CHAPTER 11
HYBRID ROCKETS

11.1 HYBRID ROCKETS VERSUS CONVENTIONAL SYSTEMS

Liquid bi-propellant systems

A liquid bi-propellant chemical rocket system is shown schematically below. Oxidizer and fuel from separate tanks are pressure-fed or pump fed into a combustion chamber where atomization, mixing, ignition and combustion takes place. Despite the apparent simplicity of the diagram liquid rockets are extremely complex. The complexity comes from the fact that the chamber pressure is usually quite high and one or both of the propellants may be cryogenic. In addition, the liquids are usually fed into the combustion chamber at very high mass flow rates requiring high performance turbopumps. Many of the most spectacular rocket failures have involved liquid bi-propellant systems.

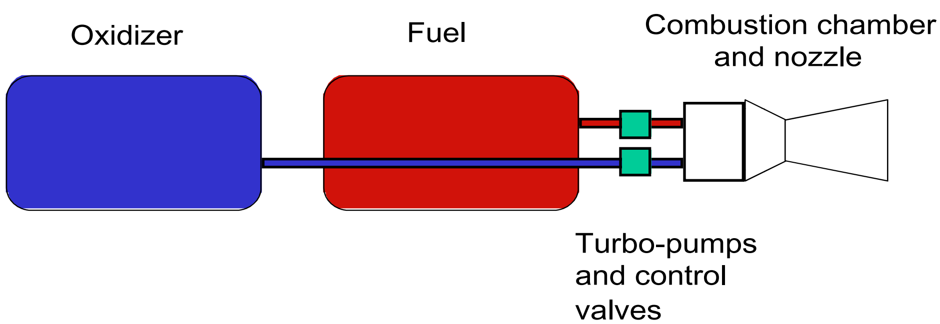


Figure 11.1 Schematic of a liquid bi-propellant rocket system

Perhaps the most widely recognizable liquid engines are the space shuttle main engines that burn hydrogen and oxygen. These engines also make use of a pre-burner where most of the oxygen is burned with a small amount of hydrogen to raise the temperature of the gases that are injected into the main combustion chamber along with the rest of the hydrogen. The hydrogen is also used to regeneratively cool the rocket chamber and nozzle. Many different oxidizers are used in bi-propellant systems. The two most popular are LOx (liquid oxygen) and N2O4. These are both very energetic oxidizers and burn readily with hydrocarbon fuels such as kerosene and alcohol as well as hydrazine (N2H4). The ideal specific impulse of kerosene burning with LOx is approximately 360 seconds.

Liquid rockets can be throttled by controlling the flow of fuel and oxidizer while keeping the ratio of oxidizer to fuel flow the same. Wide throttle ratios are somewhat difficult to achieve however because of the reduced mixing that can occur at low liquid flow rates. Liquid rockets are subject to a variety of instabilities and the design and development of a new injector and combustion chamber is an expensive multi-year process.

Solid systems

The sketch below depicts a solid rocket system. Though mechanically much simpler than liquids the solid rocket is complicated by the use of an explosive mixture of fuel and oxidizer. that involves a very complex and expensive manufacturing process. In addition solid rockets require stringent safety precautions in manufacture, handling and launch.

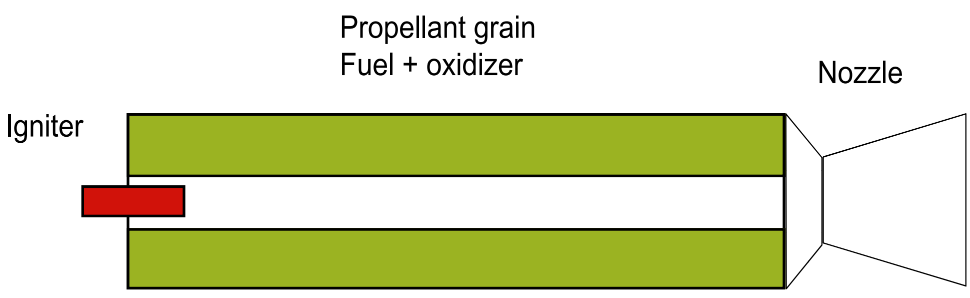


Figure 11.2 Schematic of a solid rocket motor.

