzles*. A significant portion of the nozzle structure is submerged within the combustion chamber or case, as shown in Figs. 14–1 to 14–3. Submerging the nozzle reduces the overall motor length somewhat, which in turn reduces the vehicle length and its inert mass. It is important for length-limited applications such as silo- and submarine-launched strategic missiles as well as their upper stages, and space motor propulsion systems. Reference 14–8 describes the sloshing of trapped molten aluminum oxide that can accumulate in the groove around a submerged noz-

*This section was revised and rewritten by **Terry A. Boardman** of Thiokol Corporation, a Division of Cordant Technologies.

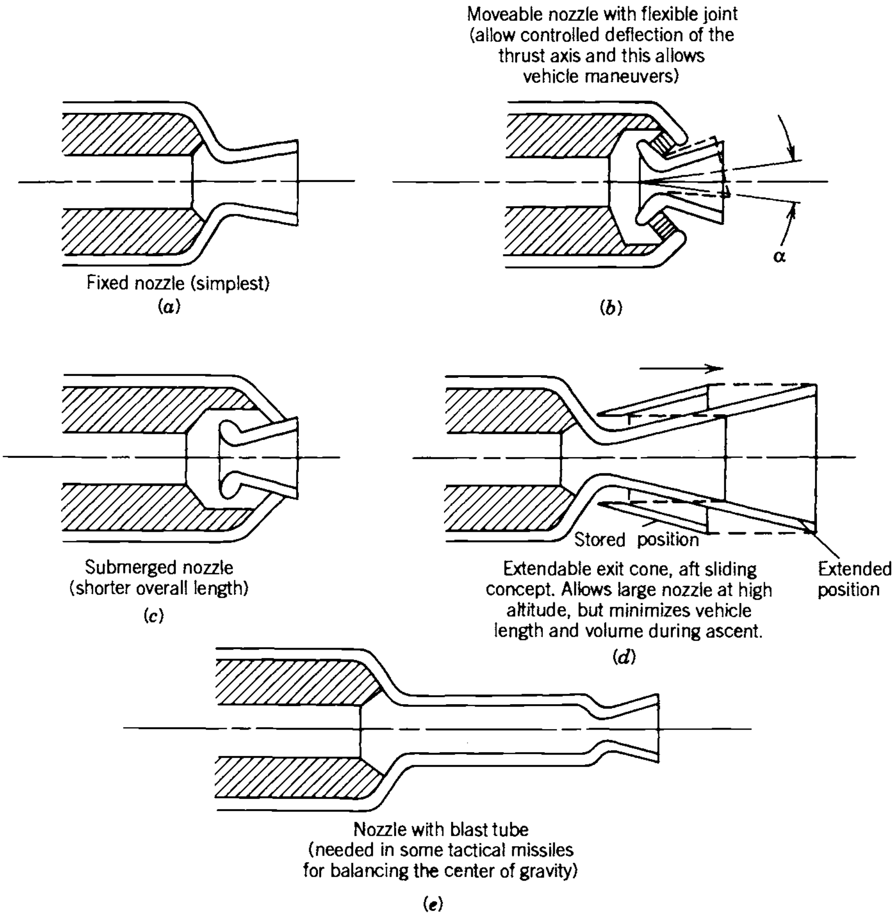


FIGURE 14–6. Simplified diagrams of five common nozzle configurations.

zle. This accumulation is undesirable, but can be minimized by good design.

4. *Extendible Nozzle.* Commonly referred to as an extendible exit cone, or EEC, although it is not always exactly conical. It is used on strategic missile propulsion upper-stage systems and upper stages for space launch vehicles to maximize motor-delivered specific impulse. As shown in Fig. 11–3, it has a fixed low-area-ratio nozzle section which is enlarged to a higher area ratio by mechanically adding a nozzle cone extension piece. The extended nozzle improves specific impulse by doubling or tripling the initial expansion ratio, thereby significantly increasing the nozzle thrust coefficient. This system thus allows a very high expansion ratio nozzle to be packaged in a relatively short length, thereby reducing vehicle inert mass. The nozzle cone extension is in its retracted position during the

boost phase of the flight and is moved into place before the motor is started but after separation from the lower stage. Typically, electromechanical or turbine-driven ball screw actuators deploy the exit cone extension.

5. *Blast-Tube-Mounted Nozzle*. Used with tactical air- and ground-launched missiles with diameter constraints to allow space for aerodynamic fin actuation or TVC power supply systems. The blast tube also allows the rocket motor's center of gravity (CG) to be close to or ahead of the vehicle CG. This limits the CG travel during motor burn and makes flight stabilization much easier.

Each motor usually has a single nozzle. A few larger motors have had four movable nozzles, which are used for thrust vector control.

Design and Construction

Almost all solid rocket nozzles are ablatively cooled. The general construction of a solid rocket nozzle features steel or aluminum shells (housings) that are designed to carry structural loads (motor operating pressure and nozzle TVC actuator load are the biggest), and composite ablative liners which are bonded to the housings. The ablative liners are designed to insulate the steel or aluminum housings, provide the internal aerodynamic contour necessary to efficiently expand combustion gases to generate thrust, and to ablate and char in a controlled and predictable manner to prevent the buildup of heat which could damage or substantially weaken the structural housings or the bonding materials. Solid rocket nozzles are designed to ensure that the thickness of ablative liners is sufficient to maintain the liner-to-housing adhesive bond line below the temperature that would degrade the adhesive structural properties during motor operation. Nozzle designs are shown in Figs. 1–5, 11–1 to 11–4, and 14–7.

The construction of nozzles ranges from simple single-piece non-movable graphite nozzles to complex multipiece nozzles capable of moving to control the direction of the thrust vector. The simpler, smaller nozzles are typically for applications with low chamber pressure, short durations (perhaps less than 10 sec), low area ratios, and/or low thrust. Typical small, simple built-up nozzles are shown in Fig. 14–7. Complex nozzles are usually necessary to meet more difficult design requirements such as providing thrust vector controls, operating at high chamber pressures (and thus at higher heat transfer rates) and/or higher altitudes (large nozzle expansion ratios), producing very high thrust levels, and surviving longer motor burn durations (above 30 sec).

Figures 14–8 and 14–9 illustrate the design features of the largest and one of the most complex solid rocket nozzles currently in production. This nozzle is used on the Reusable Solid Rocket Booster (RSRM) to provide 71.4% of the lift-off thrust of the Space Shuttle launch vehicle shown in Figs. 1–13 and 14–2. The nozzle is designed to provide large structural and thermal safety margins

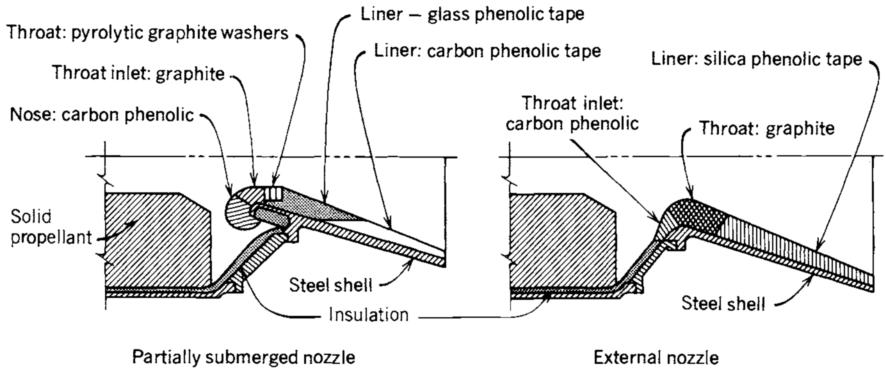
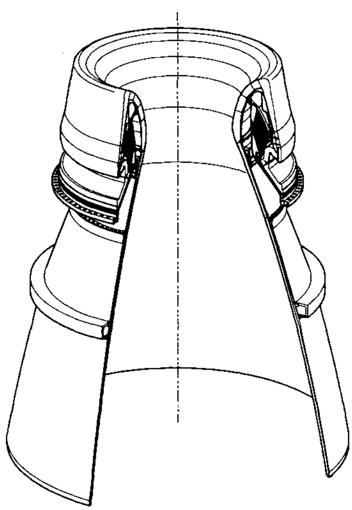


FIGURE 14–7. Nozzle designs for small solid propellant motors employ ablative heat sink wall pieces and graphite throat inserts resistant to high temperatures, erosion, and oxidation. The pyrolytic washers or disks are so oriented that their high conductivity direction is perpendicular to the nozzle axis.

during the shuttle booster's 123.7 sec burn time and consists of nine carbon cloth phenolic ablative liners bonded to six steel and aluminum housings. The housings are bolted together to form the structural foundation for the nozzle. A flexible bearing (described further in Chapter 16), made of rubber vulcanized to steel shims, enables the nozzle to vector omniaxially up to eight degrees from centerline to provide thrust vector control. Since the metal housings are recov-



RSRM Nozzle Characteristics

Type	Contoured or bell
Thrust vector control	Flexible bearing
Expansion area ratio	7.72
Throat diameter	53.86 in.
Exit diameter	149.64 in.
Total length	178.75 in.
Nozzle weight	23, 941 lbf
Maximum pressure	1.016 psi
Maximum thrust (vac.)	3, 070, 000 lbf
Burn time	123.7 sec
Materials	
Housings	Steel and aluminum
Liners	Carbon cloth phenolic

FIGURE 14–8. External quarter section view of nozzle configuration of the Space Shuttle reusable solid rocket motor (RSRM). (Courtesy of Thiokol Propulsion, a Division of Cordant Technologies.)

