4

Solid propellant rocket motors

Considering the complexities of the liquid propellant rocket engine, it does not seem remarkable that so much attention has been given to the design and development of the much simpler solid propellant motor. This has a range of applications: the main propulsion system for small and medium launchers; as a simple and reliable third stage for orbital injection; and most of all as a strap-on booster for many modern heavy launchers. The solid propellant is storable, and is relatively safe to handle; no propellant delivery system is required, and this produces a huge improvement in reliability and cost. There are two main disadvantages: the motor cannot be controlled once ignited (although the thrust profile can be preset), and the specific impulse is rather low because of the low chemical energy of the solid propellant.

4.1 BASIC CONFIGURATION

Thermodynamically a solid-fuelled rocket motor is identical to a liquid-fuelled engine. The hot gas produced by combustion is converted to a high-speed exhaust stream in exactly the same way, and so the nozzle, the throat and the restriction in the combustion chamber leading to the throat are all identical in form and function. The thrust coefficient is calculated in the same way as for a liquid-fuelled engine, as is the characteristic velocity. The theoretical treatment in Chapter 2 serves for both.

The hot gas is produced by combustion on the hollow surface of the solid fuel block, known as the *charge*, or *grain*. In most cases the grain is bonded to the wall of the combustion chamber to prevent access of the hot combustion gases to any surface of the grain not intended to burn, and to prevent heat damage to the combustion chamber walls. The grain contains both fuel and oxidant in a finely divided powder form, mixed together and held by a binder material.

Figure 4.1 shows a typical solid-motor configuration. In comparison with the liquid rocket combustion chamber it is very simple. It consists of a casing for the

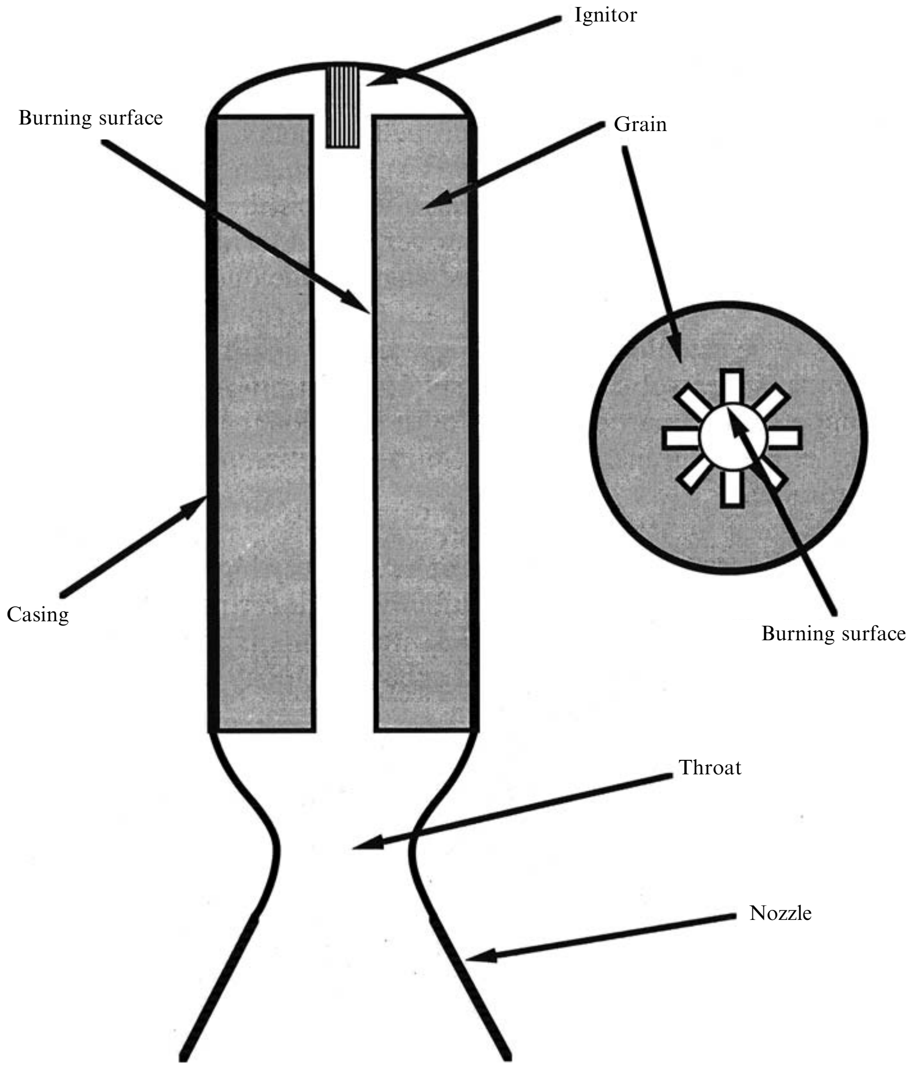

Figure 4.1. Schematic of a solid-fuelled rocket motor.

propellant, which joins to a nozzle of identical geometry to that of a liquid-fuelled engine. Once the inner surface of the grain is ignited, the motor produces thrust continuously until the propellant is exhausted.

The fundamental simplicity of the solid propellant rocket enables wide application. The exhaust velocity is not very high—the most advanced types can produce about 2,700 m s^{-1}—but the absence of turbo-pumps and separate propellant tanks, and the complete absence of complicated valves and pipelines, can produce a high mass ratio, low cost, or both. In addition, the reliability is very high, due to the small number of individual components compared with a liquid-fuelled engine. The one big disadvantage is that the device cannot be test fired, and so the reliability has to be established by analogy and by quality control. This has important implications when solid motors are used in human space flight. The two areas in which solid motors excel are as strap-on boosters and as upper stages, particularly for orbit insertion or for circularisation of elliptical transfer orbits. Solid propellants are, by definition, storable.

As a booster, a solid motor can have a very high mass-flow rate and therefore high thrust, while the engineering complexity and cost can be low in a single use item. This is ideal for the early stages of a launch where high exhaust velocity is not an issue. To produce the same thrust with a liquid-fuelled rocket would not require such a large engine, because of the higher specific chemical energy of some liquid propellants, but it would be much more costly and less reliable. Very large solid boosters can be made and fuelled in sections which are then bolted together, which again makes for simplicity of construction and storage of what would otherwise be a very large unit.

As a final stage the solid motor is again reliable, and is well adapted to high mass-ratio. While the dead weight of a liquid stage includes turbo-pumps and empty tanks for two separate propellants, the dead weight of a solid stage is just the casing and the nozzle. The casing for upper stages is often made of composite materials, reducing the mass even further. It is also convenient to make such a stage with a spherical or quasispherical form, so as to minimise the mass of containing walls.

4.2 THE PROPERTIES AND THE DESIGN OF SOLID MOTORS

In comparison with a liquid-fuelled engine, the solid motor is very simple, and the design issues are therefore fewer. There is no injector, and no propellant distribution system. Design issues related to the propellant are mostly concerned with selection of the propellant type and the mounting and protection of the propellant in the casing, and ignition is similar to that of a liquid-fuelled engine. There are no propellant tanks, but the casing has to contain the propellant and also behave as a combustion chamber. For boosters the casing is large, and to combine large size with resistance to high combustion pressure is very different from the same issue in a liquid system where the requirements are separated. Cooling is totally different, because there are no liquids involved and heat dissipation has to be entirely passive.

Thrust stability—which for a liquid-fuelled rocket is dependent only on a steady supply of propellant once the chamber and injector have been optimised—is very complicated for a solid propellant. Here the supply of combustible material is dependent on conditions in the combustion chamber, and there are increased chances for instabilities to arise and propagate. Associated with stability is thrust control. For a liquid rocket the thrust is actively controlled by the rate of supply of propellants, and in the majority of cases it is stabilised at a constant value. For a solid rocket the thrust depends on the rate of supply of combustible propellant; this depends on the pressure and temperature at the burning surface, and it cannot actively be controlled. In the same way, a liquid-fuelled engine can be shut down by closing valves, whereas the solid motor continues to thrust until all the propellant is exhausted. These design issues are central to the correct performance of a solid propellant rocket motor.

