KARPAGAM ACADEMY OF HIGHER EDUCATION

(Deemed to be University Established Under Section 3 of UGC Act, 1956)

Eachanari post, Coimbatore-641021. INDIA.

FACULTY OF ENGINEERING

DEPARTMENT OF MECHANICAL ENGINEERING

UG / B.Tech -AEROSPACE ENGINEERING

17BTAR503 ROCKET PROPULSION 3 0 0 3 100

OBJECTIVES:

To study in detail about gas turbines, ramjet, fundamentals of rocket propulsion and chemical rockets

UNIT - I FUNDAMENTALS OF ROCKET PROPULSION

Operating principle – Rocket equation, Definitions. Performance parameters, Staging and Clustering – Classification of rockets -Specific impulse of a rocket - internal ballistics-Rocket nozzle and performance- Rocket nozzle classification–Numerical Problems.

UNIT - II CHEMICAL ROCKETS

Molecular mass, specific heat ratio, Energy release during combustion, Stoichiometry & mixture ratio, Criterion for choice of propellant, Solid propellants, requirement, composition and processing. Liquid propellants, energy content, storability-Numerical problems

UNIT - III SOLID PROPULSION SYSTEMS

Solid propellant rockets – Classifications- Selection criteria of solid propellants – Important hardware components of solid rockets – Propellant grain design considerations- Burn rate, burn rate index for stable operation, mechanism of burning, ignition and igniter types

UNIT - IV LIQUID PROPULSION SYSTEMS

Liquid propellant rockets – Classifications- Selection of liquid propellants – Thrust control in liquid rockets – Cooling in liquid rockets – Limitations of hybrid rockets – Relative advantages of liquid rockets over solid rockets- Numerical Problems.

UNIT - V ADVANCED PROPULSION TECHNIQUES

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:

S.NO.	AUTHOR(S)	TITLE OF THE BOOK	PUBLISHER	YEAR OFPUBLICATION
1.	Sutton G.P	Rocket PropulsionElements	John Wiley &Sons Inc.,New York.	2011
2.	Ramamurthi.K	Rocket Propulsion	Macmillan Publishing Co, India. First edition	2010

S.No.	AUTHOR(S)	TITLE OF THE BOOK	PUBLISHER	YEAR OF PUBLICATION
1.	Cohen, H., Rogers, G.F.C. and Saravanamuttoo	Gas Turbine Theory	Longman Co.,	2003
2.	Gorden, C.V.	Aero Thermodynamics of Gas Turbine and Rocket Propulsion	AIAA Education Series, New York.	2004
3.	Mathur, M., and Sharma R.P.	GasTurbines and JetandRocket Propulsion	StandardPublishers, New Delhi.	2010
4.	George P. Sutton, Oscar Biblarz	Rocket Propulsion Elements	John Wiley & Sons, New York.	2017

WEB REFERENCE:

www.aircraftenginedesign.com
www.aticourses.com/rockets.htm
www.grc.nasa.gov
exploration.grc.nasa.gov/education/rocket/rktth1.html
www.scribd.com

KARPAGAM UNIVERSITY
COIMBATORE – 21
FACULTY OF ENGINEERING
DEPARTMENT OF MECHANICAL ENGINEERING

COURSE PLAN

Subject Name : **Rocket Propulsion**
Subject Code : **17BTAR303** **(Credits - 04)**
Name of the Faculty : **Mrs.P.JAYAPRADHA**
Designation : **ASSISTANT PROFESSOR**
Year/Semester/Section : **II/III**
Branch : **B.TECH- AEROSPACE ENGINEERING**

Sl. No.	No. of Periods	Topics to be Covered	Support Materials
UNIT - I FUNDAMENTALS OF ROCKET PROPULSION			
1.	1	Basics of Rocket Propulsion	T [1]
2.	1	Operating principle	T [1]
3.	1	Rocket equation	T [1]
4.	1	Definitions. Performance parameters	T [1]
5.	1	Staging and Clustering	T [2]
6.	1	Classification of rockets	T [2]
7.	1	Specific impulse of a rocket	T [2]
8.	1	Internal ballistics	T [2]
9.	1	Rocket nozzle and performance	T [3]
10.	1	Rocket nozzle classification	T [3]
11.	1	Numerical Problems	T [1]
12.	1	**Tutorial: Numerical Problems**	T [3]
Total No. of Hours Planned for Unit - I			**12**

Sl. No.	No. of Periods	Topics to be Covered	Support Materials
UNIT - II CHEMICAL ROCKETS			
13.	1	Introduction to Chemical Rockets	T [3]
14.	1	Molecular mass, specific heat ratio	T [3]
15.	1	Energy release during combustion	T [3]
16.	1	Stoichiometry & mixture ratio,	T [3]
17.	1	Stoichiometry & mixture ratio,	T [3]
18.	1	Criterion for choice of propellant	T [3]
19.	1	Solid propellants, requirement,	T [3]
20.	1	Composition and processing	T [3]

21.	1	Liquid propellants	T [3]
22.	1	Energy content, storability-Numerical problems	T [3]
23.	1	**Tutorial: Numerical problems**	T [3]
Total No. of Hours Planned for Unit - II			**11**

Sl. No.	No. of Periods	Topics to be Covered	Support Materials
UNIT - III SOLID PROPULSION SYSTEMS			
24.	1	Solid propellant rockets	T [3]
25.	1	Classifications-	T [3]
26.	2	Selection criteria of solid propellants	T [3]
27.	1	Important hardware components of solid rockets	T [3]
28.	1	Propellant grain design considerations	T [3]
29.	2	Burn rate, burn rate index for stable operation,	T [3]
30.	1	Mechanism of burning,	T [2]
31.	1	Ignition and igniter types	T [2]
Total No. of Hours Planned for Unit - III			**10**

Sl. No.	No. of Periods	Topics to be Covered	Support Materials
UNIT - IV LIQUID PROPULSION SYSTEMS			
32.	1	Liquid propellant rockets	T [1]
33.	1	Classifications	T [1]
34.	2	Selection of liquid propellants	T [1]
35.	1	Thrust control in liquid rockets	T [1]
36.	2	Cooling in liquid rockets	T [1]
37.	1	Limitations of hybrid rockets	T [1]
38.	1	Relative advantages of liquid rockets over solid rockets-	T [1]
39.	1	Numerical Problems	T [1]
Total No. of Hours Planned for Unit - IV			**10**

Sl. No.	No. of Periods	Topics to be Covered	Support Materials
UNIT - V ADVANCED PROPULSION TECHNIQUES			
40.	1	Electric rocket propulsion	R [4]
41.	1	types of electric propulsion techniques	R [4]
42.	1	Ion propulsion	R [4]
43.	1	Nuclear rocket	R [4]

44.	1	Nuclear rocket	R [4]
45.	2	Comparison of performance of these propulsion systems with chemical rocket propulsion systems	R [4]
46.	2	Future applications of electric propulsion systems - Solar sail.	R [4]
47.	1	**Discussion on Competitive Examination related Questions / University previous year questions**	
Total No. of Hours Planned for Unit - V			**10**

TOTAL PERIODS :53

TEXT BOOKS

T [1] – Yunus A. Cengel and Michael A. Boles, “Thermodynamics an engineering approach” McGraw Hill Higher education,2011.
T [2] – Nag, P. K “Engineering Thermodynamics” McGraw Hill, New Delhi, 2013.
T[3]- R.K. Rajput “A Text book of Engineering Thermodynamics” Fourth Edition, 2013.

REFERENCES

R [1] - Rayner Joel, “Basic Engineering Thermodynamics” Addison Wesley, New York,1996.
R [2] - Holman, J, “Thermodynamics, 4th Edition” Tata McGraw Hill, PNew Delhi, 1998
R [3] - Michael Moran, J., and Howard Shapiro, N, Fundamentals of Engineering Thermodynamics,4th Edition, John Wiley & Sons, New York, 2000.
R [4] - Fundamentals of Engineering Thermodynamics, E, Prentice –Hall, India, 2005.

WEBSITES

W [1] - http://www.brighthubengineering.com/hvac/64884-different-types-of-air-compressors
W [2] - www.airconditioning-systems.com/air-conditioner-compressor.html
W [3] - www.daveycompressor.com/differenttype.html

I. CONTINUOUS INTERNAL ASSESSMENT : 40 Marks

(Internal Assessment Tests: 30, Attendance: 5, Assignment/Seminar: 5)

II. END SEMESTER EXAMINATION : 60 Marks

TOTAL : 100 Marks

UNIT-1

FUNDAMENTALS OF ROCKET PROPULSION

Operating principle – Rocket equation, Definitions. Performance parameters, Staging and Clustering – Classification of rockets -Specific impulse of a rocket - internal ballistics-Rocket nozzle and performance- Rocket nozzle classification–Numerical Problems.

Rockets – Special Features and Applications

Historical Reference

- The basic principles of all propulsive devices lie with the laws of motion due to Newton (17^{th} Century AD). These laws are phenomenological and therefore one can expect that even before Newton there may have existed many devices working on the principles of reaction.
- Rockets working directly on the principle of reaction are perhaps the simplest of the propulsive engines.
- The reciprocating engines and gas turbine engines are relatively more complex.
- The Chinese are credited with the invention of rockets probably in 12-14^{th} century AD.
- Indians used the rockets as effective weapons in late 18^{th} century against British and in 19^{th} century, the rockets became a part of the warfare in Europe. But it was only in the early part of the present century that man has recognized the full potential of rocket owing to the interests in space travel/satellite technology and like.
- Tsiolkovsky (USSR 1903) Goddard (USA, 1912) and Oberth (1921) are the pioneers of modern rocketry.
- The liquid propellant rocket owe their genesis to these people.
- The German V-2 rockets (25 tons, 65 sec, LOX-Alchol) and the post-second World war progress in rocketry are too familiar to all.

Principle

All the conventional propulsion systems work by causing a change of momentum in a working fluid in a direction opposite to the intended motion. Rockets fall under the category of direct acting engines – since the energy liberated by the chemical process is directly used to obtain thrust. Being non-air breathing devices the basic component of a rocket are

(i) Combustion chamber where exothermic processes produces gases at high temperature and pressure, and

(ii) Nozzle, which accelerate the fluid to high velocities and discharge them into surrounding atmosphere thereby deriving the desired force or thrust.

Some Special Features

The non-air breathing nature of rockets makes them very distinct among the propulsive devices.

(a) The reaction system does not depend on the surrounding atmosphere. There are no velocity limitations and altitude ceiling.

(b) Since it has to carry its own oxidizer required for combustion reaction, the specific propellant consumption is very high. Rockets consume approximately 15kg/kg-hr of propellant compared to about 1 kg/kg-hr of fuel by turbojet engine.

(c) High pressure operation is possible and hence the ratio of energy liberation per unit volume (and also unit weight of hardware) is very high.

(d) Main part of the rockets contains no moving element. Hence there is no constraint on internal aerodynamics and the reliability is high. This also implies quick response times, which makes them ideal control components.

With the above features, it is clear that the rockets are the most suitable power plants for

(i) High altitude and space applications where atmospheric oxygen is not available, eg. Launch vehicles and satellite control rockets.

(ii) All applications where high thrust are required for short duration: missiles, boosters, JATO etc.

Rockets in Space Applications

There are a variety of rockets when it comes to launching and satellite control. Many of these are non-chemical in nature but are restricted to extremely low thrust levels.

Sl No	Type	Order of Magnitude of Thrust (N)	F/W	Operational Time	I_{sp} (sec)	Applications
1	Solar sail (not a rocket in fact)	10^{-5}	10^{-4}	Years	∞	Satellite Altitude control
2	Electric Prop. (Electro thermal, Electro Static, Electro Magnetic)	10^{-6} - 10^{-2}	10^{-5}-10^{-3}	Years	150 - 6000	Satellite control, stabilization, orbit maneuver
3	Stored cold gas (N_2, NH_3 etc.)	10^{-2} - 10^{-1}	~ 10^{-3}	Years	50 - 100	-do-
4	Nuclear Rocket	upto 10^5	20-30	Minute to hours	800	Interplanetary and space travel
5	Chemical Rocket (Solid, Liquid and Hybrid)	upto 10^7	upto 80	Seconds to minutes@	150-450	Launch vehicles, Missiles, Control rockets, Sounding rockets, JATO etc

@ shuttle main engine operate for about 8 min at a time but over 7 hrs cumulatively.

Classification of Chemical Rockets

Depending on the context, the chemical rockets are classified in many ways as follows:

(a) Type of propellant: Solid, Liquid (mono propellant / bipropellant and hybrid rockets)
(b) Application: Launch vehicle, ABM's, JATO's, ICBM, IRBM, SAM etc.
(c) Size of Unit (and thrust level sometimes): 10 ton, 100 kg etc.
(d) Type of subsystem: Turbopump fed, clustering, grain type etc.

Specific Impulse

- The specific impulse of a rocket, I_{sp}, is the ratio of the thrust to the flow rate of the weight ejected, that is

$$I_{sp} = \frac{F}{\dot{m}g} \qquad (2.23)$$

where F is thrust, $\dot{m}$ is the rate of mass flow, and g is the acceleration of gravity at ground level.

- Specific impulse is expressed in seconds. When the thrust and the flow rate remain constant throughout the burning of the propellant, the specific impulse is the time for which the rocket engine provides a thrust equal to the weight of the propellant consumed.

- For a given engine, the specific impulse has different values on the ground and in the vacuum of space because the ambient pressure is involved in the expression for the thrust. It is therefore important to state whether specific impulse is the value at sea level or in a vacuum.

- There are a number of losses within a rocket engine, the main ones being related to the inefficiency of the chemical reaction (combustion) process, losses due to the nozzle, and losses due to the pumps.

- Overall, the losses affect the efficiency of the specific impulse. This is the ratio of the real specific impulse (at sea level, or in a vacuum) and the theoretical specific impulse obtained with an ideal nozzle from gases coming from a complete chemical reaction. Calculated values of specific impulse are several percent higher than those attained in practice.

- From Equation (2.8) we can substitute $\dot{m}C$ for F in Equation (2.23), thus obtaining

$$I_{sp} = \frac{C}{g}, \quad \text{or}$$

$$C = g\,I_{sp} \qquad (2.24)$$

- Equation (2.24) is very useful when solving Equations (2.18) through (2.21). It is rare we are given the value of C directly, however rocket engine specific impulse is a commonly given parameter.

Internal Ballistics

The parameters that govern the burning rate and mass discharge rate of rocket motors are called internal ballistic properties; they include

- r – propellant burning rate (velocity of consumption), m/sec or mm/sec or in/sec.
- K – ratio of burning surface to throat area, A_b/A_t
- σ_p – temperature sensitivity of burning rate, expressed as percent change of burning rate per degree change in propellant temperature at a particular value of chamber pressure.
- π_K - temperature sensitivity of pressure expressed as percent change of chamber pressure per degree change in propellant temperature at a particular value of K,

and the influences caused by pressure, propellant ingredients, gas velocity, or acceleration. The subsequent solid propellant rocket parameters are performance parameters; they include thrust, ideal exhaust velocity, specific impulse, propellant mass fraction, flame temperature, temperature limits and duration.

Propellant Burning Rate

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.**

Solid propellant burns normal to its surface. The (average) burning rate, r, is defined as the regression of the burning surface per unit time. For a given propellant, the burning rate is mainly dependent on the pressure, p, and the initial temperature, T_i, of the propellant. Burning rate is also a function of propellant composition. For composite propellants it can be increased 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

Apart from the propellant formulation and propellant manufacturing process, burning ratein 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)

Burning rate data are usually obtained in three ways – namely, from testing by:

1. Standard strand burners, often called Crawford burners
2. Small-scale ballistic evaluation motors
3. Full-scale motors with good instrumentation

A strand burner is a small pressure vessal (usually with windows) in which a thin strand or bar of propellant is ignited at one end and burned to the other end. The strand can be inhibited with an external coating so that it will burn only on the exposed cross-sectional surface; chamber pressure is simulated by pressurizing the container with inert gas. The burning rate can be measured by electric signals from embedded wires, by ultrasonic waves, or by optical means. The burning rate measured on strand burners is usually lower than that obtained from actual rocket motor firing (by 4 to 12%) because it does not truly simulate the hot chamber environment of an actual rocket motor. Also small ballistic evaluation motors usually have a slightly lower burning rate than full-scale large motors, because of scaling factors.

During development of a new or modified solid propellant, it is tested extensively or characterized. This includes the testing of the burn rate (in several different ways) under different temperatures, pressures, impurities, and conditions. It also requires measurements of physical, chemical, and manufacturing properties, ignitability, aging, sensitivity to various energy inputs or stimuli (e.g., shock, friction, fires), moister absorption, compatibility with other materials (liners, insulators, cases), and other characteristics. It is a lengthy, expensive, often hazardous program with many tests, samples, and analyses.

The burning rate of propellant in a motor is a function of many parameters, and at any instant governs the mass flow rate $\dot{m}$ of hot gas generated and flowing from the motor (stable combustion);

$$\dot{m} = A_b \, r \, \rho_b$$

Here A_b is the burning area of the propellant grain, r the burning rate, and ρ_b the solid propellant density prior to motor start. The total mass m of effective propellant burned can be determined by integrating the above equation,

$$m = \int \dot{m}\, dt = \rho_b \int A_b\, r\, dt$$

Where A_b and r vary with time and pressure.

Burning Rate Relation with Pressure

Classical equations relating to burning rate are helpful in preliminary design, data extrapolation, and understanding the phenomena. Unless otherwise stated, burning rate is expressed for 70^{o}F or 294 K propellant (prior to ignition) burning at a reference chamber pressure of 1000 psia or 6.895 MPa. For most production-type propellant the burning rate is approximated as a function of chamber pressure, at least for a limited range of chamber pressures, which is given as

$$r = a P^{n}$$

where r, the burn rate, is usually in centimeter per second and chamber pressure P is in MPa; a is an empirical constant influenced by ambient temperature. Also a is known as the temperature coefficient and it is NOT dimensionless. The burning rate exponent n, sometimes called the combustion index, is independent of the initial grain temperature and describes the influence of chamber pressure on the burning rate.

Burning Rate Relation with Temperature

Temperature affects chemical reaction rates and the initial ambient temperature of a propellant grain prior to combustion influences burning rate.

The sensitivity of burning rate to propellant temperature can be expressed in the form of temperature coefficient, the two most common being

$$\sigma_p = \left(\frac{\partial \ln r}{\partial T}\right)_p = \frac{1}{r}\left(\frac{\partial r}{\partial T}\right)_p$$

$$\pi_K = \left(\frac{\partial \ln p}{\partial T}\right)_K = \frac{1}{P}\left(\frac{\partial P}{\partial T}\right)_K$$

with σ_p , temperature sensitivity of burning rate and π_K, temperature sensitivity of pressure.

The coefficient σ_p (typically 0.001 – 0.009 / K) for a new propellant is usually calculated from strand burner test data, and π_K (typically 0.067 – 0.278 % / ^{o}C) from small-scale or full-scale motors. Mathematically, these coefficients are the partial derivatives of the natural logarithm of the burning rate r or the chamber pressure p, respectively, with respect to propellant temperature T.

The values of π_K and σ_p depend primarily on the nature of the propellant burning rate, the composition, and the combustion mechanism of the propellant. It is possible to derive a relationship between the two temperature sensitivities, namely

$$\pi_K = \frac{1}{1-n}\sigma_p$$

This formula is usually valid when the three variables are constant over the chamber pressure and temperature range.

The temperature sensitivity σ_p can be also expressed as

$$\sigma_p = \left(\frac{\partial \ln(aP^n)}{\partial T}\right)_p = \frac{1}{a}\frac{da}{dT}$$

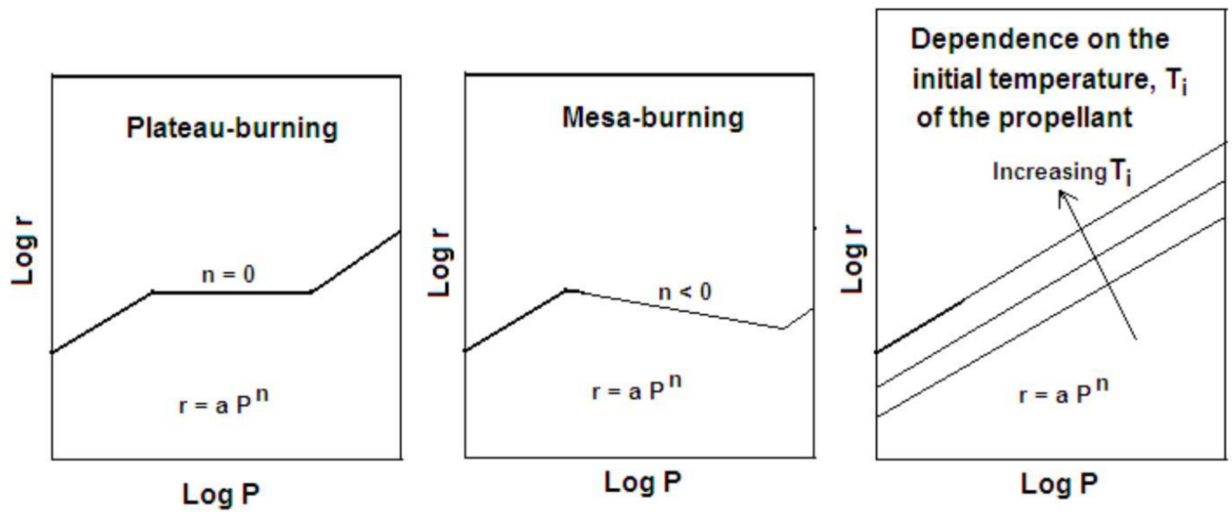


Fig. Burning rate versus pressure

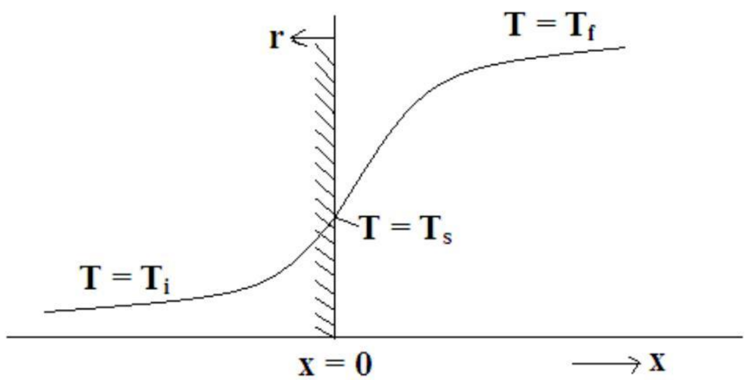


Fig. Coordinate system and temperature profile for a stationary burning solid propellant

Equilibrium chamber pressure

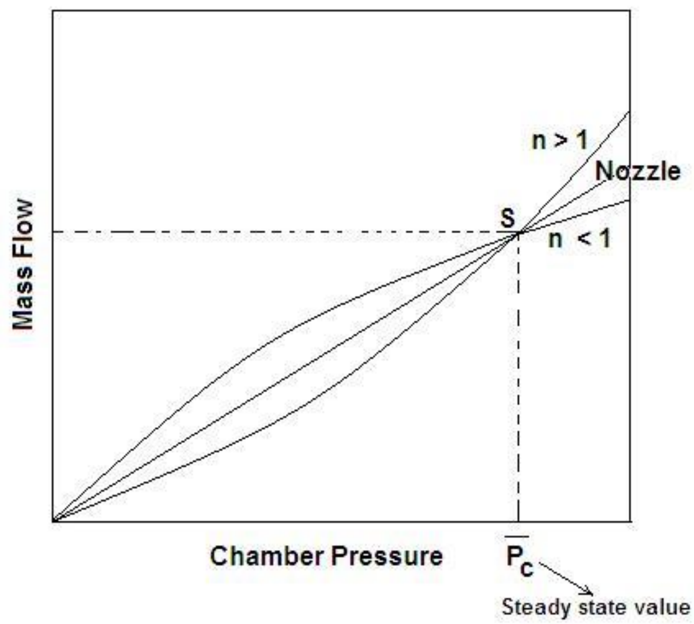


Fig. The balance of mass in a solid rocket motor with reference to the burning rate index, n

In the above figure the straight line through the origin and point 'S' depicts the mass flow through the nozzle as a function of P_c. At point S there is a balance between mass production and outflux of the mass. At higher pressures (> $\bar{P}_c$) the mass flow through the nozzle is larger than the production at the burning surface in case n < 1 and the reverse happens for n > 1. Thus if n < 1 the pressure will drop to its steady-state value $\bar{P}_c$. Note that when n < 1 even at higher chamber pressure rocket motor will back to its designed equilibrium chamber pressure and ensure a stable operation. On the other hand when n>1 these types of situations will possibly lead to over-pressurization and rupture of the rocket motor or depressurization and flame out.

Erosive Burning

Erosive burning refers to the increase in the propellant burning caused by the high-velocity flow of combustion gases over the burning propellant surface. It can seriously affect the performance of solid propellant rocket motors. It occurs primarily in the port passages or perforations of the grain as the combustion gases flow toward the nozzle; it is more likely to occur ehen the port passage cross-sectional area A is small relative to the throat area A_t with a port-to-throat area ratio of 4 or less. The high velocity near the burning surface and the turbulent mixing in the boundary layers increase the heat transfer to the solid propellant and thus increase the burning rate.

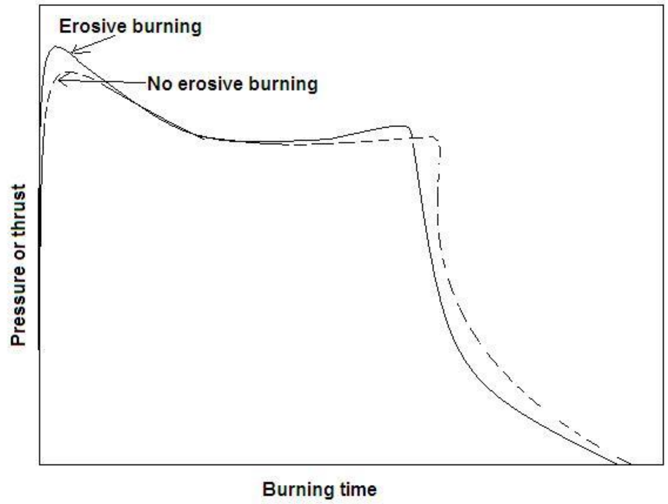


Fig. Typical pressure-time curve with and without erosive burning

Erosive burning increases the mass flow and thus also the chamber pressure and thrust during the early portion of the burning for a particular motor (see above Fig.). Erosive burning causes early burnout of the web, usually at the nozzle end, and exposes the insulation and aft closure to hot combustion gas for a longer period of time; this usually requires more insulation layer thickness (and more inert mass) to prevent local thermal failure. In designing motors, erosive burning is either avoided or controlled to be reproducible from one motor to the next.

Total burning rate = steady state burning rate (aP_c^n) + erosive burning

Basic Performance Relations

One basic performance relation derived from the principle of conservation of matter. The propellant mass burned per unit time has to equal the sum of the change in gas mass per unit time in the combustion chamber grain cavity and the mass flowing out through the exhaust nozzle per unit time.

$$A_b r \rho_b = \frac{d}{dt}(\rho_1 V_1) + A_t P_1 \sqrt{\frac{k}{RT_1}\left(\frac{2}{k+1}\right)^{(k+1)/(k-1)}}$$

The term on the left side of the equation gives the mass rate of gas generation. The first term on the right gives the change in propellant mass in the gas volume of the combustion chamber, and the last term gives the nozzle flow. The burning rate of propellant is r; A_b is the propellant burning area; ρ_b is the solid propellant density; ρ_1 is the combustion gas density; V_1 is the chamber gas cavity volume, which becomes larger as the propellant is expended; A_t is the throat area; P_1 is the chamber pressure; T_1 is the absolute chamber temperature, which is usually assumed to be constant; and k is the specific heat ratio of the combustion gases. During startup the changing mass of propellant in the grain cavity becomes important.

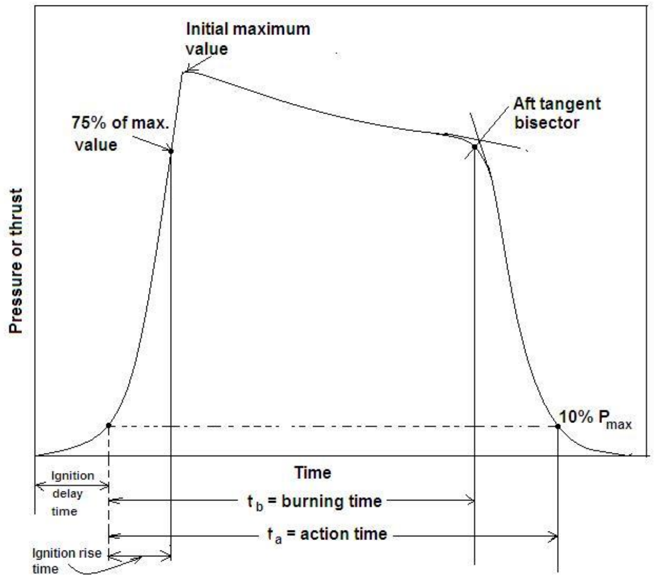


Fig. Definition of burning time and action time

Isentropic Flow through Nozzles

In a converging diverging nozzle a large fraction of the thermal energy of the gases in the chamber is converted into kinetic energy. As will be explained, the gas pressure and temperature drop dramatically and gas velocity can reach values in excess of around 3.2 km/sec. This is a reversible, essentially isentropic flow process.

If a nozzle inner wall has a flow obstruction or a wall protrusion (a piece of weld splatter or slag), then the kinetic gas energy is locally converted back into thermal energy essentially equal to the stagnation temperature and stagnation pressure in the chamber. Since this would lead quickly to a local overheating and failure of the wall, nozzle inner walls have to be smooth without any protrusion.