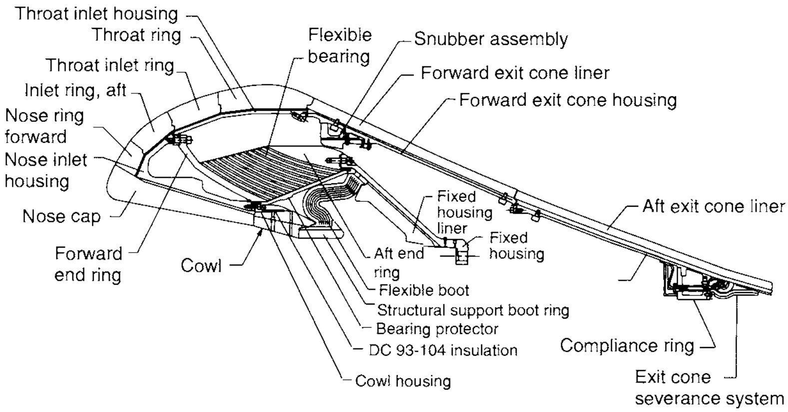


FIGURE 14–9. Section through movable nozzle shown in Fig. 14–8 with component identification. (Courtesy of Thiokol Propulsion, a Division of Cordant Technologies.)

ered and reused after flight, an exit cone severance system (a circumferential linear shaped charge) is used to cut off a major section of the aft exit cone just below the aft exit cone aluminum housing to minimize splashdown loading on the remaining components.

From a performance perspective, the primary nozzle design task is to efficiently expand gas flow from the motor combustion chamber to produce thrust. Simple nozzles with noncontoured conical exit cones can be designed using the basic thermodynamic relationships presented in Chapter 3 to determine throat area, nozzle half angle, and expansion ratio. A more complex contoured (bell-shaped) nozzle is used to reduce the divergence loss, improve the specific impulse slightly, and reduce nozzle length and mass. Section 3.4 gives data on designing bell-shaped nozzles with optimum wall contour (to avoid shock waves) and minimum impact of particulates in the exhaust gas.

Two-dimensional, two-phase, reacting gas method-of-characteristics flow codes are used to analyze the gas-particle flow in the nozzle and determine the optimal nozzle contour which maximizes specific impulse while yielding acceptable erosion characteristics. Such codes provide analytical solutions to all identified specific impulse loss mechanisms which result in less than ideal performance. An example is given in Table 14–3.

Figure 14–10 illustrates the amount of carbon cloth phenolic liner removed by chemical erosion and particle impingement, the liner char depth, and gas temperature and pressure at selected locations in the RSRM nozzle. Erosion on the nose cap (1.73 in.) is high primarily as a result of impingement by Al_2O_3 particles traveling down the motor bore. The impact of the particles mechanically removes the charred liner material. In contrast, the radial throat erosion of 1.07 in. results primarily from the carbon liner material reacting chemically

TABLE 14–3. Calculated Losses in the Space Shuttle Booster RSRM Nozzle

Theoretical specific impulse (vacuum conditions)	278.1 sec
Delivered specific impulse (vacuum conditions)	268.2 sec
Losses (calculated):	*(9.9 sec total)*
Two-dimensional two-phase flow (includes divergence loss)	7.4 sec
Throat erosion (reduces nozzle area ratio)	0.9 sec
Boundary layer (wall friction)	0.7 sec
Submergence (flow turning)	0.7 sec
Finite rate chemistry (chemical equilibrium)	0.2 sec
Impingement (of Al_2O_3 particles on nozzle wall)	0.0 sec
Shock (if turnback angle is too high or nozzle length too low)	0.0 sec
Combustion efficiency (incomplete burning)	0.0 sec

with oxidizing species in the combustion gas flow at the region of greatest heat transfer. At the throat location, impingement erosion is essentially zero because Al_2O_3 particles are traveling parallel to the nozzle surface.

The acronym ITE is often used; it means *integral throat/entrance* and refers to a single-piece nozzle throat insert that also includes a part of the converging entry section. ITE nozzle inserts can be seen in Figs. 1–5, 11–1, 11–3, and 11–4.

Nozzle throat erosion causes the throat diameter to enlarge during operation, and is one of the problems encountered in nozzle design. Usually, a throat area increase larger than 5% is considered unacceptable for most solid rocket appli-

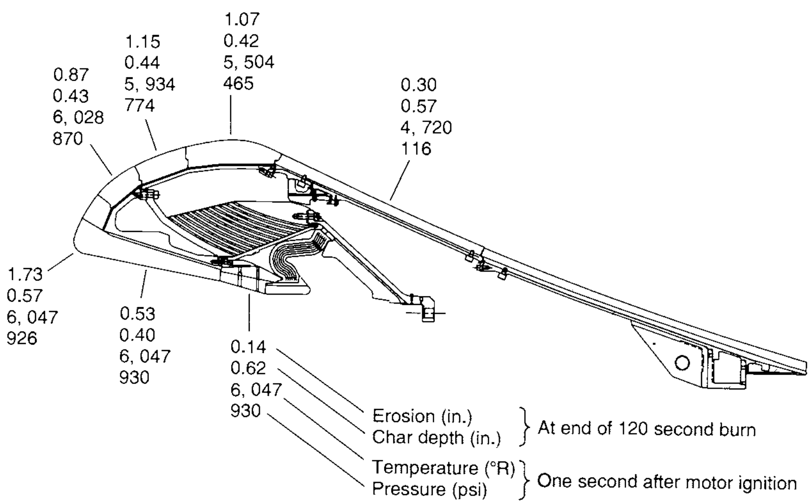


FIGURE 14–10. Erosion measurements and char depth data of the carbon fiber phenolic material of the nozzle of the Space Shuttle reusable solid rocket motor. (Courtesy of Thiokol Propulsion, a Division of Cordant Technologies.)

cations, since it causes a reduction in thrust and chamber pressure. Erosion occurs not only at the throat region (typically, at 0.01 to 0.25 mm/sec or 0.004 to 0.010 in./sec), but also at the sections immediately upstream and downstream of the throat region, as shown in Fig. 14–10. Nozzle assemblies typically lose 3 to 12% of their initial inert mass. Erosion is caused by the complex interaction between the high-temperature, high-velocity gas flow, the chemically aggressive species in the gas, and the mechanical abrasion by particles. The carbon in the nozzle material reacts with species like O_2, O, OH, or H_2O and is oxidized; the cumulative concentration of these species is an indication of the likely erosion. Tables 5–6 and 5–7 give chemical concentrations for the exhaust species from aluminized propellant. Fuel-rich propellants (which contain little free O_2 or O) and propellants where some of the gaseous oxygen is removed by aluminum oxidation show less tendency to cause erosion. Uneven erosion of a nozzle causes thrust misalignment.

Finding the optimum nozzle wall contour requires an analysis (computer codes for bell-shaped nozzles using the method of characteristics are mentioned in Section 3.4) to determine the wall contour which most rapidly turns the gas to near axial flow without introducing shock waves or impinging excessive aluminum oxide (Al_2O_3) particles on the nozzle wall. Figure 3–14 illustrates the key parameters which govern design of the nozzle contour; the initial angle θ_i (angle through which supersonic flow is turned immediately downstream of the nozzle throat), the throat to exit plane length L, the exit plane exit angle θ_e, and the turnback angle $\theta_i - \theta_e$. With solid or liquid particles in the exhaust the impingement can be minimized with an initial angle typically between 20 and 26° and a turnback angle of typically 10 to 15°. The length reduction of a bell-shaped nozzle (with solid particles in the gas) is typically 80 to 90% of the length of an equivalent conical nozzle with 15° half angle. The nozzle throat inlet contour is generally based on a hyperbolic spiral that uniformly accelerates the combustion gas flow to supersonic velocity at the throat plane.

