Module – 1

INTRODUCTION

Syllabus:

Introduction: Space launch Vehicles and military missiles, function, types, role, mission, mission profile, thrust profile, propulsion system, payload, staging, control and guidance requirements, performance measures, design, construction, operation, similarities and differences. Some famous space launch vehicles and strategic missiles.

1.1 Definitions

1.1.1 Rocket and Missile

- A rocket is a self-contained, self-propelled projectile that carries its own supplies of fuel and oxygen.
- The word applies equally to projectiles for military use (bombardment) and civilian use (signaling, lifesaving, fireworks).
- It has, since World War II, been applied only to self-propelled projectiles without onboard guidance systems.
- A rocket designed to be installed in a vehicle (car, aircraft, and spacecraft) as a propulsion system is, technically, a “rocket motor” or a “rocket engine.”
- The word “missile” (for centuries, just a synonym for “projectile”) now refers exclusively to a self-contained, self-propelled projectile with some form of guidance system.
- Missiles are usually powered by rocket motors, but they need not be.

1.1.2 Space Launch Vehicle

- A launch vehicle is a rocket-powered vehicle used to lift satellites and spacecraft into orbit around the Earth.
- A launch vehicle or carrier rocket is a rocket propelled vehicle used to carry a payload from Earth's surface to space, usually to Earth orbit or beyond.
- A launch system includes the launch vehicle, launch pad, vehicle assembly and fueling systems, range safety, and other related infrastructure.

- Most of the launch vehicles used since the beginning of the Space Age in 1957 have been adapted from military missiles, but the term “launch vehicle” also applies to manned cargo carrying spacecraft like the space shuttle.
- The term “booster rocket” is often used by the public and the mainstream press to describe unmanned launch vehicles.
- Professionals tend to avoid it, however, because “booster” also refers to a self-contained, solid-propellant rocket motor used to enhance the thrust of a launch vehicle or missile for specific missions.

1.1.3 Spacecraft

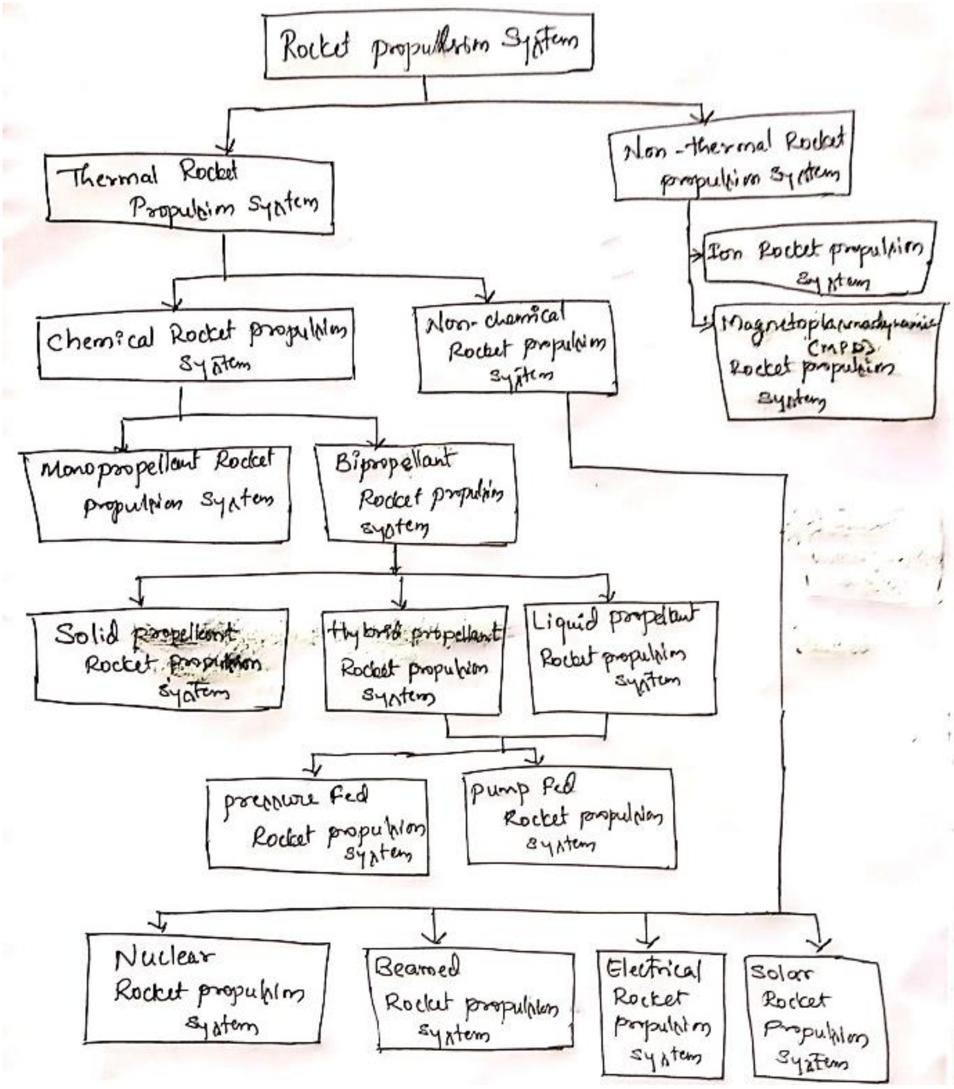
- A spacecraft is a vehicle (with or without a human crew) capable of operating beyond Earth’s atmosphere.
- It may leave Earth under its own power (like the space shuttle) or be carried into space by a launch vehicle (like the Soyuz).
- It is different from a satellite because it can be steered by a human pilot or controllers on the ground.
- Spacecraft need not, in theory, use rocket propulsion. Plans exist for spacecraft propelled by ground-based lasers or “solar sails” designed to catch the streams of charged particles given off by the Sun.
- So far, however, every spacecraft to travel beyond Earth’s atmosphere has done so under rocket power.

1.2 Rocket Propulsion System

- Reaction propulsion/rocket propulsion, wherein both the fuel and the oxidizer, generating the hot gases expended through a nozzle, are carried as part of the rocket engine.
- Rocket propulsion is a class of jet propulsion that produces thrust by ejecting stored matter, called the propellant.
- Rocket propulsion systems can be classified according to
  - The type of energy source (chemical, nuclear, or solar).
  - The basic function (booster stage, sustainer, attitude control, orbit station keeping, etc.).
  - The type of vehicle (aircraft, missile, assisted take-off, space vehicle, etc.).
  - Size.

- ➢ Type of propellant.
- ➢ Type of construction.
- ➢ Number of rocket propulsion units used in a given vehicle.

- Another way is to classify by the method of producing thrust.
- A thermodynamic expansion of a gas is used in the majority of practical rocket propulsion concepts.
- The internal energy of the gas is converted into the kinetic energy of the exhaust flow and the thrust is produced by the gas pressure on the surfaces exposed to the gas.



1.2.1 Chemical Rocket Propulsion

- The energy from a high-pressure combustion reaction of propellant chemicals, usually a fuel and an oxidizing chemical, permits the heating of reaction product gases to very high temperatures (2500 to 4100°C).
- These gases subsequently are expanded in a nozzle and accelerated to high velocities (1800 to 4300 m/sec). Since these gas temperatures are about twice the melting point

of steel, it is necessary to cool or insulate all the surfaces that are exposed to the hot gases.

- According to the physical state of the propellant, there are several different classes of chemical rocket propulsion devices.

1.2.1.1 Liquid Propellant Rocket

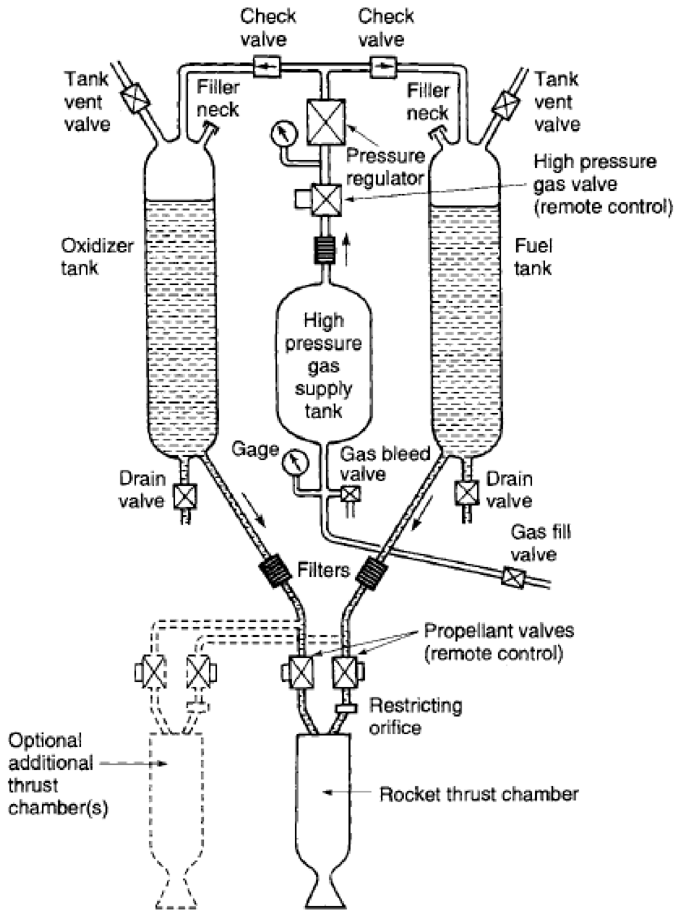


FIGURE 1–3. Schematic flow diagram of a liquid propellant rocket engine with a gas pressure feed system. The dashed lines show a second thrust chamber, but some engines have more than a dozen thrust chambers supplied by the same feed system. Also shown are components needed for start and stop, controlling tank pressure, filling propellants and pressurizing gas, draining or flushing out remaining propellants, tank pressure relief or venting, and several sensors.

- Liquid propellant rocket engines use liquid propellants that are fed under pressure from tanks into a thrust chamber.
- The liquid bipropellant consists of a liquid oxidizer (e.g., liquid oxygen) and a liquid fuel (e.g., kerosene).
- A monopropellant is a single liquid that contains both oxidizing and fuel species; it decomposes into hot gas when properly catalyzed.
- Gas pressure feed systems are used mostly on low thrust, low total energy propulsion systems, such as those used for attitude control of flying vehicles, often with more than one thrust chamber per engine.