Nozzle exit velocity can be derived as,

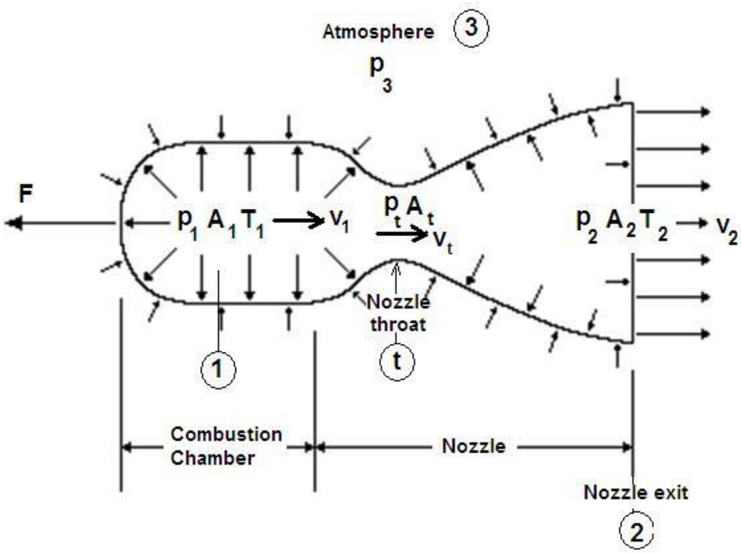
$$v_2 = \sqrt{\frac{2k}{k-1} R T_1 \left[1 - \left(\frac{P_2}{P_1}\right)^{(k-1)/k}\right] + v_1^2}$$

This relation also holds for any two points within the nozzle. Note that when the chamber section is large compared to the nozzle throat section, the chamber velocity or nozzle approach velocity is comparatively small and the v_1^2 can be neglected. The chamber temperature T_1 is at the nozzle inlet and, under isentropic condition, differ little from the stagnation temperature or (for a chemical rocket) from combustion temperature. This leads to an important simplified expression of the exhaust velocity v_2, which is often used in the analysis.

$$v_2 = \sqrt{\frac{2k}{k-1} R T_1 \quad \frac{P_2}{P_1}^{(k-1)/k}}$$

$$= \sqrt{\frac{2k}{k-1} \frac{R' T_o}{M} \; 1 - \frac{P_2}{P_1}^{(k-1)/k}}$$

Thrust and Thrust Coefficient



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

$$F = C_F A_t P_1$$

Where C_F is the thrust coefficient, which can be derived as a function of gas property k, the nozzle area ratio (A_2/A_t), and the pressure ratio across the nozzle p_1/p_2 , but independent of chamber temperature. For any fixed pressure ratio (p_1/p_3) the thrust coefficient C_F and the thrust F have a peak when $p_2 = p_3$. This peak value is known as optimum thrust coefficient.

$$C_F = \sqrt{\frac{2k^2}{k-1}\left(\frac{2}{k+1}\right)^{(k+1)/(k-1)}\left[1-\left(\frac{p_2}{p_1}\right)^{(k-1)/k}\right]} + \frac{p_2 - p_3}{p_1}\frac{A_2}{A_t}$$

Effective Exhaust Velocity

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

$$c = I_{sp}\, g_o = \frac{F}{\dot{m}}$$

It is usually given in meters per second.

The concept of weight relates to the gravitational attraction at or near sea level, but in space or outer satellite orbits, “weight” signifies the mass multiplied by an arbitrary constant, namely g_o. In system international (SI) or metric system of units I_{sp} can be expressed simply in “seconds”, because of the use of the constant g_o.

Specific Propellant Consumption

Specific propellant consumption is the reciprocal of the specific impulse.

Mass Ratio

The mass ratio of a vehicle or a particular vehicle stage is defined to be the final mass m_f (after rocket operation has consumed all usable propellant) divided by initial mass m_o (before rocket operation).

$$\text{Mass Ratio, MR} = \frac{m_f}{m_o}$$

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

Propellant Mass Fraction, ζ

Propellant mass fraction 'ζ' is defined as the ratio of propellant mass 'm_p' to the initial mass 'm_o'

$$\zeta = \frac{m_p}{m_o}$$

$$m_o = m_f + m_p$$

Characteristic Velocity

The characteristic velocity has been used frequently in the rocket propulsion literature. It is represented by a symbol C^*. It is defined as,

$$C^* = \frac{p_1 A_t}{\dot{m}}$$

The characteristic velocity is used in comparing the relative performance of different chemical rocket propulsion system designs and propellants. It is basically a function of the propellant characteristics. It is easily determined from data of $\dot{m}$, p_1, and A_t. It relates to the efficiency of the combustion and is essentially independent of nozzle characteristics. However, the specific impulse and the effective exhaust velocities are functions of the nozzle geometry, such as the nozzle area ratio.

Problems:

Q1.

A rocket projectile has the following characteristics: Initial mass = 200kg

Mass after rocket operation = 130 kg

Payload, nonpropulsive structure, etc., = 110 kg

Rocket operation duration = 3.0 sec

Average specific impulse of propellant = 240 sec

Determine the vehicle's (i) mass ratio, (ii) propellant mass fraction, (iii) propellant flow rate, (iv) thrust, (v) thrust-to-weight ratio, (vi) acceleration of the vehicle, (vii) effective exhaust velocity, (viii) total impulse, and (ix) the impulse to weight ratio.

Solution:

(i) Mass ratio of vehicle, m_f/m_p = 130/200 = 0.65

Mass ratio of rocket system = m_f/m_o = (130-110)/(200-110) = 20/90 = 0.222

(ii) Propellant mass fraction = $(m_o - m_f)/m_o$ = (90-20)/90 = 0.778

(iii)Propellant mass flow rate = 70/3 = 23.3 kg/sec

(iv) Thrust = $I_{sp} \dot{m} g_o$ = 240 x 23.3 x 9.81 = 54.857 N

(v) Thrust-to-weight ratio of the vehicle is

Initial value = 54,857 / (200 x 9.81) = 28

Final value = 54,857 / (130 x 9.81) = 43

(vi) Maximum acceleration of the vehicle is 43 x 9.81 = 421 m/sec^2

(vii) The effective exhaust velocity is

$c = I_{sp} g_o$ = 240 x 9.81 = 2354 m/sec

(viii) Total impulse = I_{sp} w = 240 x 70 x 9.81 = 164, 808 N-sec

This result can also be obtained by multiplying the thrust by the duration.

(ix) The impulse to weight ratio of the propulsion system is

= 164,808 / [(200-110) x 9.81] = 187

Q2. The following measurements were made in a sea level test of a solid propellant rocket motor:

Burn duration = 40 sec

Initial mass before test = 1210 kg

Mass of rocket motor after test = 215 kg Average thrust = 62,250 N

Chamber pressure = 7.00 MPa

Nozzle exit pressure = 0.070 MPa

Nozzle throat diameter = 0.855 m

Nozzle exit diameter = 0.2703 m

Determine $\dot{m}, v_2, c^*, c$ and I_{sp} at sea level, and c and I_{sp} at 1000 and 25,000 m altitude.

Assume an invariant thrust and mass flow rate and negligible short start and stop transients.

Solution:

Mass flow rate = (initial motor mass – final motor mass)/burn time = (1210 – 215) / 40 = 24.9 kg/sec

The nozzle areas at the throat and exit are

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

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

The actual exhaust velocity

$$v_2 = \frac{(F - (p_2 - p_3) A_2)}{\dot{m}}$$

= (62,250 – (0.070-0.1013) 10^6 x 0.0574) / 24.9

= 2572 m/sec

$$C^* = \frac{p_1 A_t}{\dot{m}}$$

= 7.00 x 10^6 x 0.00574/24.9 = 1613 m/sec

I_{sp} = 62,250 / (24.9 x 9.81) = 255 sec

c = 255 x 9.81 = 2500 m/sec

For altitudes of 1000 and 25, 000 m the ambient pressure (see atmospheric table) is 0.898 and 0.00255 MPa.

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

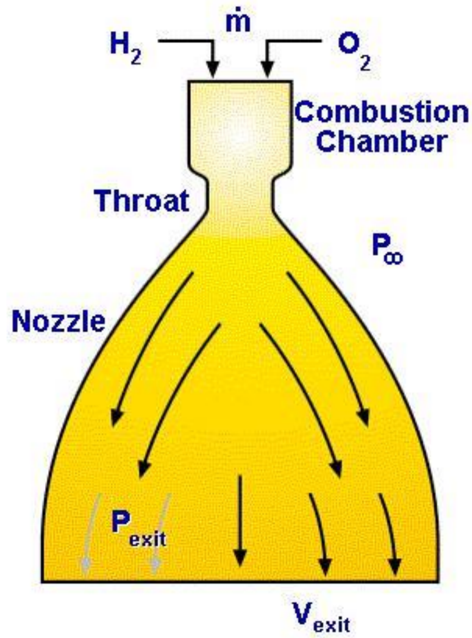
At 1000 m altitude,
c = 2572 + (0.070-0.898) x 10^6 x 0.0574/24.9 = 2527 m/sec
I_{sp} = 2527/9.81 = 258 sec

At 25,000 m altitude,
c = 2572 + (0.070-0.00255) x 10^6 x 0.0574/24.9 = 2727 m/sec
I_{sp} = 2727/9.81 = 278 sec

Rocket Nozzles

Purpose:

The nozzle is the component of a rocket or air-breathing engine that produces thrust. This is accomplished by converting the thermal energy of the hot chamber gases into kinetic energy and directing that energy along the nozzle's axis, as illustrated below.



Simple representation of a rocket nozzle

Although simplified, this figure illustrates how a rocket nozzle works. The propellant is composed of a fuel, typically liquid hydrogen (H_2), and an oxidizer, typically liquid oxygen (O_2). The propellant is pumped into a combustion chamber at some rate $\dot{m}$ (mdot) where the fuel and oxidizer are mixed and burned. The exhaust gases from this process are pushed into the throat region of the nozzle. Since the throat is of less cross-sectional area than the rest of the engine, the gases are compressed to a high pressure. The nozzle itself gradually increases in cross-sectional area allowing the gases to expand. As the gases do so, they push against the walls of the nozzle creating thrust.

Mathematically, the ultimate purpose of the nozzle is to expand the gases as efficiently as possible so as to maximize the exit velocity (v_{exit}). This process will maximize the thrust (F) produced by the system since the two are directly related by the equation

$$F = \dot{m}\, v_{exit} + (p_{exit} - p_{\infty})\, A_{exit}$$

where

F = thrust force

$\dot{m}$ = mass flow rate

v_{exit} = exhaust gas velocity at the nozzle exit

p_{exit} = pressure of the exhaust gases at the nozzle

exit p_{∞} = ambient pressure of the atmosphere

A_{exit} = cross-sectional area of the nozzle exit

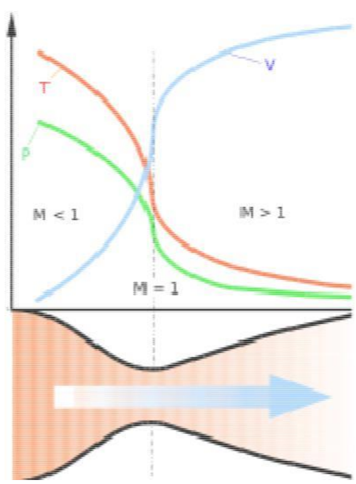
Expansion Area Ratio:

In theory, the only important parameter in rocket nozzle design is the expansion area ratio ( ), or the ratio of exit area (A_{exit}) to throat area (A_{throat}).

$$\varepsilon = \frac{A_{exit}}{A_{throat}}$$

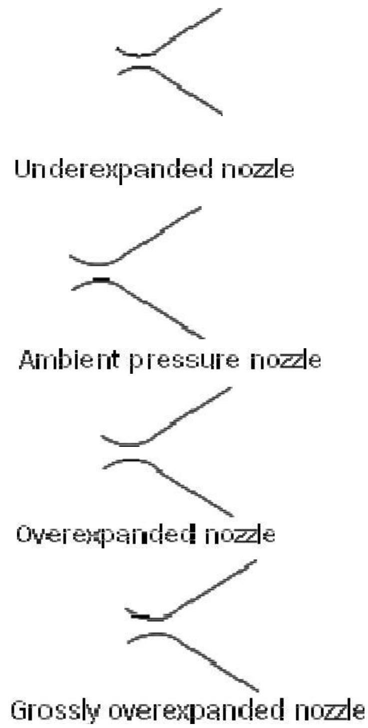
Fixing all other variables (primarily the chamber pressure), there exists only one such ratio that optimizes overall system performance for a given altitude (or ambient pressure). However, a rocket typically does not travel at only one altitude. Thus, an engineer must be aware of the trajectory over which a rocket is to travel so that an expansion ratio that maximizes performance over a range of ambient pressures can be selected.

Nevertheless, other factors must also be considered that tend to alter the design from this expansion ratio-based optimum. Some of the issues designers must deal with are nozzle weight, length, manufacturability, cooling (heat transfer), and aerodynamic characteristics.



Typical temperatures (T) and pressures (p) and speeds (v) in a De Laval Nozzle

Maximum thrust for a rocket engine is achieved by maximizing the momentum contribution of the equation without incurring penalties from over expanding the exhaust. This occurs when $P_e = P_{amb}$. Since ambient pressure changes with altitude, most rocket engines spend very little time operating at peak efficiency.



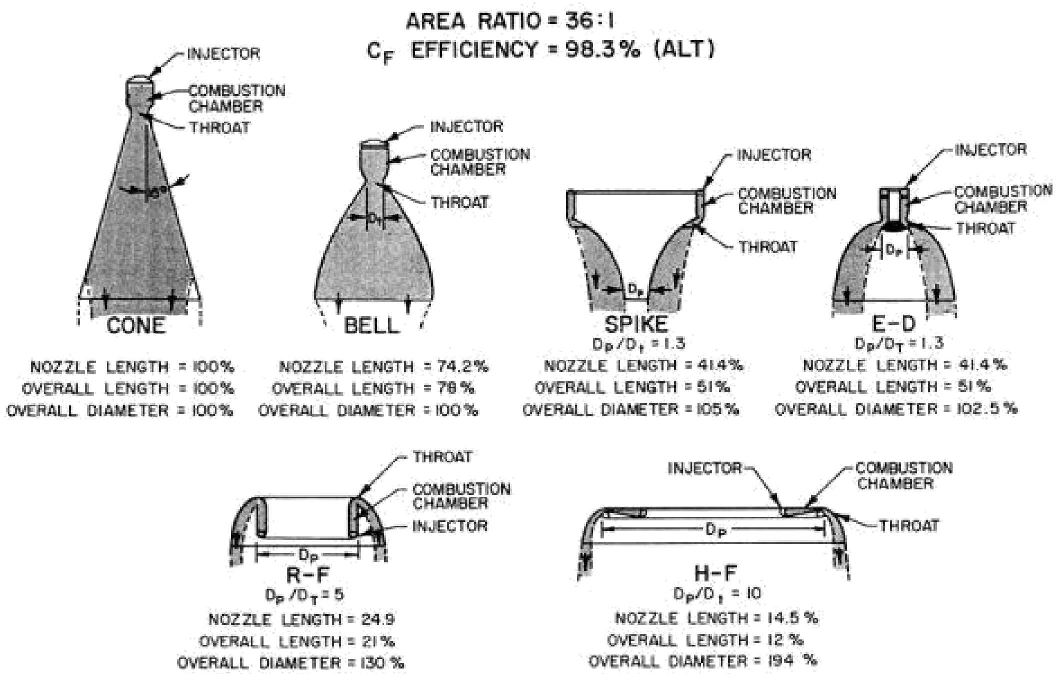
If the pressure of the exhaust jet varies from atmospheric pressure, nozzles can be said to be underexpanded, ambient or overexpanded. If under or overexpanded then loss of efficiency occurs, grossly overexpanded nozzles lose less efficiency, but the exhaust jet is usually unstable. Rockets become progressively more underexpanded as they gain altitude. Note that almost all rocket engines will be momentarily grossly overexpanded during startup in an atmosphere.

Rocket Nozzle Shapes

Not all rocket nozzles are alike, and the shape selected usually depends on the application. This section discusses the basic characteristics of the major classes of nozzles used today.

Nozzle Comparisons:

To date three major types of nozzles, the cone, the bell or contoured, and the annular or plug, have been employed. Each class satisfies the design criteria to varying degrees. Examples of these nozzle types can be seen below.



Size comparison of optimal cone, bell, and radial nozzles for a given set of conditions

Conical Nozzle:

The conical nozzle was used often in early rocket applications because of its simplicity and ease of construction. The cone gets its name from the fact that the walls diverge at a constant angle. A small angle produces greater thrust, because it maximizes the axial component of exit velocity and produces a high specific impulse (a measure of rocket efficiency). The penalty, however, is a longer and heavier nozzle that is more complex to build. At the other extreme, size and weight are minimized by a large nozzle wall angle. Unfortunately, large angles reduce performance at low altitude because the high ambient pressure causes overexpansion and flow separation.

Bell Nozzle:

The bell, the most commonly used nozzle shape, offers significant advantages over the conical nozzle, both in size and performance. Referring to the above figure, note that the bell consists of two sections. Near the throat, the nozzle diverges at a relatively large angle but the degree of diveregence tapers off further downstream. Near the nozzle exit, the diveregence angle is very small. In this way, the bell is a compromise between the two extremes of the conical nozzle since it minimizes weight while maximizing performance. The most important design issue is to contour the nozzle to avoid oblique shocks and maximize performance. However, we must remember that the final bell shape will only be the optimum at one particular altitude.

Annular Nozzles:

The annular nozzle, also sometimes known as the plug or "altitude-compensating" nozzle, is the least employed of those discussed due to its greater complexity. The term "annular" refers to the fact that combustion occurs along a ring, or annulus, around the base of the nozzle. "Plug" refers to the centerbody that blocks the flow from what would be the center portion of a traditional nozzle. "Altitude-compensating" is sometimes used to describe these nozzles since that is their primary advantage, a quality that will be further explored later.

Before describing the various forms of annular nozzles, it is useful to mention some key differences in design parameters from the conical or bell nozzles. The expansion area ratio for a traditional nozzle has already been discussed. When considering an annular nozzle, the area of the centerbody (A_{plug}) must also be taken into account.

$$\varepsilon = \frac{A_{exit} - A_{plug}}{A_{throat}}$$

Another parameter particular to this type of nozzle is the annular diameter ratio, D_p / D_t, or the ratio of the centerbody diameter to that of the throat. The ratio is used as a measure of the nozzle geometry for comparison with other plug nozzle shapes. Typical values of this ratio appear in the above figure.

Annular Nozzles I

Having introduced the three principal families of nozzle shapes, we will now look more closely at the two major subclasses of annular, or plug, nozzles.

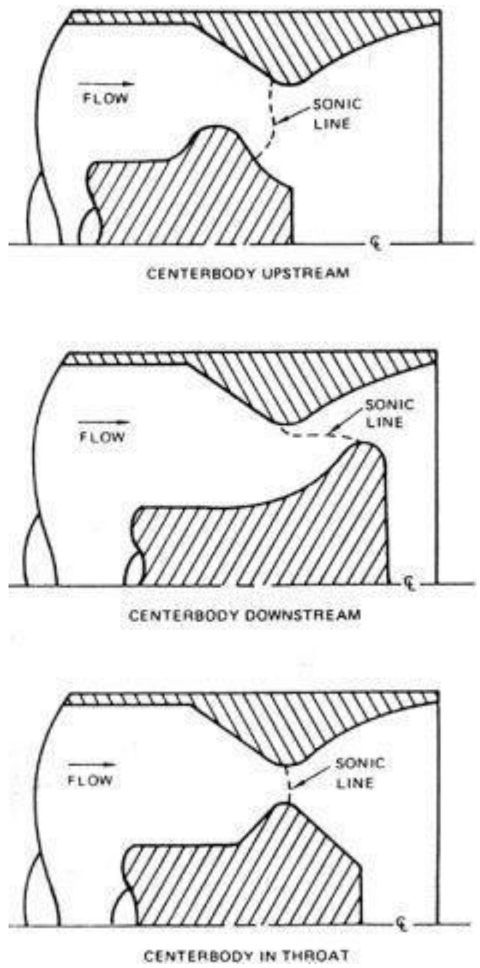
Radial Out-Flow Nozzles:

Two major types of plug nozzles have been developed to date. They are distinguished by the method in which they expand the exhaust, outward or inward. The radial out-flow nozzle was the subject of much research in the late 1960s and early 1970s. Examples of this type are the expansion-deflection (E-D), reverse-flow (R-F), and horizontal-flow (H-F) nozzles shown in the figure above.

The name of each of these nozzles indicates how it functions. The expansion-deflection nozzle works much like a bell nozzle since the exhaust gases are forced into a converging throat region of low area before expanding in a bell-shaped nozzle. However, the flow is deflected by a plug, or centerbody, that forces the gases away from the center of the nozzle. Thus, the E-D is a radial out-flow nozzle.

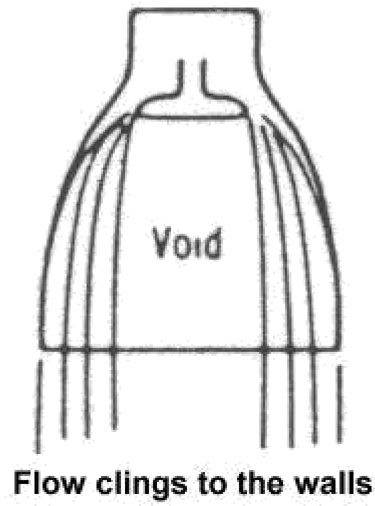
The reverse-flow nozzle gets its name because the fuel is injected from underneath, but the exhaust gases are rotated 180° thereby reversing their direction. Similarly, the fuel in the horizontal-flow nozzle is injected sideways, but the exhaust is rotated 90°.

Judging by the amount of literature obtained on this subject, little work has been done on the R-F and H-F nozzles, and they will not be considered further. The E-D, on the other hand, has been one of the most studied forms of annular nozzles. While similar in nature to the bell nozzle, the most notable difference is the addition of a centerbody. As shown below, this "plug" may be located upstream of, downstream of, or in the throat, with each location resulting in better performance for a given set of operating conditions.



Comparison of centerbody locations in Expansion-Deflection nozzles [from Conley et al, 1984]

The purpose of the centerbody is to force the flow to remain attached to, or to stick to, the nozzle walls, as illustrated in the following figure.



Expansion-deflection nozzle flow behavior at low altitude
[from Sutton, 1992]

This behavior is desirable at low altitudes because the atmospheric pressure is high and may be greater than the pressure of the exhaust gases. When this occurs, the exhaust is forced inward and no longer exerts force on the nozzle walls, so thrust is decreased and the rocket becomes less efficient. The centerbody, however, increases the pressure of the exhaust gases by squeezing the gases into a smaller area thereby virtually eliminating any loss in thrust at low altitude.

Annular Nozzles II

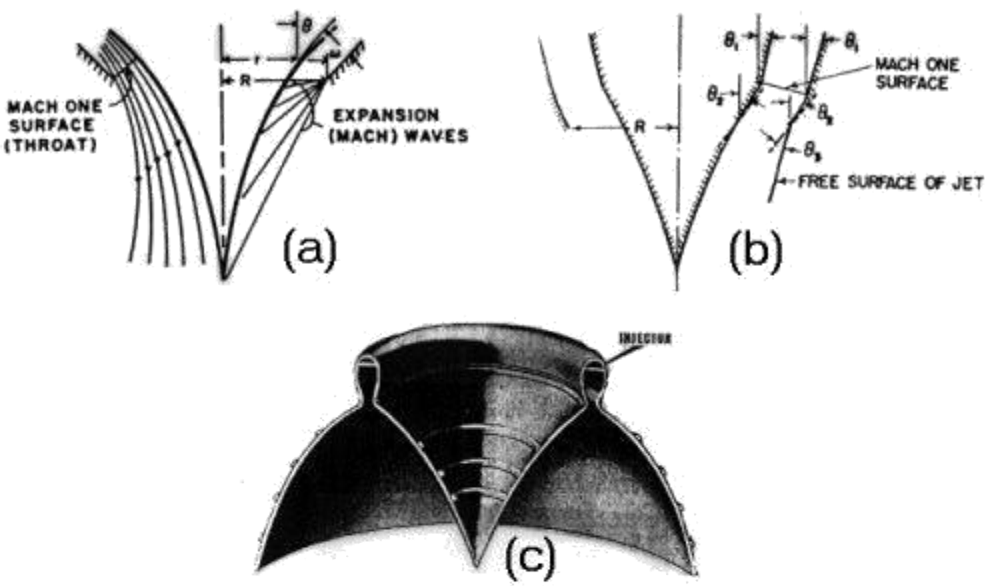
Having introduced the three principal families of nozzle shapes and discussed the radial out-flow nozzle, we will now look more closely at the second class of annular nozzles.

Radial In-Flow Nozzles:

The second major variety of annular nozzles is the radial in-flow type, exemplified by the spike shown above.

This type of nozzle, named for the prominent spike centerbody, is often described as a bell turned inside out. However, the nozzle shown above is only one of many possible spike configurations. Variations of this design, shown below, include

(a) a traditional curved spike with completely external supersonic expansion
(b) a similar shape in which part of the expansion occurs internally
(c) a design similar to the expansion-deflection nozzle in which all expansion occurs internally.



Comparison of spike nozzles with (a) external expansion, (b) internal-external expansion, and (c) internal expansion [from Berman and Crimp, 1961]

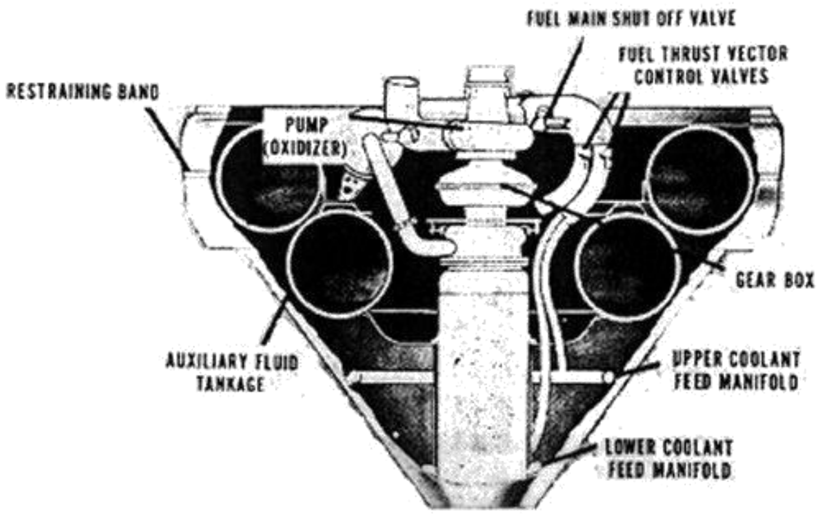
Note that each of the above spike nozzles features a curved, pointed spike, the most ideal shape. This spike shape allows the exhaust gases to expand through an isentropic, or constant entropy, process. In so doing, the nozzle efficiency is maximized and no energy is lost because of turbulent mixing in the exhaust flow. While the isentropic spike may be most efficient, it also tends to be prohibitively long and heavy. However, theoretical studies have shown that replacing the curved shape by a much shorter and easier to construct cone results in very little performance loss. The following graph illustrates that the thrust decreases by less than 1% for cone half-angles up to 30°. Furthermore, the graph gives an indication of how much the spike length can be reduced by employing a cone-shaped spike.

Aerospike Nozzles

A further subclass of the radial in-flow family of spike nozzles is known as the aerospike.

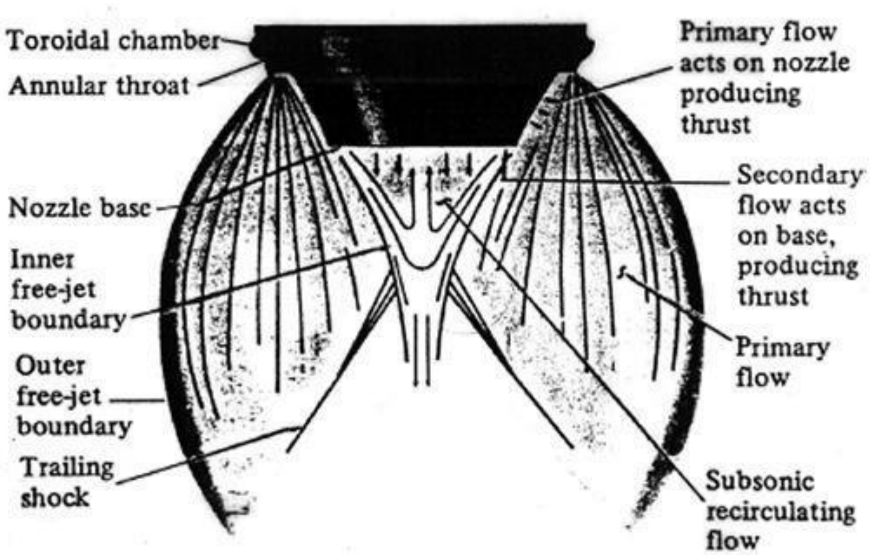
Aerospike Nozzles:

Previously, we discussed methods of reducing the length of a spike nozzle centerbody by replacing the ideal spike with a conical spike. While this method does indeed result in a much shorter nozzle length, we can go even further by removing the pointed spike altogether and replacing it with a flat base. This configuration is known as a truncated spike, an example of which is shown below.



Example of a truncated, conical spike [from Berman and Crimp, 1961]

As any fluid dynamicist recognizes, the significant disadvantage of the "flat" plug is that a turbulent wake forms aft of the base at high altitudes resulting in high base drag and reduced efficiency. However, this problem can be greatly alleviated in an improved version of the truncated spike that introduces a "base bleed," or secondary subsonic flow, into the region aft of the base.

