INSTITUTE OF AERONAUTICAL ENGINEERING

(Autonomous)

Dundigal, Hyderabad -500 043

AERONAUTICAL ENGINEERING

COURSE HANDOUT

Course Title	**SPACE PROPULSION**				
Course Code	AAE012				
Programme	B.Tech				
Semester	VI	AE			
Course Type	Core				
Regulation	IARE - R16				
Course Structure	**Theory**			**Practical**	
	Lectures	**Tutorials**	**Credits**	**Laboratory**	**Credits**
	3	1	4	-	-
Chief Coordinator	Dr. Praveen Kumar Balguri, Associate Professor				
Course Faculty	Dr. Praveen Kumar Balguri, Associate Professor Mr. Shiva Prasad U, Assistant Professor				

COURSE OBJECTIVES:

I	Appraise various space missions, parameters to be considered for designing trajectories and rocket mission profiles.
II	Classify the different chemical rocket propulsion systems, types of igniters and performance considerations of rockets.
III	Discuss the working principle of solid and liquid propellant rockets and gain basic knowledge of hybrid rocket propulsion.
IV	Illustrate electric propulsion techniques, ion and nuclear rocket and the performances of different advanced propulsion systems.

COURSE OUTCOMES (COs):

The course should enable the students to:	
CO 1	Evaluate various space missions, parameters to be considered for designing trajectories and rocket mission profiles.
CO 2	Classify the different chemical rocket propulsion systems, types of igniters and performance considerations of rockets.

CO 3	Discuss the working principle of solid propellant rockets, propellant grain designs and combustion.
CO 4	Demonstrate the working principle of liquid propellant rockets, feed systems and gain basic knowledge of hybrid rocket propulsion.
CO 5	Illustrate electric propulsion techniques, ion and nuclear rocket and the performances of different advanced propulsion systems.

COURSE LEARNING OUTCOMES (CLOs):

1. Demonstrate the basic principles of space propulsion and its applications in different types of orbits.
2. Describe the concept of orbital elements and basic orbital equations.
3. Adapt the concepts of vertical takeoff and landing for space applications and launch trajectories.
4. Explain the operating principle of rocket engine and demonstrate the rocket equation.
5. Discuss the different Newton's laws of motion and the relation of thrust generation to different laws of motion
6. Describe the different types of propulsion systems and preliminary concepts in nozzle less
7. Propulsion and air augmented rockets.
8. Demonstrate the salient features of solid propellants rockets and estimate the grain configuration
9. Designs suitable for different missions.
10. Understand the erosive burning, combustion instability and burners
11. Remember the applications and advantages of solid propellant rockets
12. Recognize the salient features of liquid propellants rockets, various feed systems and injectors.
13. Understand the thrust control cooling, heat transfer problems, combustion instability in liquid
14. propellant rockets
15. Understand the peculiar problems associated with operation of cryogenic engines in different missions.
16. Recognize the standard and reverse hybrid systems, combustion mechanism, applications and limitations.
17. Understand the different types of Electric, Ion and Nuclear propulsion systems.
18. Identify the future applications of electric propulsion system

SYLLABUS

Unit-I	PRINCIPLES OF ROCKET PROPULSION	Classes: 10
History of rockets, Newton's third law, orbits and space flight, types of orbits, basic orbital equations, elliptical transfer orbits, launch trajectories, the velocity increment needed for launch, the thermal rocket engine, concepts of vertical takeoff and landing, SSTO and TSTO, launch assists.		
Unit-II	**FUNDAMENTALS OF ROCKET PROPULSION**	**Classes: 09**
Operating principle, Rocket equation, Specific impulse of a rocket, internal ballistics, Rocket nozzle classification, Performance characteristics of rockets, air augmented rockets, pulse rocket motors, static testing of rockets and instrumentation, safety considerations.		
Unit-III	**SOLID ROCKET PROPULSION**	**Classes: 10**
Salient features of solid propellant rockets, selection criteria of solid propellants, estimation of solid propellant adiabatic flame temperature, propellant grain design considerations. Types of igniters, Erosive burning in solid propellant rockets, combustion instability, strand burner and T-burner,		

applications and advantages of solid propellant rockets.		
Unit -IV	**LIQUID AND HYBRID ROCKET PROPULSION**	**Classes: 08**
Salient features of liquid propellant rockets, selection of liquid propellants, various feed systems and injectors for liquid propellant rockets, thrust control cooling in liquid propellant rockets and the associated heat transfer problems, combustion instability in liquid propellant rockets, peculiar problems associated with operation of cryogenic engines, introduction to hybrid rocket propulsion, standard and reverse hybrid systems, combustion mechanism in hybrid propellant rockets, applications and limitations.		
Unit -V	**ADVANCED PROPULSION TECHNIQUES**	**Classes: 08**
Electric rocket propulsion, types of electric propulsion techniques, Ion propulsion, Nuclear rocket, comparison of performance of these propulsion systems with chemical rocket propulsion systems, future applications of electric propulsion systems, Solar sail.		
Text Books:		
1. Turner, M.J.L., Rocket and Spacecraft Propulsion, 2nd Edition, MIT Press, 1922. 2. Sutton, G.P., "Rocket Propulsion Elements" John Wiley & Sons Inc., New York, 5th Edition, 1993 3. PHill, P.G. and Peterson, C.R., Mechanics and Thermodynamics of Propulsion, 2nd Edition, Addison Wesley, 1992.		
Reference Books:		
1. Mathur, M.L., and Sharma, R.P., "Gas Turbine, Jet and Rocket Propulsion", Standard Publishers and Distributors, Delhi, 1988. 2. Tajmar, M., Advanced Space Propulsion Systems, Springer 2003 3. Hieter and Pratt, Hypersonic Air breathing propulsion th Edition, 1993.		
Web References:		
1. https://nptel.ac.in/courses/112106073/ 2. https://www.udemy.com/rocket-science/ 3. https://ocw.mit.edu/courses/aeronautics-and-astronautics/16-522-space-propulsion-spring-2015/		
E-Text Books:		
1. https://ebooks.benthamscience.com/book/9781608052707/ 2. https://www.springer.com/in/book/9781461406068		

UNIT-1

PRINCIPLES OF ROCKET PROPULSION

1.1 THE DEVELOPMENT OF THEROCKET:

Hero of Alexandria is credited with inventing the rocket principle. He was a mathematician and inventor and devised many machines using water, air pressure. The rocket was also used as a weapon of oriental war. Hublai Hahn used it during the Japanese invasion of 1275; by the 1300s rockets were used as bombardment weapons as far west as Spain, brought west by the Mongol hordes, and the Arabs.

Konstantin Tsiolkovsky(1857−1935), a mathematics teacher, wrote about space travel, including weightlessness and escape velocity, in1883, and he wrote about artificial satellites in 1895. In a paper published in 1903 he derived the rocket equation, and dealt in detail with the use of rocket propulsion for space travel andin1924hedescribedmulti-stagerockets.Tsiolkovsky never experimented with rockets; his work was almost purely theoretical. He identified exhaust velocity as the important performance parameter; he realized that the higher temperature and lower molecular weight produced by liquid fuels would be important for achieving high exhaust velocity; and heidentifiedliquidoxygenandhydrogenassuitablepropellantsforspacerockets. He also invented the multi-stage rocket.

Goddard's inventions included the use of gyroscopes for guidance, the use of vanes in the jet stream to steer the rocket, the use of valves in the propellant lines to stop and start the engine, the use of turbo-pumps to deliver the propellant to the combustion chamber, and the use of liquid oxygen to cool the exhaust nozzle, all of which were crucial to the development of the modern rocket. He launched his first liquid-fuelled rocket from Auburn, Massachusetts, on 16 March 1926. It weighed 5 kg, was powered by liquid oxygen and petrol, and it reached a height of 12.5 meters. At the end of his 1919 paper Goddard had mentioned the possibility of sending an unmanned rocket to the Moon, and for this he was ridiculed by the Press. BecauseofhisrocketexperimentshewaslaterthrownoutofMassachusettsbythe fire officer, but he continued his work until 1940, launching his rockets in New Mexico. In 1960 the US government bought his patents for two million dollars.

1.1.1 THE RUSSIAN SPACE PROGRAMME: the Russian space programme has been the most active and focused in history. The first artificial satellite, the first man in space, the first spacecraft on the Moon, the first docking of two spacecraft, and the first space station. All of these are the achievements of Russia (or, rather, the Soviet Union). In the period from 1957 to 1959, three satellites and two successful lunar probes had been launched by the USSR, ironically fulfilling Goddard's prophesy. In 1961, Yuri Gagarin became the first man in space, and at the same time, several fly-bys of Mars and Venus were accomplished. In all there were 12 successful Russian lunar probes launched before the first Saturn V. Apart from the drive and vision of the Soviet engineers— particularly Sergei Korolev—the reason for this success lay in the fact that the Russian rockets were more powerful, and were better designed. The pre-war Russian attitude to rocketry had found a stimulus in the captured German parts, leading to the development of an indigenous culture which was to produce the best engines. It is significant that the Saturn V was the brainchild of Werner von Braun, a German, and the Vostok, Soyuz, and Molniya rockets were the brainchildren of Korolev and Glushko, who were Russian.

1.1.2 OTHER NATIONAL PROGRAMMES: Before turning to the United States' achievements in rocketry, we should remember that a number of other nations have contributed to the development of the present- day portfolio of launchers and space vehicles. There are active space and launcher programmers in the Far East, where China, Japan, India, and Pakistan all have space programmers. China and Japan both have major launcher portfolios.

1.1.3THE UNITED STATES SPACE PROGRAMME: The achievement of the United States in realizing humanity's dream of walking on the Moon cannot be overrated. Its origin in the works of Tsiolkovsky and Oberth, its national expression in the dream of Robert Goddard, and its final achievement through the will of an American president and people, is unique in human history. From what has gone before it is clear that the ambition to walk on the Moon was universal amongst those who could see the way, and did not belong to any nation or hemisphere. Nor was the technology exclusive. In fact, the Soviet Union came within an ace of achieving it. But it rested with one nation to achieve that unity of purpose without which no great endeavour can be achieved. That nation was the United States of America.

1.2 NEWTON'STHIRDLAW: Therockethadbeenapracticaldeviceformorethan1,000years beforeTsiolkovskydeterminedthedynamicsthatexplaineditsmotion.Indoingso, he opened the way to the use of the rocket as something other than an artillery weaponofdubiousaccuracy.Infact,heidentifiedtherocketasthemeansbywhich humanity could explore space. This was revolutionary earlier, fictious journeys to the Moon had made use of birds or guns as the motive force, and rockets had been discounted. By solving the equation of motion of the rocket, Tsiolkovsky was able to show that space travel was possible, and that it could be achieved using a device which was readily to hand, and only needed to be scaled up. He even identified the limitations and design issues which would have to be faced in realizing practical spacevehicle.Thedynamicsaresosimplethatitissurprisingthatithadnotbeen solved before—but this was probably due to a lack of interest perusal of dynamics books of the period reveals consistent interest in the flight of unpowered projectiles, immediately applicable to gunnery.

1.3 ORBITS ANDSPACE FLIGHT: This involves gravity, and the motion of vehicles in the Earth's gravitational field. Common experience, with a cricket ball for example, tells us that the faster a body is projected upwards, the further it goes. The science of ballistics tells us that a shell, with a certain velocity, will travel furthest in a horizontal direction, if projected at an initial angle of 45^{O}. The equations of motion of a cricket ball, or a shell, can be solved using a constant and uniform gravitational field, with very little error. This is a matter for school physics. When we consider space travel, the true shape of the gravitational field becomes important; It is radial field, with its origin in the center of the Earth. Note that the gravitational field of a spherical object is accurately represented by assuming that it acts from the Centre, with the full mass of the object. The flat Earth approximation is good enough for distances travelled which are small compared with the curvature of the Earth, but cannot be applied to space travel, where the distances are much greater.

1.3.1 TYPES OF ORBITS: For a spacecraft to achieve Earth orbit, it must be launched to an elevation above the Earth's atmosphere and accelerated to orbital velocity. The most energy efficient orbit, that is one that requires the least amount of propellant, is a direct low inclination orbit. To achieve such an orbit, a spacecraft is launched in an eastward direction from a site near the Earth's equator. The advantage being that the rotational speed of the Earth contributes to the spacecraft's final orbital speed. At the United States' launch site in Cape Canaveral (28.5 degrees north latitude) due easts launch results in a "free ride" of 1,471 km/h (914 mph). Launching a spacecraft in a direction other than east, or from a site far from the equator, results in an orbit of higher inclination. High inclination orbits are less able to take advantage of the initial speed provided by the Earth's rotation, thus the launch vehicle must provide a greater part, or all, of the energy required to attain orbital velocity. Although high inclination orbits are less energy efficient, they do have advantages over equatorial orbits for certain applications.

Below we describe several types of orbits and the advantages of each:

Geosynchronous orbits (GEO): are circular orbits around the Earth having a period of 24 hours. A geosynchronous orbit with an inclination of zero degrees is called a geostationary orbit. A spacecraft in a geostationary orbit appears to hang motionless above one position on the Earth's equator. For this reason, they are ideal for some types of communication and meteorological satellites. A spacecraft in an inclined geosynchronous orbit will appear to follow a regular figure-8 pattern in the sky once every orbit. To attain geosynchronous orbit, a spacecraft is first launched into an elliptical orbit with an apogee of 35,786 km (22,236 miles) called a geosynchronous transfer orbit (GTO). The orbit is then circularized by firing the spacecraft's engine at apogee.

Polar orbits (PO): are orbits with an inclination of 90 degrees. Polar orbits are useful for satellites that carry out mapping and/or surveillance operations because as the planet rotates the spacecraft has access to virtually every point on the planet's surface.

Walking orbits: An orbiting satellite is subjected to a great many gravitational influences. First, planets are not perfectly spherical and they have slightly uneven mass distribution. These fluctuations have an effect on a spacecraft's trajectory. Also, the sun, moon, and planets contribute a gravitational influence on an orbiting satellite. With proper planning it is possible to design an orbit which takes advantage of these influences to induce a precession in the satellite's orbital plane. The resulting orbit is called a walking orbit, or precessing orbit.

Sun synchronous orbits (SSO) :are walking orbits whose orbital plane precesses with the same period as the planet's solar orbit period. In such an orbit, a satellite crosses periapsis at about the same local time every orbit. This is useful if a satellite is carrying instruments which depend on a certain angle of solar illumination on the planet's surface. In order to maintain an exact synchronous timing, it may be necessary to conduct occasional propulsive maneuvers to adjust the orbit.

Molniya orbits:are highly eccentric Earth orbits with periods of approximately 12 hours (2 revolutions per day). The orbital inclination is chosen so the rate of change of perigee is zero, thus both apogee and perigee can be maintained over fixed latitudes. This condition occurs at inclinations of 63.4 degrees and 116.6 degrees. For these orbits the argument of perigee is typically placed in the southern hemisphere, so the satellite remains above the northern hemisphere near apogee for approximately 11 hours per orbit. This orientation can provide good ground coverage at high northern latitudes.

Hohmann transfer orbits: are interplanetary trajectories whose advantage is that they consume the least possible amount of propellant. A Hohmann transfer orbit to an outer planet, such as Mars, is achieved by launching a spacecraft and accelerating it in the direction of Earth's revolution around the sun until it breaks free of the Earth's gravity and reaches a velocity which places it in a sun orbit with an aphelion equal to the orbit of the outer planet. Upon reaching its destination, the spacecraft must decelerate so that the planet's gravity can capture it into a planetary orbit.

To reach a planet requires that the spacecraft be inserted into an interplanetary trajectory at the correct time so that the spacecraft arrives at the planet's orbit when the planet will be at the point where the spacecraft will intercept it. This task is comparable to a quarterback "leading" his receiver so that the football and receiver arrive at the same point at the same time. The interval of time in which a spacecraft must be launched in order to complete its mission is called a launch window.

1.3.2 MOTION OF PLANETS AND SATELLITES: Through a lifelong study of the motions of bodies in the solar system, Johannes Kepler (1571-1630) was able to derive three basic laws known as Kepler's laws of planetary motion. Using the data compiled by his mentor Tycho Brahe (1546-1601), Kepler found the following regularities after years of laborious calculations:

1. All planets move in elliptical orbits with the sun at one focus.
2. A line joining any planet to the sun sweeps out equal areas in equal times.
3. The square of the period of any planet about the sun is proportional to the cube of the planet's mean distance from the sun.

These laws can be deduced from Newton's laws of motion and law of universal gravitation. Indeed, Newton used Kepler's work as basic information in the formulation of his gravitational theory.

As Kepler pointed out, all planets move in elliptical orbits, however, we can learn much about planetary motion by considering the special case of circular orbits. We shall neglect the forces between planets, considering only a planet's interaction with the sun. These considerations apply equally well to the motion of a satellite about a planet.

Let's examine the case of two bodies of masses M and m moving in circular orbits under the influence of each other's gravitational attraction. The center of mass of this system of two bodies lies along the line joining them at a point C such that mr = MR. The large body of mass M moves in an orbit of constant radius R and the small body of mass m in an orbit of constant radius r, both having the same angular velocity ω.

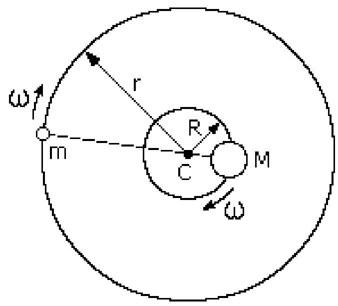


Figure1.1 Angular velocity

For this to happen, the gravitational force acting on each body must provide the necessary centripetal acceleration. Since these gravitational forces are a simple action-reaction pair, the centripetal forces must be equal but opposite in direction. That is, $m\omega^2 r$ must equal $M\omega^2 R$. The specific requirement, then, is that the gravitational force acting on either body must equal the centripetal force needed to keep it moving in its circular orbit, that is

$$\frac{GMm}{(R+r)^2} = m\omega^2 r$$

If one body has a much greater mass than the other, as is the case of the sun and a planet or the Earth and a satellite, its distance from the centre of mass is much smaller than that of the other body. If we assume that m is negligible compared to M, then R is negligible compared to r. Thus, equation then becomes

$$GM = \omega^2 r^3$$

If we express the angular velocity in terms of the period of revolution, $\omega = 2\pi/P$, we obtain

$$GM = \frac{4\pi^2 r^3}{P^2}, \quad \text{or}$$

$$P^2 = \frac{4\pi^2 r^3}{GM}$$

where P is the period of revolution. This is a basic equation of planetary and satellite motion. It also holds for elliptical orbits if we define r to be the semi-major axis (a) of the orbit.

A significant consequence of this equation is that it predicts Kepler's third law of planetary motion, that is P2~r3.

Kepler's second law of planetary motion must, of course, hold true for circular orbits. In such orbits both ω and r are constant so that equal areas are swept out in equal times by the line joining a planet and the sun. For elliptical orbits, however, both ω and r will vary with time. Let's now consider this case.

Figure 1.2 shows a particle revolving around C along some arbitrary path. The area swept out by the radius vector in a short time interval Δt is shown shaded. This area, neglecting the small triangular region at the end, is one-half the base times the height or approximately r(r ω Δt)/2.

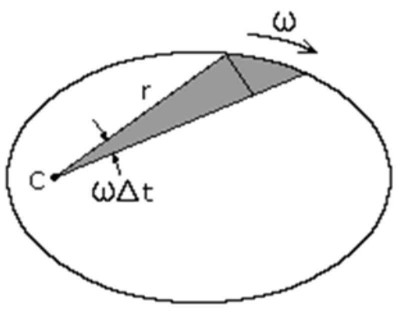


Figure 1.2 a particle revolving around C

This expression becomes more exact as Δt approaches zero, i.e. the small triangle goes to zero more rapidly than the large one. The rate at which area is being swept out instantaneously is therefore

$$\lim_{t \to 0} \left[\frac{r(r\omega\Delta t)}{2}\right] = \frac{\omega r^2}{2}$$

For any given body moving under the influence of a central force, the value ωr2 is constant.

Let's now consider two points P1 and P2 in an orbit with radii r1 and r2, and velocities v1 and v2. Since the velocity is always tangent to the path, it can be seen that if γ is the angle between r and v, then

$$v\sin\gamma = \omega r \qquad \text{----- -1}$$

where vsinγ is the transverse component of v. Multiplying through by r, we have

$$rv\sin\gamma = \omega r^2 = \text{Constant} \text{ ------2}$$

or, for two points P1 and P2 on the orbital path

$$r_1 v_1 \sin\gamma_1 = r_2 v_2 \sin\gamma_2 \text{ --------3}$$

Note that at periapsis and apoapsis, γ = 90 degrees. Thus, letting P1 and P2 be these two points we get

$$R_p V_p = R_a V_a \text{ --------4}$$

Let's now look at the energy of the above particle at points P1 and P2. Conservation of energy states that the sum of the kinetic energy and the potential energy of a particle remain constant. The kinetic energy Tof a particle is given by mv2/2 while the potential energy of gravity V is calculated by the equation -GMm/r. Applying conservation of energy we have

$$T_1 + V_1 = T_2 + V_2, \quad \text{or}$$

$$\frac{mv_1^2}{2} - \frac{GMm}{r_1} = \frac{mv_2^2}{2} - \frac{GMm}{r_2}, \quad \text{or}$$

$$v_2^2 - v_1^2 = 2GM\left(\frac{1}{r_2} - \frac{1}{r_1}\right) \qquad \text{-----------5}$$

From equations (4) and (5) we obtain

$$Vp = \sqrt{\frac{2GMR_a}{R_p(R_a+R_p)}}, \quad \text{and}$$

$$Va = \sqrt{\frac{2GMR_p}{R_a(R_a+R_p)}} \qquad \text{------------------6}$$

Rearranging terms we get

$$R_a = \frac{R_p}{\left(\frac{2GM}{R_p V_p^2} - 1\right)}, \quad \text{and}$$

$$R_p = \frac{R_a}{\left(\frac{2GM}{R_a V_a^2} - 1\right)} \qquad \text{---------7}$$

The eccentricity e of an orbit is given by

$$e = \frac{R_p V_p^2}{GM} - 1 \quad \text{--------------8}$$

If the semi-major axis a and the eccentricity e of an orbit are known, then the periapsis and apoapsis distances can be calculated by

$$R_p = a(1-e), \quad \text{and}$$

$$R_a = a(1+e)$$

$$\text{also note, } R_p + R_a = 2a \quad \text{-------------9}$$

1.4 THE VELOCITY INCREMENT NEEDED FOR LAUNCH: It is possible to calculate the total velocity increment required, without gravity loss, as follows, using the earlier formulae. Assume that the launch from the Earth's surface is the equivalen t of a transfer from a circularorbitwitharadiuswhichisthat of the Earth, via a transfer ellipse, to a 500km circular orbit. The imaginary Earth- radiuscircularorbitwouldhaveahorizontalvelocityof7,909ms^{-1}. The transfer ellipse, with perigee at the Earth's surface and apogee at 500 km altitude,hasaperigeevelocityof8,057ms^{-1}.Theapogeevelocityis7,471kms^{-1}, andthenecessarycircularvelocityis7,616ms^{-1}.Thusthetotalvelocityincrementis (8057 7616 7471) 8,203 m s^{-1}. So the velocity cost of the launch, over and abovethatneededforacircularorbitinjectionat500kmaltitude,is587ms^{-1}.This would be true if all the velocity could be given to the rocket all at once, and there were no atmosphere, but because of the gravity loss we need to include an extra allowance of velocity. This depends on the trajectory; an approximate value is 500ms^{-1},andthetotalvelocityincrementrequiredisapproximately8,700ms^{-1}.

There is a distinction between velocity increment and the actual velocity of the vehicle.Thevelocityincrementisthevelocitycalculatedfromtherocketequation, andisameasureoftheenergyexpendedbytherocket.Thevehiclevelocityisless thanthis,becauseofgravityloss,andtheenergyneededtoreachorbitalaltitude.So theactualvelocityofthevehicleinits500-kmcircularorbitis7.6kms^{-1},whilethe velocity increment is 8.7 km s^{-1}. The difference represents the energyexpended against gravity loss and potentialenergy.

The mass ratio for such a velocity increment—especially with primitive rocket fuels, giving low exhaust velocity—is too high to achieve, even with modern constructionmethods.Tsiolkovskyrealisedthis,andin1924hepublishedapaper calledCosmicRockettrains,inwhichheproposedtosolvethedifficultybyusing multistage rockets. This was the essential breakthrough which has enabled humanity, 1,000 years after the invention of the rocket, to travel inspace.

1.5 THE THERMAL ROCKET ENGINE: The rocket principle is the basis of all propulsion in space, and all launch vehicles. The twin properties of needing no external medium for the propulsion system to act upon, and no external oxidant for the fuel, enable rockets to work in any ambient conditions, including the vacuum of space. The thermal rocket is the basis of all launchers, and almost all space propulsion (although some electric propulsion uses a different principle). From these physical principles the strengths and limitations of rocket motors can be understood and appreciated. The thermal rocket motor is a heat engine,, it converts the heat, generated by burning the propellants—fuel and oxidiser, in the combustion chamber—into kinetic energy of the emerging exhaust gas. The momentum carried away by the exhaust gas provides the thrust, which accelerates the rocket. As a heat engine, the rocket is no different in principle from other heat engines, such as the steam engine or the internal combustion engine. The conversion of heat into work is the same, whether the work is done on a piston, or on a stream of exhaust gas.

1.6 GRAVITATIONAL ASSIST: In planning certain types of trajectories of spacecraft within the solar system, engineers rely on a technique called gravitational assist (also gravity assist, slingshot, or swing-by). This technique underlies the feasibility of effecting a net change in both the speed and direction of motion of a spacecraft by passage through the gravitational field of a planet or a planetary satellite, typically in order to save propellant, time, and expense.

A gravitational assist around a planet changes a spacecraft's velocity relative to the Sun by entering and leaving the gravitational field of a planet. The spacecraft accelerates as it approaches the planet and decelerates while escaping its gravitational pull. Because the planet orbits the Sun, this motion affects the spacecraft during the maneuver. To accelerate, the spacecraft flies across the trailing side of the planet, taking a small amount of the planet's orbital energy (as pictured in Figure 1.3). To decelerate, the spacecraft flies across the leading side of the planet. The sum of the kinetic energies of both bodies remains constant. A gravitational assist can therefore be used to change the spaceship's trajectory and speed relative to the Sun.

The resulting increase, or decrease, in the kinetic energy of the spacecraft appears to contradict the casual expectation that in such an encounter the kinetic energy of the spacecraft after the encounter would be the same as that before the encounter. However, the energy gained by the spaceship is equal in magnitude to that lost by the planet, though the planet's enormous mass compared to the spacecraft makes the resulting change in its speed negligibly small. These effects on the planet are so slight that they can be ignored in the calculation.