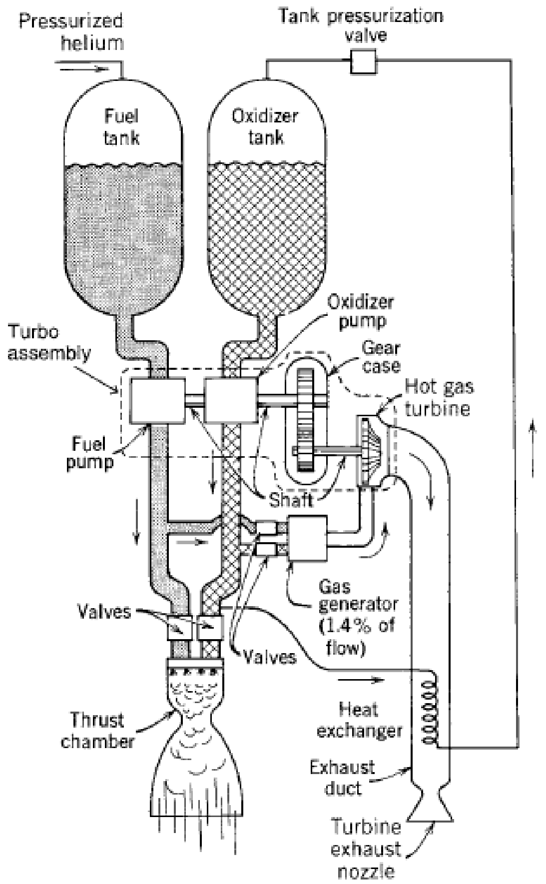


FIGURE 1–4. Simplified schematic diagram of one type of liquid propellant rocket engine with a turbopump feed system and a separate gas generator, which generates warm gas for driving the turbine. Not shown are components necessary for controlling the operation, filling, venting, draining, or flushing out propellants, filters or sensors. The turbopump assembly consists of two propellant pumps, a gear case, and a high speed turbine.

- Pump-fed liquid rocket systems are used typically in applications with larger amounts of propellants and higher thrusts, such as in space launch vehicles.
- Some liquid rocket engines permit repetitive operation and can be started and shut off at will.
- If the thrust chamber is provided with adequate cooling capacity, it is possible to run liquid rockets for periods exceeding 1 hour, dependent only on the propellant supply.
- A liquid rocket propulsion system requires several precision valves and a complex feed mechanism which includes propellant pumps, turbines, or a propellant-pressurizing device, and a relatively intricate combustion or thrust chamber.

1.2.1.2 Solid Propellant Rocket Motors

- In solid propellant rocket motors the propellant to be burned is contained within the combustion chamber or case.
- The solid propellant charge is called the grain and it contains all the chemical elements for complete burning.

- Once ignited, it usually burns smoothly at a predetermined rate on all the exposed internal surfaces of the grain.
- Initial burning takes place at the internal surfaces of the cylinder perforation and the four slots.
- The internal cavity grows as propellant is burned and consumed.
- The resulting hot gas flows through the supersonic nozzle to impart thrust.
- Once ignited, the motor combustion proceeds in an orderly manner until essentially all the propellant has been consumed.
- There are no feed systems or valves combustion or thrust chamber.

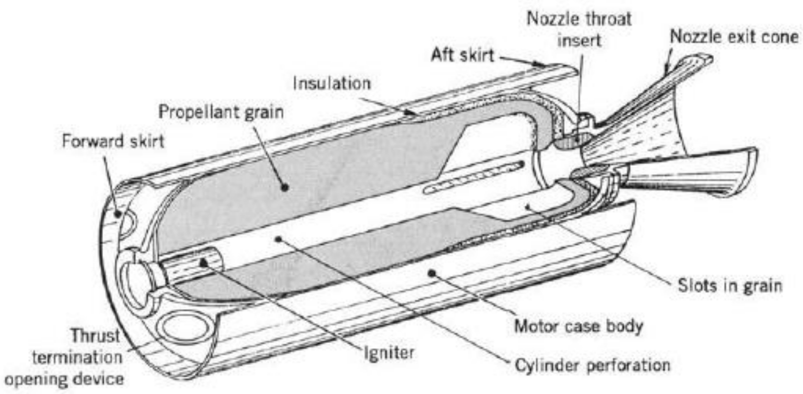
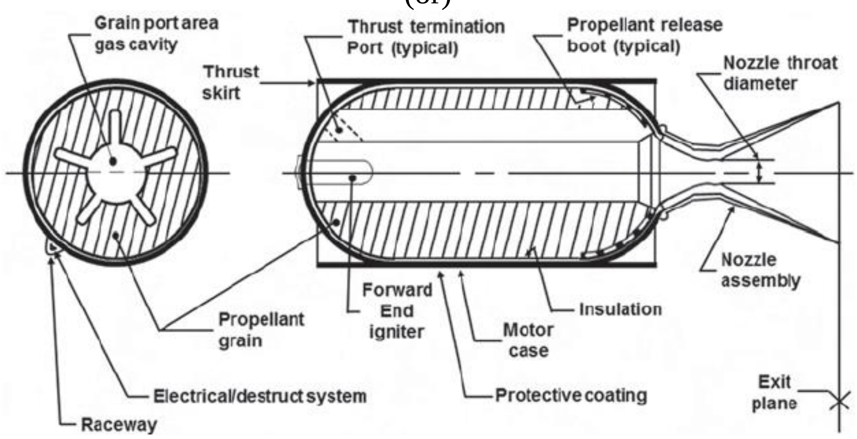


FIGURE 1–5. Simplified perspective three-quarter section of a typical solid propellant rocket motor with the propellant grain bonded to the case and the insulation layer and with a conical exhaust nozzle. The cylindrical case with its forward and aft hemispherical domes form a pressure vessel to contain the combustion chamber pressure. Adapted with permission from Reference 11–1.

(or)



1.2.1.3 Gaseous propellant rocket engines

- Gaseous propellant rocket engines use a stored high-pressure gas, such as air, nitrogen, or helium, as their working fluid or propellant.
- The stored gas requires relatively heavy tanks.
- These cold gas engines have been used on many early space vehicles as attitude control systems and some are still used today.

- Heating the gas by electrical energy or by combustion of certain monopropellants improves the performance and this has often been called warm gas propellant rocket propulsion.

1.2.1.4 Hybrid Propellant Rocket Propulsion Systems

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

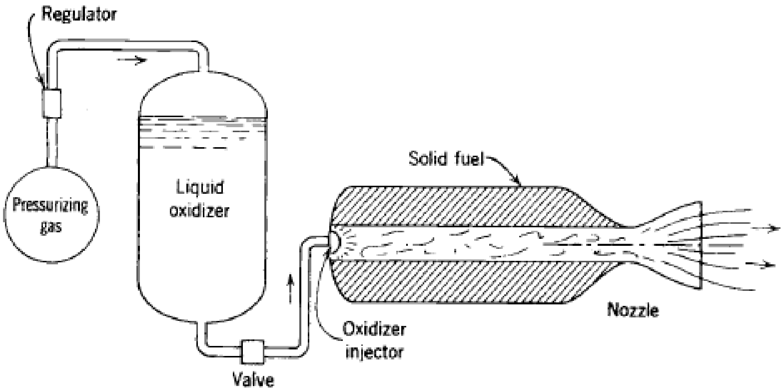


FIGURE 1–6. Simplified schematic diagram of a typical hybrid rocket engine. The relative positions of the oxidizer tank, high pressure gas tank, and the fuel chamber with its nozzle depend on the particular vehicle design.

1.2.1.5 Combinations of Ducted Jet Engines and Rocket Engines

- The Tomahawk surface-to-surface missile uses two stages of propulsion in sequence. The solid propellant rocket booster lifts the missile away from its launch platform and is discarded after its operation. A small turbojet engine sustains the low level flight at nearly constant speed toward the target.
- A ducted rocket, sometimes called 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.
- The principles of the rocket and ramjet can be combined so that the two propulsion systems operate in sequence and in tandem and yet utilize a common combustion chamber volume.
- A solid fuel ramjet uses a grain of solid fuel that gasifies or ablates and reacts with air. Good combustion efficiencies have been achieved with a patented boron-containing solid fuel fabricated into a grain similar to a solid propellant and burning in a manner similar to a hybrid rocket propulsion system.

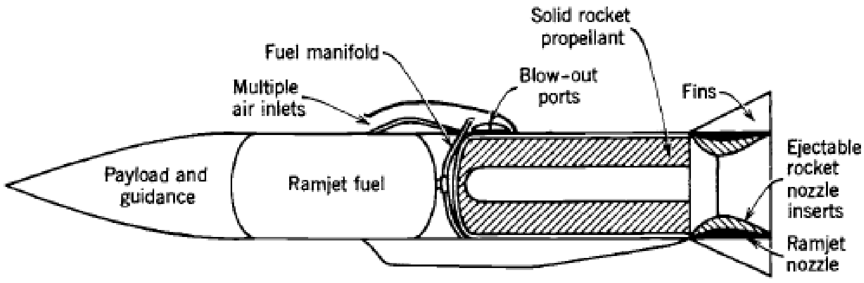
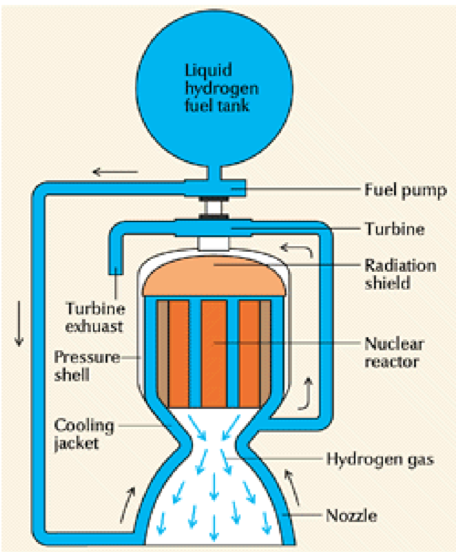


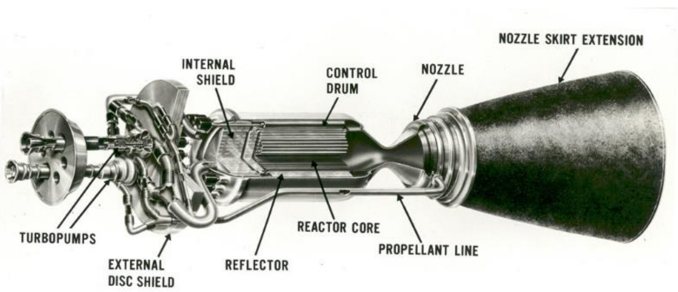
FIGURE 1–7. Elements of an air-launched missile with integral rocket–ramjet propulsion. After the solid propellant has been consumed in boosting the vehicle to flight speed, the rocket combustion chamber becomes the ramjet combustion chamber with air burning the ramjet liquid fuel.