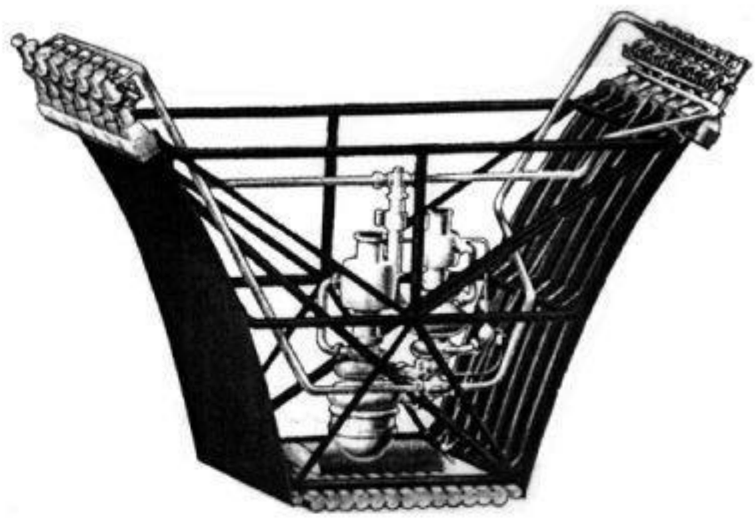


Example of an aerospike nozzle with a subsonic, recirculating flow [from Hill and Peterson, 1992]

The circulation of this secondary flow and its interaction with the engine exhaust creates an "aerodynamic spike" that behaves much like the ideal, isentropic spike. In addition, the secondary flow re-circulates upward pushing on the base to produce additional thrust. It is this artificial aerodynamic spike for which the aerospike nozzle is named.

Linear Aerospike:

All of the nozzles we have studied thus far have been annular, or circular when viewed from below. Still another variation of the aerospike nozzle is not an annular nozzle at all. A second approach, pioneered by the Rocketdyne company (now a division of Boeing) in the 1970s, places the combustion chambers in a line along two sides of the nozzle:

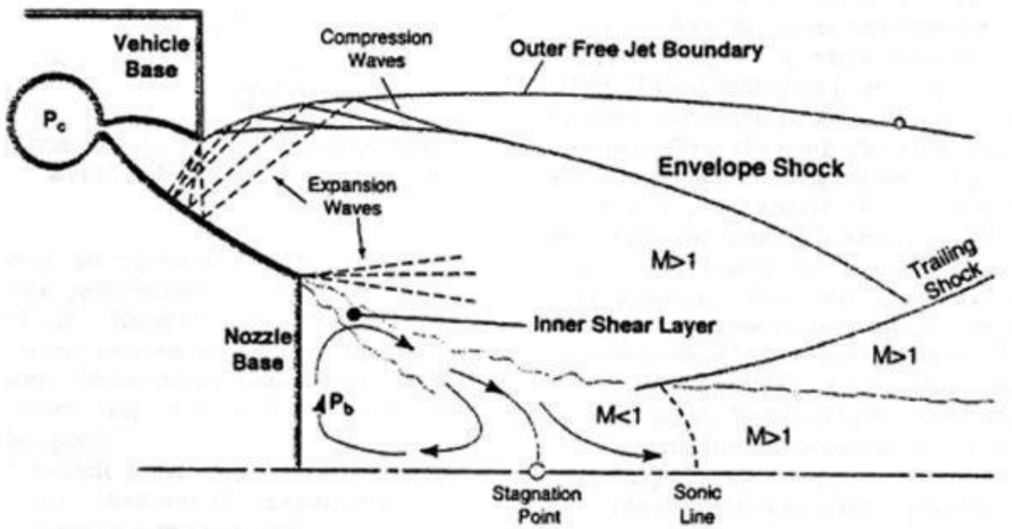


Rocketdyne RS-2200 linear aerospike engine [from Flinn, 1996]

This approach results in a more versatile design allowing the use of lower-cost modular combustors. These modules can be combined in varying configurations depending on the application.

Aerospike Flowfield:

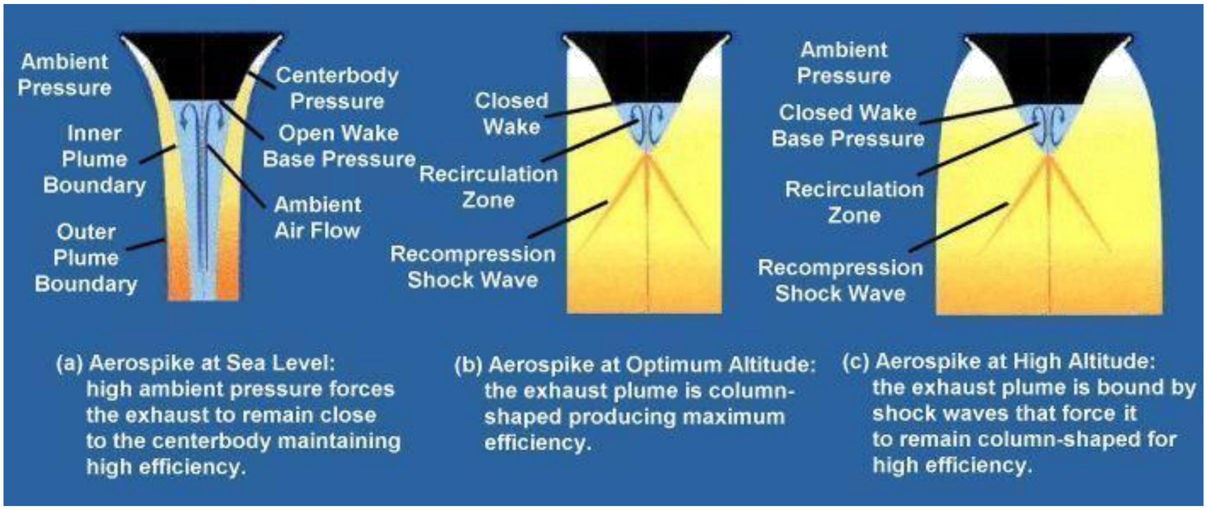
The exact nature of the exhaust flowfield behind an aerospike nozzle is currently the subject of much research. The most notable features of a typical aerospike nozzle flowfield are shown in more detail below.



Flowfield characteristics of an aerospike nozzle [from Ruf and McConnaughey, 1997]

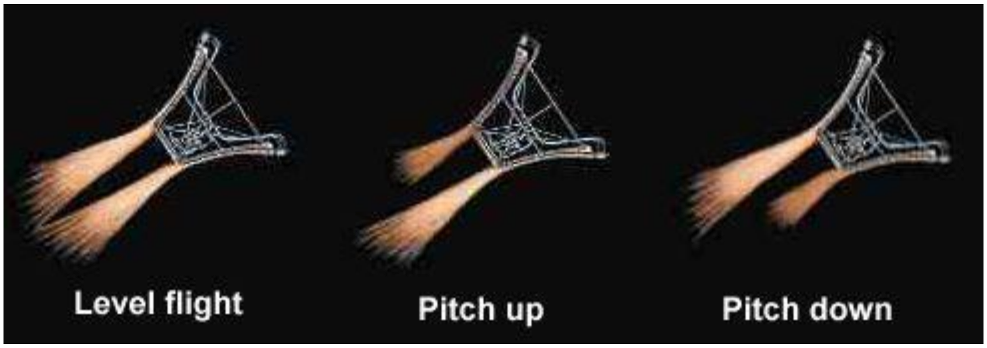
The primary exhaust can be seen expanding against the centerbody and then around the corner of the base region. The interaction of this flow with the re-circulating base bleed creates an inner shear layer. The outer boundary of the exhaust plume is free to expand to ambient pressure. Expansion waves can be seen emanating from the thruster exit lip, and these waves reflect from the centerbody contour to the free jet boundary. Compression waves are then reflected back and may merge to form the envelope shock seen in the primary exhaust.

At low altitude (high ambient pressure), the free boundary remains close to the nozzle (see below) causing the compression waves to reflect onto the centerbody and shear layer themselves. The waves impacting the centerbody increase pressure on the surface, thereby increasing the centerbody component of thrust. The waves impacting the shear layer, on the other hand, increase the circulation of the base flow thereby increasing the base component of thrust.



Aerospike nozzle behavior during flight [from Rocketdyne, 1999]

- **Thrust vectoring:** Because the combustion chambers can be controlled individually, the vehicle can be maneuvered using differential thrust vectoring. This eliminates the need for the heavy gimbals and actuators used to vary the direction of traditional nozzles.



Aerospike thrust vectoring control [from Rocketdyne, 1999]

Additional Reading

1. Sutton, G.P., "Rocket Propulsion Elements", John Wiley & Sons Inc., New York, 7th Edn., 2001.

UNIT-II

CHEMICAL ROCKETS

INTRODUCTION

The only known way to meet space-flight velocity requirements is through the use of the rocket in one of its several forms.

Rocket thrust is the reaction force produced by expelling particles at high velocity from a nozzle opening. These expelled particles may be solid, liquid, gaseous, or even bundles of radiant energy. The engine's ability to produce thrust will endure only so long as the supply of particles, or working fluid, holds out. Expulsion ofmaterial is the essence of the thrust production, and without material to expel no thrust can be produced, regardless of how much energy is available.

Because of this fundamental fact, a prime criterion for rating rocket performance is specific impulse, which provides an index of the efficiency with which a rocket uses its supply of propellant or working fluid for thrust production. For gaseous working fluids, specific impulse can be increased by (1) attaining higher temperatures in the combustion chamber and (2) increasing the proportion of lighter gases, preferably hydrogen, in the exhaust.

The other important factor in assessing the merit of a propulsion system in a given application is the weight of engine and working fluid container required, since these weights influence achievable propellant fraction.

TYPES OF ROCKET ENGINES

Rocket engines are distinguished b the type of mechanism used to produce exhaust material. The simplest "engine" is a compressed air bottle attached to a nozzle. The exhaust gas is stored in the same form as it appears in the exhaust. Ejection of compressed air, or other gas, from a nozzle is a perfectly satisfactory rocket operation for some purposes.

The most common rocket engine is the chemical type in which hot exhaust gases are produced by chemical combustion. The chemicals or propellants, are of two types, fuel and oxidizer corresponding to gasoline and oxygen in an automobile engine. Both are required for combustion. They may be solid or liquid chemicals.

In other types of rockets no chemical change takes place within the engines but the working fluid may be converted to a hot gas for ejection by the addition of heat from a nuclear reactor or some other energy source.

These and other variations of the rocket engine are discussed below.[1]

SOLID-PROPELLANT ROCKET

In the solid-chemical rocket, the fuel and oxidizer are intimately mixed together and cast into a solid mass, called a grain, in the combustion chamber (fig. 1). The propellant grain is firmly cemented to the inside of the metal or plastic case, and is usually cast with a hole down the center. This hole, called the perforation, may be shaped in various ways, as star, gear, or other more unusual outlines, The perforation shape and dimension affects the burning rate or number of pounds of gas generated per second and, thereby, the thrust of the engine.

After being ignited by a pyrotechnic device, which is usually triggered by an electrical impulse, the propellant grain burns on the entire inside surface of the perforation. The hot combustion gases pass down the grain and are ejected through the nozzle to produce thrust.

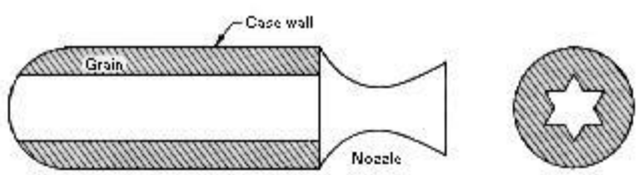


Fig.1-Schematic of solid-propellant rocket

The propellant grain usually consist of 1 of 2 types of chemical. One type is the double-base, which consists largely of nitroglycerine and nitrocellulose. It resembles smokeless gunpowder. The second type, which is now predominant, is the composite propellant, consisting of an oxidizing agent, such as ammonium nitrate or ammonium perchlorate intimately mixed with an organic or metallic fuel. Many of the fuels used are plastics, such as polyurethane.

A solid propellant must not only produce a desirable specific impulse, but it must also exhibit satisfactory mechanical properties to withstand ground handling and the flight environment.

Should the propellant grain develop a crack, for example, ignition would cause combustion to take place in the crack, with explosion as a possible result.

It can be seen from figure 1 that the case walls are protected from the hot gas by the propellant itself. Therefore, it is possible to use heat-treated alloys or plastics for case construction. The production of light-weight, high-strength cases is a major development problem in the solid-rocket field.

Since nozzles of solid rockets are exposed to the hot gas flowing through them, they must be of heavy construction to retain adequate strength at high temperature. Special inserts are often used in the region of the nozzle throat to protect the metal from the erosive effects of the flowing gas.

For vehicle guidance it is necessary to terminate thrust sharply upon command. This may be accomplished with solid rockets by blowing off the nozzle or opening vents in the chamber walls. Either of these techniques causes the pressure in the chamber to drop and, if properly done, will extinguish the flame.

The specific impulse of various solid-propellant rockets now falls in the range 175 to 250 seconds. The higher figure of 250 applies to ammonium perchlorate-biased propellants.[3]

LIQUID BIPROPELLANT CHEMICAL ROCKETS

The common liquid rocket is bipropellant; it uses two separate propellants, a liquid fuel and liquid oxidizer. These are contained in separate tanks and are mixed only upon injection into the combustion chamber. They may be fed to the combustion chamber by pumps or by pressure in the tanks (fig 2).

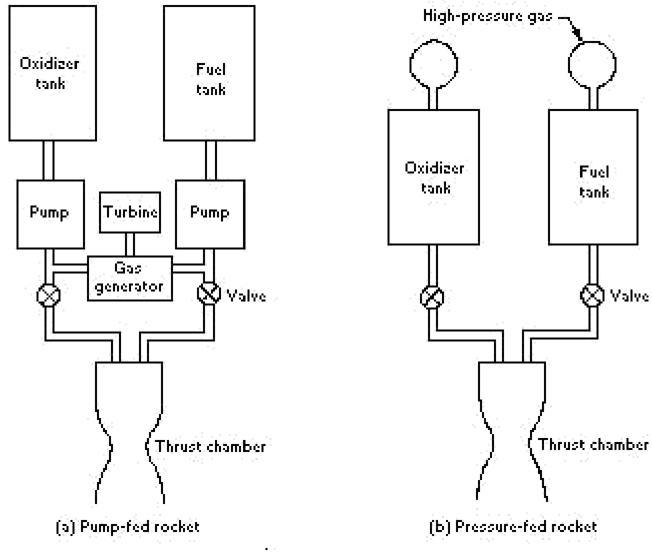


Fig. 2-Schematic of liquid-propellant rocket

Propellant flow rates must be extremely large for high-thrust engines, often hundreds of gallons per second. Pump-fed systems may require engines delivering several thousand horsepower to drive the pumps.[4] This power is usually developed by a hot gas turbine, supplied from a gas generator which is actually a small combustion chamber. The main rocket propellants can be used for the gas generator.

Thrust

Thrust is the force that propels a rocket or spacecraft and is measured in pounds, kilograms or Newtons. Physically speaking, it is the result of pressure which is exerted on the wall of the combustion chamber.

Figure 1.1 shows a combustion chamber with an opening, the nozzle, through which gas can escape. The pressure distribution within the chamber is asymmetric; that is, inside the chamber the pressure varies little, but near the nozzle it decreases somewhat. The force due to gas pressure on the bottom of the chamber is not compensated for from the outside. The resultant force F due to the internal and external pressure difference, the thrust, is opposite to the direction of the gas jet. It pushes the chamber upwards.

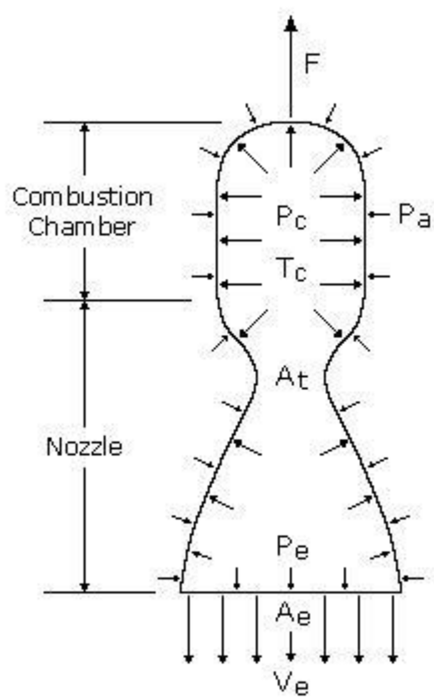


Figure 1.1

To create high speed exhaust gases, the necessary high temperatures and pressures of combustion are obtained by using a very energetic fuel and by having the molecular weight of the exhaust gases as low as possible. It is also necessary to reduce the pressure of the gas as much as possible inside the nozzle by creating a large section ratio. The section ratio, or expansion ratio, is defined as the area of the exit Aedivided by the area of the throat At.

The thrust F is the resultant of the forces due to the pressures exerted on the inner and outer walls by the combustion gases and the surrounding atmosphere, taking the boundary between the inner and outer surfaces as the cross section of the exit of the nozzle. As we shall see in the next section, applying the principle of the conservation of momentum gives

$$F = qV_e + (P_e - P_a)A_e$$

where q is the rate of the ejected mass flow, Pa the pressure of the ambient atmosphere, Pe the pressure of the exhaust gases and Ve their ejection speed. Thrust is specified either at sea level or in a vacuum.

Conservation of Momentum

The linear momentum (p), or simply momentum, of a particle is the product of its mass and its velocity. That is,

$$(1.1) \qquad p = mv$$

Newton expressed his second law of motion in terms of momentum, which can be stated as "the resultant of the forces acting on a particle is equal to the rate of change of the linear momentum of the particle". In symbolic form this becomes

$$(1.2) \qquad F = \frac{dp}{dt}$$

which is equivalent to the expression F=ma.

If we have a system of particles, the total momentum P of the system is the sum of the momenta of the individual particles. When the resultant external force acting on a system is zero, the total linear momentum of the system remains constant. This is called the principle of conservation of linear momentum. Let's now see how this principle is applied to rocket mechanics.

Consider a rocket drifting in gravity free space. The rocket's engine is fired for time Δt and, during this period, ejects gases at a constant rate and at a constant speed relative to the rocket (exhaust velocity). Assume there are no external forces, such as gravity or air resistance.

Figure 1.2(a) shows the situation at time t. The rocket and fuel have a total mass M and the combination is moving with velocity v as seen from a particular frame of reference. At a time Δ t later the configuration has changed to that shown in Figure 1.2(b). A mass ΔM has been ejected from the rocket and is moving with velocity u as seen by the observer. The rocket is reduced to mass M- ΔM and the velocity v of the rocket is changed to v+ Δv.

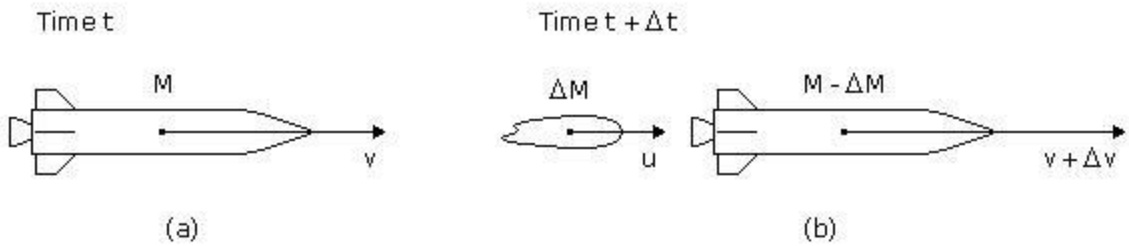


Figure 1.2

Because there are no external forces, dP/dt=0. We can write, for the time interval Δt

$$0 = \frac{\Delta P}{\Delta t} = \frac{(P_2 - P_1)}{\Delta t} \tag{1.3}$$

where P2 is the final system momentum, Figure 1.2(b), and P1 is the initial system momentum, Figure 1.2(a). We write

$$0 = \frac{[(M-\Delta M)(v+\Delta v)+\Delta M u]-Mv}{\Delta t} \tag{1.4}$$

If we let Δt approach zero, Δv/ Δt approaches dv/dt, the acceleration of the body. The quantity ΔM is the mass ejected in Δt; this leads to a decrease in the mass M of the original body. Since dM/dt, the change in mass of the body with time, is negative in this case, in the limit the quantity ΔM/ Δt is replaced by -dM/dt. The quantity u-(v+ Δv) is Vrel, the relative velocity of the ejected mass with respect to the rocket. With these changes, equation (1.4) can be written as

$$M\left(\frac{dv}{dt}\right) = (u-(v+\Delta v))\left(\frac{dM}{dt}\right), \quad \text{or}$$

$$M\left(\frac{dv}{dt}\right) = V_{rel}\left(\frac{dM}{dt}\right) \tag{1.5}$$

The right-hand term depends on the characteristics of the rocket and, like the left-hand term, has the dimensions of a force. This force is called the thrust, and is the reaction force exerted on the rocket by the mass that leaves it. The rocket designer can make the thrust as large as possible by designing the rocket to eject mass as rapidly as possible (dM/dtlarge) and with the highest possible relative speed (Vrel large).

In rocketry, the basic thrust equation is written as

$$(1.6) \qquad F = qV_e + (P_e - P_a)A_e$$

where q is the rate of the ejected mass flow, Ve is the exhaust gas ejection speed, Pe is the pressure of the exhaust gases at the nozzle exit, Pa is the pressure of the ambient atmosphere, and Ae is the area of the nozzle exit. The product qVe, which we derived above (Vrel × dM/dt), is called the momentum, or velocity, thrust. The product (Pe-Pa)Ae, called the pressure thrust, is the result of unbalanced pressure forces at the nozzle exit. As we shall see latter, maximum thrust occurs when Pe=Pa.

Equation (1.6) may be simplified by the definition of an effective exhaust gas velocity, C, defined as

$$(1.7) \qquad C = V_e + \frac{(P_e - P_a)A_e}{q}$$

Equation (1.6) then reduces to

$$(1.8) \qquad F = qC$$

Impulse & Momentum

In the preceding section we saw that Newton's second law may be expressed in the form

$$F = \frac{dp}{dt}$$

Multiplying both sides by dt and integrating from a time t1 to a time t2, we write

$$Fdt = dp$$

$$\int Fdt = p_2 - p_1, \quad \text{or}$$

$$(1.9) \qquad p_1 + \int Fdt = p_2$$

The integral is a vector known as the linear impulse, or simply the impulse, of the force F during the time interval considered. The equation expresses that, when a particle is acted upon by a force F during a given time interval, the final momentum p2 of the particle may be obtained by adding its initial momentum p1 and the impulse of the force F during the interval of time.

When several forces act on a particle, the impulse of each of the forces must be considered. When a problem involves a system of particles, we may add vectorially the momenta of all the particles and the impulses of all the forces involved. When can then write

$$(1.10) \qquad P_1 + \Sigma \int F dt = P_2$$

For a time interval Δt, we may write equation (1.10) in the form

$$(1.11) \qquad P_1 + \Sigma(F\Delta t) = P_2$$

Let us now see how we can apply the principle of impulse and momentum to rocket mechanics.

Consider a rocket of initial mass M which it launched vertically at time t=0. The fuel is consumed at a constant rate q and is expelled at a constant speed Ve relative to the rocket. At time t, the mass of the rocket shell and remaining fuel is M-qt, and the velocity is v. During the time interval Δt, a mass of fuel q Δt is expelled. Denoting by u the absolute velocity of the expelled fuel, we apply the principle of impulse and momentum between time t and time t+ Δt. Please note, this derivation neglects the effect of air resistance.

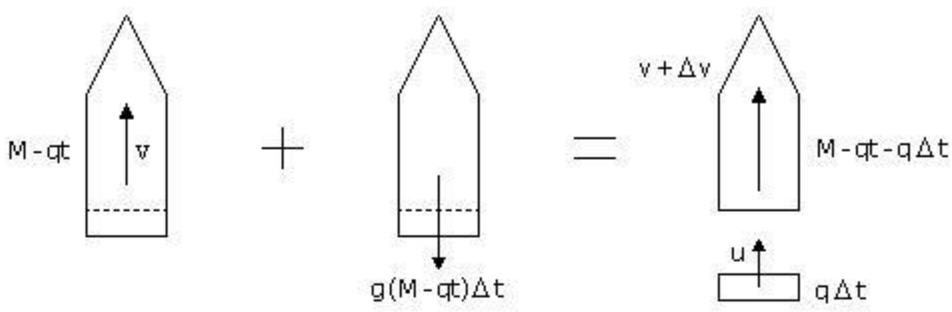


Figure 1.3

We write

$$(1.12) \qquad (M-qt)v - g(M-qt)\Delta t = (M-qt-q\Delta t)(v+\Delta v) + q\Delta t u$$

We divide through by Δt and replace u-(v+ Δv) with Ve, the velocity of the expelled mass relative to the rocket. As Δt approaches zero, we obtain

$$(1.13) \qquad -g(M-qt) = (M-qt)\left(\frac{dv}{dt}\right) - qV_e$$

Separating variables and integrating from t=0, v=0 to t=t, v=v, we obtain

$$(1.14) \qquad \int dv = \int \left(\frac{qV_e}{M-qt} - g \right) dt$$

which equals

$$(1.15) \qquad v = V_e \, LN \left(\frac{M}{M-qt} \right) - gt$$

The term -gt in equation (1.15) is the result of Earth's gravity pulling on the rocket. For a rocket drifting in space, -gt is not applicable and can be omitted. Furthermore, it is more appropriate to express the resulting velocity as a change in velocity, or ΔV. Equation (1.15) thus becomes

$$(1.16) \qquad \Delta V = V_e \, LN \left(\frac{M}{M-qt} \right)$$

Note that M represents the initial mass of the rocket and M-qt the final mass. Therefore, equation (1.16) is often written as

$$(1.17) \qquad \Delta V = V_e \, LN \left(\frac{m_o}{m_f} \right)$$

where mo/mf is called the mass ratio. Equation (1.17) is also known as Tsiolkovsky's rocket equation, named after Russian rocket pioneer Konstantin E. Tsiolkovsky (1857-1935) who first derived it.

In practical application, the variable Ve is usually replaced by the effective exhaust gas velocity, C. Equation (1.17) therefore becomes

$$(1.18) \qquad \Delta V = C \, LN \left(\frac{m_o}{m_f} \right)$$

Alternatively, we can write

$$(1.19) \qquad m_f = m_o e^{-(\Delta V/C)}$$

$$(1.20) \qquad m_o = m_f e^{(\Delta V/C)}$$

where e is a mathematical constant approximately equal to 2.71828.

For many spacecraft maneuvers it is necessary to calculate the duration of an engine burn required to achieve a specific change in velocity. Rearranging variables, we have

(1.21) $$t = \frac{m_o}{q}\left[1 - \frac{1}{e^{(\Delta V/c)}}\right]$$

Combustion & Exhaust Velocity

The combustion process involves the oxidation of constituents in the fuel that are capable of being oxidized, and can therefore be represented by a chemical equation. During a combustion process the mass of each element remains the same. Consider the reaction of methane with oxygen

$$CH_4 + 2O_2 \rightarrow CO_2 + 2H_2O$$

This equation states that one mole of methane reacts with two moles of oxygen to form one mole of carbon dioxide and two moles of water. This also means that 16 g of methane react with 64 g of oxygen to form 44 g of carbon dioxide and 36 g of water. All the initial substances that undergo the combustion process are called the reactants, and the substances that result from the combustion process are called the products.

The above combustion reaction is an example of a stoichiometric mixture, that is, there is just enough oxygen present to chemically react with all the fuel. The highest flame temperature is achieved under these conditions, however it is often desirable to operate a rocket engine at a "fuel-rich" mixture ratio. Mixture ratio is defined as the mass flow of oxidizer divided by the mass flow of fuel.

Consider the following reaction of kerosene(1) with oxygen,

$$C_{12}H_{26} + 12.5O_2 \rightarrow 12CO + 13H_2O$$

Given the molecular weight of C12H26 is 170 and that of O2 is 32, we have a mixture ratio of

$$O/F = (12.5 \times 32) / 170 = 2.35$$

which is typical of many rocket engines using kerosene, or RP-1, fuel.

The optimum mixture ratio is typically that which will deliver the highest engine performance (measured by specific impulse), however in some situations a different O/F ratio results in a better overall system. For a volume-constrained vehicle with a low-density fuel such as liquid hydrogen, significant reductions in vehicle size can be achieved by shifting to a higher O/F ratio. In that case, the losses in performance are more than compensated for by the reduced fuel tankage requirement. Also consider the example of bipropellant systems using NTO/MMH, where a mixture ratio of 1.67 results in fuel and oxidizer tanks of equal size. Equal sizing simplifies tank manufacturing, system packaging, and integration.

As we have seen previously, impulse thrust is equal to the product of the propellant mass flow rate and the exhaust gas ejection speed. The ideal exhaust velocity is given by

$$V_e = \sqrt{\left(\frac{2k}{k-1}\right)\left(\frac{R'T_c}{M}\right)\left(1-\left(\frac{P_e}{P_c}\right)^{(k-1)/k}\right)} \qquad (1.22)$$

where k is the specific heat ratio, R' is the universal gas constant (8,314.51 N-m/kmol-K in SI units, or 49,720 ft-lb/(slug-mol)-oR in U.S. units), Tc is the combustion temperature, M is the average molecular weight of the exhaust gases, Pc is the combustion chamber pressure, and Pe is the pressure at the nozzle exit.

Specific heat ratio(2) varies depending on the composition and temperature of the exhaust gases, but it is usually about 1.2. The thermodynamics involved in calculating combustion temperatures are quite complicated, however, flame temperatures generally range from about 2,500 to 3,600 oC (4,500-6,500 oF). Chamber pressures can range from about about 7 to 250 atmospheres. Pe should be equal to the ambient pressure at which the engine will operate, more on this later. Click Here for charts providing optimum mixture ratio, adiabatic flame temperature, gas molecular weight, and specific heat ratio for some common rocket propellants.

From equation (1.22) we see that high chamber temperature and pressure, and low exhaust gas molecular weight results in high ejection velocity, thus high thrust. Based on this criterion, we can see why liquid hydrogen is very desirable as a rocket fuel.

It should be pointed out that in the combustion process there will be a dissociation of molecules among the products. That is, the high heat of combustion causes the separation of molecules into simpler constituents that are then capable of recombining. Consider the reaction of kerosene with oxygen. The true products of combustion will be an equilibrium mixture of atoms and molecules consisting of C, CO, CO2, H, H2, H2O, HO, O, and O2. Dissociation has a significant effect on flame temperature.

(1) In dealing with combustion of liquid hydrocarbon fuels it is convenient to express the composition in terms of a single hydrocarbon, even though it is a mixture of many hydrocarbons. Thus gasoline is usually considered to be octane, C8H18, and kerosene is

considered to be dodecane, C12H26.

(2) Specific heat, or heat capacity, represents the amount of heat necessary to raise the temperature of one gram of a substance one degree C. Specific heat is measured at constant-pressure, CP, or at constant-volume, CV. The ratio CP/CV is called the specific heat ratio, represented by k.