The propellant regression rate for a solid rocket is proportional to the chamber pressure according to a relation of the form

$$\dot{r} = \alpha P_{t2}{}^{n} . \qquad (11.1)$$

where $n < 1$. Probably the most well known solids are the large re-usable space shuttle boosters. Each uses approximately a million pounds of propellant and produces roughly three million pounds of thrust. The fuel is mainly aluminum in a polymer binder (Hydroxyl Terminated Polybutadiene HTPB) and the oxidizer is ammonium perchlorate (AP) which is one of the more popular solid oxidizers. In general solid rockets use somewhat less energetic oxidizers than liquids and the specific impulse of solids is generally lower. The ideal specific impulse of the shuttle booster propellant is approximately 290 seconds. Recently ammonium perchlorate has been found in the ground water near many of the rocket propellant processing plants across the US and concerns have been raised about the possible environmental impact of this chlorinated compound.

Hybrid systems

The sketch below depicts a hybrid rocket. The hybrid normally uses a liquid oxidizer that burns with a solid fuel although reverse hybrids such as liquid hydrogen burning with solid oxygen have been studied.

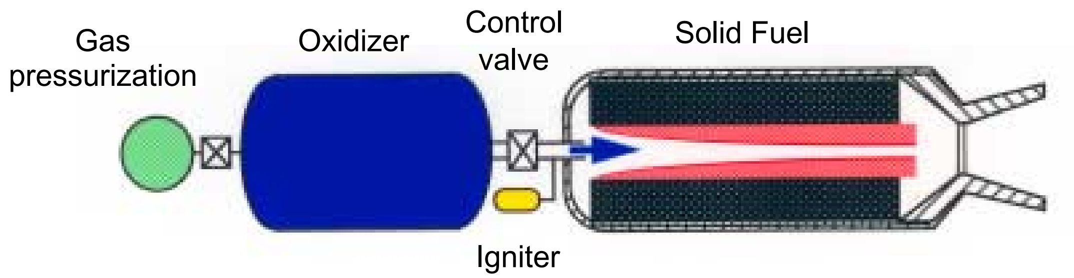


Figure 11.3 Schematic of a hybrid rocket

The fuel regression rate of the hybrid follows a relation of the form

$$\dot{r} = \alpha G_o^{\ n} \tag{11.2}$$

where G_0 is the oxidizer mass flux and the exponent is generally in the range $0.6 < n < 0.8$. The rate of fuel mass generation is

$$\dot{m}_f = \rho_f A_b \dot{r} \tag{11.3}$$

where ρ_f is the fuel density and A_b is the fuel burning area.

Over most of the useful oxidizer flux range of a hybrid the regression rate is independent of the chamber pressure. This important characteristic enables the chamber pressure to be a free variable in the motor design. Although the hybrid seems to lie somewhere between a liquid and a solid system it has advantages that are unique and not enjoyed by liquids or solids.

The hybrid is inherently safe. In the event of a structural failure, oxidizer and fuel cannot mix intimately leading to a catastrophic explosion that might endanger personnel or destroy a launch pad. The hybrid rocket requires one rather than two liquid containment and delivery systems. The complexity is further reduced by omission of a regenerative cooling system for both the chamber and nozzle. Throttling control in a high regression rate single-port hybrid is simpler because it alleviates the requirement to match the momenta of the dual propellant streams during the throttling operation. Throttle ratios up to 10 have been common in hybrid motors. The fact that the fuel is in the solid phase makes it very easy to add performance enhancing materials such as aluminum powder to the fuel. This enables the hybrid to gain an Isp advantage over a comparable hydrocarbon fueled liquid system.

The hybrid rocket has been known for over 50 years, but wasn't given serious attention until the 1960's. The primary motivation was the non-explosive character of the fuel, which led to safety in both operation and manufacture. The fuel could be fabricated at any conventional commercial site and even at the launch complex with no danger of explosion. Thus a large cost saving could be realized both in manufacture and launch operation. Additional advantages over the solid rocket are: greatly reduced sensitivity to cracks and de-bonds in the propellant, better specific impulse, throttle-ability to optimize the trajectory during atmospheric launch and orbit injection and the ability to thrust terminate on demand. The products of combustion are environmentally benign unlike conventional solids that produce acid forming gases such as hydrogen chloride.