1.2.2 Non-Chemical Rocket Propulsion

1.2.2.1 Nuclear Rocket Engines

- Three different types of nuclear energy sources have been investigated for delivering heat to a working fluid, usually liquid hydrogen, which subsequently can be expanded in a nozzle and thus accelerated to high ejection velocities (6000 to 10,000 m/sec).
- However, none can be considered fully developed today and none have flown.
- They are the fission reactor, the radioactive isotope decay source, and the fusion reactor.
- All three types are basically extensions of liquid propellant rocket engines. The heating of the gas is accomplished by energy derived from transformations within the nuclei of atoms. In chemical rockets the energy is obtained from within the propellants, but in nuclear rockets the power source is usually separate from the propellant.



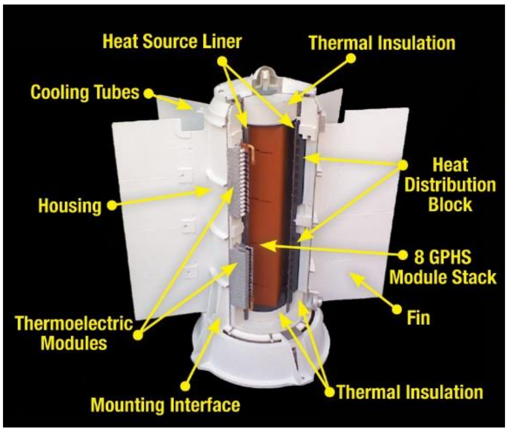


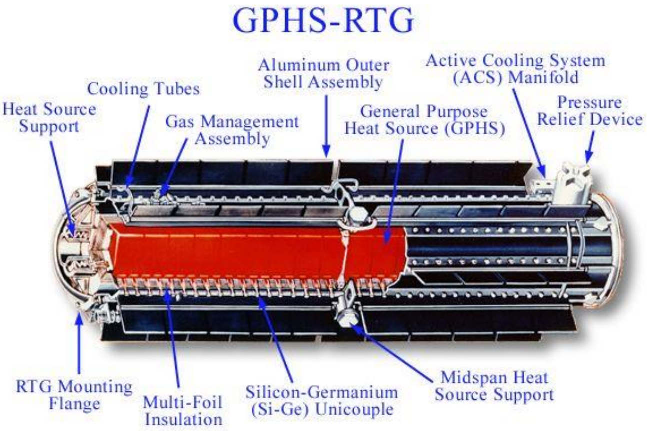
Fission:

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

Isotope Decay Engine:

- In the isotope decay engine a radioactive material gives off radiation, which is readily converted into heat.
- Isotope decay sources have been used successfully for generating electrical power in space vehicles and some have been flown as a power supply for satellites and deep space probes.
- The released energy can be used to raise the temperature of a propulsive working fluid such as hydrogen or perhaps drive an electric propulsion system.
- It provides usually a lower thrust and lower temperature than the other types of nuclear rocket.
- As yet, isotope decay rocket engines have not been developed or flown.





Fusion:

- Fusion is the third nuclear method of creating nuclear energy that can heat a working fluid.
- A number of different concepts have been studied. To date none have been tested and many concepts are not yet feasible or practical.
- Concerns about an accident with the inadvertent spreading of radioactive materials in the earth environment and the high cost of development programs have to date prevented a renewed experimental development of a large nuclear rocket engine.
- Unless there are some new findings and a change in world attitude, it is unlikely that a nuclear rocket engine will be developed or flown in the next few decades.

1.2.2.2 Electric Rocket Propulsion

- In all electric propulsion the source of the electric power (nuclear, solar radiation receivers, or batteries) is physically separate from the mechanism that produces the thrust.
- This type of propulsion has been handicapped by heavy and inefficient power sources.
- The thrust usually is low, typically 0.005 to 1 N.
- In order to allow a significant increase in the vehicle velocity, it is necessary to apply the low thrust and thus a small acceleration for a long time (weeks or months).
- Three types:
  - ➢ Electrothermal rocket propulsion – thermodynamic expansion.
  - ➢ Electrostatic or Ion propulsion engine – Non-thermodynamic expansion - works in vacuum.
  - ➢ Electromagnetic or Magnetoplasma engine - Non-thermodynamic expansion – works in vacuum.

Electrothermal Rocket Propulsion:

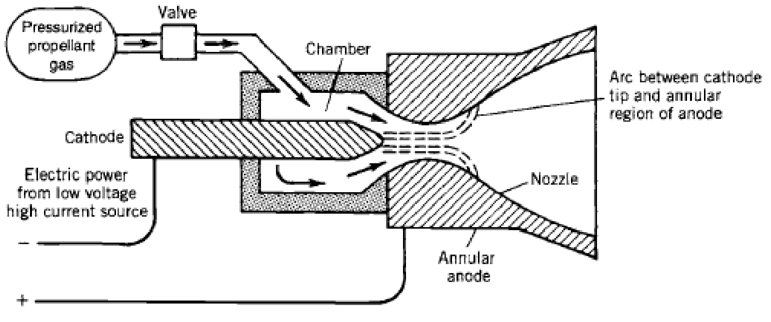


FIGURE 1–8. Simplified schematic diagram of arc-heating electric rocket propulsion system. The arc plasma temperature is very high (perhaps 15,000 K) and the anode, cathode, and chamber will get hot (1000 K) due to heat transfer.

- Electrothermal rocket propulsion most resembles the previously mentioned chemical rocket units; propellant is heated electrically (by heated resistors or electric arcs) and the hot gas is then thermodynamically expanded and accelerated to supersonic velocity through an exhaust nozzle.
- These electrothermal units typically have thrust ranges of 0.01 to 0.5 N, with exhaust velocities of 1000 to 5000 m/sec, and ammonium, hydrogen, nitrogen, or hydrazine decomposition product gases have been used as propellants.

Ion Rocket (Non-Thermal Rocket Propulsion):

- In an ion rocket a working fluid (typically, xenon) is ionized (by stripping off electrons) and then the electrically charged heavy ions are accelerated to very high velocities (2000 to 60,000 m/sec) by means of electrostatic fields.
- The ions are subsequently electrically neutralized; they are combined with electrons to prevent the buildup of a space charge on the vehicle.

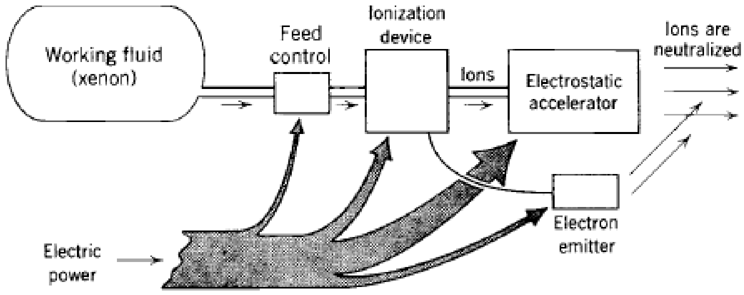


FIGURE 1–9. Simplified schematic diagram of a typical ion rocket, showing the approximate distribution of the electric power.

Magnetoplasma Rocket (Non-Thermal Rocket Propulsion):

- In the magnetoplasma rocket an electrical plasma (an energized hot gas containing ions, electrons, and neutral particles) is accelerated by the interaction between electric currents and magnetic fields and ejected at high velocity (1000 to 50,000 m/sec).

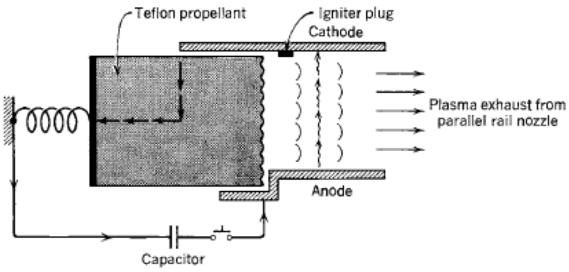


FIGURE 1–10. Simplified diagram of a rail accelerator for self-induced magnetic acceleration of a current-carrying plasma. When the capacitor is discharged, an arc is struck at the left side of the rails. The high current in the plasma arc induces a magnetic field. The action of the current and the magnetic field causes the plasma to be accelerated at right angles to both the magnetic field and the current, namely in the direction of the rails. Each time the arc is created a small amount of solid propellant (Teflon) is vaporized and converted to a small plasma cloud, which (when ejected) gives a small pulse of thrust. Actual units can operate with many pulses per second.

- There are many different types and geometries. A simple pulsed (not continuously operating) unit with a solid propellant is shown in above figure.
- This type has had a good flight record as a spacecraft attitude control engine.

1.2.2.3 Solar Rocket

- Several technologies exist for harnessing solar energy to provide the power for spacecraft and also to propel spacecraft using electrical propulsion.
- Solar cells generate electric power from the sun's radiation. They are well developed and have been successful for several decades.
- Most electric propulsion systems have used solar cells for their power supply.

Solar Thermal Rocket:

- An attractive concept, the solar thermal rocket, has large diameter optics to concentrate the sun's radiation (e.g., by lightweight precise parabolic mirrors or Fresnel lenses) onto a receiver or optical cavity.

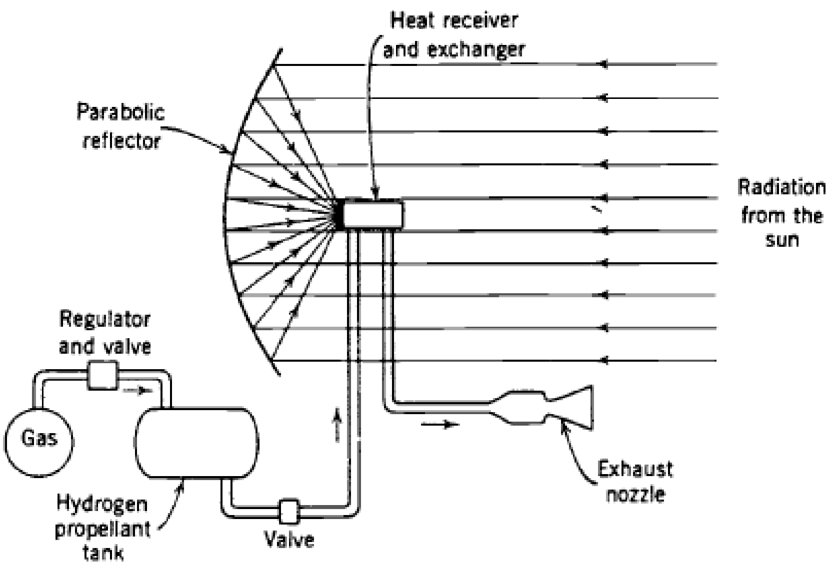


FIGURE 1-11. Simplified schematic diagram of a solar thermal rocket concept.