Specific Impulse

The specific impulse of a rocket, Isp, is the ratio of the thrust to the flow rate of the weight ejected, that is

$$(1.23) \qquad I_{sp} = \frac{F}{q g_o}$$

where F is thrust, q is the rate of mass flow, and go is standard gravity (9.80665 m/s2).

Specific impulse is expressed in seconds. When the thrust and the flow rate remain constant throughout the burning of the propellant, the specific impulse is the time for which the rocket engine provides a thrust equal to the weight of the propellant consumed.

For a given engine, the specific impulse has different values on the ground and in the vacuum of space because the ambient pressure is involved in the expression for the thrust. It is therefore important to state whether specific impulse is the value at sea level or in a vacuum.

There are a number of losses within a rocket engine, the main ones being related to the inefficiency of the chemical reaction (combustion) process, losses due to the nozzle, and losses due to the pumps. Overall, the losses affect the efficiency of the specific impulse. This is the ratio of the real specific impulse (at sea level, or in a vacuum) and the theoretical specific impulse obtained with an ideal nozzle from gases coming from a complete chemical reaction. Calculated values of specific impulse are several percent higher than those attained in practice.

From Equation (1.8) we can substitute qC for F in Equation (1.23), thus obtaining

$$I_{sp} = \frac{C}{g_o}, \quad \text{or}$$

$$(1.24) \qquad C = I_{sp} g_o$$

Equation (1.24) is very useful when solving Equations (1.18) through (1.21). It is rare we are given the value of C directly, however rocket engine specific impulse is a commonly given parameter from which we can easily calculate C.

Another important figure of merit for evaluating rocket performance is the characteristic exhaust velocity, C* (pronounced "C star"), which is a measure of the energy available from the combustion process and is given by

$$C^* = \frac{P_c A_t}{q} \qquad (1.25)$$

where Pc is the combustion chamber pressure and At is the area of the nozzle throat. Delivered values of C* range from about 1,333 m/s for monopropellant hydrazine up to about 2,360 m/s for cryogenic oxygen/hydrogen.

Rocket Engines

A typical rocket engine consists of the nozzle, the combustion chamber, and the injector, as shown in Figure 1.4. The combustion chamber is where the burning of propellants takes place at high pressure. The chamber must be strong enough to contain the high pressure generated by, and the high temperature resulting from, the combustion process.

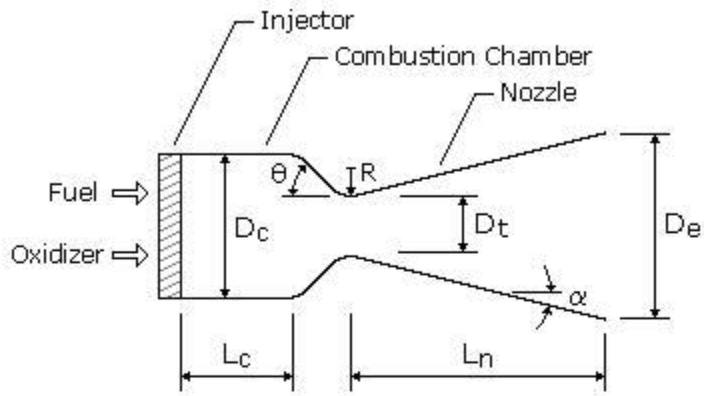


Figure 1.4

Because of the high temperature and heat transfer, the chamber and nozzle are usually cooled. The chamber must also be of sufficient length to ensure complete combustion before the gases enter the nozzle.

Nozzle

The function of the nozzle is to convert the chemical-thermal energy generated in the combustion chamber into kinetic energy. The nozzle converts the slow moving, high pressure, high temperature gas in the combustion chamber into high velocity gas of lower pressure and temperature. Since thrust is the product of mass and velocity, a very high gas velocity is desirable. Nozzles consist of a convergent and divergent section. The minimum flow area between the convergent and divergent section is called the nozzle throat. The flow area at the end of the divergent section is called the nozzle exit area. The nozzle is usually made long enough (or the exit area is great enough) such that the pressure in the combustion chamber is reduced at the nozzle exit to the pressure existing outside the nozzle. It is under this condition, Pe=Pa where Pe is the pressure at the nozzle exit and Pa is the outside ambient pressure, that thrust is maximum and the nozzle is said to be adapted, also called optimum or correct expansion. When Pe is greater than Pa, the nozzle is under-extended. When the opposite is true, it is over-extended.

We see therefore, a nozzle is designed for the altitude at which it has to operate. At the Earth's surface, at the atmospheric pressure of sea level (0.1 MPa or 14.7 psi), the discharge of the exhaust gases is limited by the separation of the jet from the nozzle wall. In the cosmic vacuum, this physical limitation does not exist. Therefore, there have to be two different types of engines and nozzles, those which propel the first stage of the launch vehicle through the atmosphere, and those which propel subsequent stages or control the orientation of the spacecraft in the vacuum of space.

The nozzle throat area, At, can be found if the total propellant flow rate is known and the propellants and operating conditions have been selected. Assuming perfect gas law theory, we have

$$(1.26) \qquad A_t = \frac{q}{P_t}\sqrt{\frac{R'T_t}{Mk}}$$

where q is the propellant mass flow rate, Pt is the gas pressure at the nozzle throat, Tt is the gas temperature at the nozzle throat, R' is the universal gas constant, and k is the specific heat ratio. Pt and Tt are given by

$$(1.27) \qquad P_t = P_c\left(1+\frac{k-1}{2}\right)^{-k/(k-1)}$$

$$(1.28) \qquad T_t = \frac{T_c}{\left(1+\frac{k-1}{2}\right)}$$

where Pc is the combustion chamber pressure and Tc is the combustion chamber flame temperature.

The hot gases must be expanded in the diverging section of the nozzle to obtain maximum thrust. The pressure of these gases will decrease as energy is used to accelerate the gas. We must find that area of the nozzle where the gas pressure is equal to the outside atmospheric pressure. This area will then be the nozzle exit area.

Mach number Nm is the ratio of the gas velocity to the local speed of sound. The Mach number at the nozzle exit is given by the perfect gas expansion expression

$$N_m^2 = \left(\frac{2}{k-1}\right)\left[\left(\frac{P_c}{P_a}\right)^{(k-1)/k} - 1\right] \quad (1.29)$$

where Pa is the pressure of the ambient atmosphere.

The nozzle exit area, Ae, corresponding to the exit Mach number is given by

$$A_e = \left(\frac{A_t}{N_m}\right)\left[\frac{1+\left(\frac{k-1}{2}\right)N_m^2}{(k+1)/2}\right]^{\left(\frac{k+1}{2(k-1)}\right)} \quad (1.30)$$

The section ratio, or expansion ratio, is defined as the area of the exit Ae divided by the area of the throat At.

For launch vehicles (particularly first stages) where the ambient pressure varies during the burn period, trajectory computations are performed to determine the optimum exit pressure. However, an additional constraint is the maximum allowable diameter for the nozzle exit cone, which in some cases is the limiting constraint. This is especially true on stages other than the first, where the nozzle diameter may not be larger than the outer diameter of the stage below. For space engines, where the ambient pressure is zero, thrust always increases as nozzle expansion ratio increases. On these engines, the nozzle expansion ratio is generally increased until the additional weight of the longer nozzle costs more performance than the extra thrust it generates.

(For additional information, please see Supplement #1: Optimizing Expansion for Maximum Thrust.)

Since the flow velocity of the gases in the converging section of the rocket nozzle is relatively low, any smooth and well-rounded convergent nozzle section will have very low energy loses. By contrast, the contour of the diverging nozzle section is very important to performance, because of the very high flow velocities involved. The selection of an optimum nozzle shape for a given expansion ratio is generally influenced by the following design considerations and goals: (1) uniform, parallel, axial gas flow at the nozzle exit for maximum momentum vector, (2) minimum separation and turbulence losses within the nozzle, (3) shortest possible nozzle length for minimum space envelope, weight, wall friction losses, and cooling requirements, and (4) ease of manufacturing.

Conical nozzle: In early rocket engine applications, the conical nozzle, which proved satisfactory in most respects, was used almost exclusively. A conical nozzle allows ease of manufacture and flexibility in converting an existing design to higher or lower expansion ratio without major redesign.

The configuration of a typical conical nozzle is shown in Figure 1.4. The nozzle throat section has the contour of a circular arc with radius R, ranging from 0.25 to 0.75 times the throat diameter, Dt. The half-angle of the nozzle convergent cone section, θ, can range from 20 to 45 degrees. The divergent cone half-angle, α, varies from approximately 12 to 18 degrees. The conical nozzle with a 15-degree divergent half-angle has become almost a standard because it is a good compromise on the basis of weight, length, and performance.

Since certain performance losses occur in a conical nozzle as a result of the nonaxial component of the exhaust gas velocity, a correction factor, λ, is applied in the calculation of the exit-gas momentum. This factor (thrust efficiency) is the ratio between the exit-gas momentum of the conical nozzle and that of an ideal nozzle with uniform, parallel, axial gas-flow. The value of λ can be expressed by the following equation:

$$\lambda = \frac{1+\cos\alpha}{2} \qquad (1.31)$$

Bell nozzle: To gain higher performance and shorter length, engineers developed the bell-shaped nozzle. It employs a fast-expansion (radial-flow) section in the initial divergent region, which leads to a uniform, axially directed flow at the nozzle exit. The wall contour is changed gradually enough to prevent oblique shocks.

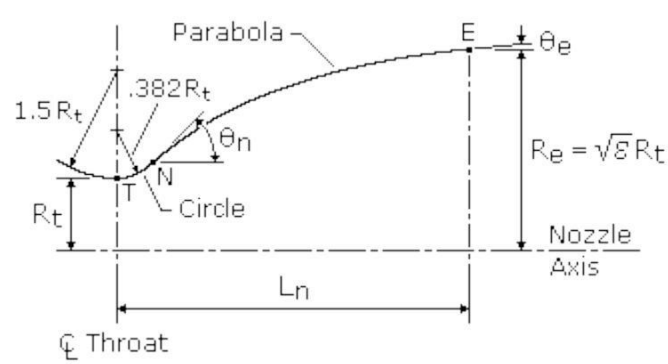


Figure 1.5

An equivalent 15-degree half-angle conical nozzle is commonly used as a standard to specify bell nozzles. For instance, the length of an 80% bell nozzle (distance between throat and exit plane) is 80% of that of a 15-degree half-angle conical nozzle having the same throat area, radius below the throat, and area expansion ratio. Bell nozzle lengths beyond approximately 80% do not significantly contribute to performance, especially when weight penalties are considered. However, bell nozzle lengths up to 100% can be optimum for applications stressing very high performance.

One convenient way of designing a near optimum thrust bell nozzle contour uses the parabolic approximation procedures suggested by G.V.R. Rao. The design configuration of a parabolic approximation bell nozzle is shown in Figure 1.5. The nozzle contour immediately upstream of the throat T is a circular arc with a radius of 1.5 Rt. The divergent section nozzle contour is made up of a circular entrance section with a radius of 0.382 Rt from the throat T to the point N and parabola from there to the exit E.

Design of a specific nozzle requires the following data: throat diameter Dt, axial length of the nozzle from throat to exit plane Ln (or the desired fractional length, Lf, based on a 15-degree conical nozzle), expansion ratio ε, initial wall angle of the parabola θn, and nozzle exit wall angle θe. The wall angles θn and θe are shown in Figure 1.6 as a function of the expansion ratio. Optimum nozzle contours can be approximated very accurately by selecting the proper inputs. Although no allowance is made for different propellant combinations, experience has shown only small effect of the specific heat ratio upon the contour.

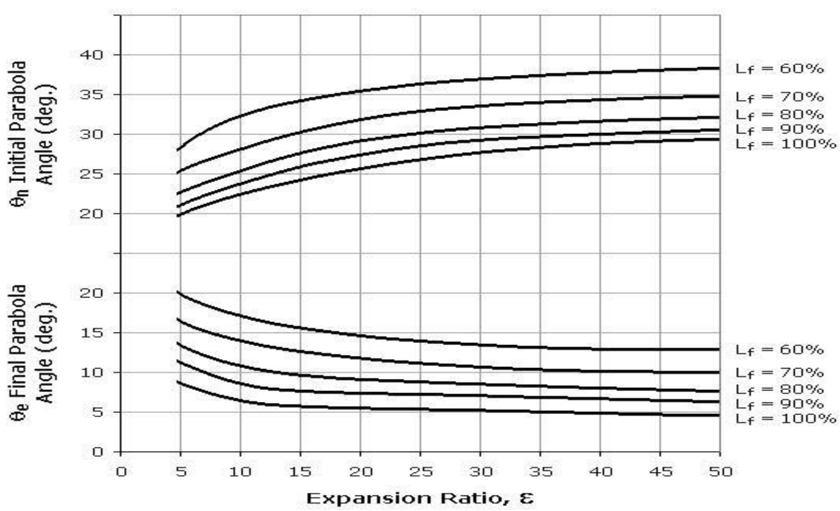


Figure 1.6

Combustion Chamber

The combustion chamber serves as an envelope to retain the propellants for a sufficient period to ensure complete mixing and combustion. The required stay time, or combustion residence time, is a function of many parameters. The theoretically required combustion chamber volume is a function of the mass flow rate of the propellants, the average density of the combustion products, and the stay time needed for efficient combustion. This relationship can be expressed by the following equation:

$$V_c = qVt_s \quad (1.32)$$

where Vc is the chamber volume, q is the propellant mass flow rate, V is the average specific volume, and ts is the propellant stay-time.

A useful parameter relative to chamber volume and residence time is the characteristic length, L* (pronounced "L star"), the chamber volume divided by the nozzle sonic throat area:

$$L^* = \frac{V_c}{A_t} \quad (1.33)$$

The L* concept is much easier to visualize than the more elusive "combustion residence time", expressed in small fractions of a second. Since the value of At is in nearly direct proportion to the product of q and V, L* is essentially a function of ts.

The customary method of establishing the L* of a new thrust chamber design largely relies on past experience with similar propellants and engine size. Under a given set of operating conditions, such as type of propellant, mixture ratio, chamber pressure, injector design, and chamber geometry, the value of the minimum required L* can only be evaluated by actual firings of experimental thrust chambers. Typical L* values for various propellants are shown in the table below. With throat area and minimum required L* established, the chamber volume can be calculated by equation (1.33).

Table 1: Chamber Characteristic Length, L*

Propellant Combination	L*, cm
Nitric acid/hydrazine-base fuel	76-89
Nitrogen peroxide/hydrazine-base fuel	76-89

Hydrogen peroxide/RP-1 (including catalyst bed)	152-178
Liquid oxygen/RP-1	102-127
Liquid oxygen/ammonia	76-102
Liquid oxygen/liquid hydrogen (GH2 injection)	56-71
Liquid oxygen/liquid hydrogen (LH2 injection)	76-102
Liquid fluorine/liquid hydrogen (GH2 injection)	56-66
Liquid fluorine/liquid hydrogen (LH2 injection)	64-76
Liquid fluorine/hydrazine	61-71
Chlorine trifluoride/hydrazine-base fuel	51-89

Three geometrical shapes have been used in combustion chamber design - spherical, near-spherical, and cylindrical - with the cylindrical chamber being employed most frequently in the United States. Compared to a cylindrical chamber of the same volume, a spherical or near-spherical chamber offers the advantage of less cooling surface and weight; however, the spherical chamber is more difficult to manufacture and has provided poorer performance in other respects.

The total combustion process, from injection of the reactants until completion of the chemical reactions and conversion of the products into hot gases, requires finite amounts of time and volume, as expressed by the characteristic length L*. The value of this factor is significantly greater than the linear length between injector face and throat plane. Thecontraction ratio is defined as the major cross-sectional area of the combuster divided by the throat area. Typically, large engines are constructed with a low contraction ratio and a comparatively long length; and smaller chambers employ a large contraction ratio with a shorter length, while still providing sufficient L* for adequate vaporization and combustion dwell-time.

As a good place to start, the process of sizing a new combustion chamber examines the dimensions of previously successful designs in the same size class and plotting such data in a rational manner. The throat size of a new engine can be generated with a fair degree of confidence, so it makes sense to plot the data from historical sources in relation to throat diameter. Figure 1.7 plots chamber length as a function of throat diameter (with approximating equation). It is important that the output of any modeling program not be slavishly applied, but be considered a logical starting point for specific engine sizing.

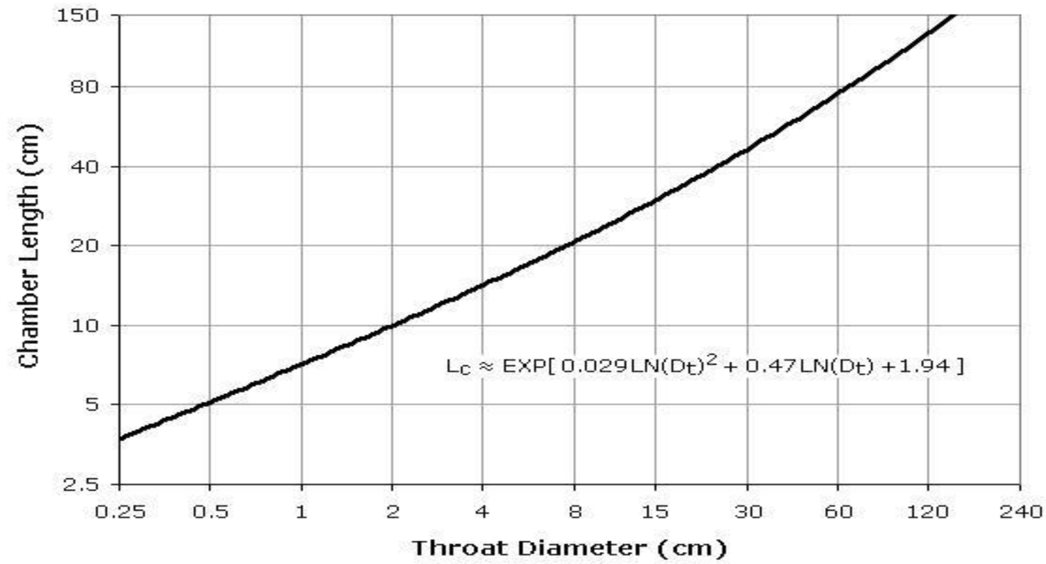


Figure 1.7

The basic elements of a cylindrical thrust-chamber are identified in Figure 1.4. In design practice, it has been arbitrarily defined that the combustion chamber volume includes the space between the injector face and the nozzle throat plane. The approximate volume of the combustion chamber can be expressed by the following equation:

$$(1.34) \qquad V_c = \frac{\pi}{24}\left[6 L_c D_c^2 + \frac{D_c^3 - D_t^3}{\tan\theta}\right]$$

Rearranging equation (1.34) we get the following, which can be solved for the chamber diameter via iteration:

$$(1.35) \qquad D_c = \sqrt{\frac{D_t^3 + \frac{24}{\pi}\tan\theta\, V_c}{D_c + 6\tan\theta\, L_c}}$$

Injector

The injector, as the name implies, injects the propellants into the combustion chamber in the right proportions and the right conditions to yield an efficient, stable combustion process. Placed at the forward, or upper, end of the combustor, the injector also performs the structural task of closing off the top of the combustion chamber against the high pressure and temperature it contains. The injector has been compared to the carburetor of an automobile engine, since it provides the fuel and oxidizer at the proper rates and in the correct proportions, this may be an appropriate comparison. However, the injector, located directly over the high-pressure combustion, performs many other functions related to the combustion and cooling processes and is much more important to the function of the rocket engine than the carburetor is for an automobile engine.

No other component of a rocket engine has as great an impact upon engine performance as the injector. In various and different applications, well-designed injectors may have a fairly wide spread in combustion efficiency, and it is not uncommon for an injector with C* efficiency as low as 92% to be considered acceptable. Small engines designed for special purposes, such as attitude control, may be optimized for response and light weight at the expense of combustion efficiency, and may be deemed very satisfactory even if efficiency falls below 90%. In general, however, recently well-designed injection systems have demonstrated C* efficiencies so close to 100% of theoretical that the ability to measure this parameter is the limiting factor in its determination. High levels of combustion efficiency derive from uniform distribution of the desired mixture ratio and fine atomization of the liquid propellants. Local mixing within the injection-element spray pattern must take place at virtually a microscopic level to ensure combustion efficiencies approaching 100%.

Combustion stability is also a very important requirement for a satisfactory injector design. Under certain conditions, shock and detonation waves are generated by local disturbances in the chamber, possibly caused by fluctuations in mixing or propellant flow. These may trigger pressure oscillations that are amplified and maintained by the combustion processes. Such high-amplitude waves - referred to as combustion instability - produce high levels of vibration and heat flux that can be very destructive. A major portion of the design and development effort therefore concerns stable combustion. High performance can become secondary if the injector is easily triggered into destructive instability, and many of the injector parameters that provide high performance appear to reduce the stability margin.

Power Cycles

Liquid bipropellant rocket engines can be categorized according to their power cycles, that is, how power is derived to feed propellants to the main combustion chamber. Described below are some of the more common types.

Gas-generator cycle: The gas-generator cycle, also called open cycle, taps off a small amount of fuel and oxidizer from the main flow (typically 3 to 7 percent) to feed a burner called a gas generator. The hot gas from this generator passes through a turbine to generate power for the pumps that send propellants to the combustion chamber. The hot gas is then either dumped overboard or sent into the main nozzle downstream. Increasing the flow of propellants into the gas generator increases the speed of the turbine, which increases the flow of propellants into the main combustion chamber, and hence, the amount of thrust produced. The gas generator must burn propellants at a less-than-optimal mixture ratio to keep the temperature low for the turbine blades. Thus, the cycle is appropriate for moderate power requirements but not high-power systems, which would have to divert a large portion of the main flow to the less efficient gas-generator flow.

As in most rocket engines, some of the propellant in a gas generator cycle is used to cool the nozzle and combustion chamber, increasing efficiency and allowing higher engine temperature.

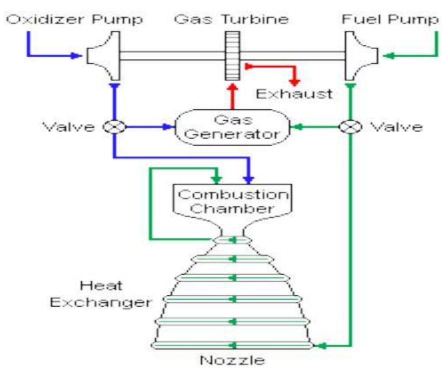


Fig. 1.8 - GAS GENERATOR

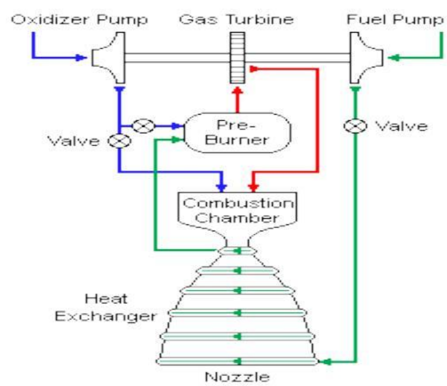


Fig. 1.9 - STAGED COMBUSTION

Staged combustion cycle: In a staged combustion cycle, also called closed cycle, the propellants are burned in stages. Like the gas-generator cycle, this cycle also has a burner, called a preburner, to generate gas for a turbine. The preburner taps off and burns a small amount of one propellant and a large amount of the other, producing an oxidizer-rich or fuel-rich hot gas mixture that is mostly unburned vaporized propellant. This hot gas is then passed through the turbine, injected into the main chamber, and burned again with the remaining propellants. The advantage over the gas-generator cycle is that all of the propellants are burned at the optimal mixture ratio in the main chamber and no flow is dumped overboard. The staged combustion cycle is often used for high-power applications. The higher the chamber pressure, the smaller and lighter the engine can be to produce the same thrust. Development cost for this cycle is higher because the high pressures complicate the development process. Further disadvantages are harsh turbine conditions, high temperature piping required to carry hot gases, and a very complicated feedback and control design.

Staged combustion was invented by Soviet engineers and first appeared in 1960. In the West, the first laboratory staged combustion test engine was built in Germany in 1963.

Expander cycle: The expander cycle is similar to the staged combustion cycle but has no preburner. Heat in the cooling jacket of the main combustion chamber serves to vaporize the fuel. The fuel vapor is then passed through the turbine and injected into the main chamber to burn with the oxidizer. This cycle works with fuels such as hydrogen or methane, which have a low boiling point and can be vaporized easily. As with the staged combustion cycle, all of the propellants are burned at the optimal mixture ratio in the main chamber, and typically no flow is dumped overboard; however, the heat transfer to the fuel limits the power available to the turbine, making this cycle appropriate for small to midsize engines. A variation of the system is the open, or bleed, expander cycle, which uses only a portion of the fuel to drive the turbine. In this variation, the turbine exhaust is dumped overboard to ambient pressure to increase the turbine pressure ratio and power output. This can achieve higher chamber pressures than the closed expander cycle although at lower efficiency because of the overboard flow.

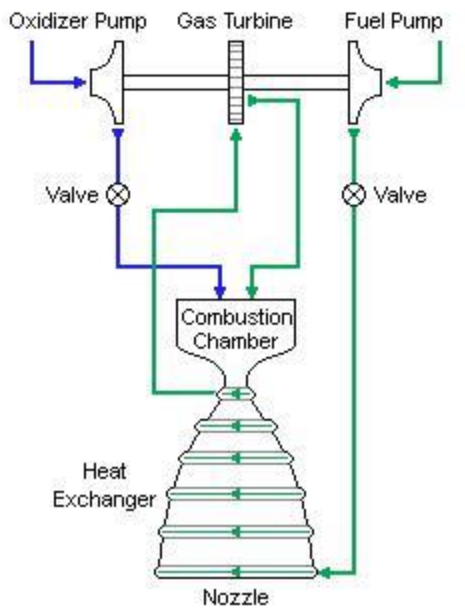


Fig. 1.10 - EXPANDER

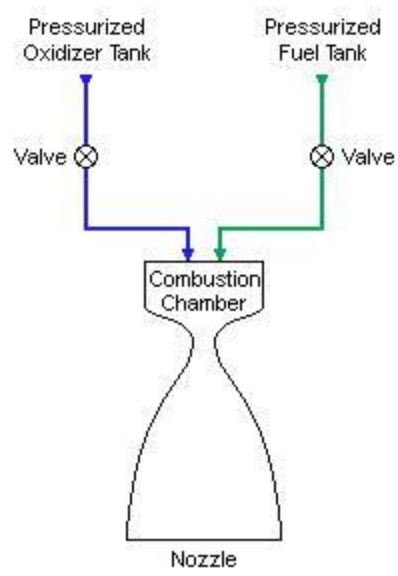


Fig. 1.11 - PRESSURE-FED

Pressure-fed cycle: The simplest system, the pressure-fed cycle, does not have pumps or turbines but instead relies on tank pressure to feed the propellants into the main chamber. In practice, the cycle is limited to relatively low chamber pressures because higher pressures make the vehicle tanks too heavy. The cycle can be reliable, given its reduced part count and complexity compared with other systems.

Engine Cooling

The heat created during combustion in a rocket engine is contained within the exhaust gases. Most of this heat is expelled along with the gas that contains it; however, heat is transferred to the thrust chamber walls in quantities sufficient to require attention.

Thrust chamber designs are generally categorized or identified by the hot gas wall cooling method or the configuration of the coolant passages, where the coolant pressure inside may be as high as 500 atmospheres. The high combustion temperatures (2,500 to 3,600o K) and the high heat transfer rates (up to 16 kJ/cm2-s) encountered in a combustion chamber present a formidable challenge to the designer. To meet this challenge, several chamber cooling techniques have been utilized successfully. Selection of the optimum cooling method for a thrust chamber depends on many considerations, such as type of propellant, chamber pressure, available coolant pressure, combustion chamber configuration, and combustion chamber material.

Regenerative cooling is the most widely used method of cooling a thrust chamber and is accomplished by flowing high-velocity coolant over the back side of the chamber hot gas wall to convectively cool the hot gas liner. The coolant with the heat input from cooling the liner is then discharged into the injector and utilized as a propellant.

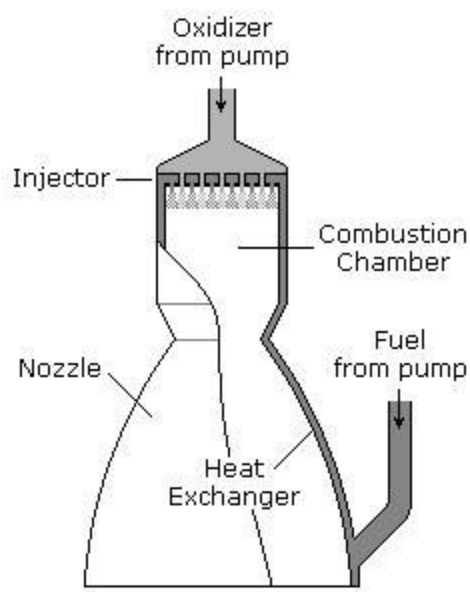


Fig. 1.12 - REGENERATIVE COOLING

Earlier thrust chamber designs, such as the V-2 and Redstone, had low chamber pressure, low heat flux and low coolant pressure requirements, which could be satisfied by a simplified "double wall chamber" design with regenerative and film cooling. For subsequent rocket engine applications, however, chamber pressures were increased and the cooling requirements became more difficult to satisfy. It became necessary to design new coolant configurations that were more efficient structurally and had improved heat transfer characteristics.

This led to the design of "tubular wall" thrust chambers, by far the most widely used design approach for the vast majority of large rocket engine applications. These chamber designs have been successfully used for the Thor, Jupiter, Atlas, H-1, J-2, F-1, RS-27 and several other Air Force and NASA rocket engine applications. The primary advantage of the design is its light weight and the large experience base that has accrued. But as chamber pressures and hot gas wall heat fluxes have continued to increase (>100 atm), still more effective methods have been needed.

One solution has been "channel wall" thrust chambers, so named because the hot gas wall cooling is accomplished by flowing coolant through rectangular channels, which are machined or formed into a hot gas liner fabricated from a high-conductivity material, such as copper or a copper alloy. A prime example of a channel wall combustion chamber is the SSME, which operates at 204 atmospheres nominal chamber pressure at 3,600 K for a duration of 520 seconds. Heat transfer and structural characteristics are excellent.