While the solid-fuelled rocket is essentially a single-use item, the cost of large boosters is very high, and the necessary engineering quality of some components—specifically the casing—may make them suitable for reuse. This was a design feature for both the Space Shuttle and the Ariane 5 solid boosters. The Space Shuttle boosters are recovered, and the segments are reused. The Ariane 5 boosters are also recovered, but only for post-flight inspection.

4.3 PROPELLANT COMPOSITION

While there is a wide choice of propellant composition for liquid-fuelled rocket engines, the choice is considerably more narrow for solid propellants. Rather than selecting a particular propellant for a particular purpose, each manufacturer has its own optimised propellant mixture. The basic sold propellant consists of two or more chemical components which react together to produce heat and gaseous products. Solid propellants have been used since the earliest times, and until this twentieth century were based on gunpowder—a mixture of charcoal, sulphur and saltpetre. Modern propellants do not differ in fundamentals from these early mixtures. The oxidant is usually one of the inorganic salts such as potassium nitrate (saltpetre) although chlorates and perchlorates are now more commonly used. The fuels sometimes include sulphur, and carbon is present in the form of the organic binder.

As with any other type of rocket, the aim is to achieve the highest combustion temperature together with the lowest molecular weight in the exhaust. The difficulty with solid oxidants is that they are mostly inorganic and contain metal atoms. These lead to higher molecular weight molecules in the exhaust. Similarly the solid fuels generally have a higher atomic weight than hydrogen, and so again the molecular weight in the exhaust is driven up. The chemical energy, per unit mass of propellant, can be the same as for the main liquid propellants, and so the combustion temperature is similar. A particular problem is that some of the combustion products may form solid particles at exhaust temperatures. This affects the performance of the nozzle in converting heat energy into gas flow. All of these properties affect the performance of solid motors.

The charge of propellant in a solid rocket motor is often called the *grain*. The basic components of the grain are fuel, oxidant, binder, and additives to achieve burning stability and stability in storage. The finished charge must also be strong enough to resist the forces induced by vehicle motion and thrust. It must also be thermally insulating to prevent parts of the grain—other than the burning surface—from reaching ignition temperature.

In the past, two different kinds of solid propellant have been used. The first kind is the mixture of inorganic oxidants with fuels, as described above. This is the most commonly used today. The other type is based on nitrated organic substances such as nitroglycerine and nitrocellulose. These came into use as gun propellants, after gunpowder, and it was natural that they should be considered as rocket propellants. These materials have the property that they contain the oxidant and fuel together in a single molecule or group of molecules. Heat induces a reaction in which the complex organic molecule breaks down, which produces heat and gaseous oxides of nitrogen, carbon and hydrogen. The molecular weight of such gas mixtures is rather low, giving an advantage in terms of exhaust velocity. These propellants are termed *homogeneous* propellants, for obvious reasons. They are not used for launcher boosters and most orbital change motors, because they have been superseded by more advanced, heterogeneous, propellants.

The fundamental requirement is to develop high thrust per unit mass. As discussed in Chapter 2, this requires a high combustion temperature and low molecular weight of the combustion products. In general a relatively high temperature of combustion is easy to achieve, but it is impossible to have the same low molecular weight of the products achievable with liquid hydrogen and liquid oxygen. The presence of carbon and the byproducts of the inorganic oxidants, potassium and sodium salts, produces a higher molecular weight and hence a lower exhaust velocity. Referring back to Chapters 1 and 2, we can see that high molecular weight does not prevent the solid motor from developing high thrust, which is just a matter of high mass flow and throat area. High ultimate vehicle velocity is harder to achieve with a solid motor because of the low exhaust velocity. A typical value would be about $2{,}700\,\mathrm{m\,s^{-1}}$. For final stages, optimisation is directed towards improving the mass ratio rather than the exhaust velocity.

In modern propellants metallic powders are often added to increase the energy release and hence the combustion temperature. Aluminium is usual, and in this case the exhaust products will contain aluminium oxide, which has a high molecular weight and is refractory, and so is in the form of small solid particles. Particles in the exhaust stream reduce efficiency: they travel more slowly than the surrounding high-velocity gas, and they radiate heat more effectively (as black bodies) and therefore reduce the energy in the stream. The loss of exhaust velocity may be balanced by the higher combustion temperature and an increase in effective density of the exhaust gases. This increases the mass flow and hence the thrust. High thrust is applicable for a first-stage booster where ultimate velocity is not as important as the thrust at lift-off. In designing a motor for high thrust, increasing the exhaust density may be preferable to an increase in throat diameter and hence in overall size of the booster; the mass ratio is also increased if the grain density is higher. The presence of particles in the exhaust produces the characteristic dense white 'smoke' seen when the boosters ignite. The exhaust from a liquid-fuelled engine is usually transparent.

The most commonly used modern solid propellant is based on a polybutadiene synthetic rubber binder, with ammonium perchlorate as the oxidiser, and some 12–16% of aluminium powder. The boosters for the Space Shuttle use this type of propellant, as do the boosters for Ariane 5 and many upper stages. The combustion temperature without the aluminium is about 3,000 K with 90% of ammonium perchlorate. The addition of 16–18% aluminium increases the temperature to

3,600 K for the Ariane 5 booster, and the oxidiser concentration is reduced correspondingly.

The chemical composition of the exhaust is approximately 32% aluminium oxide, 20% carbon monoxide, 16% water, 12% hydrogen chloride, 10% nitrogen, 7% carbon dioxide and 3% chlorine and hydrogen. A major part of the aluminium oxide, which is initially in the vapour phase, condenses into solid particles in the nozzle, but fortunately this does not contribute to the molecular weight in the expanding gases: Al_2O_3 has a molecular weight of 102. The combined effect of the gaseous components is to produce an average molecular weight of about 25. The combustion parameter is 12, giving a characteristic velocity of 1,700 m s^{-1}. The particles will reduce the mean exhaust velocity because of the effects mentioned above. The quoted vacuum exhaust velocity is 2,700 m s^{-1}, which is fairly close to the theoretical value if we assume a reasonable thrust coefficient. So this *two-phase flow*—in which the exhaust gases follow the normal expansion, cooling and acceleration, alongside particles which are accelerated by the gas—does not reduce the exhaust velocity very much. If the particles were to evaporate then a very high molecular weight gas would result, producing a very low exhaust velocity. This solid propellant is therefore rather efficient in producing high thrust and a reasonable exhaust velocity.

4.3.1 Additives

In a heterogeneous propellant the oxidant is the main constituent by mass, and the binder—usually polybutadiene rubber, a hydrocarbon—is the fuel. Aluminium is also present at 16–18%, and other materials are added to improve performance or safety. Carbon is present to render the propellant opaque to infrared radiation, so that the propellant cannot be internally ignited by heat radiated through the bulk material from the burning surface; it produces the characteristic black colour. Plasticisers are added to improve moulding and extrusion of the material. Other materials, such as inorganic salts, are added to control burning and to achieve the desirable value for the pressure-burning rate index. This is necessary for the so-called 'double base propellants'—those consisting mainly of nitroglycerine and nitrocellulose. For the heterogeneous propellants the oxidants themselves act in this way. Iron oxide is added at about the 1% level to assist smooth combustion, and waxes are also added to some propellants to lubricate extrusion.

Having arrived at an optimum composition for the main constituents, the additives are included to produce stability, storage qualities and mechanical strength. The latter is an important property. The whole mass of propellant—which is sometimes the biggest single mass in the whole vehicle (a single Ariane 5 booster weighs 260 tonnes)—has to be accelerated by the thrust. The propellant has to support this acceleration, without rupture or significant distortion, and also has to transfer the combustion pressure to the casing and maintain its integrity. The development of large boosters depends to some extent on the physical strength of the propellant.

4.3.2 Toxic exhaust

Launch vehicle boosters are fired close to the ground, and most of the exhaust is dispersed over a wide area of the launch site. While the products of liquid engines are mostly harmless, the chlorine in the oxidants of solid boosters produces hydrogen chloride, and particulates can also be dangerous. Thus it is important that during lift-off the booster exhaust is channelled away safely by water-cooled open ducts. Of course, once the rocket is in flight this is beyond control, and dilution of the exhaust products by the atmosphere, as they fall to earth, has to be relied upon. For most launch sites the area is evacuated and cleaned down after a launch; this may take several days.