Heat Absorption and Nozzle Materials

Rocket motors never reach thermal equilibrium during their firing. The temperatures of all components exposed to the heat flow increase continuously during operation. In a good thermal design the critical locations reach a maximum allowable temperature a short time after the motor stops running. The nozzle components rely on their heat-absorbing capacity (high specific heat and high energy demand for material decomposition) and slow heat transfer (good insulation with low thermal conductivity) to withstand the stresses and strains imposed by the thermal gradients and loads. The maximum allowable temperature for any of the motor materials is just below the temperature at which excessive degradation occurs (the material loses strength, melts, becomes too soft, cracks, pyrolyses, unglues, oxidizes too rapidly). The operating duration is limited by the design and amount of heat-absorbing and insulating material pieces. Stated in a different way, the objective is to design a nozzle with just sufficient heat-absorbing material mass and insulation mass at the various locations within the nozzle, so that its structures and joints will do the job for the duration of the application under all likely operating conditions.

The selection and application of the proper material is the key to the successful design of a solid rocket nozzle. Table 14–4 groups various typical nozzle materials according to their usage. The high-temperature exhaust of solid rockets presents an unusually severe environment for the nozzle materials, especially when metalized propellants are employed.

About 60 years ago nozzles were made out of a single piece of *molded polycrystalline graphite* and some were supported by metal housing structures. They eroded easily, but were low in cost. We still use them today for short duration, low chamber pressure, low altitude flight applications of low thrust, such as in certain tactical missiles. For more severe conditions a throat insert or ITE was placed into the graphite piece; this insert was a denser, better grade of graphite; later pyrolytic graphite washers and fiber-reinforced carbon materials came into use. For a period of time tungsten inserts were used; they had very good erosion resistance, but were heavy and often cracked. *Pyrolytic graphite* was introduced and is still being used as washers for the throat insert of small nozzles, as shown in Fig. 14–7. The high-strength carbon fiber and the carbon matrix were major advances in high-temperature materials. For small and medium-sized nozzles, ITE pieces were then made of *carbon–carbon*, which is an abbreviation for carbon fibers in a carbon matrix (see Ref. 14–9). The orientation of the fibers can be two-dimensional (2D) or three-dimensional (3D), as described below. Some properties of all these materials are listed in Tables 14–4 and 14–5. For large nozzles the then existing technology did not allow the fabrication of large 3D carbon–carbon ITE pieces, so layups of carbon fiber (or silicon fiber) cloth in a phenolic matrix were used.

The regions immediately upstream and downstream of the throat have less heat transfer, less erosion, and lower temperatures than the throat region, and less expensive materials are usually satisfactory. This includes various grades of graphite, or ablative materials, strong high-temperature fibers (carbon or silica) in a matrix of phenolic or epoxy resins, which are described later in this section. Figure 16–6 shows a movable nozzle with multilayer *insulators* behind the graphite nozzle pieces directly exposed to heat. These insulators (between the very hot throat piece and housing) limit the heat transfer and prevent excessive housing temperatures.

In the *diverging exit section* the heat transfer and temperatures are even lower and similar, but less capable and less expensive materials can be used here. This exit segment can be built integral with the nozzle throat segment (as it is in most small nozzles), or it can be a separate one- or two-piece subassembly which is then fastened to the smaller diameter throat segment. Ablative materials without oriented fibers as in cloth or ribbons, but with short fibers or insulating ceramic particles, can be used here. For large area ratios (upper stages and space transfer), the nozzle will often protrude beyond the vehicle's boat tail surface. This allows radiation cooling, since the exposed exit cone can

TABLE 14-4. Typical Motor Nozzle Materials and Their Functions

Function	Material	Remarks
Structure and pressure container (housing)	Aluminum	Limited to 515°C (959°F)
	Low carbon steel, high-strength steels, and special alloys	Good between 625 and 1200°C (1100 and 2200°F), depending on material; rigid and strong
Heat sink and heat-resistant material at inlet and throat section; severe thermal environment and high-velocity gas, with erosion	Molded graphite	For low chamber temperatures and low pressures only; low cost
	Pyrolitic graphite	Has anisotropic conductivity
	Tungsten, molybdenum, or other heavy metal	Heavy, expensive, subject to cracking; resists erosion
	Carbon or Kevlar fiber cloth with phenolic or plastic resins	Sensitive to fiber orientation. Ablative materials Used with large throats
	Carbon–carbon	Three- or four dimensional interwoven filaments, strong, expensive, limited to 3300°C (6000°F)
Insulator (behind heat sink or flame barrier); not exposed to flowing gas	Ablative plastics, with fillers of silica or Kevlar, phenolic resins	Want low conductivity, good adhesion, ruggedness, erosion resistance; can be filament wound or impregnated cloth layup with subsequent machining
Flame barrier (exposed to hot low-velocity gas)	Ablative plastics (same as insulators but with less filler and tough rubber matrix)	Lower cost than carbon–carbon; better erosion resistance than many insulators
	Carbon, Kevlar or silica fibers with phenolic or epoxy resin	Cloth or ribbon layups; woven and compressed, glued to housng
	Carbon–carbon	Higher temperature than others, three-dimensional weave or layup
Nozzle exit cone	Ablative plastic with metal housing structure	Heavy, limited duration; cloth or woven ribbon lay-ups, glued to housing
	Refractory metal (tantalum, molybdenum)	Radiation cooled, strong, needs coating for oxidation resistance; can be thin, limited to 1650°C (3000°F), unlimited duration
	Carbon–carbon, may need gas seal	Radiation cooled, higher allowable temperature than metals; two- or three-dimensional weave, strong, often porous

TABLE 14–5. Comparison of Properties of Molded and Pyrolytic Graphite, Carbon–Carbon, Carbon Cloth, and Silica Cloth Phenolic

	ATJ Modern Graphite	Pyrolytic Graphite	Three-Dimensional Carbon Fibers in a Carbon Matrix	Carbon Cloth Phenolic	Silica Cloth Phenolic
Density ($lbm/in.^3$)	0.0556	0.079	0.062 to 0.072	0.053	0.062
Thermal expansion (in./in./°F)	0.005 to 0.007	0.00144 (warp) 0.0432 (fill)	$1-9 \times 10^{-6}$	8.02×10^{-6}	7.6×10^{-6}
Thermal conductivity (Btu/in.-sec/°F) at room temperature	1.2×10^{-3} 1.5×10^{-3}	4.9×10^{-5} (warp)[a] 4.2×10^{-5} (fill)[a]	2 to 21×10^{-5} (warp)[a] 8 to 50×10^{-5} (fill)[a]	2.2×10^{-3} (warp)[a]	1.11×10^{-3} (warp)[a]
Modulus of elasticity (psi) at room temperature	1.5×10^6 (warp)[a] 1.2×10^6 (fill)[a]	4.5×10^6 (warp)[a] 1.5×10^6 (fill)[a]	35 to 80×10^6	2.86×10^{-6} (warp)[a] 2.91×10^{-6} (fill)[a]	3.17×10^{-6} (warp)[a] 2.86×10^{-6}(fill)[a]
Shear modulus (psi)	—	0.2×10^6 (warp)[a] 2.7×10^6 (fill)[a]	—	0.81×10^6	0.80×10^6
Erosion rate (typical) (in./sec)	0.004 to 0.006	0.001 to 0.002	0.0005 to 0.001	0.005 to 0.010	0.010 to 0.020

[a] *Warp* is in direction of principal fibers. *Fill* is at right angles to warp.

reject heat by radiation to space. Lightweight thin high temperature metals (niobium, titanium, stainless steel, or a thin carbon–carbon shell) with radiation cooling have been used in a few upper-stage or spacecraft exit cone applications. Since radiation-cooled nozzle exit sections reach thermal equilibrium, their duration is unlimited.

The *housing* or *structural support of the nozzle* uses the same material as the metal case, such as steel or aluminum. The housings are never allowed to become very hot. Some of the simpler, smaller nozzles (with one, two, or three pieces, mostly graphite) do not have a separate housing structure, but use the ITE (integral throat/entry) for the structure.

Estimates of nozzle internal temperatures and temperature distributions with time can be made using two-dimensional finite element difference methods for *transient heat transfer analyses*. These are similar in principle to the transient heat transfer method described in Section 8.3 and shown in Fig. 8–21. After firing, the nozzle temperatures reach an equilibrium value by conducting heat from the hotter inner parts, which were exposed to the hot gas, to the cooler outer pieces. Sometimes the outer pieces will exceed their limit temperatures and suffer damage after firing. The *structural analysis* (stresses and strains) of the key nozzle components is dependent on the heat transfer analysis, which determines the component temperatures. This allows use of the proper material physical properties, which are temperature dependent. The design must also allow for the thermal growth and the differential expansion of adjacent parts.

Typical materials used for the ITE (integral throat and entrance) or nozzle throat insert are listed in Table 14–5. They are exposed to the most severe conditions of heat transfer, thermal stresses, and high temperatures. Their physical properties are often anisotropic; that is, their properties vary with the orientation or direction of the crystal structure or the direction of reinforcing fibers. *Polycrystalline graphites* are extruded or molded. Different grades with different densities and capabilities are available. As already mentioned, they are used extensively for simple nozzles and for ITE parts. *Pyrolytic graphite* is strongly anisotropic and has excellent conductivity in a preferred direction. A nozzle using it is shown in Fig. 14–7. It is fabricated by depositing graphite crystals on a substratum in a furnace containing methane gas. Its use is declining, but it is still installed in current rocket motors of older design.