Typically the fuel is a polymeric hydrocarbon solid such as HTPB and the oxidizer can be any of the oxidizers used with liquid bi-propellant engines. The ideal specific impulse of a LOx-HTPB propellant combination shown in Figure 11.4 is very close to that of a LOx-kerosene liquid system.

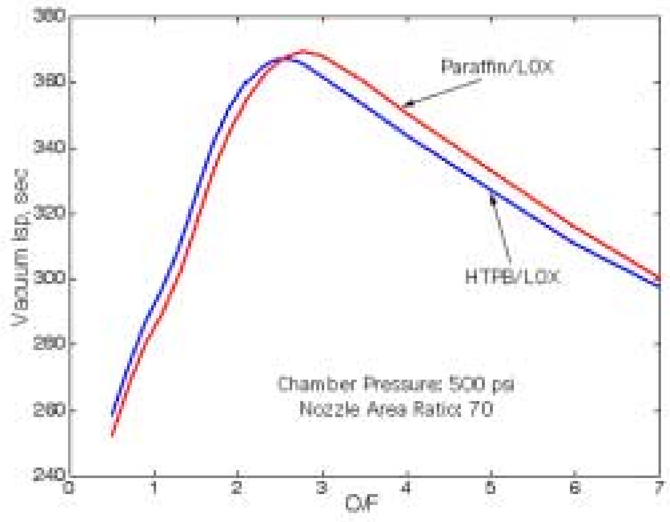


Figure 11.4 Vacuum specific impulse versus O/F ratio for two hydrocarbon fuels burning with LOx.

The main drawback of the hybrid is that the combustion process relies on a relatively slow mechanism of fuel melting, evaporation and diffusive mixing as depicted in the sketch below.

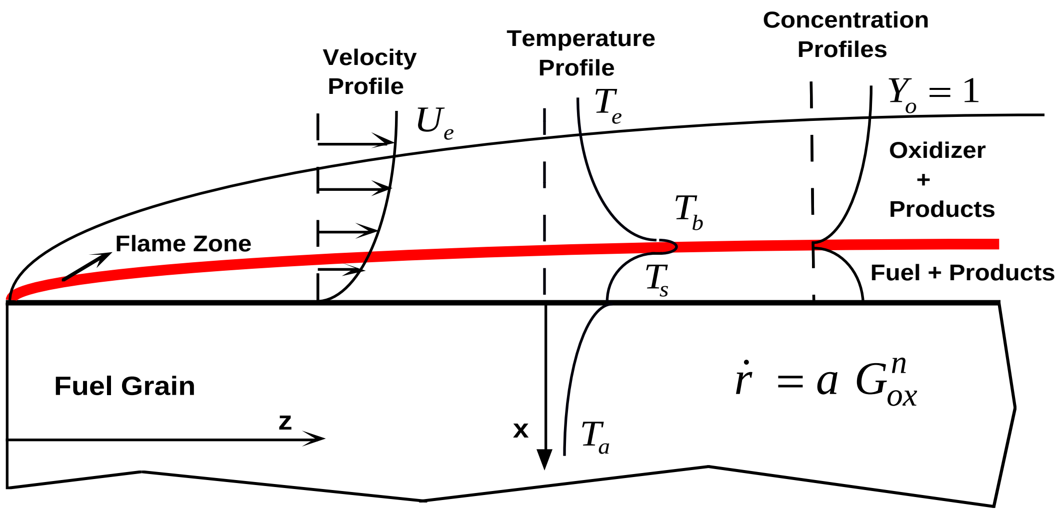


Figure 11.5 Hybrid rocket combustion scheme

As a rough comparison the regression rate in a solid rocket at a typical rocket combustion chamber pressure may be on the order of *1 cm/sec* whereas a typical hybrid using a classical polymeric fuel such as HTPB may have a regression rate on the order of *0.1 cm/sec* . To compensate for the low regression rate the surface area for burning must be increased. This is accomplished through the use of a multiport fuel grain such as that depicted below.