In addition to the regenerative cooled designs mentioned above, other thrust chamber designs have been fabricated for rocket engines using dump cooling, film cooling, transpiration cooling, ablative liners and radiation cooling. Although regenerative cooled combustion chambers have proven to be the best approach for cooling large liquid rocket engines, other methods of cooling have also been successfully used for cooling thrust chamber assemblies. Examples include:

Dump cooling, which is similar to regenerative cooling because the coolant flows through small passages over the back side of the thrust chamber wall. The difference, however, is that after cooling the thrust chamber, the coolant is discharged overboard through openings at the aft end of the divergent nozzle. This method has limited application because of the performance loss resulting from dumping the coolant overboard. To date, dump cooling has not been used in an actual application.

Film cooling provides protection from excessive heat by introducing a thin film of coolant or propellant through orifices around the injector periphery or through manifold orifices in the chamber wall near the injector or chamber throat region. This method is typically used in high heat flux regions and in combination with regenerative cooling.

Transpiration cooling provides coolant (either gaseous or liquid propellant) through a porous chamber wall at a rate sufficient to maintain the chamber hot gas wall to the desired temperature. The technique is really a special case of film cooling.

With ablative cooling, combustion gas-side wall material is sacrificed by melting, vaporization and chemical changes to dissipate heat. As a result, relatively cool gases flow over the wall surface, thus lowering the boundary-layer temperature and assisting the cooling process.

With radiation cooling, heat is radiated from the outer surface of the combustion chamber or nozzle extension wall. Radiation cooling is typically used for small thrust chambers with a high-temperature wall material (refractory) and in low-heat flux regions, such as a nozzle extension.

Solid Rocket Motors

Solid rockets motors store propellants in solid form. The fuel is typically powdered aluminum and the oxidizer is ammonium per chlorate. A synthetic rubber binder such as polybutadiene holds the fuel and oxidizer powders together. Though lower performing than liquid propellant rockets, the operational simplicity of a solid rocket motor often makes it the propulsion system of choice.

Solid Fuel Geometry

A solid fuel's geometry determines the area and contours of its exposed surfaces, and thus its burn pattern. There are two main types of solid fuel blocks used in the space industry. These are cylindrical blocks, with combustion at a front, or surface, and cylindrical blocks with internal combustion. In the first case, the front of the flame travels in layers from the nozzle end of the block towards the top of the casing. This so-called end burner produces constant thrust throughout the burn. In the second, more usual case, the combustion surface develops along the length of a central channel. Sometimes the channel has a star shaped, or other, geometry to moderate the growth of this surface.

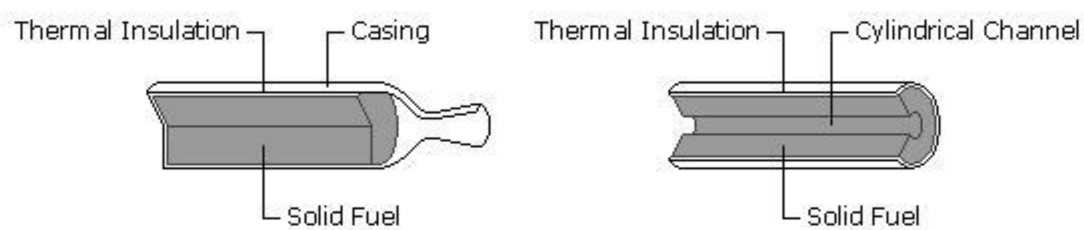


Figure 1.13

The shape of the fuel block for a rocket is chosen for the particular type of mission it will perform. Since the combustion of the block progresses from its free surface, as this surface grows, geometrical considerations determine whether the thrust increases, decreases or stays constant.

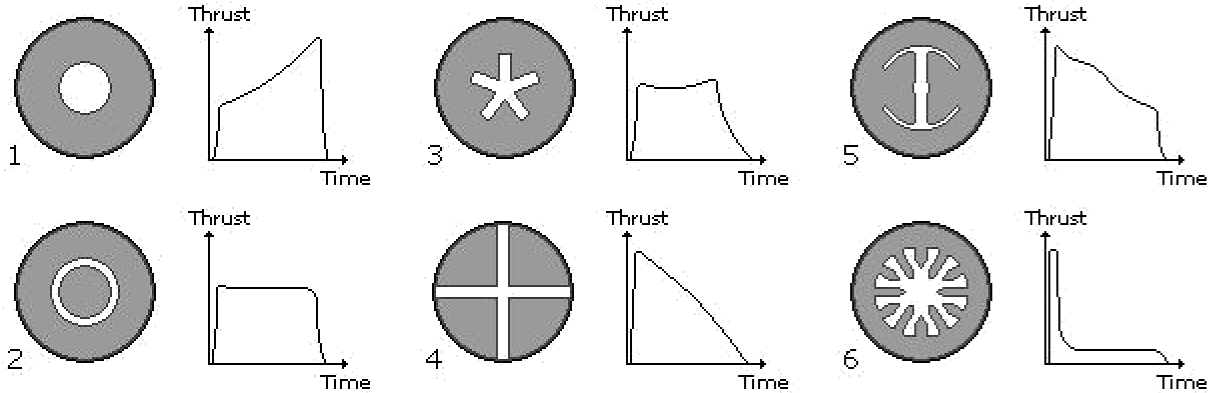


Figure 1.14

Fuel blocks with a cylindrical channel (1) develop their thrust progressively. Those with a channel and also a central cylinder of fuel (2) produce a relatively constant thrust, which reduces to zero very quickly when the fuel is used up. The five pointed star profile (3) develops a relatively constant thrust which decreases slowly to zero as the last of the fuel is consumed. The 'cruciform' profile (4) produces progressively less thrust. Fuel in a block with a 'double anchor' profile (5) produces a decreasing thrust which drops off quickly near the end of the burn. The 'cog' profile (6) produces a strong inital thrust, followed by an almost constant lower thrust.

Burn Rate

The burning surface of a rocket propellant grain recedes in a direction perpendicular to this burning surface. The rate of regression, typically measured in millimeters per second (or inches per second), is termed burn rate. This rate can differ significantly for different propellants, or for one particular propellant, depending on various operating conditions as well as formulation. Knowing quantitatively the burning rate of a propellant, and how it changes under various conditions, is of fundamental importance in the successful design of a solid rocket motor.

Propellant burning rate is influenced by certain factors, the most significant being: combustion chamber pressure, initial temperature of the propellant grain, velocity of the combustion gases flowing parallel to the burning surface, local static pressure, and motor acceleration and spin. These factors are discussed below.

Burn rate is profoundly affected by chamber pressure. The usual representation of the pressure dependence on burn rate is the Saint-Robert's Law,

$$r = a P_c^n \quad (1.36)$$

where r is the burn rate, a is the burn rate coefficient, n is the pressure exponent, and Pc is the combustion chamber pressure. The values of a and n are determined empirically for a particular propellant formulation and cannot be theoretically predicted. It is important to realize that a single set of a, n values are typically valid over a distinct pressure range. More than one set may be necessary to accurately represent the full pressure regime of interest.

Example a, n values are 5.6059* (pressure in MPa, burn rate in mm/s) and 0.35 respectively for the Space Shuttle SRBs, which gives a burn rate of 9.34 mm/s at the average chamber pressure of 4.3 MPa.

* NASA publications gives a burn rate coefficient of 0.0386625 (pressure in PSI, burn rate in inch/s).

Temperature affects the rate of chemical reactions and thus the initial temperature of the propellant grain influences burning rate. If a particular propellant shows significant sensitivity to initial grain temperature, operation at temperature extremes will affect the time-thrust profile of the motor. This is a factor to consider for winter launches, for example, when the grain temperature may be lower than "normal" launch conditions.

For most propellants, certain levels of local combustion gas velocity (or mass flux) flowing parallel to the burning surface leads to an increased burning rate. This "augmentation" of burn rate is referred to as erosive burning, with the extent varying with propellant type and chamber pressure. For many propellants, a threshold flow velocity exists. Below this flow level, either no augmentation occurs, or a decrease in burn rate is experienced (negative erosive burning).

The effects of erosive burning can be minimized by designing the motor with a sufficiently large port-to-throat area ratio (Aport/At). The port area is the cross-section area of the flow channel in a motor. For a hollow-cylindrical grain, this is the cross-section area of the core. As a rule of thumb, the ratio should be a minimum of 2 for a grain L/D ratio of 6. A greater Apart /At ratio should be used for grains with larger L/D ratios.

In an operating rocket motor, there is a pressure drop along the axis of the combustion chamber, a drop that is physically necessary to accelerate the increasing mass flow of combustion products toward the nozzle. The static pressure is greatest where gas flow is zero, that is, at the front of the motor. Since burn rate is dependant upon the local pressure, the rate should be greatest at this location. However, this effect is relatively minor and is usually offset by the counter-effect of erosive burning.

Burning rate is enhanced by acceleration of the motor. Whether the acceleration is a result of longitudinal force (e.g. thrust) or spin, burning surfaces that form an angle of about 60-90o with the acceleration vector are prone to increased burn rate.

It is sometimes desirable to modify the burning rate such that it is more suitable to a certain grain configuration. For example, if one wished to design an end burner grain, which has a relatively small burning area, it is necessary to have a fast burning propellant. In other circumstances, a reduced burning rate may be sought after. For example, a motor may have a large L/D ratio to generate sufficiently high thrust, or it may be necessary for a particular design to restrict the diameter of the motor. The web would be consequently thin, resulting in short burn duration. Reducing the burning rate would be beneficial.

There are a number of ways of modifying the burning rate: decrease the oxidizer particle size, increase or reduce the percentage of oxidizer, adding a burn rate catalyst or suppressant, and operate the motor at a lower or higher chamber pressure. These factors are discussed below.

The effect of the oxidizer particle size on burn rate seems to be influenced by the type of oxidizer. Propellants that use ammonium perchlorate (AP) as the oxidizer have a burn rate that is significantly affected by AP particle size. This most likely results from the decomposition of AP being the rate-determining step in the combustion process.

The burn rate of most propellants is strongly influenced by the oxidizer/fuel ratio. Unfortunately, modifying the burn rate by this means is quite restrictive, as the performance of the propellant, as well as mechanical properties, are also greatly affected by the O/F ratio.

Certainly the best and most effective means of increasing the burn rate is the addition of a catalyst to the propellant mixture. A catalyst is a chemical compound that is added in small quantities for the sole purpose of tailoring the burning rate. A burn rate suppressant is an additive that has the opposite effect to that of a catalyst – it is used to decrease the burn rate.

For a propellant that follows the Saint-Robert's burn rate law, designing a rocket motor to operate at a lower chamber pressure will provide for a lower burning rate. Due to the nonlinearity of the pressure-burn rate relationship, it may be necessary to significantly reduce the operating pressure to get the desired burning rate. The obvious drawback is reduced motor performance, as specific impulse similarly decays with reducing chamber pressure.

Product Generation Rate

The rate at which combustion products are generated is expressed in terms of the regression speed of the grain. The product generation rate integrated over the port surface area is

$$q = \rho_p A_b r \qquad (1.37)$$

where q is the combustion product generation rate at the propellant surface, ρ_p is the solid propellant density, Ab is the area of the burning surface, and r is the propellant burn rate.

It is important to note that the combustion products may consist of both gaseous and condensed-phase mass. The condensed-phase, which manifests itself as smoke, may be either solid or liquid particles. Only the gaseous products contribute to pressure development. The condensed-phase certainly does, however, contribute to the thrust of the rocket motor, due to its mass and velocity.

Chamber Pressure

The pressure curve of a rocket motor exhibits transient and steady state behavior. The transient phases are when the pressure varies substantially with time – during the ignition and start-up phase, and following complete (or nearly complete) grain consumption when the pressure falls down to ambient level during the tail-off phase. The variation of chamber pressure during the steady state burning phase is due mainly to variation of grain geometry with associated burn rate variation. Other factors may play a role, however, such as nozzle throat erosion and erosive burn rate augmentation.

Monopropellant Engines

By far the most widely used type of propulsion for spacecraft attitude and velocity control is monopropellant hydrazine. Its excellent handling characteristics, relative stability under normal storage conditions, and clean decomposition products have made it the standard. The general sequence of operations in a hydrazine thruster is:

When the attitude control system signals for thruster operation, an electric solenoid valve opens allowing hydrazine to flow. The action may be pulsed (as short as 5 ms) or long duration (steady state).

The pressure in the propellant tank forces liquid hydrazine into the injector. It enters as a spray into the thrust chamber and contacts the catalyst beds.

The catalyst bed consists of alumina pellets impregnated with iridium. Incoming hydrazine heats to its vaporizing point by contact with the catalyst bed and with the hot gases leaving the catalyst particles. The temperature of the hydrazine rises to a point where the rate of its decomposition becomes so high that the chemical reactions are self-sustaining.

By controlling the flow variables and the geometry of the catalyst chamber, a designer can tailor the proportion of chemical products, the exhaust temperature, the molecular weight, and thus the enthalpy for a given application. For a thruster application where specific impulse is paramount, the designer attempts to provide 30-40% ammonia dissociation, which is about the lowest percentage that can be maintained reliably. For gas-generator application, where lower temperature gases are usually desired, the designer provides for higher levels of ammonia dissociation.

Finally, in a space thruster, the hydrazine decomposition products leave the catalyst bed and exit from the chamber through a high expansion ratio exhaust nozzle to produce thrust.

Monopropellant hydrazine thrusters typically produce a specific impulse of about 230 to 240 seconds.

Other suitable propellants for catalytic decomposition engines are hydrogen peroxide and nitrous oxide, however the performance is considerably lower than that obtained with hydrazine - specific impulse of about 150 s with H2O2 and about 170 s with N2O.

Monopropellant systems have successfully provided orbit maintenance and attitude control functions, but lack the performance to provide weight-efficient large ΔV maneuvers required for orbit insertion. Bipropellant systems are attractive because they can provide all three functions with one higher performance system, but they are more complex than the common solid rocket and monopropellant combined systems. A third alternative are dual mode systems. These systems are hybrid designs that use hydrazine both as a fuel for high performance bipropellant engines and as a monopropellant with conventional low-thrust catalytic thrusters. The hydrazine is fed to both the bipropellant engines and the monopropellant thrusters from a common fuel tank.

Cold gas propulsion is just a controlled, pressurized gas source and a nozzle. It represents the simplest form of rocket engine. Cold gas has many applications where simplicity and/or the need to avoid hot gases are more important than high performance. The Manned Maneuvering Unit used by astronauts is an example of such a system.

Staging

Multistage rockets allow improved payload capability for vehicles with a high ΔV requirement such as launch vehicles or interplanetary spacecraft. In a multistage rocket, propellant is stored in smaller, separate tanks rather than a larger single tank as in a single-stage rocket. Since each tank is discarded when empty, energy is not expended to accelerate the empty tanks, so a higher total ΔV is obtained. Alternatively, a larger payload mass can be accelerated to the same total Δ V. For convenience, the separate tanks are usually bundled with their own engines, with each discard able unit called a stage.

Multistage rocket performance is described by the same rocket equation as single-stage rockets, but must be determined on a stage-by-stage basis. The velocity increment, ΔVi, for each stage is calculated as before,

$$\Delta V_i = C_i LN\left(\frac{m_{oi}}{m_{fi}}\right) \tag{1.38}$$

where moi represents the total vehicle mass when stage i is ignited, and mfi is the total vehicle mass when stage i is burned out but not yet discarded. It is important to realize that the payload mass for any stage consists of the mass of all subsequent stages plus the ultimate payload itself. The velocity increment for the vehicle is then the sum of those for the individual stages where n is the total number of stages.

$$\Delta V_{total} = \sum_{i=1}^{n} \Delta V_i \tag{1.39}$$

We define the payload fraction as the ratio of payload mass to initial mass, or mpl /mo.

For a multistage vehicle with dissimilar stages, the overall vehicle payload fraction depends on how the ΔV requirement is partitioned among stages. Payload fractions will be reduced if the Δ V is partitioned sub optimally. The optimal distribution may be determined by trial and error. A ΔV distribution is postulated and the resulting payload fraction calculated. The ΔV distribution is varied until the payload fraction is maximized. Once the ΔV distribution is selected, vehicle sizing is accomplished by starting with the uppermost or final stage (whose payload is the actual deliverable payload) and calculating the initial mass of this assembly. This assembly then forms the payload for the previous stage and the process repeats until all stages are sized. Results reveal that to maximize payload fraction for a given ΔV requirement:

UNIT III

SOLID PROPULSION SYSTEM

A solid-propellant rocket or solid rocket is a rocket with a rocket engine that uses solid propellants (fuel/oxidizer). Solid-fuel rockets can remain in storage for long periods, and then reliably launch on short notice, they have been frequently used in military applications such as missiles.

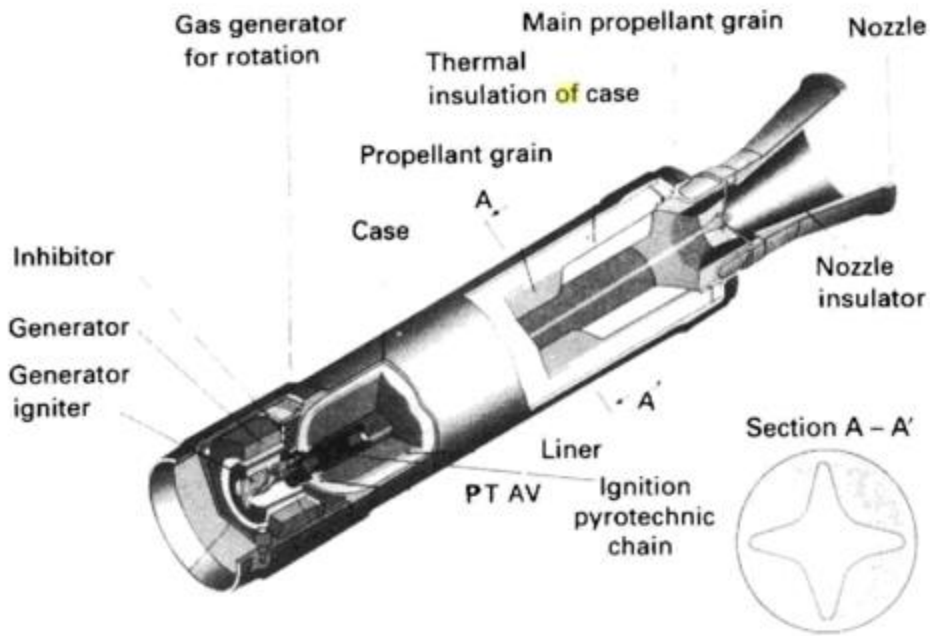


FIG. 1.1. Typical rocket motor.

Design:

Design begins with the total impulse required, which determines the fuel/oxidizer mass. Grain geometry and chemistry are then chosen to satisfy the required motor characteristics.

The following are chosen or solved simultaneously. The results are exact dimensions for grain, nozzle, and case geometries:

- The grain burns at a predictable rate, given its surface area and chamber pressure.
- The chamber pressure is determined by the nozzle orifice diameter and grain burn rate.
- Allowable chamber pressure is a function of casing design.
- The length of burn time is determined by the grain "web thickness".

The grain may or may not be bonded to the casing. Case-bonded motors are more difficult to design, since the deformation of the case and the grain under flight must be compatible.

Common modes of failure in solid rocket motors include fracture of the grain, failure of case bonding, and air pockets in the grain. All of these produce an instantaneous increase in burn surface area and a corresponding increase in exhaust gas production rate and pressure, which may rupture the casing. Another failure mode is casing seal failure.

Seals are required in casings that have to be opened to load the grain. Once a seal fails, hot gas will erode the escape path and result in failure. This was the cause of the Space Shuttle *Challenger* disaster.

A solid rocket motor (SRM) is a machine that provides thrust. Every SRM provides a different amount of thrust depending on payload, destination and other factors. But thrust is not the only performance variable of a SRM.

For example the mass of the propellant is also a major factor. As the propellant burns, it's mass decreases, allowing higher acceleration. Therefore performance assessment models also look at specific impulse, mass flow and other variables.

This model will only examine a few aspects of the performance of a SRM, although many others can also be examined. The solid rocket motor is one of two major classes of motors, where liquid propellant is the other kind. What makes solid unique versus the liquid propellant, is the simplicity of the motor.

The solid propellant contains the fuel and oxidizer. It is therefore enough to ignite the propellant and no other chemical has to be added for combustion to take place. There are different propellant compositions, usually double base or composite, that all have different properties.

Burn rate, ignition temperature, flame temperature and other properties that can all be found using experimentation and modeling. The simplicity of the solid propellant is also it's drawback. Requiring nothing external to burn, the SRM's burning cannot be regulated in-flight by providing more or less fuel, as in engines.

A form of active, real time, control of the SRM's thrust is not possible. Once ignited it will continue to burn until combustion stops. Combustion stops when the propellant has depleted or when it is not possible to sustain combustion conditions (temperature and pressure).

The only form of active thrust control is a thrust termination system. A thrust termination system allows shutdown of the motor, but this usually destroys the motor, and does not allow re-ignition. These systems are usually used as a failsafe to stop failing rockets, or when a stage rocket is separated from the main rocket. Active control of an SRM is not possible and therefore interest lies in a form of passive control.

It is possible to determine in advance at which phase of the SRM's burning it will provide more or less thrust. A model that calculates when a rocket provides more or less thrust is able to predict a way of passive control of the thrust of an SRM.

SOLID PROPELLANTS

Solid propellants are classified into the following four categories:

(a) Double-base or homogeneous propellants,
(b) Composite or heterogeneous propellants,
(c) Composite modified double-base propellants, and
(d) Nitramine propellants.

The composition and salient features of these propellants are discussed in the following sections.

Double-base Propellants

The propellants comprise nitrocellulose (NC) and nitroglycerine (NG) mixed together at the molecular level to form a homogeneous substance. NC constitutes the fuel. Part of the hydroxyl radicals (OH) in cellulose, which consists of carbon, hydrogen and oxygen in a linear structure $[C_6H_{10}O_5]_n$ is substituted by the nitrate radical ONO_2 in NC to give a molecular structure $[C_6H_{10-x}O_{5-x}(NO_3)_x]_n$. The part of hydroxyl radicals, replaced by ONO_2, is represented by 'x' in the above molecular formula and would vary depending on the amount of nitration.

NG, whose chemical formula is $[C_3H_5(ONO_2)_3]$, is based on glycerine or propane triol $[C_3H_5(OH)_3]$. The OH radical of glycerine is replaced by ONO_2 to form NG. NG is oxidiser-rich. The chemical structure of NG is aliphatic with a straight chain structure shown below:

$$\begin{array}{l} H_2C - ONO_2 \\ \;\;| \\ HC - ONO_2 \\ \;\;| \\ H_2C - ONO_2 \end{array}$$

Both NC and NG can be used singly as propellants since they have both oxidiser and fuel in them. These are known as single base propellants.

When NC and NG are used together, they constitute double-base propellant. NG forms a colloid when properly compounded with NC in the presence of a few additives comprising a plasticiser, such as, diethyl phthalate and triacetin, and stabiliser, such as, diphenylamine. Plasticisers not only increase the fluidity but also make the propellant less sensitive to ignition from impact.

Composite Propellants

The propellant is heterogeneous with a solid crystalline oxidiser, such as, ammonium perchlorate (NH_4ClO_4) dispersed in a polymeric fuel. Metal aluminium powder is also added to enhance the energy released during combustion. The polymer binds together the discrete crystalline oxidiser particles and the metal powder to form a tough rubbery mass. The polymeric fuel is, therefore, referred to as binder.

(a) Oxidiser

Ammonium Perchlorate (AP) is generally used for the oxidiser. Unlike other solid oxidisers, such as, ammonium nitrate (NH_4NO_3) and potassium nitrate (KNO_3), AP dissociates easily and is not very hygroscopic, and hence is preferred. There are many energetic perchlorate-containing compounds with lower negative heats of formation, such as, nitronium perchlorate (NO_2ClO_4), hydrazinium perchlorate ($N_2H_5ClO_4$), and hydrazinium diperchlorate [$N_2H_6(ClO_4)_2$]. However, they have poorer compatibility with the polymeric binder and have lower thermal stability with the result that processing of the propellant and storage becomes problematic. More energetic oxidisers, such as, hydrazinium nitroformate [$N_2H_5C(NO_2)_3$] are being investigated for high performance composite propellants.

Toxic gases, such as hydrochloric acid formed during combustion of AP and hydrazinium perchlorates with polymeric fuels, pollute the environment. The possibilities of using environmental-friendly energetic oxidiser ammonium dinitramide (ADN) are being studied.

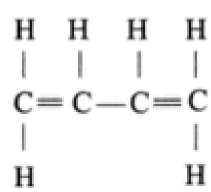
(b) Polymeric Fuel

The polymeric fuel called binder consists of H, C and O atoms (sometimes N and S also). The atomic number of sulphur is 16 compared to 6 for carbon and 1 for hydrogen. Sulphur is, therefore, not a desirable constituent since its atomic mass is high and would result in combustion products with larger molecular mass.

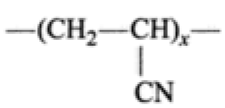
Saturated hydrocarbons (alkanes), in which carbon atoms are attached to each other by means of single bonds, have small but negative values of heats of formation and make good binders. Alkenes with one double bond between carbon atoms, alkadienes with two double bonds, alkynes with one triple bond and alkadiynes with multiple triple bonds between carbon atoms are said to be unsaturated. These have large and negative values of heat of formation.

The hydrocarbons could have a straight chain of alkanes, alkenes, alkadienes, alkynes and alkadiynes or could have ring chains in which case they are known as cycloalkanes, cycloalkenes, etc. These straight chain and ring chain hydrocarbons are known as aliphatic compounds and are distinctly different from the aromatic hydrocarbons which have six carbon rings with three double bonds (benzene).

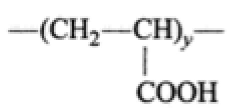
The aliphatic compounds especially those with simpler bond structures are to be preferred as they have smaller negative values of heat of formation. However, most of them like methane, ethane, propane and butane are gases. Polymers with multiple cyclo-butadienes in a linear chain structure are, therefore, chosen. These are called Polybutadienes. The butadiene structure comprising alkadienes with four carbon atoms and two double bonds is shown as follows.



The polybutadiene chain structure is $-(CH_2{=}CH - CH{=}CH_2)_n-$. Here n is the number of butadiene groups. The polybutadiene chain could be attached to poly-acrylo-nitrile group and acrylic acid groups given by:



Acrylo-nitrile



Acrylic acid

to form polybutadiene acrylic acid acrylonitrile (PBAN).

Carboxy and nitrile groups get attached to the butadiene chain to form a highly cross-linked chain network. The PBAN binder with the linear and cross chains has lower ultimate tensile strength with smaller elongation. It has been used for making propellant grains for large rockets, such as, the solid rocket booster of the Space Shuttle. The structure of the PBAN binder is given below:

$$-(CH_2{=}CH-CH{=}CH_2)_n-(\underset{\displaystyle COOH}{\underset{|}{CH_2-CH}})_x-(\underset{\displaystyle CN}{\underset{|}{CH_2-CH}})_y-$$

The randomness of the cross linking can be reduced by removing the carboxyl groups in PBAN and locating them at the ends of the butadiene chain. This provides better mechanical properties to the binder. The binder with the carboxyl radicals located in a controlled way at the ends of the butadiene chain is known as Carboxy Terminated Poly Butadiene (CTPB).

$$-(CH_2{=}CH-CH{=}CH_2)_{n1}-(\underset{\displaystyle COOH}{\underset{|}{CH_2-CH}})_{x1}-(CH_2{=}CH-CH{=}CH_2)_{n2}-$$

If the carboxyl radical at the chain ends are replaced by hydroxyl (OH) radical, the binder would give higher performance in a propellant. This is because H in the OH radical has a lower atomic number than C in COOH and the energetics of the hydrogen is superior to carbon. The butadiene chains terminated with OH at the ends is called Hydroxy Terminated Poly Butadiene (HTPB). High performance propellants make use of HTPB.

$$HO-(CH_2{=}CH-CH{=}CH_2)_n-OH$$

HTPB and PBAN binders are most widely used for the manufacture of composite propellants.

The molecular mass of the binders are large considering the long chain, typical values ranging from 30,000 to 1,00,000 kg/kmole.

Polyvinyl chloride (PVC) was used earlier as the binder. It consists of the chain:

$$-(CH_2-\overset{\displaystyle Cl}{\overset{|}{CH}})_x-$$

It gives poor performance as it contains less carbon and hydrogen.

Polyurethanes have been used in the past like PVC. They are good energy-wise but contain oxygen and are sensitive to moisture. The chemical structure of polyurethane is given below:

$$(O-(CH_2)_n-O-\overset{\displaystyle O}{\overset{\|}{C}}-NH-(CH_2)_n-\overset{\displaystyle O}{\overset{\|}{C}})_m-$$

Certain high energy polymers, such as, Glycidyl Azide Polymer (GAP), contain larger percentage of hydrogen and provide higher performance.

Metals:

Metal powders, such as, aluminium are also used as fuel in order to enhance the energy release from combustion since metal combustion is highly exothermic. Aluminium is a light metal and is, therefore, preferable. The aluminium oxide formed as a product of combustion was seen to have a very high negative value of heat of formation (Fig. 4.3).

The use of metal hydrides instead of metals has been thought of with a view to have as much hydrogen in the propellant as possible. The hydrogen, as seen earlier, would contribute to enhancing the specific impulse by lowering the molecular mass of the combustion products. The hydrides are not very compatible with polymeric binders and their use has not been demonstrated in practice.