The *carbon–carbon* material is made from carefully oriented sets of *carbon fibers* (woven, knitted, threaded, or laid up in patterns) in a *carbon matrix*. Two-dimensional (2D) material has fibers in two directions, 3D has fibers oriented in three directions (at right angle to each other), and 4D has an extra set of fibers at about a 45° angle to the other three directions. An organic liquid resin is injected into the spaces between the fibers. The assembly is pressurized, the filler is transformed into a carbon char by heating and is compacted by further injection and densification processes. The graphitization is then performed at temperatures higher than 2000°C. This material is expensive but suited to nozzle applications. Highly densified material is superior in high heat transfer regions, such as the throat. The multidirectional fiber reinforcements allow them to better withstand the high thermal stresses introduced by the steep temperature gradients within the component.

Ablative Materials. These are not only commonly used in the nozzles of rocket motors, but also in some insulation materials. They are usually a composite material of high-temperature organic or inorganic high strength fibers, namely high silica glass, aramids (Kevlar), or carbon fibers, impregnated with organic plastic materials such as phenolic or epoxy resin. The fibers may be individual strands or bands (applied in a geometric pattern on a winding machine), or come as a woven cloth or ribbon, all impregnated with resin.

The ablation process is a combination of surface melting, sublimation, charring, evaporation, decomposition in depth, and film cooling. As shown in Fig. 14–11, progressive layers of the ablative material undergo an endothermic degradation, that is, physical and chemical changes that absorb heat. While some of the ablative material evaporates (and some types also have a viscous liquid phase), enough charred and porous solid material remains on the surface to preserve the basic geometry and surface integrity. Upon rocket start the ablative material acts like any thermal heat sink, but the poor conductivity causes the surface temperature to rise rapidly. At 650 to 800 K some of the resins start to decompose endothermically into a porous carbonaceous char and pyrolysis gases. As the char depth increases, these gases undergo an endothermic cracking process as they percolate through the char in a counterflow direction to the heat flux. These gases then form an artificial fuel-rich, protective, relatively cool, but flimsy boundary layer over the char.

Since char is almost all carbon and can withstand 3500 K or 6000 R, the porous char layer allows the original surface to be maintained (but with a rough surface texture) and provides geometric integrity. Char is a weak material and can be damaged or abraded by direct impingement of solid particles in the gas. Ablative material construction is used for part or all of the chambers

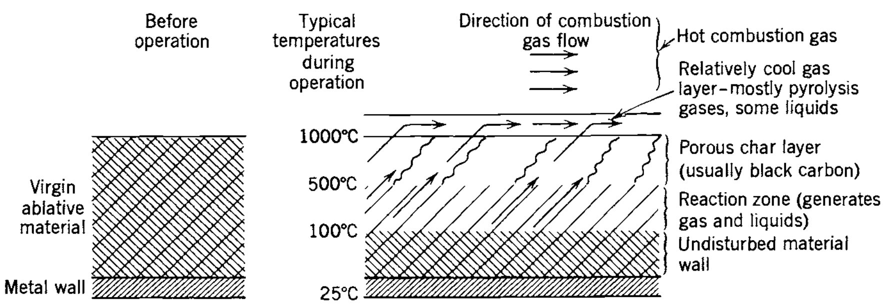


FIGURE 14–11. Zones in an ablative material during rocket operation with fibers at 45° to the flow.

and/or nozzles shown in Figs. 1–5, 6–10, 11–1 to 11–4, and 14–10.

Ablative parts are formed either by high-pressure molding (~ 55 to 69 MPa or 8000 to 10,000 psi at 149°C or 300°F) or by tapewrapping on a shaped mandrel followed by an autoclave curing process at 1000 to 2000 psi pressure and 300°F temperature. *Tapewrapping* is a common method of forming very large nozzles. The wrapping procedure normally includes heating the shaped mandrel (~ 54°C or 130°F), heating the tape and resin (66 to 121°C or 150 to 250°F), pressure rolling the tape of fiber material and the injected resin in place while rolling (~ 35,000 N/m or 200 lbf/in. width), and maintaining the proper rolling speed, tape tension, wrap orientation, and resin flow rate. Experience has proven that as-wrapped density is an important indicator of procedural acceptability, with the desired criterion being near 90% of the autoclaved density. Resin content usually ranges between 25 and 35%, depending on the fabric-reinforcing material and the particular resin and its filler material. Normally, the mechanical properties of the cured ablative material, and also the durability of the material during rocket operation, correlate closely with the cured material density. Within an optimal density range, low density usually means poor bonding of the reinforcing layers, high porosity, low strength, and high erosion rate.

In liquid propellant rockets, ablatives have been effective in very small thrust chambers (where there is insufficient regenerative cooling capacity), in pulsing, restartable spacecraft control rocket engines, and in variable-thrust (throttled) rocket engines. Figure 6–10 shows an ablative nozzle extension for a large liquid propellant rocket engine.

The heat transfer properties of the many available ablative and other fiber-based materials will depend on their design, composition, and construction. Figure 14–12 shows several common fiber orientation and approaches. The *orientation of fibrous reinforcements*, whether in the form of tape, cloth, filaments, or random short fibers, has a marked impact on the erosion resistance of composite nozzles (for erosion data see Figure 14–10). When perpendicular to the gas flow, the heat transfer to the wall interior is high because of the short conducting path. Good results have been obtained when the fibers are at 40 to 60° relative to the gas flow over the surface. Nozzle fabrication variables present wide variations in nozzle life for a given design; the variables include the

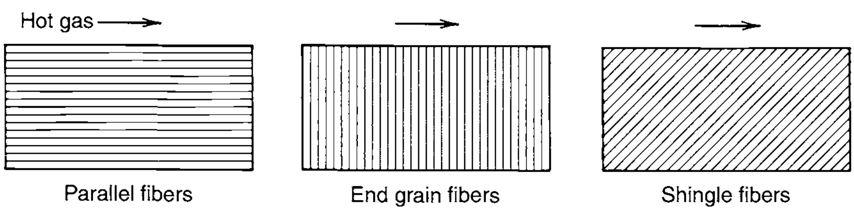


FIGURE 14–12. Simplified sketches of three different types of fiber-reinforced ablative materials.

method of wrapping, molding, and curing, resin batch processes, and resin sources.

14.3. IGNITER HARDWARE

In Section 13.2 the process of ignition was described, and in Section 12.5 some of the propellants used in igniters were mentioned briefly. In this section we discuss specific igniter types, locations, and their hardware (see Ref. 14–10).

Since the igniter propellant mass is small (often less than 1% of the motor propellant) and burns mostly at low chamber pressure (low I_s), it contributes very little to the motor overall total impulse. It is the designer's aim to reduce the igniter propellant mass and the igniter inert hardware mass to a minimum, just big enough to assure ignition under all operating conditions.

Figure 14–13 shows several alternative locations for igniter installations. When mounted on the forward end, the gas flow over the propellant surface helps to achieve ignition. With aft mounting there is little gas motion, particularly near the forward end; here ignition must rely on the temperature, pressure, and heat transfer from the igniter gas. If mounted on the nozzle, the igniter hardware and its support is discarded shortly after the igniter has used all its propellants and there is no inert mass penalty for the igniter case. There are two basic types: pyrotechnic igniters and pyrogen igniters; both are discussed below.

Pyrotechnic Igniters

In industrial practice, pyrotechnic igniters are defined as igniters (other than pyrogen-type igniters as defined further on) using solid explosives or energetic propellant-like chemical formulations (usually small pellets of propellant which

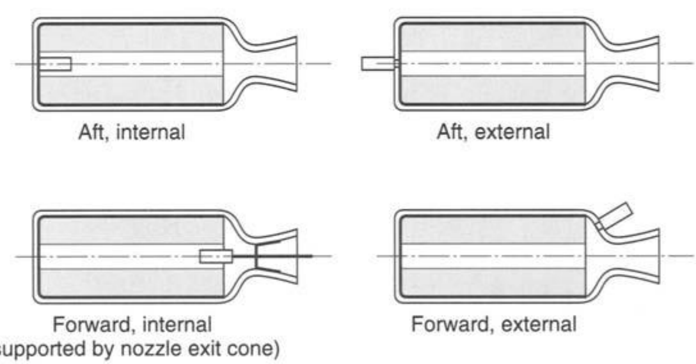


FIGURE 14–13. Simple diagrams of mounting options for igniters. Grain configurations are not shown.

give a large burning surface and a short burning time) as the heat-producing material. This definition fits a wide variety of designs, known as bag and carbon igniters, powder can, plastic case, pellet basket, perforated tube, combustible case, jellyroll, string, or sheet igniters. The common pellet-basket design in Fig. 14–14 is typical of the pyrotechnic igniters. Ignition of the main charge, in this case pellets consisting of 24% boron–71% potassium perchlorate–5% binder, is accomplished by stages; first, on receipt of an electrical signal the initiator releases the energy of a small amount of sensitive powdered pyrotechnic housed within the initiator, commonly called the squib or the primer charge; next, the booster charge is ignited by heat released from the squib; and finally, the main ignition charge propellants are ignited.