- The receiver is made of high temperature metal (such as tungsten or rhenium) and has a cooling jacket or heat exchanger.
- It heats a working fluid, usually liquid hydrogen, up to perhaps 2500°C and the hot gas is controlled by hot gas valves and exhausted through one or more nozzles.
- The large mirror has to be pointed toward the sun and this requires the mirror to be adjustable in its orientation.
- Performance can be two to three times higher than that of a chemical rocket and thrust levels in most studies are low (1 to 10 N).

Solar Sail:

- The solar sail is another concept. It is basically a big photon reflector surface.
- The power source for the solar sail is the sun and it is external to the vehicle.
- Approaches using nuclear explosions and pulsed nuclear fusion have been analyzed, but are not yet feasible.
- Concepts for transmitting radiation energy (by lasers or microwaves) from earth stations to satellites have been proposed, but are not yet developed.

1.3 Applications of Rockets

- Space Launch Vehicles or space boosters
- Spacecraft
- Missiles
- Primary engines for research airplanes
- Assist-take-off rockets for airplanes
- Ejection of crew escape capsules and stores
- Personnel "propulsion belts”
- Propulsion for target drones
- Weather sounding rockets
- Signal rockets
- Decoy rockets
- Spin rockets
- Vernier rockets
- Underwater rockets for torpedoes and missiles
- The throwing of lifelines to ships, and "Fourth of July" rockets

1.4 Definitions and Fundamentals/Performance Parameters

- Rocket propulsion is an exact but not a fundamental subject, and there are no basic scientific laws of nature peculiar to propulsion. The basic principles are essentially those of mechanics, thermodynamics, and chemistry.
- Propulsion is achieved by applying a force to a vehicle that is, accelerating the vehicle or, alternatively, maintaining a given velocity against a resisting force. This propulsive force is obtained by ejecting propellant at high velocity.

Total impulse (I_t)

The thrust force F integrated over the burning time t.

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

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

Specific impulse (I_s)

The total impulse per unit weight of propellant

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

g_0 - st. accⁿ of gravity @ sea level.

$\dot{m}$ - total mass flow rate

Mass Ratio (MR)

The final mass m_f (after rocket operation has consumed all usable propellant) divided by m_0 (before rocket operation)

$$MR = m_f / m_0$$

Total effective propellant weight (W)

The product of m_p and g_0

m_p - total effective propellant mass.

Weight flow rate ($\dot{w}$)

The product of $\dot{m}$ and g_0

$g_0 = 9.8066\, m/s^2$

Effective exhaust velocity (c)

Average equivalent velocity at which propellant is ejected from the vehicle.

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

Propellant mass fraction (ζ)

The fraction of propellant mass m_p in an initial mass m_0.

$$\zeta = m_p / m_0 = m_0 - m_f / m_0 = m_p / (m_p + m_f)$$

$$m_0 = m_f + m_p.$$

Impulse to weight ratio

The total impulse I_t divided by the initial propellant loaded vehicle weight w_0.

$$\frac{I_t}{w_0} = \frac{I_t}{(m_f + m_p) g_0} = \frac{I_s}{\frac{m_f}{m_p} + 1}$$

Thrust to weight ratio $\frac{F}{w_0}$

The acceleration that the engine is capable of giving to its own loaded propulsion system mass.

1.4.1 Thrust

The thrust is the force produced by a rocket propulsion system acting upon a vehicle. In a simplified way, it is the reaction experienced by its structure due to the ejection of matter at high velocity.

Rocket propulsion differs from these devices primarily in the relative magnitude of the accelerated masses and velocities. In rocket propulsion relatively small masses are involved which are carried within the vehicle and ejected at high velocities.

The thrust, due to a change in momentum, is given below. The thrust and the mass flow are constant and the gas exit velocity is uniform and axial. This force represents the total propulsion force when the nozzle exit pressure equals the ambient pressure.

$$F = \frac{dm}{dt} v_2 = \dot{m} v_2 = \frac{\dot{w}}{g_0} v_2$$

The pressure of the surrounding fluid (i.e., the local atmosphere) gives rise to the second contribution that influences the thrust. Figure 2-1 shows schematically the external pressure acting uniformly on the outer surface of a rocket chamber and the gas pressures on the inside of a typical thermal rocket engine.

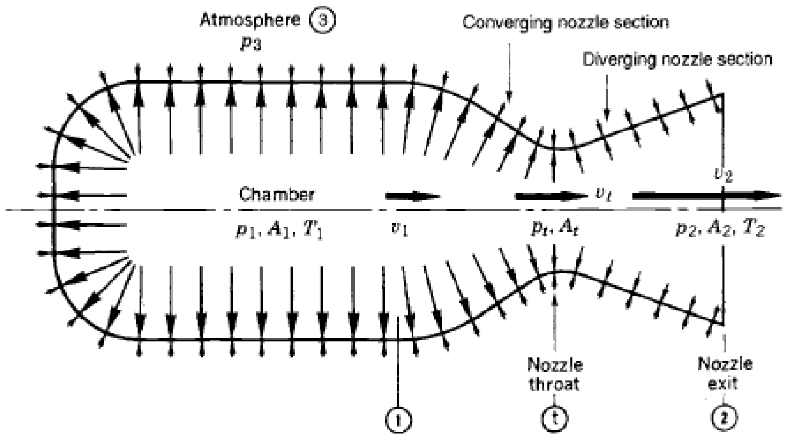


FIGURE 2–1. Pressure balance on chamber and nozzle interior walls is not uniform. The internal gas pressure (indicated by length of arrows) is highest in the chamber (p_1) and decreases steadily in the nozzle until it reaches the nozzle exit pressure p_2. The external or atmospheric pressure p_3 is uniform. At the throat the pressure is p_t. The four subscripts (shown inside circles) refer to the quantities A, v, T, and p at specific locations.

The size of the arrows indicates the relative magnitude of the pressure forces. The axial thrust can be determined by integrating all the pressures acting on areas that can be projected on a plane normal to the nozzle axis. The forces acting radially outward are appreciable, but do not contribute to the axial thrust because a rocket is typically an axially symmetric chamber. The conditions prior to entering the nozzle are essentially stagnation conditions. Because of a fixed nozzle geometry and changes in ambient pressure due to variations in altitude, there can be an imbalance of the external environment or atmospheric pressure P_3 and the local pressure P_2 of the hot gas jet at the exit plane of the nozzle. Thus, for a steadily operating rocket propulsion system moving through a homogeneous atmosphere, the total thrust is equal to

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

The first term from above equation is the momentum thrust represented by the product of the propellant mass flow rate and its exhaust velocity relative to the vehicle. The second term represents the pressure thrust consisting of the product of the cross-sectional area at the nozzle exit A_2 (where the exhaust jet leaves the vehicle) and the difference between the exhaust gas pressure at the exit and the ambient fluid pressure.

If the exhaust pressure is less than the surrounding fluid pressure, the pressure thrust is negative. Because this condition gives a low thrust and is undesirable, the rocket nozzle is usually so designed that the exhaust pressure is equal or slightly higher than the ambient fluid pressure. When the ambient atmosphere pressure is equal to the exhaust pressure, the pressure term is zero and the thrust is the same as:

$$F = \frac{dm}{dt} v_2 = \dot{m} v_2 = \frac{\dot{w}}{g_0} v_2$$

In the vacuum of space $P_3 = 0$ and the thrust becomes:

$$F = \dot{m} v_2 + p_2 A_2$$

The pressure condition in which the exhaust pressure is exactly matched to the surrounding fluid pressure ($P_2 = P_3$) is referred to as the rocket nozzle with optimum expansion ratio.

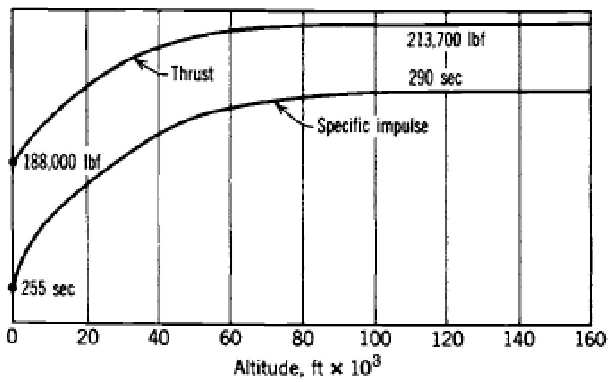


FIGURE 2–2. Altitude performance of RS 27 liquid propellant rocket engine used in early versions of the Delta launch vehicle.

1.4.2 Exhaust Velocity

The effective exhaust velocity is applies to all rockets that thermodynamically expand hot gas in a nozzle and, indeed, to all mass expulsion systems. From thrust equation and for constant propellant mass flow this can be modified to:

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

When $P_2 = P_3$, the effective exhaust velocity c is equal to the average actual exhaust velocity of the propellant gases v_2. When $P_2 \neq P_3$ then $c \neq v_2$. The second term of the right hand side from above equation is usually small in relation to v_2; thus the effective exhaust velocity is usually close in value to the actual exhaust velocity. When $c = v_2$ the thrust can be rewritten as:

$$F = (\dot{w}/g_0)v_2 = \dot{m}c$$

The characteristic velocity has been used frequently in the rocket propulsion literature. Its symbol c*, pronounced "cee-star," is defined as:

$$c^* = p_1 A_t / \dot{m}$$

The characteristic velocity c* is used in comparing the relative performance of different chemical rocket propulsion system designs and propellants.

1.4.3 Example Problems

Example 2–1. A rocket projectile has the following characteristics:

Initial mass	200 kg
Mass after rocket operation	130 kg
Payload, nonpropulsive structure, etc.	110 kg
Rocket operating duration	3.0 sec
Average specific impulse of propellant	240 sec

Determine the vehicle's mass ratio, propellant mass fraction, propellant flow rate, thrust, thrust-to-weight ratio, acceleration of vehicle, effective exhaust velocity, total impulse, and the impulse-to-weight ratio.

SOLUTION. Mass ratio of vehicle (Eq. 2–8) $\mathbf{MR} = m_f/m_0 = 130/200 = 0.65$; mass ratio of rocket system $\mathbf{MR} = m_f/m_0 = (130 - 110)/(200 - 110) = 0.222$. Note that the empty and initial masses of the propulsion system are 20 and 90 kg, respectively.