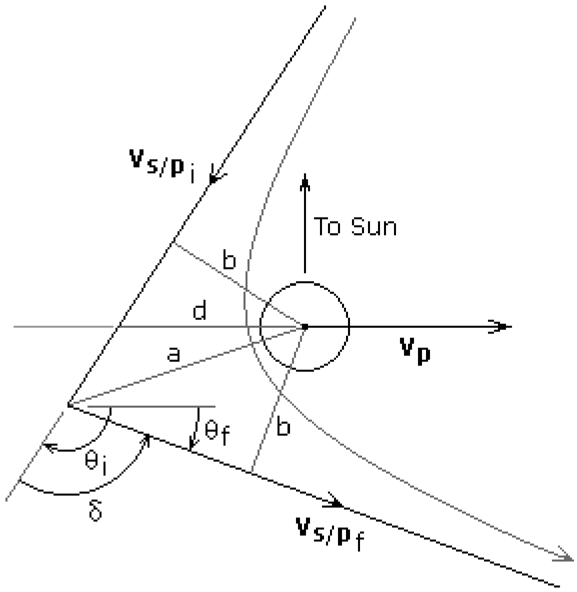


Figure 1.3 shows the motion of a spacecraft relative to a planet during a gravity assist maneuver. Encounters in space require the consideration of three dimensions; however, an approximate solution to the gravitational assist problem can be found using a simplified two-dimensional model.

The following conditions are assumed:

- Orbits of planet and spacecraft are coplanar.
- Y-axis is parallel to the planet's position vector, positive outward from Sun.
- X-axis is in the orbital plane normal to the Y-axis, positive in the prograde direction.
- Planet's velocity (Vp) and flight path angle (ϕp) are given.
- Spacecraft's initial velocity (Vsi), flight path angle (ϕsi), and miss distance (d) are given.

UNIT 2
FUNDAMENTALS OF ROCKET PROPULSION

2.1 INTRODUCTION

Rocket engine: A vehicle or device propelled by one or more rocket engines, especially such a vehicle designed to travel through space.

A projectile weapon carrying a warhead that is powered and propelled by rockets.

A projectile firework having a cylindrical shape and a fuse that is lit from the rear.

Missile: An object or weapon that is fired, thrown, dropped, or otherwise projected at a target; a projectile.

2.2 PROPELLANT: Propellant is the chemical mixture burned to produce thrust in rockets and consists of a fuel and an oxidizer. A fuel is a substance that burns when combined with oxygen producing gas for propulsion. An oxidizer is an agent that releases oxygen for combination with a fuel. The ratio of oxidizer to fuel is called the mixture ratio. Propellants are classified according to their state - liquid, solid, or hybrid.

The gauge for rating the efficiency of rocket propellants is specific impulse, stated in seconds. Specific impulse indicates how many pounds (or kilograms) of thrust are obtained by the consumption of one pound (or kilogram) of propellant in one second. Specific impulse is characteristic of the type of propellant, however, its exact value will vary to some extent with the operating conditions and design of the rocket engine.

2.2.1Liquid Propellants**:**In a liquid propellant rocket, the fuel and oxidizer are stored in separate tanks, and are fed through a system of pipes, valves, and turbopumps to a combustion chamber where they are combined and burned to produce thrust. Liquid propellant engines are more complex than their solid propellant counterparts; however, they offer several advantages. By controlling the flow of propellant to the combustion chamber, the engine can be throttled, stopped, or restarted.

A good liquid propellant is one with a high specific impulse or, stated another way, one with a high speed of exhaust gas ejection. This implies a high combustion temperature and exhaust gases with small molecular weights. However, there is another important factor that must be taken into consideration: the density of the propellant. Using low-density propellants means that larger storage tanks will be required, thus increasing the mass of the launch vehicle. Storage temperature is alsoimportant.

A propellant with a low storage temperature, i.e. a cryogenic, will require thermal insulation, thus further increasing the mass of the launcher. The toxicity of the propellant is likewise important. Safety hazards exist when handling, transporting, and storing highly toxic compounds. Also, some propellants are very corrosive; however, materials that are resistant to certain propellants have been identified for use in rocket construction.

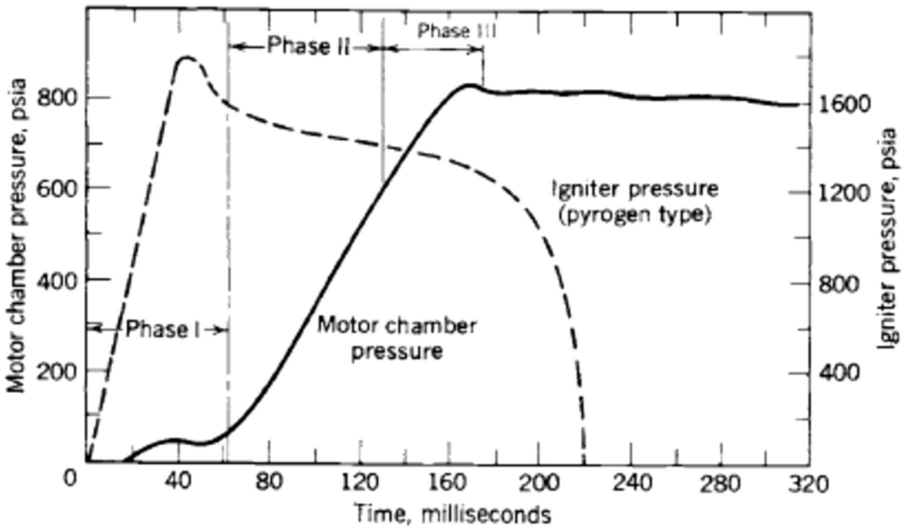
2.2.2Solid Propellants**:**Solid propellant motors are the simplest of all rocket designs. They consist of a casing, usually steel, filled with a mixture of solid compounds (fuel and oxidizer) that burn at a rapid rate, expelling hot gases from a nozzle to produce thrust. When ignited, a solid propellant burns from the center out towards the sides of the casing. The shape of the center channel determines the rate and pattern of the burn, thus providing a means to control thrust.

Unlike liquid propellant engines, solid propellant motors cannot be shut down. Once ignited, they will burn until all the propellant is exhausted.
There are two families of solids propellants: homogeneous and composite. Both types are dense, stable at ordinary temperatures, and easily storable.

2.2.3Hybrid Propellants:Hybridpropellant engines represent an intermediate group between solid and liquid propellant engines. One of the substances is solid, usually the fuel, while the other, usually the oxidizer, is liquid. The liquid is injected into the solid, whose fuel reservoir also serves as the combustion chamber. The main advantage of such engines is that they have high performance, similar to that of solid propellants, but the combustion can be moderated, stopped, or even restarted. It is difficult to make use of this concept for vary large thrusts, and thus, hybrid propellant engines are rarely built.

2.3 IGNITION SYSTEM IN ROCKETS: This section is concerned with the mechanism or the process for initiating the combustion of a solid propellant grain. Solid propellant ignition consists of a series of complex rapid events, which start on receipt of a signal (usually electric) and include heat generation, transfer of the heat from the igniter to the motor grain surface, spreading the flame over the entire burning surface area, filling the chamber free volume (cavity) with gas, and elevating the chamber pressure without serious abnormalities such as overpressures, combustion oscillations, damaging shock waves, hangfires (delayed ignition), extinguishment, and chuffing. The *igniter* in a solid rocket motor generates the heat and gas required for motor ignition.
Motor ignition must usually be complete in a fraction of a second for all but the very large motors. The motor pressure rises to an equilibrium state in a very short time, as shown in Fig. 2.1. Conventionally, the ignition process is divided into three phases for analytical purposes:

Phase I, Ignition time lag: the period from the moment the igniter receives a signal until the first bit of grain surface burns.
Phase II, Flame-spreading interval: the time from first ignition of the grain surface until the complete grain burning area has been ignited.
Phase III, Chamber-filling interval: the time for completing the chamber filling process and for reaching equilibrium chamber pressure and flow



Source from Rocket Propulsion elements by George. P. Sutton

Figure2.1 Typical ignition pressure transient portion of motor chamber pressure time trace with igniter pressure trace and ignition process phases shown.

The ignition will be successful once enough grain surfaces is ignited and burning, so that the motor will continue to raise its own pressure to the operating chamber pressure. The critical process seems to be a gas-phase reaction above the burning surface, when propellant vapors or decomposition products interact with each other and with the igniter gas products. If the igniter is not powerful enough, some grain surfaces may burn for a short time, but the flame will be extinguished.

Satisfactory attainment of equilibrium chamber pressure with full gas flow is dependent on

(1) Characteristics of the igniter and the gas temperature, composition and flow issuing from the igniter,

(2) Motor propellant composition and grain surface ignitability,

(3) heat transfer characteristics by radiation and convection between the igniter gas and grain surface,

(4) Grain flame spreading rate, and

(5) The dynamics of filling the motor free volume with hot gas.

The quantity and type of caloric energy needed to ignite a particular motor grain in the prevailing environment has a direct bearing on most of the igniters' design parameters--particularly those affecting the required heat output. The ignitability of a propellant at a given pressure and temperature is normally shown as a plot of ignition time versus heat flux received by the propellant surface, as shown in Fig. 2.2;

These data are obtained from laboratory tests. Ignitability of a propellant is affected by many factors, including

(1) The propellant formulation,

(2) The initial temperature of the propellant grain surface,

(3) The surrounding pressure,

(4) The mode of heat transfer,

(5) Grain surface roughness,

(6) Age of the propellant,

(7) The composition and hot solid particle content of the igniter gases,

(8) The igniter propellant and its initial temperature,

(9) The velocity of the hot igniter gases relative to the grain surface, and

(10) The cavity volume and configuration.

The ignition time becomes shorter with increases in both heat flux and chamber pressure. If a short ignition delay is required, then a more powerful igniter will be needed. The radiation effects can be significant in the ignition transient case. In Section 1.4 we describe an analysis and design for igniters.

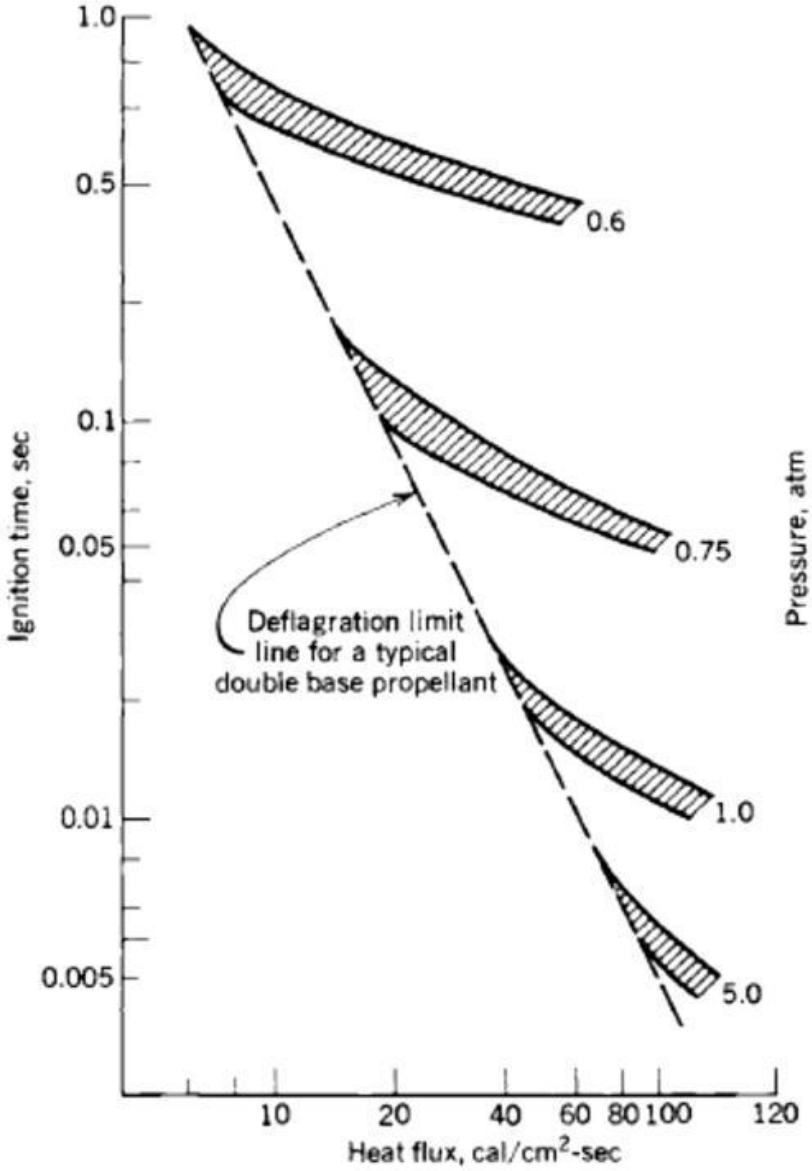


Figure 2.2 Propellant ignitability curves

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

Fig. 2.3 shows several alternative locations for igniter installations. When mounted on the forward end, the gas flow over the propellant surface helps to achieve ignition. With aft mounting there is little gas motion, particularly near the forward end; here ignition must rely on the temperature, pressure, and heat transfer from the igniter gas. If mounted on the nozzle, the igniter hardware and its support is discarded shortly after the igniter has used all its propellants and there is no inert mass penalty for the igniter case.

There are two basic types: pyrotechnic igniters and pyrogen igniters; both are discussed below.

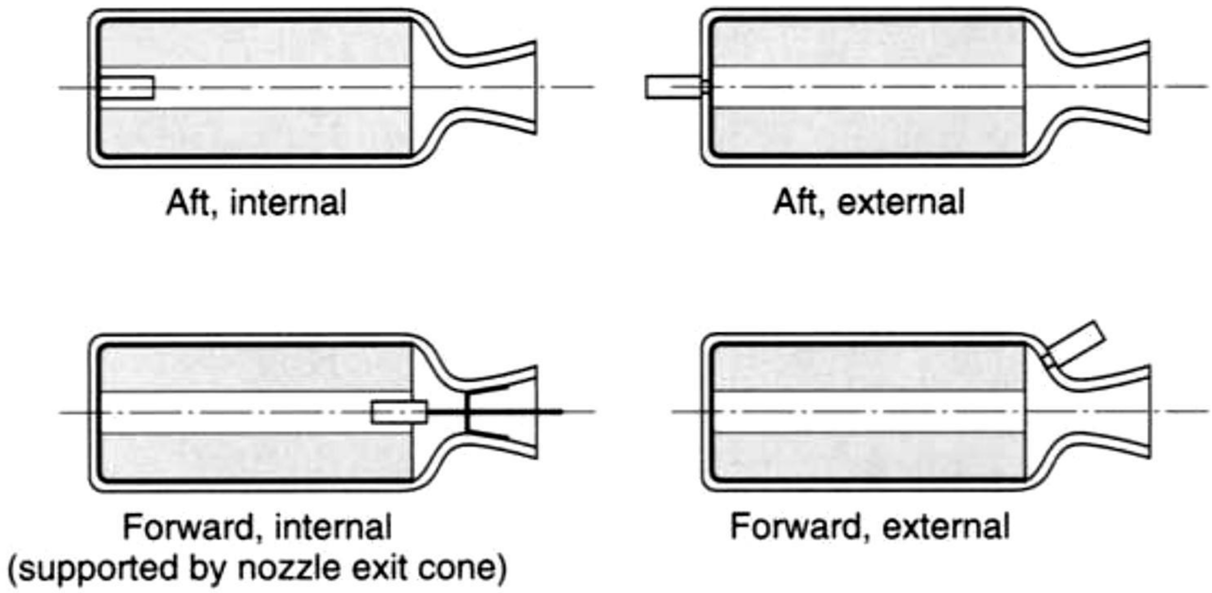


Figure 2.3 Simple Diagrams Of Mounting Options For Igniters.

2.4.1 Pyrotechnic Igniters: In industrial practice, pyrotechnic igniters are defined as igniters (other than pyrogen-type igniters as defined further on) using solid explosives or energetic propellant-like chemical formulations (usually small pellets of propellant which give a large burning surface and a short burning time) as the heat-producing material. This definition fits a wide variety of designs, known as bag and carbon igniters, powder can, plastic case, pellet basket, perforated tube, combustible case, jellyroll, string, or sheet igniters. The common pellet-basket design in Fig. 4 is typical of the pyrotechnic igniters. Ignition of the main charge, in this case pellets consisting of 24% boron-71% potassium perchlorate-5% binder, is accomplished by stages; first, on receipt of an electrical signal the initiator releases the energy of a small amount of sensitive powdered pyrotechnic housed within the initiator, commonly called the squib or the primer charge; next, the booster charge is ignited by heat released from the squib; and finally, the main ignition charge propellants are ignited.

2.4.2surface-bonded or grain-mounted igniter:A special form of pyrotechnic igniter is the surface-bonded or grain-mounted igniter. Such an igniter has its initiator included within a sandwich of flat sheets; the layer touching the grain is the main charge of pyrotechnic. This form of igniter is used with multipulse motors with two or more end-burning grains. The ignition of the second and successive pulses of these motors presents unusual requirements for available space, compatibility with the grain materials, life, and the pressure and temperature.

2.4.3 Pyrogen Igniters: A pyrogen igniter is basically a small rocket motor that is used to ignite a larger rocket motor. The pyrogen is not designed to produce thrust. All use one or more nozzle orifices, both sonic and supersonic types, and most use conventional rocket motor grain formulations and design technology. Heat transfer from the pyrogen to the motor grain is largely convective, with the hot gases contacting the grain surface as contrasted to a highly radiative energy emitted by pyrotechnic igniters.

2.5 TOTAL IMPULSE: It is the thrust force F (which can vary with time) integrated over the burning time t.

$$I_t = \int_0^t F dt$$

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

$$I_t = Ft$$

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

2.5.1 SPECIFIC IMPULSE: The specific impulse is the total impulse per unit weight of propellant. It isan important figure of merit of the performance of a rocket propulsion system, similar in concept to the miles per gallon parameter used with automobiles. Ahigher number means better performance.

If the total mass flow rate of propellant is m(dot)and the standard acceleration of gravity at sealevel go is 9.8066 m/sec 2 or 32.174 ft/sec2, then

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

This equation will give a time-averaged specific impulse value for any rocketpropulsion system, particularly where the thrust varies with time. During transient conditions (during start or the thrust buildup period, the shutdown period, or during a change of flow or thrust levels) values of Is can be obtained byintegration or by determining average values for F and**m(dot)** for short time intervals.

For constant thrust and propellant flow this equation can be simplified;below, mp is the total effective propellant mass.

$$I_s = I_t/(m_p g_0)$$

In a rocket nozzle the actual exhaust velocity is not uniform over the entireexit cross-section and does not represent the entire thrust magnitude. Thevelocity profile is difficult to measure accurately. For convenience a uniformaxial velocity c is assumed which allows a one-dimensional description of theproblem. This effective exhaust velocity c is the average equivalent velocity atwhich propellant is ejected from the vehicle.

It is defined as

$$c = I_s g_0 = F/\dot{m}$$

2.6ROCKET NOZZLE CLASSIFICATION:

Under-and Over-Expanded Nozzles:
An under-expanded nozzle discharges the fluid at an exit pressure greater thanthe external pressure because the exit area is too small for an optimum areáratio. The expansion of the fluid is therefore incomplete within the nozzle, andmust take place outside. The nozzle exit pressure is higher than the local atmosphericpressure.
In an over-expanded nozzle the fluid attains a lower exit pressure than the atmosphere as it has an exit area too large for optimum. The phenomenon of over-expansion for a supersonic nozzle is shown in Fig. 3-9, with typical pressure measurements of superheated steam along the nozzle axis and differentback pressures or pressure ratios. Curve AB shows the variation of pressurewith the optimum back pressure corresponding to the area ratio.

2.6.1 NOZZLE CONFIGURATIONS

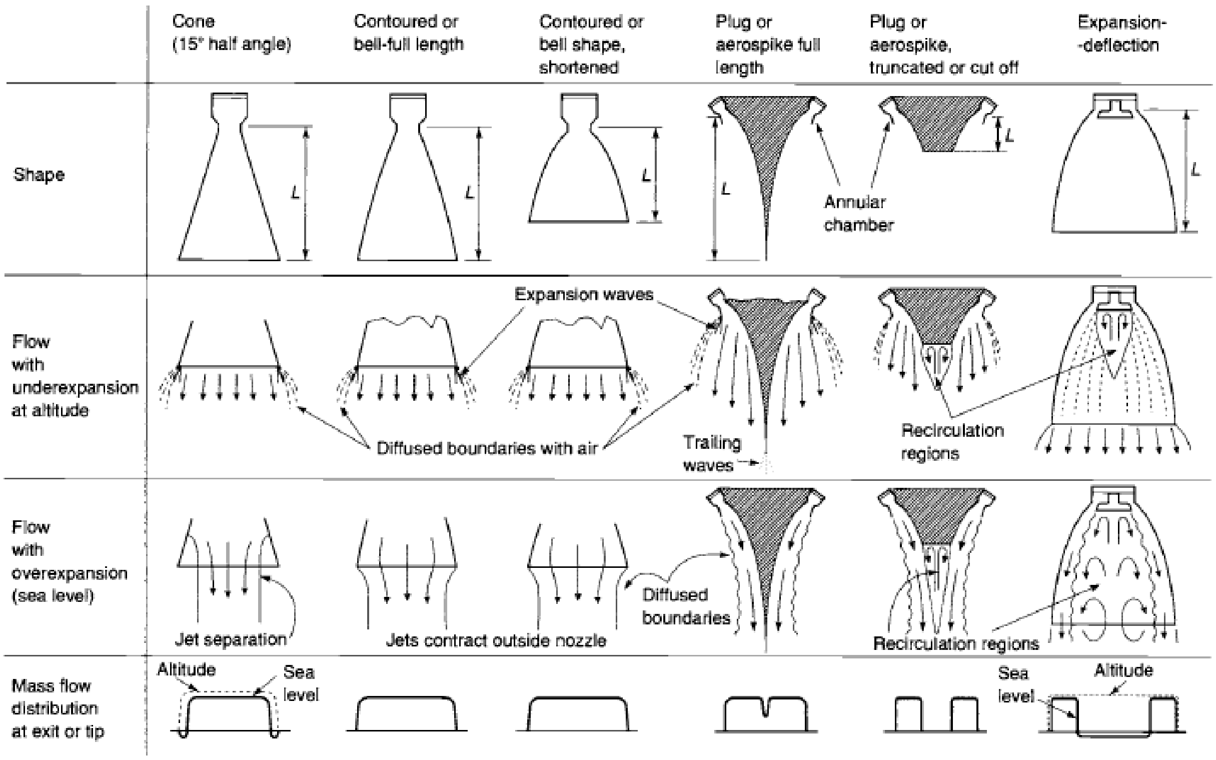


Figure2.4 Simplified diagrams of several different nozzle configurations and their flow effects.

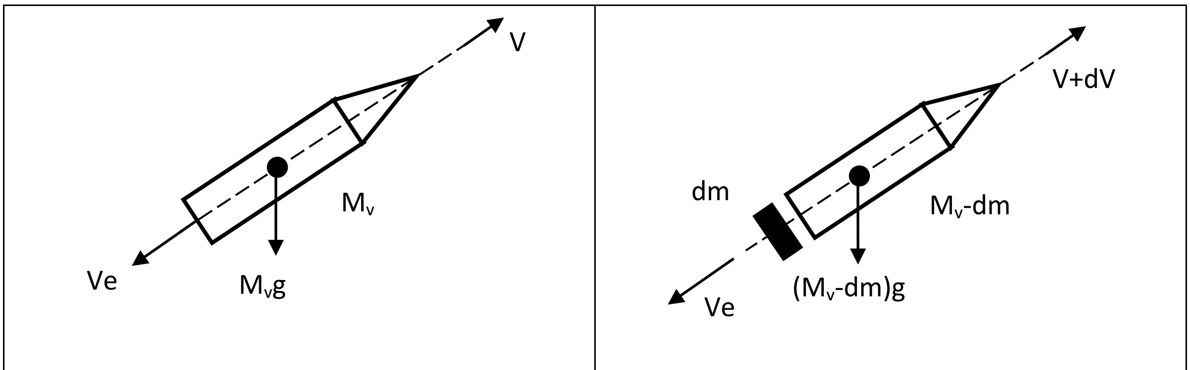
2.7 TSIOLKOVSKY'S ROCKET EQUATION:

A large fraction (typically ~90%) of the mass of a chemical rocket is propellant, thus it is important to consider the change in mass of the vehicle as it accelerates. The goal is to arrive at an expression which relates the change in velocity of a rocket to the change in its mass as well as any external forces that are acting on it. The analysis is performed using Newton's 2nd Law, Equation 1, which states that the time rate of change of momentum is equal to the sum of the forces acting on the system.

$$\sum \vec{F} = \frac{d}{dt}\left(m\vec{V}\right) \qquad \text{------1}$$

The resulting expression is called the Rocket Equation and it may be used to relate specific impulse to the performance of a rocket. There are several ways to do this by applying conservation of momentum.

The first step is to apply the momentum theorem differentially to a rocket in accelerating flight. In the figures below the coordinate system is aligned to the axis of the rocket and parallel to both the direction of flight and the direction of the exhaust velocity. The positive direction is aligned with the direction of flight and gravity acts perpendicular to the Earth's center and at an angle q relative to the body attached coordinate system.



System at t	System (Rocket + Expelled Mass) at t + dt

Figure 2.5 Mechanics and Thermodynamics of Propulsion

The basic idea is that at time t the rocket has a mass Mv and is traveling at a velocity (as measured by an inertial observer) of V. Note that both Mv and V are functions of time. During a small time increment, dt, the rocket has expelled a small mass, dm, such that at time t+dt, the mass of the rocket is Mv-dm. The small mass, dm, is expelled from the rocket at a velocity, Ve, relative to the rocket. The expulsion of this mass during the time dt leads to an increase in the velocity of the rocket such that V(t+dt)=V(t)+dV. The table below summarizes each of these terms:

	Time	Mass	Velocity	Momentum
Rocket	t	Mv	V	MvV
Rocket Only	t + dt	Mv - dm	V + dV	(Mv - dm)(V + dV)

Expelled Mass Only	t + dt	dm	V + dV - Ve	dm(V + dV - Ve)

Table 1: Summary of Initial and Final System Momentum in Inertial Frame

Pay careful attention to the final velocity and momentum of the expelled mass. The point to notice is that this velocity must be expressed in an inertial system. An observer stationed on the rocket would measure the velocity and momentum of the expelled mass as Ve and dmVe, respectively. However, recall that Equation 1 must be applied in an inertial reference frame, such as an observer located on the ground. An observer stationed on the ground would measure the velocity of the expelled mass as the vector sum of the velocity of the rocket (traveling in the positive direction) and the velocity of the expelled mass (traveling in the negative direction) relative to the rocket at time t+dt as V+dV-Ve..