4.3.3 Thrust stability

The overall thrust profile can be controlled by the shape of the charge, but other factors are important in understanding the way a solid motor performs, the most important of which is the stability of the thrust. In the liquid-fuelled engine the chamber pressure is usually constant and, with the mass flow rate, is determined by the rate at which the propellants are delivered through the injectors. On the other hand, in the solid motor the mass flow rate is not determined by external supply but by the rate at which the surface of the burning charge is consumed, which itself is a function of the pressure in the combustion chamber.

Because of this peculiarity of solid propellant systems the rate of supply of combustible propellant increases with pressure, and stable burning is not necessarily a given. If we arbitrarily assume that the rate of consumption of the grain depends on pressure as $m = ap^{\beta}$, then the value of β controls the stability, as follows. From Chapter 2 we see that the mass flow rate out of the chamber depends linearly on the pressure. Thus if $\beta > 1$, the supply of gas from the burning grain increases faster with pressure than the rate of exhaust, and an uncontrolled rise in burning rate and pressure could result from a small initial increase. Similarly a small initial decrease in pressure could result in a catastrophic drop in burning rate. Home-made rockets tend to exhibit one or other of these distressing tendencies. If on the other hand, $\beta < 1$ then the rate of change of burning rate is always less than the (linear) rate of change of mass flow through the exhaust, and the pressure in the chamber will stabilise after any positive or negative change in burning rate. This problem—which does not occur with liquid propellant engines—is a primary consideration in the design of the solid motor and in grain composition and configuration. Some additives are used to achieve the correct dependency, where this does not arise naturally. Typical values of β range from 0.4 to 0.7.

The rate of burning of the propellant, expressed as a linear recession rate of the burning surface, depends on the rate of heat supply to the surface from the hot gas. This heat evaporates the propellant. The recession rate should be constant, under constant pressure conditions, with $\beta < 1$. There is also another effect which can change the rate at which the surface recedes: *erosive* burning, which occurs because of the velocity of the gas over the surface. With a liquid-fuelled engine, it is a fair assumption that the velocity of the gas in the combustion chamber is small and constant; it is finite because the gas has to leave the chamber. Because of the length of a solid propellant combustion chamber the gas accelerates down the void; the velocity near the nozzle can be quite large. The conditions of burning at the upper and lower portions of the charge are then different. At the top, the hot gas is fairly stagnant, while near the bottom it is moving fast, and constantly supplying energy to the burning surface. The result of this is a faster evaporation and a faster recession rate near the nozzle. If this is not checked or allowed for, then the charge can burn through at the nozzle end before the upper portion is exhausted. This may lead to failure of the casing or an unforeseen decrease in thrust, neither of which is pleasant. Paradoxically, it can be ameliorated by designing the hollow void within the grain to have an increasing cross-sectional area towards the nozzle. For constant flow rate the increased cross-section requires a corresponding decrease in the velocity, and in this way the effects of erosive burning are counterbalanced.

4.3.4 Thrust profile and grain shape

The pressure in the chamber, and hence the thrust, depends on the rate at which the grain is consumed. The pressure depends on the recession rate and on the area of the burning surface, and the mass flow rate depends on the volume of propellant consumed per second. The shape of the charge can be used to preset the way the area of the burning surface evolves with time, and hence the temporal thrust profile of the motor. The pressure and the thrust are independent of the increase in chamber volume as the charge burns away, and depend only on the recession rate and the area of the burning surface.

The simplest thrust profile comes from linear burning of a cylindrical grain (as with a cigarette): a constant burning area produces constant thrust. This shape, however, has disadvantages: the burning area is limited to the cylinder cross-section, and the burning rim would be in contact with the wall of the motor. Active cooling of the wall is of course not possible with a solid motor, and this type of charge shape can be used only for low thrust and for a short duration because of thermal damage to the casing.

The most popular configuration involves a charge in the form of a hollow cylinder, which burns on its inner surface. This has two practical advantages: the area of the burning surface can be much larger, producing higher thrust, and the unburned grain insulates the motor wall from the hot gases. In the case of a simple hollow cylinder, the area of the burning surface increases with time, as do the pressure and the thrust. If a constant thrust is desired, the inner cross-section of the grain should be formed like a cog, the teeth of which penetrate part way towards the outer surface. The area of burning is thus initially higher, and the evolving surface profile corresponds roughly to constant area and hence constant thrust. Other shapes for the grain produce different thrust profiles, depending on the design. Figure 4.2 illustrates some examples. It is common in large boosters to mix the profiles; for example, the forward segment may have a star or cog profile, while the aft segments may have a circular profile. In this way the thrust profile can be fine-

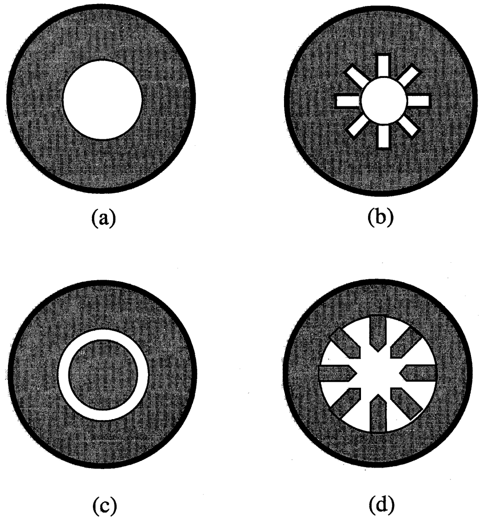

Figure 4.2. Cross-sections of grains.

tuned. It is also common to have at least one segment with a tapered profile to ameliorate erosive burning and to modify the thrust profile.

The thrust profiles associated with the shapes in Figure 4.2 can be understood from simple geometric arguments. The recession rate is assumed to be constant over the whole exposed area, which in the diagrams may be assumed to be proportional to the length of the perimeter of the burning surface. Type (a), called ‘progressive’, is the simplest to understand. Here the circumference of the circular cross-section increases linearly with time, as does the area of the burning surface, and there is a linear increase in mass flow rate and hence in thrust. Type (b)—which is perhaps the most commonly used—produces a quasi-constant thrust, because the initial burning area is quite large due to the convolutions of the cog shape; as the cog ‘teeth’ burn away the loss of burning area is compensated by the increasing area of the cylindrical part. This profile is simple to cast, and is effective in producing an almost constant mass flow rate. Type (c) produces a perfectly flat thrust profile, because burning takes place both on the outer surface of the inner rod and on the inner surface of the outer cylinder. The decrease in burning area of the outer surface of the rod is exactly compensated by the increasing burning area on the inner surface of the cylinder. This type of grain profile is difficult to manufacture and sustain because of the need to support the rod through the hot gas stream. It is not used for space vehicles. The final example (d) shows how an exotic profile can be used to tailor the thrust profile for a particular purpose. The narrow fins of propellant initially produce a very high surface area, and so the thrust is initially very high. Once they have burned away then a low and slowly increasing thrust is produced by the cylindrical section. When the diameter of the burning cross-section is large, the area changes more slowly than in the initial stages. Such a profile may be useful for strong acceleration followed by sustained flight.

Ambient temperature has a significant effect on the rate of burning and hence on the thrust profile. At first it may seem surprising that this has not arisen for liquid-fuelled rockets. However, they are much less sensitive to ambient temperature, because the temperature of the propellant and the supply rate are determined by the conditions in the combustion chamber and not by outside effects. For the solid propellant rocket this is not the case. The rate of evaporation of the combustible material from the burning surface of the grain depends on the rate at which the material is heated. This depends both on the rate of supply of heat from the combustion (which we have already dealt with) and on the temperature of the grain itself. If it is cold then more heat has to be supplied to reach evaporation point. The grain is massive, and is itself a good insulator, which means that during waiting time on the launch pad, or in space, it can slowly take up the temperature of its surroundings. This will not change appreciably over the short time of the burn because of the large heat capacity of the mass, and its good insulating properties. The burning rate will therefore shift, depending on the temperature of the grain. Variations of as much as a factor of two between $-15°C$ and $20°C$ have been reported. This affects the thrust profile, which could be a serious matter. It appears that the same factor which affects pressure sensitivity—β in the pressure index—also affects temperature dependence. Small values of β are beneficial here, and specific additives can also reduce the temperature effect. Even so, solid motors should not be used outside their specified temperature limits—particularly for launchers, for which a predictable thrust profile is very important.

