INSTITUTE OF AERONAUTICAL ENGINEERING

LECTURE NOTES

ROCKET AND MISSILES

B. Tech VIII SEMESTER

Prepared by

V.PHANINDER REDDY (IARE 10741)

Contents

List of Figures

Abbreviations

TVC	**T**hrust **V**ector **C**ontrol
LOX	**L**iquid **OX**ygen
LVDT	**L**iquid **P**ropellant **R**ocket **E**ngine
RC	**R**einforced **C**oncrete

Symbols

D^{el}	Elasticity tensor
σ	Stress tensor
ε	Strain tensor
V_{eq}	Equivalent velocity
$\dot{m}$	Mass flow rate
I_{sp}	Specific Impulse
c	Effective exhaust velocity
I_t	Total impulse
υ	Exhaust velocity
m_p	Propellant mass
m_e	Empty mass
A_{ex}	Exit Area
p_{ex}	Exhaust pressure
P_{SL-a}	Ambient pressure at sea level
F_{SL-a}	Sea level thrust of the rocket
$\dot{W}_{sp}$	Specific propellant consumption rate
C_w	Weight flow coefficient
C_f	Thrust coefficient

Chapter 1

ROCKET DYNAMICS

1.1 Introduction

In a broad sense, propulsion is the act of changing the motion of a body. Propulsion mechanisms provide forces which move bodies that are initially at rest, change a constant velocity motion, or overcome retarding forces when a body is propelled through a medium. There are two essential elements in any propulsive mechanism: the energy source and an energy conversion device to transpose the energy into the form most suitable for propulsion. In an automobile, for example, a chemical combustion process of fuel with air furnishes the energy input, which is then transformed in an engine into thermal energy of a gas and subsequently by transformation into the mechanical energy through a rotating shaft and wheels to imparting momentum to the vehicle.

1.2 Rocket

A rocket engine is the device or mechanism that converts the energy into suitable form and ejects stored matter to derive momentum. The working fluid or the ejected matter in rocket propulsion is called the propellant .

1.2.1 Classification of Rockets

Among many possible energy sources, four are considered to be useful in rocket propulsion: the chemical combustion reaction, nuclear reaction, captured radiation energy from an emitter such as the sun, and Jet, Rocket, Nuclear, Ion and Electric Propulsion electric energy which is stored or created in the vehicle. Accordingly, the various propulsion devices can be categorized into

1. Chemical propulsion
2. Nuclear energy propulsion
3. Solar energy propulsion and
4. Electric energy propulsion

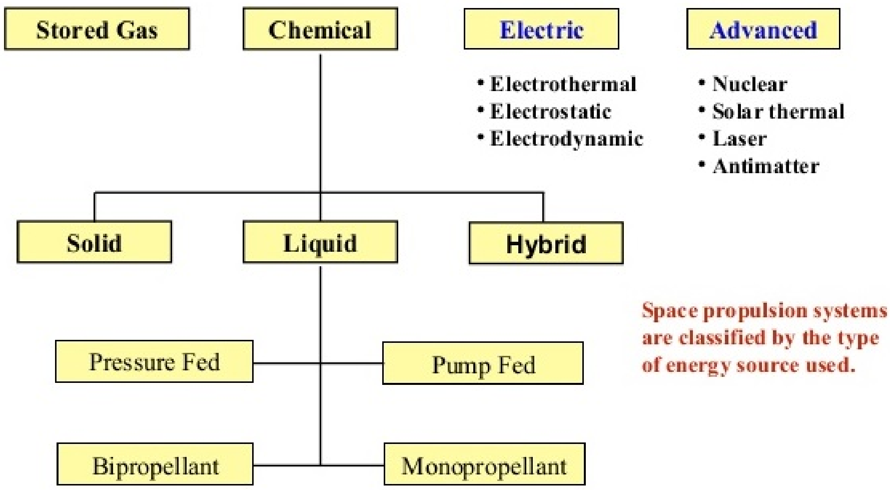

FIGURE 1.1: Classification of Rockets

1.2.1.1 Chemical Propulsion:

1. Solid propellant rockets

   It consist of a case or tube in which the propellants are packed. Modern rockets uses cases made of a thin and lightweight metal such as aluminum. Making the case from thin metal reduces the overall weight of the structure and increases flight performance. However, the heat from the burning propellants could easily melt through the metal. To prevent this, the inner walls of the case have to be insulated.

The upper end of the rocket is closed off and capped with a payload section or recovery parachutes. The lower end of the rocket is constricted with a narrow opening called the throat, above a larger cone-shaped structure, called the nozzle. By constricting the opening, the throat causes the combustion products to accelerate greatly as they race to the outside (second law). The nozzle aims the exhaust straight downward so that the rocket travels straight upward (third law).

To appreciate how the throat of the rocket accelerates the combustion products, turn on the water for a garden hose. Open the nozzle to the widest setting. Water slowly flows out. Next, reduce the opening of the nozzle. Water quickly shoots out in a long stream (second law) and the hose pushes back on you (third law). The propellant in solid rockets is packed inside the insulated case. It can be packed as a solid mass or it may have a hollow core. When packed as a solid mass, the propellant burns from the lower end to the upper end. Depending upon the size of the rocket, this could take a while.

With a hollow core, the propellants burn much more rapidly because the entire face of the core is ignited at one time. Rather than burning from one end to the other, the propellant burns from the core outward, Solid Propellant Rocket End-burning and hollow core rockets towards the case. The advantage of a hollow core is that the propellant mass burns faster, increasing thrust (second law). To make solid rockets even more powerful, the core doesn't have to be round. It can have other shapes that increase the surface area available for burning. The upper ends of the space shuttle SRBs had star-shaped cores. When ignited, the large surface area of the star points boosted liftoff thrust. In about one minute, however, the points burned off, and the thrust diminished somewhat. This was done on purpose because the space shuttle begins accelerating through the sound barrier. Passing through causes vibrations that are diminished by the temporary thrust reduction of the SRBs (second law).

2. Liquid propellant rockets

   They are an invention of the twentieth century. They are far more complex than solid rockets. Generally, a liquid rocket has two large tanks within its body. One tank contains a fuel, such as kerosene or liquid hydrogen. The other tank contains liquid oxygen. When the liquid rocket engine is fired, high-speed pumps force the propellants into a cylindrical or spherical combustion chamber. The fuel and oxidizer mix as they are sprayed into the chamber. There they ignite, creating huge quantities of combustion products that shoot through the throat and are focused downward by the nozzle.

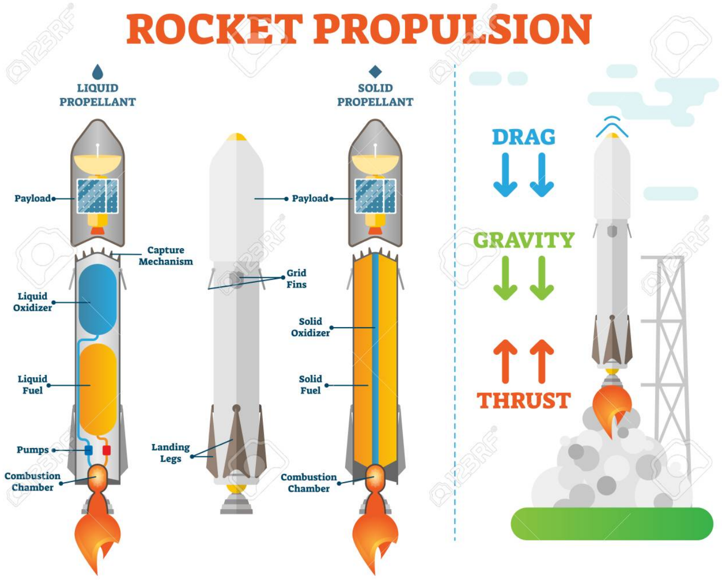

FIGURE 1.2: Classification of Chemical Rockets

1.2.1.2 Nuclear Thermal Propulsion(NTP):

NTP systems work by pumping a liquid propellant, most likely hydrogen, through a reactor core. Uranium atoms split apart inside the core and release heat through fission. This physical process heats up the propellant and converts it to a gas, which is expanded through a nozzle to produce thrust.

NTP rockets are more energy dense than chemical rockets and twice as efficient. Engineers measure this performance as specific impulse, which is the amount of thrust you can get from a specific amount of propellant. The specific impulse of a chemical rocket that combusts liquid hydrogen and liquid oxygen is 450 seconds, exactly half the propellant efficiency of the initial target for nuclear-powered rockets (900 seconds). This is because lighter gases are easier to accelerate. When chemical rockets are burned, they produce water vapor, a much heavier byproduct than the hydrogen that is used in a NTP system. This leads to greater efficiency and allows the rocket to travel farther on less fuel.

NTP systems won't be used on Earth. Instead, they'll be launched into space by chemical rockets before they are turned on. NTP systems are not designed to produce the amount of thrust needed to leave the Earth's surface.

NTP systems offer greater flexibility for deep space missions. They can reduce travel times to Mars by up to 25 percent and, more importantly, limit a flight crew's exposure to cosmic radiation. They can also enable broader launch windows that are not dependent on orbital alignments and allow astronauts to abort missions and return to Earth if necessary.

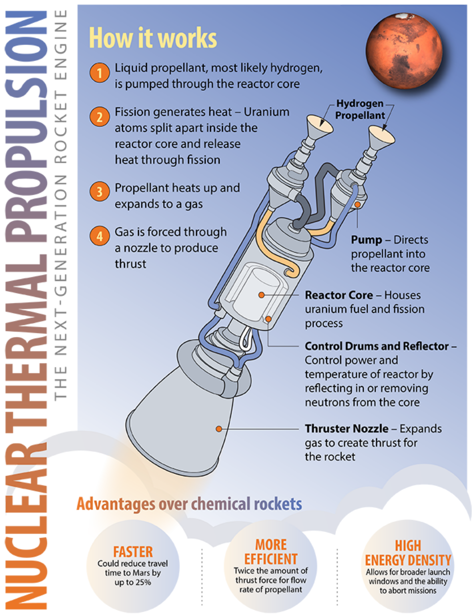

FIGURE 1.3: Nuclear Propulsion

1.2.1.3 Ion Thruster Propulsion:

An ion thruster or ion drive is a form of electric propulsion used for spacecraft propulsion. It creates thrust by accelerating ions using electricity. The positively charged ions migrate toward grids that contain thousands of very precisely aligned holes (apertures) at the aft end of the ion thruster. The first grid is the positively charged electrode (screen grid). A very high positive voltage is applied to the screen grid, but it is configured to force the discharge plasma to reside at a high voltage. As ions pass between the grids, they are accelerated toward a negatively charged electrode (the accelerator grid) to very high speeds (up to 90,000 mph).

The positively charged ions are accelerated out of the thruster as an ion beam, which produces thrust. The neutralizer, another hollow cathode, expels an equal amount of electrons to make the total charge of the exhaust beam neutral. Without a neutralizer, the spacecraft would build up a negative charge and eventually ions would be drawn back to the spacecraft, reducing thrust and causing spacecraft erosion.

Electrons produced by the discharge cathode are attracted to the dis- charge chamber walls, which are charged to a high positive potential by the voltage applied by the thruster's discharge power supply. Neutral propellant is injected into the discharge chamber, where the electrons bombard the propellant to produce positively charged ions and release more electrons.

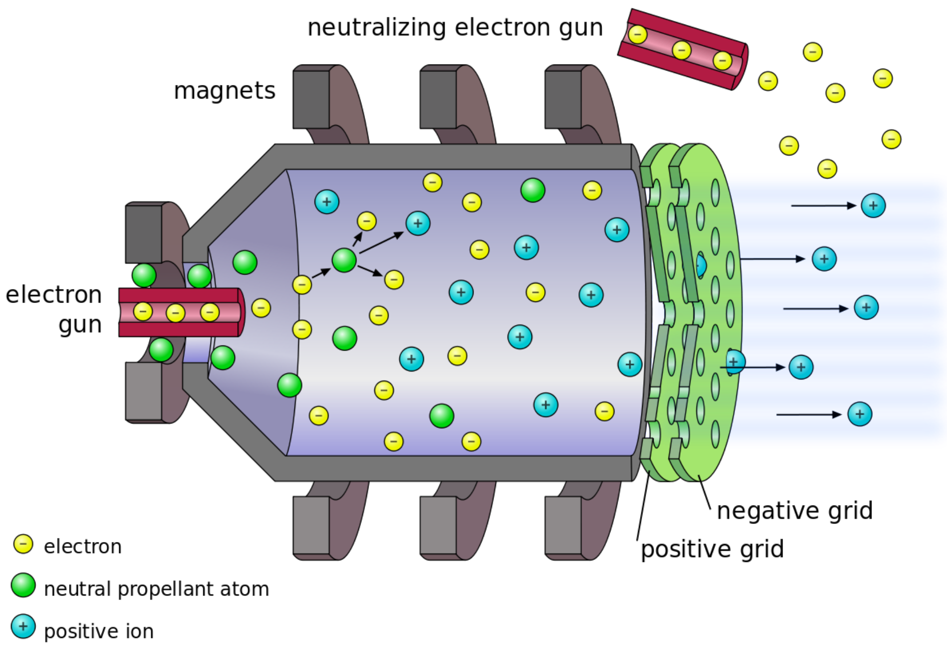

FIGURE 1.4: Ion Propulsion

1.3 Missile

Missile, a rocket-propelled weapon designed to deliver an explosive warhead with great accuracy at high speed. Missiles vary from small tactical weapons that are effective out to only a few hundred feet to much larger strategic weapons that have ranges of several thousand miles.

1.3.1 Classification of Missiles

Missiles are generally classified on the basis of their Type, Launch Mode, Range, Propulsion, Warhead and Guidance Systems. All missiles contain some form of guidance and control mechanism and are therefore often referred to as guided missiles. Launch vehicles are the rocket-powered systems that provide transportation from the earth's surface into the environment of space. A propeller-driven underwater missile is called a torpedo, and a guided missile powered along a low, level flight path by an air-breathing jet engine is called a cruise missile.

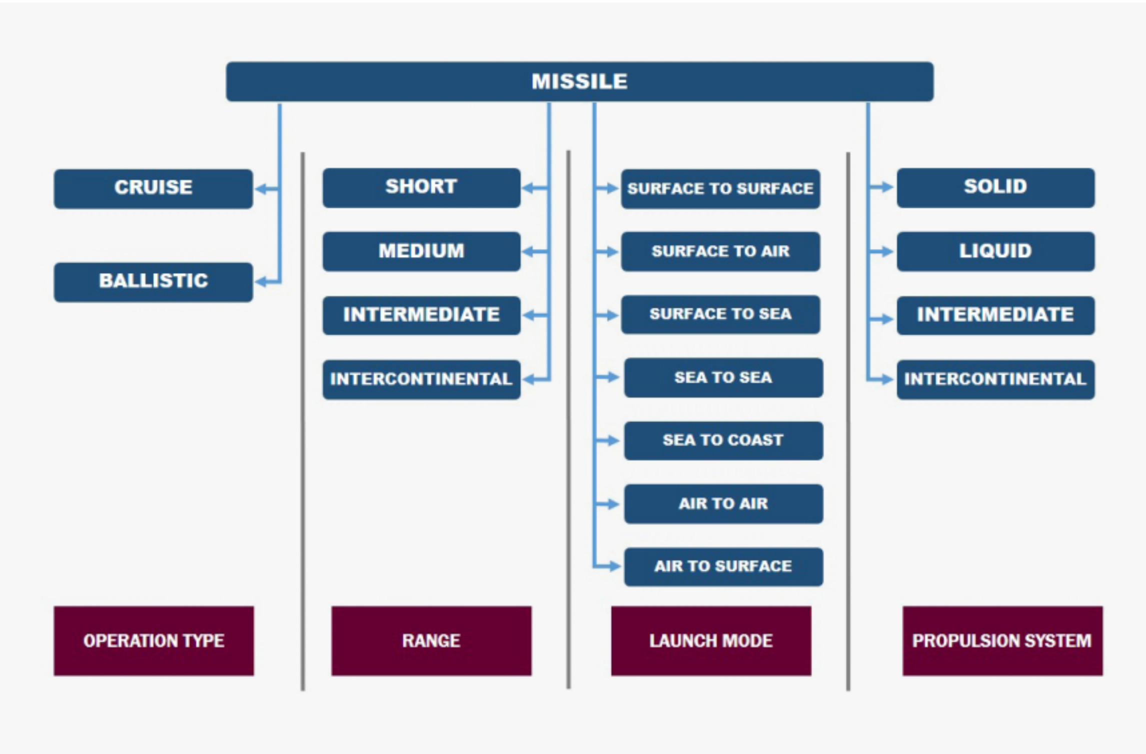

FIGURE 1.5: Classification of Missiles

1.3.1.1 On the Basis of Type:

1. Cruise Missile: A cruise missile is an unmanned self-propelled (till the time of impact) guided vehicle that sustains flight through aerodynamic lift for most of its flight path and whose primary mission is to place an ordnance or special payload on a target. They fly within the earth's atmosphere and use jet engine technology. These vehicles vary greatly in their speed and ability to penetrate defenses.Cruise missiles can be categorized by size,

speed (subsonic or supersonic), range and whether launched from land, air, surface ship or submarine.

Depending upon the speed such missiles are classified as:

(a) Subsonic cruise missile

(b) Supersonic cruise missile

(c) Hypersonic cruise missile

Subsonic cruise missile flies at a speed lesser than that of sound. It travels at a speed of around 0.8 Mach. The well- known subsonic missile is the American Tomahawk cruise missile. Some other examples are Harpoon of USA and Exocet of France.

Supersonic cruise missile travels at a speed of around 2-3 Mach i.e.; it travels a kilometer approximately in a second. The modular design of the missile and its capability of being launched at different orientations enable it to be integrated with a wide spectrum of platforms like warships, submarines, different types of aircraft, mobile autonomous launchers and silos. The combination of supersonic speed and warhead mass provides high kinetic energy ensuring tremendous lethal effect. BrahMos is the only known versatile supersonic cruise missile system which is in service.

Hypersonic cruise missile travels at a speed of more than 5 Mach. Many countries are working to develop hypersonic cruise missiles. BrahMos Aerospace is also in the process of developing a hypersonic cruise missile, BrahMos-II, which would fly at a speed greater than 5 Mach.

2. Ballistic Missile: A ballistic missile is a missile that has a ballistic trajectory over most of its flight path, regardless of whether or not it is a weapon-delivery vehicle. Ballistic missiles are categorized according to their range,maximum distance measured along the surface of earth's ellipsoid from the point of launch to the point of impact of the last element of their payload. The missile carry a huge payload. The carriage of a deadly warhead is justified by the distance the missile travels. Ballistic missiles can be launched from ships and land based facilities. For example, Prithvi I, Prithvi II, Agni I, Agni II and Dhanush ballistic missiles are currently operational in the Indian defense forces.

FIGURE 1.6: Missiles on the basis of its type

1.3.1.2 On the basis of Launch Mode:

1. Surface-to-Surface Missile: A surface-to-surface missile is a guided projectile launched from a hand-held, vehicle mounted, trailer mounted or fixed installation. It is often powered by a rocket motor or sometimes fired by an explosive charge since the launch platform is stationary.

2. Surface-to-Air Missile: A surface-to-air missile is designed for launch from the ground to destroy aerial targets like aircrafts, helicopters and even ballistic missiles. These missiles are generally called air defense systems as they defend any aerial attacks by the enemy.

3. Surface (Coast)-to-Sea Missile: A surface (coast)-to-sea missile is designed to be launched from land to ship in the sea as targets.

4. Air-to-Air Missile: An air-to-air missile is launched from an aircraft to destroy the enemy aircraft. The missile flies at a speed of 4Mach.

5. Air-to-Surface Missile: An air-to-surface missile is designed for launch from military aircraft and strikes ground targets on land, at sea or both. The missiles are basically guided via laser guidance, infrared guidance and optical guidance or via GPS signals. The type of guidance depends on the type of target.

6. Sea-to-Sea Missile: A sea-to-sea missile is designed for launch from one ship to another ship.

7. Sea-to-Surface (Coast) Missile: A sea-to-surface missile is designed for launch from ship to land based targets.

8. Anti-Tank Missile: An anti-tank missile is a guided missile primarily designed to hit and destroy heavily-armored tanks and other armored fighting vehicles. Anti-tank missiles could be launched from aircraft, helicopters, tanks and also from shoulder mounted launcher.

1.3.1.3 On the basis of Range:

This type of classification is based on maximum range achieved by the missiles. The basic classification is as follows:

1. Short Range Missile

2. Medium Range Missile

3. Intermediate Range Ballistic Missile

4. Intercontinental Ballistic Missile

1.3.1.4 On the basis of Propulsion:

1. Solid Propulsion: Solid fuel is used in solid propulsion. Generally, the fuel is aluminum powder. Solid propulsion has the advantage of being easily stored and can be handled in fuelled condition. It can reach very high speeds quickly. Its simplicity also makes it a good choice whenever large amount of thrust is needed.

2. Liquid Propulsion: The liquid propulsion technology uses liquid as fuel. The fuels are hydrocarbons. The storage of missile with liquid fuel is difficult and complex. In addition, preparation of missile takes considerable time. In liquid propulsion, propulsion can be controlled easily by restricting the fuel flow by using valves and it can also be controlled even under emergency conditions. Basically, liquid fuel gives high specific impulse as compared to solid fuel.

3. Hybrid Propulsion: There are two stages in hybrid propulsion - solid propulsion and liquid propulsion. This kind of propulsion compensates the disadvantages of both propulsion systems and has the combined advantages of the two propulsion systems.

4. Ramjet: A ramjet engine does not have any turbines unlike turbojet engines. It achieves compression of intake air just by the forward speed of the air vehicle. The fuel is injected and ignited. The expansion of hot gases after fuel injection and combustion accelerates the exhaust air to a velocity higher than that at the inlet and creates positive push. However, the air entering the engine should be at supersonic speeds. So, the aerial vehicle must be moving in supersonic speeds. Ramjet engines cannot propel an aerial vehicle from zero to supersonic speeds.

5. Scramjet: Scramjet is an acronym for Supersonic Combustion Ramjet. The difference between scramjet and ramjet is that the combustion takes place at supersonic air velocities through the engine. It is mechanically simple, but vastly more complex aerodynamically than a jet engine. Hydrogen is normally the fuel used.