Now write these terms as the change in momentum of the system from the final state (t+dt) and the initial state (t), where the system is the rocket plus incremental mass.

$$momentum_{final} = (M_V - dm)(V + dV) + dm(V + dV - V_e) \quad \ldots \qquad \ldots\ldots 2$$

$$momentum_{initial} = M_V V \qquad \ldots\ldots 3$$

$$momentum_{final} - momentum_{initial} = M_V dV - V_e dm \quad \ldots\ldots \qquad \ldots\ldots 4$$

Equation 1 can now be rewritten as:

$$\sum \vec{F} dt = M_v dV - V_e dm \qquad \ldots\ldots 5$$

Compare Equation 5 to the result of a control volume analysis and you will find that the result is, of course, identical. We can now look at two important cases involving the expressions for the change in momentum of the rocket system.

Case 1: No external surface or body forces acting on the rocket vehicle

In this case Equation 5 is equal to zero, and we can solve this expression for dV:

$$dV = \frac{V_e dm}{M_V} \qquad ----6$$

Also note that the incremental mass that was ejected from the vehicle may be written as:

$$dm = \dot{m} dt = -\frac{dM_V}{dt} dt \qquad \ldots 7$$

In this expression $\dot{m}$ is the propellant mass flow rate, and this expression simply says that the change in mass of the vehicle during dt (which is decreasing, hence the negative sign) is equal to the mass of the expelled mass dm. This makes sense from the conservation of mass standpoint. Putting Equation 7 into Equation 6 gives:

$$dV = -\frac{V_e}{M_V}\left(\frac{dM_V}{dt}dt\right) = -V_e\frac{dM_V}{M_V}$$ ….8

Here we again make the assumption that the exit velocity of the ejected mass is a constant. This expression is now ready for integration. The limits of integration on the left integral are from the initial velocity to the final velocity and the limits of integration for the expression on the right hand side of the equal sign are from the initial mass to the final mass. This is shown below:

$$\int_{V_0}^{V_f} dV = -V_e \int_{M_0}^{M_f} \frac{dM_V}{M_V}$$ …..9

$$V_{final} - V_{inital} = \Delta V = -V_e \ln\left(\frac{M_{final}}{M_{initial}}\right) = V_e \ln\left(\frac{M_{initial}}{M_{final}}\right)$$ ----10

The final mass, Mfinal, is sometimes referred to as the burnout mass, and as its name implies this is the mass of the rocket when all the fuel as been expended. We can define a ratio, R, that relates the initial mass to the final, burnout mass of the rocket as:

$$R \equiv \frac{M_{initial}}{M_{burnout}}$$ ----11

Putting this expression into equation 8, gives:

$$\Delta V = V_e \ln R$$ ----12

Again, recall the assumptions on Equation 12. No forces (pressure, drag, gravity, etc.) are acting on the vehicle.

Case 2: External surface and body forces acting on the rocket vehicle

In this case we will consider pressure forces of the non-deal expansion, as well as gravity and drag acting on the rocket. The sum of these forces is expressed below:

$$\sum F = (P_e - P_a)A_e - D - M_V g \cos\theta \quad \text{----13}$$

I have represented the drag force simply by D. The last term on the right hand side is the gravity term, and a valid question to ask is: What is the correct mass to use in this term? As we shall discuss below, the final result must be solved by integration over time and the mass of the vehicle should be updated at each time step, dt. One could use the mass at the beginning of the time step (which is done above), the end of the time step (Mv-dm) or an average of these two, which would be expressed as (Mv-dm/2). If the time step is sufficiently small, accurate results will be obtained in all cases.

Equation 5 holds exactly for the case with external forces, but the difference in momentum between the initial and final state of the system is not zero. The difference between the final momentum and the initial momentum of the system is equal to the impulse $\sum Fdt$. We can express this as:

$$M_V dV - V_e dm = [(P_e - P_a)A_e - D - M_V g \cos\theta]dt \quad \text{----14}$$

Again apply Equation 7 and rearrange some terms to yield:

$$M_V dV = [(P_e - P_a)A_e + \dot{m}V_e - D - M_V g \cos\theta]dt \quad \text{----15}$$

Next we can combine the pressure and momentum flux terms to an equivalent velocity, c:

$$c = V_e + \left(\frac{P_e - P_a}{\dot{m}}\right)A_e \quad \text{----16}$$

For the case where the exhaust pressure, pe, is equal to the ambient pressure, pa, we have:

$$dV = -V_e \frac{dM_V}{M_V} - \frac{D}{M_V}dt - g \cos\theta dt \quad \text{----17}$$

For the case where pe ≠ pa, replace Ve with c from Equation 16. Equation 17 is called the Rocket Equation. Neglecting drag and assuming vertical flight:

$$dV = -V_e \frac{dM_V}{M_V} - gdt \qquad \text{-----18}$$

Integrating we arrive at:

$$V = -V_e \ln\left(\frac{M_{final}}{M_{initial}}\right) - gt \qquad \text{-----19}$$

This is exactly the set of expressions derived in the text book in Section 10.3.

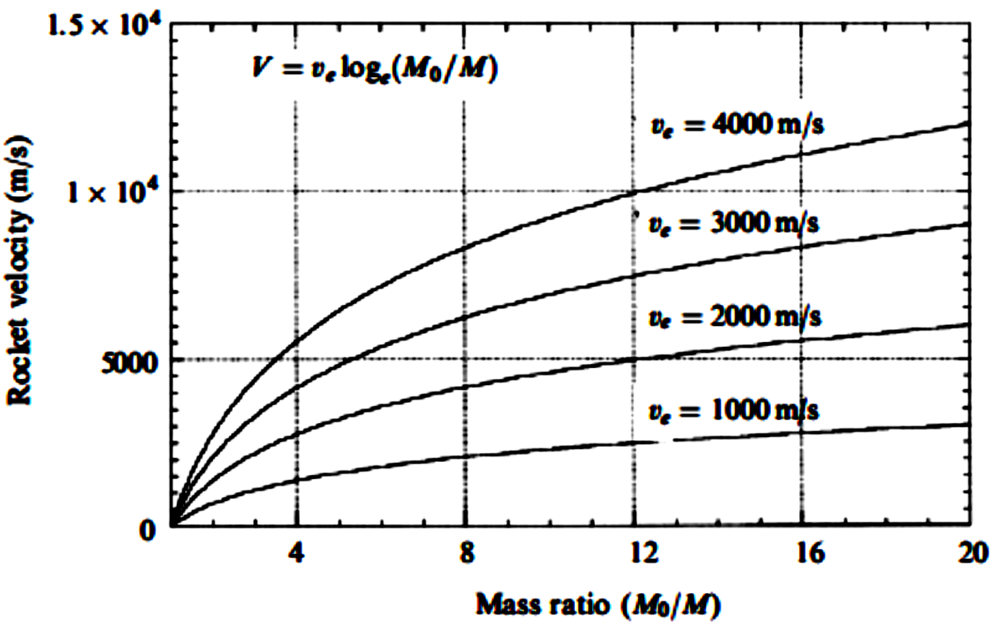
One other interesting aspect is to relate Equation 19 to the specific impulse, which is defined as the thrust divided by the fuel weight flow:

$$I_{sp} = \frac{T}{\dot{m}g} \approx \frac{\dot{m}V_e}{\dot{m}g} = \frac{V_e}{g} \qquad \text{-----20}$$

Substituting Equation 20 into Equation 19, we arrive at another useful form of the Rocket Equation:

$$V = g\left[I_{sp} \ln\left(\frac{M_{initial}}{M_{final}}\right) - t\right] \qquad \text{-----21}$$

We can view equation 19 as being similar to the Breguet Range Equation for aircraft. It presents the overall dependence of the principal performance parameters for a rocket (velocity, V), on the efficiency of the propulsion system (Isp), and the structural design (ratio of the total mass to structural mass – since the initial mass is the fuel mass plus the structural mass and the final mass is only the structural mass).



Source from Rocket Propulsion elements by George. P. Sutton

Figure 2.3 Tsiolkovsky's rocket equation

2.8 ROCKET ENGINEPERFORMANCE:

Tsiolkovsky was faced with the dynamics of a vehicle, the mass of which is decreasingasajetofmatterisprojectedrearwards.Asweshallseelater,theforce that projects the exhaust is the same force that propels the rocket. It partakes in Newton's third law—'action and reaction are equal and opposite', where 'action' meansforce.Theacceleratingforceisrepresented,usingNewton'slaw,as

$$F = Ma$$

Inthisequation,thethrustoftherocketisexpressedintermsofthe mass flow rate, m, and the effective exhaust velocity,ve.

So the energy released by the burning propellant appears as a fast-moving jet ofmatter, anda rocket accelerating in the opposite direction. Newton's law can be appliedtothisdynamicalsystem,andthedecreasingmasscanbetakenintoaccount, using some simple differential calculus. The resultant formula which Tsiolkovsky obtained for the vehicle velocity V is simple andrevealing,,

$$V = v_e \log_e \frac{M_0}{M}$$

Here M_0 is the mass of the rocket at ignition, and Mis the current mass of the rocket.Theonlyotherparametertoenterintotheformulaisve,theeffectiveexhaust velocity. This simple formula is the basis of all rocket propulsion. The velocity increaseswithtimeasthepropellantisburned.Itdependsonthenaturallogarithm of the ratio of initial to current mass; that is, on how much of the propellant has been burned. For a fixed amount of propellant burned, it also depends on the exhaust velocity—how fast the mass is beingexpelled.
This is shown in Figure 2.3, where the rocket velocity is plotted as a functionof themassratio.Themassratio,oftenwrittenasR,isjusttheratiooftheinitial to the currentmass,,

$$R = \frac{M_0}{M}$$

In most cases, the final velocity of the rocket needs to be known, and here the appropriatevalueisthemassratiowhenallthefuelisexhausted.Unlessotherwise stated, the final mass ratio should beassumed.
Therocketequationshowsthatthefinalspeeddependsupononlytwonumbers,,
thefinalmassratio,andtheexhaustvelocity.Itdoesnotdependonthethrust,rather
surprisingly,orthesizeoftherocketengine,orthetimetherocketburns,oranyotherparameter.Clearly,ahigherexhaustvelocityproducesahigherrocketvelocity,
andmuchoftheeffortinrocketdesigngoesintoincreasingtheexhaustvelocity
Gunpowder, and the range of propellants used for nineteenth century rockets,producedanexhaustvelocityaround2,000ms^{-1},oralittlemore.Themost advanced liquid-fuelled chemical rockets today produce an exhaust velocity of, at best,4,500ms^{-1}.Thereisnowhereelsetogo,,thisisclosetothetheoreticallimitof chemical energyextraction.
Toachieveahighrocketvelocity,themassratiohastobelarge.Themassratiois
definedastheratioofvehicle-plus-propellantmass,tovehiclemass.Intheseterms,a

massratioof,say,5indicatethat80%oftheinitialmassoftherocketisfuel.Thisis verydifferentfromacar,forinstance,whichhasatypicalemptymassof1.5tones, andafuelmassof40kg;amassratioof1.003.Soarocketvehicleisnothinglikeany otherkindofvehicle,becauseoftherequirementtohaveamassratioconsiderably greater than 1. The most obvious feature about a rocket like the Saturn V, or the SpaceShuttle,isitssheersizecomparedwithitspayload.TheSaturnVcarriedthree men on an eight-day journey, and weighed 3,000 tones. Most of this weight was fuel.

We know thattherocketcantravelfasterthanthespeedofits exhaust.Thisseemscounter-intuitivewhenthinkingintermsoftheexhaustpushing against something. In fact, the exhaust is not pushing against anything at all, and onceithasleftthenozzleoftherocketengineithasnofurthereffectontherocket. All the action takes place inside the rocket, where a constant accelerating forcesis being exerted on the inner walls of the combustion chamber and the inside of the nozzle.So,whilethespeedoftherocketdependsonthemagnitudeoftheexhaust velocity,itcanitselfbemuchgreater.Astationaryobserver sees the rocket and its exhaust passing by, both moving in the same direction, althoughtherocketismovingfasterthantheexhaust.

Thepointatwhichtherocket speed exceeds the exhaust speed is when the mass ratio becomes equal to e, or 2.718,the base of natural logarithms. It should also be kept in mind that theaccelerating forceisindependentofthespeedoftherocket;howeverfastitgoes,thethrustisstill thesame.Sowithaverylargemassratio,averyhighspeedcanbeattained.Abig enough rocket could, in principle, reach α Centauri within a fewcenturies.

Itisaswelltomentionherethatarocketcarriesbothitsfuelanditsoxidizer,and needs no intake of air to operate, like, for example, a jet engine. It can therefore function in a vacuum—and in fact works better, because air pressure retards the exhaust and reduces the thrust. It also works, rather inefficiently, under water, providedthatthecombustionchamberpressureexceedsthehydrostaticpressure; those who have cast a weighted firework into water can vouch for this.

Tsiolkovsky also calculated how fast a rocket needs to travel to reach space,

he realized, from the rocket equation, that there was a limit. It is obvious from Figure 2.3 that after a certain point, increasing the mass of fuel has a diminishing effectonthevelocitygain—notwithstandingwhatwehavesaidaboutαCentauri.If wetakethecurveforanexhaustvelocityof1,000ms^{-1}—alreadyaboutthespeedof sound—we can see that a speed of 3,000 m s^{-1} is about the limit that can be reasonably achieved. A higher mass ratio would produce a higher velocity, but withadiminishingreturn.Figure2.3hasawildlyoptimisticordinate,,amassratioof 10 is almost impossible to achieve, particularly with a sophisticated high exhaust velocity engine. Those working, at the moment, on single stage to orbit rockets, wouldbehappytoachieveamassratioofaround8.SowhileTsiolkovskywasable tocalculatethevelocityachievablebyaparticularrocket,wewouldnodoubthave beendisappointedwiththenumbersthatderivedfromhiscalculations.Heknewthat avelocityof11kms^{-1}wasneededtoescapetheEarth’sgravitationalfield.Faced with a gunpowder rocket, having at most about 2 km s^{-1} of exhaust velocity, the necessary mass ratio would have been wholly impossible toachieve.

Naturally, the first thing to do was to consider increasing the exhaust velocity. Tsiolkovskyknewthatthiswasamatterofcombustiontemperatureandmolecular weight,whichcouldbehandledbynineteenthcenturychemistry.Hequicklyrealized that liquid-fuelled rockets, using pure hydrogen and oxygen, could produce a considerable increase in exhaust velocity—in excess of 4,000 m s^{-1}.Referring to thegraph,escapevelocitybeginstoappearpossible.Amassratioofabout14isa less daunting task, but was still extremely difficult to achieve.

2.9 TEST FACILITIES AND SAFEGUARDS TES TEST FACILITIES AND SAFEGUARDS FACILITIES AND SAFEGUARDS:

For chemical rocket propulsion systems, each test facility usually has the followingmajor systems or components:

1. A test cell or test bay where the article to be tested is mounted, usually in a special test fixture. If the test is hazardous, the test facility must haveprovisions to protect operating personnel and to limit damage in case ofan accident.
2. An instrumentation system with associated computers for sensing, maintaining,measuring, analyzing, correcting, and recording various physicaland chemical parameters. It usually includes calibration systems andtimers to accurately synchronize the measurements.
3. A control system for starting, stopping, and changing the operating conditions.
4. Systems for handling heavy or awkward assemblies, supplying liquidpropellant, and providing maintenance, security, and safety.
5. For highly toxic propellants and toxic plume gases it has been required tocapture the hazardous gas or vapor (firing inside a closed duct system),remove almost all of the hazardous ingredients (e.g., by wet scrubbingand/or chemical treatment), allow the release of the nontoxic portion ofthe cleaned gases, and safely dispose of any toxic solid or liquid residues from the chemical treatment. With exhaust gas containing fluorine,for example, the removal of much of this toxic gas can be achieved byscrubbing it with water that contains dissolved calcium; it will then form calcium fluoride, which can be precipitated and removed.
6. In some tests specialized test equipment and unique facilities are neededto conduct static testing under different environmental conditions orunder simulated emergency conditions. For example, high and low ambienttemperature tests of large motors may require a temperature-controlledenclosure around the motor; a rugged explosion-resistantfacility is needed for bullet impact tests of propellant-loaded missile systemsand also for cook-off tests, where gasoline or rocket fuel is burnedwith air below a stored missile. Similarly, special equipment is needed for vibration testing, measuring thrust vector forces and moments in threedimensions, or determining total impulse for very short pulse durationsat low thrust.

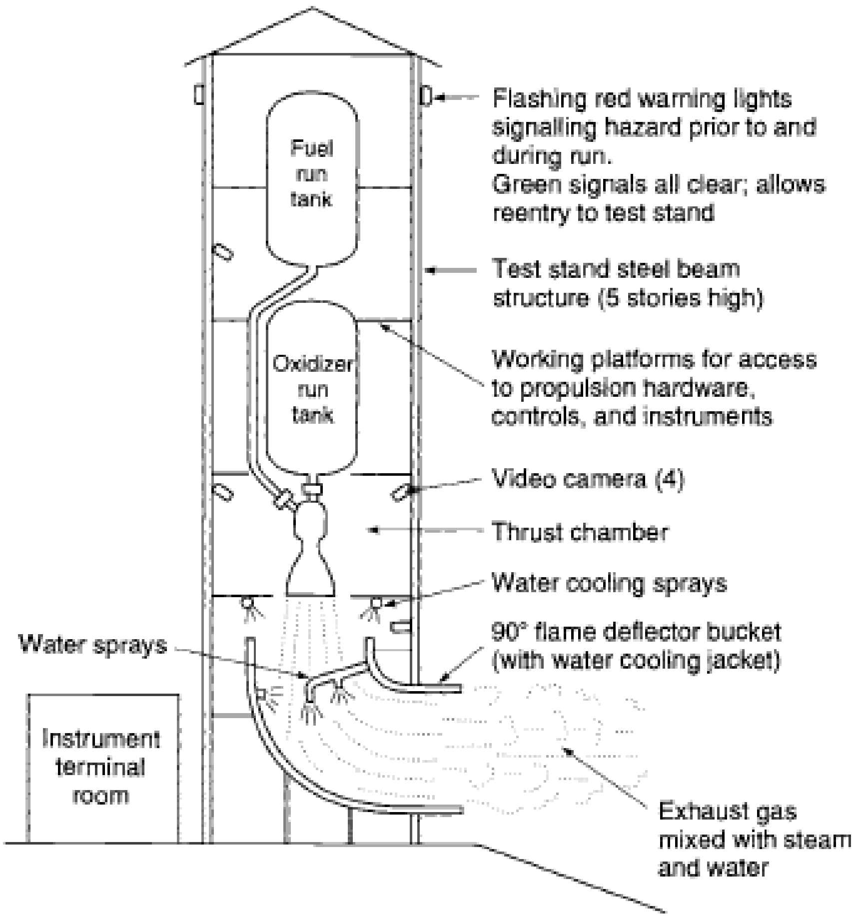


Figure 2.4 Simplified sketch of a typical static test stand for a large liquid propellantthrust chamber firing vertically downward

2.9.1 SAFETY PROVISIONS IN A MODERN TEST FACILITY INCLUDE THE FOLLOWING:

1. Concrete-walled blockhouse or control stations for the protection ofpersonnel and instruments remote from the actual rocketpropulsion location.
2. Remote control, indication, and recording of all hazardous operationsand measurements; isolation of propellants from the instrumentationand control room.
3. Automatic or manual water deluge and fire-extinguishing systems.
4. Closed circuit television systems for remotely viewing the test.
5. Warning signals (siren, bells, horns, lights, speakers) to notify personnelto clear the test area prior to a test, and an all-clear signal when theconditions are no longer hazardous.
6. Quantity and distance restrictions on liquid propellant tankage andsolid propellant storage to minimize damage in the event of explosions;separation of liquid fuels and oxidizers.
7. Barricades around hazardous test articles to reduce shrapnel damage inthe event of a blast.
8. Explosion-proof electrical systems, spark-proof shoes, and non-sparkhand tools to prevent ignition of flammable materials.
9. For certain propellants also safety clothing , includingpropellant- and fire-resistant suits, face masks and shields, gloves, specialshoes, and hard hats.
10. Rigid enforcement of rules governing area access, smoking, safetyinspections, and so forth.
11. Limitations on the number of personnel that may be in a hazardousarea at any time.

2.9.2 INSTRUMENTATION AND DATA MANAGEMENT:

Some of the physical quantities measuredin rocket testing are as follows:

1. Forces (thrust, thrust vector control side forces, short thrust pulses).
2. Flows (hot and cold gases, liquid fuel, liquid oxidizer, leakage).
3. Pressures (chamber, propellant, pump, tank, etc.).
4. Temperatures (chamber walls, propellant, structure, and nozzle).
5. Timing and command sequencing of valves, switches, igniters, etc.
6. Stresses, strains, and vibrations (combustion chamber, structures, propellantlines, accelerations of vibrating parts).
7. Time sequence of events (ignition, attainments of full pressure).
8. Movement and position of parts (valve stems, gimbal position, deflectionof parts under load or heat).Voltages, frequencies, and currents in electrical or control subsystems.
9. Visual observations (flame configuration, test article failures, explosions)
10. Using high-speed cameras or video cameras.

Special quantities such as turbopump shaft speed, liquid levels in propellanttanks, burning rates, flame luminosity, or exhaust gas composition.

UNIT-3
SOLID ROCKET PROPULSION

3.1 INTRODUCTION

Solid propellant motors are the simplest of all rocket designs. They consist of a casing, usually steel, filled with a mixture of solid compounds (fuel and oxidizer) that burn at a rapid rate, expelling hot gases from a nozzle to produce thrust. When ignited, a solid propellant burns from the center out towards the sides of the casing. The shape of the center channel determines the rate and pattern of the burn, thus providing a means to control thrust. Unlike liquid propellant engines, solid propellant motors cannot be shut down. Once ignited, they will burn until all the propellant is exhausted.

3.1.1 Salient features of solid propellant rockets:

There are two families of solids propellants: homogeneous and composite. Both types are dense, stable at ordinary temperatures, and easily storable.

Homogeneous propellants are either simple base or double base. A simple base propellant consists of a single compound, usually nitrocellulose, which has both an oxidation capacity and a reduction capacity. Double base propellants usually consist of nitrocellulose and nitroglycerine, to which a plasticiser is added. Homogeneous propellants do not usually have specific impulses greater than about 210 seconds under normal conditions. Their main asset is that they do not produce traceable fumes and are, therefore, commonly used in tactical weapons. They are also often used to perform subsidiary functions such as jettisoning spent parts or separating one stage from another.

Modern composite propellants are heterogeneous powders (mixtures) that use a crystallized or finely ground mineral salt as an oxidizer, often ammonium perchlorate, which constitutes between 60% and 90% of the mass of the propellant. The fuel itself is generally aluminum. The propellant is held together by a polymeric binder, usually polyurethane or polybutadienes, which is also consumed as fuel. Additional compounds are sometimes included, such as a catalyst to help increase the burning rate, or other agents to make the powder easier to manufacture. The final product is rubber like substance with the consistency of a hard rubber eraser.

Composite propellants are often identified by the type of polymeric binder used. The two most common binders are polybutadiene acrylic acid acrylonitrile (PBAN) and hydroxy-terminator polybutadiene (HTPB). PBAN formulations give a slightly higher specific impulse, density, and burn rate than equivalent formulations using HTPB. However, PBAN propellant is the more difficult to mix and process and requires an elevated curing temperature. HTPB binder is stronger and more flexible than PBAN binder. Both PBAN and HTPB formulations result in propellants that deliver excellent performance, have good mechanical properties, and offer potentially long burn times.

Solid propellant motors have a variety of uses. Small solids often power the final stage of a launch vehicle, or attach to payloads to boost them to higher orbits. Medium solids such as the Payload Assist Module (PAM) and the Inertial Upper Stage (IUS) provide the added boost to place satellites into geosynchronous orbit or on planetary trajectories.
The Titan, Delta, and Space Shuttle launch vehicles use strap-on solid propellant rockets to provide added thrust at liftoff. The Space Shuttle uses the largest solid rocket motors ever built and flown. Each booster contains 500,000 kg (1,100,000 pounds) of propellant and can produce up to 14,680,000 Newtons (3,300,000 pounds) of thrust.