The propellant mass fraction (Eq. 2–9) is

$$\zeta = (m_0 - m_f)/m_0 = (90 - 20)/90 = 0.778$$

The propellant mass is $200 - 130 = 70$ kg. The propellant mass flow rate is $\dot{m} = 70/3 = 23.3$ kg/sec,

The thrust (Eq. 2–5) is

$$F = I_s \dot{w} = 240 \times 23.3 \times 9.81 = 54{,}857 \text{ N}$$

The thrust-to-weight ratio of the vehicle is

$$\begin{aligned} &\text{initial value } F/w_0 = 54{,}857/(200 \times 9.81) = 28 \\ &\text{final value } 54{,}857/(130 \times 9.81) = 43 \end{aligned}$$

The maximum acceleration of the vehicle is $43 \times 9.81 = 421$ m/sec^2. The effective exhaust velocity (Eq. 2–6) is

$$c = I_s g_0 = 240 \times 9.81 = 2354 \text{ m/sec}$$

The total impulse (Eqs. 2–2 and 2–5) is

$$I_t = I_s w = 240 \times 70 \times 9.81 = 164{,}808 \text{ N-sec}$$

This result can also be obtained by multiplying the thrust by the duration. The impulse-to-weight ratio of the propulsion system (Eq. 2–11) is

$$I_t/w_0 = 164{,}808/[(200 - 110)9.81] = 187$$

Example 2–2. 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	0.0855 m
Nozzle exit diameter	0.2703 m

Determine $\dot{m}$, v_2, c^*, c, and I_s at sea level, and c and I_s at 1000 and 25,000 m altitude. Assume an invariant thrust and mass flow rate and negligible short start and stop transients.

SOLUTION. The mass flow rate $\dot{m}$ is determined from the total propellant used (initial motor mass − final motor mass) and the burn time.

$$\dot{m} = (1210 - 215)/40 = 24.9 \text{ kg/sec}$$

The nozzle areas at the throat and exit are

$$A_t = \pi D^2/4 = \pi \times 0.0855^2/4 = 0.00574 \text{ m}^2$$
$$A_2 = \pi D^2/4 = \pi \times 0.2703^2/4 = 0.0574 \text{ m}^2$$

Equation 2–14 is to be solved for v_2, the actual average exhaust velocity.

$$\begin{aligned} v_2 &= F/\dot{m} - (p_2 - p_3)A_2/\dot{m} \\ &= 62{,}250/24.9 - (0.070 - 0.1013)10^6 \times 0.0574/24.9 \\ &= 2572 \text{ m/sec} \end{aligned}$$

The characteristic velocity and effective exhaust velocity are found from Eqs. 2–6 and 2–18 for sea level conditions.

$$\begin{aligned} c^* &= p_1 A_t/\dot{m} = 7.00 \times 10^6 \times 0.00574/24.9 = 1613 \text{ m/sec} \\ I_s &= F/\dot{m}g_0 = 62{,}250/(24.9 \times 9.81) = 255 \text{ sec} \\ c &= I_s g_0 = 255 \times 9.81 = 2500 \text{ m/sec} \end{aligned}$$

For altitudes of 1000 and 25,000 m the ambient pressure (see Appendix 2) is 0.0898 and 0.00255 MPa. From Eq. 2–16 the altitude values of c can be obtained.

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

At 1000 m altitude,

$$\begin{aligned} c &= 2572 + (0.070 - 0.0898) \times 10^6 \times 0.0574/24.9 = 2527 \text{ m/sec} \\ I_s &= 2527/9.81 = 258 \text{ sec} \end{aligned}$$

At 25,000 m altitude,

$$\begin{aligned} c &= 2572 + (0.070 - 0.00255) \times 10^6 \times 0.0574/24.9 = 2727 \text{ m/sec} \\ I_s &= 2727/9.80 = 278 \text{ sec} \end{aligned}$$

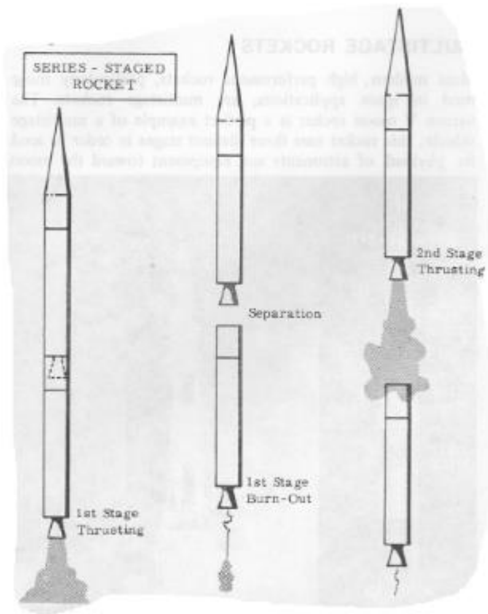
1.5 Multistage/Multistep/Multistate Rocket

- A multistage or multistep rocket is a series of individual vehicles or stages each with its own structure, tanks and engines.

- The stages are so connected that each operates in turn accelerating the remaining stages and the payload before being detached from them.
- Multistage rocket vehicles permit higher vehicle velocities, more payloads for space vehicle and ballistic missile.
- After the useful propellant is fully consumed in a particular stage, the remaining empty mass of that expanded stage is dropped from the vehicle and the operation of the propulsion system of the near step or stage is started.
- The last at top stage which is usually the smallest carries the payload.
- The empty mass of expanded stage or step is separated from the remainder of the vehicle, because it avoids the expenditure of additional energy for faster accelerating a useless mass.
- The first or lowest stage, often called a booster stage is the longest and it requires the largest thrust and largest total impulse.
- The thrust usually become smaller with each subsequent stage also known as upper stage or sustainer stage.
- The spacecraft is that part of a launch vehicle that carries the payload. It is the only part of the vehicle that goes into orbit or deep space.

1.5.1 Types of Multistage Rockets

1.5.1.1 Serial Staging

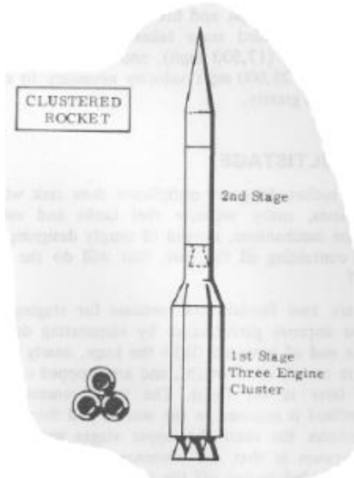


- Small, second stage rocket that is placed on top of a larger first stage rocket.
- The first stage is ignited at launch and burns through the powered ascent until its propellants are exhausted.

- The first stage engine is then extinguished, the second stage separates from the first stage, and the second stage engine is ignited.
- The payload is carried a top the second stage into orbit.

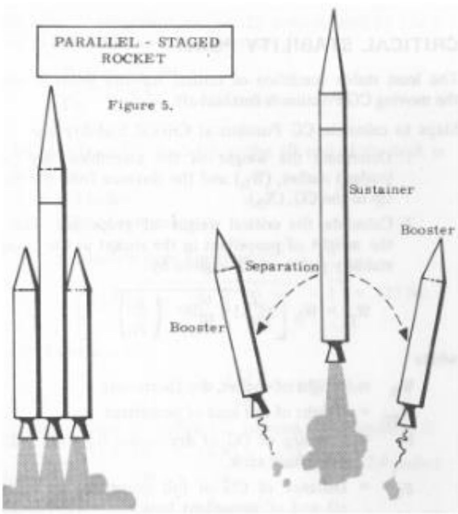
1.5.1.2 Clustered Rocket

- A popular method for producing a large first stage has been cluster several rockets together to provide greater combined thrust without actually having to build a large rocket.
- To form high thrust first stages by strapping or attaching them together.



1.5.1.3 Parallel Staging

- Several small first stages are strapped onto to a central sustainer rocket.
- At launch, all of the engines are ignited.



- When the propellants in the strap-on's are extinguished, the strap-on rockets are discarded.
- The sustainer engine continues a top the sustainer rocket into orbit.

- Used on the space shuttle.
- The discarded solid rocket boosters are retrieved from the ocean, re-filled with propellant, and used again on the shuttle.

1.5.2 Advantages of Staging

- Once the fuel is exhausted, the space and structure which contained it and the motors themselves itself are dropped being no longer useful. Thus rocket lightens itself.
- The thrust of next stages is able to provide more acceleration than if the earlier stage were till affected.
- Rocket performance is improved by eliminating dead weight.
- The capability of rocket to reduce thrust in midflight thus avoids severe acceleration from both men and instruments.

1.5.3 Disadvantages of Staging

- Staging requires the vehicle to lift motors which are not yet being used; this means that the first stages must produce higher thrusts than they supposed to do.
- Staging also makes the entire rocket more complex and harder to build.
- Each staging event is a significant point of failure during a launch, with a possibility of separation failure, ignition failure and stage collision.

1.6 Propulsion System Criteria for Selection

Many criteria used in selecting a particular rocket propulsion system are peculiar to the particular mission or vehicle application. 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 (H2, Ar, N2, 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.

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. 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. 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. 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.

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.

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).

1.7 Propulsion System 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, and the experience of those responsible for making the selection, and the available time and resources to examine the alternate propulsion systems.

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.

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 involves both the system organization (and 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.