6. Cryogenic: Cryogenic propellants are liquefied gases stored at very low temperatures, most frequently liquid hydrogen as the fuel and liquid oxygen as the oxidizer. Cryogenic propellants require special insulated containers and vents which allow gas to escape from the evaporating liquids. The liquid fuel and oxidizer are pumped from the storage tanks to an expansion chamber and injected into the combustion chamber where they are mixed and ignited by a flame or spark. The fuel expands as it burns and the hot exhaust gases are directed out of the nozzle to provide thrust.

1.3.1.5 On the basis of Warhead :

1. Conventional Warhead: A conventional warhead contains high energy explosives. It is filled with a chemical explosive and relies on the detonation of the explosive and the resulting metal casing fragmentation as kill mechanisms.

2. Strategic Warhead: In a strategic warhead, radio active materials are present and when triggered they exhibit huge radio activity that can wipe out even cities. They are generally designed for mass annihilation.

1.3.1.6 On the basis of Guidance Systems:

1. Wire Guidance: This system is broadly similar to radio command, but is less susceptible to electronic counter measures. The command signals are passed along a wire (or wires)dispensed from the missile after launch.

2. Command Guidance: Command guidance involves tracking the projectile from the launch site or platform and transmitting commands by radio, radar, or laser impulses or along thin wires or optical fibers. Tracking might be accomplished by radar or optical instruments from the launch site or by radar or television imagery relayed from the missile.

3. Terrain Comparison Guidance: Terrain Comparison (TERCOM) is used invariably by cruise missiles. The system uses sensitive altimeters to measure the profile of the ground directly below and checks the result against stored information.

4. Terrestrial Guidance: This system constantly measures star angles and compares them with the pre-programmed angles expected on the missile's intended trajectory. The guidance system directs the control system whenever an alteration to trajectory is required.

5. Inertial Guidance: This system is totally contained within the missile and is programmed prior to launch. Three accelerometers, mounted on a platform space-stabilized by gyros, measure accelerations along three mutually perpendicular axes; these accelerations are then integrated twice, the first integration giving velocity and the second giving position. The system then directs the control system to preserve the pre-programmed trajectory. This systems are used in the surface-to-surface missiles and in cruise missiles.

6. Beam Rider Guidance: The beam rider concept relies on an external ground or ship-based radar station that transmits a beam of radar energy towards the target. The surface radar tracks the target and also transmits a guidance beam that adjusts its angle as the target moves across the sky.

7. Laser Guidance: In laser guidance, a laser beam is focused on the target and the laser beam reflects off the target and gets scattered. The missile has a laser seeker that can detect even miniscule amount of radiation. The seeker provides the direction of the laser scatters to the guidance system.The missile is launched towards the target,the seeker looks out for the laser

reflections and the guidance system steers the missile towards the source of laser reflections that is ultimately the target.

8. RF and GPS Reference: RF (Radio Frequency) and GPS (Global Positioning System) are examples of technologies that are used in missile guidance systems. A missile uses GPS signal to determine the location of the target. Over the course of its flight, the weapon uses this information to send commands to control surfaces and adjusts its trajectory. In a RF reference, the missile uses RF waves to locate the target.

Differentiate between tactical and strategic missiles

Ans. A tactical missile is used for attacking or defending ground troops, nearby military or strategic installations, military aircraft, or war missiles. Strategic missiles with a range of 3000 km or more have been two- or three stage surface-to-surface rocket-propelled missiles. Early designs used liquid propellant rocket engines and some are still in service.

Guided missiles

When missiles are launched from an aircraft at a relatively high initial velocity, or when projectiles are given stability by spinning them on their axis, their accuracy of reaching a target is increased two- to ten-fold, compared to a simple fin-stabilized rocket launched from rest. These are called guided missiles. In guided air-to-air and surface-to-air rocket-propelled missiles the time of flight to a given target, usually called the time to target tt, is an important flight performance parameter.

1.4 Airframe components

There are many parts that make up a rocket. For design and analysis, engineers group parts which have the same function into systems. There are four major systems in a full scale rocket; the structural system, the payload system, the guidance system, and the propulsion system.

The structural system, or frame, is similar to the fuselage of an airplane. The frame is made from very strong but light weight materials, like titanium or aluminum, and usually employs long

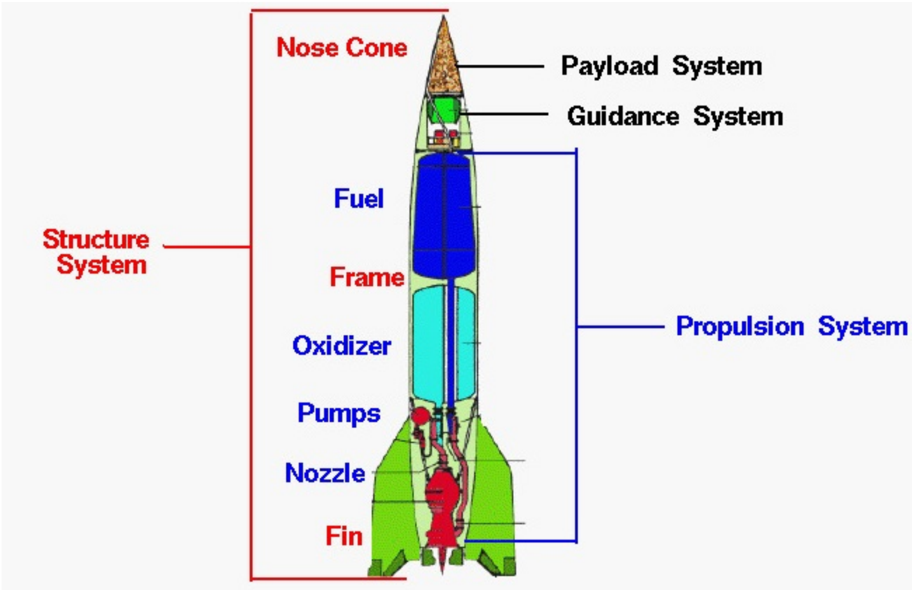

FIGURE 1.7: Airframe components of a Rocket

"stringers" which run from the top to the bottom which are connected to "hoops" which run around around the circumference. The "skin" is then attached to the stringers and hoops to form the basic shape of the rocket. The skin may be coated with a thermal protection system to keep out the heat of air friction during flight and to keep in the cold temperatures needed for certain fuels and oxidizers. Fins are attached to some rockets at the bottom of the frame to provide stability during the flight.

The payload system of a rocket depends on the rocket's mission. Many countries developed guided ballistic missiles armed with nuclear warheads for payloads. The same rockets were modified to launch satellites with a wide range of missions; communications, weather monitoring, spying, planetary exploration, and observatories, like the Hubble Space Telescope. Special rockets were developed to launch people into earth orbit and onto the surface of the Moon.

The guidance system of a rocket may include very sophisticated sensors, on-board computers, radars, and communication equipment to maneuver the rocket in flight. Many different methods have been developed to control rockets in flight. The V2 guidance system included small vanes in the exhaust of the nozzle to deflect the thrust from the engine. Modern rockets typically rotate the nozzle to maneuver the rocket. The guidance system must also provide some level of stability so that the rocket does not tumble in flight.

Most of a full scale rocket is propulsion system. There are two main classes of propulsion systems, liquid rocket engines and solid rocket engines. The V2 used a liquid rocket engine consisting of fuel and oxidizer (propellant) tanks, pumps, a combustion chamber with nozzle, and the associated plumbing. The Space Shuttle, Delta II, and Titan III all use solid rocket strap-ons.

The various rocket parts described above have been grouped by function into structure, payload, guidance, and propulsion systems. There are other possible groupings. For the purpose of weight determination and flight performance, engineers often group the payload, structure, propulsion structure (nozzle, pumps, tanks, etc.), and guidance into a single empty weight parameter. The remaining propellant weight then becomes the only factor that changes with time when determining rocket performance.

1.5 Inertia and Non Inertia Reference frames:

A reference frame is specified by an ordered set of three mutually orthogonal, possibly time dependent, unit-length direction vectors. A reference frame has an associated center. A coordinate system specifies a mechanism for locating points within a reference frame. Velocity is a frame dependent quantity but acceleration is frame independent. If force is frame dependent, then Newton's law will be valid in all frames $\overrightarrow{F} = m * \overrightarrow{a}$

1.5.1 Types of Reference frames

1.5.1.1 Inertial Reference frame

A frame of reference in which isolated object(object that experience no real forces) is found to move with constant velocity is called as Inertial frame of reference. There can be infinite number of Inertial frames but relative velocity between them must be constant.
In most of the cases this reference frame is fixed to the ground because we want to find out what

is the force respect to a ground fixed coordinate system. For example in this case of the rocket, we have to send the rocket out from the ground and we want to find out what is the force required to actually or what is the thrust required to actually accelerate the rocket away from the ground.
The force acting depends on the coordinate system, the choice of the coordinate system. For example, if you take this rocket, then suppose for an observer who is sitting inside the rocket, he throws a ball upwards, so when he does that, if force is required to throw the ball upwards and that force is different than if a observer who is sitting on the ground has to throw a ball at the same speed but considering the acceleration of the rocket. So, if you at the ball which is thrown by the observer sitting inside the rocket from a ground fixed coordinate system the ball has an acceleration imparted by the observer inside the rocket as well as the acceleration of the rocket itself.

1.5.1.2 Non-Inertial Reference frame

In practical applications we often encounter situations where we have to use a accelerating control volume. In the derivation of Reynolds transport theorem we have assumed that the that our coordinate system for defining the fluid velocity is fixed to the control volume. If the control volume is accelerating, it means that the reference frame is also accelerating and such a reference frame is known as a non-inertial frame of reference. There are many non-inertial (that is, accelerating) frames that one needs to consider, such as elevators, merry-go-rounds, and so on.
One such application is when we try to find using integral analysis the thrust acting on the rocket. While we try to do that, we actually put a control volume around the rocket and try to find the force but a rocket moves, it accelerates and the control volume attached to the rocket also have to accelerate. So, now we will see how to change our equations to accommodate this acceleration of the reference frame which is attached to the control volume. So, this is the case of an accelerating control volume.

1.6 Aerodynamic Forces on a Rocket

Aerodynamic forces are generated and act on a rocket as it flies through the air. Forces are vector quantities having both a magnitude and a direction. The single aerodynamic force is broken into two components: **the drag force** which is opposed to the direction of motion, and **the lift force** which acts perpendicular to the direction of motion. Aerodynamic forces are mechanical forces. They are generated by the interaction and contact of a solid body with a fluid, a liquid or a gas. Aerodynamic forces are not generated by a force field, in the sense of the gravitational field,or an electromagnetic field.

Aerodynamic forces are used differently on a rocket than on an airplane.

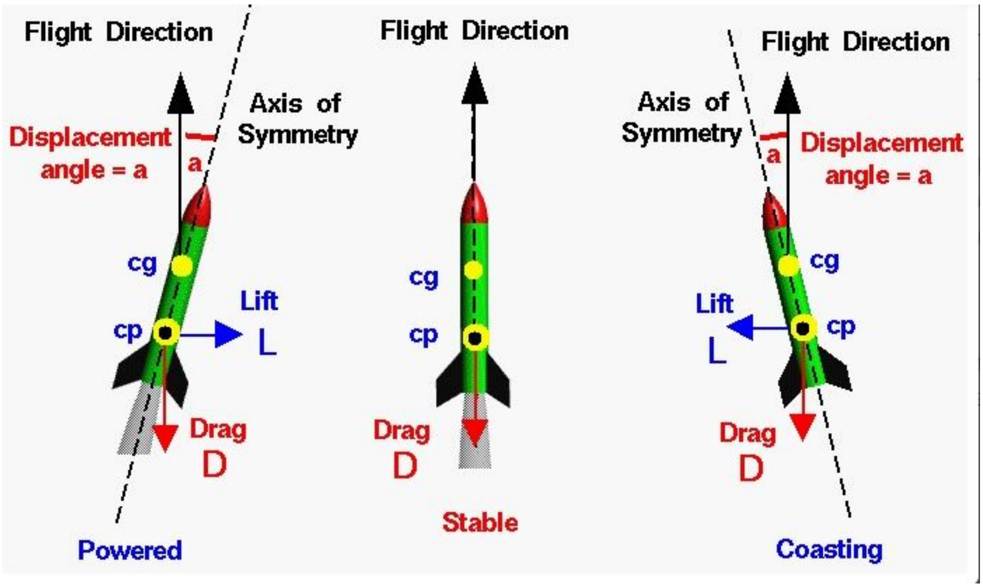

FIGURE 1.8: Aerodynamic forces of a Rocket

1. On an airplane, Lift is used to overcome the weight of the aircraft, but on a rocket, thrust is used in opposition to weight. Because the center of pressure is not normally located at the center of gravity of the rocket, aerodynamic forces can cause the rocket to rotate in flight.

2. The lift of a rocket is a side force used to stabilize and control the direction of flight. Lift occurs when a flow of gas is turned by a solid object. The flow is turned in one direction, and the lift is generated in the opposite direction, according to Newton's third law of action and reaction.

3. While most aircraft have a high lift to drag ratio, the drag of a rocket is usually much greater than the lift.

Two very important points are found on a rocket: center of gravity (CG) and center of pressure (CP). **Center of gravity** is the point on the z axis (center axis through the length of the rocket) where the amount of mass on both sides of that point is equal. If you are balancing a body with uniform mass distribution, the center of gravity will be in the middle of the object.

Center of pressure is the point along the rocket z axis with the same amount of surface area on both sides. For an object with a simple mathematical shape, it can be found with a simple integral. Generally, it is difficult to calculate and can either be found experimentally—in a wind tunnel—or numerically.

Stability: It is important to know where the CG and CP are in absolute measures and relative to each other. The rocket will always rotate around the center of gravity during flight, and gravity act on that singular point. However, the drag and lift forces do act on the center of pressure, and this decide how stable the rocket is. Stability is usually judged by the stability margin (SM), where the distance between the center of gravity and center of pressure is divided by the diameter d of the rocket body. $SM =$

There are two ways to stabilize rockets: active and passive. Active stabilization is using rocket engines (like gimballing the main thrusters or using smaller engines called Vernier thrusters) to control the attitude of the rocket. Active controlling is expensive and complex, but on large rockets it is necessary to use it.

On smaller rockets, as the ones launched at Andøya Space Center for science, one usually does not need to control the rocket attitude after lift-off and the rocket is then stabilized passively using a controlled spin. The spin is usually induced by the fins by aerodynamic forces.

1.7 Rocket Performance parameters

1. Equivalent velocity $V_{eq} = V_e + (p_e - p_0) * \frac{A_e}{\dot{m}}$

2. Total Impulse: It is the thrust force F integrated over the burning time t.

   $I = F * \triangle t = I = \int F * \triangle t$

   $I = \int \dot{m} * V_{eq} * dt$

   Remember that $\dot{m}$ is the mass flow rate; it is the amount of exhaust mass per time that comes out of the rocket. Assuming the equivalent velocity remains constant with time, we can integrate the equation to get: $I = m * V_{eq}$

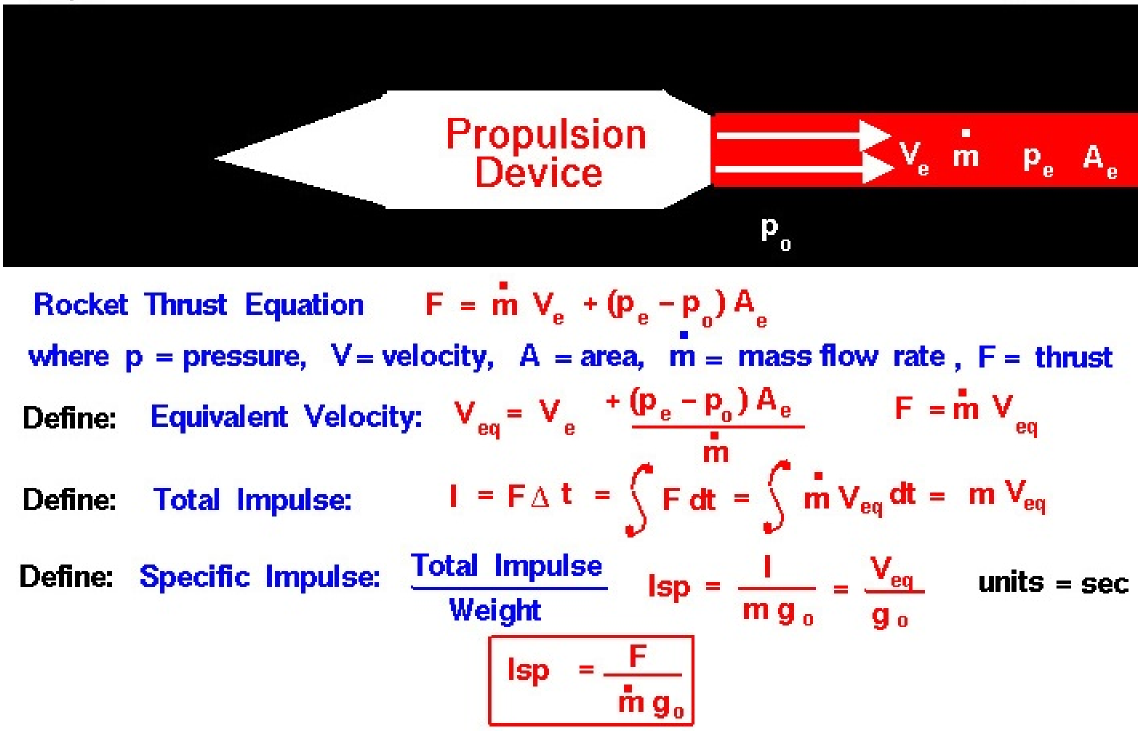

FIGURE 1.9: Rocket Performance parameters

where m is the total mass of the propellant. We can divide this equation by the weight of the propellants to define the specific impulse.

3. Specific Impulse:The specific impulse is the total impulse per unit weight of propellant. It is an important figure of merit of performance of the rocket system. $I_{sp} = \frac{V_{eq}}{g_0}$
$I_{sp} = \frac{F}{\dot{m}g_0}$

The performance of rocket is determined largely by the rocket-propellant combination and the total amount of usable propellant. The performance of propellants is characterized by the specific impulse, a measure of thrust produced per unit of propellant consumed per second. The unit of specific impulse is sec. The velocity that can be achieved by a rocket is directly proportional to the specific impulse of its propellants.

4. Effective Exhaust Velocity c: In a rocket nozzle, the actual exhaust velocity is not uniform over the exit cross section. For convenience, a uniform exit velocity is assumed which allows a one-dimensional description of the flow. The effective exhaust velocity c is the average equivalent velocity at which propellant is ejected from the vehicle. It is defined as

$c = I_{sp} * g_0$

The effective exhaust velocity c is given in m/sec.

5. Mass Ratio MR: The mass ratio of a vehicle is defined to be the final mass (after the rocket has consumed all usable propellant) divided by mass (before rocket operation). $M_{initial} = m_f = m_e + m_p$
$M_{final} = m_e$
$MR = m_f / m_e$
Mass ratio MR = The final mass is the mass of the vehicle after the rocket has ceased to operate when all the useful propellant mass is consumed and ejected. The final mass includes mass of guidance devices, navigational gear, payload, flight control system, vehicle structure tanks, control surfaces, communication equipment and unusable propellant etc

6. The Impulse-to-weight Ratio: The impulse to weight ratio of the propulsion system is defined as the total impulse divided by the initial vehicle weight . A high value indicates an efficient design. Impulse-to-weight ratio =
$\frac{I_t}{W} = \frac{I_t}{m_{vehicle} * g_0} = \frac{I_t}{(m_f + m_p) * g_0} = \frac{I_{sp}}{(\frac{m_f}{m_p}) + 1}$

7. The Thrust-to-weight Ratio: It is a measure of the acceleration (in multiples of g0) that the engine is capable of providing to its own mass. $\frac{F}{W} = \frac{F}{m_{vehicle} * g_0}$
Launch Vehicle: $\frac{F}{W_{initial}} > 1.0$
Orbit-Transfer vehicle: $\frac{F}{W} > 0.2$
Orbit-Maintenance Vehicle: $\frac{F}{W} > 0.1$

1.8 General Thrust Equation

The forces on a rocket change dramatically during a typical flight. During powered flight the propellants of the propulsion system are constantly being exhausted from the nozzle. As a result, the weight and mass of the rocket is constantly changing. Because of the changing mass, we cannot use the standard form of Newton's second law of motion to determine the acceleration and velocity of the rocket. This figure shows a derivation of the change in velocity during powered flight while accounting for the changing mass of the rocket. In this derivation, we are going to neglect the effects of aerodynamic lift and drag. We can add these effects to the final answer.

Let us begin by considering the rocket drawing on the left of the figure. M is the instantaneous mass of the rocket, u is the velocity of the rocket, v is the velocity of the exhaust from the rocket, A is the area of the exhaust nozzle, p is the exhaust pressure and p0 is the atmospheric pressure. During a small amount of time dt a small amount of mass dm is exhausted from the rocket. This changes the mass of the rocket and the velocity of the rocket and we can evaluate the change in momentum of the rocket as

Change in rocket momentum $= M*(u+du) - M*u = M*du$

We can also determine the change in momentum of the small mass dm that is exhausted at velocity υ as change in exhaust momentum $= dm*(u-\upsilon) - dm*u = -dm*\upsilon$.

So the total change in momentum of the system (rocket + exhaust) is change in system momentum= $M*du - dm*v$.

as shown on the figure. Now consider the forces acting on the system, neglecting the drag on rocket. The weight of the rocket is $M*g$ (gravitational constant) acting at an angle a to the flight path.

The pressure force is given by $(p-p_0)*A$ acting in the positive u direction.

Then the total force on the system is

force on the system $= (p-p_0)*A - M*g*cos(a)$

The change in momentum of the system is equal to the impulse on the system which is equal to the force on the system times the change in time dt. So we can combine the previous two equations:

$M*du - dm*\upsilon = [(p-p_0)*A - M*g*cos(a)]*dt$

If we ignore the weight force, and perform a little algebra, this becomes $M*du = [(p-p_0)*A]*dt + dm*\upsilon$

Now the exhaust mass dm is equal to the mass flow rate $\dot{m}$ times the increment of time dt. So we can write the last equation as

$M*du = [(p-p_0)*A + \dot{m}*\upsilon]*dt$

We introduce the equivalent exit velocity V_{eq} which is defined as

$V_{eq} = \upsilon + (p-p_0)*A/\dot{m}$

If we substitute the value of V_{eq} into the momentum equation we have

$Mdu = V_{eq} * \dot{m} * dt$

$\dot{m} * dt$ is the amount of change of the instantaneous mass of the rocket.