A special form of pyrotechnic igniter is the *surface-bonded* or *grain-mounted igniter*. Such an igniter has its initiator included within a sandwich of flat sheets; the layer touching the grain is the main charge of pyrotechnic. This form of igniter is used with multipulse motors with two or more end-burning grains. The ignition of the second and successive pulses of these motors presents unusual requirements for available space, compatibility with the grain materials, life, and the pressure and temperature resulting from the booster grain operation. Advantages of the sheet igniter include light weight, low volume, and high heat flux at the grain surface. Any inert material employed (such as wires and electric ceramic insulators) is usually blown out of the motor nozzle during ignition and their impacts have caused damage to the nozzle or plugged it, particularly if they are not intentionally broken up into small pieces.

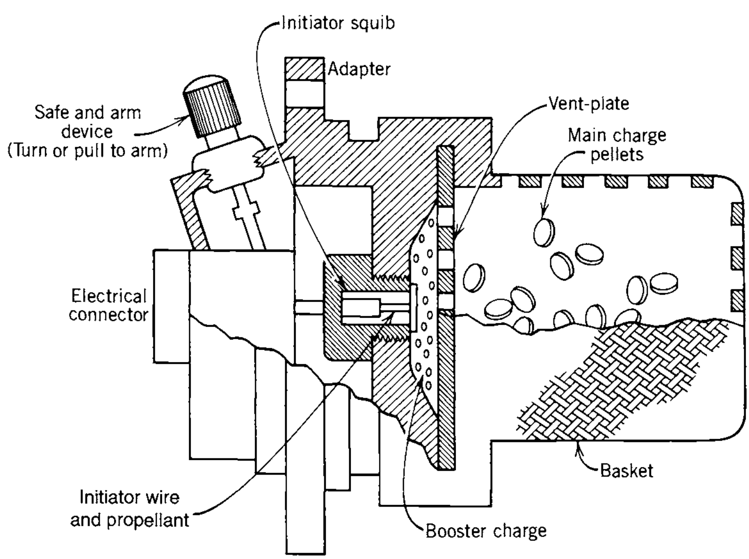


FIGURE 14–14. Typical pyrotechnic igniter with three different propellant charges that ignite in sequence.

Pyrogen Igniters

A pyrogen igniter is basically a small rocket motor that is used to ignite a larger rocket motor. The pyrogen is not designed to produce thrust. All use one or more nozzle orifices, both sonic and supersonic types, and most use conventional rocket motor grain formulations and design technology. Heat transfer from the pyrogen to the motor grain is largely convective, with the hot gases contacting the grain surface as contrasted to a highly radiative energy emitted by pyrotechnic igniters. Figures 11–1, 11–2, and 11–20 illustrate rocket motors with a typical pyrogen igniter. The igniter in Fig. 16–5 has three nozzles and a cylindrical grain with high-burn-rate propellant. For pyrogen igniters the initiator and the booster charge are very similar to the designs used in pyrotechnic igniters. Reaction products from the main charge impinge on the surface of the rocket motor grain, producing motor ignition. Common practice on the very large motors is to mount externally, with the pyrogen igniter pointing its jet up through the large motor nozzle. In this case, the igniter becomes a piece of ground-support equipment.

Two approaches are commonly used to safeguard against motor misfires, or inadvertent motor ignition; one is the use of the classical *safe and arm device* and the second is the *design of safeguards* into the initiator. Energy for unintentional ignition—usually a disaster when it happens—can be (1) static electricity, (2) induced current from electromagnetic radiation, such as radar, (3) induced electrical currents from ground test equipment, communication apparatus, or nearby electrical circuits in the flight vehicle, and (4) heat, vibration, or shock from handling and operations. Functionally, the safe and arm device serves as an electrical switch to keep the igniter circuit grounded when not operating; in some designs it also mechanically misaligns or blocks the ignition train of events so that unwanted ignition is precluded even though the initiator fires. When transposed into the *arm* position, the ignition flame can be reliably propagated to the igniter's booster and main charges.

Electric initiators in motor igniters are also called squibs, glow plugs, primers, and sometimes headers; they always constitute the initial element in the ignition train and, if properly designed, can be a safeguard against unintended ignition of the motor. Three typical designs of initiators are shown in Fig. 14–15. Both (*a*) and (*b*) structurally form a part of the rocket motor case and generically are headers. In the integral diaphragm type (*a*) the initial ignition energy is passed in the form of a shock wave through the diaphragm activating the acceptor charge, with the diaphragm remaining integral. This same principle is also used to transmit a shock wave through a metal case wall or a metal insert in a filament-wound case; the case would not need to be penetrated and sealed. The header type (*b*) resembles a simple glow plug with two high-resistance bridgewires buried in the initiator charge. The exploding bridgewire design (*c*) employs a small bridgewire (0.02 to 0.10 mm) of low-resistance material, usually platinum or gold, that is exploded by application of a high-voltage discharge.

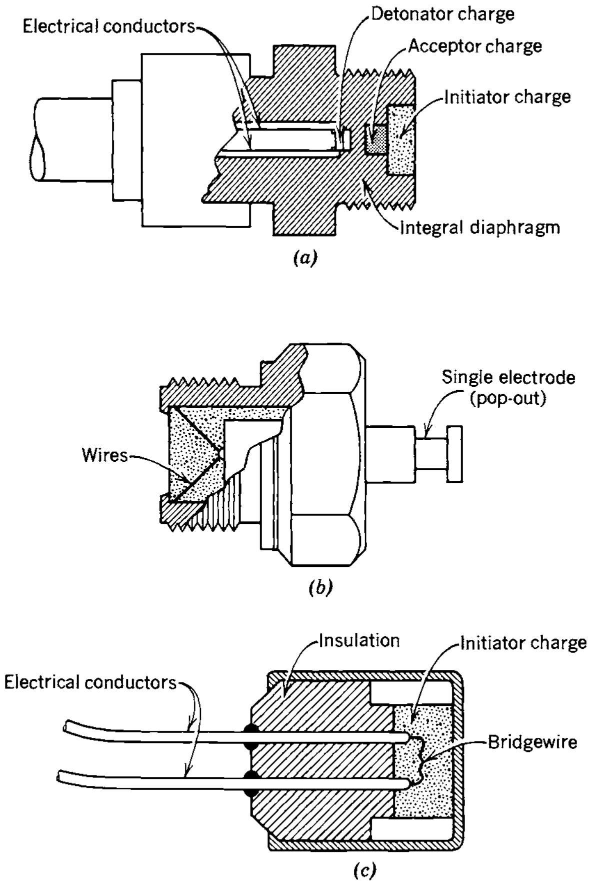


FIGURE 14–15. Typical electric initiators; (a) integral diaphragm type; (b) header type with double bridgewire; (c) exploding bridgewire type.

The safeguard aspect of the initiator appears as a basic design feature in the form of (1) minimum threshold electrical energy required for activation, (2) voltage blockage provisions (usually, air gaps or semiconductors in the electrical circuit), or (3) responsiveness only to a specific energy pulse or frequency band. Invariably, such safeguards compromise to some degree the safety provided by the classical safe and arm device.

A new method of initiating the action of an igniter is to use laser energy to start the combustion of an initiator charge. Here there are no problems with induced currents and other inadvertent electrical initiation. The energy from a small neodymium/YAG laser, external to the motor, travels in fiber-optical glass cables to the pyrotechnic initiator charge (Ref. 14–11). Sometimes an optical window in the case or closure wall allows the initiator charge to be inside the case.

Igniter Analysis and Design

The basic theories of initiating ignition, heat transfer, propellant decomposition, deflagration, flame spreading, and chamber filling are common to the design and application of pyrotechnic and pyrogen igniters. In general, the mathematical models of the physical and chemical processes that must be considered in the design of igniters are far from complete and accurate. See Chapter 5 by Hermance and Chapter 6 by Kumar and Kuo in Ref. 13–1, and Ref. 14–10.