SELECTION CRITERIA OF SOLID PROPELLANTS:

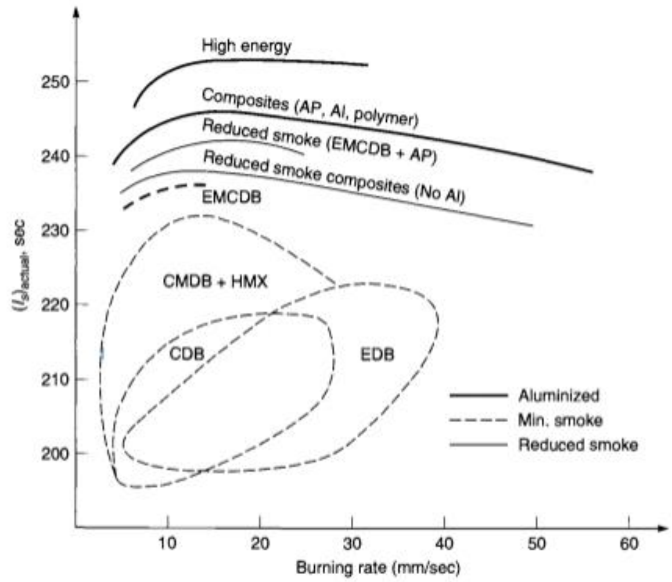


FIGURE 12–1. Estimated actual specific impulse and burning rate for several solid propellant categories. (Adapted and reproduced from Ref. 12–1 with permission of the American Institute of Aeronautics and Astronautics [AIAA].)

CRITERIA FOR SELECTION

Several selection criteria may be in conflict with each other. For example, some propellants with a very high specific impulse are more likely to experience combustion instabilities.

In liquid propellant systems, where the oxidizer tank is pressurized by a solid propellant gas generator and where the fuel-rich hot gases are separated by a thin flexible diaphragm from the oxidizer liquid, there is a trade-off between a very compact system and the potential for a damaging system failure (fire, possible explosion, and malfunction of system) if the diaphragm leaks or tears.

In electric propulsion, high specific impulse is usually accompanied by heavy power generating and conditioning equipment.

Actual selection will depend on the balancing of the various selection factors in accordance with their importance, benefits, or potential impact on the system, and on quantifying as many of these selection factors as possible through analysis, extrapolation of prior experience/data, cost estimates, weights, and/or separate tests.

For example, the monitoring of extra sensors can prevent the occurrence of certain types of failure and thus enhance the propulsion system reliability, yet the extra sensors and control components contribute to the system complexity and their possible failures will reduce the overall reliability.

The selection process may also include feedback when the stated propulsion requirements cannot be met or do not make sense, and this can lead to a revision of the initial mission requirements or definition.

Mission Definition

Purpose, function, and final objective of the mission of an overall system are well defined and their implications well understood. There is an expressed need for the mission, and the benefits are evident. The mission requirements are well defined. The payload, flight regime, vehicle, launch environment, and operating conditions are established. The risks, as perceived, appear acceptable. The project implementing the mission must have political, economic, and institutional support with assured funding. The propulsion system requirements, which are derived from mission definition, must be reasonable and must result in a viable propulsion system.

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 (see Ref. 17-5). Benefits of achieving the mission should appear to justify costs. Investment in new facilities should be low. Few, if any, components should require expensive materials. For commercial applications, such as communications satellites, the return on investment must look attractive. No need to hire new, inexperienced personnel, who need to be trained and are more likely to make expensive errors.

System Performance

The propulsion system is designed to optimize vehicle and system performance, using the most appropriate and proven technology. Inert mass is reduced to a practical minimum, using improved materials and better understanding of loads and stresses. Thrust-time profiles and number of restarts must be selected to optimize the vehicle mission. Vehicles must operate with adequate performance for all the possible conditions (pulsing, throttling, temperature excursions, etc.). Vehicles should be storable 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 the selected 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 personnel injury, damage to equipment, facilities, or the environment is minimal. Certain mishaps or failures will result in a change in the operating condition or the safe shutdown of the propulsion system. Applicable safety standards must be obeyed. Inadvertent energy input to the propulsion system (e.g., bullet impact, external fire) should not result in a detonation. The probability for any such drastic failures should be very low. Safety monitoring and inspections must have proven effective in identifying and preventing a significant share of possible incipient failures (see Ref. 17-6). Adequate safety factors must be included in the design. Spilled liquid propellants should cause no undue hazards. All systems and procedures must conform to the safety standards. Launch test range has accepted the system as being safe enough to launch.

Solid Propellant Rocket Advantages

- Simple design (few or no moving parts).
- Easy to operate (little preflight checkout).
- Ready to operate quickly.
- Will not leak, spill, or slosh.
- Sometimes less overall weight for low total impulse application.
- Can be throttled or stopped and restarted (a few times) if preprogrammed.
- Can provide TVC, but at increased complexity.
- Can be stored for 5 to 25 years.
- Usually, higher overall density; this allows a more compact package, a smaller vehicle (less drag).
- Some propellants have nontoxic, clean exhaust gases, but at a performance penalty.
- Some grain and case designs can be used with several nozzles.
- Thrust termination devices permit control over total impulse.
- Ablation and gasification of insulator, nozzle, and liner materials contribute to mass flow and thus to total impulse.
- Some tactical missile motors have been produced in large quantities (over 200,000 per year). Can be designed for recovery, refurbishing, and reuse (Space Shuttle solid rocket motor).

Solid Propellant Rocket Disadvantages

- Explosion and fire potential is larger; failure can be catastrophic; most cannot accept bullet impact or being dropped onto a hard surface.
- Many require environmental permit and safety features for transport on public conveyances.
- Under certain conditions some propellants and grains can detonate.
- Cumulative grain damage occurs through temperature cycling or rough handling; this limits the useful life. If designed for reuse, it requires extensive factory rework and new propellants. Requires an ignition system.
- Each restart requires a separate ignition system and additional insulation--in practice, one or two restarts. Exhaust gases are usually toxic for composite propellants containing ammonium perchlorate.

- Some propellants or propellant ingredients can deteriorate (self-decompose) in storage.
- Most solid propellant plumes cause more radio frequency attenuation than liquid propellant plumes.
- Only some motors can be stopped at random, but motor becomes disabled (not reusable).
- Once ignited, cannot change predetermined thrust or duration. A moving pintle design with a variety throat area will allow random thrust changes, but experience is limited.
- If propellant contains more than a few percent particulate carbon, aluminum, or other metal, the exhaust will be smoky and the plume radiation will be intense.
- Integrity of grain (cracks, unbonded areas) is difficult to determine in the field.
- Thrust and operating duration will vary with initial ambient grain temperature and cannot be easily controlled. Thus the flight path, velocity, altitude, and range of a motor will vary with the grain temperature.
- Large boosters take a few seconds to start.
- Thermal insulation is required in almost all rocket motors.
- Cannot be tested prior to use.
- Needs a safety provision to prevent inadvertent ignition, which would lead to an unplanned motor firing. Can cause a disaster.

SOLID PROPELLANT GRAIN DESIGN

To design a solid propellant grain is to conceive and to define a grain which satisfies various requirements. This chapter describes the methods and procedures used today to design propellant grains. It describes and analyses:

- the various types of grain and the various families of propellant which are available and used today,
- the detailed requirements that a solid propellant grain must satisfy,
- the methods which are used to precisely define the propellant, the architecture and the configuration of the grain, and more specifically the methods used in order to ensure required ballistic performances though maintaining structural integrity of the grain (which is submitted to mechanical loads all through its life),
- an overview on a method of solid propellant grain reliability assessment.

Propellant grain

Two main configurations — free-standing grain and case-bonded grain — with various central port geometries are used to fulfill the required performance objectives.

- Free-standing grains. Free-standing grains are contained inside a cylindrical plastic cartridge (PVC, etc.). They are secured inside the case by various support elements such as wedges, springs or grids.
- Case-bonded grains. These are obtained by casting the propellant, before polymerization has occurred, directly into a case already provided with thermal insulation. Additional manufacturing steps (molding, curing, machining, control) required for the propellant grain are performed on the loaded case.

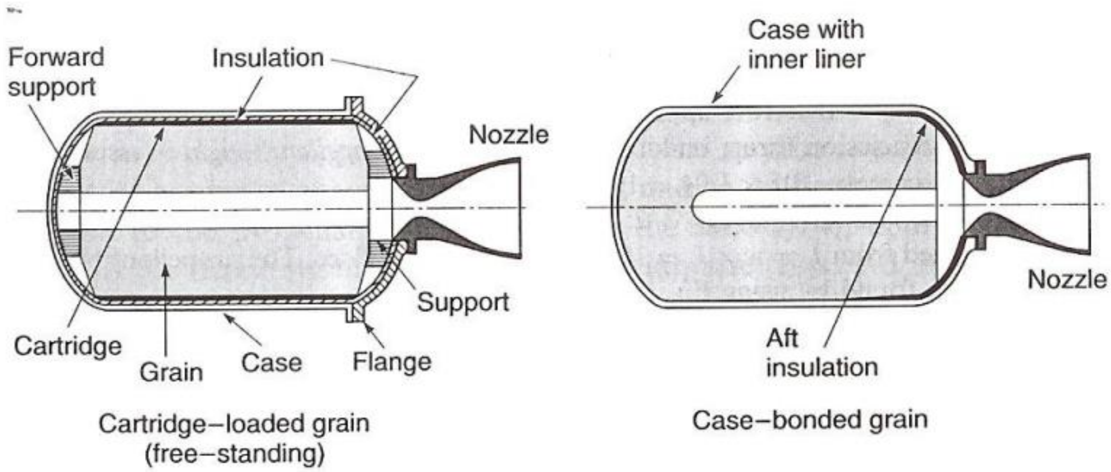


FIGURE 11–14. Simplified schematic diagrams of a free-standing (or cartridge-loaded) and a case-bonded grain.

Case-bonded grain configurations

When propellant grains have an outer diameter larger than 500 mm or a weight of more than 300 kg, they are almost always case-bonded. High-performance, middle-sized grains (outer diameter between 100 mm and

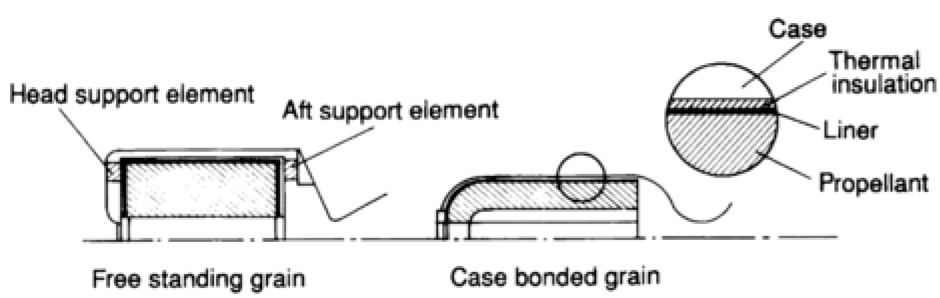


FIG. 2.1. Case bonded grain.

500 mm, weight between 10 and 300 kg) are case-bonded, but free-standing middle-sized grains are very common. For small rocket motors free-standing grains are generally used.

Case-bonded grains generally have a central port, the outer surface of the grain is bonded by a liner (and a thermal insulation) to the motor case. During firing, combustion of the propellant is initiated on the internal surface of the central port and proceeds radially toward the case (and to a certain extent longitudinally depending on the exact geometry). Exact grain geometry is obtained during manufacture of the grain either by direct casting in the case around the mandrel or by machining the port after casting and curing have been completed.

Axisymmetric configurations

AXIL: Axisymmetric grain with annular slots. The slots are circular; their axis is the same as the grain axis. They are located all along the central port (Fig. 2).

AXAR: Axisymmetric grain with annular slots. This configuration is similar to AXIL, except that the slots are located near the aft-end of the central port (Fig. 3).

CONOCYL (contraction of cone and cylinder): Axisymmetric grain with annular slots. The tips of the annular slots are inclined toward grain head-end so that a part of the grain is cone-shaped (Fig. 4).

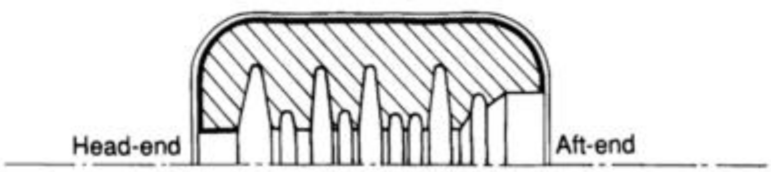


FIG. 2.2. Axil configuration.

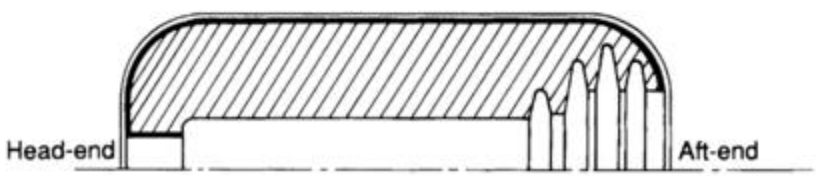


FIG. 2.3. Axar configuration.

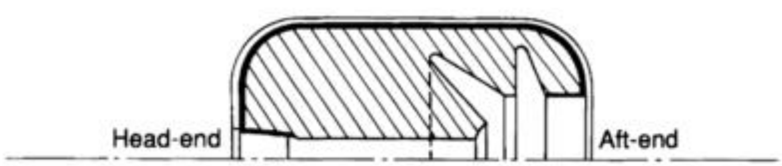


FIG. 2.4. Conocyl configuration.

Cylindrical configurations

STAR: The cross-section of the central port has the shape of an *n*-points star. The contour of the star is constant along the axis [in some cases it may be slightly tapered for manufacture practicality (Fig. 5)].

WAGON WHEEL: The cross-section of the central port looks like a wagon wheel (Fig. 6). Numerous parent configurations exist, such as dendrite, anchor and dogbone configurations.

Three-dimensional geometries

FIG. 2.5. Star configuration.

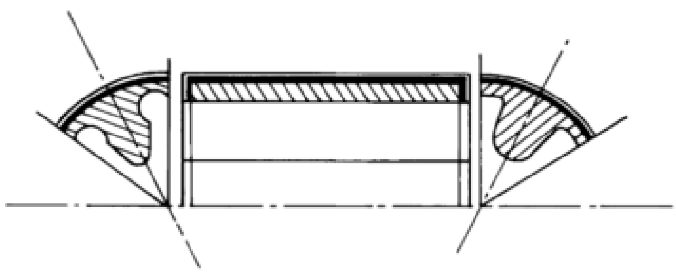


FIG. 2.6. Wagon wheel configuration.

General principles for selection of grain configuration

For selection of the grain configuration, the main factors which are taken into account are:

- volume available for the propellant grain;
- grain length to diameter ratio (L/D);
- grain diameter to web thickness ratio (D/e);
- thrust versus time curve: this gives a good idea of what should be the burning area versus web burned curve (neutral, regressive, progressive, dual-level);

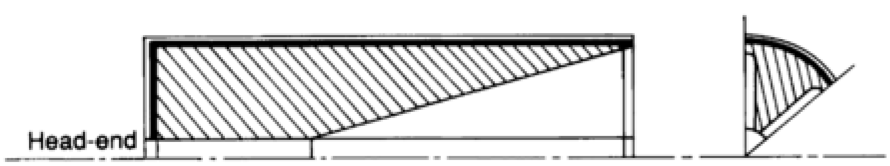


FIG. 2.8. Trumpet configuration.

- volumetric loading fraction: this can be estimated from required total impulse and actual specific impulse of available propellants;
- critical loads (thermal cycles, pressure rise at ignition, acceleration, internal flow);
- manufacture practicality, which depends on case geometry (some grain configurations are more or less easy to obtain);
- fabrication cost: this can be the critical factor for selecting a given configuration.

There is no definite procedure to select a grain configuration in order to satisfy a set of requirements, because there are often several technical solutions to the propulsion problem.

Special requirements

Other requirements are necessary to the designer in order to define a satisfactory propellant grain:

- Maximum weight of propellant grain.
- Maximum weight of total inert (thermal insulation, liner and restrictor).
- Maximum axial and transverse acceleration undergone by the propellant grain during operation of the rocket motor.
- Rocket spin rate (for instance for unguided rockets).
- Dispersions on pressure, thrust, total impulse and burning time have to be specified. Depending on the corresponding requirements, manufacturing process and control operations may be strongly affected and thus the cost of the grain also.
- Plume characteristics (emission and transmission in the visible, infrared and electromagnetic wavelengths range).

Influence of Initial Temperature of Propellant on Burn Rate

The effect of initial temperature is contained in the coefficient in the burn rate law $r = a\, p^n$. The influence of variations of the initial temperature is expressed by a temperature-sensitivity parameter π_r, defined as the fractional variation in r due to unit temperature change at constant pressure:

$$\pi_r = \frac{1}{r}\frac{dr}{dT}\Big|_p$$

Burn rate of the double-base propellants are more sensitive to the changing propellant temperatures due to the chemical reactions within the propellant in the foam zone. Typical values of π_r at a pressure of 7 MPa are about $5 \times 10^{-3}\ ^{\circ}C^{-1}$ for the double-base propellants and about $3 \times 10^{-3}\ ^{\circ}C^{-1}$ for composite propellants. The sensitivity-parameter π_r for AP and HMX burning are lower at about $2 \times 10^{-3}\ ^{\circ}C^{-1}$ at 7 MPa.

The burn rate of a propellant at any temperature T can be determined from its known value at temperature T_0, say r_{T_0}, if π_r is given. Equation 5.10 is written for the specified constant value of p as:

$$\frac{dr}{r} = \pi_r\, dT$$

On integrating from the reference value of T_0 at which the burn rate (at given pressure) is r_{T_0}, we have the value r at temperature T at the same pressure p as:

$$\ln\left(\frac{r}{r_{T_0}}\right) = \pi_r \int_{T_0}^{T} dT$$

$$r = r_{T_0} \exp[\pi_r (T - T_0)]$$

CHOICE OF INDEX *n* FOR STABLE OPERATION OF SOLID PROPELLANT ROCKETS

A smaller value of the index n in the burn rate law $r = a\,p^n$ is desirable in order to have a stable burning in a solid propellant rocket. The reason for this may be understood by considering the mass generation and mass leaving a rocket having a burning surface area S_b and a nozzle throat of area A_t, as shown in Fig. The rate of mass generation from the propellant surface $\dot{m}_{\text{gen}}$ is $S_b r \rho_p$, where ρ_p is the propellant density and r is the burn rate. The rate at which hot gases leave the nozzle $\dot{m}_n$ is $p\,A_t/C^*$, where C^* is the characteristic velocity of the propellant and p is the chamber pressure. The rate of mass accumulation in the chamber is the difference between the mass generation and the mass leaving through the nozzle; it is represented as:

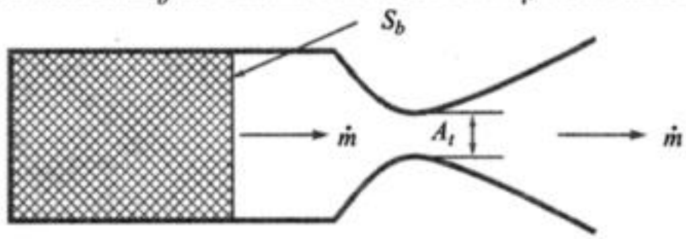


Fig. Mass Generation and Mass Depletion

$$\left.\frac{dm}{dt}\right|_{\text{accumulation}} = \dot{m}_{\text{gen}} - \dot{m}_n$$

Substituting the values of mass generation and mass leaving, and noting that for steady state conditions there is no mass accumulation in the rocket, we have:

$$S_b r \rho_p - \frac{1}{C^*} p\,A_t = 0$$

The burn rate law $r = ap^n$ when incorporated in the above equation, gives the steady state or equilibrium pressure as:

$$p = \left\{\rho_p\, a\, C^* \left(\frac{S_b}{A_t}\right)\right\}^{\frac{1}{1-n}}$$

This equilibrium pressure is denoted by p_{eq}. If n is near unity, the value of $1/(1-n)$ would be very large and any small variation of S_b, A_t, ρ_p or a propellant characteristic which influences C^* and a would have a significant influence on p_{eq}.

The influence can be incorporated by expressing the mass of gas m at any instant of time in the chamber in terms of the volume of gas in the chamber (V) using the ideal gas equation:

$$m = \frac{pV}{RT}$$

R is the specific gas constant and T is the gas temperature. Substituting in eq. 5.13 and simplifying, we have:

$$\frac{d}{dt}\left(\frac{pV}{RT}\right) = S_b \rho_p a p^n - \frac{pA_t}{C^*}$$

The gas temperature does not vary significantly and the left side of the above equation gives:

$$\frac{d}{dt}\left(\frac{pV}{RT}\right) = \frac{V}{RT}\frac{dp}{dt} + \frac{p}{RT}\frac{dV}{dt}$$

Noting that $\frac{dV}{dt} = S_b a p^n$ and $\frac{p}{RT} = \rho_g$ = density of the gas, we have

$$\frac{V}{RT}\frac{dp}{dt} = S_b a p^n (\rho_p - \rho_g) - \frac{pA_t}{C^*}$$

For steady state conditions, $\frac{dp}{dt} = 0$.

The equilibrium value of pressure p_{eq} is:

$$p_{eq} = \left\{ a C^* (\rho_p - \rho_g)\left(\frac{S_b}{A_t}\right)\right\}^{\frac{1}{1-n}}$$

limit of ρ_g being much smaller than ρ_p is written as:

$$\frac{V}{RT}\frac{dp}{dt} = S_b a p^n \rho_p - \frac{pA_t}{C^*} = \dot{m}_{gen} - \dot{m}_n$$

If the terms $\dot{m}_{gen} = S_b a p^n \rho_p$ and $\dot{m}_n = \frac{pA_t}{C^*}$ were to be plotted as a function of p for given values of S_b, A_t, ρ_p, C^* and a, we observe that $\dot{m}_n$ increases linearly with increase of pressure. The mass generation rate $\dot{m}_g$, however, sharply increases with p when $n > 1$ while the rate of its increase comes down with increase of p when $n < 1$. These trends are shown in Figs. 5.13 and 5.14 for $n > 1$ and $n < 1$ respectively. The point of intersection O of $\dot{m}_g$ and $\dot{m}_n$ curves gives the steady state or equilibrium pressure p_{eq} obtained in eq. 5.15.

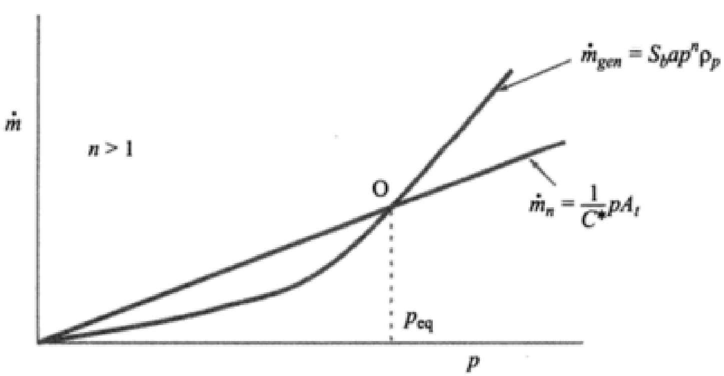


Fig. 5.13 Unstable Operation for $n > 1$

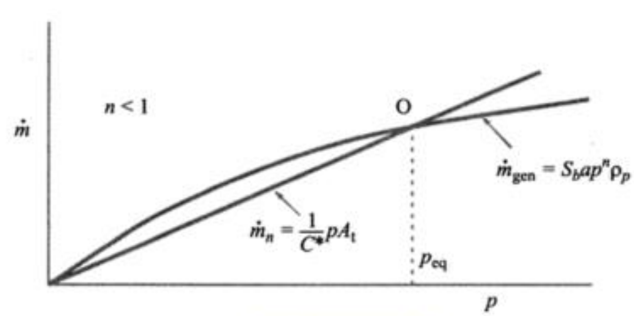


Fig. 5.14 Stable Operation for $n < 1$

The condition for stable operation of a solid propellant rocket is $n < 1$. If increase in the volume of gas due to consumption of the propellant is to be accounted, we have by comparing eqs. 5.19 and 5.21,

the condition for stable operation as: $n < \dfrac{1}{1-\dfrac{\rho_g}{\rho_p}}$.

The Ignition of Solid Propellants.

In simplest terms, ignition consists in supplying energy from an external source to the propellant surface to produce a chemical and thermal state approximating that characteristic of steady burning. If the ignition system is well matched to the requirements of the propellant, energy will be supplied at a rate and in a quantity sufficient to lead to a close approximation of the final steady state.

With the removal of the ignition stimulus, the normal combustion processes will take over the control of the reaction without discontinuity. If the energy supply from the igniter is not matched to the requirements of the propellant, various abnormalities such as overignition, igniter peaks, ignition delays, hangfires, and chuffing may occur in going from the transient to the steady state.

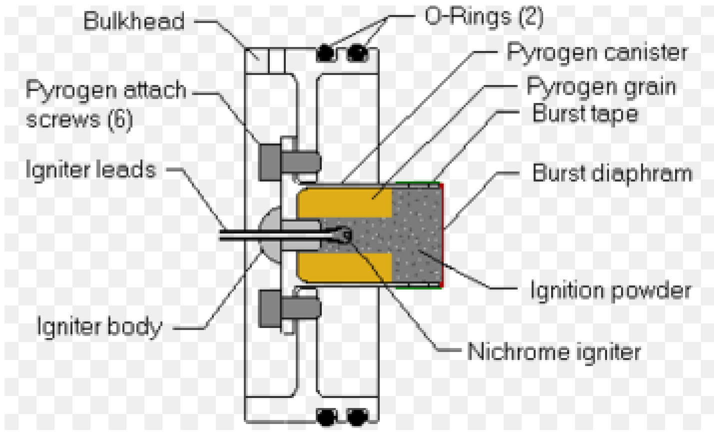
The ignition of gun or rocket propellant charges is usually accomplished by means of a small charge of black powder set off with an electric match. Less frequently, a metal oxidizer or other pyrotechnic mixture, finely divided propellant, or nitrocellulose is used as the main igniter charge.

In such a system, energy transfer to the surface of the propellant grain can occur principally through conductive and convective transfer from the hot igniter gas, through radiative transfer from the hot gas and incandescent solid particles, and through the impingement of hot solid particles from the igniter onto the propellant surface.

Ignition has been accomplished by each of these mechanisms under conditions such that the alternate methods of energy transfer could not play a significant part. The relative importance which the various mechanisms will assume in a particular case will depend on the composition of igniter and propellant and on the geometric parameters of the charge and igniter system.

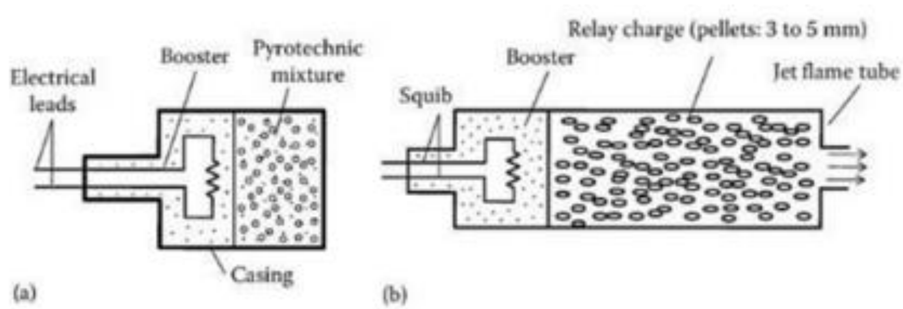
The use of black powder affords conditions favorable to all three mechanisms of energy transfer. The use of "gasless" igniters of the metal oxidizer type largely precludes the possibility of conductive and convective transfer from the gas. The effectiveness of this type of igniter must be due to the presence of hot solid particles coming in contact with the propellant surface and perhaps to radiation from the hot particles.

Pyrogen Igniter: This is basically a small rocket motor used to ignite a larger rocket motor. For Pyrogen igniters, the initiator and the booster charge are similar to that of pyrotechnic igniters. Reaction products from the main charge impinge on the surface of the rocket motor grain, producing motor ignition.



Comparison between Pyrotechnic and Pyrogen Igniters

	Pyrotechnic	Pyrogen
1. Main charge	Pyrotechnic propellant	Fast-burning propellant
2. Configuration	Powders/pellets	Cast as grain
3. Action time	Shorter burn time (100–200 ms)	Longer burn time (250–1000 ms)
4. Applications	In small solid-propellant rocket engines	In large solid-propellant rocket engines
5. Relative merits/ demerits	1. Performance moderately controllable	1. Performance closely controllable
	2. Thrust law depends on test chamber volume	2. Thrust law independent of test chamber volume
	3. Simple construction and processing	3. Complex construction and processing
	4. Lower cost	4. Higher cost



:6 Schematic of pyrotechnic igniters: (a) basket igniter; (b) jet flame pyrotechnic igniter.

Table 4.8-1 Common Types of Ignition Mechanisms

Type	Comments	No. of start
1) Gasoline-soaked cotton cloth or rags burning with air	Used by Goddard and amateurs in 1920s; flown in 1926; now abandoned	Single
2) Pyrotechnic or solid propellant	Goddard used in 1920s and 1930s with TCs; used with GG beginning about 1950	Single
Same but with rotating wheel and four solid cartridges	TC of German V-2 LPRE	Single
3) Electric power, spark plug, hot-wire glow plug,	Ignition prechamber in large LPREs, with small flow of both propellants; LOX/RP-1, or LOX/LH_2; used in space shuttle main engine	Multiple
4) Initial slug of hypergolic fuel	Aluminum triethyl will ignite with LOX or air, used in F-1, RD-170, H-1	Multiple

Pyrotechnic initiators are often controlled electrically (called electro-pyrotechnic initiators), e.g. using a heated bridge wire or a *bridge resistor*. They are somewhat similar to blasting caps or other detonators, but they differ in that there is no intention to produce a shock wave. An example of such pyrotechnic initiator is an electric match.

The energetic material used, often called **pyrogen**, is usually a pyrotechnic composition made of a fuel and oxidizer, where the fuel produces a significant amount of hot particles that cause/promote the ignition of the desired material.

Common compositions

One of the most common initiators is **ZPP**, or **zirconium – potassium perchlorate** – a mixture of metallic zirconium and potassium perchlorate. It is also known as the **NASA Standard Initiator**. It yields rapid pressure rise, generates little gas, emits hot particles when ignited, is thermally stable, has long shelf life, and is stable under vacuum. It is sensitive to static electricity.