The sign of this term is negative because the rocket is losing mass as the propellants are exhausted.

$\dot{m} * dt = -dM$

Substituting into the momentum equation:

$Mdu = -V_{eq}dM$

$du = -V_{eq}dM/M$

We can now integrate this equation:

$\triangle u = -V_{eq}ln(M)$

where $\triangle u$ represents the change in velocity, and ln is the symbol for the natural logarithmic function.

The limits of integration are from the initial mass of the rocket to the final mass of the rocket.

The instantaneous mass of the rocket M, the mass is composed of two main parts, the empty mass m_e and the propellant mass m_p.

The empty mass does not change with time, but the mass of propellants on board the rocket does change with time: $M(t) = m_e + m_p(t)$

Initially, the full mass of the rocket mf contains the empty mass and all of the propellant at lift off.

At the end of the burn, the mass of the rocket contains only the empty mass:

$M_{initial} = m_f = m_e + m_p$

$M_{final} = m_e$

Substituting for these values we obtain:

$$\triangle u = V_{eq}ln(m_f/m_e)$$

This equation is called the **ideal rocket equation**.

There are several additional forms of this equation which we list here: Using the definition of the propellant mass ratio MR

$MR = m_f/m_e$

$\triangle u = V_{eq} * ln(MR)$

V_{eq} is related to the specific impulse I_{sp} : $V_{eq} = I_{sp} * g_0$

where g_0 is the gravitational constant. So the change in velocity can be written in terms of the specific impulse of the engine:

$$\triangle u = I_{sp} * g_0 * ln(MR)$$

If we have a desired $\triangle u$ for a maneuver, we can invert this equation to determine the amount of propellant required: $MR = \exp^{\frac{\triangle u}{(I_{sp}*g_0)}}$

Rocket thrust in the atmosphere:

If the exit area is A_{ex}, the exit pressure p_{ex}, and the altitude ambient pressure $p_a(p_{SL-a}$at sea level), then the altitude thrust is less than the thrust in a vacuum by the amount p_a.

Sea level thrust of the rocket, $F_{SL-a} = \dot{m}V_{ex} + A_{ex} * (p_{ex} - p_{SL-a})$

$F_j = \dot{m}V_{ex} + A_{ex} * (p_{ex} - p_a)$

Thus thrust at any altitude is

$F_j = F_{SL-a} + A_{ex} * (p_{SL-a} - p_a)$

$F_j = F_{SL-a} + A_{ex} * p_{SL-a} * (1 - \delta)$

where $\delta = \frac{p_a}{p_{SL-a}}$ pressure drop with altitude.

Rocket thrust in Vacuum:

$F_j = \dot{m} * V_{ex} + A_{ex} * p_{ex}$

$F_j = \frac{\dot{W}}{g} * V_{eq} + A_{ex} * p_{ex}$

From these equations the specific impulse(at sea level) is given as

$I_{sp} = \frac{F_{SL-a} + A_{ex} * p_{SL-a} * (1-\delta)}{\dot{m} * g}$

In vacuum

$I_{sp} = \frac{V_{ex}}{g}$ The characteristics of a rocket is also signified by a parameter called characteristics velocity,

$V^* = \frac{V_{ex}}{C_f}$

$C_f = \frac{F_j}{p_c A_t}$

C_f is the Thrust coefficient

p_c is combustion chamber pressure and A_t nozzle throat area

If weight flow rate of propellant is given as one can define a specific propellant consumption rate as

$\dot{W}_{sp} = \frac{\dot{W}}{F} = \frac{g}{I_{sp}}$

and weight flow coefficient as

$C_w = \frac{\dot{W}}{p_c A_t}$

Chapter 2

SOLID PROPULSION AND PYROTECHNICS

2.1 Introduction

Solid propellant motors are the simplest of all rocket designs. In solid propellant rocket motors-and the word ”motor” is as common to solid rockets as the word ”engine” is to liquid rockets. Solid propellant rocket motors have been credited with having no moving parts. This is still true of many, but some motor designs include movable nozzles and actuators for vectoring the line of thrust relative to the motor axis.They consist of a casing, usually steel, filled with a mixture of solid compounds (fuel and oxidizer) that burn at a rapid rate, expelling hot gases from a nozzle to produce thrust. When ignited, a solid propellant burns from the center out towards the sides of the casing.

In comparison to liquid rockets, solid rockets are usually relatively simple, are easy to apply (they often constitute most of the vehicle structure), and require little servicing; they cannot be fully checked out prior to use, and thrust cannot usually be randomly varied in flight There are two families of solids propellants: homogeneous and composite. Both types are dense, stable at ordinary temperatures, and easily storable.They also make affordable and compact high thrust levels, which are very difficult to reach with liquid rocket engines. Further, the solid nature of the fuel renders.The rockets exceptionally durable. They can be constructed, fuelled and then put to storage for numerous years without the performance characteristics changing noticeably when finally ignited. These properties result in SRMs having excellent scalability and are thus used in a wide variety of applications and environments. This includes thrusters controlling attitude of nano satellites to full fledged booster rockets such as the Reusable Solid Rockets Motor (RSRM) on the Space Shuttle.

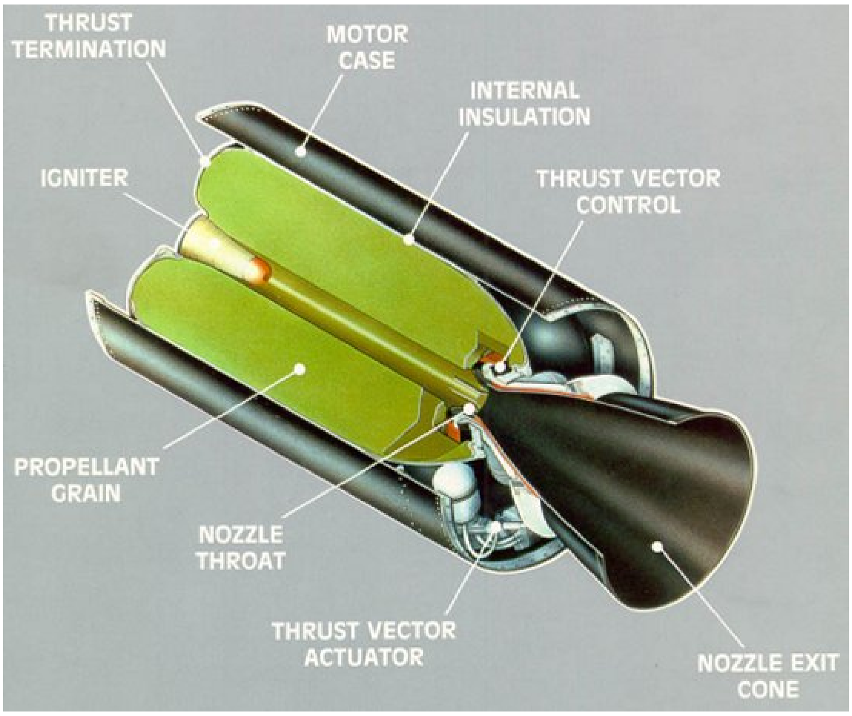

FIGURE 2.1: Solid rocket motor

2.2 Motor Components

2.2.0.1 Casing

The casing makes up the external shell of an SRM. It acts as a supporting structure and combustion chamber with the grain occupying a majority of space inside. The latter implies that the casing also serves as a high pressure vessel. Casings are thus typically built of high-strength steel alloys or filament-reinforced composite materials capable of withstanding very high loads. Common configurations include oblate spheroids and cylinders with either elliptical or hemispherical heads. Ratio between the length and diameter of cylindrical casings typically ranges between 2 and 5 in most launcher related applications. An overly low ratio has negative influence on the axial compressive drag loads exerted during ascent whereas a disproportionately high value adversely affects handling, stability and rigidity. This renders the rocket difficult to control whilst also induces buckling and bending problems.

2.2.0.2 Nozzle

SRM nozzles allow the expansion and acceleration of combustion gases through a converging diverging nozzle. The geometry and mounting technique employed has a significant effect on the thrust produced and thus, a number of nozzle categories currently exist.

1. For space launch applications **submerged nozzles** are used extensively. These nozzles have a large portion of the structure sunken into the combustion chamber that reduces the motor length and inert mass. This is especially beneficial for upper stage rockets due to the implied limitation of the length and mass of the inter-stage structure.
2. **Fixed nozzles:** Fixed nozzles are generally not submerged and do not provide thrust vector control.
3. **Movable nozzles:** Movable nozzles can provide pitch and yaw control and two are needed for roll control. Movable nozzles are typically submerged and use a flexible sealed joint or bearing with two actuators.
4. **Extendible nozzle:** Extended nozzle improves specific impulse by doubling or tripling the initial expansion ratio, thereby significantly increasing the nozzle thrust coefficient.

2.2.0.3 Igniters

Two categories of igniters are found in SRMs, pyrotechnic igniters and pyrogen igniters. The first type utilizes explosives like black powder or small pellets of propellants that once ignited, results in a large surface burning area. The heat produced by this process then ignites the main grain. This type is commonly used for small to medium sized SRMs. The second type consists principally of miniature rocket motors with fast burning grains. Once ignited, the resulting flame

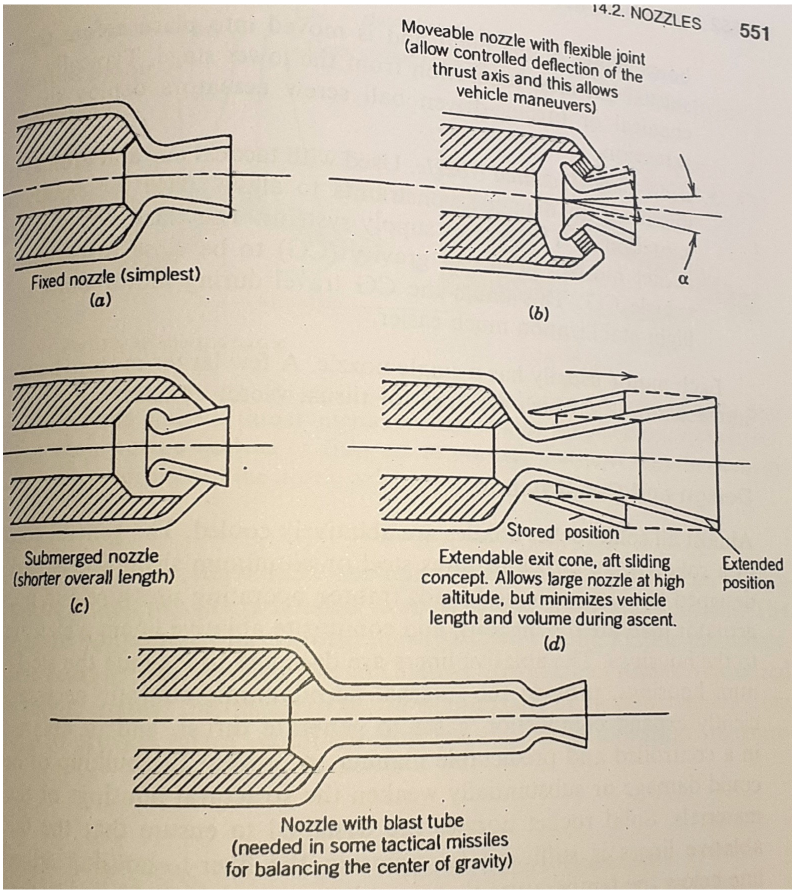

FIGURE 2.2: Classification of Nozzles in SRM

spreads throughout the gas cavity area. The hot gases of the flame then interact and effectively ignite the main grain. This type of igniters is more common amongst larger SRMs. For both types, igniters are often located at the forward end, opposite side of the nozzle. The shape of igniters varies to a greater extent depending on the ignition method and implementation utilized. However, as igniters tend to be small in relation to the main grain, a first approximation of modeling them as simple cylinders is acceptable

2.2.0.4 Insulations

With flame temperature of between 1500K-3000K in typical combustion chambers during combustion, insulation in-between the grain, internal surface of the casing and other vital components is vital. Similar to nozzles, insulation inside the casing commonly employs ablation as the main method of cooling.

2.2.0.5 Grain

The grain is the mass of processed solid propellant inside the rocket motor. In a typical SRM, up to 96 percent of the total mass consists of propellant grain For any propellant, additives may control the burnrate, either to accelerate or to slow the rate. All propellants are processed into a similar basic geometric form, referred to as a propellant grain. As a rule, propellant grains are cylindrical in shape to fit neatly into a rocket motor in order to maximize volumetric efficiency. The grain may consist of a single cylindrical segment or may contain many segments.
It is important to recognize that the burning area of a propellant grain is a key parameter in determining the performance of a rocket motor. The primary function of a propellant grain is to produce combustion products at a prescribed flowrate defined by: Performance of a grain is usually measured in a thrust vs time diagram.

2.3 Classification of Solid rocket Motors

2.3.0.1 On the basis of Case design:

In rocketry the designing of casing and hardware plays a vital role which is the replication of pressure vessel design. The temperatures induced inside the casing are of the order of 1000 C to 3000 C. The final temperature experienced by the casing at it's external surface (after insulation) is 100 C .To withstand these high temperatures some ablative liners are provided inside the casing.

2.3.0.2 On the basis of Grain Installation:

There are two methods of holding the grain in case: Cartridge-loaded and case-bonded grains.
Cartridge-loaded or freestanding grains are manufactured separately from the case (by extrusion or by casting into a cylindrical mold or cartridge) and then loaded into or assembled into the case. In case-bonded grains the case is used as a mold and the propellant is cast directly into the case and is bonded to the case or case insulation. Free-standing grains can more easily be replaced if the propellant grain has aged excessively. Cartridge-loaded grains are used in some small tactical missiles and a few medium-sized motors. They often have a lower cost and are easier to inspect.

The case-bonded grains give a somewhat better performance, a little less inert mass (no holding device, support pads, and less insulation), a better volumetric loading fraction, are more highly stressed, and often somewhat more difficult and expensive to manufacture. Today almost all larger motors and many tactical missile motors use case bonding.

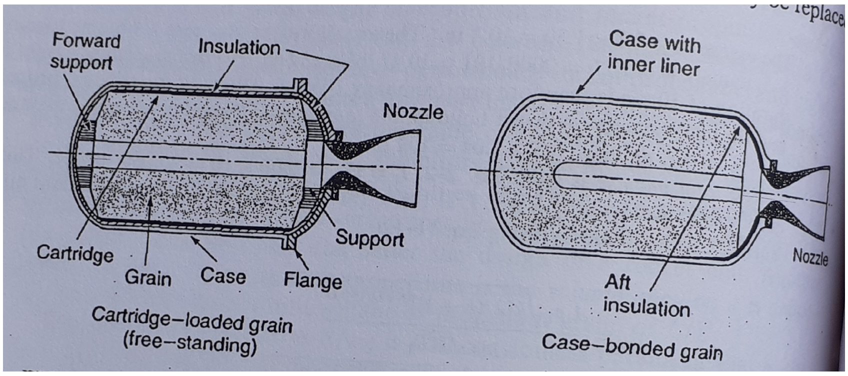

FIGURE 2.3: Classification of SRM based on Grain Installation

2.3.0.3 On the basis of Thrust action:

There are three methods in which Solid rocket motors are classified based on thrust action: Progressive, Regressive and Neutral burning.

1. Progressive Burning: Burn time during which thrust, pressure, and burning surface area increase.
2. Regressive Burning: Burn time during which thrust, pressure, and burning surface area decrease.
3. Neutral Burning: Motor burn time during which thrust, pressure, and burning surface area remain approximately constant, typically within about ±15 percentage. Many grains are neutral burning.

2.3.0.4 On the basis of Grain Configuration:

Solid grains are also classed by the shapes of their exposed burning surfaces and the manner in which the propellants are burned out of the case Cylindrical Grain: A grain in which the internal

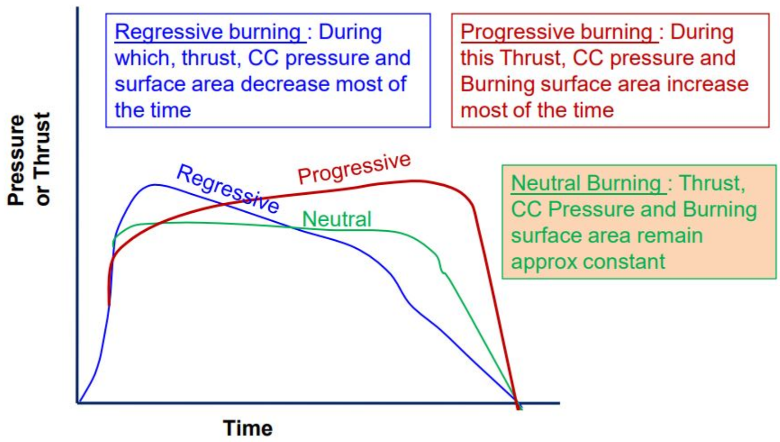

FIGURE 2.4: Classification of SRM based on Thrust action

cross section is constant along the axis regardless of perforation shape. The thrust (and chamber pressure) that a rocket motor generates is proportional to the burning area at any particular instant in time. This is referred to as the instantaneous burning area. The burning surface at any point recedes in the direction normal (perpendicular) to the surface at that point, the result being a relationship between burning surface and web distance burned that depends almost entirely on the grain initial shape and restricted (inhibited) boundaries. This important concept is illustrated in Figure, where the contour lines represent the core shape at successive moments in time during the burn. Notice that the shape of the thrust-time curve changes, with the vertical lines corresponding the the same successive moments during the burn.

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 initial thrust, followed by an almost constant lower thrust. As can be seen , the star grain provides an approximately neutral burn, as the surface area remains fairly constant throughout the burn duration. A neutral burn is usually desirable because it provides for greater efficiency in delivery of total impulse, as a nozzle operates most efficiently at a constant chamber pressure.

These are some of the cross sectional designs and the sidewise views of the grains of the solid propellant rockets. Now, you can see here some of them are mentioned as restricted burning types. The restricted burning times have these solid and linings around them, so a and b in this figure are restricted burnings. Around this, are the linings, which restrict the burning. They do not allow those surfaces to be participating in the combustion process and hence, this is called end burning. It means only one end is open for combustion or burning and all the other surfaces are closed to the burning process. In the second one b, we have an internal burning, which means the internal surface actually has a star shape. This star shape is open to combustion or open to burning all the other surfaces including the two ends, specially this end is restricted and is not allowed for combustion or burning. Only the internal star shaped cross section is opened for burning. Now, the question here is why this star shape or any such shape is created. Essentially, if you look in closely compared to a circle, if you have a circle over here, the surface area of this star shape is substantially more than any shape that you put here; whether circle or ellipse or square. A star shaped would have more surface area of burning and this more surface area actually gives you faster burning. So, the star shaped has been created for enhanced burning capability of the rocket.

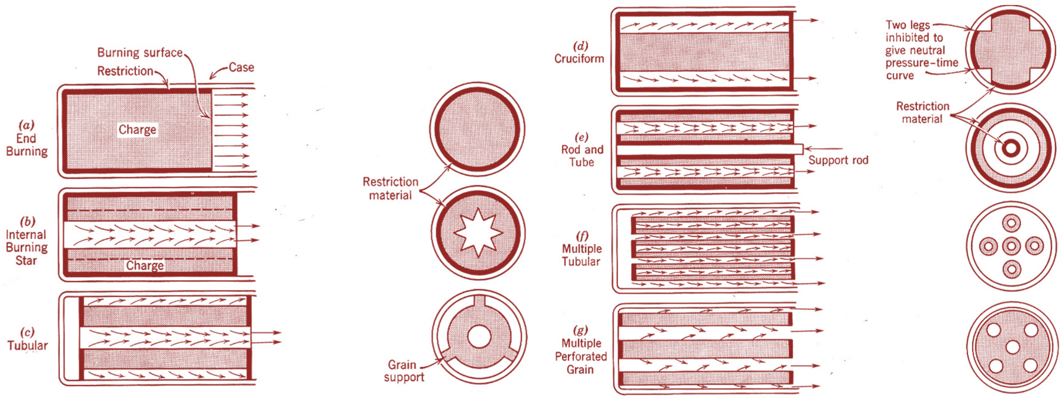

FIGURE 2.5: Various grain configurations1

2.3.0.5 On the basis of Propellant:

Most modern solid-propellant grains belong to one of two classes, double-base or composite grain. The double-base propellant is a mixture of two very energetic compounds, either one of which alone would make a rocket propellant. Usually the two constituents are nitroglycerin $[C_3H_5(ONO_2)_3]$ and nitrocellulose $[C_6H_7O_2(ONO_2)_3]$. As the chemical formulas indicate, both the fuel (carbon and hydrogen) and the oxidizer (oxygen) atoms are contained in each of these molecules; both

substances are monopropellants which burn without any added oxidizer. The nitrocellulose provides physical strength to the grain, while nitroglycerin is a high-performance and fast-burning propellant.
A **composite grain** is so named because it is formed of a mixture of two or more unlike compounds into a composite material with the burning properties and strength characteristics desired. None of these constituent compounds would make a good propellant by itself; instead, one is usually the fuel component, another the oxidizer. The most modern of the composite propellants use a rubbery polymer which acts as the fuel and as a binder for the crumbly oxidizer powder. The oxidizer is generally a finely ground nitrate or perchlorate crystal, as, for example, potassium nitrate ($KN0_3$J or ammonium perchlorate (NH_4CI0_4). The composite mixture can be mixed and poured like cake batter, cast ninto molds or into the motor case itself, and made to set (cure) like hard rubber or concrete.