Analysis and design of igniters, regardless of the type, depend heavily on experimental results, including past successes and failures with full-scale motors. The effect of some of the important parameters has become quite predictable, using data from developed motors. For example, Fig. 14–16 is of benefit in estimating the mass of igniter main charge for motors of various sizes (motor free volume). From these data,

$$m = 0.12(V_F)^{0.7} \tag{14–4}$$

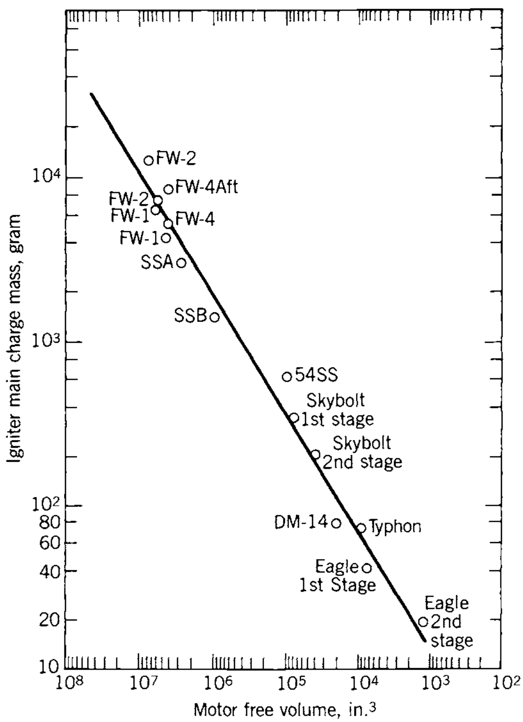


FIGURE 14–16. Igniter charge mass versus motor free volume, based on experience with various-sized rocket motors using AP/Al composite propellant. (Data with permission from Ref. 14–12.)

where m is the igniter charge in grams and V_F is the motor free volume in cubic inches or the void in the case not occupied by propellant. A larger igniter mass flow means a shorter ignition delay. The ignition time events were shown in Fig. 13–3.

14.4. ROCKET MOTOR DESIGN APPROACH

Although there are some common elements in the design of all solid propellant rocket motors, there is no single, well-defined procedure or design method. Each class of application has some different requirements. Individual designers and their organizations have different approaches, background experiences, sequences of steps, or emphasis. The approach also varies with the amount of available data on design issues, propellants, grains, hardware, or materials, with the degree of novelty (many "new" motors are actually modifications of proven existing motors), or the available proven computer programs for analysis.

Usually the following items are part of the preliminary design process. We start with the requirements for the flight vehicle and the motor, such as those listed in Table 14–6. If the motor to be designed has some similarities to proven existing motors, their parameters and flight experience will be helpful in reducing the design effort and enhancing the confidence in the design. The selection of the propellant and the grain configuration are usually made early in the preliminary design; propellant selection was discussed in Chapter 12 and grains in Chapter 11. It is not always easy for the propellant to satisfy its three key requirements, namely the performance (I_s), burning rate to suit the thrust–time curve, and strength (maximum stress and strain). A well-characterized propellant, a proven grain configuration, or a well-tested piece of hardware will usually be preferred and is often modified somewhat to fit the new application. Compared to a new development, the use of proven propellant, grain design, or hardware components avoids many analyses and tests.

An analysis of the structural integrity should be undertaken, at least in a few of the likely places, where stresses or strains might exceed those that can be tolerated by the grain or the other key components at the limits of loading or environmental conditions. An analysis of the nozzle should be done, particularly if the nozzle is complex or includes thrust vector control. Such a nozzle analysis was described briefly in an earlier section of this chapter. If gas flow analysis shows that erosive burning is likely to happen during a portion of the burning duration, it must be decided whether it can be tolerated, or whether it is excessive and a modification of the propellant, the nozzle material, or the grain geometry needs to be made. Usually a preliminary evaluation is also done of the resonances of the grain cavity with the aim of identifying possible combustion instability modes (see Chapter 13). Motor performance analysis, heat transfer, and stress analyses in critical locations will usually be done.

TABLE 14-6. Typical Requirements and Constraints for Solid Rocket Motors

Requirement Category	Examples
Application	Definition of mission, vehicle and propulsion requirements, flight paths, maneuvers, environment
Functional	Total impulse, thrust–time curve, ignition delay, initial motor mass, specific impulse, TVC angles and accelerations, propellant fraction, class 1.1 to 1.3, burn time, and tolerances on all of these parameters
Interfaces	Attachments to vehicle, fins, TVC system, power supply, instruments, lifting and transport features, grain inspection, control signals, shipping container
Operation	Storage, launch, flight environment, temperature limits, transport loads or vibrations, plume characteristics (smoke, toxic gas, radiation), life, reliability, safe and arm device functions, field inspections
Structure	Loads and accelerations imposed by vehicle (flight maneuvers), stiffness to resist vehicle oscillations, safety factors
Insensitive munitions (military application)	Response to slow and fast cook-off, bullet impact, sympathetic detonation, shock tests
Cost and schedule	Stay within the allocated time and money
Deactivation	Method of removing/recycling of propellants, safe disposal of over-age motors
Constraints	Limits on volume, length, or diameter; minimum acceptable performance, maximum cost

There is considerable interdependence and feedback between the propellant formulation, grain geometry/design, stress analysis, thermal analysis, major hardware component designs, and their manufacturing processes. It is difficult to finalize one of these without considering all the others, and there may be several iterations of each. Data from tests of laboratory samples, subscale motors, and full-scale motors have a strong influence on these steps.

Preliminary layout drawings or CAD (computer-aided design) images of the motor with its key components will be made in sufficient detail to provide sizes and reasonably accurate dimensions. For example, a preliminary design of the thermal insulation (often with a heat transfer analysis) will provide preliminary dimensions for that insulator. The layout is used to estimate volumes, inert masses, or propellant masses, and thus the propellant mass fraction.

If any of these analyses or layouts show a potential problem or a possible failure to meet the initial requirements or constraints, then usually a modification of the design, possibly of the propellant, or of the grain configuration may need to be made. The design process needs to be repeated with the changed motor design. If the proposed changes are too complex or not effective, then a change in the motor requirements may be the cure to a particular problem of noncompliance with the requirements. It is common to have several iterations in the preliminary design and the final design. Any major new feature can result in additional development and testing to prove its performance, reliability, operation, or cost; this means a longer program and extra resources.

A simplified diagram of one particular approach to motor preliminary design and development activities for a rocket motor is shown in Fig. 14–17. Not shown in this diagram are many other steps, such as igniter design and tests, liner/insulating selection, thrust vector control design and test, reliability analysis, evaluation of alternative designs, material specifications, inspection/quality control steps, safety provision, special test equipment, special test instrumentation, and so on.

If the performance requirements are narrow and ambitious, it will be necessary to study the cumulative tolerances of the performance or of various other parameters. For example, practical tolerances may be assigned to the propellant density, nozzle throat diameter (erosion), burn rate scale factor, initial burning surface area, propellant mass, or pressure exponent. These, in turn, reflect themselves into tolerances in process specifications, specific inspections, dimensional tolerances, or accuracy of propellant ingredient weighing. Cost is always a major factor and a portion of the design effort will be spent looking for lower-cost materials, simpler manufacturing processes, fewer assembly steps, or lower-cost component designs. For example, tooling for casting, mandrels for case winding, and tooling for insulator molding can be expensive. The time needed for completing a design can be shortened when there is good communication and a cooperative spirit between designers, propellant specialists, analysts, customer representatives, manufacturing people, test personnel, or vendors concerned with this effort. Reference 14–13 deals with some of the uncertainties of a particular booster motor design, and Ref. 14–14 discusses design optimization.

A *preliminary project plan* is usually formulated simultaneously with the preliminary design work. A decade or more ago the project plan was made after the preliminary design was completed. With today's strong emphasis on low cost, the people working on the preliminary designs have also to work on reducing costs on all components and processes. The project plan reflects decisions and defines the number of motors and key components to be built, the availability and lead time of critical materials or components, the type and number of tests (including aging or qualification tests); it identifies the manufacturing, inspection, and test facilities to be used, the number and kind of personnel (and when they will be needed), or any special tooling or fixtures. These decisions and data are needed to make a realistic *estimate of cost* and a *preliminary schedule*. If these exceed the allowable cost or the desired delivery schedule, then some changes have to be made. For example, this may include changes in the number of units to be built, the number and types of tests, or a redesign for easier, less costly assembly. However, such changes must not compromise reliability or performance. It is difficult to