We recall from Chapter 1 that, for orbital manoeuvres, the ultimate velocity of the vehicle depends on the exhaust velocity and mass ratio, and not on the thrust profile. Provided that the total impulse produced by the motor is predictable, the exact thrust profile is not important. Active temperature control of a solid motor in space would require far too much electrical power. But given the above argument, variation in thrust profile due to temperature changes is less important for this application.

4.4 INTEGRITY OF THE COMBUSTION CHAMBER

The combustion chamber of a liquid-fuelled engine is rather small. It is just big enough in diameter to allow proper mixing, and long enough to allow evaporation of propellant droplets. The combustion chamber of a solid motor is also the fuel store, and is large. In addition, since high thrust is usually the main requirement, the throat diameter is larger. The pressures experienced by each of them are about the same in modern rockets—about 50 bar. However, designing a large vessel to accommodate high pressure and high temperature is much more difficult than the equivalent task of designing a smaller vessel. The skin has to take the pressure, and as the diameter increases the thickness has to increase; and because of the large surface area this has a major effect on the mass. In general, high-tensile steels are used. 4SCDN-4-10 high-strength low alloy steel is used for the Ariane 5 boosters.

4.4.1 Thermal protection

The walls of the vessel cannot be cooled by the propellant as in the liquid-fuelled engine, and this imposes a considerable difficulty. As in the case of liquid-fuelled combustion chambers, the temperature of combustion is much higher than the softening point of most metals. The combustion products cannot be allowed to contact the walls for any extended period, or disaster will result. The best solution is to bond the propellant to the walls and to cover the remaining inside surfaces with a refractory insulating layer. This technique is known as *case bonding*, and is used in most modern solid motors. The grain burns only on its inside surface, so the propellant acts as an insulator. Boosters are normally used only once, and so any residual damage caused to the walls when the propellant is exhausted is not important. In fact, a thin layer of propellant usually remains after burn-out, due to the sudden drop in pressure, which extinguishes the combustion. Where particular care is required on manned missions, and for potentially reusable casings, a layer of insulating material is also placed between the grain and the casing before it is bonded in.

The lack of any active means of cooling for solid rocket components would make them unusable if the time factor were not important. The motor has only to operate for a short time, and after this time it does not matter if components exceed their service temperature, although it is, of course, important if they are to be reused. So, provided the fatal rise in temperature of the casing or the throat of the nozzle can be delayed till after burn-out, then the motor is perfectly safe to use. The means for doing this were developed and used in the early space programme, for atmospheric re-entry. The conditions and requirements are the same: to keep the important parts cool, for a limited time, against a surface temperature higher than the melting point of metals. The method used is called *ablative* cooling. The surface is, in fact, a composite structure (Figure 4.3). Furthest removed from the source of heat is the metallic structural component, which provides the strength and stiffness if it is kept cool. This is covered with many layers of non-metallic material, which have a dual purpose. Undisturbed, they provide good heat insulation, but when exposed to the full effects of the hot gases they evaporate slowly, or *ablate*. This process extracts heat of vaporisation from the gas layers nearest the surface, and forms an insulating cool gas layer analogous to that provided by film cooling in a liquid rocket engine. The materials used are combinations of silica fibres, phenolic resins and carbon fibres. The material is refractory, but because it is fibrous and flexible it does not crack, and retains its strength and integrity even while being eaten away by the hot gases. Needless to say, a crack in the insulator would allow hot gases to penetrate to the wall. The development of these ablative insulators was a vital step in the development both of solid motors and re-entry capsules. The time factor is of course important. The process works only for a certain length of time, after which the heat reaches the structural material and the component fails. The time can be extended by including one or more *heat sinks* in the construction. These are thick pieces of metal with a high thermal capacity, which are used locally to slow down the rise in temperature of a sensitive component. They cannot be used for casings, but

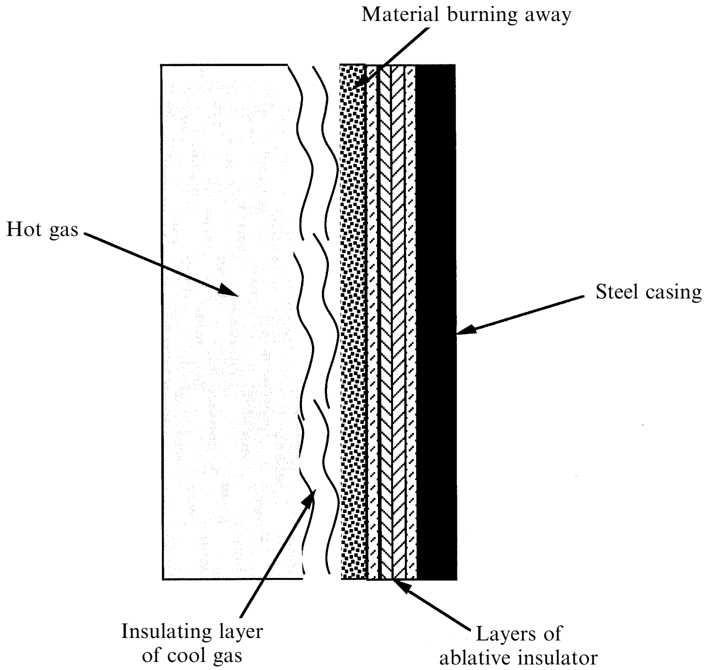

Figure 4.3. Thermal protection.

are often used in nozzle throats in which the sensitive structural components are smaller.

The propellant is generally cast or extruded into the required shape, and is then inserted into the chamber and bonded to the wall. The end faces of the charge—which should not burn—are coated with an insulating inhibitor, and the other surfaces of the chamber are insulated as described.

These activities are much easier if the motor is made in sections, and large ones may consist of several identical cylindrical sections as well as the top cap and the rear section containing the nozzle. The grain is cast in identical forms to match the sections, and each form is separately case-bonded into its section. For very large boosters the grain is cast directly into the casing section after the insulation is installed. The sections are then joined together to make the complete booster. This technique is not used for third stages or orbital change boosters in which the mass ratio advantage of a quasi-spherical shape for the motor is paramount; here the grain is formed as a unit and bonded into the case. These types of motor often have carbon fibre reinforced plastic (CFRP; called graphite reinforced plastic (GRP) in the United States) walls to improve the mass ratio.

4.4.2 Inter-section joints

The joints between sections have to be gas tight, and they also have to transmit the forces arising from the high thrust of the boosters. The combustion pressure of

50 bar is sufficient to cause some deformation of the cylindrical sections, and the joints must be proof against this. The forces involved are testified to by the large number of fasteners obvious when looking at a booster. Each section case is a cylinder with the wall as thin as possible (about 12 mm) to minimise mass. At each end there has to be a sturdy flange to take the fasteners and to properly transmit the forces to the cylindrical wall. Turning the whole section from solid material is a safe approach, but is costly, and other methods of forming the flanges—such as flow turning—can be employed.

There are two kinds of joint between sections: the *factory* joint and the *field* joint. The factory joint is assembled before the charge is installed, and results from the need to make up large booster casings from steel elements of a manageable size. These joints can be protected by insulation before the grain is installed, and are relatively safe. The field joint is so called because it is made 'in the field'—that is, at the launch site—and is used because of the impossibility of transporting and handling a complete booster. Field joints allow the booster to be assembled from ready-charged sections, more or less at the launch pad. They have two safety issues: they are made under field conditions away from the factory; and they cannot be protected with insulation in the same way as a factory joint, because the two faces of the propellant charge come together on assembly, and access to the inner surface of the joint is impossible.

The simplest pressure seal is an O-ring located in a groove in one flange, and clamped by the surface of the mating flange. This is a well tried and reliable seal, but requires very stiff and heavy flanges otherwise the flexing of the structure under thrust could open the seal. Organic seals like O-rings are generally not resistant to high temperature, and should be protected. This is all the more difficult because at the junction of two sections in a large motor, such as a field joint, the grain is not continuous, so that it is possible for hot gases to reach the intersection joint in the casing. In general, for a simple solid-fuelled rocket the thrust acts axially, and so the loads on the joints are even and the effect is to close the joint even more firmly. For strap-on boosters there is the possibility of a bending load caused by the asymmetry of the structure. Simple face joints are used quite safely in solid-fuelled rockets, although with strap-on boosters a different joint is required.