Another common igniter formula is **BPN**, **BKNO3**, or **boron – potassium nitrate**, a mixture of 25% boron and 75%potassium nitrate by weight. It is used e.g. by NASA. It is thermally stable, stable in vacuum, and its burn rate is independent of pressure.

In comparison with black powder, BPN burns significantly hotter and leaves more of solid residues, therefore black powder is favored for multiple-use systems.

Metal hydride-oxidizer mixtures replace the metal with its corresponding hydride. They are generally safer to handle than the corresponding metal-oxidizer compositions. During burning they also release hydrogen, which can act as a secondary fuel. Zirconium hydride, titanium hydride, and boron hydride are commonly used.

ZHPP (**zirconium hydride – potassium perchlorate**) is a variant of ZPP that uses zirconium hydride instead of pure zirconium. It is significantly safer to handle than ZPP.

THPP (titanium hydride potassium perchlorate) is a mixture of titanium(II) hydride and potassium perchlorate. It is similar to ZHPP. Like ZHPP, it is safer to handle than titanium-potassium perchlorate.

Titanium-boron composition is one of the hottest pyrotechnic reactions in common usage. It is solid-state, gasless. It can be used as a pyrotechnic initiator or for heating confined gas to perform mechanical work.

Nickel-aluminium laminates can be used as electrically initiated pyrotechnic initiators. Nano Foil is such material, commercially available.

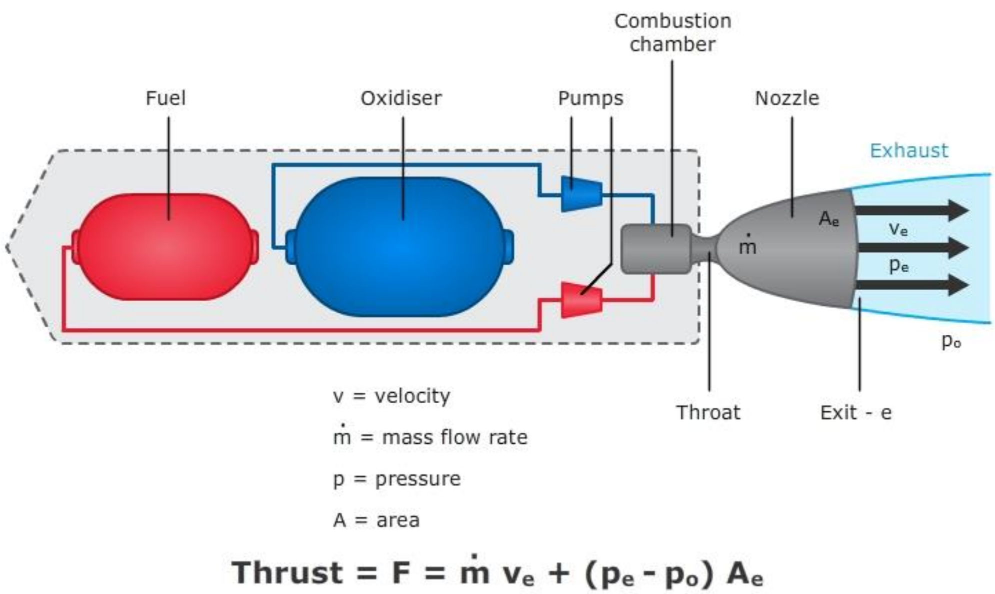
UNIT - IV

LIQUID PROPULSION SYSTEMS

A **liquid-propellant rocket** or **liquid rocket** is a rocket engine that uses liquid propellants. Liquids are desirable because their reasonably high density allows the volume of the propellant tanks to be relatively low, and it is possible to use lightweight centrifugal turbo pumps to pump the propellant from the tanks into the combustion chamber, which means that the propellants can be kept under low pressure. This permits the use of low-mass propellant tanks, resulting in a high mass ratio for the rocket.

Liquid rockets can be monopropellant rockets using a single type of propellant, bipropellant rockets using two types of propellant, or more exotic tri propellant rockets using three types of propellant. Some designs are throttle able for variable thrust operation and some may be restarted after a previous in-space shutdown. Liquid propellants are also used in hybrid rockets, in which a liquid oxidizer is generally combined with a solid fuel.

Bipropellant liquid rockets generally use a liquid fuel, such as liquid hydrogen or a hydrocarbon fuel such as RP-1, and a liquid oxidizer, such as liquid oxygen. The engine may be a cryogenic rocket engine, where the fuel and oxidizer, such as hydrogen and oxygen, are gases which have been liquefied at very low temperatures.



Liquid propellants comprise liquid fuels and liquid oxidisers. They are classified according to their energy content, their ignitability and their storage. In the following, we examine the different liquid propellants based on the above classification.

Energy Content of Propellant

Energy refers to the heat of combustion. It was seen earlier in this chapter that higher energy release is obtained from fuels and oxidisers having small and negative values of heats of formation or positive heats of formation. The fuels considered were kerosene, hydrazine, mono-methyl hydrazine and hydrogen while the oxidisers consisted oxygen, N_2O_4 and HNO_3.

Liquid propellants are categorised as per their specific impulse into:

(a) Low-energy propellants
(b) Medium-energy propellants, and
(c) High-energy propellants.

The classification is said to be based on energy though it is actually done on the basis of specific impulse. The specific impulse takes into account the molecular mass of the combustion products in addition to the energy released by the propellant.

Low-energy Propellants

The low energy propellants are those which give I_{sp} at sea-level conditions less than about 3000 Ns/kg. They comprise liquid oxygen (LO_2) and alcohol which was successfully used in the V2 rockets during the Second World War. Liquid oxygen and kerosene, also known as rocket propellant (RP) and certain combinations of oxidisers and fuels, such as, nitric acid with aniline, xylidine or hydrazine and N_2O_4 with hydrazine, mono-methyl hydrazine and unsymmetrical di-methyl hydrazine also constitute the low energy propellants. The characteristics of these propellants are discussed in the following:

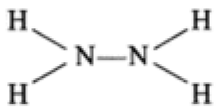
Aniline (C_6H_7N) is an aromatic amine in which the amine radical NH_2 is attached to the benzene chain. In the case of xylidine ($C_8H_{11}N$) two methyl radicals are additionally attached to the benzene chain.

Both aniline and xylidine react spontaneously with nitric acid (HNO_3) when brought in contact in the liquid phase itself. The combustion products, however, contain a lot of soot considering the aromatic nature of the fuel. The energy release is low since the heats of formation of both the fuels (about –100 kJ/mole) and HNO_3 (about –200 kJ/mole) are somewhat large and negative. These propellants are used only when high performance is not a requirement.

The oxygen content in HNO_3 is enhanced by dissolving about 15 per cent NO_2 in it. Red fumes are obtained from the high NO_2 content and the mixture is known as Red Fuming Nitric Acid (RFNA). The incorporation of a small quantity of NO_2 content in HNO_3 (about 0.5 per cent) is known as White Fuming Nitric Acid (WFNA).

HNO_3 is very corrosive. The addition of about 0.5 to 0.8 per cent of hydrofluoric acid (HF) to RFNA inhibits the corrosion of the storage vessel and the inhibited RFNA is termed as Inhibited Red Fuming Nitric Acid (IRFNA). IRFNA with aniline and xylidine have been extensively used in the past especially for low performing missiles.

Fuel hydrazine (N_2H_4) has a much simpler molecular structure with bonds as illustrated below:



It has a positive heat of formation and dissociates in the presence of a catalyst to liberate heat. It is, therefore, sometimes used as a single propellant without the need for an oxidiser and is, therefore, called a monopropellant. Similarly, the oxidiser hydrogen peroxide (H_2O_2) liberates heat during its decomposition and can be used as a monopropellant. N_2H_4 combines readily with oxidisers, such as, HNO_3 and N_2O_4 giving much higher heat release than RFNA with aniline and xylidine and forms combustion products with lower molecular mass.

Medium-energy Propellants

The use of UDMH, N_2H_4 and Az50 with LO_2 gives sea-level I_{sp} between 3,000 and 3,500 Ns/kg. The use of kerosene with liquid Fluorine, which is an extremely reactive oxidiser also provides the higher sea-level I_{sp} between 3,000 and 3,500 Ns/kg. Such of the propellant combinations having I_{sp} in this range are known as medium energy propellants. The medium energy propellants are not popular and only UDMH-LO_2 combination has been used in practice. The use of Fluorine has been difficult in view of its toxicity and reactivity.

High-energy Propellants

High energy propellants have sea-level I_{sp} greater than 4000 Ns/kg. Two propellant combinations comprising Liquid Hydrogen (LH_2) with LO_2 and with Liquid Fluorine (LF) fall in this category. Hydrogen contributes to a significantly reduced molecular mass of the combustion gases. Since LH_2, LO_2 and LF are liquids only at cryogenic temperatures, these propellant combinations [LH_2/LO_2, LH_2/LF] are known as cryogenic propellants. The boiling point of LF is about 5°C lower than LO_2. LF is seldom used for reasons of reactivity cited earlier. The only high energy propellant used in practice is LH_2-LO_2.

Common Physical Hazards

Although the several categories of hazards are described below, they do not all apply to every propellant. The hazards are different for each specific propellant and must be carefully understood before working with that propellant. The consequences of unsafe operation or unsafe design are usually also unique to several propellants.

Corrosion.

Various propellants, such as nitrogen tetroxide or hydrogen peroxide, have to be handled in containers and pipelines of special materials. If the propellant were permitted to become contaminated with corrosion products, its physical and chemical properties could change sufficiently to make it unsuitable for rocket operation. The corrosion of the gaseous reaction products is important in applications in which the reaction products are likely to damage structure and parts of the vehicle or affect communities and housing near a test facility or launch site.

Explosion Hazard.

Some propellants, such as hydrogen peroxide and nitromethane, are unstable and tend to detonate under certain conditions of impurities, temperature, and shock. If liquid oxidizers (e.g., liquid oxygen) and fuels are mixed together they can be detonated. Unusual, rare flight vehicle launch or transport accidents have caused such mixing to occur (see Refs. 7-2 and 7-3).

Fire Hazard.

Many oxidizers will start chemical reactions with a large variety of organic compounds. Nitric acid, nitrogen tetroxide, fluorine, or hydrogen peroxide react spontaneously with many organic substances. Most of the fuels are readily ignitable when exposed to air and heat.

Accidental Spills.

Unforeseen mishaps during engine operation and traffic accidents on highways or railroads while transporting hazardous materials, including propellants, have on occasion caused spills, which expose people to unexpected fires, or potential health hazards. The U.S. Department of Transportation has rules for marking and containing hazardous materials during transport and also guidelines for emergency action (see Ref. 7-4).

Health Hazards.

Many propellants are toxic or poisonous, and special precautions have to be taken to protect personnel. Fluorine, for example, is very poisonous. Toxic propellant chemicals or poisonous exhaust species can enter the human body in several ways. The resulting health disorders are propellant specific. Nitric acid can cause severe skin burn and tissue disintegration. Skin contact with aniline or hydrazine can cause nausea and other adverse health effects. Hydrazine and its derivatives, such as dimethylhydrazine or hydrazine hydrate, are known carcinogens (cancer-causing substances). Many propellant vapors cause eye irritation, even in very small concentration. Inadvertent swallowing of many propellants can also cause severe health degradation.

The inhalation of certain toxic exhaust gases or gaseous or vaporized propellants is perhaps the most common health hazard. It can cause severe damage if the exposure is for long duration or in concentrations that exceed established maximum threshold values. In the United States the **Occupational Safety and Health Administration (OSHA)** has established limits or thresholds on the allowable exposure and concentration for most propellant chemicals**.**

The corrosion, explosion, and fire hazards of many propellants put severe limitations on the materials, the handling, and the design of rocket-propelled vehicles and their engine compartments. Not only is the rocket system itself exposed to the hazardous propellant, but adjacent personnel, structural parts, electrical and other vehicle equipment, and test and launch facilities have to be properly protected against the effects of possible leaks, fumes, and fires or explosions from propellant accumulations

Advantages of liquid propellants

- Usually highest specific impulse; for a fixed propellant mass, this increases the vehicle velocity increment and the attainable mission velocity.
- Can be randomly throttled and randomly stopped and restarted; can be efficiently pulsed (some small thrust sizes over 250,000 times). Thrust-time profile can be randomly controlled; thisallows a reproducible flight trajectory.
- Cutoff impulse can be controllable wth thrust termination device (better control of vehicle terminal velocity).
- Can be largely checked out just prior to operation. Can be tested at full thrust on ground or launch pad prior to flight.
- Can be designed for reuse after field services and checkout.
- Thrust chamber (or some part of the vehicle) can be cooled and made lightweight.
- Storable liquid propellants have been kept in vehicle for more than 20 years and engine can be ready to operate quickly.
- With pumped propulsion feed systems and large total impulse, the inert propulsion system mass
- (including tanks) can be very low (thin tank walls and low tank pressure), allowing a high propellant mass fraction.
- Most propellants have nontoxic exhaust, which is environmentally acceptable.
- Same propellant feed system can supply several thrust chambers in different parts of the vehicle.
- Can modify operating conditions during firing to prevent some failures that would otherwise result in the loss of the mission or vehicle.
- Can provide component redundancy (e.g., dual check valves or extra thrust chamber) to enhance reliability.
- With multiple engines, can design for operation with one or more shutoff (engine out capability).
- The geometry of low-pressure tanks can be designed to fit most vehicles' space constraints (i.e., mounted inside wing or nose cone).
- The placement of propellant tanks within the vehicle can minimize the travel of the center of gravity during powered flight. This enhances the vehicle's flight stability and reduces control forces.

- Plume radiation and smoke are usually low.

Disadvantages:

- Relatively complex design, more parts or components, more things to go wrong.
- Cryogenic propellants cannot be stored for long periods except when tanks are well insulated and escaping vapors are recondensed. Propellant loading occurs at the launch stand and requires cryogenic propellant storage facilities.
- Spills or leaks of several propellants can be hazardous, corrosive, toxic, and cause fires, but this can be minimized with gelled propellants.
- More overall weight for most short-duration, low-total-impulse applications (low propellant mass fraction).
- Nonhypergolic propellants require an ignition system.
- Tanks need to be pressurized by a separate pressurization subsystem. This can require highpressure inert gas storage (2000 to 10,000 psi) for long periods of time.
- More difficult to control combustion instability.
- Bullet impact will cause leaks, sometimes a fire, but usually no detonations; gelled propellants can minimize or eliminate these hazards.
- A few propellants (e.g., red fuming nitric acid) give toxic vapors or fumes.
- Usually requires more volume due to lower average propellant density and the relatively inefficient packaging of engine components.
- If vehicle breaks up and fuel and oxidizer are intimately mixed, it is possible (but not likely) for an explosive mixture to be created.
- Sloshing in tank can cause a flight stability problem, but it can be minimized with baffles.
- If tank outlet is uncovered, aspirated gas can cause combustion interruption or combustion vibration.
- Smoky exhaust (soot) plume can occur with some hydrocarbon fuels.
- Needs special design provisions for start in zero gravity.
- With cryogenic liquid propellants there is a start delay caused by the time needed to cool the system flow passage hardware to cryogenic temperatures.
- Life of cooled large thrust chambers may be limited to perhaps 100 or more starts.
- High-thrust unit requires several seconds to start.

CHAPTER 17

SELECTION OF ROCKET PROPULSION SYSTEMS

With few exceptions, design problems have several possible engineering solutions from which to select. In this chapter we discuss in general terms the process of selecting propulsion systems for a given mission. Three specific aspects are covered in some detail:

1. A comparison of the merits and disadvantages of liquid propellant rocket engines with solid propellant rocket motors.
2. Some key factors used in evaluating particular propulsion systems and selecting from several competing candidate rocket propulsion systems.
3. The interfaces between the propulsion system and the flight vehicle and/or the overall system.

A propulsion system is really a subsystem of a flight vehicle. The vehicle, in turn, can be part of an overall system. An example of an overall system would be a communications network with ground stations, computers, transmitters, and several satellites; each satellite is a flight vehicle and has an attitude-control propulsion system with specific propulsion requirements. The length of time in orbit is a system parameter that affects the satellite size and the total impulse requirement of its propulsion system.

Subsystems of a vehicle system (such as the structure, power supply, propulsion, guidance, control, communications, ground support, or thermal control) often pose conflicting requirements. *Only through careful analyses and system engineering studies is it possible to find compromises that allow all subsystems to operate satisfactorily and be in harmony with each other.* The subject of engineering design has advanced considerably in recent times and general references such as Ref. 17–1 should be consulted for details. Other works address the design of space systems (e.g., Refs. 17–2, 17–3) and the design of liquid propellant engines (e.g., Ref. 17–4).

All mission (overall system), vehicle, and propulsion system *requirements* can be related to either performance, cost, or reliability. For a given mission, one of these criteria is usually more important than the other two. There is a strong interdependence between the three levels of requirements and the three categories of criteria mentioned above. Some of the characteristics of the propulsion system (which is usually a second-tier subsystem) can have a strong influence on the vehicle and vice versa. An improvement in the propulsion performance, for example, can have a direct influence on the vehicle size, overall system cost, or life (which can be translated into reliability and cost).

17.1. SELECTION PROCESS

The selection process is a part of the overall design effort for the vehicle system and its rocket propulsion system. The selection is based on a *series of criteria*, which are based on the requirements and which will be used to evaluate and compare alternate propulsion systems. This process for determining the most suitable rocket propulsion system depends on the application, the ability to express many of the characteristics of the propulsion systems quantitatively, the amount of applicable data that are available, the experience of those responsible for making the selection, and the available time and resources to examine the alternate propulsion systems. What is described here is one somewhat idealized selection process as depicted in Fig. 17–1, but there are alternate sequences and other ways to do this job.

All propulsion selections start with a definition of the overall system and its mission. The mission's objectives, payload, flight regime, trajectory options, launch scenarios, probability of mission success, and other requirements have to be defined, usually by the organization responsible for the overall system. Next, the vehicle has to be defined in conformance with the stated flight application. Only then can the propulsion system requirements be derived for the specific mission and/or vehicle. For example, from the mission requirements it is possible to determine the required mass fraction, the minimum specific impulse, and the approximate total propellant mass, as shown in Chapter 4. Furthermore, this can include propulsion parameters such as thrust–time profile, propellant mass fraction, allowable volume or envelope, typical pulsing duty cycle, ambient temperature limits, thrust vector control needs, vehicle interfaces, likely number of units to be built, prior applicable experience, time schedule requirements, and cost limits.

Since the total vehicle's performance, flight control, operation, or maintenance are usually critically dependent on the performance, control, operation, or maintenance of the rocket propulsion system (and vice versa), the process will usually go through several iterations in defining both the vehicle and propulsion requirements, which are then documented. This iterative process

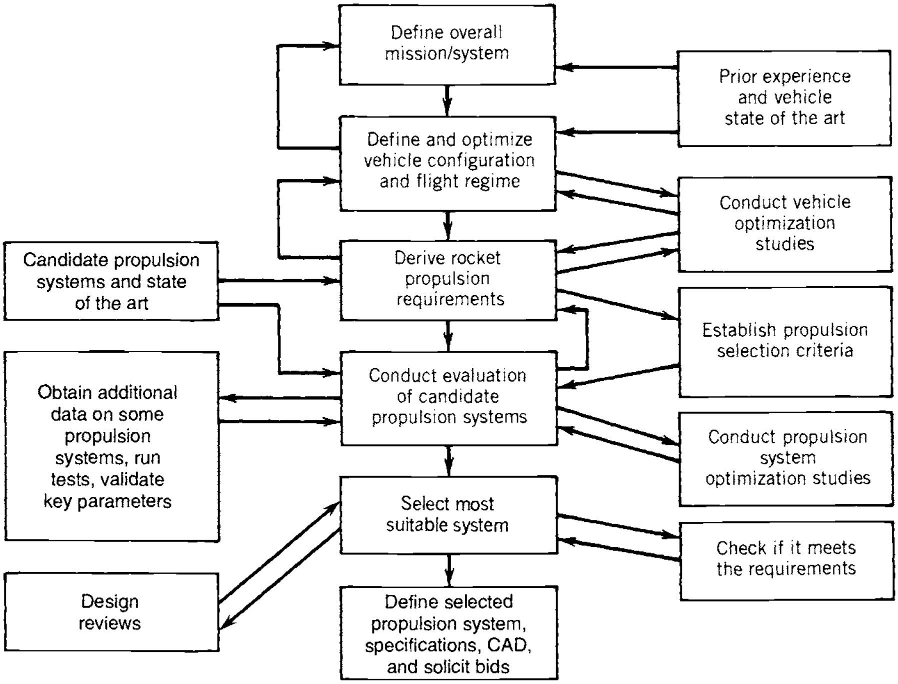


FIGURE 17–1. Idealized process for selecting propulsion systems.

involves both the system organization (or the vehicle/system contractor) and one or more propulsion organizations (or rocket propulsion contractors). Documentation can take many forms; electronic computers have expanded their capability to network, record, and retrieve documents.

A number of competing candidate systems are usually evaluated. They may be proposed by different rocket propulsion organizations, perhaps on the basis of modifications of some existing rocket propulsion system, or may include some novel technology, or may be new types of systems specifically configured to fit the vehicle or mission needs. In making these evaluations it will be necessary to compare several candidate propulsion systems with each other and to rank-order them (in accordance with the selection criteria) on how well they meet each requirement. This requires analysis of each candidate system and also, often, some additional testing. For example, statistical analyses of the functions, failure modes, and safety factors of all key components can lead to quantitative reliability estimates. For some criteria, such as safety or prior related experience, it may not be possible to compare candidate systems quantitatively but only somewhat subjectively.

Various rocket parameters for a particular mission need to be optimized. Trade-off studies are used to determine the number of thrust chambers, engines or motors, optimum chamber pressure, best packaging of the propulsion system(s), optimum mixture ratio, optimum number of stages in a multistage vehicle, best trajectory, optimum nozzle area ratio, number of nozzles, TVC (thrust vector control) concept, optimum propellant mixture ratio or solid propellant formulation, and so on. These trade-off studies are usually aimed at achieving the highest performance, highest reliability, or lowest cost for a given vehicle and mission. Some of these optimizations are needed early in the process to establish propulsion criteria, and some are needed in evaluating competing candidate propulsion systems.

Early in the selection process a tentative recommendation is usually made as to whether the propulsion system should be a solid propellant motor, a liquid propellant engine, an electrical propulsion system, or some other type. Each type has its own regime of thrust, specific impulse, thrust-to-weight ratio (acceleration), or likely duration, as shown in Table 2–1 and Fig. 2–5; these factors are listed for several chemical rocket engines and several types of non-chemical engines. Liquid engines and solid motors are covered in Chapters 6 to 14, hybrids in Chapter 15.

If an existing vehicle is to be upgraded or modified, its propulsion system is usually also improved or modified (e.g., higher thrust, more total impulse, or faster thrust vector control). While there might still be some trade-off studies and optimization of the propulsion parameters that can be modified, one normally does not consider an entirely different propulsion system as is done in an entirely new vehicle or mission. Also, it is rare that an identical rocket propulsion system is selected for two different applications; usually, some design changes and interface modifications are necessary to adapt an existing rocket propulsion system to another application. Proven existing and qualified propulsion systems, that fit the desired requirements, usually have an advantage in cost and reliability.

Electric propulsion systems have a set of unique applications with low thrusts, low accelerations, trajectories exclusively in space, high specific impulse, long operating times, and generally a relatively massive power supply system. They perform well in certain space transfer and orbit maintenance missions. With more electric propulsion systems flying than ever before, the choice of proven electric propulsion thruster types is becoming larger. These systems, together with design approaches, are described in Chapter 19 and Ref. 17–3.

When a chemical rocket is deemed most suitable for a particular application, the selection has to be made between a liquid propellant engine, a solid propellant motor, or a hybrid propulsion system. Some of the major advantages and disadvantages of liquid propellant engines and solid propellant motors are given in Tables 17–1 to 17–4. These lists are general in nature; some items can be controversial, and a number are restricted to particular applications. Items from this list can be transformed into evaluation criteria. For a specific mission, the relevant items on these lists would be rank-ordered in accordance with their relative importance. A quantification of many of the items would be

TABLE 17–1. Solid Propellant Rocket Advantages

Simple design (few or no moving parts).
Easy to operate (little preflight checkout).
Ready to operate quickly.
Will not leak, spill, or slosh.
Sometimes less overall weight for low total impulse application.
Can be throttled or stopped and restarted (a few times) if preprogrammed.
Can provide TVC, but at increased complexity.
Can be stored for 5 to 25 years.
Usually, higher overall density; this allows a more compact package, a smaller vehicle (less drag).
Some propellants have nontoxic, clean exhaust gases, but at a performance penalty.
Some grain and case designs can be used with several nozzles.
Thrust termination devices permit control over total impulse.
Ablation and gasification of insulator, nozzle, and liner materials contribute to mass flow and thus to total impulse.
Some tactical missile motors have been produced in large quantities (over 200,000 per year).
Can be designed for recovery, refurbishing, and reuse (Space Shuttle solid rocket motor).

TABLE 17–2. Liquid Propellant Rocket Advantages

Usually highest specific impulse; for a fixed propellant mass, this increases the vehicle velocity increment and the attainable mission velocity.
Can be randomly throttled and randomly stopped and restarted; can be efficiently pulsed (some small thrust sizes over 250,000 times). Thrust–time profile can be randomly controlled; this allows a reproducible flight trajectory.
Cutoff impulse can be controllable wth thrust termination device (better control of vehicle terminal velocity).
Can be largely checked out just prior to operation. Can be tested at full thrust on ground or launch pad prior to flight.
Can be designed for reuse after field services and checkout.
Thrust chamber (or some part of the vehicle) can be cooled and made lightweight.
Storable liquid propellants have been kept in vehicle for more than 20 years and engine can be ready to operate quickly.
With pumped propulsion feed systems and large total impulse, the inert propulsion system mass (including tanks) can be very low (thin tank walls and low tank pressure), allowing a high propellant mass fraction.
Most propellants have nontoxic exhaust, which is environmentally acceptable.
Same propellant feed system can supply several thrust chambers in different parts of the vehicle.
Can modify operating conditions during firing to prevent some failures that would otherwise result in the loss of the mission or vehicle.
Can provide component redundancy (e.g., dual check valves or extra thrust chamber) to enhance reliability.
With multiple engines, can design for operation with one or more shutoff (engine out capability).
The geometry of low-pressure tanks can be designed to fit most vehicles' space constraints (i.e., mounted inside wing or nose cone).
The placement of propellant tanks within the vehicle can minimize the travel of the center of gravity during powered flight. This enhances the vehicle's flight stability and reduces control forces.
Plume radiation and smoke are usually low.

TABLE 17–3. Solid Propellant Rocket Disadvantages

Explosion and fire potential is larger; failure can be catastrophic; most cannot accept bullet impact or being dropped onto a hard surface.
Many require environmental permit and safety features for transport on public conveyances.
Under certain conditions some propellants and grains can detonate.
Cumulative grain damage occurs through temperature cycling or rough handling; this limits the useful life.
If designed for reuse, it requires extensive factory rework and new propellants.
Requires an ignition system.
Each restart requires a separate ignition system and additional insulation—in practice, one or two restarts.
Exhaust gases are usually toxic for composite propellants containing ammonium perchlorate.
Some propellants or propellant ingredients can deteriorate (self-decompose) in storage.
Most solid propellant plumes cause more radio frequency attenuation than liquid propellant plumes.
Only some motors can be stopped at random, but motor becomes disabled (not reusable).
Once ignited, cannot change predetermined thrust or duration. A moving pintle design with a variety throat area will allow random thrust changes, but experience is limited.
If propellant contains more than a few percent particulate carbon, aluminum, or other metal, the exhaust will be smoky and the plume radiation will be intense.
Integrity of grain (cracks, unbonded areas) is difficult to determine in the field.
Thrust and operating duration will vary with initial ambient grain temperature and cannot be easily controlled. Thus the flight path, velocity, altitude, and range of a motor will vary with the grain temperature.
Large boosters take a few seconds to start.
Thermal insulation is required in almost all rocket motors.
Cannot be tested prior to use.
Needs a safety provision to prevent inadvertent ignition, which would lead to an unplanned motor firing. Can cause a disaster.

TABLE 17–4. Liquid Propellant Rocket Disadvantages

Relatively complex design, more parts or components, more things to go wrong.
Cryogenic propellants cannot be stored for long periods except when tanks are well insulated and escaping vapors are recondensed. Propellant loading occurs at the launch stand and requires cryogenic propellant storage facilities.
Spills or leaks of several propellants can be hazardous, corrosive, toxic, and cause fires, but this can be minimized with gelled propellants.
More overall weight for most short-duration, low-total-impulse applications (low propellant mass fraction).
Nonhypergolic propellants require an ignition system.
Tanks need to be pressurized by a separate pressurization subsystem. This can require high-pressure inert gas storage (2000 to 10,000 psi) for long periods of time.
More difficult to control combustion instability.
Bullet impact will cause leaks, sometimes a fire, but usually no detonations; gelled propellants can minimize or eliminate these hazards.
A few propellants (e.g., red fuming nitric acid) give toxic vapors or fumes.
Usually requires more volume due to lower average propellant density and the relatively inefficient packaging of engine components.
If vehicle breaks up and fuel and oxidizer are intimately mixed, it is possible (but not likely) for an explosive mixture to be created.
Sloshing in tank can cause a flight stability problem, but it can be minimized with baffles.
If tank outlet is uncovered, aspirated gas can cause combustion interruption or combustion vibration.
Smoky exhaust (soot) plume can occur with some hydrocarbon fuels.
Needs special design provisions for start in zero gravity.
With cryogenic liquid propellants there is a start delay caused by the time needed to cool the system flow passage hardware to cryogenic temperatures.
Life of cooled large thrust chambers may be limited to perhaps 100 or more starts.
High-thrust unit requires several seconds to start.

needed. These tables apply to generic rocket propulsion systems; they do not cover systems that use liquid–solid propellant combinations.