3.2 SELECTION CRITERIA OF SOLID PROPELLANTS:

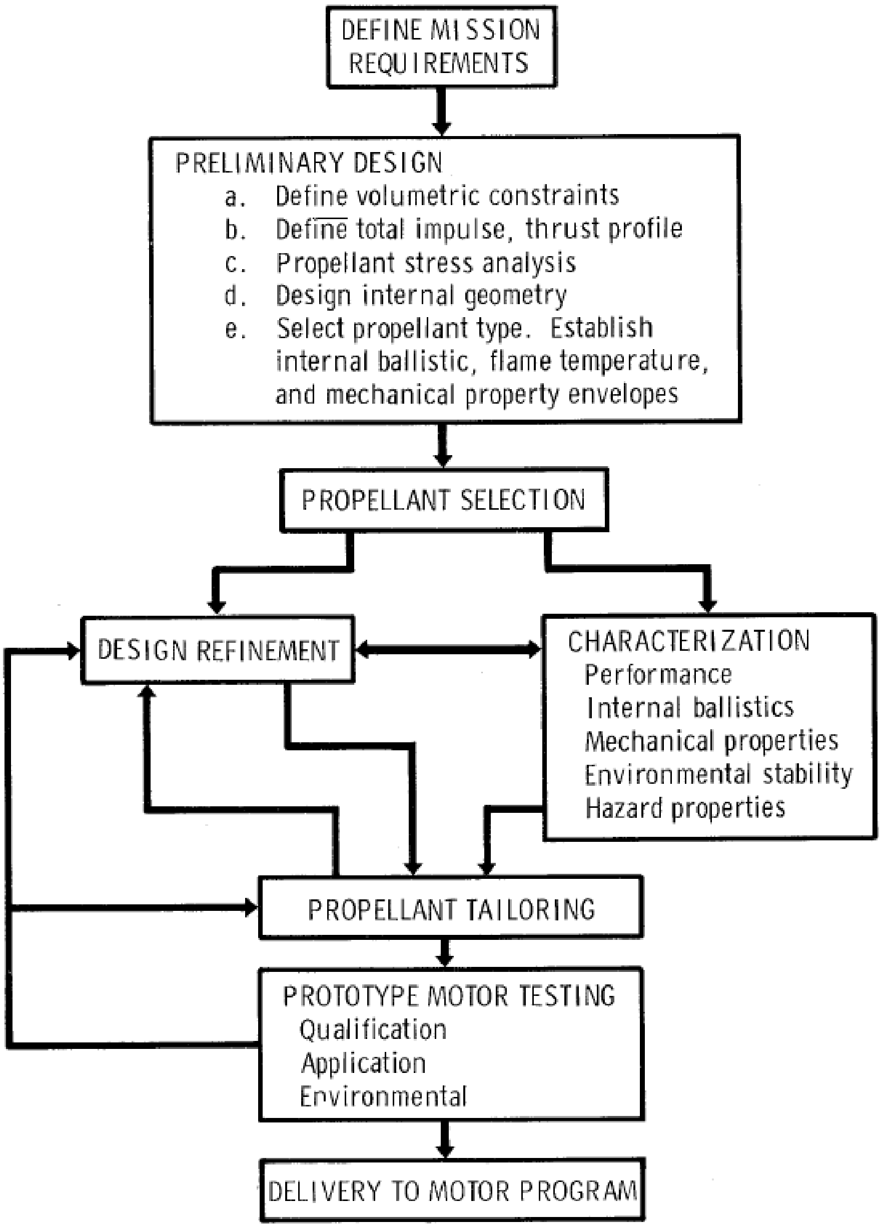


Figure 3.1 propellant selection and tailoring in design criteria

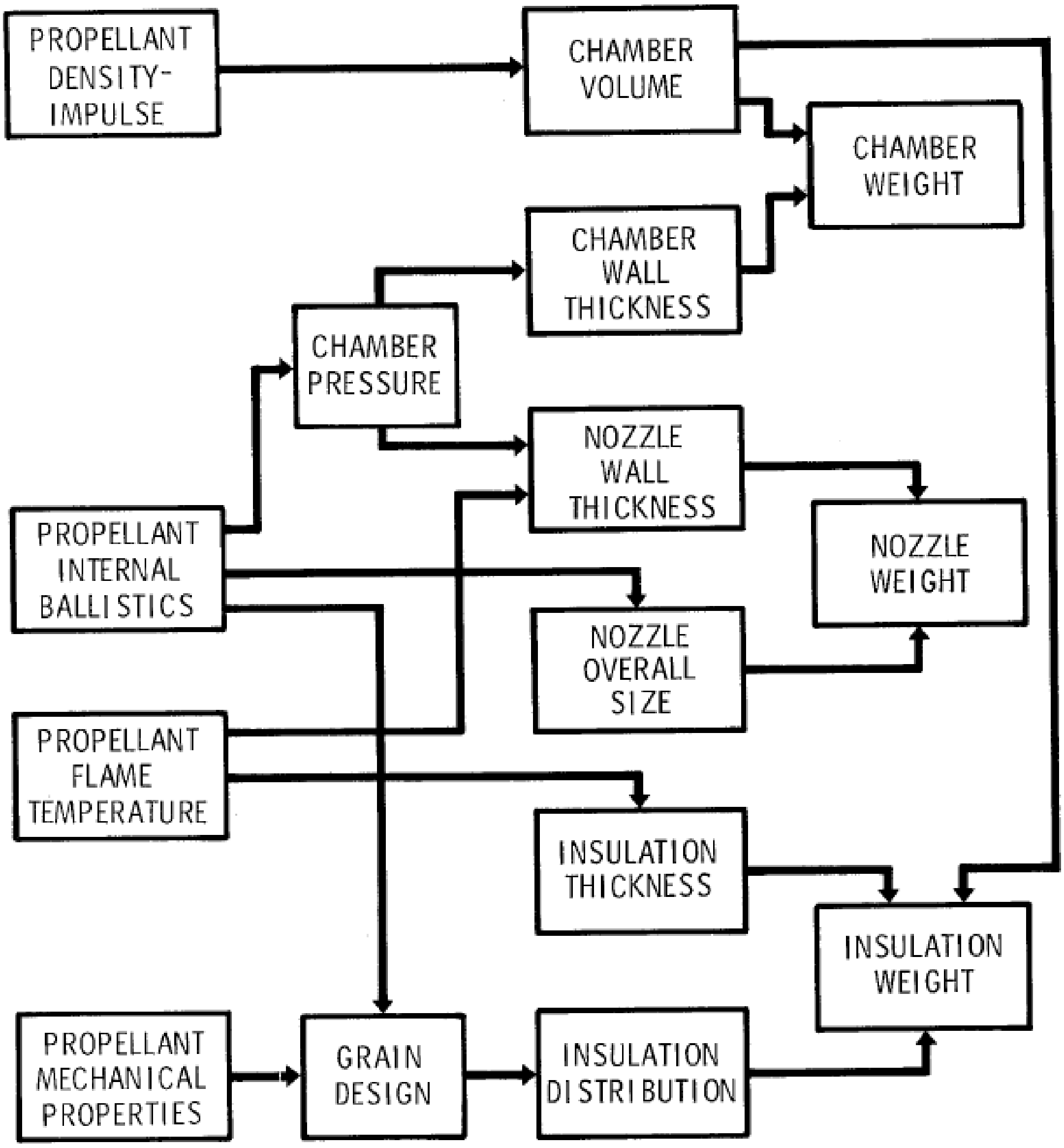


Figure 3.2: influence of propellant properties

3.3 ESTIMATION OF SOLID PROPELLANT ADIABATIC FLAME TEMPERATURE:

3.3.1 Adiabatic Flame Temperature: For a combustion process that takes place adiabatically with no shaft work, the temperature of the products is referred to as the adiabatic flame temperature. This is the maximum temperature that can be achieved for given reactants. Heat transfer, incomplete combustion, and dissociation all result in lower temperature. The maximum adiabatic flame temperature for a given fuel and oxidizer combination occurs with a stoichiometric mixture (correct proportions such that all fuel and all oxidizer are consumed). The amount of excess air can be tailored as part of the design to control the adiabatic flame temperature. The considerable distance between present temperatures in a gas turbine engine and the maximum adiabatic flame temperature at stoichiometric conditions is shown in Figure , based on a compressor exit temperature of $1200°\mathrm{F}$ (922 K).

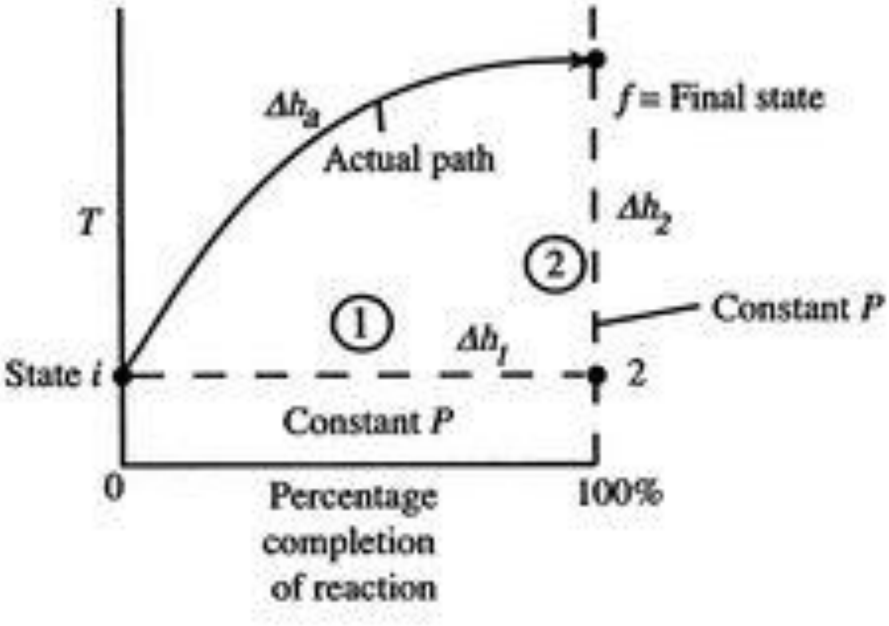


Figure 3.3: Schematic of adiabatic flame temperature

An initial view of the concept of adiabatic flame temperature is provided by examining two reacting gases, at a given pressure, and asking what the end temperature is. The process is shown schematically in Figure 3.3, where temperature is plotted versus the percentage completion of the reaction. The initial state is i and the final state is f , with the final state at a higher temperature than the initial state. The solid line in the figure shows a representation of the ``actual'' process.

To see how we would arrive at the final completion state the dashed lines break the state of reaction change into two parts. Process (1) is reaction at constant T and P . To carry out such a process, we would need to extract heat. Suppose the total amount of heat extracted per unit mass is q_1 . The relation between the enthalpy changes in Process (1) is

$$h_2 - h_i = -q_1 = (h_f^\circ)_{\text{unit mass}},$$

where q_1 is the ``heat of reaction.''

For Process (2), we put this amount back into the products to raise their temperature to the final level. For this process,

$$h_f - h_2 = q_1,$$

or, if we can approximate the specific heat as constant (using some appropriate average value)

$$c_{\mathrm{p,avg}}(T_f - T_2) = q_1.$$

For the overall process there is no work done and no heat exchanged so that the difference in enthalpy between initial and final states is zero:

$$\Delta h_1 + \Delta h_2 = \Delta h_{\mathrm{adiabatic}} = 0.$$

The temperature change during this second process is therefore given by (approximately)

$$(T_f - T_2) = \frac{q_1}{c_{\mathrm{p,avg}}} = \frac{|(h_f^\circ)_{\mathrm{unit\ mass}}|}{c_{\mathrm{p,avg}}}. \quad (1)$$

The value of the adiabatic flame temperature given in Equation (1) is for 100% completion of the reaction. In reality, as the temperature increases, the tendency is for the degree of reaction to be less than 100%. For example, for the combustion of hydrogen and oxygen, at high temperatures the combustion product (water) dissociates back into the simpler elemental reactants. The degree of reaction is thus itself a function of temperature that needs to be computed. We used this idea in discussing the stoichiometric ramjet, when we said that the maximum temperature was independent of flight Mach number and hence of inlet stagnation temperature. It is also to be emphasized that the idea of a constant (average) specific heat, $c_{\mathrm{p,avg}}$, is for illustration and not inherently part of the definition of adiabatic flame temperature.

3.4 PROPELLANT GRAIN DESIGN CONSIDERATIONS:

The rocket motor's operation and design depend on the combustion characteristics of the propellant, its burning rate, burning surface, and grain geometry. The branch of applied science describing these is known as internal ballistics; The burning surface of a propellant grain recedes in a direction essentially perpendicular to the surface. The rate of regression, usually expressed in cm/sec, mm/sec, or in./sec, is the burning rate r. In Fig. 3.5 we can visualize thechange of the grain geometry by drawing successive burning surfaces with a constant time interval between adjacent surface contours. Figure 3.5 shows this for a two-dimensional grain with a central cylindrical cavity with five slots. Success in rocket motor design and development depends significantly on knowledge of burning rate behavior of the selected propellant under allmotor operating conditions and design limit conditions.
Burning rate is afunction of the propellant composition. For composite propellants it can beincreased by changing the propellant characteristics:

1. Add a burning rate catalyst, often called burning rate modifier (0.1 to 3.0% of propellant) or increase percentage of existing catalyst.
2. Decrease the oxidizer particle size.
3. Increase oxidizer percentage.
4. Increase the heat of combustion of the binder and/or the plasticizer.
5. Imbed wires or metal staples in the propellant.

Aside from the propellant formulation and propellant manufacturing process, burning rate in a full-scale motor can be increased by the following:
1. Combustion chamber pressure.

2. Initial temperature of the solid propellant prior to start.
3. Combustion gas temperature.
4. Velocity of the gas flow parallel to the burning surface.
5. Motor motion (acceleration and spin-induced grain stress).

The grain is the shaped mass of processed solid propellant inside the rocket motor. The propellant material and geometrical configuration of the grain determine the motor performance characteristics. The propellant grain is a cast, molded, or extruded body and its appearance and feel is similar to that of hard rubber or plastic. Once ignited, it will burn on all its exposed surfaces to form hot gases that are then exhausted through a nozzle. A few rocket motors have more than one grain inside a single case or chamber and very few grains have segments made of different propellant composition (e.g., to allow different burning rates). However, most rockets have a single grain. There are two methods of holding the grain in the case, as seen in Fig. 3.4. Cartridge-loaded or freestanding grains are manufactured separately from the case (by extrusion or by casting into a cylindrical mold or cartridge) and then loaded into or assembled into the case. In case-bonded grains the case is used as a mold and the propellant is cast directly into the case and is bonded to the case or case insulation. Free-standing grains can more easily be replacedloaded) and a case-bonded grain.

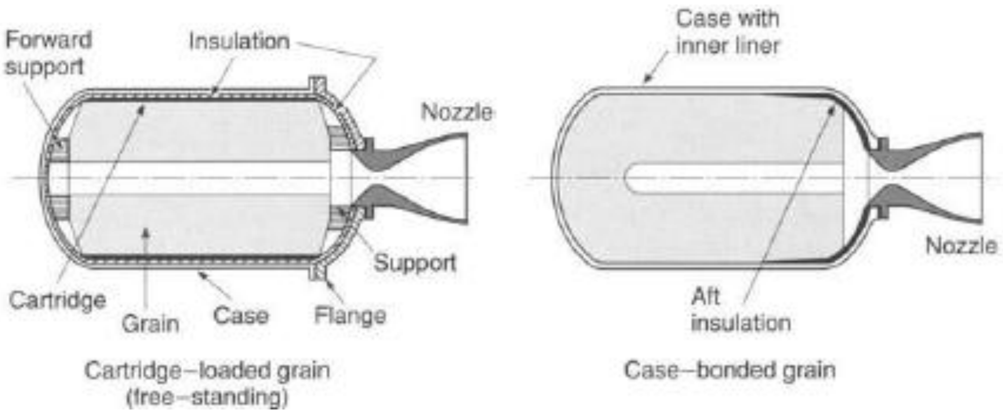


Fig. 3.4 Simplified schematic diagrams of a free-standing (or cartridge-loaded) and a case-bonded grain.

If the propellant grain has aged excessively. Aging is discussed in the next chapter. Cartridge-loaded grains are used in some small tactical missiles and a few medium-sized motors. They often have a lower cost and are easier to inspect. The case-bonded grains give a somewhat better performance, a little less inert mass (no holding device, support pads, and less insulation), a better volumetric loading fraction, are more highly stressed, and often somewhat more difficult and expensive to manufacture. Today almost all larger motors and many tactical missile motors use case bonding.

3.5 EROSIVE BURNING IN SOLID PROPELLANT ROCKETS

Definitions and terminology important to grains include:
Configuration: The shape or geometry of the initial burning surfaces of a grain as it is intended to operate in a motor.

Cylindrical Grain: A grain in which the internal cross section is constant along the axis regardless of perforation shape.
Neutral Burning: Motor burn time during which thrust, pressure, and burning surface area remain approximately constant, typically within about +15%. Many grains are neutral burning.
Perforation: The central cavity port or flow passage of a propellant grain; its cross section may be a cylinder, a star shape, etc.
Progressive Burning: Burn time during which thrust, pressure, and burning surface area increase.
Regressive Burning: Burn time during which thrust, pressure, and burning surface area decrease
Sliver: Unburned propellant remaining (or lost--that is, expelled through the nozzle) at the time of web burnout.

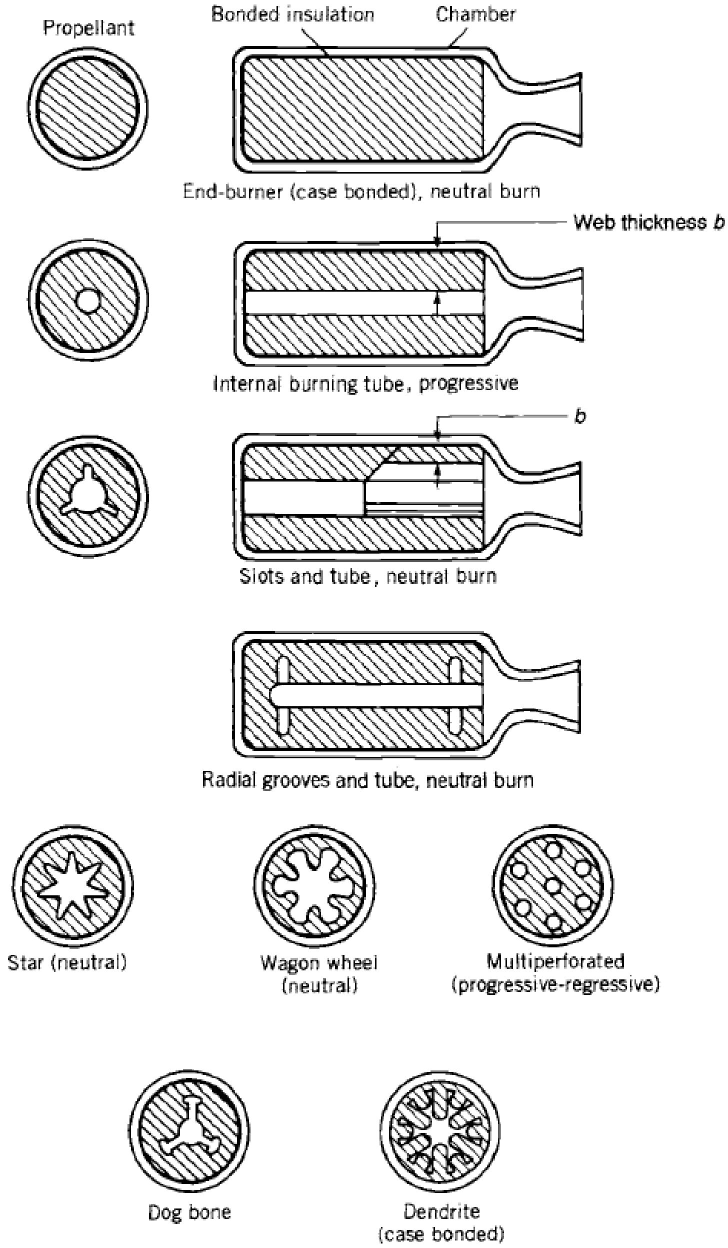


Figure3.5 classification of grain according to their pressure -time

Source from Rocket Propulsion elements by George. P. Sutton

A grain has to satisfy several interrelated requirements:

1. From the flight mission one can determine the rocket motor requirements. They have to be defined and known before the grain can be designed. They are usually established by the vehicle designers. This can include total impulse, a desired thrust-time curve and a tolerance thereon, motor mass, ambient temperature limits during storage and operation, available vehicle volume or envelope, and vehicle accelerations caused by vehicle forces (vibration, bending, aerodynamic loads, etc.).
2. The grain geometry is selected to fit these requirements; it should be compact and use the available volume efficiently, have an appropriate burn surface versus time profile to match the desired thrust-time curve, and avoid or predictably control possible erosive burning. The remaining unburned propellant slivers, and often also the shift of the center of gravity during burning, should be minimized. This selection of the geometry can be complex.
3. The propellant is usually selected on the basis of its performance capability (e.g., characteristic velocity), mechanical properties (e.g., strength), ballistic properties (e.g., burning rate), manufacturing characteristics, exhaust plume characteristics, and aging properties. If necessary, the propellant formulation may be slightly altered or "tailored" to fit exactly the required burning time or grain geometry.
4. The structural integrity of the grain, including its liner and/or insulator, must be analyzed to assure that the grain will not fail in stress or strain under all conditions of loading, acceleration, or thermal stress. The grain geometry can be changed to reduce excessive stresses.
5. The complex internal cavity volume of perforations, slots, ports, and fins increases with burning time. These cavities need to be checked for resonance, damping, and combustion stability.
6. The processing of the grain and the fabrication of the propellant should be simple and low cost.

3.6 COMBUSTION INSTABILITY:

There seem to be two types of combustion instability: a set of acoustic resonancesor pressure oscillations, which can occur with any rocket motor, and avortex shedding phenomenon, which occurs only with particular types ofgrains.

*3.6.1 Acoustic Instabilities:*When a solid propellant rocket motor experiences unstable combustion, thepressure in the interior gaseous cavities (made up by the volume of the port orperforations, fins, slots, conical or radial groves) oscillates by at least 5% andoften by more than 30% of the chamber pressure. When instability occurs, theheat transfer to the burning surfaces, the nozzle, and the insulated case walls isgreatly increased; the burning rate, chamber pressure, and thrust usuallyincrease; but the burning duration is thereby decreased. The change in thethrust-time profile causes significant changes in the flight path, and at timesthis can lead to failure of the mission. If prolonged and if the vibration energylevel is high, the instability can cause damage to the hardware, such as overheatingthe case and causing a nozzle or case failure. Instability is a conditionthat should be avoided and must be carefully investigated and remedied if itoccurs during a motor development program. Final designs of motors must befree of such instability.

There are fundamental differences with liquid propellant combustion behavior.In liquid propellants there is fixed chamber geometry with a rigid wall;liquids in feed systems and in injectors that are not part of the oscillating gas inthe combustion chamber can interact strongly with the pressure fluctuations. In solid propellant motors the geometry of the oscillating cavity increases in sizeas burning proceeds and there are stronger damping factors, such as solidparticles and energy-absorbing viscoelastic materials. In general, combustioninstability problems do not occur frequently or in every motor development,and, when they do occur, it is rarely the cause for a drastic sudden motorfailure or disintegration. Nevertheless, drastic failures have occurred.Undesirable oscillations in the combustion cavity propellant rocket motorsis a continuing problem in the design, development, production, and even long-term(10 yr) retention of solid rocket missiles. While acoustically "softer" thana liquid rocket combustion chamber, the combustion cavity of a solid propellantrocket is still a low-loss acoustical cavity containing a very large acousticalenergy source, the combustion process itself. A small fraction of the energyreleased by combustion is more than sufficient to drive pressure vibrations toan unacceptable level.

Combustion instability can occur spontaneously, often at some particulartime during the motor burn period, and the phenomenon is usually repeatablein identical motors. Both longitudinal and transverse waves (radial and tangential)can occur. The pressure oscillations increase in magnitude, and the thrustand burning rate also increase. The frequency seems to be a function of thecavity geometry, propellant composition, pressure, and internal flame field. Asthe internal grain cavity is enlarged and local velocities change, the oscillationoften abates and disappears. The time and severity of the combustion vibrationtend to change with the ambient grain temperature prior to motor operation.

For a simple grain with a cylindrical port area, the resonant transversemode oscillations (tangential and radial) correspond roughly for liquid propellant thrust chambers. The longitudinal or axialmodes, usually at a lower frequency, are an acoustic wave traveling parallelto the motor axis between the forward end of the perforation and the convergentnozzle section. Harmonic frequencies of these basic vibration modescan also be excited. The internal cavities can become very complex and caninclude igniter cases, movable as well as submerged nozzles, fins, cones, slots,star-shaped perforations, or other shapes, as described in the section on graingeometry ,determination of the resonant frequencies of complexcavities is not always easy. Furthermore, the geometry of the internalresonating cavity changes continually as the burning propellant surfacesrecede; as the cavity volume becomes larger, the transverse oscillation frequenciesare reduced.

The bulk mode, also known as the Helmholtz mode, L* mode, or chuffingmode, is not a wave mode as described above. It occurs at relatively lowfrequencies (typically below 150 Hz and sometimes below 1 Hz), and thepressure is essentially uniform throughout the volume. The unsteady velocityis close to zero, but the pressure rises and falls. It is the gas motion (in and outof the nozzle) that corresponds to the classical Helmholtz resonator mode,similar to exciting a tone when blowing across the open mouth of a bottle. It occurs at low values of L* , sometimes duringthe ignition period, and disappears when the motor internal volume becomeslarger or the chamber pressure becomes higher. Chuffing is the periodic lowfrequencydischarge of a bushy, unsteady flame of short duration (typically lessthan 1 sec) followed by periods of no visible flame, during which slow outgassingand vaporization of the solid propellant accumulates hot gas in thechamber. The motor experiences spurts of combustion and consequent pressurebuildup followed by periods of nearly ambient pressure. This dormantperiod can extend for a fraction of a second to a few seconds.

A useful method of visualizing unstable pressure waves is shown in Figs. It consists of a series of Fourier analyses of themeasured pressure vibration spectrum, each taken at a different time in theburning duration and displayed at successive vertical positions on a time scale,providing a map of amplitude versus frequency versus burning time. This figureshows a low-frequency axial mode and two tangential modes, whose frequencyis reduced in time by the enlargement of the cavity; it also shows the timing ofdifferent vibrations, and their onset and demise.

The initiation or triggering of a particular vibration mode is still not wellunderstood but has to do with energetic combustion at the propellant surface.

A sudden change in pressure is known to be a trigger, such as when a piece ofbroken-off insulation or unburned propellant flows through the nozzle andtemporarily blocks all or a part of the nozzle area (causing a momentarypressure rise).

The shifting balance between amplifying and damping factors changes duringthe burning operation and this causes the growth and also the abatement ofspecific modes of vibration. The response of a solid propellant describes thechange in the gas mass production or energy release at the burning surfacewhen it is stimulated by pressure perturbations. When a momentary highpressure peak occurs on the surface, it increases the instantaneous heat transferand thus the burning rate, causing the mass flow from that surface to alsoincrease. Velocity perturbations along the burning surface are also believed tocause changes in mass flow. Phenomena that contribute to amplifying thevibrations, or to gains in the acoustic energy, are:

1. The dynamic response of the combustion process to a flow disturbance or the oscillations in the burning rate. This combustion response can bedetermined from tests of T-burners. The response function depends on the frequency of these perturbations and the propellant formulation. The combustion response may not be in a phase with the disturbance. The effects of boundary layers on velocity perturbations have been investigated.
2. The interactions of flow oscillation with the main flow, similar to the basis for the operation of musical wind instruments or sirens.
3. The fluid dynamic influence of vortexes.

Phenomena that contribute to a diminishing of vibration or to damping areenergy-absorbing processes; they include the following:

1. Viscous damping in the boundary layers at the walls or propellant surfaces.
2. Damping by particles or droplets flowing in an oscillating gas/vapor flow is often substantial. The particles accelerate and decelerate by being "dragged" along by the motion of the gas, a viscous flow process that absorbs energy. The attenuation for each particular vibration frequency is an optimum at a particular size of particles; high damping for low frequency oscillation (large motors) occurs with relatively large solid particles (8 to 20 ~tm); for small motors or high-frequency waves the best damping occurs with small particles (2 to 6 ~tm). The attenuation drops off sharply if the particle size distribution in the combustion gas is not concentrated near the optimum for damping.
3. Energy from longitudinal and mixed transverse/longitudinal waves is lost out through the exhaust nozzle. Energy from purely transverse waves does not seem to be damped by this mechanism.
4. Acoustic energy is absorbed by the viscoelastic solid propellant, insulator, and the motor case; its magnitude is difficult to estimate.

The propellant characteristics have a strong effect on the susceptibility toinstability. Changes in the binder, particle-size distribution, ratio of oxidizer tofuel, and burn-rate catalysts can all affect stability, often in ways that are notpredictable. All solid propellants can experience instability. As a part of characterizinga new or modified propellant (e.g., determining its ballistic, mechanical,aging, and performance characteristics), many companies now alsoevaluate it for its stability behavior, as described below.

3.7 STRAND BURNER AND T-BURNER: In contrast with liquid rocket technology, an accepted combustion stabilityrating procedure does not now exist for full-scale solid rockets. Undertakingstability tests on large full-scale flight-hardware rocket motors is expensive, andtherefore lower-cost methods, such as subscale motors, T-burners, and othertest equipment, have been used to assess motor stability.

The best known and most widely used method of gaining combustion stability-related data is the use of a T-burner, an indirect, limited method that doesnot use a full-scale motor. Standard T-burner hasa 1.5-in. internal diameter double-ended cylindrical burner vented at its midpoint.

Venting can be through a sonic nozzle to theatmosphere or by a pipe connected to a surge tank which maintains a constantlevel of pressure in the burner cavity. T-burner design and usage usually concentrateon the portion of the frequency spectrum dealing with the transverse oscillations expected in a full-scale motor. The desired acoustical frequency, tobe imposed on the propellant charge as it burns, determines the burner length(distance between closed ends).