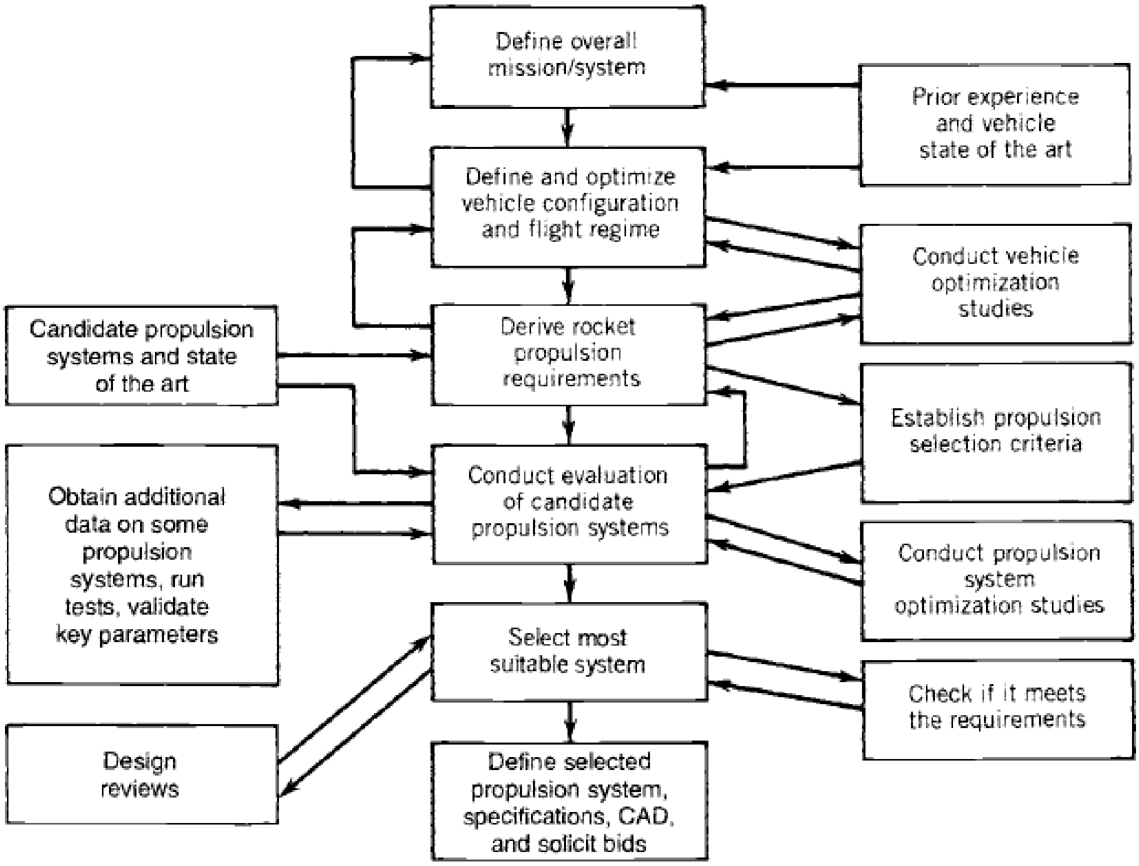


FIGURE 17–1. Idealized process for selecting propulsion systems.

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.

1.8 Mission Profile

1.8.1 Mission Profile of Space Launch Vehicle

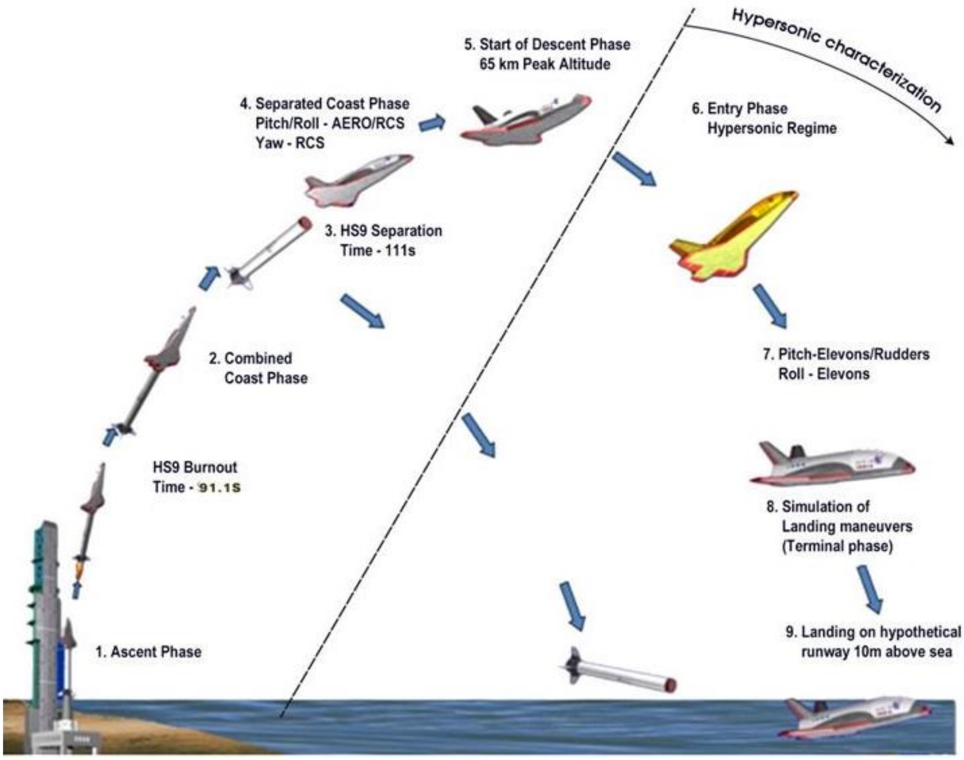


Fig: Mission profile of RLV TD (Reusable Launch Vehicle Technology Demonstrator)

The TDV (Technology Demonstrator Vehicle) weighing 1750 Kgs was propelled as part of the ascent by a slow burning 11-meter long solid rocket booster that burnt out around 90 seconds at an altitude of 33Kms. The booster and TDV continued in a combined coast up to 44Kms altitude when the dynamic pressure has reduced enough for a safe separation. The TDV continued on an unpowered coast after separation of the booster up to an altitude of 65 Kms with a peak Mach number of around 5 before starting its descent.

The descent of the space plane from that altitude had a reentry around Mach 4 that ended with a splash down in the sea, 412 Kms from the launch pad.

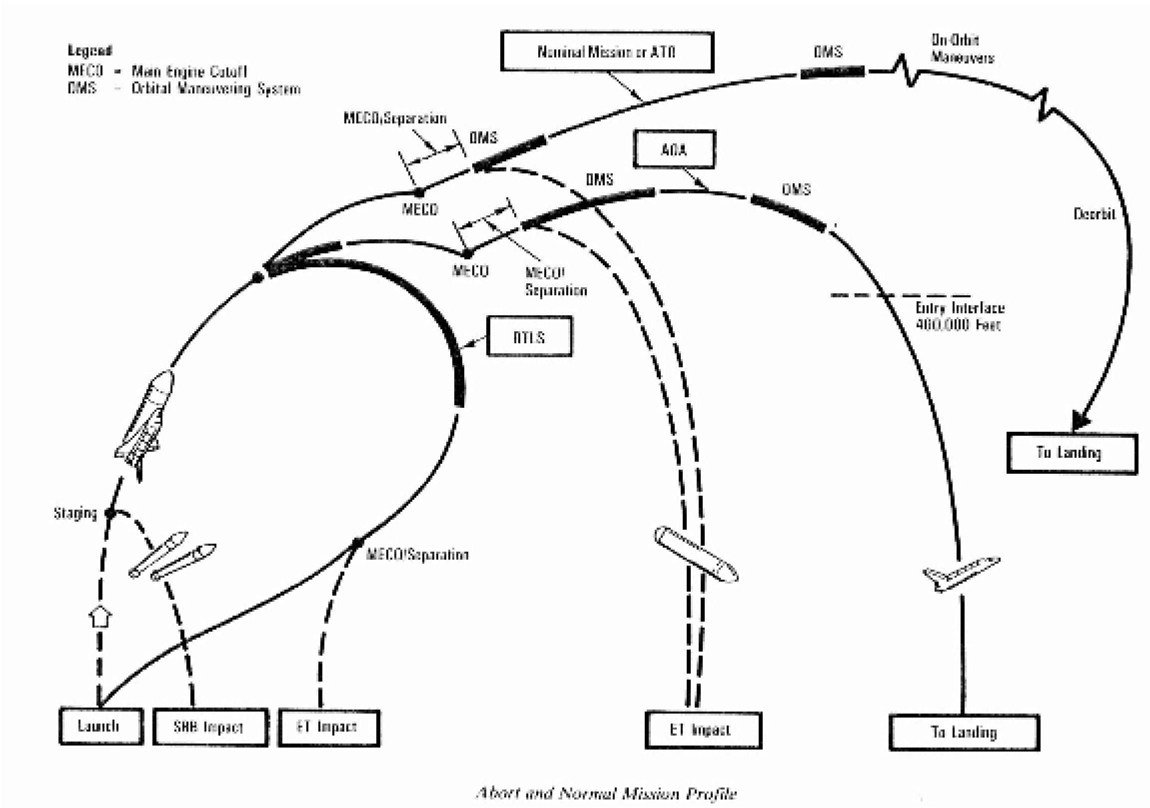


Fig: Mission profile of space launch vehicle - The Shuttle Orbiter

1.8.2 Mission Profile of Strategic Missile

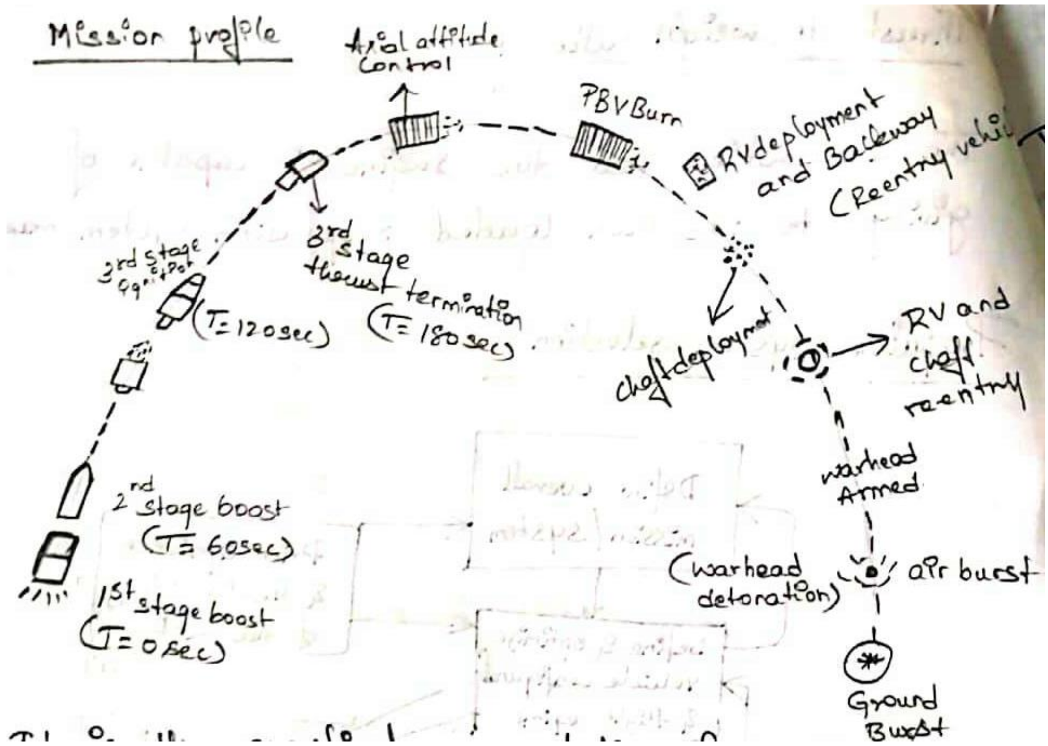


Fig: Mission profile of ballistic missile

- In order to cover the defined altitudes, the each stages will be undergoing the process of ignition and finally will reach the required altitude.
- The purpose of axial altitude control is to maintain or monitor the direction or orientation of missile.
- The deployment of PBV (Post Boost Vehicle) burn is to give the required thrust to the missile in order to enter the missile in to the earth atmosphere.
- With the help of Re-entry Vehicle (RV) and chaff (the inedible parts of the grain), the missile will be reaching the targeted place.
- Once it reaches to the targeted place, the warhead begins. The warhead detonation may occurs in air or ground.