To reduce mass and to give some protection from flexing, overlap joints are used between sections. By overlapping the joints the pressure in the motor compresses the O-ring seals, while flexure can cause only small transverse movements of the joint faces.

It is essential that the casing is gas-tight at the operating pressure of 50–60 bar, as any leakage would allow hot gas to reach sensitive components. Being organic, O-rings cannot withstand even moderately high temperatures, and should be kept within a reasonable range around room temperature. They cannot be exposed to the hot combustion products, and a thermal barrier has to be placed between the hot gases and the seal—it does not form a seal itself and is simply there to protect the O-ring. Typically this thermal barrier is a flexible silicone sealant or thermal 'putty'. Provided the seal is intact, then a small amount of gas leaks past the thermal protection and equalises the pressure. This small amount of gas cools and does no harm to the pressure seal. While the thermal protection remains intact, the O-rings are safe, but if either the sealant leaks or the O-ring leaks then the joint can fail. If the sealant leaks then hot gas can reach the O-ring and cause it to scorch or melt. If the O-ring leaks, then the cool gas supporting the sealant leaks away, and the sealant flows, which again may allow hot gas to reach the O-ring.

This kind of failure caused the loss of *Challenger*. An external temperature a little above 0° C may have caused the O-ring in the aft field joint to become stiff, and unable to follow the movements of the opposing flanges so as to remain leak-tight. Hot gas leaked past the thermal putty and destroyed part of the O-ring. High-temperature combustion products then escaped and damaged the main propellant tank containing liquid hydrogen and liquid oxygen. The loss of the Space Shuttle and of seven lives then became inevitable.

The joint was redesigned after the disaster, both to eliminate the thermal putty by replacing it with a rubber 'J' seal, and to use three O-rings in series. The 'J' seal is configured to seal more tightly when the internal pressure rises. It is interesting to contemplate that rubber and silicone rubber—both of which are used domestically—can, under the right circumstances, resist the most undomestic temperatures and pressures.

4.4.3 Nozzle thermal protection

The nozzle and throat are protected from the heat of the exhaust by using similar techniques to those used to protect the casing. Here the problem is more severe because of the high velocity of the exhaust gases. The main structure of the throat and nozzle is made of steel, but many layers of ablative insulator are applied to the inside. A heat sink is also used at the throat to reduce the transfer of heat to the steel structure. Most of the thrust is developed on the walls of the nozzle, and so the structure needs to remain within its service temperature until burn-out. Ablation, heat diffusion into the heat sink, and the thermally insulating properties of the throat lining keep the steel cool long enough to do its job. After burn-out it does not matter if the outer structure becomes too hot. It is worth mentioning that without such a lining the steel would reach its melting point in less than one second, but the lining prolongs this by a factor of about 200.

4.5 IGNITION

Solid motors are used for two applications, both of which require ignition which is as stable and reliable as for a liquid-fuelled engine. The main propulsion unit for a rocket stage or satellite orbital injection system requires timely ignition in order to achieve the eventual orbit. A booster forms one of a group of motors which must develop thrust together. In all cases a pyrotechnic igniter is used. Pyrotechnic devices have an extensive and reliable heritage for space use, in a variety of different applications. To ignite a solid motor, a significant charge of pyrotechnic material is needed to ensure that the entire inner surface of the grain is simultaneously brought to the ignition temperature: 25 kg of pyrotechnic is used in the Ariane 5 solid boosters. It is itself ignited by a redundant electrical system.

4.6 THRUST VECTOR CONTROL

For orbital injection, thrust vector control is not normally needed, as the burn is too short to require the spacecraft to change its course while the motor is firing. Thrust vector control is essential for solid boosters because their thrust dominates the thrust of a launcher for the first few minutes, and so course corrections require the booster thrust to be diverted.

The technique of liquid injection applicable to small solid propellant launchers cannot produce sufficient transverse thrust to manoeuvre a large launch vehicle like the Ariane or the Space Shuttle. It requires a moveable nozzle, mounted on a gimballed flexible bearing so that it can be traversed by about 6°C in two orthogonal directions. Large forces are needed to move the nozzle quickly, and the motion is contrived using hydraulic rams controlled by electrical signals from the vehicle's attitude control system. The flexible joint has to be protected from the heat of the combustion products by flaps of material similar to that used to insulate the joints and casing.

4.7 TWO MODERN SOLID BOOSTERS

As current examples, we shall describe two important solid boosters: those of the Space Shuttle and of Ariane 5. Their similarities reflect both the similarity in application and the relative maturity of the technology.

4.7.1 The Space Shuttle SRB

Table 4.1 shows that the SRB is about twice the size of the Ariane 5 MPS. It develops a thrust of 10 MN. The casing consists of eight steel segments flow-turned with the appropriate flanges. The fore and aft sections are fitted with the igniter and nozzle respectively. The casing sections are joined in pairs by factory joints which are then thermally protected by thick rubber seals, and the inner walls of the casing are protected by insulating material to which the propellant will be bonded. The propellant is mixed and cast into each pair of segments, with a mandrel of the appropriate shape to form the hollow core. The booster pairs are thrust-matched by filling the appropriate segments of each pair together, from the same batch of propellant. Insulating material and inhibitor are applied to the faces, which are not to burn. The filled sections are then transported to the launch site where they are assembled, using the field joints, into the complete booster.

The nozzle is made of layers of glass and carbon fibre material bonded together to form a tough composite structure which can survive temperatures up to 4,200 K. This composite is then bonded onto the inside of the steel outer cone, which provides the structural support. Rings attached to the cone provide the anchorage for the hydraulic actuators. Inboard is the flexible joint which allows the nozzle to be tilted.

Table 4.1. Two modern solid boosters.

	SRB (Space Shuttle)	MPS (Ariane 5)
Thrust (individual)	10.89 MN	5.87 MN
Thrust (fractional)	71% at lift-off	90% at lift-off
Expansion ratio	11.3	≈10
Exhaust velocity	2,690 m s^{-1}	2,690 m s^{-1}
Temperature	3,450	3,600
Pressure	65 bar	60 bar
Total mass	591 T	267 T
Propellant mass	500 T	237 T
Dry mass	87.3 T	30 T
Burn-time	124 s	123 s
Charge shape		
Upper section	11 point cog	23 point cog
Middle section	Truncated cone	Cylinder
Lower section	Truncated cone	Truncated cone
Propellant	Ammonium perchlorate 69.6%	Ammonium perchlorate 68%
	Aluminium powder 16%	Aluminium powder 18%
	Polymeric binder 14%	Polymeric binder 14%
	Additive iron oxide 0.4%	
Factory joints	4	6
Field joints	3	2
Length	45.4 m	27 m
Diameter	3.7 m	3.0 m
Casing	Steel 12 mm	4SCDN-4-10 steel

In addition to the propellant in its casing, each booster has a redundant hydraulic system to displace the nozzles by $\pm 8°$. There are two actuators, one for each orthogonal direction of displacement. The nose cap of the booster also contains avionics and recovery beacons.

The casing segments are reused. After recovery from the sea the casings are cleaned, inspected and pressure tested to ensure they are sound. Some are discarded because of damage, which is mostly caused by impact with the ocean. The segments are then dimensionally matched, as the combustion chamber pressure can permanently increase the diameter by several fractions of a millimetre. Once cleared for reuse they are refilled with propellant. The sections are rated for 20 re-uses.

The propellant core shapes are intended to produce a ‘sway-backed’ thrust curve rather than a constant thrust. This produces a period of lower thrust some 50 seconds into the flight, while the vehicle is passing through the region of maximum dynamic pressure (see Chapter 5). This is the period when the product of velocity and air pressure is at a maximum and when the possibility of damage by aerodynamic forces is greatest, and the risk is minimised if the thrust is reduced for a short time. The Space Shuttle main engines are also throttled back for this period.

At launch the two boosters provide 71% of the total thrust, effectively forming the first stage. Their short nozzles are adapted for sea-level operation, and the huge mass flow rate—almost five tonnes per second—provides the high thrust necessary for lift-off.