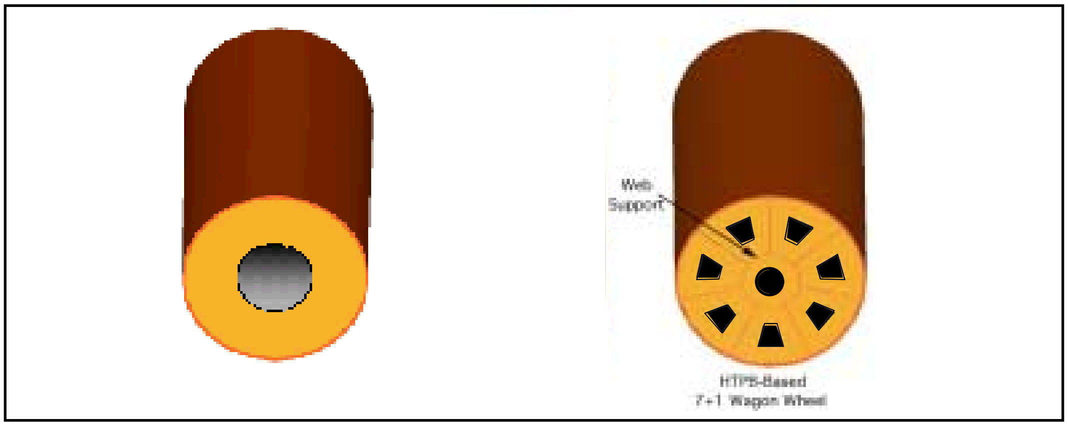


Figure 11.6   Single versus multiport (wagon wheel) grain design

The most obvious problem with the multiport design is that the amount of fuel that can be loaded into a given volume is reduced leading to an increase in the vehicle diameter for a given total fuel mass. There are other problems. The grain must be produced in segments and each segment must be supported structurally adding weight and complexity. In addition it is very difficult to get each port to burn at the same rate. If one burns slightly faster than another then the oxidizer will tend to follow the path of least resistance leading to further disparity in the oxidizer flow rate variation from port to port. Toward the end of burning the port that reaches the liner first forces the motor to be shut down prematurely leading to an inordinately large sliver fraction of unburned fuel. Small pressure differences from port to port can lead to grain structural failure and loss of fuel fragments through the nozzle. Aside from possible damage to the nozzle, the resulting increase in the overall O/F ratio leads to a reduction of the specific impulse and an increase in the nozzle throat erosion rate. Due to the high erosion the nozzle area ratio decreases excessively leading to an additional loss of specific impulse.

11.2 HISTORICAL PERSPECTIVE

Early hybrid rocket development and flight test programs were initiated both in Europe and the U.S. in the 1960's. The European programs in France and Sweden involved small sounding rockets, whereas the American flight programs were target drones (Sandpiper, HAST, and Firebolt) which required supersonic flight in the upper atmosphere for up to 5 minutes. These latter applications were suitable for the conventional hybrid because its very low burning rate was ideal for a long duration sustainer operation.

Despite the very low regression rate of the fuel, in the late 1960's Chemical Systems Division of United Technologies (CSD/UTC) investigated motor designs of larger diameters that could produce high thrust suitable for space launch vehicles. They experimented with a 38 inch diameter motor delivering 40,000 lbs. of thrust. In order to achieve a high mass flow rate, a motor with 12 ports in the fuel grain was required. Although the motor was successfully fired several times, it was recognized that the volumetric fuel loading efficiency was compromised, which would lead to a deficit in vehicle performance.

Interest in the hybrid was revived again in the late 1970's when concern was expressed for the storage and handling of the large solid propellant segments of the Shuttle booster. The storage of potentially explosive grains is costly in terms of requirements for reinforced structures and interline distance separation. The same safety concern arose again after the Challenger disaster, where it was recognized that a thrust termination option might have avoided the failure. This concern was heightened when, a few months later, there was a Titan failure, caused by an explosion of one of the solid boosters.