2.4 Burning Process

A correct chemical mixture of fuel and oxidizer will support combustion when exposed to high temperature and gas flow; it will continue to burn as long as the gaseous products of combustion are allowed to escape from the burning surface. The rate at which hot gases are produced by the burning propellant depends on the total area over which burning is occurring A_b; the rate at which burning is progressing into the propellant $\dot{r}$; and the density of propellant being transformed into gas ρ.
The flow of combustion gases off the burning surface is described by the rate equation:

$$\dot{w} = A_b \dot{r} \rho \tag{2.1}$$

It is a characteristic of any solid propellant that its burning rate at any point on the

1. The composition of the propellant at that point.

2. The pressure of the gases surrounding the point

3. The temperature of the grain at that point just as the ”burning zone” approaches

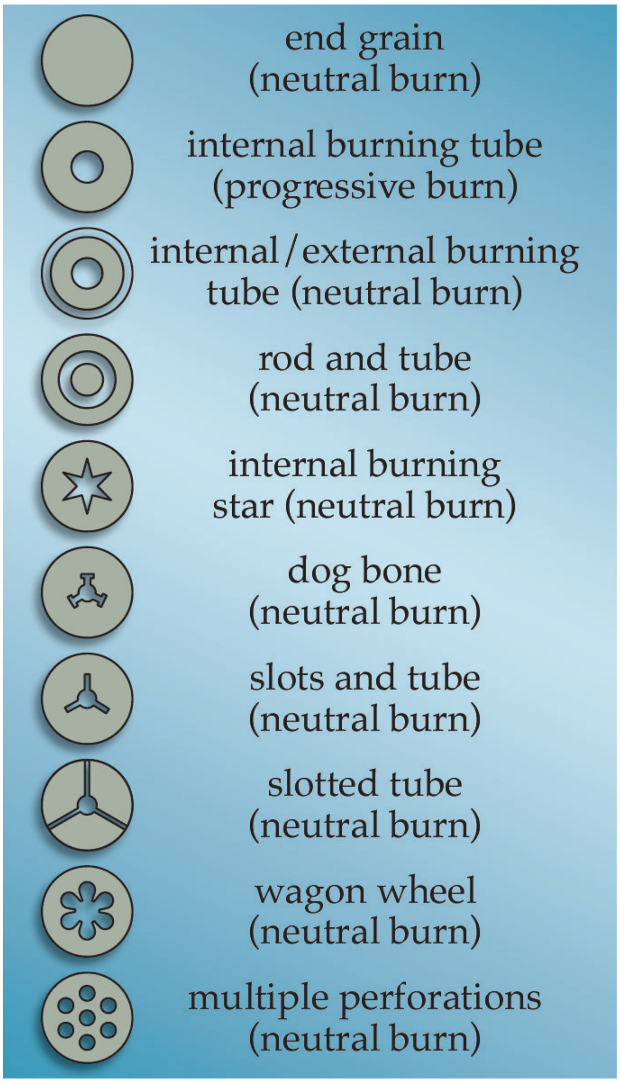

FIGURE 2.6: Classification of Grains

These characteristics at each point on the grain are averaged for the entire grain in grain surface is determined by: the general, solid-propellant, burning-rate equation

$$\dot{r} = a_B P_c^n \tag{2.2}$$

where $\dot{r}$ is the instantaneous burn rate (in. /sec), a_B is the burn-rate constant (which varies slightly with the overall temperature of the grain), P_c^n, is the instantaneous motor chamber pressure (psi), and n is the burn-rate exponent for the particular propellant (typical values range from 0.4 to almost 1. 0).

Combining the burning-rate 2.2 with the weight-flow-of-gas-produced equation 2.1 yields

$$\dot{w} = A_b a_B P_c^n \rho \tag{2.3}$$

Now, for any rocket device, the thrust produced by expansion of exhaust gases through a nozzle can be expressed as

$$F = C_F A_t P_c \tag{2.4}$$

where F is the thrust (Ib), C_F is the nozzle thrust coefficient (a constant which is a measure of the expansion efficiency of the nozzle and the properties of the propelling gases), A_t is the nozzle throat area, and P_c is the rocket chamber pressure.
But, thrust is also given by the equation

$$W = I_{sp} \dot{W} \tag{2.5}$$

$$\tag{2.6}$$

where $\dot{W}$ is the gas weight flaw through the nozzle (lb/sec) and I_{sp} is the engine specific impulse (a measure of propellant energy release and efficiency of gas expansion through the nozzle). Combining 2.5 and 2.4 gives

$$\dot{W} = \frac{F}{I_{sp}} = \frac{C_F A_t P_c}{I_{sp}} \tag{2.7}$$

In the rocket, a steady-state condition is reached when the rate of gas produced equals the rate of gas flow out of the chamber.

$$\dot{W}_{produced} = \dot{W}_{out} \tag{2.8}$$

or, from 2.3 and 2.7

$$A_b a_B P_c^n \rho = \frac{C_F A_t P_c}{I_{sp}} \tag{2.9}$$

At any instant during the firing, everything in 2.9 is invariable except the chamber pressure. Thus, the pressure in the rocket chamber stabilizes at the instantaneous value found by solving 2.9:

$$P_c = (\frac{C_F A_t}{A_b a_B I_{sp} \rho})^{n-1} \tag{2.10}$$

And so, the motor designer can control the pressure at which the rocket will operate by:
(a) Selecting the propellant, thereby fixing I_{sp}, ρ, a_B and n.
(b) Designing a nozzle size and configuration, thereby fixing A_t and C_F.
(c) Designing the grain burning surface to make A_b vary as desired during the firing.
Solid-motor grain design concentrates on the problem of tailoring the thrust curve by configuring the burning surface area to give the desired thrust with time Thrust curves are typically progressive, regressive, neutral, or a combination of these, as shown in . Also noted are some of the grain port shapes which will produce these thrust variations by the manner in which their burning surfaces vary in area as burning proceeds.
Another major design problem comes in the elimination of long thrust tail off, or decay period, at the end of rocket firing. Long tail off time wastes propellant by burning it inefficiently at low pressure for a relatively long time. Long tail off also endangers the motor case by exposing it to hot gases while it is no longer protected by propellant. Short, abrupt tail off is desirable but difficult to achieve, particularly in complex star grain designs. In such grains, the nature of the burning-surface shape gives decreased burning area near the end of the firing because of residual propellant slivers.

2.5 Internal Ballistics

The rocket motor's operation and design depend on the combustion characteristics of the propellant, its burning rate, burning surface, and grain geometry. The branch of applied science describing these is known as internal ballistics. The burning surface of a propellant grain recedes in a direction essentially perpendicular to the surface. The rate of regression, usually expressed in cm/ sec, mm/sec, or in./sec, is the burning rate r.
Burning rate is a function of the 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 or increase percentage of existing catalyst.
2. Decrease the oxidizer particle size.
3. Increase oxidizer percentage.
4. Increase the heat of combustion of the binder and/or the plasticizer.
5. Imbed wires or metal staples in the propellant.

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

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

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

1. Standard strand burners, often called Crawford burners. A strand burner is a small pressure vessel (usually with windows) in which a thin strand or bar of propellant is ignited at one end and burned to the other end. The burning rate can be measured by electric signals from embedded wires, by ultrasonic waves, or by optical means
2. Small-scale ballistic evaluation motors.
3. Full-scale motors with good instrumentation.

Erosive burning refers to the increase in the propellant burning rate 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.

2.6 Igniter Design

The igniter in a solid rocket motor generates the heat and gas required for motor ignition. Motor ignition must usually be complete in a fraction of a second for all but the very large motors. Conventionally, the ignition process is divided into three phases for analytical purposes:

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

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

2.7 Igniter Hardware

Since the igniter propellant mass is small (often less than 1 propellant) and burns mostly at low chamber pressure (low / 5), it contributes very little to the motor overall total impulse. It is the designer's aim to reduce the igniter propellant mass and the igniter inert hardware mass to a minimum, just big enough to assure ignition under all operating conditions. There are two basic types: pyrotechnic igniters and pyrogen igniters.

2.7.0.1 Pyrotechnic igniters

In industrial practice, pyrotechnic igniters are defined as igniters using solid explosives or energetic propellant-like chemical formulations (usually small pellets of propellant which give a large burning surface and a short burning time) as the heat-producing material.

1. Firstly, on receipt of an electrical signal the initiator releases energy of a small amount of sensitive powdered pyrotechnic housed within the initiator, commonly called the squib or the primer charge.

2. Secondly, the booster charge is ignited by heat released from the squib.

3. Finally, the main ignition charge propellants are ignited.

A special form of pyrotechnic igniter is the surface-bonded or grain-mounted igniter. Such an igniter has its initiator included within a sandwich of flat sheets; the layer touching the grain is the main charge of pyrotechnic. This form of igniter is used with multipulse motors with two or more end-burning grains.

2.7.0.2 Pyrogen igniters

A pyrogen igniter is basically a small rocket motor that is used to ignite a larger rocket motor. The pyrogen is not designed to produce thrust. All use one or more nozzle orifices, both sonic and supersonic types, and most use conventional rocket motor grain formulations and design technology. Heat transfer from the pyrogen to the motor grain is largely convective, with the hot gases contacting the grain surface as contrasted to a highly radiative energy emitted by pyrotechnic igniters. For pyrogen igniters the initiator and the booster charge are very similar to the designs used in pyrotechnic igniters. Reaction products from the main charge impinge on the surface of the rocket motor grain, producing motor ignition.-
Two approaches are commonly used to safeguard against motor misfires, or inadvertent motor ignition; one is the use of the classical safe and arm device and the second is the design of safeguards into the initiator. Functionally, the safe and arm device serves as an electrical switch to keep the igniter circuit grounded when not operating; in some designs it also mechanically misaligns or

blocks the ignition train of events so that unwanted ignition is precluded even though the initiator fires.

2.8 Pyrotechnics

Pyrotechnics refers to making fire by chemical reaction, with the goal to produce light, heat, noise, or gases. It is always done by combustion of a fuel and an oxidizer (a red-ox reaction), but distinguishes from normal combustion in the speed: combustion refers to slow processes whereas pyrotechnics is associated to almost instantaneous combustion (solid-rocket propellants being in between).
In aerospace technology pyrotechnics refer to a broad family of sophisticated devices utilizing explosive, propellant and pyrotechnics to accomplish:

Initiation, Jettison, Release, Valving, time delay and Actuation.

For pyrotechnics to be effective, fuel and oxidizer must be premixed (double-base pyrotechnics) or, even better, they should be part of the same molecule (single-base pyrotechnics) with zero or slightly positive oxygen balance, they should be highly exothermic, and they should be in condense form and generate a lot of gas. Nitrogen atoms are found in most explosives, because they yield nitrogen molecules that release great energy and expanding gases. In double-base pyrotechnics, the oxidizing agents may be nitrates, chlorate, peroxides, oxides, chromate's and perchlorate (the best), all providing oxygen. The reducing agents may be charcoal (carbon), sulfur, or metal powders. Notice that all practical double-base pyrotechnics are powder solids mixed-up, with some gluing agent to keep them bounded, because liquid mixtures are too unstable.

2.8.0.1 Classification

By physical state pyrotechnics may be grouped as:

1. Solids. The majority of cases, because they are more stable and easier to handle.

2. Liquids. Very unstable even if single-base, as nitrocellulose and nitroglycerine. Separate liquids, like LH2 and LOX used in cryogenic rockets, are treated as combustion processes.

3. Gases. There are no single-base pyrotechnic gases (they would decompose), and premixed explosive gases are considered under normal combustion.

By use, pyrotechnics are grouped as:

1. Explosives. Substances that, by chemical decomposition, generate a supersonic reaction wave, propagating at several km/s within the material, generating a lot of hot expanding gases. They are also called high-explosives, and the process is known as detonation; e.g. dynamite. Sensitive materials that can be exploded by a relatively small amount of heat or pressure are called primary explosives (e.g. nitroglycerine, lead azide), and more stable materials secondary explosives (e.g. TNT, ANFO).

2. Propellants. Substances that, by chemical decomposition, generate a subsonic reaction wave, propagating at a few cm/s or m/s within the material, generating a lot of hot expanding gases. They are also called low-explosives, and the process is known as deflagration, as in combustion; e.g. black powder.

2.8.0.2 Applications

According to their purpose, pyrotechnics may be classified as:

1. Blasters, for mining, tunneling, demolition, quick-release devices, and weaponry (warhead). They are high-explosives that undergo supersonic combustion when detonated by a low-explosive or shock-wave (they slowly burn if just approached by a flame). If the blast is just to cause an abrupt noise, with insignificant blasting, the device is called a firecracker (see below).

2. Propellants, for rockets and weaponry. They generate a large gas stream (like all other pyrotechnics) that is channeled with one free end to give propulsive thrust to a projectile or to the combustor body. The main difference between rocket propellants and gun propellants is the working pressure reached, which in rockets is around 10 MPa, and in guns more than 100 MPa, with the consequent change in burning rate).

3. Launch Escape Tower (LET) separation
4. Separation rocket ignition
5. Booster stage/Lunar Module separation
6. Forward heat shield jettison
7. Spacecraft/Lunar Module Adapter panel separation
8. Lunar Module landing gear deployment
9. Lunar Module propulsion systems pressurization and activation
10. Parachute deployment and release
11. Electrical circuit opening and closing
12. Line/cable cutting – timed and delayed-time
13. Spacecraft vehicle destruction, if loss of control or other catastrophe.

2.9 Thrust vector Control

In addition to providing a propulsive force to a flying vehicle, a rocket propulsion system can provide moments to rotate the flying vehicle and thus provide control of the vehicle's attitude and flight path. It directs thrust in a direction other than parallel to the vehicles longitudinal axis. The reasons for TVC are:

1. to willfully change a flight path or trajectory (e.g., changing the direction of the flight path of a target-seeking missile);
2. to rotate the vehicle or change its attitude during powered flight;
3. to correct for deviation from the intended trajectory or the attitude during powered flight; or
4. to correct for thrust misalignment of a fixed nozzle in the main propulsion system during its operation, when the main thrust vector misses the vehicle's center of gravity.

Several TVC mechanisms that have been used in production vehicles are:

1. **Rotating Nozzle**: The rotating nozzle has no throat movement. These nozzles work in pairs and are slant-cut to create an area of under expansion of exhaust gases on one side of the nozzle. This creates an unbalanced side load and the inner wall of the longer side of the nozzle. Rotation of the nozzles moves this side load to any point desired and provides roll, yaw and pitch control. This system is simple but produces slow changes in the velocity vector. Rotating nozzles are usually supplemented with some other form of TVC.

2. **Swiveled Nozzle** The swiveled nozzle changes the direction of the throat and nozzle. It is similar to gimbaling in liquid propellant engines. The main drawback in using this method is the difficulty in fabricating the seal joint of the swivel since this joint is exposed to extremely high pressures and temperatures.

3. **Movable Control Surfaces**- Movable Control Surfaces physically deflect the exhaust or create voids in the exhaust plume to divert the thrust vector. This method includes jet vanes, jet tabs, and mechanical probes. These TVC approaches are all based on proven technology with low actuator power required. They suffer from erosion and cause thrust loss with any deflection. A similar system is the jet avator, a slipring or collar at the nozzle exit which creates an under expansion region. The jet avator is a movable surface which allows the under expanded region to be moved 360 degrees around the rocket nozzle to produce pitch and yaw control. This system was developed for the Polaris SLBM.

2.10 Main Failure Modes

1. Case breach: Local burning-through of the rocket case (see Figure 1 in Home section) which can result in catastrophic impact on the TVC system (see Figure 1, and Figure 2 (in Home section) for the test firing of TD31).

2. Case burst: Nozzle blocking or bore choking which results in overpressure in a combustion chamber.

3. Nozzle Failure: Deformations of the nozzle which, in particular, can reduce the thrust being generated. This effect can be induced by ablation process and abrupt breaking off of large

pieces of the propellant. These pieces or a cloud of solid particles in the exhaust gases accumulating at the nozzle inlet can block temporarily the nozzle throat (transient nozzle blocking fault). A nonuniform failure of the nozzle (such as loosing a chunk of the aft exit cone, or partially failing a joint) will result in a non-axial component of thrust. A failure would also result in the plume moving closer to the aft skirt, causing increased heating and adversely affecting the TVC system.

4. Bore choking: Bore choking occurs when the propellant deforms (bulges) radially inward and disrupts the exhaust gas flow, causing a choked flow condition inside the motor. Bore choking can be most likely realized near radial slots and segment joints between two sections with a smaller radius of the aft section. This critical effect is typically caused by localized areas of low pressure arising near such inhomogeneous. Development testing has shown that this fault was observed, for example, in the primary construction of the Titan IV (see Figure 3). Bore choking has the potential of causing booster over-pressure and catastrophic failure.

5. Debonding: Potentially large parts of the propellant debond from the liner and become loose. They can bend and stick inside the bore. In the large rocket with the large aspect ratio of the bore volume the depleted propellant can significantly obscure the bore volume leading to chocking.

6. Propellant structural failure: Critical defects are cracks and voids in solid propellant and slots of booster joint segments. These defects can stimulate the increase of local burning rate that can result in abruption of lager enough piece of the propellant. This piece can stick to a narrow place of the burning propellant or choke minimum cross section of the nozzle. This can cause a sharp catastrophic jump of the booster trust and overpressure in the chamber head.

7. Combustion instabilities: Instabilities of combustion in the system.

8. Structural Failure: Large-scale buckling in the case or first to second stage coupling could result in a non-linear vehicle causing excessive aerodynamic drag. Small-scale buckling may alter stress/strain levels in the case.

9. Ignition failure: Failure of ignition in system.

2.11 Important parameters

Silver The solid propellant residue left unburnt after web extinguishes.

Liner The purpose of linear is to extinguish the flame in a solid rocket motor and insulate the case.

Web The maximum radial thickness of solid propellant grain.

The Volumetric Loading Fraction

is defined as the fraction of grain volume to case volume, and relates the volumetric efficiency of the motor, as well as a measure of performance efficiency:

$$V_1 = \frac{V_p}{V_a} = \frac{I_t}{\rho_p V_a} \tag{2.11}$$

where V_p is the grain volume, V_a is the available chamber volume, It is the total impulse (deliverable), and I_{sp} is the propellant specific impulse.

Ullage

The volume of the gas space above the propellant in a propellant tank. It is expressed as percentage of propellant volume.

Web Thickness

The minimum dimension from the port surface to liner interface measured radially.

Web Fraction

is the ratio of propellant web thickness to grain outer radius, and is given by:

$$V_1 = \frac{D-d}{D} = \frac{2rt_b}{D} \tag{2.12}$$

where t_b is the motor burn time. Clearly, to maximize burn duration, it is necessary to maximize the web fraction (i.e. thickness). The ”price” for maximizing web thickness is reduction of the grain core diameter.

Port-to-Throat area ratio

is given by the flow channel cross-sectional area to the nozzle throat cross-sectional area:

$$\frac{A_p}{A_t} = \frac{\pi D^2(1 - V_1)}{4A_t} \tag{2.13}$$

Length-to-Diameter ratio

is the grain overall length in relation to the grain outer diameter. This parameter is very significant in motor design, as larger L/D values tend to result in greater erosive burning effects (including negative erosive burning). High L/D values tend to generate high mass flow rate differentials along the grain length, and may be best served with a tapered core or stepped core diameters (largest nearer the nozzle).

2.12 Desirable Properties of Solid Rocket Motor

Solid rocket motor should have high release of chemical energy, they should have lower molecular weight, no deterioration of mechanical and chemical properties during storage. Most of the solid propellant rockets are created in specialized factories. They have to be stored for long period, during which there should not be any drop in the mechanical or chemical properties because they need to retain them during their actual operation inside the rocket chamber. They also need to be ensured that during this storage period, which could be many months and they must be unaffected by the atmosphere or atmospheric conditions. They should not be amenable to high temperature and pressure. For combustion, initiation combustion needs to take place at certain temperature and pressure. It should not get into combustion, before that temperature or pressure is arrived. So, these are some of the basic properties for solid propellant rockets that are used for choosing the solid propellants.

2.13 Solid Rocket Motor Performance

There are two major indicators of rocket system performance, specific impulse I and mass fraction $M.F$:

$$I_{sp} = \frac{Thrust}{Rate of propellant usage}$$

$$M.F. = \frac{(Initial mass) - (Burnout mass)}{Initial mass}$$

$$M.F. = \frac{Weight of propellant}{Weight of total propulsion system}$$

Solid propellants are typically less energetic than the better liquid-propellant combinations. Modern solids have sea-level $I_s p$ values in the range of 220 to 250 seconds, compared with over 350 seconds for the liquid-oxygen/liquid-hydrogen combination. On the other hand, solid-rocket mass fractions can be quite high because there are no valves, piping, or pumps to add to the inert weight. High-performance upper-stage solid motors typically attain mass fractions nearing 0.95 through the use of filament wound glass cases and refractory-lined nozzles.
Even the large solid boosters have mass fractions exceeding 0.90, a value which liquid-fueled missiles with very thin tank walls (e. g., Atlas) can barely achieve. The solid rocket's real advantages are its strength, since the propellant grain has considerable strength of its own and also acts as a stiffener and shock dampener, and its instant readiness, since there are no fuel tanks to be filled just prior to firing and launch.

2.14 Classification of Nozzles for solid propellant rocket motors

Nozzles for solid propellant rocket motors can be classified into five categories.

1. Fixed Nozzle. Simple and used frequently in tactical weapon propulsion systems for short-range air, ground, and sea-launched missiles, also as strap-on propulsion for space launch vehicles such as Atlas and Delta, and in spacecraft motors for orbital transfer. Typical throat

diameters are between 0.25 and 5 in. for tactical missile nozzles and approximately l O in. for strap-on motors. Fixed nozzles are generally not submerged and do not provide thrust vector control (although there are exceptions).

2. Movable Nozzle. Provides thrust vector control for the flight vehicle. One movable nozzle can provide pitch and yaw control and two are needed for roll control. Movable nozzles are typically submerged and use a flexible sealed joint or bearing with two actuators 90 degrees apart to achieve omniaxial motion. Movable nozzles are primarily used in long-range strategic propulsion ground- and sea launched systems (typical throat diameters are 7 to 15 in. for the first stage and 4 to 5 in. for the third stage) and in large space launch boosters such as the Space Shuttle reusable solid rocket motor, Titan boost rocket motor, and Ariane V solid rocket booster, with throat diameters in the 30 to 50 in. range.

3. Submerged Nozzles. A significant portion of the nozzle structure is submerged within the combustion chamber or case. Submerging the nozzle reduces the overall motor length somewhat, which in turn reduces the vehicle length and its inert mass. It is important for length-limited applications such as silo- and submarine-launched strategic missiles as well as their upper stages, and space motor propulsion systems.