4.7.2 The Ariane MPS

The MPS (*Moteur à Propergol Solide*) is similar in many respects to the SRB, and about half the size. It has a 3-metre diameter compared with the 5 metres of the SRB. The thrust, at nearly 6 MN, contributes 90% of the Ariane 5 lift-off thrust. The propellant is very similar, with percent level differences in the composition of ammonium perchlorate oxidant and aluminium powder fuel. The binder—a poly-butadiene rubber—may well be somewhat different in detail from the Thiokol rubber used for the SRB. The additives are not specified.

The booster consists of seven sections (Figure 4.4). The forward section contains the igniter and the aft section the nozzle, and both the forward and aft domes are protected with ablative insulating material. The forward section, which has a rough forged bulkhead to contain the pressure, is charged with 23 tonnes of propellant with a cog-shaped inner void; this is done in Europe, prior to shipment to the launch site at Kourou. The remaining segments are charged, at the launch site, with locally manufactured propellant. The middle segment consists of three casing elements pinned together with factory joints. The MPS joints are overlapping joints with transverse pins, and the inner walls are protected with silica and Kevlar fibre insulator (GSM 55 and EG2) before the propellant is cast into the segment. This thermal protection covers the factory joints. The mandrel for the casting produces a shallow truncated cone shape to the grain void. The lower segment is constructed in the same way, and the mandrel here produces a steeper conical form to the void, opening towards the nozzle. The forward grain burns away in the first 15 seconds, while the two lower grains, each of 107 tonnes, burn for 123 seconds. This produces the 'sway-backed' thrust profile which reduces the thrust around maximum dynamic pressure.

The nozzle is made of composite materials incorporating carbon–carbon and phenolic silica materials. It is supported by a lightweight metallic casing, to which the 35-tonne servoactuators are connected by a strong ring. The nozzle can be traversed by $\pm 6^\circ$ for thrust vector control.

The boosters are recovered after launch, but presently there is no plan for reuse of casing segments. The main purpose of recovery is post-flight inspection of seals and components to ensure that they are functioning correctly throughout the flight. Post-flight inspection, for example, studies the seals and thermal protection to confirm that it remains intact. Some 10–12 mm of the aft dome protection ablates away during flight, in the region directly adjacent to the nozzle. The throat diameter is 895 mm, and 38 mm of the thermal protection was ablated away during a test firing. This is allowed for in the design, and demonstrates how thermal protection is provided by this technique.

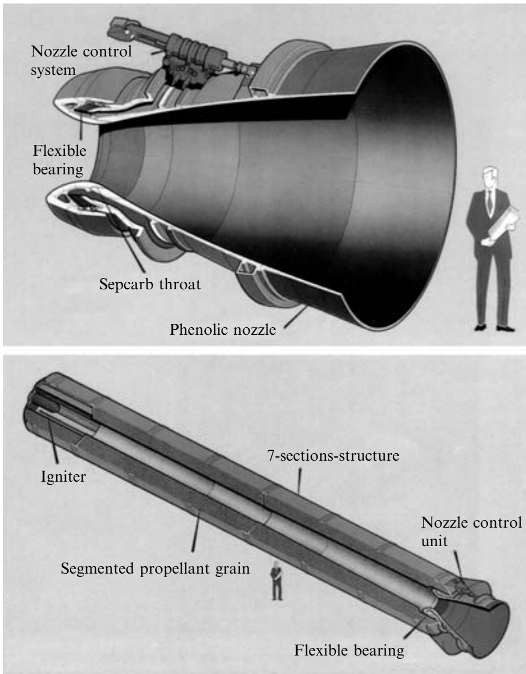

Figure 4.4. The Ariane MPS solid booster.

The Space Shuttle and Ariane 5 represent the state-of-the-art heavy launcher capability presently available. The booster technology is mature, as evidenced by the similarity in techniques. Solid propellant boosters of this power represent the best way of increasing the in-orbit payload capability of large expendable launchers. Because of their tough construction and early burn-out, boosters are eminently recoverable, and their reuse is a factor in the economics of launchers. The Space Shuttle has shown the way in reuse, but caution still exercises a strong restraint, as the cost of quality assurance needed to ensure safe reuse is a significant fraction of the cost of new components. The next step in making space more accessible may come from the development of fully reusable launchers and the single stage to orbit. For these, liquid propellants are appropriate, but it is likely that hybrids and intermediate developments will still utilise solid propellant motors. Small launchers make considerable use of all solid propellant propulsion, and this is a market where cost and reliability indicate its continued use.

4.8 HYBRID ROCKET MOTORS

While the solid rocket motor contains both fuel and oxidant in the charge or grain, the hybrid motor only has the fuel in the charge; the oxidant is introduced as a liquid

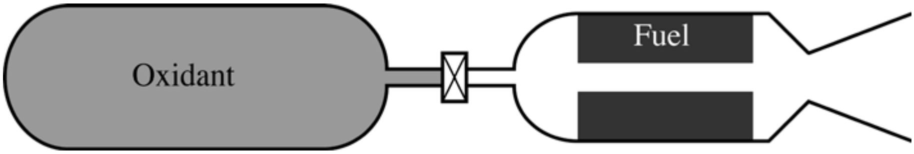

Figure 4.5. Schematic of a hybrid rocket motor.

propellant, injected and atomised just as in the case of a liquid-fuelled rocket engine (Figure 4.5). This kind of motor was investigated historically, but was then neglected, except for a few specialised applications. More recently it has come to prominence for two specific applications. Amateur rocket builders find its safety and simplicity attractive, and there are many commercially available hybrid motors available for this purpose; many schoolboy rocket engineers are cutting their teeth on the hybrid motor. There are however also applications of hybrid motors that are suitable for commercial, and even human spaceflight. The most spectacular new use for hybrid motors was on the recent X-prize winning, manned sub-orbital vehicle, SpaceShipOne, which achieved an altitude of 120 km. with a human pilot. This vehicle was entirely developed and launched by private industry. Again, the relative safety and simplicity of the hybrid motor was the key factor in its selection. The safety and reliability of the hybrid motor come about because of the separation of the fuel and oxidant, both in location and in physical state: the one in liquid form, the other a solid. Where, in the case of a solid motor, the intimate mixture of powdered oxidant and fuel only requires ignition in order to burn, the hybrid motor requires two separate steps to burn: first, the solid fuel must be heated and vaporised, and then the oxidant has to be introduced. This makes the possibility of an accident much less likely. Indeed, accidental contact between cold fuel and oxidiser has no effect, because they are in different material states. The second obvious advantage of the hybrid motor over the solid motor is the fact that it can be shut down, just as a liquid-fuelled engine can, simply by closing the valve supplying the liquid oxidant. These two features make the hybrid motor suitable for human space flight, and of course for amateur use.

4.8.1 Hybrid motor history

Sergei Korolev, of course, appears in the first recorded launch of a hybrid rocket on August 3rd, 1933. The GIRD 09 rocket developed a thrust of 500 N and reached an altitude of 1500 metres. This was in fact the first Russian flight of a liquid-fuelled rocket. The oxidant, liquid oxygen, was pressure-fed by its own vapour and so the system was very simple; this technique is still used in modern hybrid motors. The solid fuel was a mixture of petroleum with collophonium, a natural gum-resin. Rocket societies in the German-speaking world, and in the United States, continued small-scale trials, including tests conducted by Herman Oberth near Vienna in 1938–9 using liquid oxygen and a tar–wood–potassium nitrate fuel. Others in this period used wood and coal as the fuels. After WWII, the first recorded use of polyethylene as the fuel was in 1951, in the US; hydrogen peroxide was the oxidant. Then developments continued in the US, France, and Germany, with many types of fuel and oxidant; but the rockets themselves were small. The introduction of metal powders (lithium) to the fuel, and exotic oxidants like mixtures of fluorine and liquid oxygen, led to some higher thrust motors in the US, notably by the United Technologies Corporation (UTC); these produced tens of kilo-Newtons of thrust. This led to the Dolphin sea-launched hybrid rocket (1984), which had a thrust of 175 kN using polybutadiene rubber as the fuel, and liquid oxygen as the oxidant. The AMROC company then tested a number of high-thrust motors up to 324 kN. again using polybutadiene as the fuel, with liquid oxygen. The aim was to develop a spacecraft launch vehicle. However, following the failure of the engine test vehicle SET-1, which had a single H1500 engine developing a vacuum thrust of 931 kN and carrying 25 tonnes of propellant, AMROC ceased activities, and it was the company SpaceDev that took over the development of large hybrid motors. The SpaceShipOne motor, delivered by SpaceDev, developed 74 kN of thrust and burned 2.4 tonnes of propellant. This was really an upper-stage motor as the ship was carried to altitude by a specially developed aircraft.