In the last decade or so, two significant hybrid efforts occurred. One was the formation of the American Rocket Company (AMROC), an entrepreneurial industrial company devoted entirely to the development of large hybrid boosters. The second, with encouragement from NASA, was the formation of the Hybrid Propulsion Industry Action Group (HPIAG) composed of both system and propulsion companies devoted to exploring the possible use of hybrids for the Shuttle booster and other launch booster applications. Both efforts ran into technical stumbling blocks, basically caused by the low regression rate fuels, which resulted in large diameter motors with many ports to satisfy thrust requirements. The resulting configuration not only compromised potential retrofit for the Shuttle and Titan boosters but also raised questions about internal ballistic performance of a thin web multi-port motor, especially toward the end of burning. Although AMROC had many successful tests in 51 inch diameter motors, they ran into difficulties when the motor was scaled to 6 foot diameter and 250,000 lb. thrust. The low regression rate of the fuel dictated a 15 port grain design and problems of poor grain integrity were the result. In 1995 AMROC filed for bankruptcy.

Several hybrid propulsion programs were initiated in the late 80’s and early 90’s. The Joint Government/Industry Research and Development (JIRAD) program involved the testing of 11 and 24 inch diameter hybrid motors at the Marshall Space Flight Center. Another hybrid program initiated during the early 90’s was DARPA’s Hybrid Technology Options Project (HyTOP). The goal of this program was to develop the HyFlyer launch vehicle and demonstrate the feasibility of hybrid boosters for space applications. The members of the HyTOP team were AMROC, Martin Marietta and CSD/UTC.

The Hybrid Propulsion Demonstration Program (HPDP) began in March 1995. The goal of the HPDP was to enhance and demonstrate several critical technologies which are essential for the full scale development of hybrid rocket boosters for space launch applications. The government and industry participants in the program are NASA, DARPA, Lockheed Martin, CSD/UTC, Thiokol, Rocketdyne, Allied Signal and Environmental Aeroscience Corporation. Even though the tasks of the HPDP program included systems studies and subscale testing, the main objective of the program was the design and fabrication of a 250,000 pound thrust test-bed. The design of the motor was guided by the subscale motor tests performed under the JIRAD program. The wagon wheel 7+1 multiport fuel grain is made of conventional hydroxyl-terminated-polybutadiene (HTPB)/ Escorez fuel. The motor was fired for short times in July 1999. The motor exhibited large pressure oscillations and unequal burning rates in the various ports. Problems related to low regression rate inherent in conventional hybrids fuels were not solved.

The most recent advance in hybrid rockets occurred in the Fall of 2004 when SpaceShipOne carried a pilot to over 328,000 feet to win the Ansari X-prize. This privately funded, sub-orbital flight ushered in a new era in space tourism.

Figure 11.7 Space Ship One carried aloft by the White Knight

The figure below depicts the various components of this suborbital launch system.

Figure 11.8 Various systems of Space Ship One. Note the hybrid rocket on the left.

The propulsion system used for this project used a four port motor fueled by HTPB with nitrous oxide (N2O) as the oxidizer. Although the flight of Space Ship One was a great success it was not exactly a walk in the park for the pilot. The quotes below from Aviation Week create a pretty sobering picture of the flight.

Neither SpaceDev or eAc met Scaled's wishes. The SpaceDev design, which has four longitudinal ports in the rubber fuel for enough burning area for high thrust, comes on with a bang, producing maximum thrust at the start—not the smooth ramp-up envisioned to turn the corner. The peak thrust is only about 85% of the desired plateau, and declines steadily from there, according to the Sept. 18 SETP presentation. Despite the early start, this still means that to get enough total impulse to loft SpaceShipOne above 100 km., the motor has to run longer than desired, in the very thin atmosphere where control is tenuous.

The eAc motor didn't ignite until five sec. after the switch was thrown, and then also came on with a bang, but the initial combustion instabilities were less. It has a single port and compensates for the lower burning area with fuel additives to increase burning rate. But not enough, because it only made about 65% of the desired thrust. That required the burn time to be even longer for sufficient total impulse, extending the engine run farther out of the atmosphere.