4. Extendible Nozzle. Commonly referred to as an extendible exit cone, or EEC, although it is not always exactly conical. It is used on strategic missile propulsion upper-stage systems and upper stages for space launch vehicles to maximize motor-delivered specific impulse. It has a fixed low-area ratio nozzle section which is enlarged to a higher area ratio by mechanically adding a nozzle cone extension piece. The extended nozzle improves specific impulse by doubling or tripling the initial expansion ratio, thereby significantly increasing the nozzle thrust coefficient. This system thus allows a very high expansion ratio nozzle to be packaged in a relatively short length, thereby reducing vehicle inert mass. The nozzle cone extension is in its retracted position during the boost phase of the flight and is moved into place before the motor is started but after separation from the lower stage. Typically, electromechanical or turbine-driven ball screw actuators deploy the exit cone extension.

5. Blast-Tube-Mounted Nozzle. Used with tactical air- and ground-launched missiles with diameter constraints to allow space for aerodynamic fin actuation or TVC power supply systems. The blast tube also allows the rocket motor's center of gravity (CG) to be close to

or ahead of the vehicle CG. This limits the CG travel during motor burn and makes flight stabilization much easier. Each motor usually has a single nozzle. A few larger motors have had four movable nozzles, which are used for thrust vector control.

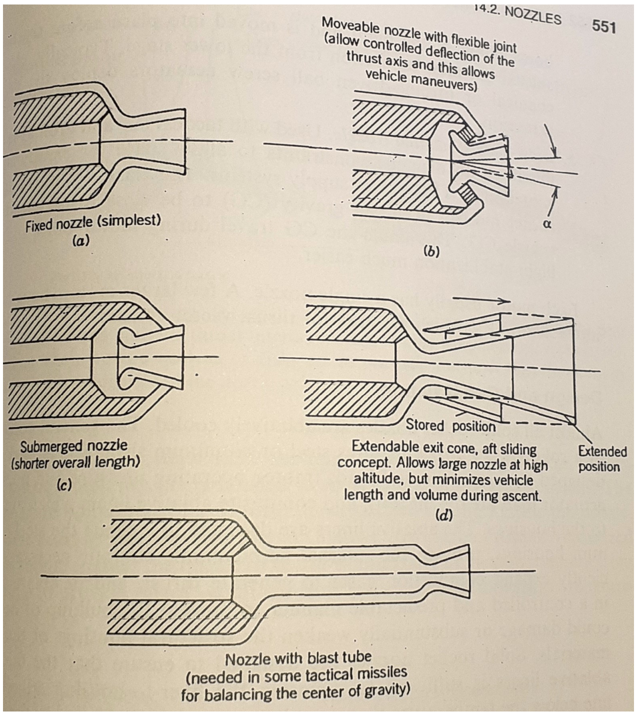

FIGURE 2.7: Classification of nozzles in SRM

Chapter 3

LIQUID PROPULSION AND CONTROL SYSTEMS

3.1 Introduction

Liquid propellant engines pioneered by Pedro Paulet in 19th century. The first static test firing took place in 1923. Robert Goddard flies first liquid propellant engine (LOX/gasoline) March 16, 1926 in Auburn. V-2 (LOX/ethanol) developed in the 1930s. Early proponents of liquid propulsion include Tsiolkovsky, Goddard, and Oberth.

Liquid propellant rocket engines are widely used and play a very important role in aerospace. The function of a LPRE is to generate thrust through chemical reactions, which usually release thermal energy from the chemical energy of the propellants. The pressure generated from the thermal energy imparts a momentum to the reaction products. Then a momentum in the opposite direction is imparted to the rocket and propels a vehicle in space. A LPRE system usually consists of thrust chamber assembly, propellant feed system, turbine-drive system (for turbo-pump LPRE), and propellant control system, etc. A liquid propellant rocket engine is very complex and difficult to design and analyze because of many coupled subsystems and their extreme working conditions. Physical experiments under various conditions are also expensive. Hence it's critical to utilize models to facilitate the design and analysis process of LPRE.

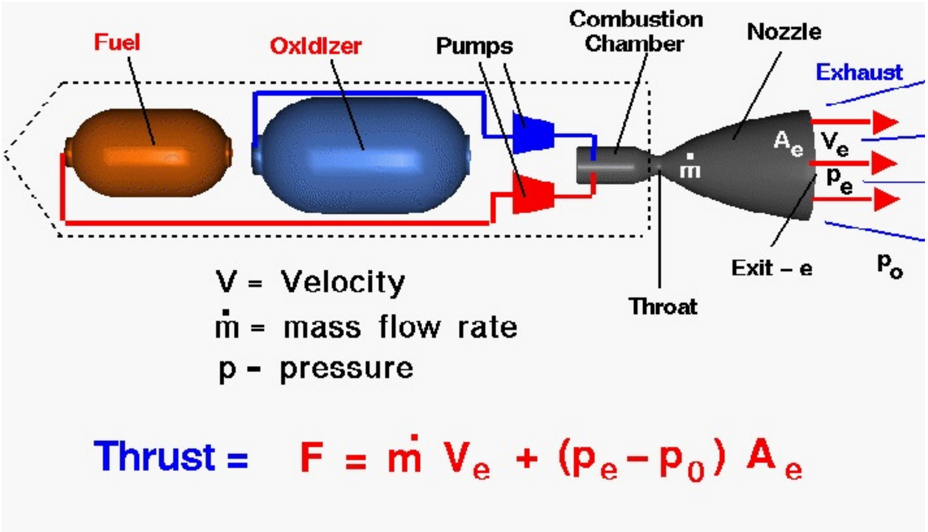

FIGURE 3.1: Liquid Rocket Engine

3.2 Classification

There are two categories, namely **Booster propulsion and Auxiliary propulsion** Booster propulsion are used for boosting a payload and imparting a significant velocity increase to a payload. It is used in Booster stage and upper stages of launch vehicles, large missiles. The total impulse of Booster propulsion is very high. The thrust level is High, ranging from 4500 N up to 7,900,000 N. Time elapsed to reach full thrust is Up to several seconds. The propellants used are Storable liquids and cryogenic liquids. Auxiliary propulsion is used for Attitude control, minor space maneuvers, trajectory corrections, orbit maintenance. Its applications include Spacecraft, satellites, top stage of anti-ballistic missile, space rendezvous. The total impulse of Auxiliary propulsion is low. Number of thrust chambers per engine are between 4 and 24. The thrust level is small usually from 0.001 up to 4500 N,

Liquid propellant rocket engine systems can be classified also on the basis of propellants used.

A **Bipropellant** rocket unit has two separate liquid propellants, an oxidizer and a fuel. They are stored separately and are not mixed outside the combustion chamber. The majority of liquid propellant rockets have been manufactured for bipropellant applications.

A **Monopropellant** contains an oxidizing agent and combustible matter in a single substance. It may be a mixture of several compounds or it may be a homogeneous material, such as hydrogen peroxide or hydrazine. Hydrazine is being used extensively as a monopropellant in small attitude and trajectory control rockets for the control of satellites and other spacecraft and also as a hot gas generator.

Monopropellants are stable at ordinary atmospheric conditions but decompose and yield hot com-

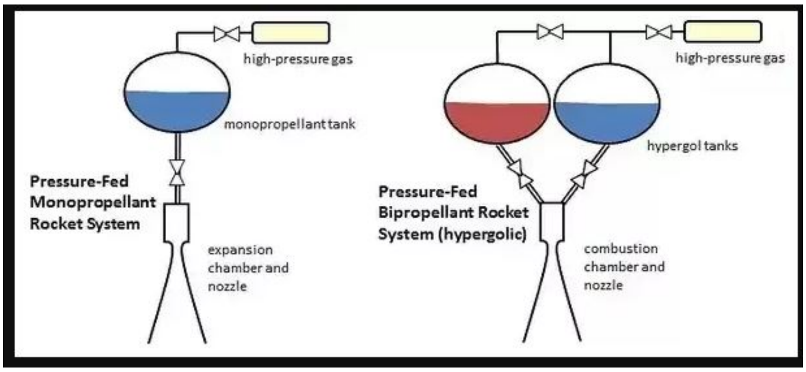

FIGURE 3.2: Monopropellant Rocket Engine

bustion gases when heated or catalyzed.

A **Cold gas propellant** (e.g., nitrogen) is stored at very high pressure, gives a low performance, allows a simple system and is usually very reliable. It has been used for roll control and attitude control.

A **Cryogenic propellant** is liquified gas at low temperature, such as liquid oxygen (- 1 83°C) or liquid hydrogen (-253°C). Provisions for venting the storage tank and minimizing vaporization losses are necessary with this type. Methane (CH_4) is a cryogenic hydrocarbon fuel. It is denser than liquid hydrogen and relatively low in cost.

Storable propellants (e.g., nitric acid or gasoline) are liquid at ambient temperature and can be stored for long periods in sealed tanks. Space storable propellants are liquid in the environment of space; this storability depends on the specific tank design, thermal conditions, and tank pressure. An example is ammonia. Nitrogen tetroxide is a storable propellant oxidizer and is used in the Titan missile together with a fuel mixture consisting of hydrazine and unsymmetrical dimethyl-hydrazine. It is also used with monomethylhydrazine fuel in the Space Shuttle orbital maneuver system and reaction control system and in many spacecraft propulsion systems.

A **Gelled propellant** is a thixotropic liquid with a gelling additive. It behaves like a jelly or thick paint. It will not spill or leak readily, can flow under pressure, will burn, and is safer in some respects.

Liquid propellant rocket engine systems can be classified in several other ways. They can be reusable (like the Space Shuttle main engine or a booster rocket engine for quick ascent or maneuvers of fighter aircraft) or suitable for a single flight only (as the engines in the Atlas or Titan launch vehicles) and they can be restartable, like a reaction control engine, or single firing, as in a space launch vehicle. They can also be categorized by their propellants, application, or stage, such as an upper stage or booster stage, their thrust level, and by the feed system type (pressurized or turbo pump).

3.2.1 Advantages of Liquid Propellant Engines over Solid Propelleant Engines

Liquid propellant engines have a number of advantages over solid propellant engines.

1. A wider array of propellant combinations are available for different applications. Some of these require an ignition system and others simply ignite on contact. Monomylmethylhydrozene (fuel) and nitrogen tetroxide (oxidizer) ignite spontaneously. These are called hypergolic propellants. With hypergolic propellants, a rocket engine does not need an ignition system. Hypergolic propellants are great for attitude control rockets like those that will be arrayed around the Orion service module.
2. Another advantage of liquid propellants is that they can be controlled. Adjusting their flow into the Liquid propellant rocket combustion chamber adjusts the amount of thrust produced.
3. Furthermore, liquid engines can be stopped and restarted later. It is very difficult to stop a solid propellant rocket once it is started, and thrust control is limited.

3.3 Design Considerations of Bi-propellant systems

1. Plumbing in the pressurization system has to: Provide isolation of the high pressure pressurant tank(s) from the relatively low pressure propellant tanks. Prevent migration and mixing of propellant vapors, if a pressurization system common to both propellants is used.
2. Prevent mixing of propellants (except in thrusters, of course).
3. Maintain control of the flow of both propellants such that the thruster inlet conditions stay within acceptable limits.

4. Control pressure drops in the system such that effects associated with pressurant coming out of solution are within acceptable limits

5. Bi-propellant systems often provide for the isolation of the pressurization system during long periods of system inactivity, or after enough of the propellant has been expelled that blow-down operations are possible.

6. Dual mode systems often provide for the isolation of the oxidizer system after the bi-propellant main engine has been used, either leaving the hydrazine tank in blow-down, or keeping the hydrazine tank at regulated pressure.

3.4 Engine Cycles

There are four methods which are used for transporting liquid propellants from their storage tanks to the rocket motor. They are:

Cycle	Application	Specifics	
Pressure Drive	Upper Stage, Space Propulsion	Low Pressure Low Thrust	< 2 MPa < 40 kN
Gas Generator	Booster, Core, Upper Stage	Medium Pressure Large Thrust Range	< 15 MPa 60 kN - 7 MN
Expander	Upper Stage	Low-Medium Pressure Small Thrust Range	3 - 7 MPa 80 - 200 kN
Staged Combustion	Booster, Core, Upper Stage	High Pressure Large Thrust Range	13 - 26 MPa 80 kN - 8 MN
Full Flow Staged Combustion	Booster, Core (nothing's flying yet)	High Pressure Large Thrust Range	> 30 MPa up to 10 MN

FIGURE 3.3: Applications of Engine Cycles

1. Gas pressurization system
2. Pump pressurization system.
3. Gas generator System.
4. Expander System.

3.4.1 Pressure fed system

Gas pressurization system is simplest cycle for rocket propulsion and it relies on a pressurant to force propellant from the tanks to the combustor. Thrust is limited due to the size of the pressurant tank. In this system, an inert gas (nitrogen) is used. This gas is stored at high pressure and is supplied through pressure regulator valves to force the liquid propellants through the lines, control valves, injector plate and into the combustion chamber.

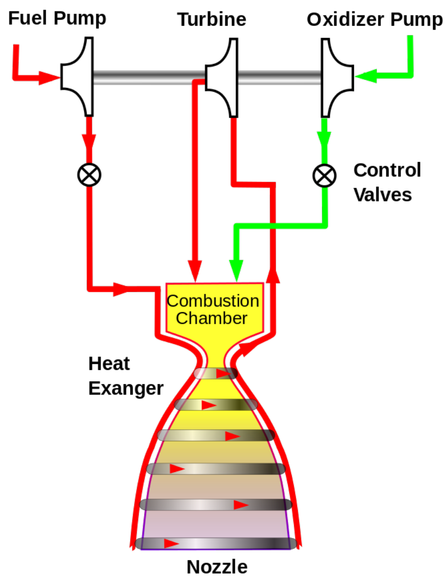

FIGURE 3.4: Pressure fed Rocket Engine

3.4.2 Turbo pump fed system

In turbopump system liquid oxidizer and fuel are stored at low pressure so that tanks are light in weight and forced into rocket motor at high pressure by fuel and oxidizer pumps. The power driving the pumps is supplied by gas turbine which is supplied with steam and oxygen obtained by decomposing hydrogen peroxide by a catalyst.

Because of the use of third liquid (H_2O_2), gas turbine, pumps and additional lines that are necessary, pump pressurization system is more complex than gas pressurization system. The design of pumps is the biggest problem because any leakage of liquids will lead to explosion. Liquid oxidizers are generally acids, liquid oxygen, concentrated hydrogen peroxide (H_2O_2), nitrogen tetra-oxide (N_2O_4) etc. and hence special pump materials are required for the oxidizer pump and impeller.

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

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

3.4.3 Expander fed system

Next let us look at the another cycle, which is called expander cycle as the name suggest there is some kind of expansion taking place in expander cycle. So, here what happens, there is a schematic fuel comes through a valve, and then it is channelized all around the combustion chamber, and the nozzle. So, the fuel flows through this channels and absorb some of the waste heat, because of this absorption of this heat - the fuel gets expanded, and gets converted into gas. And there after that this gas is supplied across a turbine, and because of that the turbine starts to rotate, this turbine is connected by a shaft to this fuel pump. So, the pump starts to volt and therefore fresh fuel is sucked in. Now, the gas after it crosses the turbine is fed back into the combustion chamber. So, therefore, there is no wastage of fuel and everything is essentially burnt.

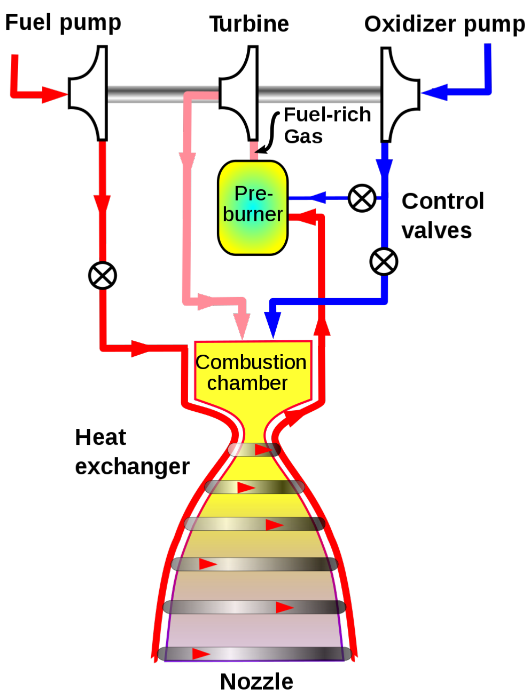

FIGURE 3.5: Expander combustion cycle

3.4.4 Stage Combustion cycle

The forth type of cycle is the stage combustion cycle. In stage combustion cycle the liquid fuel comes in goes around the combustion chamber gets heated up, but its fed into the pre burner, oxidizer directly comes in here part of the oxidizer is taken from the supply and put into the pre burner. Pre burner burns this fuel and complete all the fuel earlier in the gas generator it was all the fuel which was going to the pre burner. Here all the fuel goes to the pre burner. So, as you see you can see there is a fairly fuel reach combustion going on, and then it goes to the turbine, and then the entire product is fed into the combustor again. So, nothing is lost and it is already a vapor phase now. So, your fuel is now coming as a vapor of product and fuel, because some of the product is created here. So, this is the stage combustion cycle, some of the propellant is burn in the pre burner and resulted hot gases used to power the engines turbine, and the pumps. The exhausted gas is then injected into the main combustor along with the rest of the propellant and combustion is completed. So, all the cycle gases are going through the combustion chamber, therefore overall efficiency does not suffer any loss. So, we have fairly high combustion efficiency. This combustion

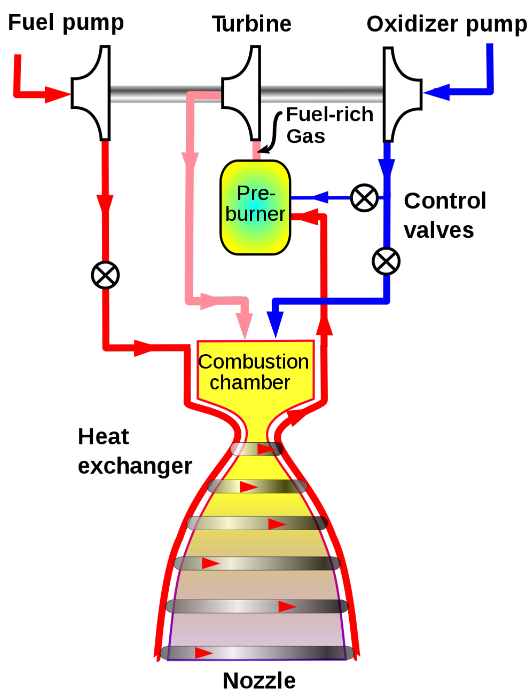

FIGURE 3.6: Staged combustion cycle

cycle is often called close cycle, because everything is going round. And the cycle is closed as propellant produce product goes to the chamber as oppose to the open cycle, where some of these were discarded right in the open circle some of gas generator cycle, some of it were discarded. Stage combustion gives an abundance of power, which permits very high chamber pressure, we can go to very high chamber pressure by this schematic, very high chamber pressure on the other hand means high expansion ratio nozzles are possible. and because of this high expansion ratio nozzle, we can get fairly good pressure at take of as well even at ambient pressure, we can get fairly high pressure in the combustion chamber to allow even take off, because it is possible to operate at very high pressures.

3.5 Valve

The propellant valves in high thrust units handle relatively large flows at high service pressures. Therefore, the forces necessary to actuate the valves are large. Hydraulic or pneumatic pressure, controlled by pilot valves, operates the larger valves; these pilot valves are in turn actuated by a solenoid or a mechanical linkage. Essentially this is a means of power boost.

3.5.0.1 Classification of Valves Used in Liquid Propellant Rocket Engines

Two valves commonly used in pressurized feed systems are isolation valves and latch valves. They require power for brief periods during movements, such as to open or shut, but need no power when latched or fastened into position. A very simple and very light valve is a burst diaphragm. It is essentially a circular disk of material which blocks a pipeline and is designed so that it will fail and burst at a predetermined pressure differential. Burst diaphragms are positive seals and prevent leakage, but they can be used only once. Pressure regulators are special valves which are used frequently to regulate gas pressures. Usually the discharge pressure is regulated to a predetermined standard pressure value by continuously throttling the flow, using a piston, flexible diaphragm, or electromagnet as the actuating mechanism.

3.6 Combustion Instabilities

Development of resonant fluctuations in chamber pressure can produce vibration, and in extreme cases, destruction of the engine. Even if the damages does not ensue, the fluctuating pressure, and the associated fluctuations in combustion temperature reduce the efficiency with which propellant energy is converted into thrust. Thus, particular effort has to be made in the design to remove or limit the magnitude of these fluctuations in the chamber pressure.

3.6.1 Pogo Oscillation

Pogo oscillation is a self-excited vibration in liquid-propellant rocket engines caused by combustion instability.[1] The unstable combustion results in variations of engine thrust, causing variations of acceleration on the vehicle's flexible structure, which in turn cause variations in propellant pressure and flow rate, closing the self-excitation cycle. The name is a metaphor comparing the longitudinal vibration to the bouncing of a pogo stick. Pogo oscillation places stress on the frame of the vehicle, which in severe cases can be dangerous.

3.6.2 Sloshing

Sloshing means any motion of free liquid surface in its container. It is caused by any distrubance to partially filled containers. Depending on the type of distrubance, shape of container, the free liquid surface can experience different types of motion including simple planar, rotational, irregular beating, symmetric asymmetric, quasi-periodic and chaotic. Important examples include propellant slosh in spacecraft tanks and rockets (especially upper stages), and the free surface effect (cargo slosh) in ships and trucks transporting liquids (for example oil and gasoline). During a rocket's ascent phase, sloshing of propellants in its fuel tanks can lead to serious trajectory control problems because the sloshing natural frequencies are close to the controller frequencies and this likely instability is compounded by the large propellant masses involved. Shortly after it reached orbit in August 1969, NASA's spin-stabilized Applications Technology Satellite 5 (ATS5) began to wobble, sending the spacecraft into an unplanned flat spin and crippling the mission. It was later found that this event was caused by excessive fuel slosh, creating a long-standing concern about this phenomenon.