The nozzle location, midway between the ends of the burner, minimizesattenuation of fundamental longitudinal mode oscillations (in the propellantgrain cavity). Theoretically, an acoustic pressure node exists at the center andantinodes occur at the ends of the cavity. Acoustic velocity nodes are out ofphase with pressure waves and occur at the ends of the burner. Propellantcharges are often in the shape of discs or cups cemented to the end faces ofthe burner. The gas velocity in the burner cavity is kept intentionally low(Mach 0.2 or less) compared with the velocity in a full-scale motor. Thispractice minimizes the influence of velocity-coupled energy waves and allowsthe influence of pressure-coupled waves to be more clearly recognized.

Use of the T-burner for assessing the stability of a full-scale solid rocketpresupposes valid theoretical models of the phenomena occurring in both theT-burner and the actual rocket motor; these theories are still not fully validated.
In addition to assessing solid rocket motor combustion stability, the Tburneralso is used to evaluate new propellant formulations and the importanceof seemingly small changes in ingredients, such as a change in aluminumpowder particle size and oxidizer grind method.
Once instability has been observed or predicted in a given motor, themotor design has to fix the problem. There is no sure method for selecting theright remedy, and none of the cures suggested below may work. The usualalternatives are:

1. Changing the grain geometry to shift the frequencies away from the undesirable values. Sometimes, changing fin locations, port cross-section profile, or number of slots has been successful.
2. Changing the propellant composition. Using aluminum as an additive has been most effective in curing transverse instabilities, provided that the particle-size distribution of the aluminum oxide is favorable to optimumdamping at the distributed frequency. Changing size distributionand using other particulates (Zr, A1203, or carbon particles) has been effective in some cases. Sometimes changes in the binder have worked.
3. Adding some mechanical device for attenuating the unsteady gas motions or changing the natural frequency of cavities. Various inert resonance rods, baffles, or paddles have been added, mostly as a fix to an existingmotor with observed instability. They can change the resonance frequenciesof the cavities, introduce additional viscous surface losses, but also cause extra inert mass and potential problems with heat transfer or erosion.

Combustion instability has to be addressed during the design process,usually through a combination of some mathematical simulation, understandingsimilar problems in other motors, studies of possible changes, and supportingexperimental work (e.g., T-burners, measuring particle-sizedistribution). Most solid propellant rocket companies have in-house twoandthree-dimensional computer programs to calculate the likely acousticmodes (axial, tangential, radial, and combinations of these) for a given grain/motor, the initial and intermediate cavity geometries, and the combustiongas properties calculated from thermochemical analysis. Data on combustionresponse (dynamic burn rate behavior) and damping can be obtainedfrom T-burner tests. Data on particle sizes can be estimated from priorexperience or plume measurements.
Estimates of nozzle losses,friction, or other damping need to be included. Depending on the balancebetween gain and damping, it may be possible to arrive at conclusions on thegrain's propensity to instability for each specific instability mode that isanalyzed. If unfavorable, either the grain geometry or the propellant usuallyhas to be modified. If favorable, full-scale motors have to be built andtested to validate the predicted stable burning characteristics. There is alwaysa trade-off between the amount of work spent on extensive analysis, subscaleexperiments and computer programs (which will not always guarantee astable motor), and taking a chance that a retrofit will be needed after full-scalemotors have been tested. If the instability is not discovered until afterthe motor is in production, it is often difficult, time consuming, and expensiveto fix the problem.

3.8 APPLICATIONS AND ADVANTAGES OF SOLID PROPELLANT ROCKETS:

Major Application Categories for Solid Propellant Rocket Motors

Two general types of solid propellants are in use. The first, the so called double-base propellant, consists of nitrocellulose and nitroglycerine, plus additives in small quantity. There is no separate fuel and oxidizer. The molecules are unstable, and upon ignition break apart and rearrange themselves, liberating large quantities of heat. These propellants lend themselves well to smaller rocket motors. They are often processed and formed by extrusion methods, although casting has also been employed.

The other type of solid propellant is the composite. Here, separate fuel and oxidized chemicals are used, intimately mixed in the solid grain. The oxidizer is usually ammonium nitrate, potassium chlorate, or ammonium chlorate, and often comprises as much as four-fifths or more of the whole propellant mix. The fuels used are hydrocarbons, such as asphaltic-type compounds, or plastics. Because the oxidizer has no significant structural strength, the fuel must not only perform well but must also supply the necessary form and rigidity to the grain. Much of the research in solid propellants is devoted to improving the physical as well as the chemical properties of the fuel.

Ordinarily, in processing solid propellants the fuel and oxidizer components are separately prepared for mixing, the oxidizer being a powder and the fuel a fluid of varying consistency. They are then blended together under carefully controlled conditions and poured into the prepared rocket case as a viscous semisolid. They are then caused to set in curing chambers under controlled temperature and pressure.

Solid propellants offer the advantage of minimum maintenance and instant readiness. However, the more energetic solids may require carefully controlled storage conditions, and may offer handling problems in the very large sizes, since the rocket must always be carried about fully loaded. Protection from mechanical shocks or abrupt temperature changes that may crack the grain is essential.

UNIT IV
LIQUID AND HYBRID ROCKET PROPULSION

4.1 SALIENT FEATURES OF LIQUID PROPELLANT ROCKETS: The propellants, which are the working substance of rocket engines, constitutethe fluid that undergoes chemical and thermodynamic changes. The term liquidpropellant embraces all the various liquids used and may be one of thefollowing:

1. Oxidizer (liquid oxygen, nitric acid, etc.)
2. Fuel (gasoline, alcohol, liquid hydrogen, etc.).
3. Chemical compound or mixture of oxidizer and fuel ingredients, capableof self-decomposition.
4. Any of the above, but with a gelling agent.

A bipropellant rocket unit has two separate liquid propellants, an oxidizerand a fuel. They are stored separately and are not mixed outside the combustionchamber. The majority of liquid propellant rockets have been manufacturedfor bipropellant applications.
A monopropellant contains an oxidizing agent and combustible matter in asingle substance. It may be a mixture of several compounds or it may be ahomogeneous material, such as hydrogen peroxide or hydrazine.
Monopropellants are stable at ordinary atmospheric conditions but decomposeand yield hot combustion gases when heated or catalyzed.
A cold gas propellant (e.g., nitrogen) is stored at very high pressure, gives alow performance, allows a simple system and is usually very reliable. It hasbeen used for roll control and attitude control.
A cryogenic propellant is liquefied gas at low temperature, such as liquidoxygen (-183°C) or liquid hydrogen (-253°C). Provisions for venting thestorage tank and minimizing vaporization losses are necessary with this type.
Storable propellants (e.g., nitric acid or gasoline) are liquid at ambient temperatureand can be stored for long periods in sealed tanks. Space storablepropellants are liquid in the environment of space; this storability depends onthe specific tank design, thermal conditions, and tank pressure. An example is ammonia.
A gelled propellant is a thixotropic liquid with a gelling additive. It behaveslike a jelly or thick paint. It will not spill or leak readily, can flow underpressure, will burn, and is safer in some respects.

4.2 SELECTION OF LIQUID PROPELLANTS:

Mission Definition: Purpose, function, and final objective of the mission of an overall system are well defined andtheir implications well understood. There is an expressed need for the mission, and the benefitsare evident. The mission requirements are well defined. The payload, flight regime, vehicle,launch environment, and operating conditions are established. The risks, as perceived, appearacceptable. The project implementing the mission must have political, economic, andinstitutional support with assured funding. The propulsion system requirements, which arederived from mission definition, must be reasonable and must result in a viable propulsionsystem.

Affordability (Cost): Life cycle costs are low. They are the sum of R&D costs, production costs, facility costs,operating costs, and decommissioning costs, from inception to the retirement of the system. Benefits of achieving the mission should appear to justify costs. Investment innew facilities should be low. Few, if any, components should require expensive materials. Forcommercial applications, such as communications satellites, the return on investment mustlook attractive. No need to hire new, inexperienced personnel, who need to be trained and aremore likely to make expensive errors.

System Performance: The propulsion system is designed to optimize vehicle and system performance, using the mostappropriate and proven technology. Inert mass is reduced to a practical minimum, usingimproved materials and better understanding of loads and stresses. Residual (unused)propellant is minimal. Propellants have the highest practical specific impulse without unduehazards, without excessive inert propulsion system mass, and with simple loading, storing, andhandling. Thrust-time profiles and number of restarts must be selected tooptimize the vehicle mission. Vehicles must operate with adequate performance for all thepossible conditions (pulsing, throttling, temperature excursions, etc.). Vehicles should bestorable over a specified lifetime. Will meet or exceed operational life. Performance parameters(e.g., chamber pressure, ignition time, or nozzle area ratio) should be near optimum for theselected mission. Vehicle should have adequate TVC. Plume characteristics are satisfactory.

Survivability (Safety): All hazards are well understood and known in detail. If failure occurs, the risk of personnelinjury, damage to equipment, facilities, or the environment is minimal. Certain mishaps orfailures will result in a change in the operating condition or the safe shutdown of thepropulsion system. Applicable safety standards must be obeyed. Inadvertent energy input tothe propulsion system (e.g., bullet impact, external fire) should not result in a detonation. Theprobability for any such drastic failures should be very low. Safety monitoring and inspectionsmust have proven effective in identifying and preventing a significant share of possibleincipient failures. Adequate safety factors must be included in the design.Spilled liquid propellants should cause no undue hazards. All systems and procedures mustconform to the safety standards. Launch test range has accepted the system as being safeenough to launch.

Reliability: Statistical analyses of test results indicate a satisfactory high-reliability level. Technical risks,manufacturing risks, and failure risks are very low, well understood, and the impact on theoverall system is known. There are few complex components. Adequate storage and operatinglife of components (including propellants) have been demonstrated. Proven ability to checkout major part of propulsion system prior to use or launch. If certain likely failures occur, thesystem must shut down safely. Redundancy of key components should be provided, whereeffective. High probability that all propulsion functions must be performed within the desiredtolerances. Risk of combustion vibration or mechanical vibration should be minimal.

Controllability: Thrust buildup and decay are within specified limits. Combustion process is stable. The timeresponses to control or command signals are within acceptable tolerances. Controls need to befoolproof and not inadvertently create a hazardous condition. Thrust vector control responsemust be satisfactory. Mixture ratio control must assure nearly simultaneous emptying of thefuel and oxidizer tanks. Thrust from and duration of afterburning should be negligible.Accurate thrust termination feature must allow selection of final velocity of flight. Changingto an alternate mission profile should be feasible. Liquid propellant sloshing and pipeoscillations need to be adequately controlled. In a zero-gravity environment, a propellant tankshould be essentially fully emptied.

Maintainability: Simple servicing, foolproof adjustments, easy parts replacement, and fast, reliable diagnosis ofinternal failures or problems. Minimal hazard to service personnel. There must be easy accessto all components that need to be checked, inspected, or replaced. Trained maintenancepersonnel are available. Good access to items which need maintenance.

Geometric Constraints: Propulsion system fits into vehicle, can meet available volume, specified length, or vehiclediameter. There is usually an advantage for the propulsion system that has the smallestvolume or the highest average density. If the travel of the center of gravity has to becontrolled, as is necessary in some missions, the propulsion system that can do so withminimum weight and complexity will be preferred.

Prior Related Experience: There is a favorable history and valid, available, relevant data of similar propulsion systemssupporting the practicality of the technologies, manufacturability, performance, and reliability.Experience and data validating computer simulation programs are available. Experienced,skilled personnel are available.

Operability: Simple to operate. Validated operating manuals exist. Procedures for loading propellants,arming the power supply, launching, igniter checkout, and so on, must be simple. Ifapplicable, a reliable automatic status monitoring and check-out system should be available.Crew training needs to be minimal. Should be able to ship the loaded vehicle on public roadsor railroads without need for environmental permits and without the need for adecontamination unit and crew to accompany the shipment. Supply of spare parts must beassured. Should be able to operate under certain emergency and overload conditions.

Producibility: Easy to manufacture, inspect, and assemble. All key manufacturing processes are wellunderstood. All materials are well characterized, critical material properties are well known,and the system can be readily inspected. Proven vendors for key components have beenqualified. Uses standard manufacturing machinery and relatively simple tooling. Hardwarequality and propellant properties must be repeatable. Scrap should be minimal. Designs mustmake good use of standard materials, parts, common fasteners, and off-the-shelf components.There should be maximum use of existing manufacturing facilities and equipment. Excellentreproducibility, i.e., minimal operational variation between identical propulsion units.Validated specifications should be available for major manufacturing processes, inspection,parts fabrication, and assembly.

Schedule: The overall mission can be accomplished on a time schedule that allows the system benefits tobe realized. R&D, qualification, flight testing, and/or initial operating capability are completedon a preplanned schedule. No unforeseen delays. Critical materials and qualified suppliersmust be readily available.

Environmental Acceptability: No unacceptable damage to personnel, equipment, or the surrounding countryside. No toxicspecies in the exhaust plume. No serious damage (e.g., corrosion) due to propellant spills orescaping vapors. Noise in communities close to a test or launch site should remain withintolerable levels. Minimal risk of exposure to cancer-causing chemicals. Hazards must besufficiently low, so that issues on environmental impact statements are not contentious andapprovals by environmental authorities become routine. There should be compliance withapplicable laws and regulations. No unfavorable effects from currents generated by anelectromagnetic pulse, static electricity, or electromagnetic radiation.

Reusability: Some applications (e.g., Shuttle main engine, Shuttle solid rocket booster, or aircraft rocketassistedaltitude boost) require a reusable rocket engine. The number of flights, serviceability,and the total cumulative firing time then become key requirements that will need to bedemonstrated. Fatigue failure and cumulative thermal stress cycles can be critical in some ofthe system components. The critical components have been properly identified; methods,instruments, and equipment exist for careful check-out and inspection after a flight or test(e.g., certain leak tests, inspections for cracks, bearing clearances, etc.). Replacement and/orrepair of unsatisfactory parts should be readily possible. Number of firings before disassemblyshould be large, and time interval between overhauls should be long.

Other Criteria: Radio signal attenuation by exhaust plume to be low. A complete propulsion system, loadedwith propellants and pressurizing fluids, can be storable for a required number of yearswithout deterioration or subsequent performance decrease. Interface problems are minimal.Provisions for safe packaging and shipment are available. The system includes features thatallow decommissioning (such as to deorbit a spent satellite) or disposal (such as the saferemoval and disposal of over-age propellant from a refurbishable rocket motor).

4.3 INJECTOR: The functions of the injector are similar to those of a carburetor of an internal combustion engine. The injector has to introduce and meter the flow of liquid propellants to the combustion chamber, cause the liquids to be broken up into small droplets (a process called atomization), and distribute and mix the propellants in such a manner that a correctly proportioned mixture of fuel and oxidizer will result, with uniform propellant mass flow and composition over the chamber cross section. This has been accomplished with different types of injector designs and elements; several common types are shown in Fig. 4.1 and complete injectors are shown in Fig. 4.2.

The injection hole patternon the face of the injector is closely related to the internal manifolds or feed passages within the injector. These provide for the distribution of the propellant from the injector inlet to all the injection holes. A large complex manifold volume allows low passage velocities and good distribution of flow over the cross section of the chamber. A small manifold volume allows for a lighter weight injector and reduces the amount of "dribble" flow after the main valves are shut. The higher passage velocities cause a more uneven flow through different identical injection holes and thus a poorer distribution and wider local gas composition variation. Dribbling results in afterburning, which is an inefficient irregular combustion that gives a little "cutoff" thrust after valve closing. For applications with very accurate terminal vehicle velocity requirements, the cutoff impulse has to be very small and reproducible and often valves are built into the injector to minimize passagevolume.

Impinging-stream-type, multiple-hole injectorsare commonly used with oxygen-hydrocarbon and storable propellants. For *unlike doublet* patterns the propellants are injected through a number of separate small holes in such a manner that the fuel and oxidizer streams impinge upon each other. Impingement forms thin liquid fans and aids atomization of the liquids into droplets, also aiding distribution. Impinging hole injectors are also used for like-on-like or self-impinging patterns(fuel-on-fuel and oxidizer-on- oxidizer). The two liquid streams then form a fan which breaks up into droplets. Unlike doublets work best when the hole size (more exactly, the volume flow) of the fuel is about equal to that of the oxidizer and the ignition delay is long enough to allow the formation of fans. For uneven volume flow the triplet pattern seems to be more effective.

The non-impinging or shower head injector employs nonimpinging streams of propellant usually emerging normal to the face of the injector. It relies on turbulence and diffusion to achieve mixing. The German World War II V-2 rocket used this type of injector. This type is now not used, because it requires a large chamber volume for good combustion. Sheet or spray-type injectorsgive cylindrical, conical, or other types of spray sheets; these sprays generally intersect and thereby promote mixing and atomization. By varying the width of the sheet (through an axially moveable sleeve) it is possible to throttle the propellant flow over a wide range without excessive reduction in injector pressure drop. This type of variable area concentric tube injector was used on the descent engine of the Lunar Excursion Module and throttled over a 10:1 range of flow with only a very small change in mixture ratio.

The coaxial hollow post injectorhas been used for liquid oxygen and gaseous hydrogen injectors by most domestic and foreign rocket designers. It is shown in the lower left of Fig. 4.1. It works well when the liquid hydrogen has absorbed heat from cooling jackets and has been gasified. This gasified hydrogen flows at high speed (typically 330 m/sec or 1000 ft/sec); the liquid oxygen flows far more slowly (usually at less than 33 m/sec or 100 ft/sec) and the differential velocity causes a shear action, which helps to break up the oxygen stream into small droplets. The injector has a multiplicity of these coaxial posts on its face. This type of injector is not used with liquid storable bipropellants, in part because the pressure drop to achieve high velocity would become toohigh.

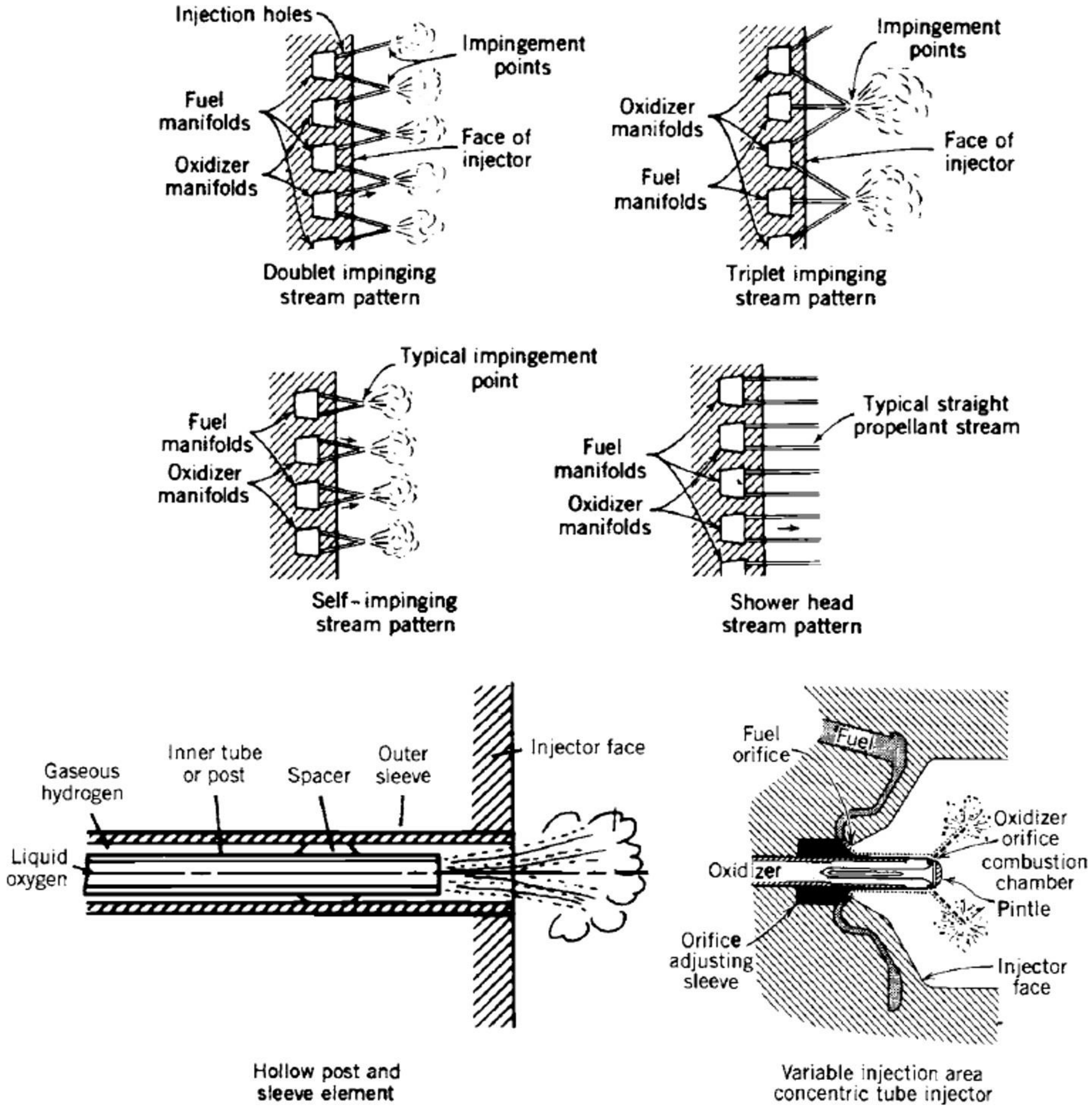


Figure 4.1 Schematic diagrams of several injector types. The movable sleeve type variable thrust injector

The SSME injector uses 600 concentric sleeve injection elements; 75 of them have been lengthened beyond the injector face to form cooled baffles, which reduce the incidence of combustion instability.

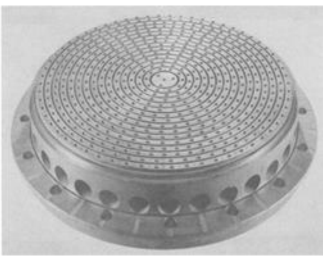
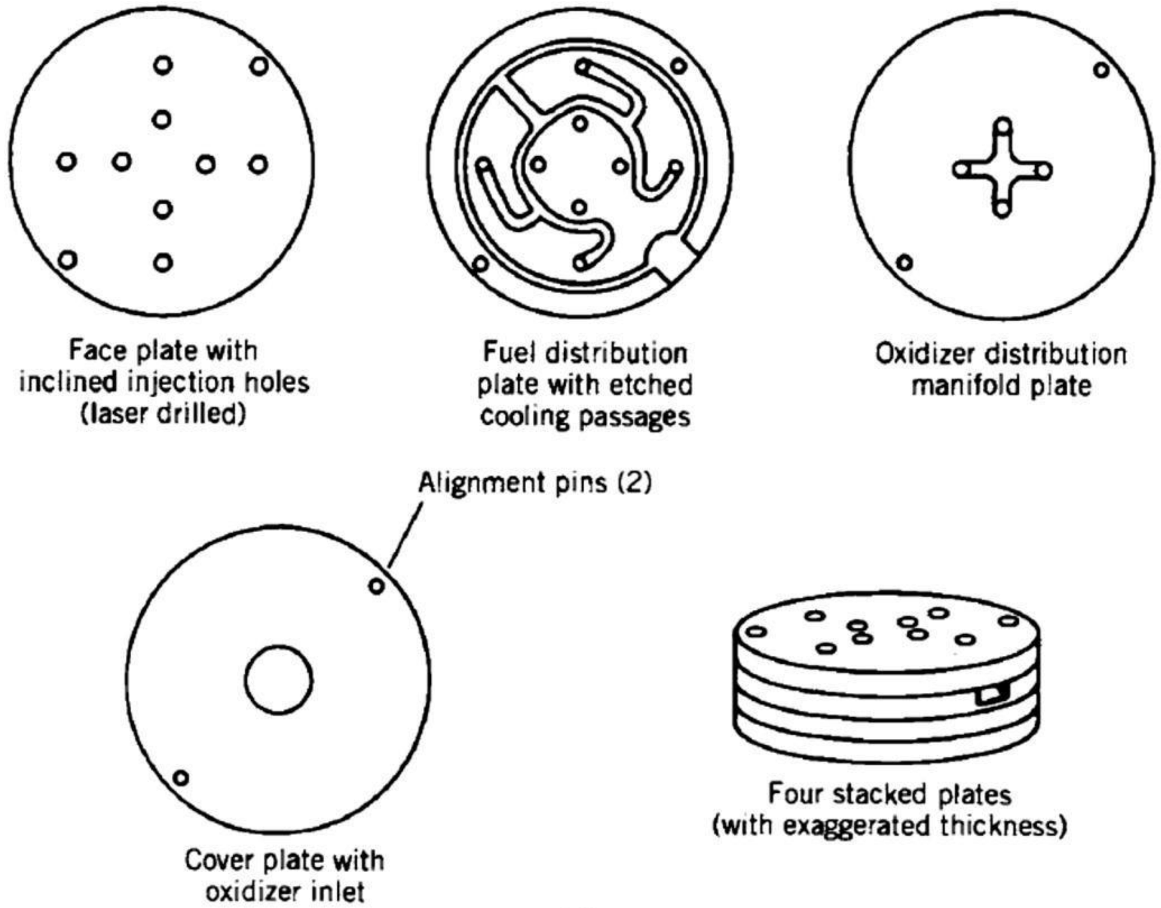


Figure 4.2: injector

Figure 4.2 Injector with 90° self-impinging (fuel-against-fuel and oxidizer-against oxidizer) -type countersunk doublet injection pattern. Large holes are inlets to fuel manifolds. Pre-drilled rings are brazed alternately over an annular fuel manifold or groove and a similar adjacent oxidizer manifold or groove.

The original method of making injection holes was to carefully drill them and round out or chamfer their inlets. This is still being done today. It is difficult to align these holes accurately (for good impingement) and to avoid burrs and surface irregularities. One method that avoids these problems and allows a large number of small accurate injecton orifices is to use multiple etched, very thin plates (often called platelets) that are then stacked and diffusion bonded together to form a monolithic structure as shown in Fig.4. 3. The photo-etched pattern on each of the individual plates or metal sheets then provides not only for many small injection orifices at the injector face, but also for internal distribution or flow passages in the injector and sometimes also for a fine-mesh filter inside the injector body. The platelets can be stacked parallel to or normal to the injector face. The finished injector has been called the platelet injector and has been patented by the Aero jet PropulsionCompany.