Longer burn time of the eAc motor was considered the more serious problem, and the contract went to SpaceDev.

The webs of rubber between the four SpaceDev ports thin out and come apart toward the end of the run. The chunks extrude through the nozzle, causing frightening shaking and explosion noises in the cockpit. It happened at least three times on one flight and Melvill thought the tail had blown off. After minutes by himself in zero-g and entry, he was relieved when chase aircraft said the spaceship appeared alright.

Mike Dornheim - Aviation Week
October 18, 2004 page 36

The conclusion from this history is that if a significantly higher burning rate fuel can be developed for the hybrid motor, the multiport difficulties mentioned above can be alleviated and a smaller, safer more efficient motor can be obtained. Although this deficiency of conventional hybrid fuels was recognized more than forty years ago, attempts to increase the burning rate by more than 50-100%, without compromising the safety and low cost features of the hybrid design, have been largely unsuccessful until recently.

11.3 HIGH REGRESSION RATE FUELS

Recent research at Stanford University has led to the identification of a class of paraffin-based fuels that burn at surface regression rates that are 3-4 times that of conventional hybrid fuels. The new fuel produces a very thin, low viscosity, low surface tension liquid layer on the fuel surface when it burns. The instability of this layer driven by the oxidizer gas flow in the port leads to the lift-off of and entrainment of droplets into the gas stream greatly increasing the overall fuel mass transfer rate. The basic mechanism is sketched below.

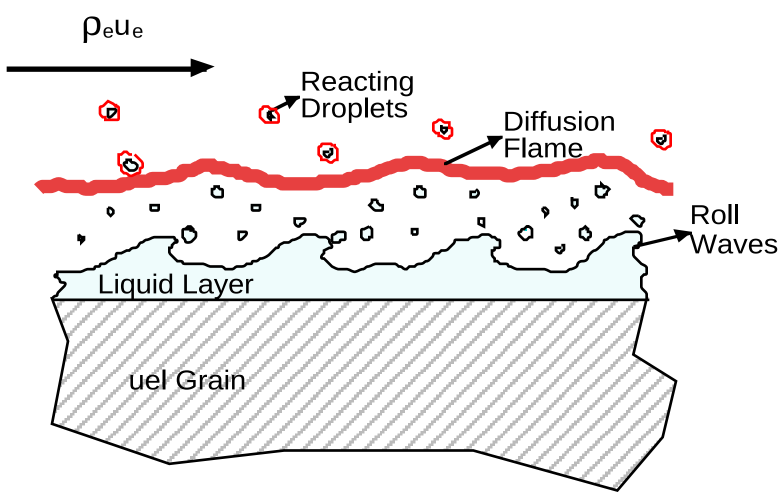


Figure 11.9 Liquid layer entrainment mechanism

In effect, this mechanism acts like a continuous spray injection system distributed along the port with most of the fuel vaporization occurring around droplets convecting between the melt layer and the flame front. Since droplet entrainment is not limited by diffusive heat transfer to the fuel from the combustion zone, this mechanism can lead to much higher surface regression rates than can be achieved with conventional polymeric fuels that rely solely on evaporation. The entrainment component of the regression rate depends on the parameters depicted below.

Dynamic pressure
thickness

$$\dot{m}_{ent} \propto \frac{P_d^{\alpha} h^{\beta}}{\sigma^{\pi} \mu_l^{\gamma}}$$

Surface tension
viscosity

The key fuel properties are low surface tension and low viscosity of the melt layer evaluated at the characteristic temperature of the layer. This forms the basis of a fundamental criterion that can be used to identify high regression rate fuels. Not all fuels that form a melt layer at the fuel surface will entrain. For example high-density-polyethelene (HDPE) which is a conventional hybrid fuel does form a melt layer but the viscosity of the liquid is four orders of magnitude larger than paraffin - too viscous to permit significant droplet entrainment.