3.6.3 Low Frquency: Chugging

This is a low frequency oscillation at a few Hertz in chamber pressure usually caused by pressure variations in feed lines due to variations in acceleration of the vehicle. Characteristic frequency range is between 10 and 200 Hz. Chugging, the first type of combustion instability stems mostly from the elastic nature of the feed systems and structures of vehicles or the imposition of propulsion forces upon the vehicle. Chugging of an engine or thrust chamber assembly can occur in a test facility, especially with low chamber pressure engines (100 to 500 psia), because of propellant pump cavitation, gas entrapment in propellant flow, tank pressurization control fluctuations, and vibration of engine supports and propellant lines. It can be caused by resonances in the engine feed system (such as an oscillating bellows inducing a periodic flow fluctuation) or a coupling of structural and feed system frequencies.

3.6.4 Buzzing

Characteristic frequency range is between 20 and 1000 Hz. Instability is a result from either flow instabilities or resonance with chamber structure. This can be caused due to insufficient pressure drop across the injectors. It generally is mostly annoying, rather than being damaging. However, in extreme cases combustion can end up being forced backwards through the injectors – this can cause explosions with mono propellants.[citation needed]

3.6.5 High frequency: Screaming mode

Characteristic frequency range is above 1000 Hz. Instability occurs due to interaction of combustion process with chamber acoustics. It is due to acoustics within the combustion chamber that often couples to the chemical combustion processes that are the primary drivers of the energy release, and can lead to unstable resonant ”screeching” that commonly leads to catastrophic failure due to thinning of the insulating thermal boundary layer. Acoustic oscillations can be excited by thermal processes, such as the flow of hot air through a pipe or combustion in a chamber. Screeching is often dealt with by detailed changes to injectors, or changes in the propellant chemistry, or vaporising the propellant before injection, or use of Helmholtz dampers within the combustion chambers to change the resonant modes of the chamber.

3.7 Thrust vector control

In a rocket, the rocket engine or motor not only provides the propulsive force but also the means of controlling its flight path by redirecting the thrust vector to provide directional control for the vehicle’s flight path. This is known as thrust vector control (TVC).

3.7.1 Various methods of vector control of Liquid Rockets

Gimbaled Engines – Some liquid propellant rockets use an engine swivel or gimbal arrangement to point the entire engine assembly. This arrangement requires flexible propellant lines, but produces negligible thrust losses for small deflection angles. This method is relatively common.

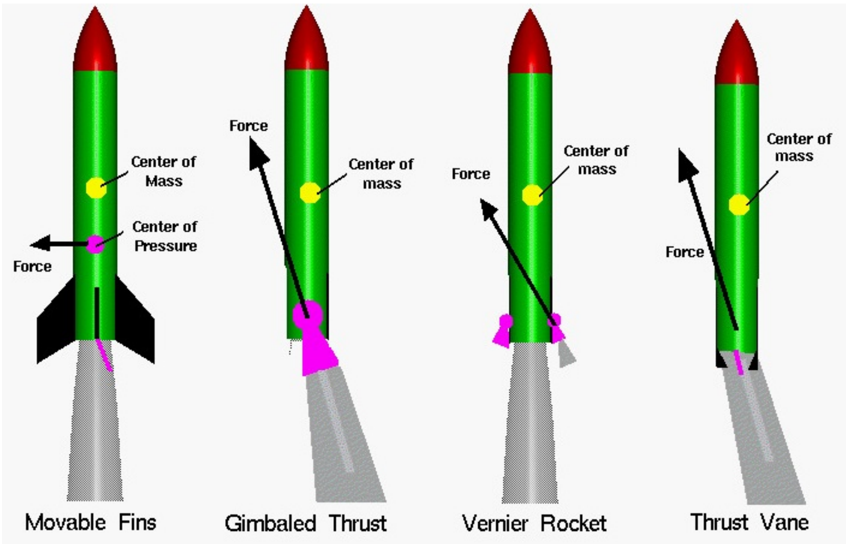

FIGURE 3.7: Thrust vector control mechanisms

Vernier Rockets: Vernier rockets are small auxiliary rocket engines. These engines can provide all attitude control, or just roll control for single engine stages during the main engine burn, and a means of controlling the rocket after the main engine has shut off.

Jet Vanes: Jet vanes are small airfoils located in the exhaust flow behind the nozzle exit plane. They act like ailerons or elevators on an aircraft and cause the vehicle to change direction by redirecting the rocket. Jet vanes are made of heat-resistant materials like carbon-carbon and other refractory substances. Unfortunately, this control system causes a two to three percent loss of thrust, and erosion of the vanes is also a major problem.

3.8 Characteristics of Bi-propellant systems

1. Small negative or preferably positive standard heats of formation of the reactant .
2. The reaction products should have low molecular weights and large negative heats of formation.
3. The propellants should have large densities in order to minimize the dead weight of storage tanks.
4. The oxidizers and reducing agents are best handled as liquids. For substances such as liquid oxygen and hydrogen special cooling units must be provided.

5. It is necessary to provide special cooling equipment for the chamber walls. At least one of the components of a bi-propellant system with a high specific heat and/or large heat of vaporization.

6. Since it may be necessary to store the propellants for long periods of time before use, good propellants should have high storage stability, i.e., they must not decompose or change chemically in any way during storage so that their use as a propellant is impaired.

7. Since propellants are chemicals which have to be handled by service personnel it may be desirable for some applications to use propellants of relatively low toxicity.

8. Propellants which are readily available and preferably also of low cost are employed.

9. The bi-propellant mixture in a liquid-fuel rocket should be spontaneously combustible with minimum time lag. Spontaneously combustible propellants are said to be hypergolic whereas non spontaneous6 propellants are said to be non hypergolic. The time lag or ignition delay is the period of time preceding steady-state combustion.

10. The reaction products should not be excessively corrosive or form solid deposits thereby leading either to increased or decreased nozzle throat diameters.

11. For application to guided missiles the exhaust gases should not interfere with the guidance method which is being used.

3.9 Advantages

1. Provides higher impulse for given propellant density; increases attainable vehicle velocity increment and mission velocity.

2. Can be randomly throttles and stopped and restarted.

3. Provides for pulsed (repetitive) operation. Some small thrust rockets allow over 250,000 times usage.

4. Better control over mission terminal velocity, with precise thrust termination devices.

5. Can be largely checked prior to operationie can be tested for full thrust operation on ground.

6. Thrust chamber smaller, can be cooled.
7. Thrust chamber can be designed for re-use after check ups.
8. Thrust chamber has thinner walls and light weight.
9. With pumped propellant feed system, inert system weight (including tanks) is lower allowing high propellant mass fraction.
10. Liquid propellants are storable in the vehicle for more than 20 years and engine can be ready for use quickly.
11. Propellant feed system can be designed to feed multiple thrust chambers.
12. Plume radiation and smoke are usually low.
13. Propellant tanks can be located such that vehicle stability is high.

3.10 Disadvantages

1. Relatively complex design with more components. Probability of failure more.
2. Spills or leaks can be hazardous, corrosive, toxic and can cause fires.
3. Fuel and oxidizer tanks need to be pressurized.
4. Needs separate feed system.
5. Cryogenic propellants cannot be stored for long periods. Storage tanks need special insulation.
6. Need separate ignition system (except for hypergolic propellants).
7. More overall weight for short duration, low total impulse application.
8. More difficult to control combustion instability.
9. A few propellants like RFNA (red fuming nitric acid) give toxic vapors and fumes
10. Need more volume due to low average density of propellant.

11. Sloshing of liquid in tanks can cause stability problem in flight.

12. Needs special design provisions for start at zero gravity.

3.11 Cooling of Combustion Chamber and Nozzles

3.11.1 Radiation cooling

This is probably the simplest method of cooling a rocket engine or motor. The method is usually used for monopropellant thrusters, gas generators, and lower nozzle sections. The interior of the combustion chamber is covered with a refractory material (graphite, pyrographite, tungsten, tantalum or molybdenum) or is simply made thick enough to absorb a lot of heat. Cooling occurs by heat loss through radiation into the exhaust plume. Radiation cooling can set an upper limit on the temperature attained by the walls of the thrust chamber. The rate of heat loss varies with the fourth power of the absolute temperature and becomes more significant as the temperature rises.

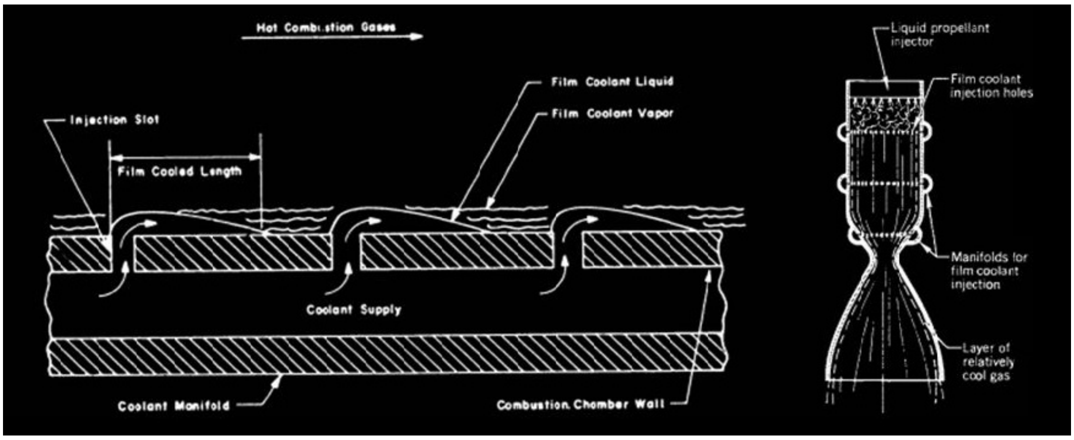

FIGURE 3.8: Film cooling mechanisms

3.11.2 Ablation cooling

In the ablation cooling method, the interior of the thrust chamber is lined with an ablative material, usually some form of fabric reinforced plastic. This material chars, melts and vaporizes in the intense heat of the nozzle. In this type of "heat sink cooling," the heat absorbed in the melting and burning (the energy alters the chemical form instead of raising its temperature) of the ablative material prevents the temperature from becoming excessively high. The charred material also serves as an insulator and protects the rocket case from overheating. The gas produced by burning the ablative material provides an area of “cooler” gas next to the nozzle walls. The synthetic organic plastic binder material is reinforced with glass fiber or a synthetic substance. Solid rocket motors use ablative cooling almost exclusively, as there are no other fluids to use to cool the nozzle throat.

3.11.3 Film cooling

With this method of cooling, liquid propellant is forced through small holes at the periphery of the injector forming a film of liquid on the interior surface of the combustion chamber. The film has a low thermal (or heat) conductivity since it readily vaporizes and protects the wall material from the hot combustion gases. Cooling results from the vaporization of the liquid which absorbs considerable heat. Film cooling is especially useful in regions where the walls become exceptionally hot, e.g., the nozzle throat area.

3.11.4 Regenerative cooling

Ans. This is the most common method of cooling for cryogenic propellant rockets. It involves circulating one of the super cooled propellants through a cooling jacket around the combustion chamber and nozzle before it enters the injector. The propellant removes heat from the walls, keeping temperatures at acceptable levels. At the same time, the temperature of the propellant rises, causing it to vaporize faster upon injection. This cooling method is often used with gas generator systems as a way to drive turbo pumps.

Chapter 4

Multistage of Rockets and Stage Separation Dynamics

The guidance system of a rocket includes very sophisticated sensors, on-board computers, radars, and communication equipment. The guidance system has two main roles during the launch of a rocket; to provide stability for the rocket, and to control the rocket during maneuvers.

Missile guidance concerns the method by which the missile receives its commands to move along a certain path to reach a target. On some missiles, these commands are generated internally by the missile computer autopilot. On others, the commands are transmitted to the missile by some external source. The missile sensor or seeker, on the other hand, is a component within a missile that generates data fed into the missile computer. This data is processed by the computer and used to generate guidance commands. Sensor types commonly used today include infrared, radar, and the global positioning system. Based on the relative position between the missile and the target at any given point in flight, the computer autopilot sends commands to the control surfaces to adjust the missile's course. The most common types of seekers or sensors used today include infrared, radar, lasers, inertial, and GPS. It is the use of these guidance systems that turns a ”dumb” weapon into a ”smart” weapon.

The guidance process

The guidance process consists of measurement of vehicle position and velocity, computation of control actions necessary to properly adjust position and velocity, and delivery of suitable adjustment commands to the vehicle's control system.

4.1 Guidance phases of flight

Guidance operations may occur in the initial, midcourse, or terminal phases of flight.

Ballistic missiles are commonly guided only during the initial flight phase, while rocket engines are burning.

A cruise type of missile, such as the Snark or Matador, uses midcourse guidance, operating continuously during cruising flight. Air-to-air missiles such as Sidewinder employ terminal guidance systems that lead the missile directly to the target on the basis of measurements on the target itself.

Any or all of these three kinds of guidance will be necessary for space flight, depending on the type of vehicle and mission involved. Generally, missile in-flight guidance is divided into three phases-boost, midcourse, and terminal. These names refer to the different parts or time periods of a trajectory.

4.1.1 Boost Phase

The boost phase of missile flight is also known as the launching phase or initial phase. It is during this period that the missile is boosted to flight speed. It lasts until the fuel supply of the booster bums up. For the medium-range (MR) missiles that use a dual-thrust rocket motor (DTRM), the booster propellant grain is consumed and bums out. For extended range (ER) missiles, the separate booster drops off at burnout.

The boost phase is very important to the flight path of the missile. The launcher and missile are aimed in a specific direction by orders from the FCS computer. This aiming establishes the line of sight (trajectory or flight path) the missile must fly along during the initial phase. At the end of

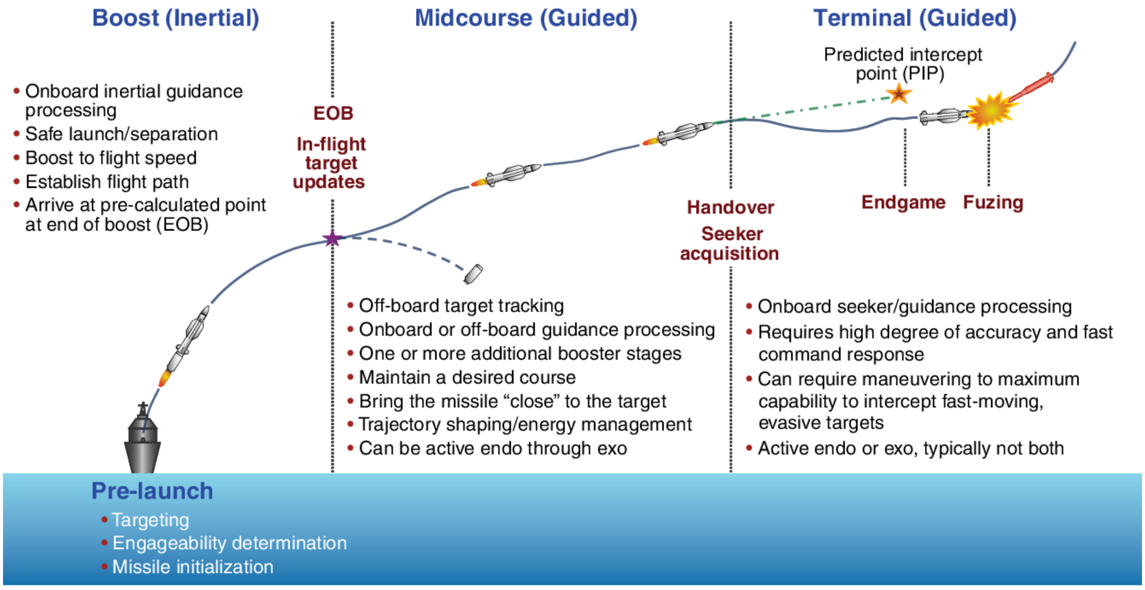

FIGURE 4.1: Guidance phases of flight

boost, the missile must be at a calculated point. Some missiles are guided during boost; others are not.

4.1.2 Midcourse Phase

The second or midcourse phase of guidance is often the longest in both distance and time. During midcourse (or cruise) guidance, the missile makes any corrections necessary to stay on the desired course. Guidance information can be supplied to the missile by various means. The object of midcourse guidance is to place the missile near the target.

4.1.3 Terminal Phase

The terminal phase of guidance brings the missile into contact or close proximity with the target. The last phase of guidance must have quick response to ensure a high degree of accuracy. Quite often the guidance system causes the missile to perform what is best described as an ”up-and-over” maneuver during the terminal phase. Essentially, the missile flies higher than the target and descends on it at intercept.

4.2 Classification of Guidance Systems

Guidance systems are divided into different categories according to whether they are designed to attack fixed or moving targets. The weapons can be divided into two broad categories: Go-Onto-Target (GOT) and Go-Onto-Location-in-Space (GOLIS) guidance systems.[4] A GOT missile can target either a moving or fixed target, whereas a GOLIS weapon is limited to a stationary or near-stationary target. The trajectory that a missile takes while attacking a moving target is dependent upon the movement of the target. Also, a moving target can be an immediate threat to the sender of the missile. The target needs to be eliminated in a timely fashion in order to preserve the integrity of the sender. In GOLIS systems, the problem is simpler because the target is not moving.
In every Go-Onto-Target system there are three subsystems: Target tracker, Missile tracker and guidance computer. The way these three subsystems are distributed between the missile and the launcher result in two different categories.

4.2.1 Beam Rider Guidance

The beam rider concept relies on an external ground- or ship-based radar station that transmits a beam of radar energy towards the target. The surface radar tracks the target and also transmits a guidance beam that adjusts its angle as the target moves across the sky. The missile is launched into this guidance beam and uses it for direction. Scanning systems onboard the missile detect the presence of the beam and determine how close the missile is to the edges of it. This information is used to send command signals to control surfaces to keep the missile within the beam. In this way, the missile ”rides” the external radar beam to the target.

Beam riding was often used on early surface-to-air missiles but was found to become inaccurate at long ranges. Limited improvement was possible using two different surface-based radar beams, but the beam rider method has been largely abandoned. The technique was used on the US Navy's Terrier ship-launched surface-to-air missile of the 1950s.

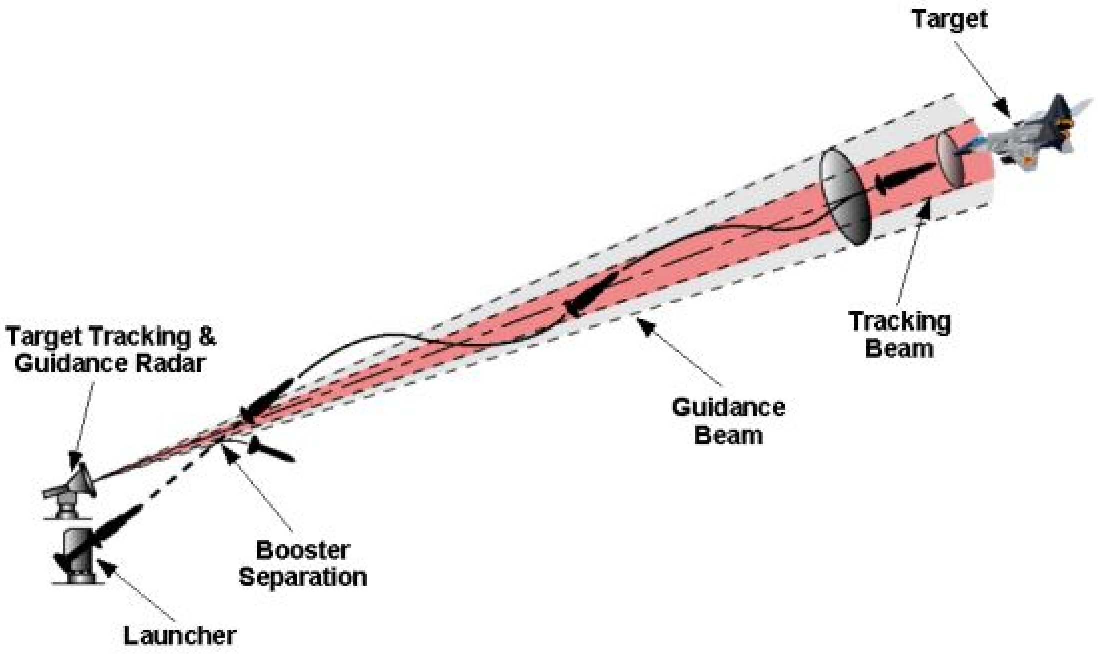

FIGURE 4.2: Beam rider guidance

4.2.2 Command Guidance

Command guidance is similar to beam riding in that the target is tracked by an external radar. However, a second radar also tracks the missile itself. The tracking data from both radars are fed into a ground based computer that calculates the paths of the two vehicles.

This computer also determines what commands need to be sent to the missile control surfaces to steer the missile on an intercept course with the target. These commands are transmitted to a receiver on the missile allowing the missile to adjust its course. An example of command guidance is the Russian SA-2 surface-to-air missile used against US aircraft in North Vietnam.

Command guidance is not limited just to radar. Another method that falls under command guidance is the use of wire guided systems. In this technique, commands are sent to the missile through a conventional wire or fiber optic cable that reels out from the missile back to its launcher. Wire guidance is often used on anti-tank missiles like TOW, which can be launched from both ground vehicles and helicopters. Many naval torpedoes fired from submarines also use wire guidance.