1.9 Similarities and Differences between Space Launch Vehicles and Missiles

1.9.1 Differences

Space Launch Vehicle	Missile
Generally launch vehicle may be guided or unguided depending upon the application.	The missile will be the guided vehicle.
Launch vehicle are rocket powered vehicles.	Missiles are sometimes self-powered rocket.
Launch vehicles have no steering ability.	Missiles have the guidance system so that it can be steered in the flight towards its target.
Usually used for destructive or non-destructive purpose depending upon application.	Missiles are used only for the destructive purpose.
The launch vehicles are used to place payload into the orbit of earth or any other planet.	Missiles are always used to carry the warheads. Ex: bombs and explosives.
The launch vehicles are cylindrical tube propelled by combustion in a direction opposite to its thrust.	Missiles anything forcibly propelled at a target by means of mechanical.
Ex: PSLV, GSLV, Space Shuttle, SpaceX Falcon-9	Ex: AGNI, PRITHVI, BRAHMOS, AKASH, NAG, AMOGH, DHANUSH

1.9.2 Similarities

The following are the similarities can be found between space launch vehicles and the missile:

- Both space launch vehicle and missile carry a payload.
- In order to ignite, both the systems will use the propellants.
- The working principle for both vehicles is same. i.e., Newton's 3^{rd} law of motion.
- The trajectories followed by both launch vehicles and missiles will be same.
- Both the systems will have the same design criteria i.e., usually cylindrical shape.

1.10 Classification of Missiles

Missiles can be classified on the basis of:

- Point of launching and impact –
  - *Air-to-air missile (AAM)*
  - *Air-to-surface missile (ASM)*
  - *Air-to-underwater missile (AUM)*
  - *Surface-to-surface missile (SSM)*
  - *Underwater-to-underwater missile (UUM)*
- Type of guidance system –
  - *Command system:*
    - The missile and the target are continuously tracked from one or more vantage points and the necessary path for the missile to intercept the target is computed and relayed to the missile by some means such as radio.
  - *Beam-riding missile:*
    - It contains a guidance system to constrain it to a beam.
    - The beam is usually a radar illuminating the target so that, if the missile stays in the beam, it will move toward the target.
  - *Homing missile:*
    - It has a seeker, which sees the target and gives the necessary directions to the missile to intercept the target.
    - The homing missile can be subdivided into classes having:

        - *Active class* – the missile illuminates the target and receives the reflected signals.
        - *Semiactive class* – the missile receives reflected signals from a target illuminated by means external to missile.
        - *Passive class* – this type of guidance system depends on a receiver in the missile sensitive to the radiation of the target itself.

- Type of trajectory taken by the missile –
    - *Ballistic missile:*
        - It follows the usual ballistic trajectory of a hurled object.
    - *Glide missile:*
        - It is launched at a steep angle to an altitude depending on the range, and the glides down on the target.
    - *Skip missile:*
        - It is launched to an altitude where the atmosphere is very rare, and then skips along on the atmospheric shell.
- Propulsive system –
    - These missiles fall into the categories of *turbojet, ramjet, rocket, etc.,*
    - If the missile receives short burst of power that rapidly accelerates it to top speed and then glides to its target, it is a *boost-glide* missile.
    - Sometimes a missile is termed *single-stage, double –stage, etc*., depending on the number of stages of its propulsive system.
- Trim and control device –
    - *Canard missile:*
        - It has a small forward lifting surface that can be used for either trim or control similar to a tail-first airplane.
    - *Wing control missile:*
        - A missile controlled by deflecting the wing surfaces.
    - *Tail control missile:*
        - A missile controlled by deflecting the tail surfaces.
    - *Cruciform missile:*
        - These sets of controls at right angles permit the missile to turn immediately in any plane without the necessity of its banking.
    - *Bank to turn missile:*

- These are like an airplane, banks into the turn to bring the normal acceleration vector as close to the vertical plane of symmetry as possible.

1.11 Thrust Profile

A thrust curve, sometimes known as a "performance curve" or "thrust profile" is a graph of the thrust of an engine or motor (usually a rocket motor) with respect to time.

Most engines do not produce linear thrust (thrust which increases at a constant rate with time). Instead, they produce a curve of some type, where thrust will slowly rise to a peak, and then fall, or "tail off". Rocket engines, particularly solid-fuel rocket engines, produce very consistent thrust curves, making this a useful metric for judging their performance.

This information is vital when designing spacecraft, particularly multistage spacecraft, since it may be advantageous to separate the engine and its associated fuel tanks and machinery before the fuel has been fully exhausted. This is because even though the engine is still producing thrust during the tail off phase, it may be so little that the spacecraft would be more efficient without it.

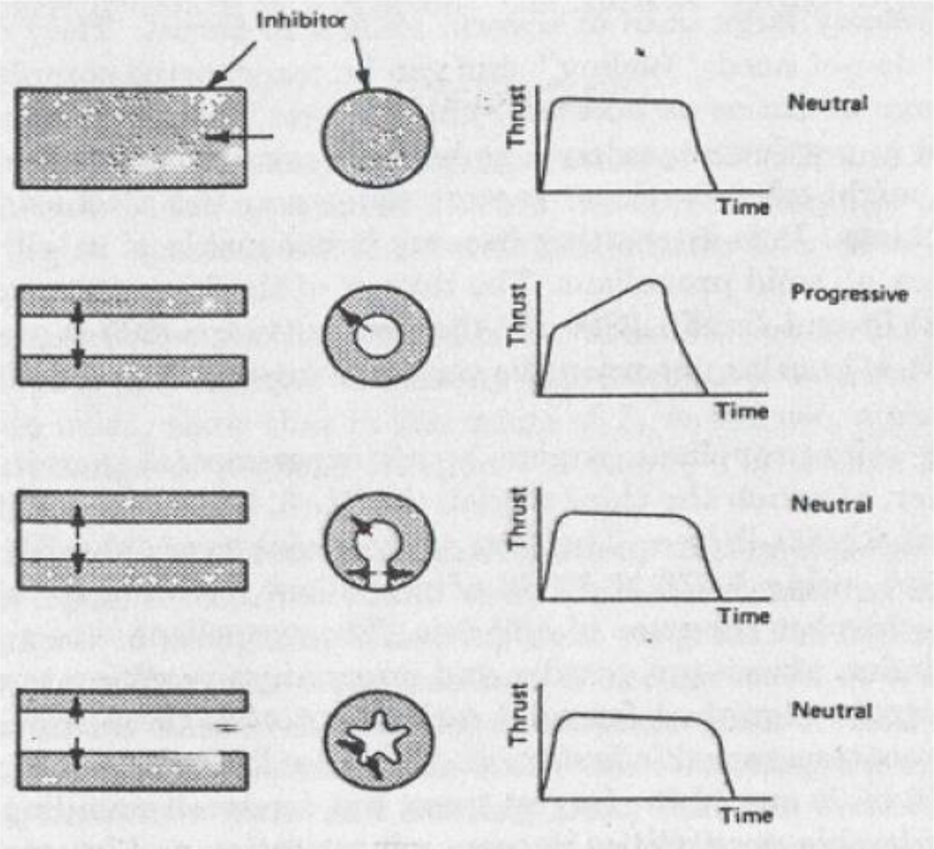


Fig: Thrust profile curve for Solid Propellant Rocket with different geometry

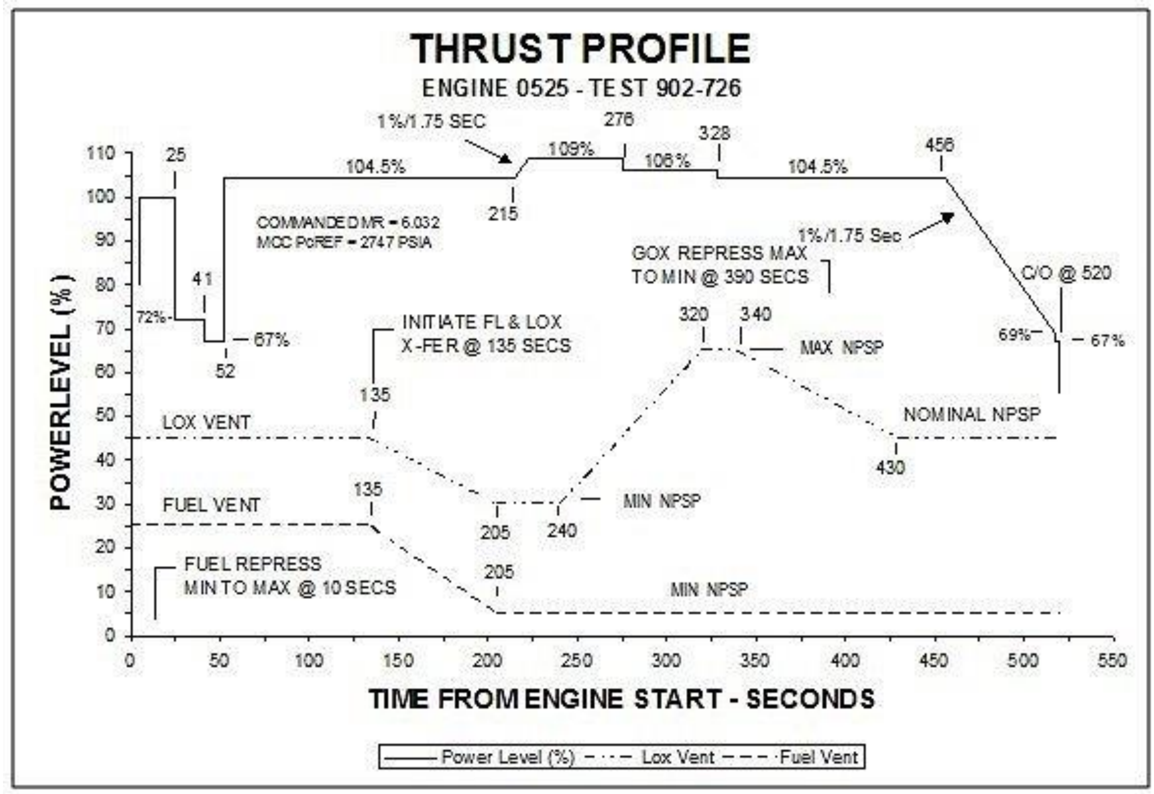


Fig: Thrust profile curve for Liquid Propellant Rocket (J-2X, RS-25, general)

1.12 Some Famous Space Launch Vehicles and Strategic Missiles

1.12.1 India's Space Launch Vehicles

During the 1960s and 1970s, India initiated its own launch vehicle programme owing to geopolitical and economic considerations. In the 1960s–1970s, the country developed a sounding rocket programme, and by the 1980s, research had yielded the Satellite Launch Vehicle-3 and the more advanced Augmented Satellite Launch Vehicle (ASLV), complete with operational supporting infrastructure. ISRO further applied its energies to the advancement of launch vehicle technology resulting in the creation of the successful PSLV and GSLV vehicles.