(a)

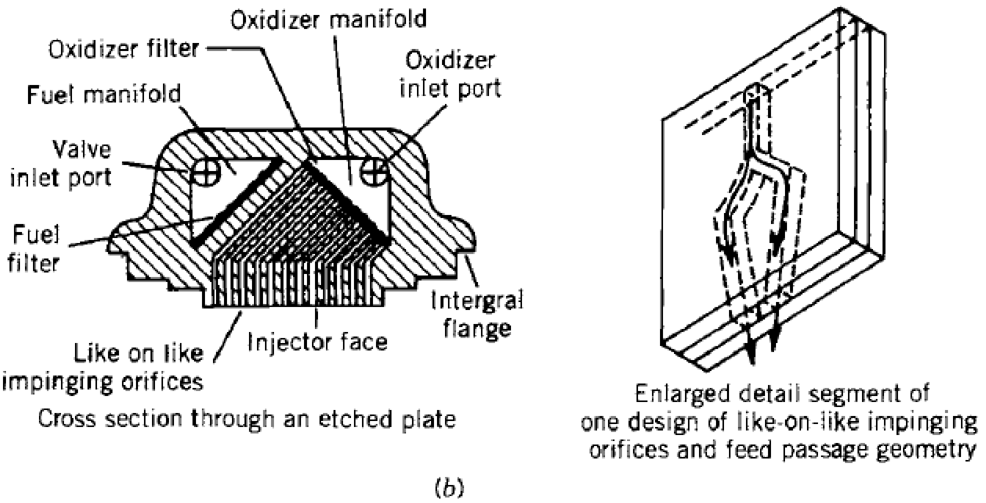


Figure 4.3Simplified diagrams of two types of injector using a bonded platelet construction technique: (a) injector for low thrust with four impinging unlike doubletliquid streams; the individual plates are parallel to the injector face; (b) Like-on-like impinging stream injector with 144 orifices; plates are perpendicular to the injector face.

4.4 PROPELLANT FEEDSYSTEMS:

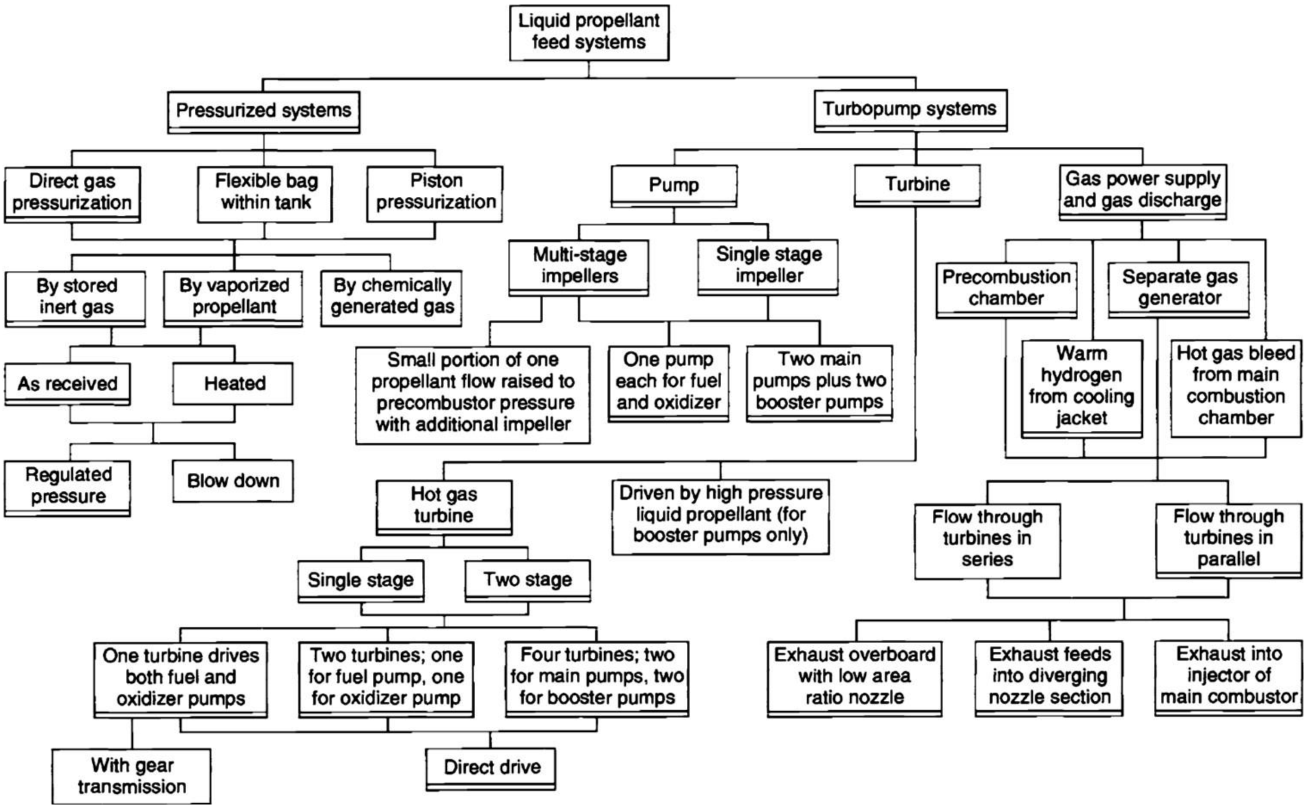


Figure 4.4Design options of fed systems for liquid propellant rocket engines. The more common types are designated with a double line at the bottom of the box.

The propellant feed system has two principal functions: to raise the pressure of the propellants and to feed them to one or more thrust chambers. The energy for these functions comes either from a high- pressure gas, centrifugal pumps, or a combination of the two. The selection of a particular feed system and its components is governed primarily by the application of the rocket, duration, number or type of thrust chambers, past experience, mission, and by general requirements of simplicity of design, ease of manufacture, low cost, and minimum inert mass. A classification of several of the more important types of feed system is shown in Fig. 4.4 and some are discussed in more detail below. All feed systems have piping, a series of valves, provisions for filling and removing (draining and flushing) the liquid propellants, and control devices to initiate, stop, and regulate their flow andoperation.

In general, a pressure feed system gives a vehicle performance superior to a turbopump system when the total impulse or the mass of propellant is relatively low, the chamber pressure is low, the engine thrust-to-weight ratio is low (usually less than 0.6), and when there are repeated short-duration thrust pulses; the heavy-walled tanks for the propellant and the pressurizing gas usually constitute the major inert mass of the engine system. In a turbopump feed systems the propellant tank pressures are much lower (by a factor of 10 to 40) and thus the tank masses are much lower (again by a factor of 10 to40).

Turbopump systems usually give a superior vehicle performance when the total impulse is large (higher Au) and the chamber pressure is higher. The pressurized feed system can be relatively simple, such as for a single operation, factory-preloaded, simple unit (with burst diaphragms instead of some of the valves), or quite complex, as with multiple restartable thrusters or reusable systems. If the propulsion system is to be reusable or is part of a manned vehicle (where the reliability requirements are very high and the vehicle's crew can monitor and override automatic commands), the feed system becomes more complex (with more safety features and redundancies) and more expensive.

The pneumatic (pressurizing gas) and hydraulic (propellant) flows in a liquid propellant engine can be simulated in a computer analysis that provides for a flow and pressure balance in the oxidizer and the fuel flow paths through the system. Some of these analyses can provide information on transient conditions (filling up of passages) during start, flow decays at cutoff, possible water hammer, or flow instabilities.

4.5 THRUST CONTROL COOLING IN LIQUID PROPELLANT ROCKETS AND THE ASSOCIATED HEAT TRANSFER PROBLEMS:

Liquid rocket engines developed for space missionsencompass a wide spectrum of performance and structuralrequirements.

Thrust levels may vary from a few Newtonsto many thousands of Newtons, with burning time fromfraction of a second to hours. In all these engines, theenergy released by the propellants must be contained insidethe thrust chamber and accelerated through the nozzle to extract the thrust. Extremely high heatflux levels andtemperature gradients are present not only in the immediatevicinity of the injector head, but also in the nozzle throatregion.

It is seen that the maximum heatflux occurs in theclose proximity to nozzle throat, and an effective cooling ofthe throat area is crucial for enhanced reliability andreusability. Regenerative cooling is the standard coolingsystem for almost all modern main stage, booster, and upperstage engines. Different cooling techniques such asfilmcooling, transpiration cooling, ablative cooling, radiationcooling, heat sink cooling and dump cooling have beendeveloped in the past to reduce regenerative cooling loadand propellant requirements. Film cooling can be employedeither at the combustion chamber or at the nozzle of arocket engine.

Liquidfilm cooling with fuel or oxidizer asthe coolant can be employed in the combustion chambers ofgas generator/expander/staged combustion cycle engines. Incase of gas generator cycle, the turbine exhaust gas can beused as a gaseousfilm coolant in the combustion chamberor nozzle sections. It is found that all these methods lead toreduced wall temperatures.The mechanism by whichfilm cooling maintains a lowercombustor wall temperature is considerably different fromthat of convective cooling. Film cooling is accomplished byinterposing a layer of coolantfluid between the surface tobe protected and the hot gas stream. Thefluid is introduceddirectly into the combustion chamber through slots or holesand is directed along the walls (Figure 4.5). A typicaltemperature distribution from the hot combustion gasesto the exterior of the chamber wall in afilm cooled

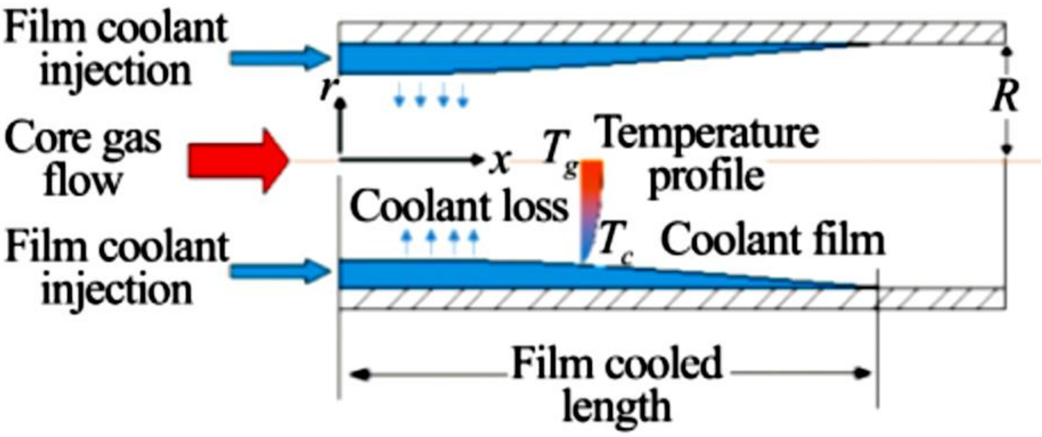


Figure 4.5: Schematic of the physical system

Combustion chamber is shown inFigure 4.6. It can beobserved that the coolantfilm produces a thermal insulationeffect and reduces the chamber wall temperature. Coolantfilm may be generated by injecting liquid fuel or oxidizerthrough wall slots or holes in the combustion chamber, orthrough the propellant injector. The cooling effect willpersist up to the throat region in the case of a shortercombustion chamber. In a fully film-cooled design, injection points are located at incremental distances along thewall length.

In liquidfilm cooling, the vaporizedfilmcoolant does not diffuse rapidly into the main gas streambut persists as a protective layer of vapor adjacent to thewall for an appreciable distance downstream from theterminus of the liquidfilm. Thefilm coolant also forms aprotectivefilm which restricts the transport of the combustion products to the wall, thus reducing the rate of oxidationof the walls.

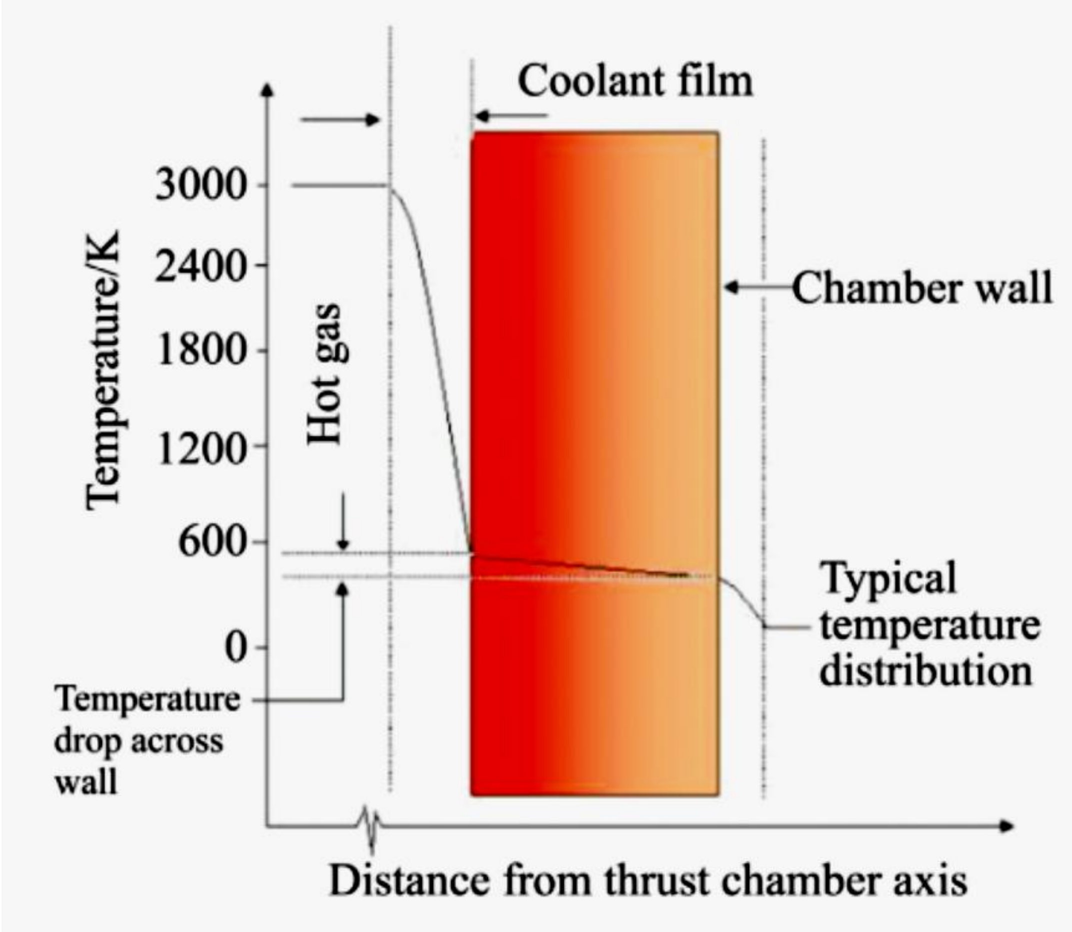


Figure 4.6 Typical temperature distribution of combustion chamberacross wall

4.6 COMBUSTION INSTABILITY IN LIQUID PROPELLANT ROCKETS:

If the process of rocket combustion is not controlled (by proper design), then combustion instabilities can occur which can very quickly cause excessive pressure vibration forces (which may break engine parts) or excessive heat transfer (which may melt thrust chamber parts). The aim is to prevent occurrence of this instability and to maintain reliable operation.

Although much progress has been made in understanding and avoiding combustion instability, new rocket engines can still be plagued by it.

Table below lists the principal types of combustion vibrations encountered in liquid rocket thrust chambers .Admittedly, combustion in a liquid rocket is never perfectly smooth; some fluctuations of pressure, temperature, and velocity are always present. When these fluctuations interact with the natural frequencies of the propellant feed system (with and without vehicle structure) or the chamber acoustics, periodic superimposed oscillations, recognized as instability, occur. In normal rocket practice smooth combustion occurs when pressure fluctuations during steady operation do not exceed about -t-5% of the mean chamber pressure. Combustion that gives greater pressure fluctuations at a chamber wall location which occur at completely random intervals is called rough combustion. Unstable combustion, or combustion instability, displays organized oscillations occurring at well-defined intervals with a pressure peak that may be maintained, may increase, or may die out. These periodic peaks, representing fairly large concentrationsof vibratory energy, can be easily recognized against the random-noise background in fig 4.7.

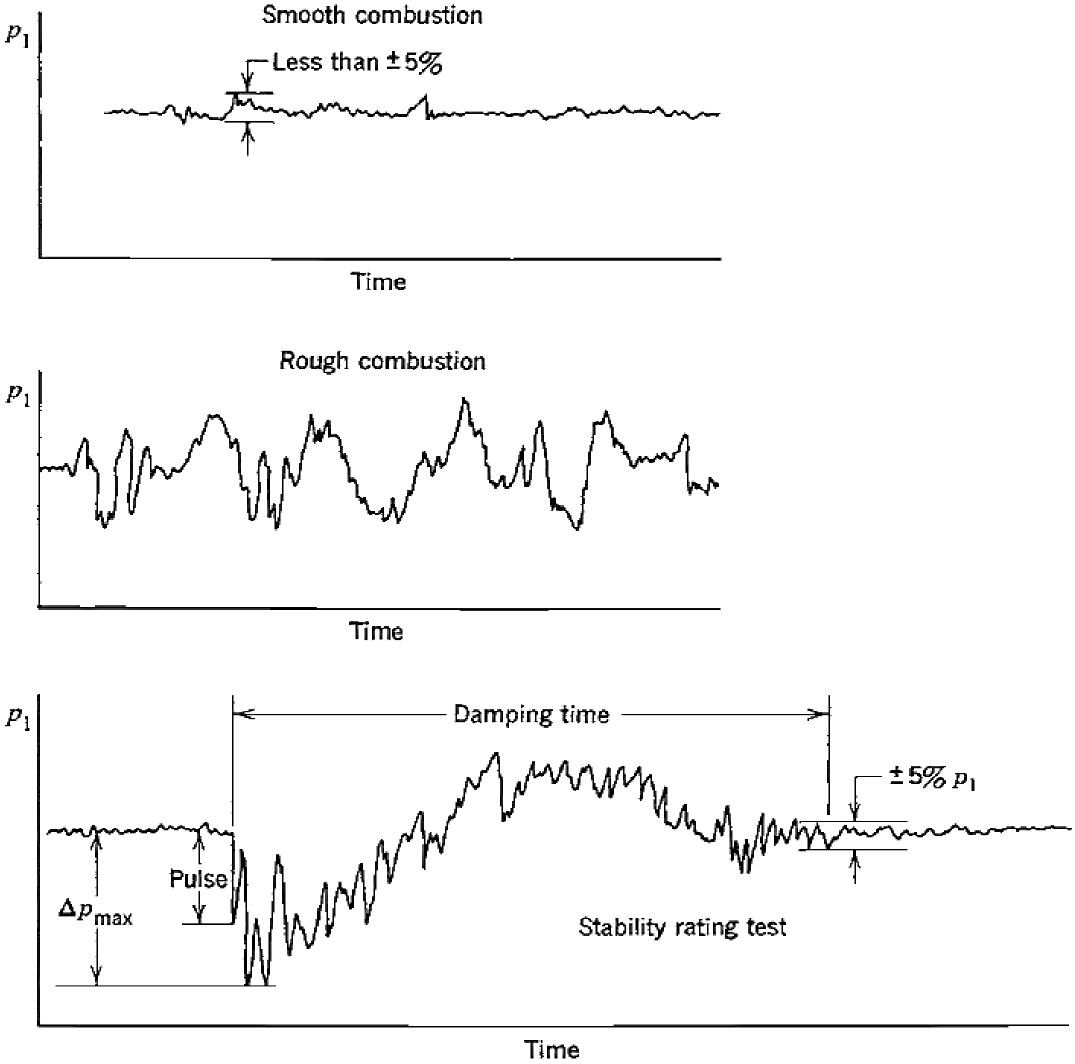


Fig. 4.7 Typical oscillogrpah traces of chamber pressure Pl with time for differentcombustion events.

4.7 PRINCIPAL TYPES OF COMBUSTION INSTABILITY:

Type and word description	Frequency	Cause relationship
Low frequency, called chugging or feed system instability	10-400	Linked with pressure interactionsbetween propellant feed system,if not the entire vehicle, andcombustion chamber
Intermediate frequency, called acoustic, a buzzing, or entropy waves	400-1000	Linked with mechanical vibrations of propulsion structure, injector manifold, flow eddies, fuel/oxidizer ratio fluctuations, and propellant feed system resonances
High frequency, called screaming, screeching, or squealing	Above 1000	Linked with combustion process forces (pressure waves) and chamber acoustical resonance properties

4.8 PROBLEMS ASSOCIATED WITH OPERATION OF CRYOGENIC ENGINES:

Peculiar problems associated with operation of cryogenic engines. The thrust comes from the rapid expansion from liquid to gas with the gas emerging from the motor at very highspeed. The energy needed to heat the fuels comes from burning them, once they are gasses.Cryogenic enginesare the highest performing rocket motors. Cryogenic engines are fundamentally different from electric motors because there isn't anything rotating in them. They're essentially reaction engines. By 'reaction' I'm referring to Newton's law: "to every action there is an equal and opposite reaction."The cryogenic (or rocket) engine throws mass in one direction, and the reaction to this is a thrust in the opposite direction.Therefore, to get the required mass flow rate, the only option was to cool the propellants down tocryogenictemperatures (below −183 °C [90 K], −253 °C [20 K]), converting them toliquid form. Hence, all cryogenic rocket engines are also, by definition, eitherliquid-propellant rocket enginesorhybrid rocket enginesIntroduction to hybrid rocket propulsion-standard and reverse hybrid systems-combustion mechanism in hybrid propellant rocketsapplications and limitations.

4.9 INTRODUCTION TO HYBRID ROCKET PROPULSION:

Rocket propulsion concepts in which one component of the propellant is stored in liquid phase while the other is stored in solid phase are called hybrid propulsion systems. Such systems most commonly employ a liquid oxidizer and solid fuel. Various combinations of solid fuels and liquid oxidizers as well as liquid fuels and solid oxidizers have been experimentally evaluated for use in hybrid rocket motors. Most common is the liquid oxidizer-solid fuel concept shown in Fig. 4.8. Illustrated here is a large pressure-fed hybrid booster configuration. The means of pressurizing the liquid oxidizer is not an important element of hybrid technology and a turbopump system could also perform this task. $ The oxidizer can be either a non-cryogenic (storable) or a cryogenic liquid, depending on the application requirements.

In this hybrid motor concept, oxidizer is injected into a precombustion or vaporization chamber upstream of the primary fuel grain. The fuel grain contains numerous axial combustion ports that generate fuel vapor to react withthe injected oxidizer. An aft mixing chamber is employed to ensure that all fuel and oxidizer are burned before exiting the nozzle.

The main advantages of a hybrid rocket propulsion system are:

(1) Safetyduring fabrication, storage, or operation without any possibility of explosionor detonation;
(2) start-stop-restart capabilities;
(3) Relatively low system cost;
(4) Higher specific impulse than solid rocket motors and higher density-specificimpulse than liquid bipropellant engines; and
(5) The ability to smoothly changemotor thrust over a wide range on demand.

The disadvantages of hybrid rocket propulsion systems are:
(1) Mixture ratioand, hence, specific impulse will vary somewhat during steady-state operationand throttling;
(2) lower density-specific impulse than solid propellant systems;
(3) some fuel sliver must be retained in the combustion chamber at end-ofburn,which slightly reduces motor mass fraction; and
(4) Unproven propulsionsystem feasibility at large scale.

4.11 STANDARD AND REVERSE HYBRID SYSTEMS :

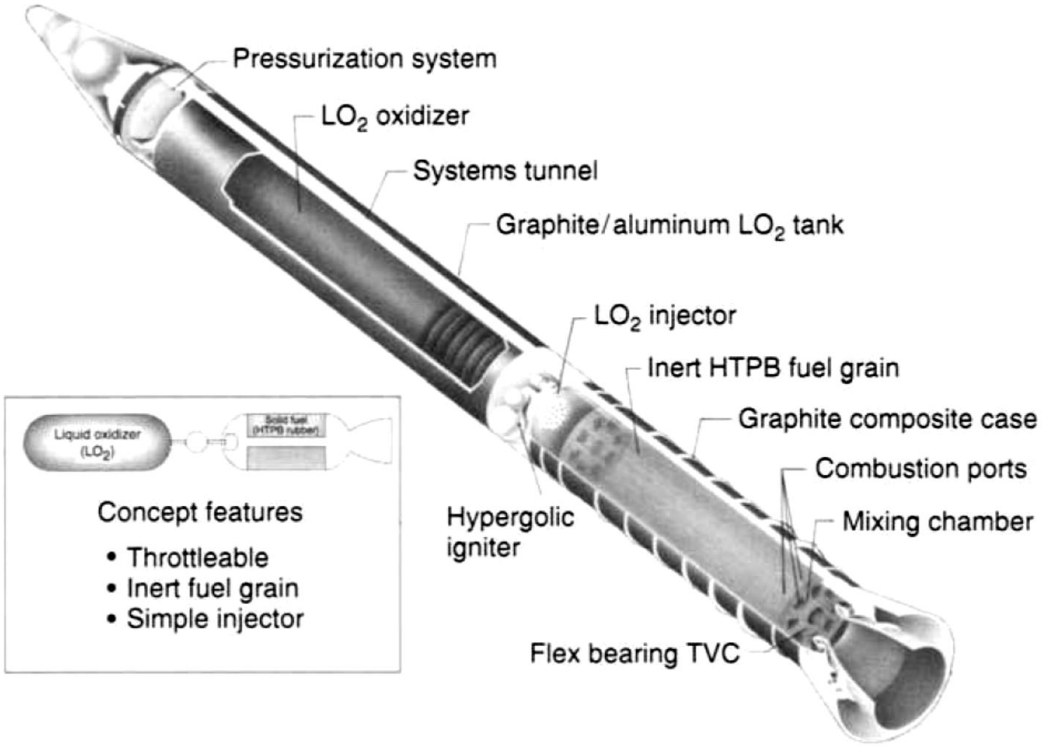


Figure 4.8. Large hybrid rocket booster concept capable of boosting the Space Shuttle. It has an inert solid fuel grain, pressurized liquid oxygen feed system, and can be throttled.

4.12 :HYBRID ROCKET CONFIGURATION:

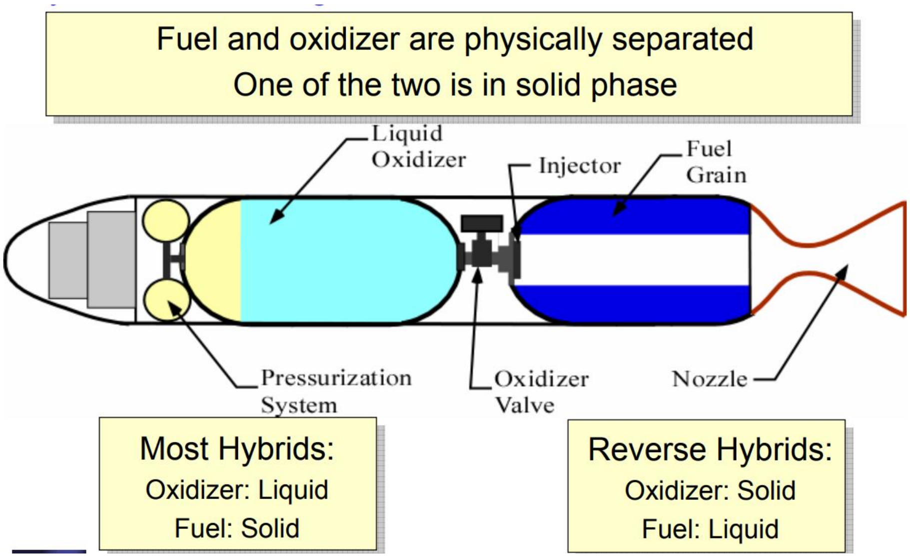


Figure 4.9 hybrid rocket

The hybrid is inherently safer than other rocket designs.The idea is to store the oxidizer as a liquid and the fuel as a solid, producing a designthat is less susceptible to chemical explosion than conventional solid and bi-propellantliquid designs. The fuel is contained within the rocket combustion chamber in the formof a cylinder with a circular channel called a port hollowed out along its axis. Uponignition, a diffusion flame forms over the fuel surface along the length of the port. Thecombustion is sustained by heat transfer from the flame to the solid fuel causing continuousfuel vaporization until the oxidizer flow is turned off. In the event of a structural failure,oxidizer and fuel cannot mix intimately leading to a catastrophic explosion that mightendanger 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 boththe chamber and nozzle. Throttling control in a hybrid is simpler because it alleviatesthe requirement to match the momenta of the dual propellant streams during the mixingprocess. Throttle ratios up to 10 have been common in hybrid motors. The fact that thefuel is in the solid phase makes it very easy to add performance enhancing materials to thefuel such as aluminum powder. In principle, this could enable the hybrid to gain an I_{sp}advantage over a comparable hydrocarbon fueled liquid system.