It has been found that members of the normal-alkane class of hydrocarbons, that are solid at room temperature for carbon numbers greater than 14, have low surface tension and viscosity at the melt layer conditions typical of hybrid rockets. These fuels, which include the paraffin waxes and polyethylene waxes, are predicted to have high regression rates at oxidizer mass fluxes covering a wide range of hybrid rocket applications. A less expensive, easier to process, more environmentally benign fuel could hardly be imagined! Making, handling and transporting traditional solid rocket propellants is very costly, but the new paraffin-based fuel is non-toxic and non-hazardous. The by-products of combustion are carbon dioxide and water. In contrast, the by-products of burning conventional solid rocket propellant often include acid forming gases such as hydrogen chloride, as well as carbon monoxide.

Regression rates 3 to 4 times the predicted classical rate have been observed in a laboratory scale motor using gaseous oxygen and an industrial grade paraffin wax. The specific impulse of a paraffin-based hybrid motor is slightly higher than that of a kerosene-based liquid motor and solid paraffin is approximately 20% more dense than liquid kerosene. Figure 11.4 shows the ideal specific impulse of paraffin wax and HTPB burning with Liquid Oxygen. The waxes comprise a wide range of molecular weight, surface tension and viscosity and therefore can be used to create mixtures whose regression rate characteristics are tailored for a given mission.

11.4 THE O/F SHIFT

Over the course of a burn at a fixed oxidizer mass flow rate there is a tendency for the oxidizer to fuel (O/F) ratio to shift to higher values as the port opens up. This can be seen from the following. For a single circular port,

$$O/F = \frac{\dot{m}_o}{\dot{m}_f} = \frac{\dot{m}_o}{\rho_f A_b \dot{r}} = \frac{\dot{m}_o}{\rho_f L \pi D \alpha \left(\frac{4\dot{m}_o}{\pi D^2}\right)^n} = \frac{\dot{m}_o^{1-n} D^{2n-1}}{4^n \pi^{1-n} \alpha \rho_f L} \tag{11.4}$$

where L is the port length and D is the port diameter. Recall that the exponent is generally in the range $0.6 < n < 0.8$. As the port diameter increases the burning area increases and the oxidizer mass flux goes down. For $n > 0.5$ the decrease in mass flux dominates the increase in burning area and the overall generation rate of fuel mass goes down. The net effect is to cause the chamber pressure and hence the thrust to decrease naturally over the course of the burn as the vehicle mass decreases. This feature is desirable for a launch system where the payload is subject to a maximum acceleration constraint. Compare this to a solid rocket where the thrust tends to increase during the burn and a throttling option is not available.

Note the relatively strong sensitivity in Figure 11.4 of the specific impulse to the O/F ratio. The change of O/F implies a change in specific impulse and a possible reduction in vehicle performance. This is a factor that must be taken into account by the designer seeking to get maximum total delivered impulse from the motor. In practice the maximum payload acceleration limit leads to a requirement that the oxidizer mass flow be throttled back while the port opens up and the two effects tend to offset one another. A typical case might be a factor of two decrease in the oxidizer mass flow rate and a factor of three increase in the port diameter. For $n = 0.62$ the net effect is less than a one percent change in O/F.

11.5 SCALE-UP TESTS

To further demonstrate the feasibility of this approach, a series of tests were carried out on intermediate scale motors at pressures and mass fluxes representative of commercial applications. A new hybrid test facility designed to study these fuels was developed by NASA and Stanford researchers at NASA Ames Research Center and came on line in September of 2001. An image from one of these tests is shown below.

Figure 11.10 Hybrid motor test at NASA Ames. Thrust is approximately 2500 pounds with a simple convergent nozzle.

The figure below shows the main results of these tests as well as earlier results of testing on a laboratory scale motor at Stanford. The results are compared with the regression rate of HTPB .