The attraction of the hybrid motor has encouraged entrepreneurs in space flight who are outside the main space agencies. While its simplicity is at first sight beguiling, there are significant development issues and disadvantages, which will be examined below. As an amateur rocket motor it is unchallenged: it is simple and safe to use and build, the propellants can be obtained freely, and in almost all configurations it works pretty well.

4.8.2 The basic configuration of a hybrid motor

The basic configuration of the hybrid thrust chamber is broadly the same as for a solid rocket motor. The fuel is cast into the casing, and there is a nozzle to develop the thrust from the hot gas produced. The same consideration has to be given to the protection of the casing, and any joints, from the hot gases produced by combustion, and the same solutions have to be applied. Thus, the grain is cast into the casing, with no parts of the casing wall accessible to the hot gas; those areas not protected by the fuel grain have to be covered with ablative coolant material. The design of the casing has to take into account the pressure developed during firing, and the requirement to optimise mass-ratio by keeping the dry mass of the motor low. In this respect, the hybrid and solid motors have identical requirements, however the grain cross-section, and the specific details of the combustion chamber, are different and reflect a fundamental difference in the operation of the hybrid motor when compared with the solid motor.

4.8.3 Propellants and ignition

In all but a few experimental motors, the oxidant is the liquid propellant and the fuel is solid. Many combinations have been tried, including coal and wood for the fuel, and most of the common liquid oxidants. For modern motors, the fuel is either a common plastic material like polyethylene or polystyrene, or the polybutadiene rubber that is the bonding agent in most modern solid propellant rockets. For the oxidant, the two most commonly used are liquid oxygen and nitrous oxide.

The engineering challenges for the fuel are the same as for the solid rocket: the propellant has to be bonded to the case, and must not move or break up during thrusting or burning. This is the reason polybutadiene rubber is used, because the same engineering as used for solid-fuelled motors will guarantee integrity of the grain. As will be seen below, combustion is somewhat different from that associated with an integrated solid propellant, and the grain is somewhat shorter compared with the length of the casing to allow open volumes at the forward and aft ends, in order to improve combustion.

The oxidant is stored in the single tank, and delivered to the combustion chamber/casing via a single pipe and valve. Almost all hybrid motors are rather low-thrust, and so pressure-fed systems are the norm. This is done either by using a small additional tank of high-pressure helium, connected to the oxidant tank via a valve, or by using the vapour pressure of the oxidant itself. This latter works well with liquid oxygen and nitrous oxide, for all but the highest thrust engines. In the case of nitrous oxide, the liquid can be stored at room temperature, under a vapour pressure of 5238 kPa or 52 bar. For liquid oxygen, the tank must be vented and the vents closed at launch. The use of vapour pressure to feed the oxidant into the combustion chamber limits the combustion chamber pressure and hence the thrust, but the reduction in complexity, and in the dry mass of the motor, is a significant advantage. For higher thrust engines, turbo-pumps would be required. Powering these would require the use of an auxiliary gas generator and a liquid fuel for it, or a bleed-off of combustion gases from the exhaust nozzle. These options are not attractive when simplicity and safety are the goals, and so, very high thrust hybrid motors are not currently under development.

The liquid oxidant requires to be atomised and vaporised before combustion can take place, and so the injectors are very similar to those used in liquid-fuelled rocket engines, with the exception that there is only one propellant to deliver. The impinging jet type is used for nitrous oxide, to encourage rapid atomisation. To avoid combustion instabilities, the pressure drop across the injector should remain high, and this limits the combustion chamber pressure, for a vapour pressure fed system.

For ignition to take place there must be sufficient of the solid fuel in the vapour phase for combustion to be possible. This requires significant thermal energy and so the most reliable approach is to use a pyrotechnic igniter, similar to those used to start solid rocket motors. A significant quantity of pyrotechnic material is ignited by electrical means, and this shoots a stream of burning material down the central bore of the fuel grain; this process ensures that the whole surface begins to evaporate, and so to provide the necessary fuel in vapour phase for combustion to start. Pyrotechnic ignition precludes the possibility of a re-start, while of course the motor can be shut down easily, by closing the oxidant valve.

Other ignition processes are used, especially in the amateur field. Where oxygen is available, some gaseous oxygen can be admitted to the combustion chamber together with combustible material. If this combustible material is gaseous, then spark ignition can be used, and, in principle, a re-start would be possible. With liquid oxygen as the oxidiser, it is comparatively simple to arrange for some of the vent gas to be admitted to the combustion chamber. But a gaseous combustible partner must be provided from an auxiliary tank; vapour phase oxidant and fuel are needed for spark ignition. Another form of electrical ignition uses the oxidant gas, and some electrically heated solid material, which will ignite in the oxidant atmosphere once it is hot enough. This is not pyrotechnic, and will work with any gaseous oxidant, but it is still a one-shot device, and precludes a re-start.

4.8.4 Combustion

Combustion in a hybrid motor differs considerably from that in either a solid motor or a liquid-fuelled engine. This comes about because the fuel and oxidant are in different physical states. In a liquid-fuelled engine the propellants are introduced to the chamber, via the injectors, in a fine spray; the droplets evaporate and the vapours mix intimately and combustion takes place. The heat for evaporation and initiation of combustion comes from the hot gases further down the combustion chamber. In the case of a solid-fuelled rocket motor, the fuel and oxidant are intimately mixed in the solid phase and cast into the grain. Heat from the already burning propellant above the grain surface, warms it by radiation and convection, evaporating the intimate mixture of fuel and oxidant, which immediately ignites and burns. In both cases, it is easy to control the oxidant–fuel ratio in the combustion zone either by the relative rate at which liquid propellants are delivered to the chamber, or by the pre-mixed ratio in the solid propellant grain. The combustion is complete and the hot gases reaching the nozzle are at the design temperature.

The hybrid motor is different (Figure 4.6). The heat from the burning gases melts and evaporates fuel from the fuel grain, so that closest to the grain surface there will be a layer of pure fuel vapour. In the centre of the port or bore, down which the oxidant vapour is flowing, there will be pure oxidant. In between there is a layered system with pure fuel closest to the grain surface, which is gradually diluted with oxidant as the distance from the grain surface increases. At some point the concentrations of fuel and oxidant will allow combustion, so there will be a burning layer which lies somewhat above the grain surface, where all the combustion takes place; the concentration of oxidant will increase past this layer until the pure oxidant zone is reached. These layers are of course all moving axially down the port, and so combustion takes place in a rapidly moving cylindrical zone somewhere between the grain surface and the axis of the port. This behaviour is different from the solid motor where burning takes place close to the grain surface and the gases above this layer are just combustion products.

For complete combustion to take place, the zone of combustion products should fill the whole core of the grain by the time the nozzle is reached. This is unlikely to happen, for finite grain lengths because there will always be some pure fuel vapour, which has evaporated from the last few centimetres of the grain, and, for oxidant supply rates required for complete combustion, there will be corresponding oxidant-

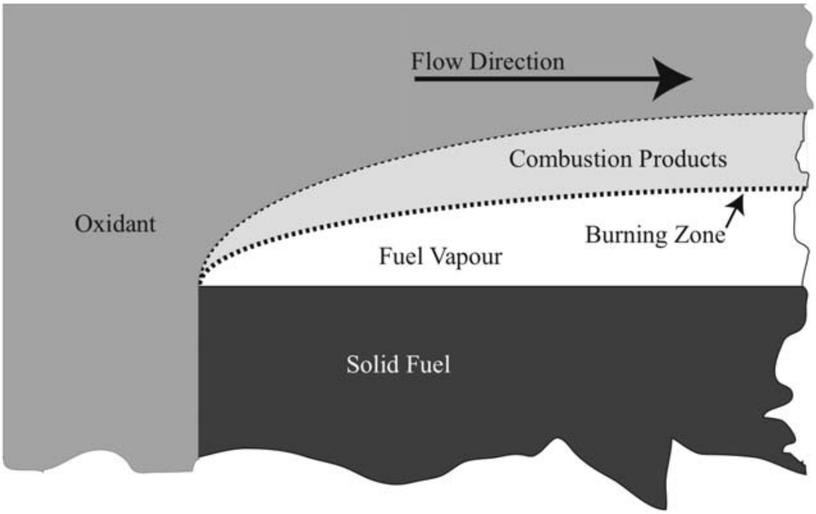

Figure 4.6. Evaporation and combustion in a hybrid rocket motor. Oxidant enters from the left and passes down the bore. The composition ranges from pure fuel vapour near the charge to pure oxidant vapour in the bore.

rich gas entering the nozzle with it. The solution is to terminate the grain early, leaving a clear volume, about equal in length to the diameter of the chamber. Vortices develop behind the grain ends, and ensure mixing of the remaining vapours; the space is long enough to allow time for combustion to complete, before the gases enter the nozzle.