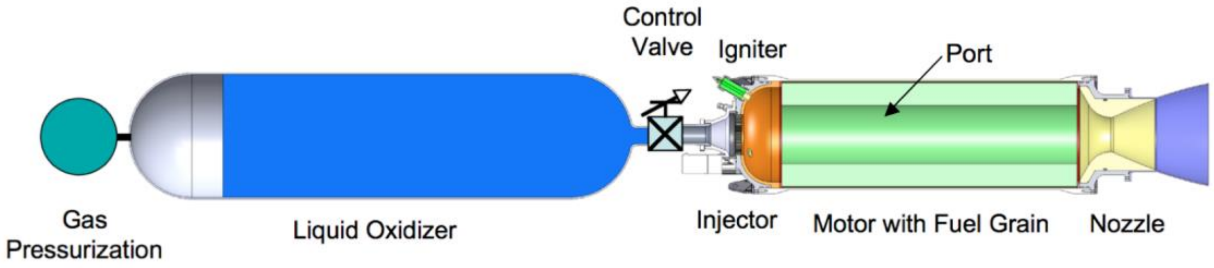


Fig. 4.10 Schematic of a hybrid rocket motor.

4.13: COMBUSTION MECHANISM IN HYBRID PROPELLANT ROCKETS

The process in a hybrid rocket combustion chamber over a large portion of the chamber constitutes the diffusive combustion in the boundary layer very close to the regressing fuel surface. The initial part is dominated by processes of impingement of the liquid oxidizer (which is sprayed by an injector) on the fuel surface and its vaporisation .Since the process is essentially diffusion-dominated, chemical kinetics, and therefore pressure, have a relatively smaller effect. It is the flux of hot gases (consisting of the products of combustion and oxidizer not yet utilised) past the fuel surface which primarily affects the regression of the fuel. The regression rate law can be deduced from the boundary layer considerations as was first accomplished by Marxman & Gilbert (1963). The law reads that rh=aG', where G is the mass flux of hot gases past the surface, a and n are constants; typically a 0.01--0.03 cm/s (flux) n, n=0.5 for laminar flow and 0.8 for turbulent flow.

Now it is possible to explain some features of hybrid engines. First,the O/F of a hybrid engine is not necessarily constant throughout the firing. Usingthe expression, we can express O/F in terms of the inner diameter of the cylindricalfuel block burning from inside outwards as

$$O/F \simeq d^{2n-1}\,(m_{ox}/\pi)^{1-n}/\rho_p aL,$$

Where d=inner diameter of the port and L is the length of the fuel block. If n=0.5, asin the ease of laminar flow, O/F is a constant and does not change during the firing;and if n =0.8, as in the case of turbulent flow, the value of O/F increases during thefiring, showing that the products become oxidiser-rich. This, in fact, causes changesin the specific impulse of the system during the firing. There are ways of combatingthis problem (see Anon 1964). One of these is to fix the initial operating point on aslightly fuel-rich side so that when the operating point moves to the oxidiser-rich side,the specific impulse does not vary by more than 1-2 Yo. Another technique whichmaintains a constant O/F level is to use two oxidiser injection points, one near thehead end and the other near the aft end. In the early part of the firing the burning isfuel-rich and aft end injection is used to optimize it. During the later part of thefiring the products of combustion from the end of the fuel block tend towardsoxidiser-richness and so aft-end injection is reduced to maintain the same O/Flevel.

The second feature concerns the low explosion hazard during storage, transportationand firing. That the explosion and fire hazard are small compared to that fora solid rocket is easy to appreciate because the solid rocket has the fuel and oxidizer imbedded in the same matrix whereas the hybrid has the solid fuel and liquid oxidizer separately stored. And in the event of an accidental initiation, the former can burnby itself, whereas the fuel in the hybrid rocket has to receive oxidiser for its combustion.

The fire hazard of the hybrid is smaller than of liquids because, in the eventof an explosion, the liquids of a liquid rocket can flow, widely spread and get mixedup, while the fuel and oxidiser in the hybrid have greater resistance to large scalemixing since the fuel is in the form of a solid.

We can further argue that a crack or a tiny hole in the fuel block of a hybrid causeslittle or no change in the performance, whereas the same crack or hole in a solidpropellant may cause explosion. To appreciate this we notice that the regression ofthe fuel occurs under the action of a heat flux from the diffusion flame. Thus any areaof the fuel which receives less heat flux will regress less. The tiny hole represents azone which is farther away from the flame and hence receives less heat flux and so willregress less. This means that the tiny hole evens out instead of becoming larger asin a solid rocket and the perturbation due to changed mass flow becomes small asburning progresses.

The third feature concerns the sensitivity of the regression rate to the nature offuel. It has already been noted that addition of even a small amount of some compoundscan disastrously alter the burning rate of a solid propellant. The situation isquite the opposite in the case of hybrids. The regression rate is negligibly dependenton the nature of fuels, even when they are as different as polystyrene and naturalrubber or polybutadiene. The reason for this lies in the counter-balancing effectcalled 'blowing effect'. This effect is simply that the burning rate does not linearlyscale with the ratio of the input heat flux to heat of phase transformation at the surface, but less (in fact, much less) strongly dependent on it. If we invoke the heatbalance at the surface of the burning fuel, we have

$$\rho_p r_h = \dot{q}'' / (\Delta h)_s,$$

where (Ah), is the heat of phase change at the surface (including that needed fordegradation) and q", the heat flux into the surface. Now, if by some mechanism rhincreases by decrease of (Ah)~, this increase in rn causes an increase in boundarylayer thickness, hence, reduction in gradients at the surface and in heat flux. The netresult is of course, an increase in regression rate but much less than is to be expectedfrom the linear relation. This effect is so significant that a 10 ~o increase in q" leavesrn virtually unaltered. Even a 35~o increase in q" causes only a 10~o increase in theregression rate (Marxman & Gilbert 1963).

Similar arguments can be used to explain why the initial temperature change causesmuch less change in the regression rate of a hybrid fuel than in the burning rate of asolid propellant.

- The purpose is to predict the regression rate.

Assumptions:

– Steady-state operation.

– Simple grain configuration (flat plate).

– No exothermic reactions in the solid grain (No oxidizer in solid phase).

– Oxidizer enters the port as a uniform gas.

– Le =Pr=1 (Le =k/D)

– No heat transfer to the ambient air through the walls of the rocket.
– All kinetic effects are neglected (Characteristic times for all chemical rxns << characteristic times for diffusion processes).
– Flame zone is infinitely thin. (Flame sheet). No oxidizer beneath the flame.
– Boundary layer is turbulent.

- Energy balance at the fuel surface: (Steady-state)

$$\dot{Q}_w = \dot{m}_f h_v = \rho_f \dot{r} h_v = (\rho v)_w h_v$$

$\dot{Q}_w$ = Total heat flux to the wall
h_v = Effective heat of gasification (Heating of the solid fuel grain + Heat of evaporation and melting + Heat of reaction for degradation of the polymer)

• End result:

$$\dot{r}(x) = AG^{0.8} x^{-0.2}$$

- Space time averaged regression rate (n ~0.5-0.8)

$$\bar{\dot{r}} = aG_o^n$$

4.14 Limitations of the Theory:

Each propellant combination has an upper and lower limit for the mass flux beyond which the model is not applicable.

- High mass fluxes Kinetic effects (Pressure dependency via the gas phase rxn rates)
- Low mass fluxes Radiation effects (Pressure dependency via the radiation effects)
- Transition to laminar boundary layer
- Cooking of the propellant (at very low regression rates)
- Dilution of the oxidizer

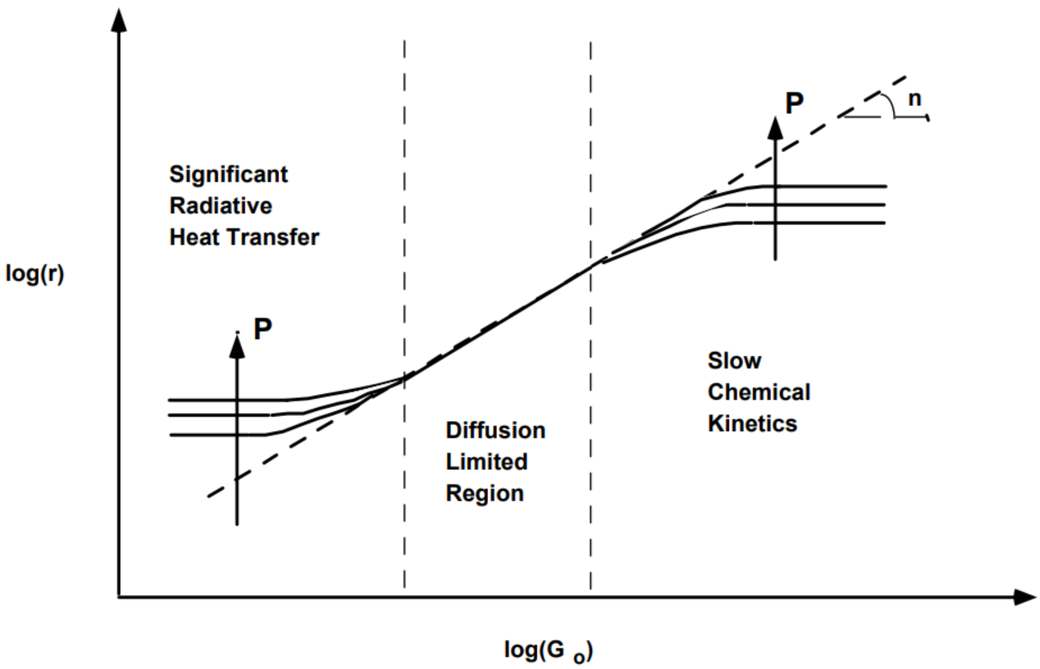


Figure 4.11 Effect of Pressure on the Regression Rate

- The heat conduction equation in the solid in reference of frame fixed to the regressing surface

$$\kappa \equiv \lambda_f / \rho_f C_{pf}$$

- During Steady state operating this expression can be integrated to yield

$$T(x) = (T_s - T_a) e^{-x/\delta_T} + T_a$$

- Here the characteristic thermal thickness can be given as

$$\delta_T = \kappa / \dot{r}$$

• Similarly the characteristics time is

$$\tau_T = \kappa / \dot{r}^2$$

• During typical operation of a polymeric hybrid fuel

$$\delta_T = 10^{-6} / 10^{-3} = 10^{-3}\ m \qquad \tau_T = 10^{-6} / 10^{-6} = 1 \sec$$

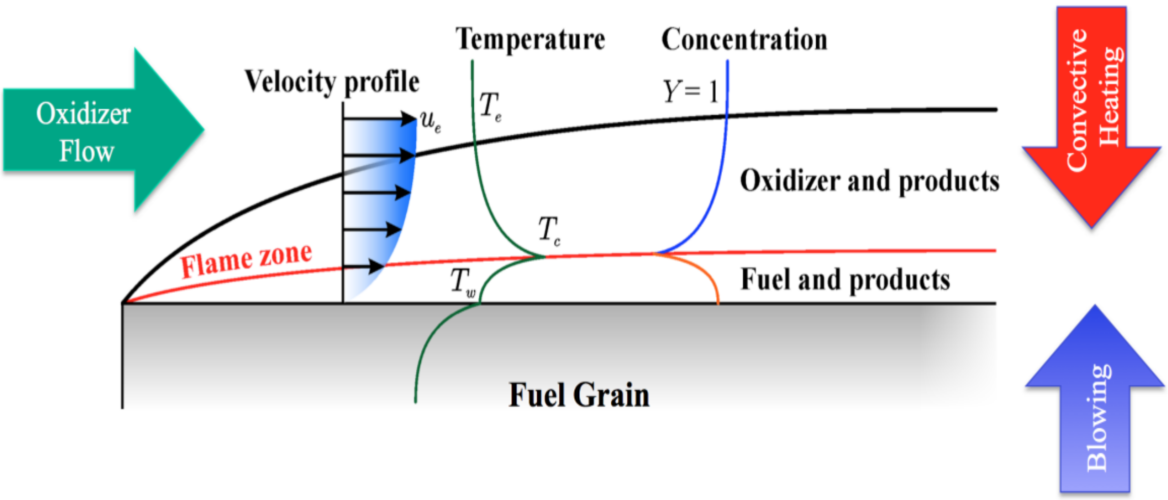


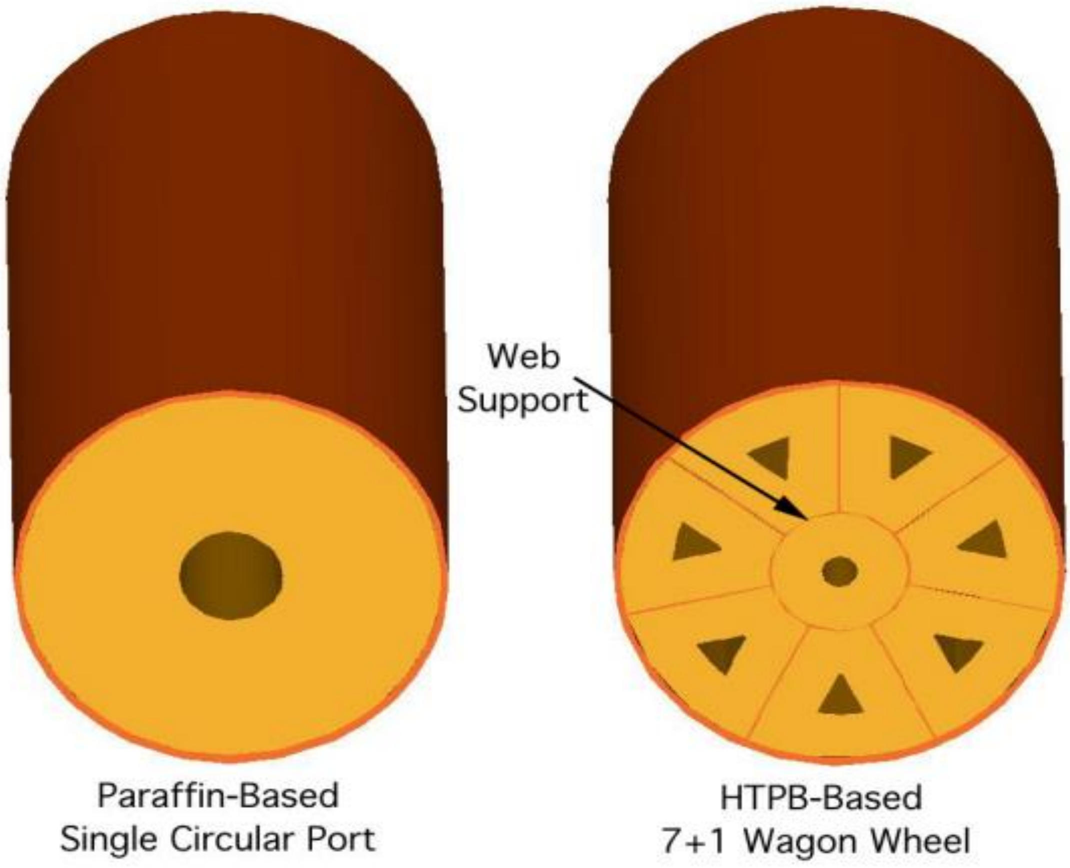
Fig. 4.12 Boundary layer combustion.

Disadvantage of Classical Hybrids:

Low Burning Rates --> Multi-port design

- Issues with multi-port design
- Excessive unburned mass fraction (i.e. typically in the 5% to 10% range).
- Complex design/fabrication, requirement for a web support structure.

• Compromised grain structural integrity, especially towards the end of the burn.
• Uneven burning of individual ports.
• Requirement for a substantial precombustion chamber or individual injectors for each port.



4.15: APPLICATIONS:

Hybrid propulsion is well suited to applications or missions requiring throttling,command shutdown and restart, long-duration missions requiring storablenontoxic propellants, or infrastructure operations (manufacturing andlaunch) that would benefit from a non-self-deflagrating propulsion system.
Such applications would include primary boost propulsion for space launchvehicles, upper stages, and satellite maneuvering systems.
Many early hybrid rocket motor developments were aimed at target missilesand low-cost tactical missile applications (Ref. 15-1). Other developmentefforts focused on high-energy upper-stage motors. In recent years developmentefforts have concentrated on booster prototypes for space launch applications.

4.16: ADVANTAGES OF HYBRID PROPELLANTS:

Compared to	Solids	Liquids
Simplicity	○ Chemically Simpler ○ Tolerant to processing errors	○ Mechanically Simpler ○ Tolerant to fabrication errors
Safety	○ Reduced Chemical Explosion hazard ○ Thrust termination and Abort possibility	○ Reduced fire hazard ○ Less prone to hard starts
Performance Related	○ Better Isp Performance ○ Throttling/Restart capability	○ Higher fuel density ○ Easy inclusion of solid performance additives (Al. Be)

Other	○ Reduced Environmental impact	○ Reduced number and mass of liquids
Cost	○ Reduced Development costs are expected ○ Reduced recurring costs are expected	

UNIT-5
ADVANCED PROPULSION TECHNIQUES

5.1 ELECTRIC ROCKET PROPULSION:

In all electric propulsion the source of the electric power (nuclear, solar radiationreceivers, or batteries) is physically separate from the mechanism thatproduces the thrust. This type of propulsion has been handicapped by heavyand inefficient power sources. The thrust usually is low, typically 0.005 to 1 N.In order to allow a significant increase in the vehicle velocity, it is necessary toapply the low thrust and thus a small acceleration for a long time (weeks ormonths).
Of the three basic types, electrothermal rocket propulsion most resembles the chemical rocket units; propellant is heated electrically(by heated resistors or electric arcs) and the hot gas is then thermodynamicallyexpanded and accelerated to supersonic velocity through an exhaust nozzle. These electrothermal units typically have thrust ranges of0.01 to 0.5 N, with exhaust velocities of 1000 to 5000 m/sec, and ammonium,hydrogen, nitrogen, or hydrazine decomposition product gases have been usedas propellants.

The two other types--the electrostatic or ion propulsion engine and the
Electromagnetic or magneto plasma engine--accomplish propulsion by differentprinciples and the thermodynamic expansion of gas in a nozzle, as such,does not apply. Both will work only in a vacuum. In an ion rocket a working fluid (typically, xenon) is ionized (by stripping off electrons)and then the electrically charged heavy ions are accelerated to very high velocities(2000 to 60,000 rn/sec) by means of electrostatic fields. The ions aresubsequently electrically neutralized; they are combined with electrons to preventthe buildup of a space charge on the vehicle.

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

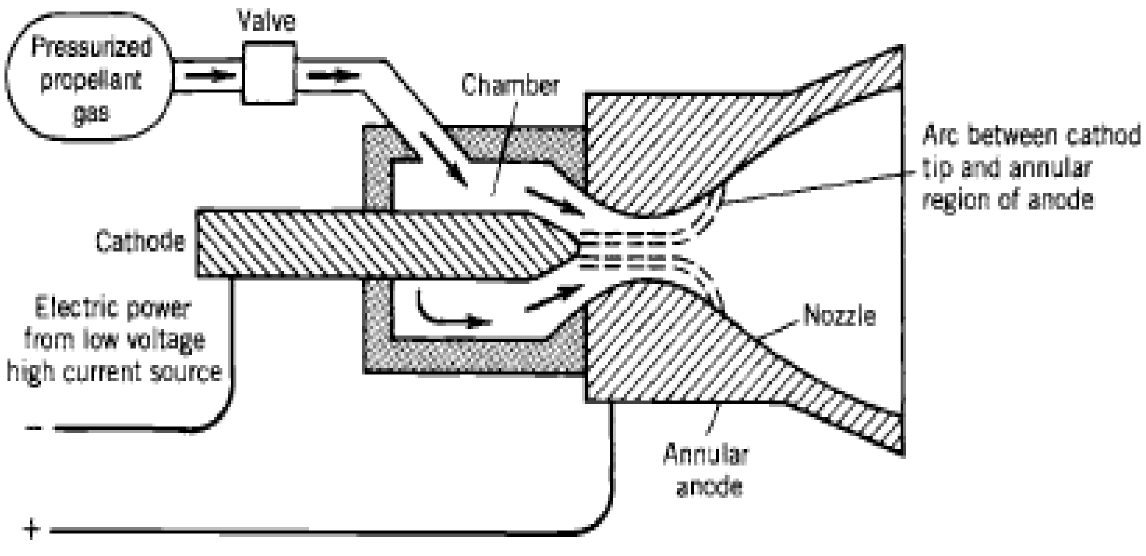


Figure: 5.1Simplified Diagram of Arc Heating Electric Propulsion

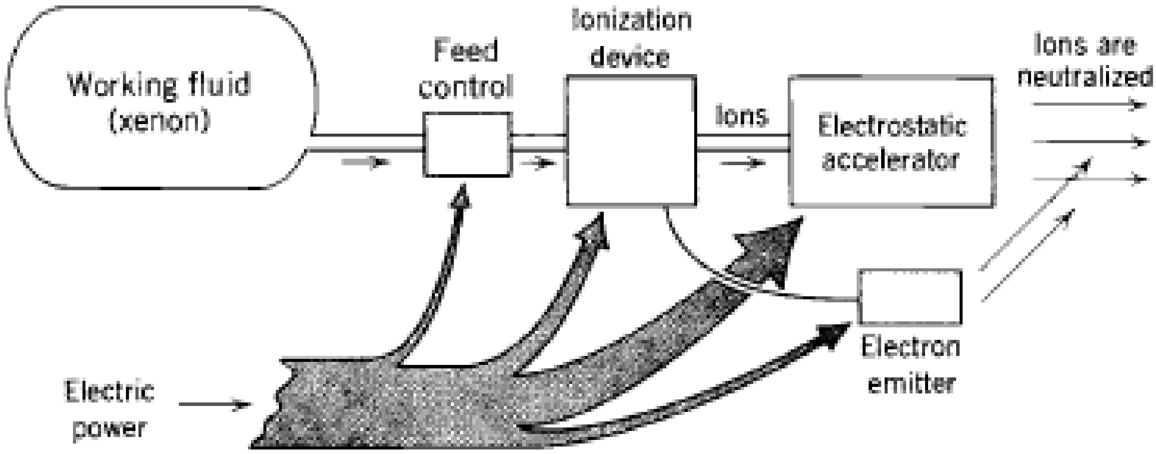


Figure. 5.2 Simplified Schematic Diagram Of A Typical Ion Rocket

5.1.1 **REVIVED INTEREST IN ELECTRIC PROPULSION**:

- The concept of electric propulsion has been known for a considerable time, and differenttypesofelectricthrusterhavebeendevelopedandtestedinspace.However,theyhaveremainedacuriosityuntilcomparativelyrecenttimes,whenitwasrealised that the requirement for high velocity increment did not apply only to ambitious space exploration missions, but to station keeping for communications satellites.

- Over the satellite's lifetime, drift from the correct orbit, induced by solar radiation pressure and gravity gradients, has to be constantly corrected. This requires a significant amount of propellant, the mass of which could be used for more communications equipment, leading to higher profitability. Increased exhaust velocity from the thrusters translates directly into decreased propellantmass. propulsion. The advantages, even for unmanned planetary missions, are significant.

- The alternative is to carry the extra propellant required for a chemical thruster into Earth orbit with the probe, which has a dramatic effect on the mass ratio at launch, and results in a serious reduction in payload. Use of electric propulsion enables unmanned interplanetary missions, requiring large velocity increments, which would otherwise be difficult using present-daylaunchers.

5.2 **PRINCIPLES OF ELECTRIC PROPULSION:** The basic principle of electric propulsion is to apply electrical energy to the propellant from an external power source. This can be done in several ways. The simplest is to heat the propellant with a hot wire coil, through which an electric current passes. This elementary approach, used in some commercial thrusters, is very successful.

More energy can be delivered from the electric current if an arc is struck through the propellant, which generates higher temperatures than the resistive approach and produces a higher exhaust velocity. Finally, electric or magnetic fields can be used directly to accelerate propellant ions to very high velocities, producing the highest exhaust velocity of all. These ion thrusters, and Kall effect thrusters are seen as the most promising for deep space applications, and they are already coming into commercial use for station keeping and interplanetary propulsion.

While for a chemical rocket the link between energy supply and propellant simplifies analysis, for electrical propulsion the power supply introduces free parameters for which we have to make estimates when deriving expectedvehicle performance.

Electric power can come from a battery, solar panels or anonboard nuclearorsolargenerator,eachofwhichhasitsownadvantagesanddisadvantages.

Whatisimportant,fromthevehicleperformancepointofview,isthepower-to-mass ratio—W/kg.

In most cases the power does not diminish with progress throughthe flight, while the mass of propellant decreases in the familiar way as the vehicle accelerates. This is in direct contrast to the chemical rocket, in which both the propellant and the available energy decrease together. Using these ideas, simple estimates of vehicle performance can beproduced.

5.3 IONPROPULSION:This is the simplest concept,, propellant is ionised, and then enters a region of strong electric field, where the positive ions are accelerated. Passing through a grid, they leave the engine as a high-velocity exhaust stream. The electrons do not leave, and so the exhaust is positively charged. Ultimately this would result in a retarding field developing between the spacecraft and the exhaust, and so an electron current is therefore discharged into the exhaust to neutralize the spacecraft. The electrons carry little momentum, and so this does not affect the thrust.