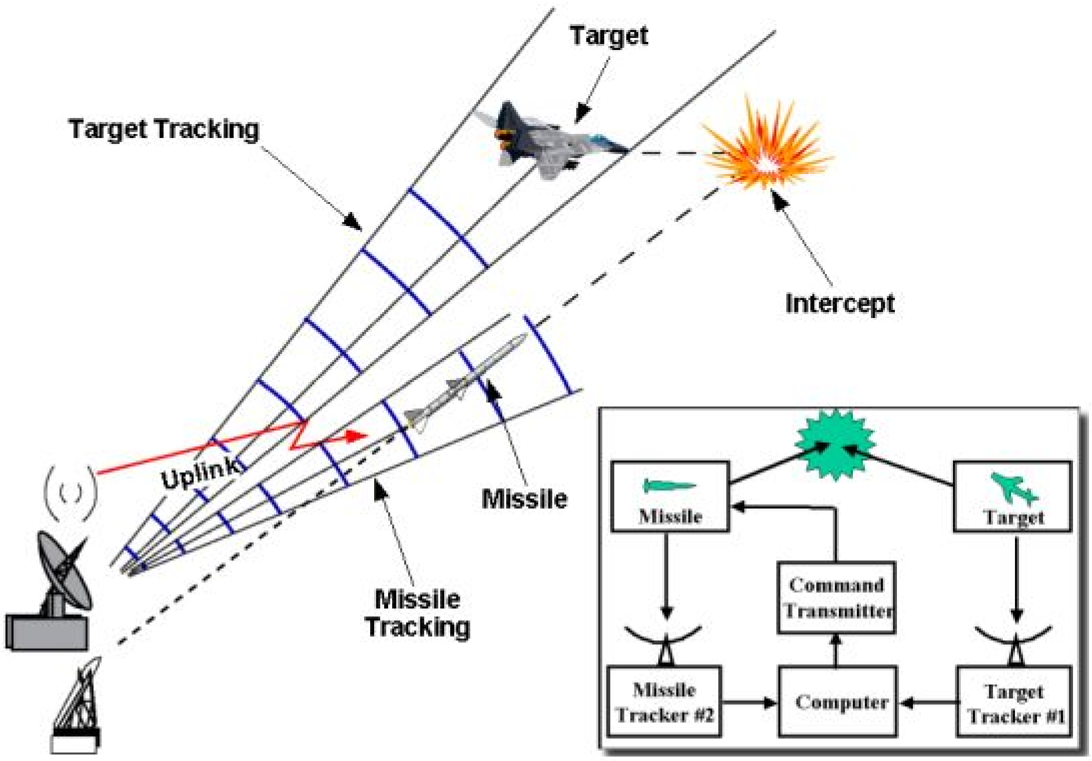

FIGURE 4.3: Command guidance

4.2.3 Homing Guidance

Homing guidance is the most common form of guidance used in anti-air missiles today. Three primary forms of guidance fall under the homing guidance umbrella–semi active, active, and passive. We will discuss each of these in turn, as well as a more unusual form called retransmission or track-via-missile homing. Semi-Active Homing Guidance

A semi-active system is similar to command guidance since the missile relies on an external source to illuminate the target. The energy reflected by this target is intercepted by a receiver on the missile. The difference between command guidance and semi-active homing is that the missile has an onboard computer in this case. The computer uses the energy collected by its radar receiver to determine the target's relative trajectory and send correcting commands to control surfaces so that the missile will intercept the target. he example shown above illustrates the guidance method used on an air-to-air missile like Sparrow. This missile relies on radar energy transmitted by the launch aircraft to track and home in on the target. This system is also sometimes referred to as bistatic meaning that the radar waves that intercept the target and those reflected back to the missile are at different angles to one another.

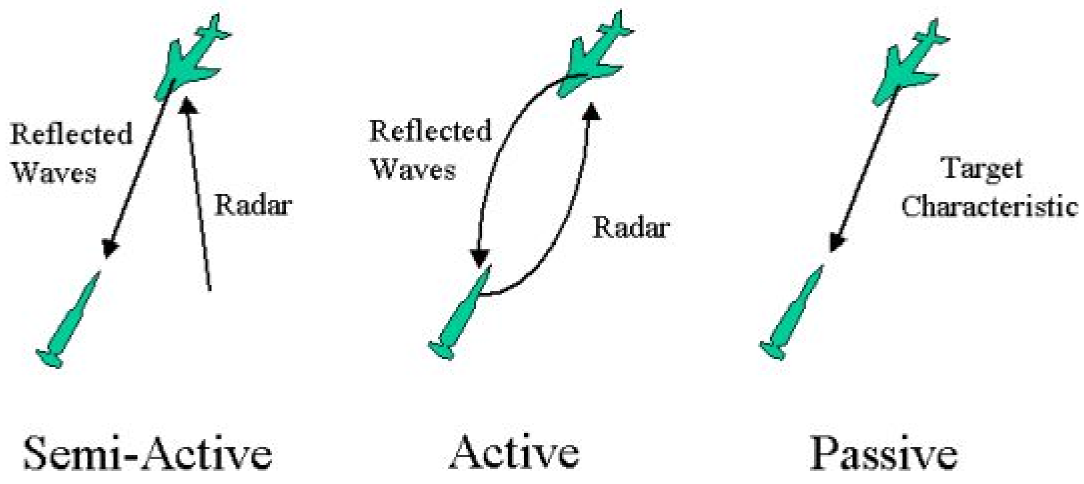

FIGURE 4.4: Homing guidance

4.2.3.1 Active Homing Guidance

Active homing works just like semi-active except that the tracking energy is now both transmitted by and received by the missile itself. No external source is needed. It is for this reason that active homing missiles are often called ”fire-and-forget” because the launch aircraft does not need to continue illuminating the target after the missile is launched. Active homing missiles typically use radar seekers to track their target. These seekers are also sometimes called monostatic because, unlike semi-active guidance, the transmitted and reflected waves are at the same angle with respect to the line of sight between the missile and target. Examples of active homing missiles include the AMRAAM air-to-air and Exocet anti-ship missiles.

4.2.3.2 Passive Homing Guidance

A passive homing system is like active in that the missile is independent of any external guidance system and like semi-active in that it only receives signals and cannot transmit. Passive missiles instead rely on some form of energy that is transmitted by the target and can be tracked by the missile seeker.

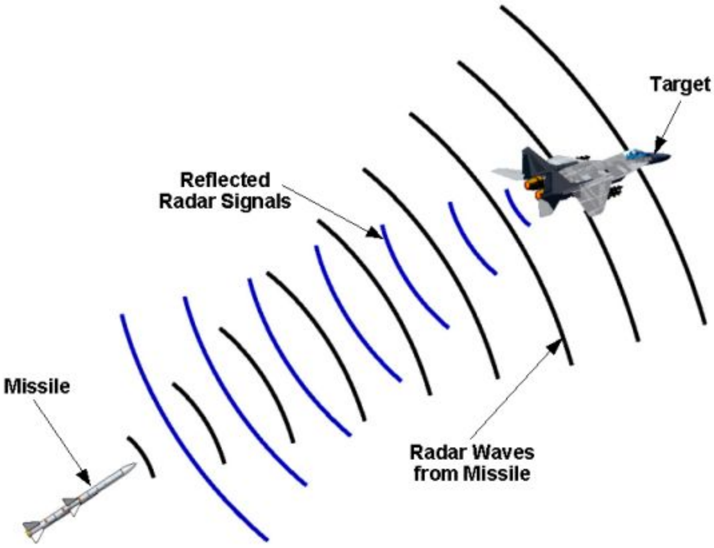

FIGURE 4.5: Active Homing Guidance

Like homing guidance, navigation guidance includes several subcategories. In this section, we will describe inertial, ranging, celestial, and geophysical navigation techniques.

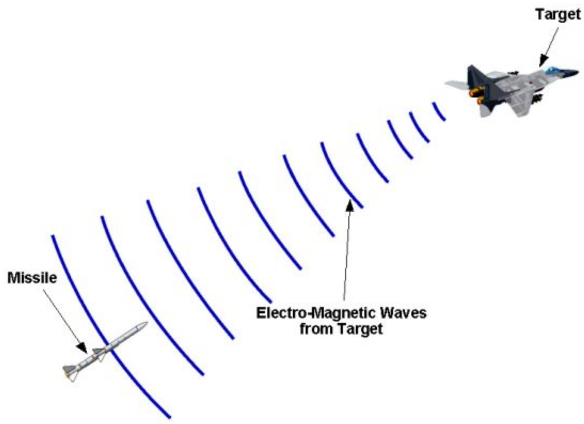

FIGURE 4.6: Passive Homing Guidance

4.2.4 Inertial Navigation Guidance

Inertial navigation relies on devices onboard the missile that sense its motion and acceleration in different directions. These devices are called gyroscopes and accelerometers. The purpose of a gyroscope is to measure angular rotation, and a number of different methods to do so have been devised. A classic mechanical gyroscope senses the stability of a mass rotating on gimbals. More recent ring laser gyros and fiber optic gyros are based on the interference between laser beams. Current advances in Micro-Electro-Mechanical Systems (MEMS) offer the potential to develop gyroscopes that are very small and inexpensive.

While gyroscopes measure angular motion, accelerometers measure linear motion. The accelerations from these devices are translated into electrical signals for processing by the missile computer autopilot. When a gyroscope and an accelerometer are combined into a single device along with a control mechanism, it is called an inertial measurement unit (IMU) or inertial navigation system (INS).
The INS uses these two devices to sense motion relative to a point of origin. Inertial navigation works by telling the missile where it is at the time of launch and how it should move in terms of both distance and rotation over the course of its flight. The missile computer uses signals from the INS to measure these motions and ensure that the missile travels along its proper programmed path.
Ranging Navigation Guidance

Unlike inertial navigation, which is contained entirely onboard the vehicle, ranging navigation depends on external signals for guidance. The earliest form of such navigation was the use of radio beacons developed primarily for commercial air service. These beacons transmit radio signals received by an aircraft in flight. Based on the direction and strength of the signals, the plane can calculate its location relative to the beacons and navigate its way through the signals.
he advent of the global positions system (GPS) has largely replaced radio beacons in both military and civilian use. GPS consists of a constellation of 24 satellites in geosynchronous orbit around the Earth. If a GPS receiver on the surface of the Earth can receive signals from at least four of these satellites, it can calculate an exact three-dimensional position with great accuracy. Missiles like JSOW and the JDAM series of guided bombs make use of GPS signals to determine where they are with respect to the locations of their targets. Over the course of its flight, the weapon uses this information to send commands to control surfaces and adjust its trajectory.

4.2.5 Celestial Navigation Guidance

Celestial navigation is one of the earliest forms of navigation devised by humans. and it saw its greatest application in the voyages of the great maritime explorers like Christopher Columbus. Celestial navigation uses the positions of the stars to determine location, especially latitude, on the surface of the Earth. This form of navigation requires good visibility of the stars, so it is only useful at night or at very high altitude. As a result, celestial navigation is seldom applied to missiles, though it has been used on many ballistic missiles like Poseidon. The missile compares the positions of the stars to an image stored in memory to determine its flight path.

4.3 Aerodynamic control systems of missiles

The primary function of a control system during powered flight is to orient and stabilize the rocket vehicle. To orient the vehicle in some desired angular direction, it is necessary to develop torque to turn the body. Control ceases and the body is stabilized when sensing instruments, usually gyroscopic devices, indicate that the proper attitude has been achieved. During the propulsion phase, control torque is usually developed either by aerodynamic forces acting on control surfaces or by rocket forces. Techniques for aerodynamic control are essentially the same as those used in conventional aircraft. At very high altitudes, however, aerodynamic surfaces become ineffective because of low air density Therefore, other means of producing torques are required in the vacuum environment of upper altitudes and space.
Most missiles do not have conventional rudders, ailerons, or elevators like those used on typical airplanes. Nonetheless, missiles do employ similar aerodynamic control surfaces in order to maneuver the vehicle during flight.

The heart of a missile is the body, equivalent to the fuselage of an aircraft. The missile body contains the guidance and control system, warhead, and propulsion system. Some missiles may consist of only the body alone, but most have additional surfaces to generate lift and provide maneuverability. Depending on what source you look at, these surfaces can go by many names. In particular, many use the generic term ”fin” to refer to any aerodynamic surface on a missile. Missile designers, however, are more precise in their naming methodology and generally consider these surfaces to fall into three major categories: canards, wings, and tail fins.

The example shown above illustrates a generic missile configuration equipped with all three sur-

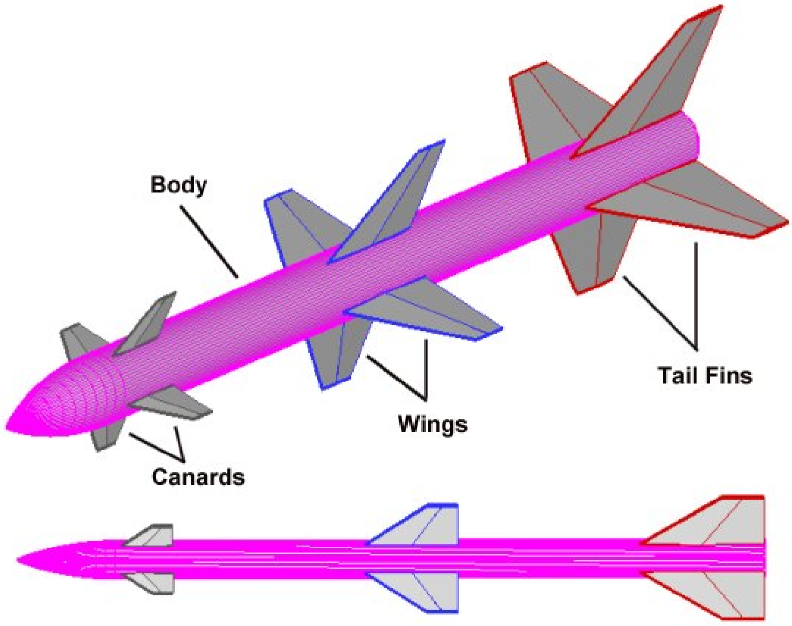

FIGURE 4.7: Generic missile configuration

faces. Often times, the terms canard, wing, and fin are used interchangeably.
Most missiles are equipped with at least one set of aerodynamic surfaces, especially tail fins since these surfaces provide stability in flight. The majority of missiles are also equipped with a second set of surfaces to provide additional lift or improved control. Very few designs are equipped with all three sets of surfaces. Whereas most aircraft have fixed horizontal and vertical tails with smaller movable rudder and elevator surfaces, missiles typically use all-moving surfaces, like those illustrated below, to accomplish the same purpose. In order to turn the missile during flight, at least one set of aerodynamic surfaces is designed to rotate about a center pivot point. In so doing, the angle of attack of the fin is changed so that the lift force acting on it changes. The changes in the direction and magnitude of the forces acting on the missile cause it to move in a different direction and allow the vehicle to maneuver along its path and guide itself towards its intended target.

Tail control is probably the most commonly used form of missile control, particularly for longer range air-to-air missiles like AMRAAM and surface-to-air missiles like Patriot and Roland. The primary reason for this application is because tail control provides excellent maneuverability at the high angles of attack often needed to intercept a highly maneuverable aircraft. Missiles using tail control are also often fitted with a non-movable wing to provide additional lift and improve range.

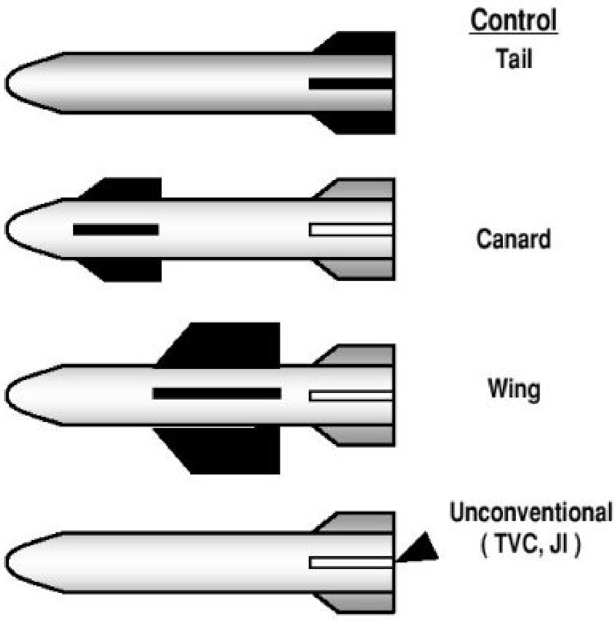

FIGURE 4.8: Canard tail control and wing configuration

Canard control is also quite commonly used, especially on short-range air-to-air missiles like AIM-9M Sidewinder. The primary advantage of canard control is better maneuverability at low angles of attack, but canards tend to become ineffective at high angles of attack because of flow separation that causes the surfaces to stall. Since canards are ahead of the center of gravity, they cause a destabilizing effect and require large fixed tails to keep the missile stable.

A further subset of canard control missiles is the split canard. Split canards are a relatively new development that has found application on the latest generation of short-range air-to-air missiles like Python 4 and the Russian AA-11. The term split canard refers to the fact that the missile has two sets of canards in close proximity, usually one immediately behind the other. The first canard is fixed while the second set is movable. The advantage of this arrangement is that the first set of canards generates strong, energetic vortices that increase the speed of the airflow over the second set of canards making them more effective. In addition, the vortices delay flow separation and allow the canards to reach higher angles of attack before stalling. This high angle of attack performance gives the missile much greater maneuverability compared to a missile with single canard control.

Unconventional Control:

Unconventional control systems is a broad category that includes a number of advanced technologies. Most techniques involve some kind of thrust vectoring. Thrust vectoring is defined as a method of deflecting the missile exhaust to generate a component of thrust in a vertical and/or horizontal direction. This additional force points the nose in a new direction causing the missile to turn. Another technique that is just starting to be introduced is called reaction jets. Reaction jets are usually small ports in the surface of a missile that create a jet exhaust perpendicular to the vehicle surface and produce an effect similar to thrust vectoring. These techniques are most often applied to high off-boresight air-to-air missiles like AIM-9X Sidewinder and IRIS-T to provide exceptional maneuverability. The greatest advantage of such controls is that they can function at very low speeds or in a vacuum where there is little or no airflow to act on conventional fins. The primary drawback, however, is that they will not function once the fuel supply is exhausted.

4.4 Multistage Rocket

Rockets use the thrust generated by a propulsion system to overcome the weight of the rocket. For full scale satellite launchers, the weight of the payload is only a small portion of the lift-off weight. Most of the weight of the rocket is the weight of the propellants. As the propellants are burned off during powered ascent, a larger proportion of the weight of the vehicle becomes the near-empty tankage and structure that was required when the vehicle was fully loaded. In order to lighten the weight of the vehicle to achieve orbital velocity, most launchers discard a portion of the vehicle in a process called staging. The idea behind staging is to improve performance by reducing the vehicle's mass on the way to orbit. Once the propellant of a stage is consumed, the empty stage which is no longer useful and only adds weight to the vehicle is discarded and the next stage is ignited. This stage then accelerates the rest of the vehicle much faster. As a result, less propellant is required to reach the desired orbit.

"A rocket having two or more engines, stacked one on top of another and firing in succession is called a multi-stage. Normally each unit, or stage, is jettisoned after completing its firing. The reason rocketeers stage models is to enable the uppermost stage to attain a very high altitude. This is accomplished by dropping mass throughout the burn so the top stage can be very light and coast a long way upward."

Advantage of multi staging is the possibility of thrust programming, as well as the possibility of adapting the engines of the subsequent stages to the altitude where they are fired, thus reducing losses due to non ideal expansion.

Another advantage of staging is that launch vehicle configuration can be optimized for the requirements of a particular mission by adjusting the amount of propellant and engine thrust, and using different types of engines, propellants and structural materials for various stages.

There are two types of rocket staging, serial and parallel. In serial staging, shown above, there is

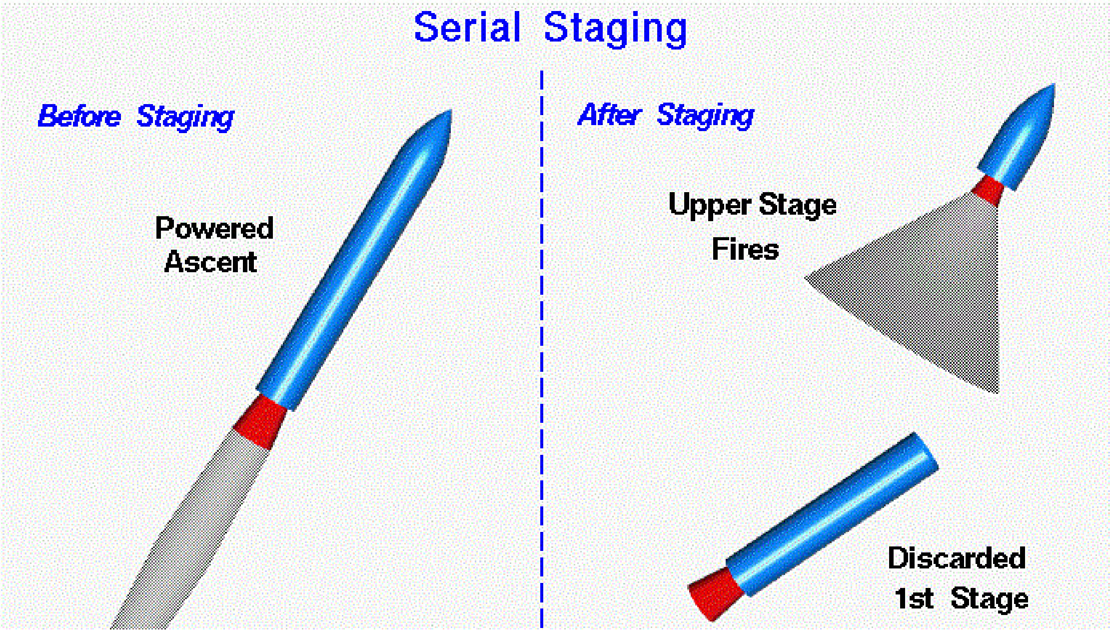

FIGURE 4.9: Series staging

a 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 atop the second stage into orbit. Serial staging was used on the Saturn V moon rockets. The Saturn V was a three stage rocket, which performed two staging maneuvers on its way to earth orbit. The discarded stages of the Saturn V were never retrieved.