A favorite student question has been: Which are better, solid or liquid propellant rockets? A clear statement of strongly favoring one or the other can only be made when referring to a specific set of flight vehicle missions. Today, solid propellant motors seem to be preferred for tactical missiles (air-to-air, air-to-surface, surface-to-air, or short-range surface-to-surface) and ballistic missiles (long- and short-range surface-to-surface) because instant readiness, compactness, and their lack of spills or leaks of hazardous liquids are important criteria for these applications. Liquid propellant engines seem to be preferred for space-launched main propulsion units and upper stages, because of their higher specific impulse, relatively clean exhaust gases, and random throttling capability. They are favored for post-boost control systems and attitude control systems, because of their random multiple pulsing capability with precise cutoff impulse, and for pulsed axial and lateral thrust propulsion on hit-to-kill defensive missiles. However, there are always some exceptions to these preferences.

When selecting the rocket propulsion system for a major new multiyear high-cost project, considerable time and effort are spent in evaluation and in developing rational methods for quantitative comparison. In part this is in response to government policy as well as international competition. Multiple studies are done by competing system organizations and competing rocket propulsion organizations; formal reviews are used to assist in considering all the factors, quantitatively comparing important criteria, and arriving at a proper selection.

17.2. CRITERIA FOR SELECTION

Many criteria used in selecting a particular rocket propulsion system are peculiar to the particular mission or vehicle application. However, some of these selection factors apply to a number of applications, such as those listed in Table 17–5. Again, this list is incomplete and not all the criteria in this table apply to every application. The table can be used as a checklist to see that none of the criteria listed here are omitted.

Here are some examples of important criteria in a few specific applications. For a spacecraft that contains optical instruments (e.g., telescope, horizon seeker, star tracker, or infrared radiation seeker) the exhaust plume must be free of possible contaminants that may deposit or condense on photovoltaic cells, radiators, optical windows, mirrors, or lenses and degrade their performance, and free of particulates that could scatter sunlight into the instrument aperture, which could cause erroneous signals. Conventional composite solid propellants and pulsing storable bipropellants are usually not satisfactory, but cold or heated clean gas jets (H_2, Ar, N_2, etc.) and monopropellant hydrazine reaction gases are usually acceptable. Another example is an emphasis on smokeless propellant exhaust plumes, so as to make visual detection of a smoke or vapor trail very difficult. This applies particularly to tactical missile applications. Only a few solid propellants and several liquid propellants would be truly smokeless and free of a vapor trail under all weather conditions.

Several selection criteria may be in conflict with each other. For example, some propellants with a very high specific impulse are more likely to experience combustion instabilities. In liquid propellant systems, where the oxidizer tank is pressurized by a solid propellant gas generator and where the fuel-rich hot gases are separated by a thin flexible diaphragm from the oxidizer liquid, there is a trade-off between a very compact system and the potential for a damaging system failure (fire, possible explosion, and malfunction of system) if the diaphragm leaks or tears. In electric propulsion, high specific impulse is usually accompanied by heavy power generating and conditioning equipment.

Actual selection will depend on the balancing of the various selection factors in accordance with their importance, benefits, or potential impact on the system, and on quantifying as many of these selection factors as possible through analysis, extrapolation of prior experience/data, cost estimates, weights, and/or separate tests. Design philosophies such as the *Taguchi methodology* and *TQM* (total quality management) can be inferred (Refs. 17–1 and 17–2). Layouts, weight estimates, center-of-gravity analyses, vendor cost estimates, preliminary stress or thermal analysis, and other preliminary design efforts are usually necessary to put numerical values on some of the selection parameters. A comparative examination of the interfaces of alternate propulsion systems is also a part of the process. Some propulsion requirements are incompatible with each other and a compromise has to be made. For example, the monitoring of extra sensors can prevent the occurrence of certain types of failure and thus enhance the propulsion system reliability, yet the extra sensors and control components contribute to the system complexity and their possible failures will reduce the overall reliability. The selection process may also include feedback when the stated propulsion requirements cannot be met or do not make sense, and this can lead to a revision of the initial mission requirements or definition.

Once the cost, performance, and reliability drivers have been identified and quantified, the selection of the best propulsion system for a specified mission proceeds. The final propulsion requirement may come as a result of several iterations and will usually be documented, for example in a *propulsion requirement specification*. A substantial number of records is required (such as engine or motor acceptance documents, CAD (computer-aided design) images, parts lists, inspection records, laboratory test data, etc.). There are many specifications associated with design and manufacturing as well as with vendors, models, and so on. There must also be a disciplined procedure for approving and making design and manufacturing changes. This now becomes the starting point for the design and development of the propulsion system.

TABLE 17.5. Typical Criteria Used in the Selection of a Particular Rocket Propulsion System

Mission Definition

Purpose, function, and final objective of the mission of an overall system are well defined and their implications well understood. There is an expressed need for the mission, and the benefits are evident. The mission requirements are well defined. The payload, flight regime, vehicle, launch environment, and operating conditions are established. The risks, as perceived, appear acceptable. The project implementing the mission must have political, economic, and institutional support with assured funding. The propulsion system requirements, which are derived from mission definition, must be reasonable and must result in a viable propulsion system.

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 (see Ref. 17–5). Benefits of achieving the mission should appear to justify costs. Investment in new facilities should be low. Few, if any, components should require expensive materials. For commercial applications, such as communications satellites, the return on investment must look attractive. No need to hire new, inexperienced personnel, who need to be trained and are more likely to make expensive errors.

System Performance

The propulsion system is designed to optimize vehicle and system performance, using the most appropriate and proven technology. Inert mass is reduced to a practical minimum, using improved materials and better understanding of loads and stresses. Residual (unused) propellant is minimal. Propellants have the highest practical specific impulse without undue hazards, without excessive inert propulsion system mass, and with simple loading, storing, and handling (the specific impulse of the propulsion system is defined in Section 2.1 and is further discussed in Section 19.1). Thrust–time profiles and number of restarts must be selected to optimize the vehicle mission. Vehicles must operate with adequate performance for all the possible conditions (pulsing, throttling, temperature excursions, etc.). Vehicles should be storable 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 the selected 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 personnel injury, damage to equipment, facilities, or the environment is minimal. Certain mishaps or failures will result in a change in the operating condition or the safe shutdown of the propulsion system. Applicable safety standards must be obeyed. Inadvertent energy input to the propulsion system (e.g., bullet impact, external fire) should not result in a detonation. The probability for any such drastic failures should be very low. Safety monitoring and inspections must have proven effective in identifying and preventing a significant share of possible incipient failures (see Ref. 17–6). Adequate safety factors must be included in the design. Spilled liquid propellants should cause no undue hazards. All systems and procedures must conform to the safety standards. Launch test range has accepted the system as being safe enough 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 the overall system is known. There are few complex components. Adequate storage and operating life of components (including propellants) have been demonstrated. Proven ability to check out major part of propulsion system prior to use or launch. If certain likely failures occur, the system must shut down safely. Redundancy of key components should be provided, where effective. High probability that all propulsion functions must be performed within the desired tolerances. Risk of combustion vibration or mechanical vibration should be minimal.

TABLE 17–5 (*Continued*)

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

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

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

Prior Related Experience
There is a favorable history and valid, available, relevant data of similar propulsion systems supporting 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. If applicable, 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 roads or railroads without need for environmental permits and without the need for a decontamination unit and crew to accompany the shipment. Supply of spare parts must be assured. Should be able to operate under certain emergency and overload conditions.

Producibility
Easy to manufacture, inspect, and assemble. All key manufacturing processes are well understood. All materials are well characterized, critical material properties are well known, and the system can be readily inspected. Proven vendors for key components have been qualified. Uses standard manufacturing machinery and relatively simple tooling. Hardware quality and propellant properties must be repeatable. Scrap should be minimal. Designs must make good use of standard materials, parts, common fasteners, and off-the-shelf components. There should be maximum use of existing manufacturing facilities and equipment. Excellent reproducibility, 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 to be realized. R&D, qualification, flight testing, and/or initial operating capability are completed on a preplanned schedule. No unforeseen delays. Critical materials and qualified suppliers must be readily available.

TABLE 17–5 (*Continued*)

Environmental Acceptability

No unacceptable damage to personnel, equipment, or the surrounding countryside. No toxic species in the exhaust plume. No serious damage (e.g., corrosion) due to propellant spills or escaping vapors. Noise in communities close to a test or launch site should remain within tolerable levels. Minimal risk of exposure to cancer-causing chemicals. Hazards must be sufficiently low, so that issues on environmental impact statements are not contentious and approvals by environmental authorities become routine. There should be compliance with applicable laws and regulations. No unfavorable effects from currents generated by an electromagnetic pulse, static electricity, or electromagnetic radiation.

Reusability

Some applications (e.g., Shuttle main engine, Shuttle solid rocket booster, or aircraft rocket-assisted altitude 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 be demonstrated. Fatigue failure and cumulative thermal stress cycles can be critical in some of the 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/or repair of unsatisfactory parts should be readily possible. Number of firings before disassembly should 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, loaded with propellants and pressurizing fluids, can be storable for a required number of years without deterioration or subsequent performance decrease. Interface problems are minimal. Provisions for safe packaging and shipment are available. The system includes features that allow decommissioning (such as to deorbit a spent satellite) or disposal (such as the safe removal and disposal of over-age propellant from a refurbishable rocket motor).

17.3. INTERFACES

In Section 2 of this chapter the interfaces between the propulsion system and the vehicle and/or overall system were identified as some of the criteria to be considered in the selection of a propulsion system. A few rocket propulsion systems are easy to integrate and interface with the vehicles. Furthermore, these interfaces are an important aspect of a disciplined design and development effort. Table 17–6 gives a partial listing of typical interfaces that have been considered in the propulsion system selection, design, and development. It too may be a useful checklist. The interfaces assure system functionality and compatibility between the propulsion system and the vehicle with its other subsystems under all likely operating conditions and mission options. Usually, an interface document or specification is prepared and it is useful to designers, operating personnel, or maintenance people.

Besides cold gas systems, a simple solid propellant rocket motor has the fewest and the least complex set of interfaces. A monopropellant liquid rocket engine also has relatively few and simple interfaces. A solid propellant motor with TVC and a thrust termination capability has additional interfaces, compared to a simple motor. Bipropellant rocket engines are more complex and the

TABLE 17–6. Typical Interfaces between Rocket Propulsion Systems and Flight Vehicle

Interface Category	Typical Detailed Interfaces
Structural	Interface (geometry/location/fastening mechanism) for mounting propulsion system
	Restraints on masses, moments of inertia, or the location of the center of gravity
	Type and degree of damping to minimize vibrations
	Attachment of vehicle components to propulsion system structure, such as wings, electrical components, TVC, or skirts
	Loads (aerodynamic, acceleration, vibrations, thrust, sloshing, dynamic interactions) from vehicle to propulsion system, and vice versa
	Dimensional changes due to loads and/or heating and means for allowing expansions or deflections to occur without overstress
	Interactions from vibration excitation
Mechanical	Interfaces for electric connectors; for pneumatic, hydraulic, propellant pipe connections
	Volume/space available and geometric interference with other subsystems
	Access for assembly, part replacement, inspection, maintenance, repair
	Lifting or handling devices, and lifting attachment locations
	Measurement and adjustment of alignment of fixed nozzles
	Matching of thrust levels when two or more units are fired simultaneously
	Sealing or other closure devices to minimize air breathing and moisture condensation in vented tanks, cases, nozzles, porous insulation, or open pipes
Power	Source and availability of power (usually electric, but sometimes hydraulic or pneumatic) and their connection interfaces
	Identification of all users of power (solenoids, instruments, TVC, igniter, sensors) and their duty cycles. Power distribution to the various users
	Conversion of power to needed voltages, dc/ac, frequencies, or power level
	Electric grounding connections of rocket motors, certain electric equipment or pyrotechnic devices, to minimize voltage buildup and prevent electrostatic discharges
	Shielding of sensitive wires and/or high-voltage components
	Telemetry and radio communications interface

TABLE 17–6 (*Continued*)

Interface Category	Typical Detailed Interfaces
	Heaters (e.g., to keep hydrazine from freezing or to prevent ice formation and accumulation with cryogenic propellants)
	Interfaces with antennas, wiring, sensors, and electronic packages located in the propulsion section of the vehicle
	Thermal management of heat generated in electric components
Propellants	Sharing of propellants between two or more propulsion systems (main thrust chambers and attitude control thrusters)
	Control of sloshing to prevent center of gravity (CG) excursions or to prevent gas from entering the liquid propellant tank outlet
	Design of solid propellant grain or liquid propellant tanks to limit CG travel
	Loading/unloading provisions for liquid propellants
	Access for X-ray inspection of grain for cracks or unbonded areas, while installed
	Access to visually inspect grain cavity for cracks
	Access to inspect cleanliness of tanks, pipes, valves
	Connection of drain pipes for turbopump seal leakage
Vehicle flight control and communications	Command signals (start/stop/throttle, etc.) interface
	Feedback signals (monitoring the status of the propulsion system, e.g., valve positions, thrust level, remaining propellant, pump speed, pressures, temperatures); telemetering devices
	Range safety destruct system
	Attitude control: command actuation in pitch, yaw, or roll; feedback of TVC angle position and slew rate, duty cycle, safety limits
	Division of control logic, computer capability, or data processing and databases between propulsion system controller, vehicle controller, test stand controller, or ground-based computer/controller system
	Number and type of fault detection devices and their connection methods
Thermal	Heat from rocket gas/exhaust plume or aerodynamic airflow will not overheat critical exposed components
	Transfer of heat between propulsion system and the vehicle
	Provisions for venting cryogenic propellant tanks overboard
	Radiators for heat rejection
	Interfaces for cooling, if any

TABLE 17–6. (*Continued*)

Interface Category	Typical Detailed Interfaces
Plume	Radiative and convective heating of vehicle by plume
	Impingement (forces and heating) of plume from attitude control nozzle with vehicle components
	Noise effects on equipment and surrounding areas
	Contamination or condensation of plume species on vehicle or payload parts, such as solar panels, optical components of instruments, or radiation surfaces
	Attenuation of radio signals
Safety	Condition monitoring and sensing of potential imminent failure and automatic remedial actions to prevent or remedy impending failure (e.g., reduce thrust or shut off one of several redundant propulsion systems)
	Arming and disarming of igniter. Access to safe & arm device
	Safe disposal of hazardous liquid propellant leaking through pump shaft seal, valve stem seal, or vented from tanks
	Designed to avoid electrostatic buildup and discharge
Ground support equipment	Interface with standby power system
	Interfaces with heating/cooling devices on ground at launch or test site
	Supply and loading method for liquid propellant, pressurizing gases, and other fluids. Also, interface with method for unloading these
	Electromechanical checkout
	Interface with ground systems for flushing, cleaning, drying the tanks and piping
	Transportation vehicles/boxes/vehicle erection devices
	Lifting devices and handling equipment
	Interface with fire extinguishing equipment on ground

number and difficulty of interfaces increase if they have a turbopump feed system, throttling features, TVC, or pulsing capability. In electric propulsion systems the number and complexity of interfaces is highest for an electrostatic thruster with pulsing capability, when compared to electrothermal systems. More complex electrical propulsion systems generally give higher values of specific impulse. If the mission includes the recovery and reuse of the propulsion system or a manned vehicle (where the crew can monitor and override the propulsion system commands), this will introduce additional interfaces, safety features, and requirements.

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Semester V

AE 2304 Propulsion II

1) What are the differences between air-breathing and Rocket engine?

Air- breathing Engine	Rocket engine
Altitude limitation	No altitude limitation
Rate of climb decreases with altitude	Rate of climb increases with altitude
Thrust decreases with altitude	Thrust increases slightly with altitude
Engine ram drag increases with flight speed.	Engine has no ram drag; constant thrust with speed.

2) What are the factors to be considered in design of a turbine?

a) Shaft rotational speed
b) Gas flow rate
c) Inlet and outlet temperatures
d) Inlet and outlet pressures
e) Exhaust velocity
f) Power output required.

3) Differentiate between impulse stage and reaction stage turbines

Impulse stage	Reaction stage
Expansion of the gas occurs only in the stator nozzles.	Expansion takes place both in stator and rotor.
The rotor blades act as directional vanes to deflect the direction of the flow.	Rotor converts the kinetic energy of the gas into work and contributes a reaction force on the rotor blades.
The relative discharge velocity of the rotor is the same as the relative inlet velocity because there is no net change in pressure between the rotor inlet and rotor exit	The relative discharge velocity of the rotor increases and the pressure decreases in the passages between rotor blades.

4) Define match point.

The match point is defined as the steady-state operating point for a gas turbine when the compressor and turbine are balanced in rotor speed, power, and flow, the operating points at the various power settings defining the operating line for the given engine configuration.

5) Define total-to-total efficiency and state when it is appropriate to use this efficiency.

Total-to-total efficiency is the ratio of the actual work done to the ideal work done corresponding to total inlet and total exit conditions.

$$\eta_{tt} = \frac{T_{01} - T_{03}}{T_{01} - T_{03}''} = \frac{h_{01} - h_{03}}{h_{01} - h_{03}''}$$

For turbojet engines, where the exhaust energy is not a loss because the gases are accelerated to a high velocity for propulsive thrust generation, total-to-total efficiency is used.

6) Define degree of reaction for a turbomachine stage.

The degree of reaction of a turbomachine stage may be defined as the ratio of the static or pressure head change occurring in the rotor to the total change across the stage.

7) What are the materials suitable for use in turbine blades?

Steel , titanium alloys and nickel based alloys are mainly used, have varying proportions of chromium and aluminium to improve the strength and corrosion resistance at high temperatures.

8) What are the different methods of blade cooling?

1) Internal air cooling
   a) Hollow blade, with or without inserts
   b) Solid blade with radial holes, with or without inserts
2) External air cooling
   a) Film cooling
   b) Transpiration or effusion cooling
   c) Root cooling
3) Internal Liquid cooling
   a) Forced convection cooling
   b) Free convection cooling
   c) Evaporative cooling
4) External liquid cooling
   a) Sweat cooling using porous blades
   b) Spray cooling

9) What are the limiting factors in gas turbine design?

Stress considerations , Operating temperatures, Blade fixing, Degree of reaction, Mach number, Outlet flow and Vibrations are some of the limiting factors.

10) What is a free vortex flow ?

A flow in which the whirl or tangential velocity varies inversely with the radius is called as the free vortex flow.

11) What is the function of a turbine ?

A part of the kinetic energy of the expanding gases is extracted by the turbine section, and this energy is transformed into shaft horsepower which is used to drive the compressor and accessories.

12) State the advantages of radial flow turbine.

a) Ruggedness and simplicity
b) Relatively inexpensive and easy to manufacture when compared to the axial-flow turbine

13) Define work ratio.

It is defined as the ratio of actual total head temperature drop to the isentropic temperature drop from the total head inlet to static outlet pressure, and is given by

$$Work\_ratio = \frac{T_{01} - T_{03}}{T_{01} - T_3''}$$

14) What are the three main sources of rotor blade stress.

1) Centrifugal tensile stress
2) The gas bending stress
3) Centrifugal bending stress when the centroid of the blade cross sections at different radii does not lie on a radial line

15) What are the conditions used to determine the steady state engine performance?

1) Continuity of flow and
2) Power balance.

16) What are the advantages of a ramjet engine ?

1) The major advantages of the ramjet engine are low weight, high thrust to weight ratio, practically no moving pars, and a large thrust percent of frontal area make it an attractive engine for satisfying the demand for a relatively low cost air breathing engine having the capability of propelling a missile at supersonic speeds with reasonable fuel consumption.
2) The ramjet engine can be operated with a higher maximum temperature in its thermodynamic cycles.

17) What are the disadvantages of a ramjet engine ?

The most serious disadvantage of the ramjet engine, from a general utilization stand point, is its inability to develop thrust at zero flight speed, so that it must be boosted or launched to a high enough flight speed to become self-sustaining.

18) What are the three distinct operating condition of a ramjet engine ?

a) Critical operation
b) Super-critical operation
c) Sub-critical operation

19) What is critical operation for a ramjet engine?

When the heat released by the burner is of such a magnitude that the back pressure at the exit to the subsonic diffuser causes the normal shock to be positioned at the inlet, then the operation is said to be critical.

20) What is Super-critical operation for a ramjet engine?

When the heat released by the burner cause the back pressure on the exit section of the diffuser to become too small for maintaining the normal shock at the inlet, then the operation is said to be super-critical.

21) What is Sub- critical operation for a ramjet engine ?

If the heat released due to combustion is such that it causes the static pressure at the exit section of the subsonic diffuser to exceed the static pressure which can be accomplished by the diffusion system, then the flow becomes choked and the normal shock wave is expelled from the diffuser and moves upstream toward the vertex of the conical centre body.

22) What are the main objectives of a burner ?

1) To obtain satisfactory combustion of the range of fuel-air ratios required for operating the engine over its flight speed and altitude ranges.
2) To achieve a high combustion efficiency under all operating conditions, especially at the design cruise condition.
3) To effect item 1 & 2 with the smallest possible drop of pressure in the combustion chamber ,and
4) To obtain the largest possible heat release and combustion temperature.

23) What are the factors affecting combustion process in a ramjet?

1) The burner geometry or configuration;
2) The physical and chemical characteristics of the fuel
3) The fuel air ratio
4) The velocity of the working fluid (air and fuel) entering the burner section

24) Name the factors to consider while selecting the fuel for ramjet engine.

1) The calorific value per unit volume
2) Calorific value per pound of air
3) The case with which it can be ignited
4) Physical properties.
5) Storage ability
6) Toxicity

25) What is an integral ram rocket?

A ducted rocket, some times called as an air-augmented rocket, combines the principles of rocket and ramjet engines, it gives higher performance (specific impulse) than a chemical rocket engine, while operating within the earth's atmosphere. Usually the term air-augmented rocket denotes mixing of air with the rocket exhaust (full-rich for after burning) in proportions that enabled the propulsion device to retain the characteristics of the rocket engine, for example, high static thrust and high thrust to weight ratio.

26) What is a rocket motor ?

A rocket motor is a device for converting the thermo chemical energy of one or more propellants into exhaust jet kinetic energy.

27) Explain Under and Over expanded nozzles.

An under expanded nozzle discharges the fluid at an exit pressure greater than external pressure because the exit area is too small for an optimum area ratio. The expansion of the fluid is there fore incomplete within the nozzle and must take place outside. The nozzle exit pressure is higher than the local atmospheric pressure.

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 area ratio.

28) Explain linear aerospike nozzles.

The linear aero spike is an altitude compensation nozzle with a variation of the round axisymmetric aero spike nozzle axis. The flow is turned on a curved contour outer diverging nozzle wall. The nozzle has been shortened and has some internal oblique shock wave losses. The hot gas flow leaving the chamber expands around a central plug.

29) What are the advantages and disadvantages of using multiple nozzles?

If a single large nozzle is replaced by a cluster of smaller nozzles (all the same cumulative thrust), then it is possible to reduce the nozzle length. Similarly, if several replace a single large thrust chamber of a liquid engine smaller thrust chambers, the nozzle length will be shorter, reducing the vehicle length and thus the vehicle structure and inert mass.

The vehicle diameter at the cluster nozzle exit is somewhat larger, the vehicle drag is somewhat higher, and there is additional engine complexity and engine mass.

30) Define effective exhaust velocity.

The effective exhaust velocity ‘c’ is the average equivalent velocity at which propellant is ejected from the vehicle.

It is given in meters per second or feet per second.

31) Define specific impulse.

The thrust per kg of air flow is known as specific thrust or specific impulse. This specific impulse is a criterion of the size of engine required for producing a given total thrust.

32) Define specific propellant consumption.

The specific propellant consumption is the reciprocal of the specific impulse. It is defined as the required propellant weight flow to produce a unit of thrust force in an equivalent rocket.

33) Define mass ratio.

The mass ratio of a vehicle or a particular vehicle stage is defined to be the final mass after rocket operation, after propellants were consumed, divided by the initial mass before rocket operation.

34) Define impulse-to-weight ratio.

The impulse to weight ratio of a complete propulsion system is defined as the total impulse divided by the initial vehicle weight or loaded vehicle weight. A high value indicates an efficient design.

35) Explain the two types of thrust.

The thrust acting on a vehicle is composed of two terms. The first term, the momentum thrust is the product of the propellant mass flow rate and exhaust velocity relative to the vehicle. The second term, the pressure thrust consists of the product of the cross-sectional area of the exhaust jet leaving the vehicle and the difference between the exhaust pressure and the fluid pressure.

36) Define grain configuration.

The shape or geometry of the initial burning surfaces of a grain as it is intended to operate in a motor.

37) What is a cylindrical grain?

A grain in which the internal cross section is constant along the axis regardless of perforation shape.

38) What is Neutral Burning?

Motor burn time during which thrust, pressure, and burning surface area remain approximately constant, typically within about ±15%. Many grains undergo neutral burning.

39) What are perforations?

The central cavity port or flow passage of a propellant grain; its cross section may be a cylinder, a star shape, etc.

40) Define progressive and regressive burning.

Burn time during which thrust, pressure, and burning surface area increases is called progressive burning

Burn time during which thrust, pressure, and burning surface area decrease is called regressive burning.

41) What is a Silver in grain terminology?

Unburned propellant remaining (or lost—that is, expelled through the nozzle) at the time of web burnout is known as silver.

42) Define burning time and action time.

Burning time, Or Effective Burning Time, t_b is the interval from 10% maximum initial pressure (or thrust) to web burnout, with web burnout usually taken as the aft tangent-bisector point on the pressure-time trace.

Action Time, t_a is the burning time plus most of the time to burn silvers; typically, the interval between the initial and final 10% pressure (or thrust) points on the pressure-time trace.

43) What is an Inhibitor?

A layer or coating of slow or non burning material (usually, a polymeric rubber type with filler materials) applied (glued, painted, dipped, or sprayed) to a part of the grain’s propellant surface to prevent burning on that surface. By preventing burning on inhibited surfaces the initial burning area can be controlled and reduced. Also called restrictor.

44) What is a Liner in grain terminology?

A sticky non-self-burning thin layer of polymeric-type material that is applied to the cases prior to casting the propellant n order to promote good bonding between the propellant and the case or the insulator. It also allows some axial motion between the grain periphery and the case.

45) What is an Internal Insulator?

An internal layer between the case and the propellant grain made of an adhesive, thermally insulating material that will not burn readily. Its purpose is to limit the heat transfer to the temperature rise of the case during rocket operation**.**

46) Define Volumetric loading fraction.

Volumetric Loading Fraction, V_f: is the ratio of propellant volume V_b to the chamber volume V_c (excluding nozzle) available for propellant, insulation, and restrictors V_b = m/p:

$$V_f = V_b / V_c = I_t / (I_s \rho_b g_0 V_c)$$

Where I_t is the total impulse, I_s the specific impulse, and ρ_b the propellant density.

47) Define propellant mixture ratio for a bi-propellant.

The propellant mixture ratio for a bipropellant is the ratio at which the oxidizer and fuel are mixed and react to give hot gases. The mixture ratio r is defined as

$$Mixture\ ratio,\quad r = {}^{m}\!/_{m_f} = \frac{Oxidizer\ massflow\ rate}{Fuel\ mass\ flow\ rate}$$

48) Mention some of the factors to be considered in selecting a liquid propellant.

1) Heat of combustion
2) Reaction rate
3) Average propellant density
4) Stability
5) Vapor pressure
6) Freezing point
7) Ignitability
8) Viscosity.
9) Specific Heat
10) Thermal Conductivity

49) What is the function of injectors.

The injector has to introduce and meter the flow of the liquid propellant to the combustion chamber, cause the liquids to be broken up into small droplets (a process called atomization),and mix the propellant in such a manner that the correctly proportionate mixture of fuel and oxidizer will result , with uniform propellant mass flow and composition over the chamber cross section.

50) What is TVC ?

Thrust vector control (TVC) is the internal change of direction of the thrust vector with respect to the symmetry axis of the rocket. By changing the direction of the thrust vector, a control moment about a lateral axis of the rocket can be generated.

51) What are the advantages and disadvantages of solid propellant rocket.

Advantages

1. Simple design (few or no moving parts).
2. Easy to operate (little preflight checkout).
3. Ready to operate quickly.
4. Will not leak, spill or slosh.
5. Thrust termination devices permits control over total

impulse. Disadvantages

1. Explosion and fire potential is large, failure can be catastrophic; most cannot accept bullet impact or being dropped on to a hard surface.
2. Many require environmental permit and safety features for transport on public conveyances.
3. If designed for reuse, it requires extensive factory rework and new propellants.
4. Requires an ignition system.
5. Each restart requires a separate ignition system and additional insulation

52) State the advantages and disadvantages of liquid propellant rocket.

Advantages

1. Highest specific impulse; for a fixed propellant mass
2. Can be largely checked out just prior to operation. Can be tested at full thrust on ground or launch pad prior to flight.
3. Can be designed for reuse after field services and checkout.
4. Thrust chamber (or some part of the vehicle) can be cooled and made light weight.

5. Most propellants have nontoxic exhaust, which is environmentally acceptable

Disadvantages

1. Relatively complex design
2. Tanks need to be pressurized by a separate pressurization subsystem. This can require high pressure inert gas storage (2000 to 10000 psi) for long periods of time.
3. More difficult to control combustion instability.
4. Usually requires more volume due to lower average propellant density and the relatively inefficient packaging of engine components.

53) What is the principle of arc jet ?

. Propellant is heated to high temperature in an electric arc and then expanded in a conventional nozzle. The high-current-density arc discharge is maintained by a sufficient voltage difference between cathode and anode.

54) Explain the principle of ion propulsion.

The principle of electrostatic or ion propulsion is simply the acceleration of charged particles by an electric filed. The propellant source feeds neutral atoms to an ion source and the positive ions generated are accelerated by one or more sets of electrodes which are maintained at zero potential, the ion source being at a high potential. The negative ions or electrons have to be returned eventually to the positively charged exhaust stream to maintain a neutral beam.

55) What is a Solar sail.

Solar sail is a big photon reflection surface. The power source for the sail is the sun and it is external to the vehicle. Approaches using nuclear explosions and pulsed nuclear fusion, has been analyzed. Concepts for transmitting radiation energy from earth stations to satellites have been proposed, but are not yet developed.

56) Explain the principle of electromagnetic thrusters.

According to electro magnetic theory, whenever a conductor carries a current perpendicular to a magnetic field, a body force is exerted on the conductor in a direction at right angles to both the current and the magnetic field. Unlike the ion engine, this acceleration process yields a neutral exhaust beam. Another advantage is the relatively high thrust density, or thrust per unit area, which is normally about 10 to 1000 times of the ion engines.