Satellite Launch Vehicle (SLV):

Status: Decommissioned

The Satellite Launch Vehicle (SLV or SLV-3) was a 4-stage solid-propellant light launcher. It was intended to reach a height of 500 kilometers and carry a payload of 40 kilograms. Its first launch took place in 1979 with two more in each subsequent year, and the final launch in 1983. Only two of its four test flights were successful.

Augmented Satellite Launch Vehicle (ASLV):

Status: Decommissioned

The Augmented Satellite Launch Vehicle (ASLV) was a five-stage solid propellant rocket with the capability of placing a 150 kilogram satellite into Low Earth Orbit. This project was started during the early 1980s to develop technologies needed for a payload to be placed into a geostationary orbit. Its design was based on Satellite Launch Vehicle. The first launch test was held in 1987, and after that three others followed in 1988, 1992 and 1994, out of which only two were successful, before it was decommissioned.

Polar Satellite Launch Vehicle (PSLV):

Status: Active

The Polar Satellite Launch Vehicle (PSLV) was a four-stage rocket with the capability of placing up to 3800 kilogram satellite into various Earth Orbit. PSLV can also launch small satellites into geostationary transfer orbit (GTO). The reliability and versatility of the PSLV is proven by the fact that it has launched, as of 2014, seventy-one satellites/spacecraft (thirty-one Indian and forty foreign) into a variety of orbits. The maximum number of satellites launched by the PSLV in a single launch is 104, in the PSLV-C37 launch on 15 February 2017.

Geosynchronous Satellite Launch Vehicle (GSLV):

Status: Active

The Geosynchronous Satellite Launch Vehicle (GSLV) is an expendable launch system developed to enable India to launch its INSAT-type satellites into geostationary orbit and to make India less dependent on foreign rockets. At present, it is ISRO's second-heaviest launch vehicle and is capable of putting a total payload of up to 4500 kilogram to low Earth orbit. The vehicle is built by India, originally with a cryogenic engine purchased from Russia, while the ISRO developed its own cryogenic engine.

Geosynchronous Satellite Launch Vehicle Mark-III (GSLV-Mk III):

Status: Active

GSLV-Mk III is a launch vehicle capable to launch 4000 kgs of satellites into geosynchronous transfer orbit (GTO) and 10000 kgs satellites into low earth orbit. GSLV-Mk III is a three-stage vehicle with a 110-tonne (120-ton) core liquid propellant stage (L-110) flanked by two 200-tonne (220-ton) solid propellant strap-on booster motors (S-200). The upper stage is cryogenic with a propellant loading of 25 tonnes (C-25). The vehicle has a lift-off mass of about 640 tonnes and is 43.43 metres tall. It allows India to become less dependent on foreign rockets for heavy lifting.

On 18 December 2014, ISRO conducted an experimental test-flight of GSLV MK III carrying a crew module, to be used in future human space missions. This suborbital test flight demonstrated the performance of GSLV Mk III in the atmosphere.

On 14 July GSLV-Mk III was supposed to launch Chandrayaan 2 but due to some technical issues regarding the helium tank it was postpone to 22 July 2019. On 22 July 2019 the GSLV-Mk III launched India's second Moon mission, Chandrayaan-2.

1.12.2 India’s Strategic Missiles

Project Devil:

Project Devil was one of two early liquid-fuelled missile projects developed by India, along with Project Valiant, in the 1970s. The goal of Project Devil was to produce a short-range surface-to-surface missile. Although discontinued in 1980 without achieving intended success, Project Devil, led to the later development of the Prithvi missile in the 1980s.

Project Valiant:

Project Valiant was one of two early liquid-fuelled missile projects developed by India, along with Project Devil in the 1970s. The goal of Project Valiant was to produce an

ICBM. Although discontinued in 1974 without achieving full success, Project Valiant, like Project Devil, helped in the development of the Prithvi missile in the 1980s.

Akash:

Akash (Sanskrit: आकाश Ākāś "Sky") is a medium-range mobile surface-to-air missile defence system developed by the Defence Research and Development Organisation (DRDO), Ordnance Factories Board and Bharat Electronics Limited (BEL) in India. The missile system can target aircraft up to 30 km away, at altitudes up to 18,000 m.

Trishul:

Trishul is a short range surface-to-air missile developed in India. It was developed by Defense Research and Development Organization as a part of the Integrated Guided Missile Development Program. It can also be used as an sea skimmer anti-ship against low flying attacking missiles.

Nag:

Nag (Sanskrit: नाग, Nāg "Cobra") is a third generation "Fire-and-forget" anti-tank missile developed in India. It is one of five missile systems developed by the Defense Research and Development Organization (DRDO) under the Integrated Guided Missile Development Program (IGMDP). Nag has been developed at a cost of ₹3 billion.

Prithvi Missile Series:

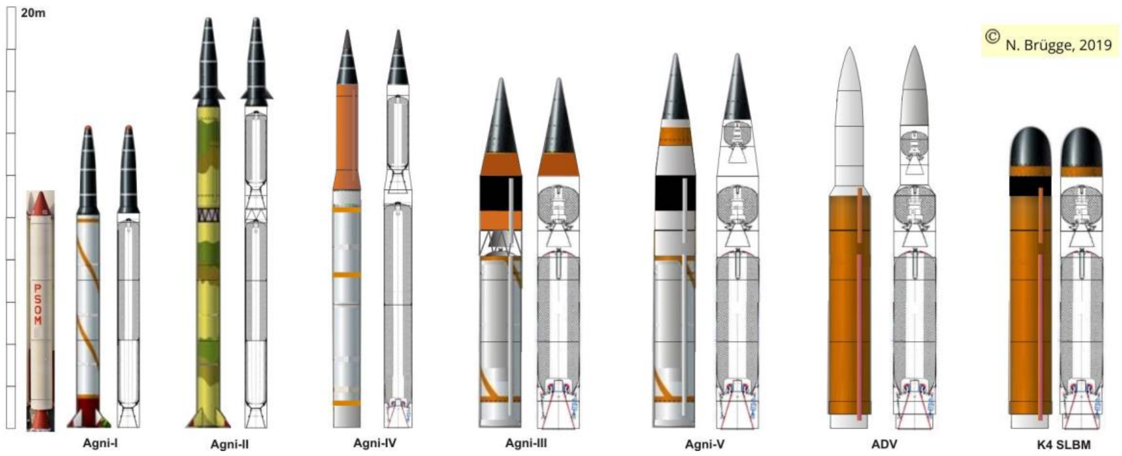
Prithvi Missiles are tactical surface-to-surface short-range ballistic missiles (SRBM).

Name	Type	Stage(s)	Range	Payload
Prithvi-I (SS-150)	SRBM	One	150 km	1000 kg
Prithvi-II (SS-250)	SRBM	Two	250 km – 350 km	500 kg – 1000 kg
Prithvi-III (SS-350)	SRBM	Two	350 km – 600 km	250 kg – 500 kg

Agni missile series:

The Agni missile series started as a "Re-Entry Vehicle" project (later rechristened as Agni Technology Demonstrator) in the IGMDP. The missiles in this series include:

Name	Type	Stage(s)	Range
Agni-I	MRBM	One	700 km – 1,200 km[15][16]
Agni-II	IRBM	Two	2,000 km – 2,500 km
Agni-III	IRBM	Two	3,000 km – 5,000 km[17]
Agni-IV	IRBM	Two	2,500 km – 3,700 km[18][19]
Agni-V	ICBM	Three	5,000 km – 8,000 km[20][21][22]
Agni-VI	ICBM	Three	10,000 km – 12,000 km[23]



K Missile series:

Type	Range	Weight	Warhead
K-15 SLBM[2]	750 km-1,500 km	6[3]-7 tonnes[4]	1 tonne
K-4 SLBM [2]	3,500 km	20 tonnes	2.5 tonnes
Air Launched [2]	200 km	2 tonnes	500 kg
K-5 SLBM	5,000 km[11]	Unspecified	1 tonne
K-6 SLBM	6,000 km	Unspecified	2-3 tonnes

Shaurya:

The Shaurya missile is a short-range surface-to-surface ballistic missile developed for use by the Indian Army. Capable of hypersonic speeds, it has a range of 600 km and is capable of carrying a payload of one-tonne conventional or nuclear warhead.

BrahMos:

Mach 3 Supersonic Cruise Missile developed in collaboration with Russia. Land Attack and Anti-ship variants in service with the Indian Army and Indian Navy. Sub-Launched and Air Launched variants under development or testing.

BrahMos II:

Mach 7 Hypersonic Cruise Missile in development collaboration with Russia.

Nirbhay:

Long Range Sub-Sonic Cruise Missile under development and testing. It was successfully test fired for second time from balasore Orissa. Able to travel at speed of 0.6 mach.

Prahaar:

Prahaar (Sanskrit:प्रहार, Strike) is a solid-fuelled Surface-to-surface guided short-range tactical ballistic missile that would be equipped with omni-directional warheads and could be used for hitting both tactical and strategic targets.

Astra:

Astra is a 'Beyond Visual Range Air-to-Air Missile' (BVRAAM) being developed for the Indian Air Force.

Helina:

A variant of NAG Missile to be launched from Helicopter is being developed under the Project named HELINA (HELIcopter launched NAg). It will be structurally different from the Nag.