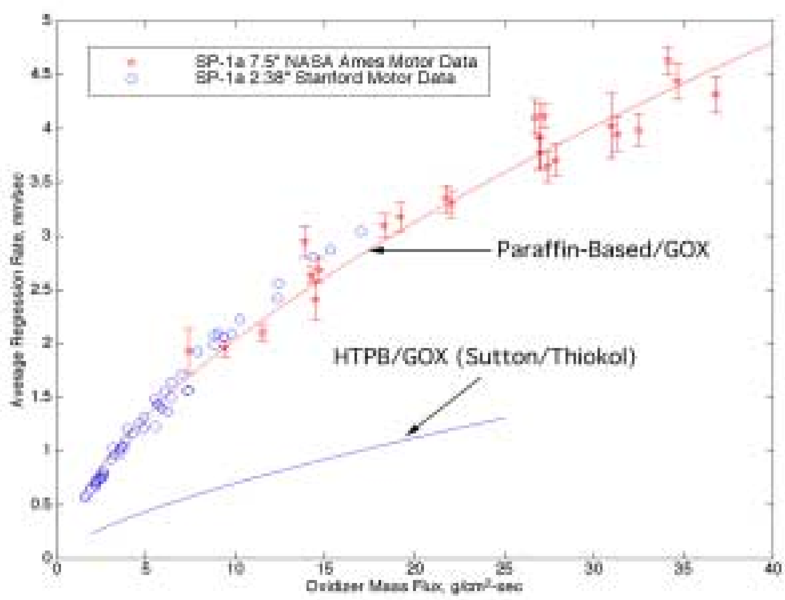


Figure 11.11 Regression rate versus oxidizer mass flux for paraffin and HTPB

The main conclusions from these tests are the following.

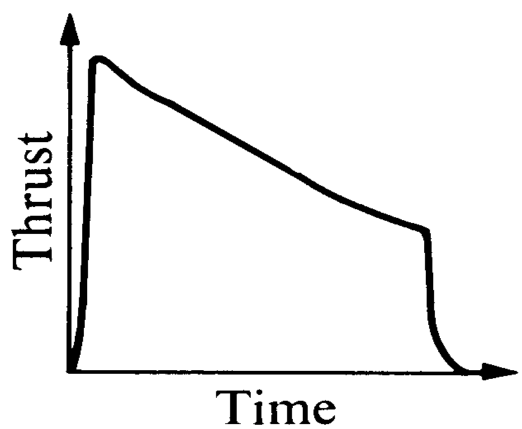
1) The regression rate behavior observed in the small scale tests at Stanford prevails when the motor is scaled up to chamber pressures and mass fluxes characteristic of operational systems. Moreover the regression rate data from large and small motors matches quite well indicating that small scale tests can be used to infer the behavior of larger motors. This is extremely useful when it comes to developing the right fuel formulation for a given mission.

2) Paraffin-based fuels provide reliable ignition and stable combustion over the entire range of mass fluxes encountered (5-60 gm/cm2-sec).

3) The fuel exhibited excellent structural integrity over the range of chamber pressures used (150-960psi).

11.6 PROBLEMS

Problem 1 - The thrust versus time history of a hybrid rocket with a circular port is shown below.



The oxidizer mass flow rate is constant during the burn. The regression rate of the fuel surface follows a law of the form

$$\dot{r} = \alpha G_o^{\ n} \tag{5}$$

where the exponent n is in the range of 0.6 to 0.8 and G_o is the oxidizer flux in the port. Briefly discuss why the thrust tends to decrease over the course of the burn.

Problem 2 - A paraffin-oxygen hybrid rocket operates in a vacuum with a $10\ cm$ diameter nozzle throat and a nozzle area ratio of 70. The motor has a cylindrical port 300 cm long. At the beginning of the burn the port is 20 cm in diameter and $O/F = 2.3$. The port diameter at the end of the burn is 60 cm. The regression rate law is

$$\dot{r} = 0.0488 G_0^{0.62}\ cm/sec \tag{11.6}$$

The fuel density is $0.93\ grams/cm^3$ and the combustion gas has $\gamma = 1.15$ and average molecular weight equal to 30. Assume the oxidizer flow rate is constant and the combustion chamber temperature remains constant over the course of the burn. Approximate the specific impulse by a mean value of 360 sec over the course of the burn.

1) Estimate the chamber pressure at the beginning of the burn.

2) Plot the diameter of the port as a function of time.

3) Plot the thrust-time history of the burn and estimate the total delivered impulse (the integral of the thrust time curve).

4) Use Figure 11.4 to estimate the specific impulse at the end of the burn.