There is a similar empty volume at the forward end of the chamber; this has a similar purpose: it is to ensure that the oxidant is completely vaporised before it encounters the hot surface of the grain. If liquid oxidant were to do so, there would be sporadic and uneven burning of the grain, leading to pitting and rough burning.

4.8.5 Grain cross-section

The mass flow rate of a hybrid engine depends on different parameters from those in a solid motor. For both, the mass flow rate depends on the rate at which the grain surface burns away—the recession rate—multiplied by the burning area. For the solid motor, the recession rate is a strong function of the chamber pressure. There is always enough oxidant present to burn the fuel released by the recession of the surface, and the recession of the surface simply depends on the heat input provided by the combustion; this depends on the combustion chamber pressure. The general formula is given in Section 4.3.3. For the hybrid motor, the recession rate depends not just on the heat input, but on the available oxidant; since the evaporated fuel zone is always oxidant-starved, the recession rate depends, not on the pressure, but on the rate of supply of the oxidant. This is familiar to us in the case of a domestic wood or coal fire: the burning rate depends on the air supply rate; fanning the fire increases the burning rate. Thus, the behaviour of the two motors is very different. The solid motor propellant needs to have its composition adjusted, with additives like iron oxide, to keep the exponent of the pressure term in the safe region—to avoid either extinction, or the opposite. For the hybrid motor, the risk of detonation is absent, and the motor can even be throttled, by reducing the oxidant supply, so burning is stable over a wide range of pressures. This is both an advantage and a disadvantage: the motor is much safer, and works over a wide range of operating parameters; but in general obtaining a high recession rate, and a high mass flow rate, is difficult. This is because the fuel has to be evaporated and mixed with oxidant before it can burn, and this is a relatively slow process. Over most of the grain bore there will be a considerable volume of unmixed oxidant and unmixed fuel vapour (see Figure 4.6). This means that the energy supply, required to heat and melt the solid fuel, is much smaller than in the case of solid propellant where the evaporated propellants immediately mix and burn.

The solution is to ensure a large surface area over which the processes can occur; this is done by having several parallel bores in the grain, typically it is divided into four bores, with the fuel walls between them forming spokes to support the grain. (Since the mass flow rate is governed by the oxidant supply, there is no need to control the thrust by special grain shapes, as in the solid motor.) The increased surface area gives increased mass flow rate, but it is not enough to bring the hybrid motor close to the huge thrusts which can be developed by solid motors. Thus high thrust has been traded away for simplicity, safety, and robust operation—this is why almost any hybrid motor, put together using the simple rules given above, will work.

4.8.6 Propulsive efficiency

The efficiency of a rocket in terms of spacecraft propulsion depends on the exhaust velocity and the achievable mass ratio; the thrust is less important, except for the first stage of a launcher. The exhaust velocity of a well-designed hybrid motor can be the same as a solid motor, or even a liquid oxygen–kerosene engine. However, for the exhaust velocity to be optimum throughout the burn, the oxidant flow has to be regulated. The surface area exposed to heat increases, as the grain burns away and releases more fuel, this has to be matched by the oxidant supply, if the supply is set up just to match the fuel available to burn at the beginning, then the mixture will become more fuel-rich with time, reducing the temperature and the exhaust velocity. Increasing the oxidant supply can mitigate this, but it requires a control system, which negates some of the simplicity of the engine. Thus, the average exhaust velocity may be considerably lower than the optimum. The thrust, as we have seen, is not high; but this does not matter for upper stages, or in-orbit propulsion. The mass ratio is unfortunately not very good. This comes about for two reasons. The casing or combustion chamber is considerably larger, and therefore heavier for a given amount of fuel, than the corresponding solid motor. This is because of the forward and aft mixing regions, which are empty of fuel, and because of the large, multiple, bores, that are necessary in order to generate sufficient thrust; a lot of the casing volume is empty of fuel. This increases the dry mass of the rocket. There is another disadvantage of the multiple bores: the structure and integrity of the fuel, and its bonding to the casing, break down much earlier than for a single cylindrical bore. In the case of a single cylindrical bore, the propellant burns away evenly, and the bond to the casing is the last thing exposed; where there are multiple bores, the walls of fuel between the bores are unsupported, and eventually may break up. This can result in pieces of the solid fuel breaking away and striking the nozzle, as well as the risk that the casing becomes exposed to the hot gases, which, as we know, is fatal. To be safe, the rocket has to be shut down early, by shutting off the oxidant supply, with a significant amount of unburnt fuel remaining. This again adversely affects the mass ratio. In summary, the hybrid motor is safe, and easy to operate, but it has low thrust and a poor mass ratio compared with other types.

4.8.7 Increasing the thrust

As we have seen, the low recession rate of the fuel results in a low thrust for these motors. There are ways in which it might be possible to increase the thrust, or alternatively, to remove the necessity for multiple bores. The requirement is to increase the rate at which the fuel evaporates from the solid grain. The rate depends on the heat input to the grain, and as we have seen this is lower that the rate in a solid propellant motor, because the evaporated fuel has to encounter and mix with the oxidant in the central region of the bore, before it can burn. The burning zone is therefore at some height above the surface of the fuel grain, and the heat input correspondingly lower.

One approach to increasing the evaporation rate, for the same heat input, is to reduce the viscosity in the layer of melted fuel on the exposed surface of the grain. Fuels such as polyethylene or polybutadiene, are long-chain polymers and, the melted fuel has a high viscosity. The surface layer is relatively undisturbed by the high-velocity gases flowing across it, and evaporation happens mainly by diffusion. If, however, the viscosity of the melted fuel can be decreased, it is possible for the high-velocity gases the generate waves on the surface, and to tear off the peaks of these waves, forming and entraining fine droplets of liquid fuel, rather like sea spray. The droplets evaporate much more quickly, and the fuel supply to the burning zone increases, so increasing the thrust—as there is always unused oxidant present. The increased burning rate also increases the heat supply to the surface of the grain, and so the evaporation rate. Some success has been achieved with this method by adding a certain amount of paraffin-wax to the fuel, to lower the viscosity.

Other approaches attempt to increase the heat supply. There are two ways. The first is to add powdered metals to the fuel. These have a higher heat of combustion, and once in the burning zone, they increase the heat supply and hence the evaporation rate. This is the same technique as applied to most solid propellants, which have a significant fraction by weight of aluminium powder included. The higher heat of combustion is complemented by a denser exhaust—it contains vaporised aluminium oxide, which increases the mass flow rate and hence the thrust. Also, the oxide condenses in the nozzle into tiny particles, forming the so-called two-phase flow. Here the expansion and acceleration of the exhaust gases ignores the aluminium oxide, and so the effective molecular weight is kept low, giving a high exhaust velocity.

The other method is to add solid oxidant to the fuel. This might seem like heresy, since the whole idea of a hybrid motor is to keep the fuel and oxidant apart. However, the amount of oxidant added is not enough to enable the grain to sustain combustion by itself. What it does do is to increase the oxidant composition in the evaporated fuel vapour, and bring the lower limit of the burning zone closer to the surface of the grain. This increases the heat input to the grain and so the evaporation rate increases.

Hybrid motors have a role to play in space propulsion, where their relative safety and simplicity show advantages. The development process is to increase the thrust so that they can be used for large human launchers. This process is by no means complete at present.