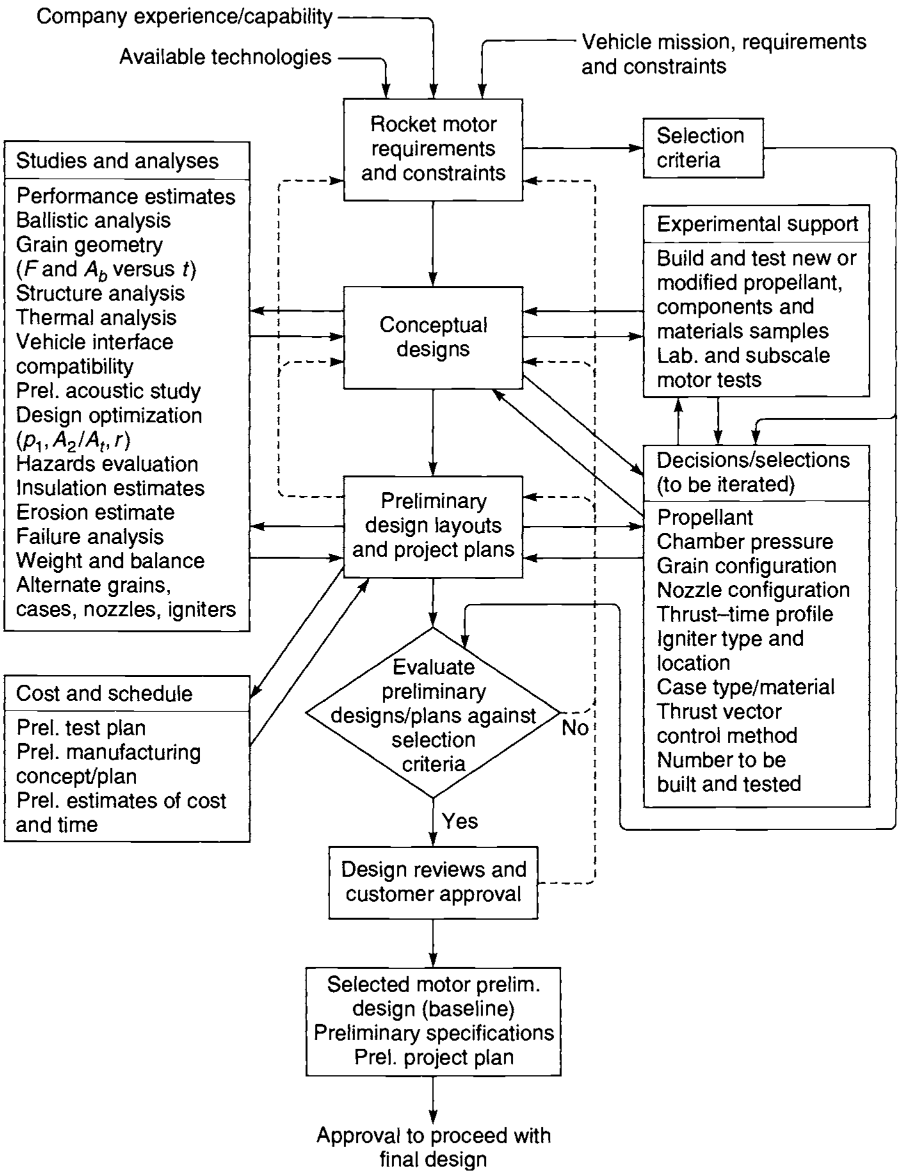


FIGURE 14-17 Simplified diagram of one approach to the preliminary design activity sequences and interrelations. Dashed lines indicate some of the feedback pathes. Some of the specific items listed here apply only to certain types of rocket motors.

make a good plan, and good cost or time estimates, when the rocket motor has not been well defined or designed in sufficient detail. These plans and estimates are therefore largely based on experience with prior successful similar rocket motors.

The final result of a preliminary design will be layout drawings or CAD images of the selected configuration, a prediction of performance, an estimate of motor mass (and, if needed, also the travel of the center of gravity), an identification of the propellant, grain, geometry, insulation, and several of the key materials of the hardware components. An estimate of predicted reliability and motor life would be accompanied by supporting data. All this information would be presented for review of the selected preliminary design. The review would be undertaken by a diverse group of motor experts, the vehicle designer, safety engineers, specialists in manufacturing, assembly, or inspection, customer representatives, analysts, and others. Here the preliminary design team explains why they selected their particular design and how it meets the requirements. With competent reviewers there usually will be suggestions for changes or further improvements. The project plan, preliminary cost estimates, and preliminary schedule are sometimes included with the design review, but more often these are presented to a different group of experts, or just to the customer's experts.

After the design review and the approval of the selected preliminary design, the detail or *final design* of all parts and components and the writing of certain specifications can begin. During manufacture and development testing some design changes may become necessary to improve the manufacture, reduce cost, or remedy a technical problem that became evident. In many organizations the final, detailed design is again submitted to a design review before manufacturing can begin. The new motor will then start its development testing. In some larger, expensive motors, that have a lot of heritage from prior proven motors, the development may consist of a single motor firing. For motors which are built in large quantities and for motors with major new features, the development and qualification may involve the testing of 10 to 30 motors. The final design ends when all detail drawings or CAD images and a final parts list are completed, and specifications for motor testing, certain manufacturing operations, or materials/component acceptance have been prepared. The detail design is considered to be completed when the motor successfully passes its development and qualification tests and begins production for deliveries.

Example 14–1. This example shows one method for making a preliminary determination of the design parameters of a solid rocket using a composite propellant. The rocket is launched at altitude and flies at constant altitude. The following data are given:

Specific impulse (actual)	$I_s = 240$ sec at altitude and 1000 psia
Burning rate	$r = 0.8$ in.,/sec at 1000 psia and 60°F
Propellant density	$\rho_b = 0.066$ lbm/in.^{3} at sea level
Specific heat ratio	$k = 1.25$
Chamber pressure, nominal	$p_1 = 1000$ psi
Desired average thrust	$F = 20,000$ lbf
Maximum vehicle diameter	$D = 16$ in.
Desired duration	$t_b = 5.0$ sec

Ambient pressure	3.0 psi (at altitude)
Vehicle payload	5010 lbm (includes structure)

Approximately neutral burning is desired.

SOLUTION

a. Basic Design. The total impulse I_t and propellant weight at sea level w_b are obtained from Eqs. 2–2 and 2–5. $I_t = Ft_b = I_s w_b = 20{,}000 \times 5.0 = 100{,}000$ lbf-sec. The propellant weight is 100,000/240 = 417 lbf. Allowing for a loss of 2% for manufacturing tolerances and slivers, the total propellant weight is $1.02 \times 417 = 425$ lbf. The volume required for this propellant V_b is given by $V_b = w_b/\rho_b = 425/0.066 = 6439$ in.^{3}. The web thickness $b = rt_b = 0.80 \times 5 = 4.0$ in.

b. Case Dimensions. The outside diameter is fixed at 16.0 in. Heat-treated steel with an ultimate tensile strength of 220,000 psi is to be used. The wall thickness t can be determined from Eq. 14–1 for simple circumferential stress as $t = (p_1/D)/(2\sigma)$. The value of p_1 depends on the safety factor selected, which in turn depends on the heating of the wall, the prior experience with the material, and so on; a safety factor of 2.0 is suggested to allow for surface scratches, combined stresses and welds, and rough field handling. The value of D is the average diameter to the center of the wall. The wall thickness t is

$$t = 2.0 \times 1000 \times 15.83/(2 \times 220{,}000) = 0.086 \text{ in.}$$

A spherical head end and a spherical segment at the nozzle end similar to Fig. 11–1 is assumed.

c. Grain Configuration. The grain will be cast into the case but will be thermally isolated from the case with an elastomeric insulator with an average thickness of 0.100 in. inside the case; the actual thickness will be less than 0.10 in. in the cylindrical and forward closure regions, but thicker in the nozzle entry area. The outside diameter D for the grain is determined from the case thickness and liner to be $16.0 - 2 \times 0.086 - 2 \times 0.10 = 15.62$ in. The inside diameter D_i of a simple hollow cylinder grain would be the outside diameter D_o minus twice the web thickness or $D_i = 15.62 - 2 \times 4.0 = 7.62$ in. For a simple cylindrical grain, the volume determines the effective length, which can be determined from the equation

$$V_b = \frac{\pi}{4} L(D_o^2 - D_i^2)$$

$$L = \frac{6439 \times 4}{\pi(15.62^2 - 7.62^2)} = 44.05 \text{ in.}$$

The web fraction would be $2b/D_o = 8/15.62 = 0.512$. The L/D_o is (approximately) $44.05/15.62 = 2.82$.