In parallel staging, as shown in this figure, 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 burning and the payload is carried atop the sustainer rocket into orbit. Parallel staging is used on the Space

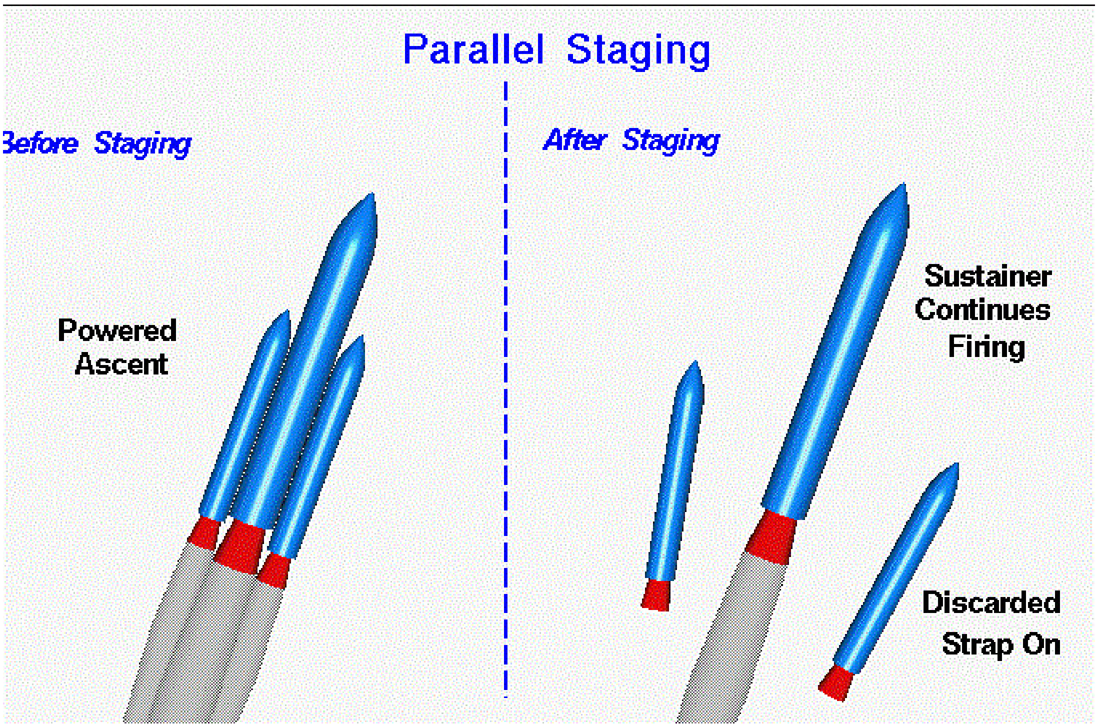

FIGURE 4.10: Parallel staging

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

Some launchers, like the Titan III's and Delta II's, use both serial and parallel staging. The Titan III has a liquid-powered, two stage Titan II for a sustainer and two solid rocket strap-ons at launch. After the solids are discarded, the sustainer engine of the Titan II burns until its fuel is exhausted. Then the second stage of the Titan II is burned, carrying the payload to orbit. The Titan III is another example of a three stage rocke

The payload mass for any stage consists of the mass of all subsequent stages plus the ultimate payload itself. A multistage vehicle with identical specific impulse, payload fraction and structure fraction for each stage is said to have similar stages. For such a vehicle, the payload fraction is maximized by having each stage provide the same velocity increment. For a multistage vehicle with dissimilar stages, the overall vehicle payload fraction depends on how the v requirement is partitioned among stages. Payload fractions will be reduced if the v is partitioned sub-optimally. In designing separation mechanisms, the following factors must be considered: (1) adequate clearance between the separating bodies; (2) shock transmission to the payload or structure of the continuing body; (3) damage to or contamination of the continuing body by debris resulting from the operation of the separation mechanism; and (4) the ability of the mechanism to withstand the natural and induced environments encountered during service.

For a mission to be successful, the separations must occur at the correct times of flight and with minimum changes in the attitude and rotational rates (i.e., tip-off errors) of the continuing body. There must be no recontact between the separating bodies, no detrimental shock loads induced in the structure, and no excessive or harmful debris. A separation mechanism that does not meet these requirements can produce attitude errors and tumble rates of the continuing body that are too large for its attitude-control system to accommodate, can damage its structure and critical equipment, and can cause failure or degradation of the mission. Failure of separation mechanisms has adversely affected mission performance in several instances; for example:

1. A Vanguard satellite failed to achieve orbit because the second stage of the launch vehicle was damaged at separation.
2. On several early satellite launches, booster stages failed to separate.
3. On a military mission, the final booster stage overtook and bumped the spacecraft after separation, damaging critical equipment in the spacecraft.
4. On a recent military mission, an extendible boom was damaged at separation and failed to extend.
5. During an Apollo launch, the pyrotechnic shock of a separation was of sufficient magnitude to close propellant-isolation valves in the reaction-control system of the spacecraft. The crew was unable to maneuver the spacecraft until the valves could be opened.

Chapter 5

DESIGN, MATERIALS AND TESTING OF ROCKETS

5.1 Design considerations in the selection of liquid rocket combustion chamber volume and shape

The volume and shape are selected after evaluating these parameters:

1. The volume has to be large enough for adequate mixing, evaporation, and complete combustion of propellants. Chamber volumes vary for different propellants with the time delay necessary to vaporize and activate the propellants and with the speed of reaction of the propellant combination. When the chamber volume is too small, combustion is incomplete and the performance is poor. With higher chamber pressures or with highly reactive propellants, and with injectors that give improved mixing, a smaller chamber volume is usually permissible.

2. The chamber diameter and volume can influence the cooling requirements. If the chamber volume and the chamber diameter are large, the heat transfer rates to the walls will be reduced, the area exposed to heat will be large, and the walls are somewhat thicker. Conversely,if the volume and cross section are small, the inner wall surface area and the inert

mass will be smaller, but the chamber gas velocities and the heat transfer rates will be increased. There is an optimum chamber volume and diameter where the total heat absorbed by the walls will be a minimum.

3. All inert components should have minimum mass. The thrust chamber mass is a function of the chamber dimensions, chamber pressure, and nozzle area ratio, and the method of cooling.

4. Manufacturing considerations favor a simple chamber geometry, such as a cylinder with a double cone bow-tie-shaped nozzle, low cost materials, and simple fabrication processes.

5. In some applications the length of the chamber and the nozzle relate directly to the overall length of the vehicle. A large-diameter but short chamber can allow a somewhat shorter vehicle with a lower structural inert vehicle mass.

6. The gas pressure drop for accelerating the combustion products within the chamber should be a minimum; any pressure reduction at the nozzle inlet reduces the exhaust velocity and the performance of the vehicle. These losses become appreciable when the chamber area is less than three times the throat area.

7. For the same thrust the combustion volume and the nozzle throat area become smaller as the operating chamber pressure is increased. This means that the chamber length and the nozzle length (for the same area ratio) also decrease with increasing chamber pressure. The performance also goes up with chamber pressure.

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5.1.1 considerations for selection of materials to be used for construction of thrust chambers of liquid rocket engine

The choice of the material for the inner chamber wall in the chamber and the throat region, which are the critical locations, is influenced by the hot gases resulting from the propellant combination, the maximum wall temperature, the heat transfer, and the duty cycle. For high-performance, high heat transfer, regeneratively cooled thrust chambers a material with high thermal conductivity and a thin wall design will reduce the thermal stresses. Copper is an excellent conductor and it will not really oxidize in fuel-rich non-corrosive gas mixtures, such as are produced by oxygen and hydrogen below a mixture ratio of 6.0. The inner walls are therefore usually made of a copper alloy (with small additions of zirconium,silver, or silicon), which has a conductivity not quite as good as pure (oxygen-free) copper but has improved high temperature strength.

5.2 Materials used for manufacturing case of rockets

Three classes of materials have been used: high-strength metals (such as steel, aluminum, or titanium alloys), wound-filament reinforced plastics, and a combination of these in which a metal case has externally wound filaments for extra strength. High-strength alloy steels have been the most common case metals, but others, like aluminum, titanium, and nickel alloys, are also used for manufacturing case of rockets.

5.2.1 Filament wound reinforced plastic cases

Filament-reinforced cases use continuous filaments of strong fibers wound in precise patterns and bonded together with a plastic, usually an epoxy resin. Their principal advantage is their lower weight. Most plastics soften when they are heated above about 180°C or 355°F; they need inserts or reinforcements to allow fastening or assembly of other components and to accept concentrated loads. The thermal expansion of reinforced plastics is often higher than that of metal and the thermal conductivity is much lower, causing a higher temperature gradient.

Typical fiber materials are, in the order of increasing strength, glass, aramids (Kevlar), and carbon. Individual fibers are very strong in tension (2400 to 6800 MPa or 350,000 to 1,000,000 psi). The fibers are held in place by a plastic binder of relatively low density; it prevents fibers slipping and thus weakening in shear or bending. In a filament-wound composite (with tension, hoop, and bending stresses) the filaments are not always oriented along the direction of maximum stress and the material includes a low-strength plastic; therefore, the composite strength is reduced by a factor of 3 to 5 compared to the strength of the filament itself. The plastic binder is usually a thermosetting epoxy material, which limits the maximum temperature to between 100 to 180°C or about 212 to 355°F.

5.2.2 Ablative material

Ans. An ablative material is a composite material of high-temperature organic or inorganic high strength fibers, namely high silica glass, aramids (Kevlar), or carbon fibers, impregnated with organic plastic materials such as phenolic or epoxy resin. The heat transfer properties of the many available ablative and other fiberbased materials will depend on their design, composition, and construction.The orientation of fibrous reinforcements, whether in the form of tape, cloth, filaments, or random short fibers, has a marked impact on the erosion resistance of composite nozzles. When perpendicular to the gas flow, the heat transfer to the wall interior is high because of the short conducting path. Good results have been obtained when the fibers are at 40 to 60° relative to the gas flow over the surface. Nozzle fabrication variables present wide variations in nozzle life for a given design; the variables include the method of wrapping, molding, and curing, resin batch processes, and resin sources.

Tapewrapping is a common method of forming very large nozzles. The wrapping procedure normally includes heating the shaped mandrel(54°C or 130°F), heating the tape and resin (66 to 121°C or 150 to 250°F), pressure rolling the tape of fiber material and the injected resin in place while rolling (35, 000 N/m or 200 !bf/in. width), and maintaining the proper rolling speed, tape tension, wrap orientation, and resin flow rate.

In liquid propellant rockets, ablatives have been effective in very small thrust chambers (where there is insufficient regenerative cooling capacity), in pulsing, restartable spacecraft control rocket engines, and in variable-thrust (throttled) rocket engines.

5.3 Criteria for the selection of materials for Aerospace applications

Based on the knowledge of following problems encountered during the life cycle of a product, material which passes the recommended tests are used for aerospace applications. Materials like aluminum alloys, titanium, magnesium alloys, stainless steel etc. and even composites are used at various places in order to meet specific requirement. The general criteria for selection of materials are as follows:

1. Critical Defect Growth
2. Corrosion Fatigue
3. Corrosion
4. Creep
5. High Cycle Fatigue
6. Hydrogen Embrittlement
7. Hydrogen Environment Embrittlement
8. Low Cycle Fatigue
9. Liquid Metal Embrittlement
10. Overload
11. Oxidation
12. Stress Assisted Grain Boundary Oxidation Crack
13. Stress Corrosion Cracking
14. Stress Rupture
15. High Temperature

Types of loads and stresses a motor case is subjected to during its operation.

(1) Temperature cycling during storage, or thermal stresses and strains)

(2) Corrosion (moisture/chemical, galvanic, stress corrosion, or hydrogen embrittlement).

(3) Space conditions: vacuum or radiation.

Case segments

For very large and long motors both the propellant grain and the motor case are made in sections which are called case segments and are mechanically attached and sealed to each other at the launch site.

characteristics of maraging steels The maraging steels have strengths up to approximately 300,000 psi in combination with high fracture toughness. The term maraging is derived from the fact that these alloys exist as relative soft low-carbon marten sites in the annealed condition and gain high strength from aging at relatively low temperatures.

constructional aspects of motor case with respect to material selection.

5.4 Metal Cases

Metal cases have several advantages compared to filament-reinforced plastic cases: they are rugged and will take considerable rough handling (required in many tactical missile applications), are usually reasonably ductile and can yield before failure, can be heated to a relatively high temperature (700 to l000°C or 1292 to 1832°F and higher with some special materials), and thus require less insulation. They will not deteriorate significantly with time or weather exposure and are easily adapted to take concentrated loads, if made thicker at a flange or boss. Since the metal case has much higher density and less insulation, it occupies less volume than does a fiber-reinforced plastic case; therefore, for the same external envelope it can contain somewhat more propellant.

High-strength alloy steels have been the most common case metals, but others, like aluminum, titanium, and nickel alloys, have also been used. Extensive knowledge exists for designing and fabricating motor cases with low-alloy steels with strength levels to 240,000 psi. The maraging steels have strengths up to approximately 300,000 psi in combination with high fracture toughness. The term maraging is derived from the fact that these alloys exist as relative soft low-carbon martensites in the annealed condition and gain high strength from aging at relatively low temperatures.

The HY steels (newer than the maraging steels) are attractive because of their toughness and resistance to tearing, a property important to motor cases and other pressure vessels because failures are less catastrophic. This toughness characteristic enables a ”leak before failure” to occur, at least during hydrostatic proof testing. The HY steels have strengths between 180,000 and 300,000 psi (depending on heat treatment and additives).

Need of using ceramic materials in missiles

In relatively small (low temperature) rockets, the interior walls of the combustion chamber and nozzle may be lined with a heat-resistant (refractory) ceramic material. The ceramic gets hot, but because it is a poor conductor of heat, it prevents the metal walls of the motor/engine from becoming overheated during the short operating period.

Structural properties of aerospace materials are affected at low and high temperatures

The very high temperatures generated in the combustion chamber transfer a great deal of heat energy to the combustion chamber and nozzle walls. This heat, if not dissipated, will cause most materials to lose strength. Without cooling the chamber and nozzle walls, the combustion chamber pressures will cause structural failure. There are many methods of cooling, all with the objective of removing heat from the highly stressed combustion chamber and nozzle.

5.5 Selection of materials

5.5.1 Re – entry nose cones

The term nose cone is used to refer to the forward most section of a rocket, guided missile or aircraft. The cone is shaped to offer minimum aerodynamic resistance. Due to the extreme temperatures involved, nose cones for high-speed applications (eg. hypersonic speeds or atmospheric reentry of orbital vehicles) have to be made of refractory materials. Pyrolytic carbon is one choice, reinforced carbon-carbon composite or HRSI ceramics are other popular choices. Other design strategy is using ablative heat shields, which get consumed during operation, disposing of excess heat that way. Materials used for ablative shields include, for example carbon phenolic, polydimethylsiloxane composite with silica filler and carbon fibers, or as in of some Chinese FSW reentry vehicles, oak wood.

5.5.2 Wing leading edges and Rocket Nozzle Throat Inserts:

Carbon/Carbon (C/C) is a lightweight, high-strength composite material capable of withstanding temperatures over 3000°C in many environments. Carbon/Carbon Composites use the strength and modulus of carbon fibers to reinforce a carbon matrix to resist the rigors of extreme environments.

Using FMI special weaving techniques, composite structures can be tailored to meet varied physical and thermal requirements through weaving architecture design. FMI woven reinforcements are impregnated with resin or pitch and carbonized to yield finished, fully dense composites using well controlled temperature and pressure processes (up to 15000 psi).
At one-tenth the density, C/C offers a high performance, cost effective alternative to refractory metals. Aerospace components commonly fabricated from C/C include rocket motor nozzle throats and exit cones, nose tips / leading edges and thermal protection systems. Reliable performance is the most critical requirement of these components. FMI C/C Composites have demonstrated reliability and reduced systems costs, especially when multiple components in an assembly can be replaced with a one-piece C/C design. Commercial applications of Carbon/Carbon materials include furnace fixturing, heatshields, load plates, heating elements and X-ray targets.

5.6 Rocket testing

5.6.1 Types of testing

Before rocket propulsion systems are put into operational use, they are subjected to several different types of tests, some of which are outlined below in the sequence in which they are normally performed.

1. Manufacturing inspection and fabrication tests on individual parts (dimensional inspection, pressure tests, x-rays, leak checks, electric continuity, electromechanical checks, etc.).

2. Component tests (functional and operational tests on igniters, valves, thrusters, controls, injectors, structures, etc.).

3. Static rocket system tests (with complete propulsion system on test stand): (a) partial or simulated rocket operation (for proper function, calibration, ignition, operation-often without establishing full thrust or operating for the full duration); (b) complete propulsion system tests (under rated conditions, off-design conditions, with intentional variations in environment or calibration). For a reusable or restartable rocket propulsion system this can include

many starts, long-duration endurance tests, and post operational inspections and reconditioning.

4. Static vehicle tests (when rocket propulsion system is installed in a restrained, nonflying vehicle or stage).
5. Flight tests: (a) with a specially instrumented propulsion system in a developmental flight test vehicle; (b) with a production vehicle.

Each of these five types of tests can be performed on at least three basic types of programs:

1. Research on and development or improvement of a new (or modified) rocket engine or motor or their propellants or components.
2. Evaluation of the suitability of a new (or modified) rocket engine or motor for a specified application or for flight readiness.
3. Production and quality assurance of a rocket propulsion system.

The first two types of programs are concerned with a novel or modified device and often involve the testing and measurement of new concepts or phenomena using experimental rockets. The testing of a new solid propellant grain, the development of a novel control valve assembly, and the measurement of the thermal expansion of a nozzle exhaust cone during firing operation are examples.

Production tests concern themselves with the measurement of a few basic parameters on production propulsion systems to assure that the performance, reliability, and operation are within specified tolerance limits. If the number of units is large, the test equipment and instrumentation used for these tests are usually partly or fully automated and designed to permit the testing, measurement, recording, and evaluation in a minimum amount of time.

5.7 Test facilities and safeguards

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

1. A test cell or test bay where the article to be tested is mounted, usually in a special test fixture. If the test is hazardous, the test facility must have provisions to protect operating personnel and to limit damage in case of an accident.

2. An instrumentation system with associated computers for sensing, maintaining, measuring, analyzing, correcting, and recording various physical and chemical parameters. It usually includes calibration systems and timers to accurately synchronize the measurements.

3. A control system for starting, stopping, and changing the operating conditions.

4. Systems for handling heavy or awkward assemblies, supplying liquid propellant, and providing maintenance, security, and safety.

5. For highly toxic propellants and toxic plume gases it has been required to capture the hazardous gas or vapor (firing inside a closed duct system), remove almost all of the hazardous ingredients (e.g., by wet scrubbing and/or chemical treatment), allow the release of the non-toxic portion of the cleaned gases, and safely dispose of any toxic solid or liquid residues from the chemical treatment. With an exhaust gas containing fluorine, for example, the removal of much of this toxic gas can be achieved by scrubbing it with water that contains dissolved calcium; it will then form calcium fluoride, which can be precipitated and removed.

6. In some tests specialized test equipment and unique facilities are needed to conduct static testing under different environmental conditions or under simulated emergency conditions. For example, high and low ambient temperature tests of large motors may require a temperature-controlled enclosure around the motor; a rugged explosion-resistant facility is needed for bullet impact tests of propellant-loaded missile systems and also for cook-off tests, where gasoline or rocket fuel is burned with air below a stored missile. Similarly, special equipment is needed for vibration testing, measuring thrust vector forces and moments in three dimensions, or determining total impulse for very short pulse durations at low thrust.

5.8 Safety provisions in a modern testfacility

It is common practice to train the test crew and go through repeated dry runs, to familiarize each person with his or her responsibilities and procedures, including the emergency procedures. Typical personnel and plant security or safety provisions in a modern test facility include the following:

1. Concrete-walled blockhouse or control stations for the protection of personnel and instruments remote from the actual rocket propulsion location.
2. Remote control, indication, and recording of all hazardous operations and measurements; isolation of propellants from the instrumentation and control room.
3. Automatic or manual water deluge and fire-extinguishing systems.
4. Closed circuit television systems for remotely viewing the test.
5. Warning signals (siren, bells, horns, lights, speakers) to notify personnel to clear the test area prior to a test, and an all-clear signal when the conditions are no longer hazardous.
6. Quantity and distance restrictions on liquid propellant tankage and solid propellant storage to minimize damage in the event of explosions; separation of liquid fuels and oxidizers.
7. Barricades around hazardous test articles to reduce shrapnel damage in the event of a blast.
8. Explosion-proof electrical systems, spark-proof shoes, and non spark hand tools to prvent ignition of flammable materials.
9. For certain propellants also safety clothing, including propellant- and fire-resistant suits, face mask and shields, gloves, special shoes, and hard hats.

5.9 Terminology for the study of atmospheric diffusion of exhaust

1. **Micrometeorology** Study and forecasting of atmospheric phenomena restricted to a region approximately 300 m above the earth's surface and a horizontal distance of approximately 5 miles.

2. **Lapse Rate**. The rate of decrease in temperature with increasing height above the earth's surface. The United States Standard Atmosphere has a lapse rate of about 6.4°C per 1000 m. Lapse rate is also affected by altitude, wind, and humidity.

3. **Inversion, or Inversion Layer**. Condition of negative lapse rate (temperature increases with increasing height). Usually formed near the ground at night.

5.10 Flight testing

Flight testing of rocket propulsion systems is always conducted in conjunction with tests of vehicles and other systems such as guidance, vehicle controls, or ground support. These flights usually occur along missile and space launch ranges, sometimes over the ocean. If a flight test vehicle deviates from its intended path and appears to be headed for a populated area, a range safety official (or a computer) will have to either cause a destruction of the vehicle, abort the flight, or cause it to correct its course. Many propulsion systems therefore include devices that will either terminate the operation (shut off the rocket engine or open thrust termination openings into rocket motor cases as) or trigger explosive devices that will cause the vehicle (and therefore also the propulsion system) to disintegrate in flight. Flight testing requires special launch support equipment, means for observing, monitoring, and recording data (cameras, radar, telemetering, etc.), equipment for assuring range safety and for reducing data and evaluating flight test performance, and specially trained personnel. Different launch equipment is needed for different kinds of vehicles. This includes launch tubes for shoulder-held infantry support missile launchers, movable turret-type mounted multiple launchers installed on an army truck or a navy ship, a transporter for larger missiles, and a track-propelled launch platform or fixed complex launch pads for spacecraft launch vehicles. The launch equipment has to have provisions for loading or placing the vehicle into a launch position, for allowing access of various equipment and connections to launch support equipment (checkout, monitoring, fueling, etc.), for aligning or aiming the vehicle, or for withstanding the exposure to the hot rocket plume at launch. During experimental flights extensive measurements are often made on the behavior of the various vehicle subsystems; for example, rocket propulsion parameters, such as chamber pressure, feed pressures, temperatures, and so on, are measured and the data are telemetered and transmitted to a ground receiving station for recording and monitoring. Some flight tests rely on salvaging and examining the test vehicle.