Theschematic(Figure5.3)showsthethrusterisdividedintotwochambers.The propellantenterstheionisationchamberintheformofneutralgasmolecules.There is a radial electric field across the chamber, and electrons are released from the cathode(whichcanbeathermionicemitter).Theelectronsareacceleratedbythe radialfield,andreachenergiesofseveraltensofelectronvolts,whichisenoughto ionise the neutral propellant atoms by collision. To extend the path length of the electrons and ensure that they encounter as many neutral atoms as possible, an axial magnetic field is provided, which makes them move in a spiral path. The ionisation therefore becomes efficient; that is, the number of ions produced, as a functionoftheelectroncurrent,ismaximised.Intheory,alltheelectricalenergyin an electrothermal thruster enters the exhaust stream, but in an electromagnetic thrustereachionintheexhausthastobecreatedwithanenergyofabout20−30eV perion.Thisenergydoesnotgointopropulsion,andislost.Thus,itisimportantto maximise the ionisationefficiency.

The ionised propellant atoms drift under a small negative field through the first gridintotheacceleratingchamber.Thegridshaveahighpotentialacrossthem,and are separated by 1−2 mm. The ions gain energy in the strong electric field and, passing through the outer grid, form theion beam.

Thereisnoneedforanozzletogeneratethethrust,becausethemotionoftheion beamisorderedandnotchaotic.

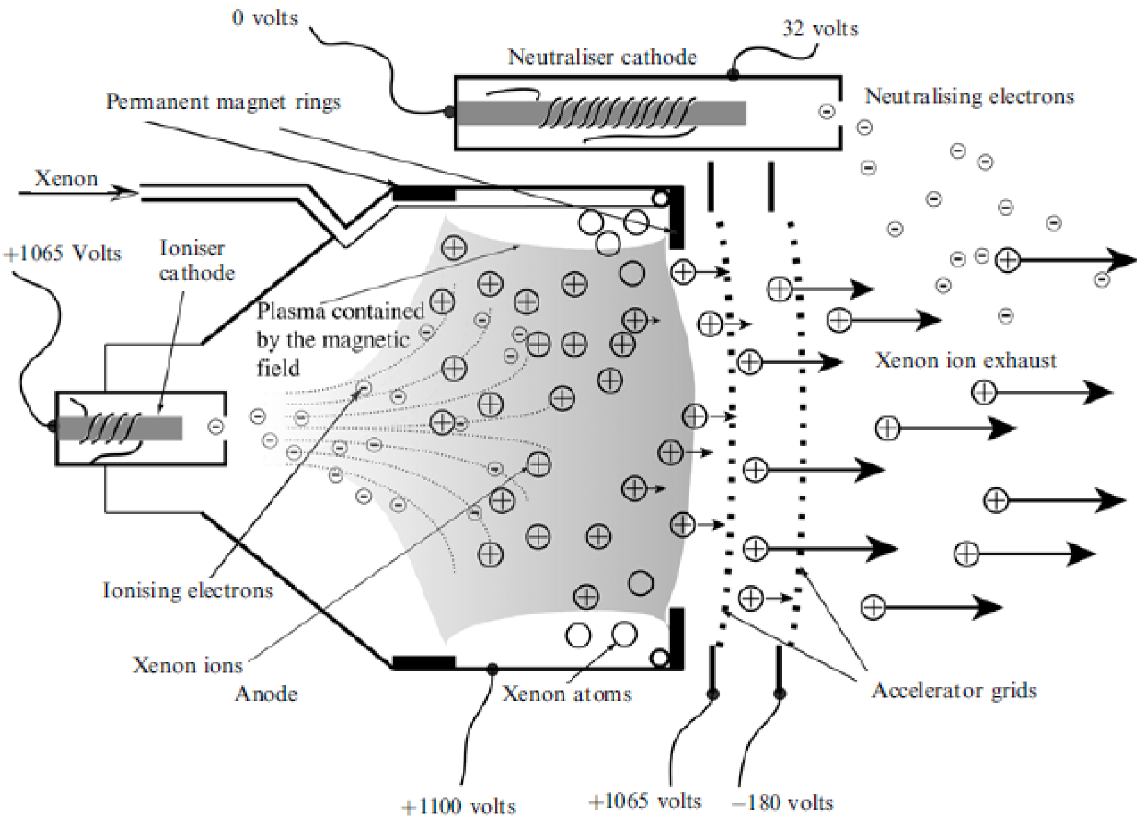


Figure 5.3 SchematicDiagramOfTheNSTARIonThruster

5.3.1 ION THRUSTER THEORY:

Theionthrusterissimpleinconcept,asdescribedabove.Thetheoryofoperationis alsorelativelysimple,andbecauseitissodifferentfromthatofathermalrocketitis usefultoincludeabriefdescriptionhere,sothatthestrengthsandlimitationscanbe appreciated.

Asinallreactionpropulsionsystems,thethrustdependsultimatelyonthetransfer of momentum from an exhaust stream to the vehicle. The exhaust velocity is straightforwardlygivenbythepotentialdifferencebetweenthegrids.Ionsdropping throughthispotentialdifferenceeachgainafixedamountofenergy,andthisconverts directlyintoavelocity.Theotherparameterinthethrustisthemassflowrate.Foran ionthrusterthisisdirectlyrelatedtothecurrentflowingbetweenthegrids,andtheion current itself becomes the exhaust stream. To increase the thrust of a given ion thruster,thecurrenthastobeincreased;butitcannotbeincreasedindefinitely,as there is a natural limit. It is this limit which we can examinetheoretically.

5.5 APPLICATIONS OF ELECTRICPROPULSION: The advantages of electric thrusters are mainly concerned with their ability to provide a high exhaust velocity, and hence to use propellant very economically. One way to look at this is to consider the following. Rearranging the rocket equation againwe find

$$V = v_e \log_e$$

$$\frac{M_0}{M} = e^{\frac{V}{v_e}}$$

$$M_0 = M + M_f$$

$$\frac{M + M_f}{M} = e^{\frac{V}{v_e}}$$

$$\frac{M_f}{M} = e^{\frac{V}{v_e}} - 1$$

Inthisinversion,theratioofpropellantmasstovehiclemassisgivenintermsofthe exhaustandvehiclevelocities.Thisisusefulincalculatingthequantityofpropellant needed for any manoeuvre for a given payload. The ratio $M_f/M$is the propellant efficiencyorfuelmultiplier,anddependsonlyontheratioofthevehiclevelocityto the exhaust velocity (Figure 5.4). we see that the propellant efficiency depends exponentially on the exhaust velocity, and this is why the high exhaust velocity provided by electric propulsion is sobeneficial.

whencomparingtheperformanceofelectricpropulsiondevicesitissensibleto includeavarietyofdiffierentmissions.Hereweshallconsiderthreecases,,station keeping for a mission lifetime of 10 years, transfer from LEO to GEO, and a nine-monthjourneytoMars.Itissometimesusefultoexpresstheperformanceintermsof the propellant to total vehicle mass, whichis

$$\frac{R-1}{R} = \frac{e^{\frac{V}{v_e}} - 1}{e^{\frac{V}{v_e}}} = 1 - e^{-\frac{V}{v_e}}$$

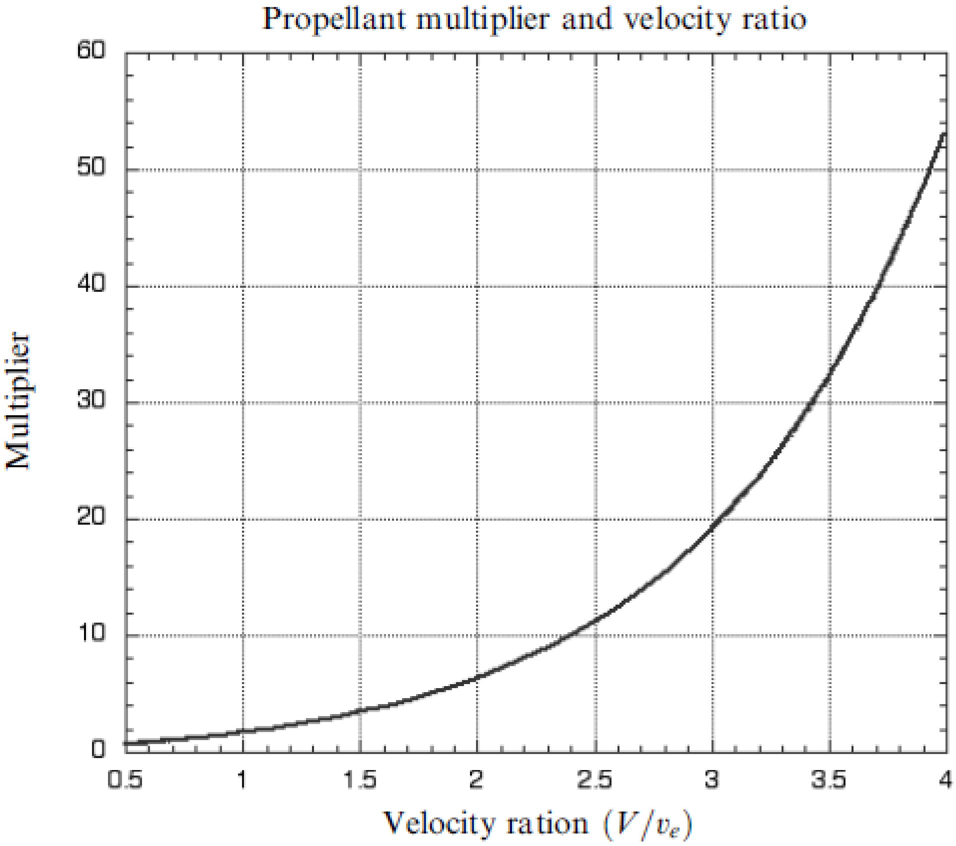


Figure:5.4 Propellant Efficiency

5.6 NUCLEAR ROCKET ENGINES:Three different types of nuclear energy sources have been investigated fordelivering heat to a working fluid, usually liquid hydrogen, which subsequentlycan be expanded in a nozzle and thus accelerated to high ejectionvelocities (6000 to 10,000 m/sec). However, none can be considered fullydeveloped today and none have flown. They are the fission reactor, the radioactive isotope decay source, and the fusion reactor. All three types arebasically extensions of liquid propellant rocket engines. The heating of the gas is accomplished by energy derived from transformations within thenuclei of atoms.

In chemical rockets the energy is obtained from withinthe propellants, but in nuclear rockets the power source is usually separatefrom the propellant.

In the nuclear fission reactor rocket, heat can be generated by the fissionof uranium in the solid reactor material and subsequently transferred to theworking fluid.). The nuclear fission rocket is primarilya high-thrust engine (above 40,000 N) with specific impulse values up to 900sec. Fission rockets were designed and tested in the 1960s. Ground testswith hydrogen as a working fluid culminated in a thrust of 980,000 N(210,000 lb force) at a graphite core nuclear reactor level of 4100 MWwith an equivalent altitude-specific impulse of 848 sec and a hydrogen temperatureof about 2500 K. There were concerns with the endurance of thematerials at the high temperature (above 2600 K) and intense radiations,power level control, cooling a reactor after operation, moderating the highenergyneutrons, and designing lightweight radiation shields for a mannedspace vehicle.

In recent years there has been renewed interest in nuclear fission rocketpropulsion primarily for a potential manned planetary exploration mission.Studies have shown that the high specific impulse (estimated in some studiesat 1100 sec) allows shorter interplanetary trip transfer times, smaller vehicles,and more flexibility in the launch time when planets are not in their optimumrelative position.In the isotope decay engine a radioactive material gives off radiation, whichis readily converted into heat. Isotope decay sources have been used successfullyfor generating electrical power in space vehicles and some have beenflown as a power supply for satellites and deep space probes. The releasedenergy can be used to raise the temperature of a propulsive working fluidsuch as hydrogen or perhaps drive an electric propulsion system. It providesusually a lower thrust and lower temperature than the other types of nuclearrocket. As yet, isotope decay rocket engines have not been developed orflown.

Fusion is the third nuclear method of creating nuclear energy that can heata working fluid. A number of different concepts have been studied. To datenone have been tested and many concepts are not yet feasible or practical.Concerns about an accident with the inadvertent spreading of radioactivematerials in the earth environment and the high cost of development programshave to date prevented a renewed experimental development of a largenuclear rocket engine.

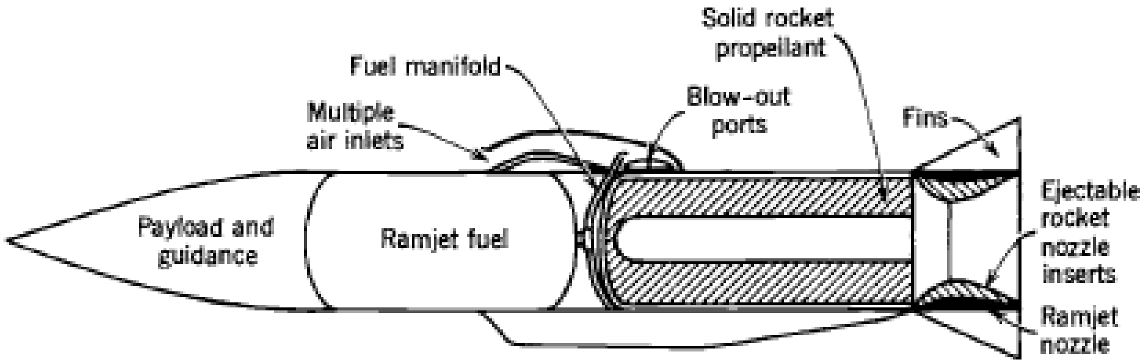


Figure 5.5 Elements Of An Air-Launched Missile With Integral Rocket-Ramjet Propulsion.

After the solid propellant has been consumed in boosting the vehicle to flightspeed, the rocket combustion chamber becomes the ramjet combustion chamber withair burning the ramjet liquid fuel.

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

Liquid propellant rocket engines use liquid propellants that are fed underpressure from tanks into a thrust chamber.* A typical pressure-fed liquid propellantrocket engine system is schematically shown in Fig. 1-3. The liquid bipropellant consists of a liquid oxidizer (e.g., liquid oxygen) and a liquid fuel(e.g., kerosene). A monopropellant is a single liquid that contains both oxidizingand fuel species; it decomposes into hot gas when properly catalyzed. A largeturbopump-fed liquid propellant rocket engine is shown in Fig. below. Gas pressure feed systems are used mostly on low thrust, low total energy propulsion systems, such as those used for attitude control of flying vehicles, often withmore than one thrust chamber per engine. Pump-fed liquid rocket systems areused typically in applications with larger amounts of propellants and higher thrusts, such as in space launch vehicles.In the thrust chamber the propellants react to form hot gases, which in turn are accelerated and ejected at a high velocity through a supersonic nozzle, thereby imparting momentum to the vehicle. A nozzle has a converging section,a constriction or throat, and a conical or bell-shaped diverging section asfurther described in the next two chapters.

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

In solid propellant rocket motors* the propellant to be burned is contained within the combustion chamber or case. The solid propellant charge is calledthe grain and it contains all the chemical elements for complete burning. Onceignited, it usually burns smoothly at a predetermined rate on all the exposedinternal surfaces of the grain. Initial burning takes place at the internal surfacesof the cylinder perforation and the four slots. The internal cavity grows as propellant is burned and consumed. The resulting hot gas flows through the

Supersonic nozzle to impart thrust. Once ignited, the motor combustion proceedsin an orderly manner until essentially all the propellant has been consumed.There are no feed systems or valves

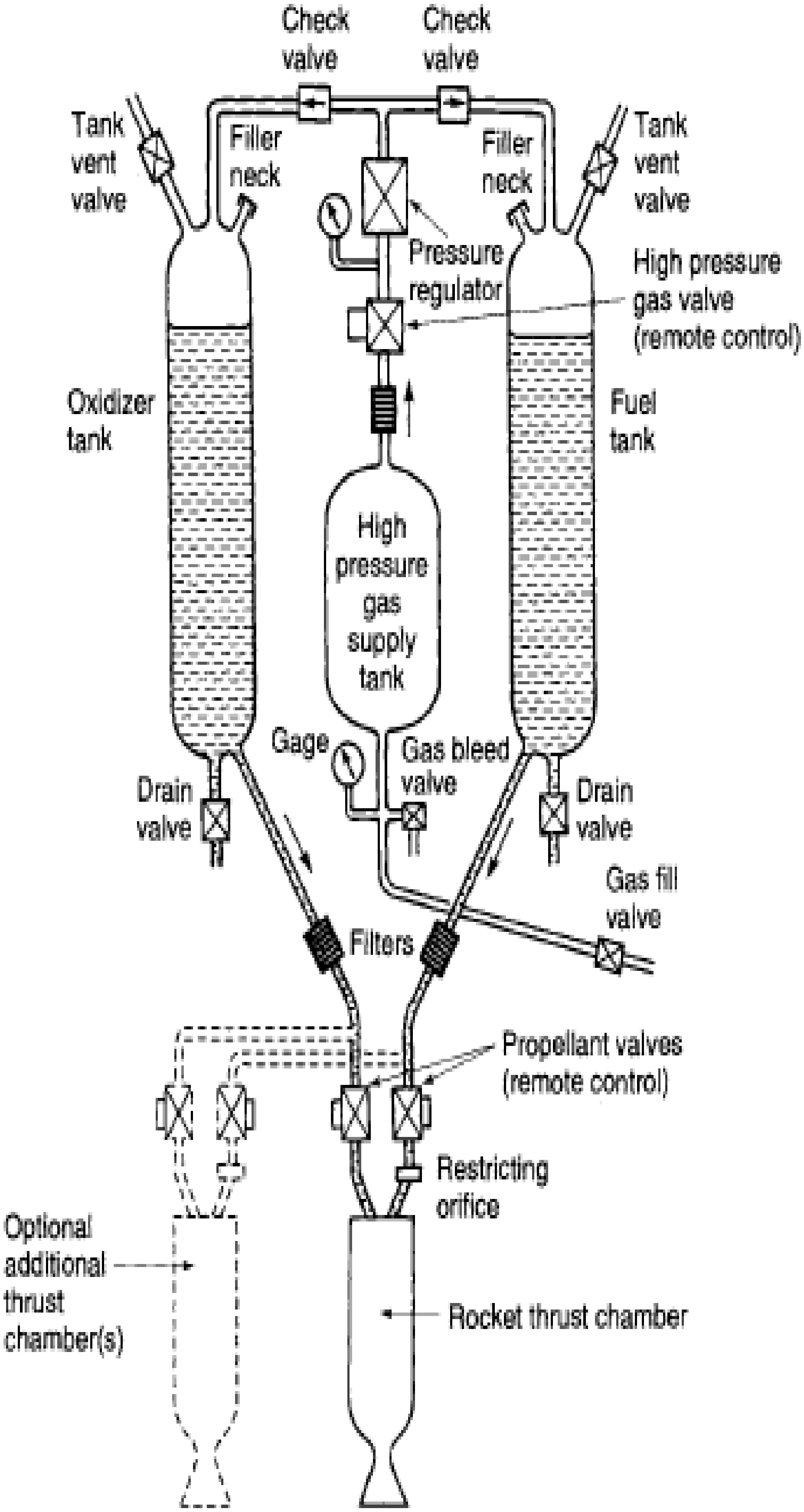


Figure5.6 Schematic flow diagram of a liquid propellant rocket engine with a gaspressure feed system.

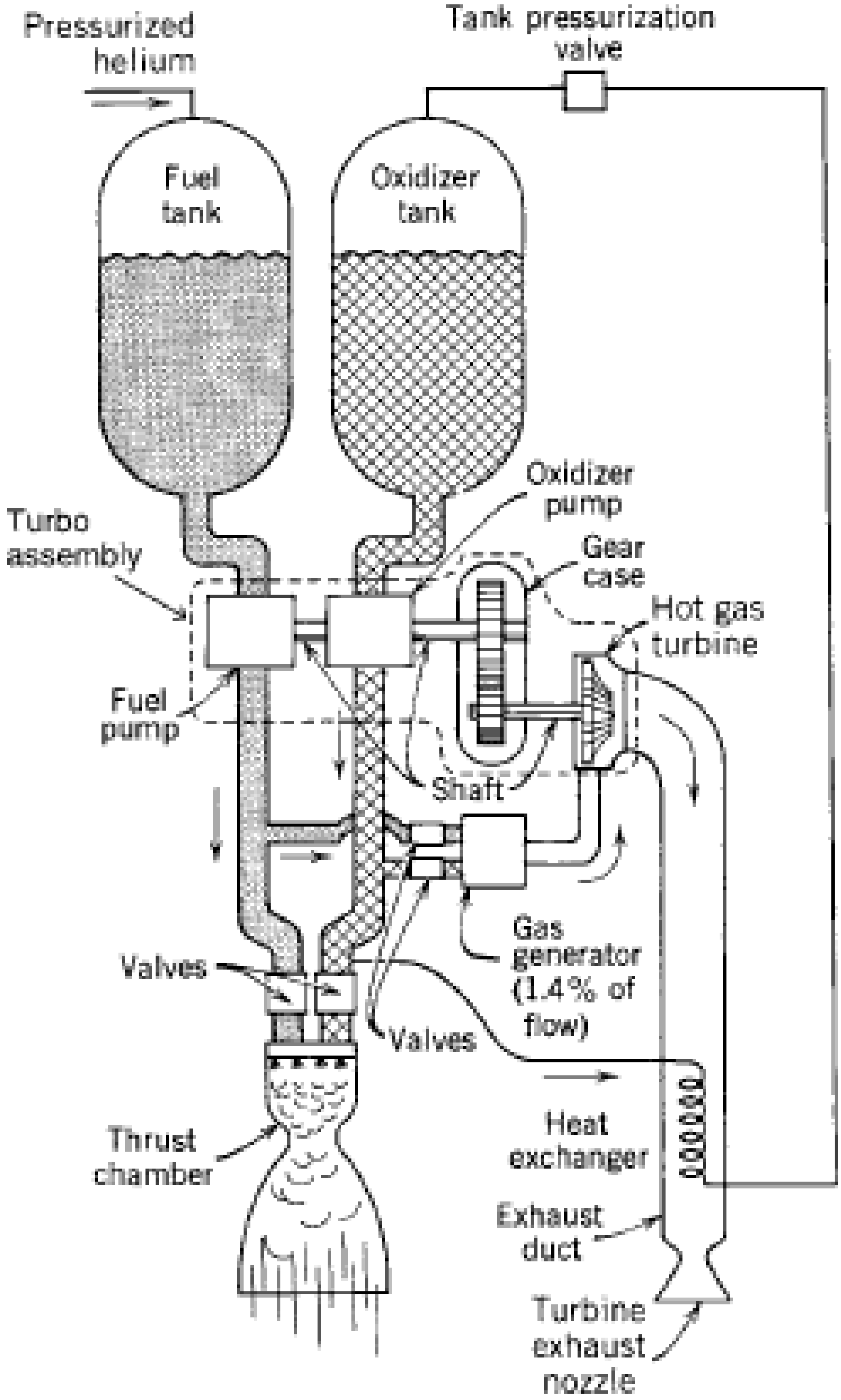


Figure5.7 Simplified schematic diagram of one type of liquid propellant rocketengine with a turbopump feed system and a separate gas generator, which generates warm gas for driving the turbine.

Hybrid propellant rocket propulsion systems use both a liquid and a solid propellant. For example, if a liquid oxidizing agent is injected into a combustion chamber filled with solid carbonaceous fuel grain, the chemical reaction produces hot combustion gases.

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

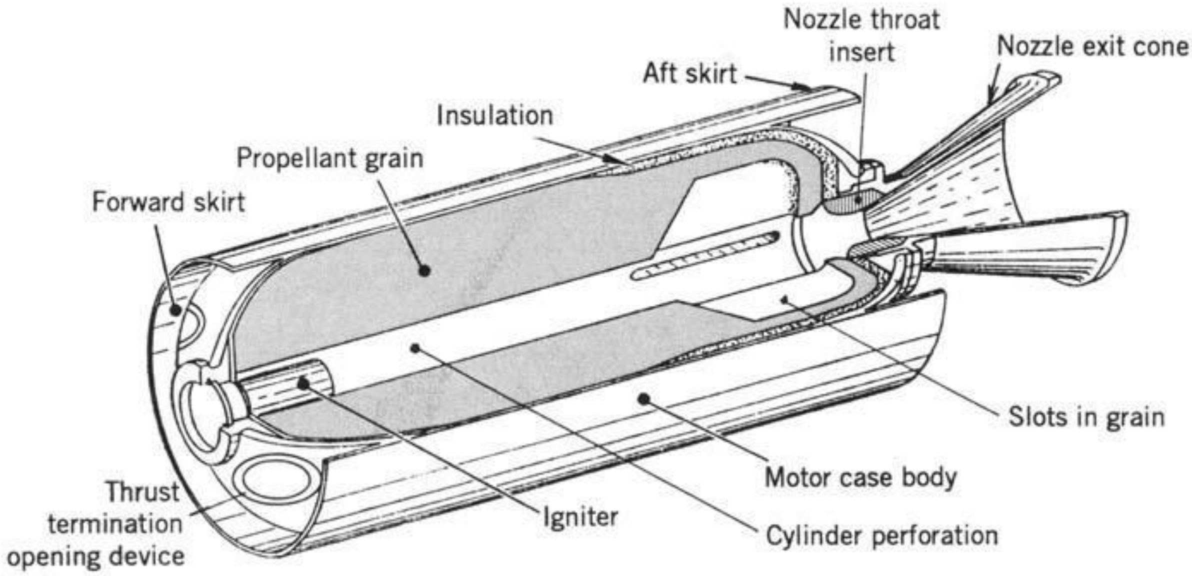


Figure5.8 Simplified perspective three-quarter section of a typical solid propellant rocket motor with the propellant grain bonded

5.8 APPLICATIONS OF ROCKET PROPULSION: Because the rocket can reach a performance unequaled by other prime movers,it has its own fields of application and does not usually compete with other propulsion devices.

Examples

Space Launch Vehicles:

Between the first space launch in 1957 and the end of 1998 approximately 4102space launch attempts have taken place in the world and all but about 129 were successful .Space launch vehicles or space boosters can be classified broadly as expendable or recoverable/reusable. Other bases of classification are the type of propellant (storable or cryogenic liquid or solid propellants), number of stages (single-stage, two-stage, etc.), size/mass of payloads or vehicles, and manned or unmanned.

Space launch vehicle, one member of the Titan family of storable propellant space launch vehicles, which is used extensively for boosting satellites into synchronous earth orbit or into escape trajectories for planetary travel. This heavy-duty launch vehicle consists of the basic 2-stage Titan III standard launch vehicle (liquid propellant rockets) supplemented by two solid propellant “strap-on motors." A fourth stage, known as the translate, permits a wide variety of manoeuvres, orbit changes, and trajectory transfers to be accomplished with the payload, which can be one or more satellites or spacecraft.

Each space launch vehicle has a specific space flight objective, such as an earth orbit or a moon landing. It uses between two and five stages, each with its own propulsion system, and each is usually fired sequentially after the lower stage is expended. The number of stages depends on the specific space trajectory, the number and types of manoeuvres, the energy content of a unit mass ofthe propellant, and other factors. The initial stage, usually called the booster stage, is the largest and it is operated first; this stage is then separated from the ascending vehicle before the second-stage rocket propulsion system is ignited and operated. Adding an extra stage permits significant increase in the payload (such as more scientific instruments or more communications gear).

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

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

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