For grains with this web fraction and this L/D_o ratio, Table 11–4 suggests the use of an internal burning tube with some fins for a cone. A conocyl configuration is selected, although a slotted tube or fins would also be satisfactory. These grain shapes are shown in Figs. 11–1 and 11–17. The initial or average burning area will be found from Eqs. 11–1 and 2–5: namely $F = \dot{w} I_s = \rho_b A_b r I_s$

$$A_b = \frac{F}{\rho_b r I_s} = \frac{20{,}000}{0.066 \times 0.8 \times 240} = 1578.3 \text{ in.}^2$$

The actual grain now has to be designed into the case with spherical ends, so it will not be a simple cylindrical grain. The approximate volume occupied by the grain is found by subtracting the perforation volume from the chamber volume. There is a full hemisphere at the head end and a partial hemisphere of propellant at the nozzle end (0.6 volume of a full hemisphere).

$$V_b = \tfrac{1}{2}(\pi/6)D_o^3(1 + 0.6) + (\pi/4)D_o^2 L - (\pi/4)D_i^2(L + D_i/2 + 0.3D_i/2) = 6439 \text{ in.}^3$$

This is solved for L, with $D = 15.62$ in. and the inside diameter $D_i = 7.62$ in. The answer is $L = 36.34$ in. The initial internal hollow tube burn area is about

$$\pi D_i(L + D_i/2 + 0.3D_i/2) = 1113 \text{ in.}^2$$

The desired burn area of 1578 in.^{2} is larger by about 465 in.^{2}. Therefore, an additional burn surface area of 465 in.^{2} will have to be designed into the cones of a conocyl configuration or as slots in a slotted tube design. Actually, a detailed geometrical study should be made analyzing the instantaneous burn surface after arbitrary short time intervals and selecting a detailed grain configuration where A_b stays approximately constant. This example does not go through a preliminary stress and elongation analysis, but it should be done.

d. *Nozzle Design*. From Chapter 3 the nozzle parameters can be determined. The thrust coefficient C_F can be found from curves of Figs. 3–6, 3–7, and 3–8 or Eq. 3–30 for $k = 1.25$ and a pressure ratio of $p_1/p_2 = 1000/3 = 333$. Then $C_F = 1.73$. The throat area is from Eq. 3–31:

$$A_t = F/p_1 C_F = 20{,}000/(1000 \times 1.73) = 11.56 \text{ in.}^2$$

The throat diameter is $D_t = 3.836$ in. The nozzle area ratio for optimum expansion (Fig. 3–6) A_2/A_t is about 27. The exit area and diameter are therefore about $A_e = 312$ in.^{2} and $D_e = 19.93$ in. However, this is larger than the maximum vehicle diameter of 16.0 in. ($A_2 = 201$ in.^{2}), which is the maximum for the outside of the nozzle exit. Allowing for an exit cone thickness of 0.10 in., the internal nozzle exit diameter D_2 is 15.80 in. and A_2 is 196 in.^{2}. This would allow only a maximum area ratio of 196/11.56 or 16.95. Since the C_F values are not changed appreciably for this new area ratio, it can be assumed that the nozzle throat area is unchanged.

This nozzle can have a thin wall in the exit cone, but requires heavy ablative materials, probably in several layers near the throat and convergent nozzle regions. The thermal and structural analysis of the nozzle is not shown here.

c. *Weight Estimate*. The steel case weight (assume a cylinder with two spherical ends and that steel weight density is 0.3 lbf/in.^{3}) is

$$
\begin{aligned}
t\pi DL\rho + (\pi/4)tD^2\rho &= 0.086\pi\ 15.83 \times 36.34 \times 0.3 \\
&\quad + 0.785 \times 0.086 \times 15.83^2 \times 0.3 \\
&= 50.9 \text{ lbf}
\end{aligned}
$$

With attachments, flanges, igniter, and pressure tap bosses, this is increased to 57.0 lb. The nozzle weight is composed of the weights of the individual parts, estimated for their densities and geometries. This example does not go through the detailed calculations, but merely gives the result of 30.2 lb. Assume an expended igniter propellant weight of 2.0 lb and a full igniter weight of 5.0 lb. The total weight then is

Case weight at sea level	57.0 lbf
Liner/insulator	14.2 lbf
Nozzle, including fasteners	30.2 lbf
Igniter case and wires	2.0 lbf
Total inert hardware weight	103.4 lbf
Igniter powder	3 lbf
Propellant (effective)	417 lbf
Unuseable propellant (2%)	8 lbf
Total weight	531.4 lbf
Propellant and igniter powder	420.0 lbf

f. *Performance*. The total impulse-to-weight ratio is 100,000/531.4 = 188.2. Comparison with I_s shows this to be an acceptable value, indicating a good performance. The total launch weight is 5010 + 531.4 = 5541 lbf, and the weight at burnout or thrust termination is 5541–420 = 5121 lbf. The initial and final thrust-to-weight ratios and accelerations are

$$
F/w = 20{,}000/5541 = 3.61
$$
$$
F/w = 20{,}000/5121 = 3.91
$$

The acceleration in the direction of thrust is 3.61 times the gravitational acceleration at start and 3.91 at burnout.

g. *Erosive Burning*. The ratio of the port area to the nozzle throat area at start is $(7.55/3.836)^2 = 3.95$. This is close to the limit of 4.0, and erosive burning is not likely to be significant. A simple analysis of erosive burning in the conical cavity should also be made, but it is not shown here.

PROBLEMS

1. In Figs. 13–4 and 14–16 it can be seen that higher pressures and higher heat transfer rates promote faster ignition. One way to promote more rapid ignition is for the nozzle to remain plugged until a certain minimum pressure has been reached, at which time the nozzle plug will be ejected. Analyze the time saving achieved by such a device, assuming that the igniter gas evolution follows Eqs. 11–3 and

11–11. Under what circumstances is this an effective method? Make assumptions about cavity volume, propellant density, etc.

2. Compare a simple cylindrical case with hemispheric ends (ignore nozzle entry or igniter flanges) for an alloy steel metal and two reinforced fiber (glass and carbon)-wound filament case. Use the properties in Table 14–2 and thin shell structure theory. Given:

Length of cylindrical portion	370 mm
Outside cylinder diameter	200 mm
Internal pressure	6 MPa
Web fraction	0.52
Insulator thickness (average)	
for metal case	1.2 mm
for reinforced plastic case	3.0 mm
Volumetric propellant loading	88%
Propellant specific gravity	1.80
Specific impulse (actual)	248 sec
Nozzle igniter and mounting provisions	0.20 kg

Calculate and compare the theoretical propulsion system flight velocity (without payload) in a gravity-free vacuum for these three cases.

3. The following data are given for a case that can be made of either alloy steel or fiber-reinforced plastic.

Type	Metal	Reinforced Plastic
Material	D6aC	Organic filament composite (Kevlar)
Physical properties		See Table 14–2
Poisson ratio	0.27	0.38
Coefficient of thermal expansions, m/m-K $\times 10^{-6}$	8	45
Outside diameter (m)	0.30	0.30
Length of cylindrical section (m)	0.48	0.48
Hemispherical ends		
Nozzle flange diameter (m)	0.16	0.16
Average temperature rise of case material during operation (°F)	55	45

Determine the growth in diameter and length of the case due to pressurization, heating, and the combined growth, and interpret the results.

4. A high-pressure helium gas tank at 8000 psi maximum storage pressure and 1.5 ft internal diameter is proposed. Use a safety factor of 1.5 on the ultimate strength. The following candidate materials are to be considered:

Kevlar fibers in an epoxy matrix (see Table 14–2)
Carbon fibers in an epoxy matrix
Heat-treated welded titanium alloy with an ultimate strength of 150,000 psi and a weight density of 0.165 lb/in.^{3}

Determine the dimensions and sea level weight of these three tanks and discuss their relative merits. To contain the high-pressure gas in a composite material that is porous, it is also necessary to include a thin metal inner liner (such as 0.016 in.-thick aluminum) to prevent loss of gas; this liner will not really carry structural loads, but its weight and volume need to be considered.

5. Make a simple sketch and determine the mass or sea level weight of a rocket motor case that is made of alloy steel and is cylindrical with hemispherical ends. State any assumptions you make about the method of attachment of the nozzle assembly and the igniter at the forward end.

Outer case and vehicle diameters	20.0 in.
Length of cylinder portion of case	19.30 in.
Ultimate tensile strength	172,000 psi
Yield strength	151,300 psi
Safety factor on ultimate strength	1.65
Safety factor on yield strength	1.40
Nozzle bolt circle diameter	12.0 in.
Igniter case diameter (forward end)	3.00 in.
Chamber pressure, maximum	1520 psi

6. Design a solid propellant rocket motor with insulation and liner. Use the AP/Al-HTPB propellant from Table 11–3 for Orbus 6. The average thrust is 3600 lbf and the average burn time is 25.0 sec. State all the assumptions and rules used in your solution and give your reasons for them. Make simple sketches of a cross section and a half section with overall dimensions (length and diameter), and determine the approximate loaded propellant mass.

7. The STAR 27 rocket motor (Fig. 11–1 and Table 11–3) has an average erosion rate of 0.0011 in./sec. (a) Determine the change in nozzle area, thrust, chamber pressure, burn time, and mass flow at cut-off. (b) Also determine those same parameters for a condition when, somehow, a poor grade of ITE material was used that had three times the usual erosion rate. Comment on the difference and acceptability.
Answer: Nozzle area increases by about (a) 5.3% and (b) 14.7% and chamber pressure at cutoff decreases by approximately the same percentage.

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