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SCHOOL OF MECHANICAL ENGINEERING
DEPARTMENT OF AEROUTICAL ENGINEERIN

Unit-1-Aerospace Propulsion- SAEA1502

1: Performance of turbojets, ramjets at high speeds – limitations. Need for supersonic combustion:

- The renewed interest in high-speed propulsion has led to increased activity in the development of the supersonic combustion ramjet engine for hypersonic flight applications.
- In this flight regime, the scramjet engine"s specific thrust exceeds that of other propulsion systems.
- Use of air breathing propulsion systems like, scramjets from takeoff to the edges of the atmosphere has the potential to reduce costs of space launch considerably.
- The hypersonic flight regime is commonly considered to begin when velocities exceed Mach 6
- Defence applications of scramjets in missiles is also very sought after due to the very short reaction times associated with high speed of the missile system
- Subsonic combustion, which technologically is easier to manage with the current knowledge, would be associated, in the hypersonic regime, with high stagnation temperatures that would lead to unacceptable dissociation levels, and hence an inability to materialize the energy rise expected through chemical reactions
- Combined cycle engines: No single-engine cycle exists that can efficiently cover the whole range of a flight from takeoff to orbit insertion; therefore combined cycles are of particular interest for the design of the scramjet cycle

Limitations of Turbojets/Turbofans and Ramjets at High Speeds:

Performance based differences between the different engine cycles are clearly illustrated in the fuel specific impulse, *Specific impulse* vs *Mach number* diagram shown.

$$I_{sp} = \frac{Thrust}{fuel\ Mass\ flow\ rate}$$

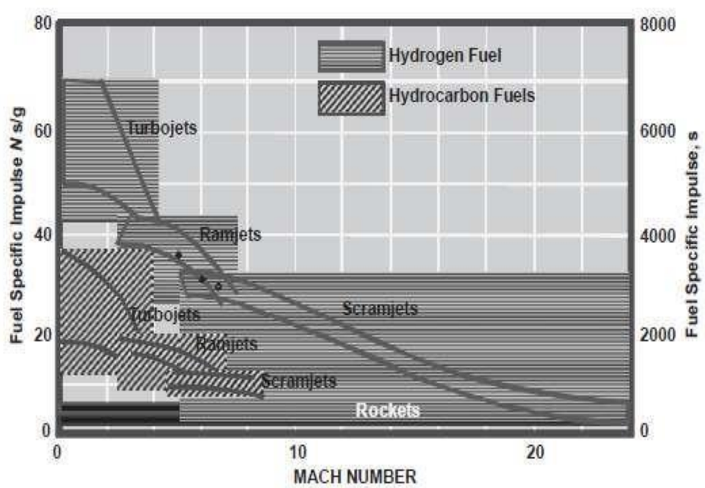


Fig:1.1 Mach 3 flight

The diagram shows that around Mach 3 flight regime the subsonic combustion ramjet becomes more efficient as a propulsive system in comparison with the turbine-based engines (turbojets of turbofans).

Around Mach 2.5, stagnation pressure captured by the intake is around 11.2,

$\frac{P_0}{P}$ = $\left[1+\frac{\gamma+1}{2}M^2\right]$ to thepower$\frac{\gamma}{\gamma-1}$ = 11.2

Assuming intake efficiency of around 60-70%, this ram pressure capture by the intake works out to 7.0. So, beyond 2.5 Mach, we do not need a compressor (and turbine combination). Ramjet is preferred in the speed range of 2-4 Mach, due to its higher specific thrust($T/_W$). Ramjets are used in military missiles like Akash, Brahmos missiles.

2. Need For Supersonic Combustion; Beyond Mach 4.0:

When the free stream flow is slowed down to subsonic speeds, the stagnation temperature is around 980 k, whereas at free stream Mach number of 6.0, it raises to 1800 k. When speeds increase to Mach 7.0, the stagnation temperature rises to 2300 k ATF, ie hydrocarbon fuel has adiabatic flame temperature of around 2300 k, so beyond Mach 7.0, heat addition is not possible by burning fuel. Therefore, heat must be added at lower stagnation temperatures ie at supersonic speeds. All hypersonic transport propulsion systems need supersonic combustion ram jets (Scramjets).

Also, beyond Mach 5, specific impulse of ramjet decays rapidly and the scramjet delivers a higher specific impulse at higher speeds. The rocket"s specific impulse is considerably lower than the other propulsion system but it offers operation capabilities from sea-level static to beyond the atmosphere which no other propulsion system mentioned here can do.

The low specific impulse of rockets, in comparison with the other propulsion systems clearly eliminates the rocket from consideration for long range cruise but as the Mach number continues to increase in the hypersonic regime the scramjet specific impulse approaches that of the rocket engine.

Since, the very high Mach numbers are expected for operation close to the edge of the atmosphere, where the continually decreasing air density will eventually require that the engine makes the transition to rocket operation for orbit insertion.

Historically, multiple-staged vehicles have been designed to operate with a single type of propulsion system for each stage. Stages are optimized for different altitude/Mach number regimes in the trajectory, increasing the overall system specific impulse.

Physical Aerodynamic Aspects

1. **Thin Shock Layers**: The oblique shock wave formed at the vehicle body is very thin and makes a much smaller angle (around 2.5). The shock waves also lie close to the body. This leads to merging of shock waves with the boundary layer, which needs to be considered while predicting the pressure distribution over the body.

2. **Entropy Layer:** Theshock wave around the blunt body (a space vehicle)in a hypersonic flow is thin, highly curved and is associated with large velocity gradients across the shock wave. The region behind the shock wave has strong thermodynamic changes and high losses and is called “entropy layer”. The entropy layer causes **high aerodynamic heating** of the surface. This requires effective cooling systems.

3. **Viscous Interaction**: The thickness of boundary layer on the surface of the vehicle is directly proportional to the Mach number. As a result the thickness of the boundary layer is very large at high Mach numbers. The thick boundary layer affects the flow outside the boundary layer called viscous interaction, which **increases the drag and aerodynamic heating**.

4. **High-Temperature effects**:The high kinetic energy flow slows down by the effect of boundary layer interaction and results in very high temperatures. Additionally, the region behind the bow shock wave is another reason for rise in temperature. The high temperatures cause chemical reactions in the flow through molecular dissociations, resulting in high zones of aerodynamic heating of the surface.

5. **Low Density Flow**: At very high altitudes beyond 60 km, air is no more a continuous medium, but rarefied and very low density medium. This alters the aerodynamic force coefficients, heat transfer coeffcients vary considerably and need to be factored in predicting vehicle aerodynamic and propulsive behavior.

6. **Problems of Combustion in High Speed Flow**:

Slow Reaction rate & Low residence time- Supersonic combustion poses following problems

(a) **Reduces O_2 Content**: At high temperatures, Oxygen and Nitrogen in the air react with each other, thereby **reducing oxygen content** available for combustion. Corresponding to M_∞ of 4.0, O_2 content is 0.21; M_∞ of 6.0, O_2 content is 0.207; further reduces at M_∞ of 9.0, O_2 content is 0.17.

(b) **Reduces Reaction Times**: At high Mach number in the combustion chamber, static pressure is low, therefore the reaction rate of combustion is slow. (Reaction time ∞p^2)

(c) **Reduces Residence Times**: As the flow is passing the combustion chamber at supersonic speeds, the residence time of air in the combustion chamber is very low.

(d) **Requires Larger Combustion Volumes**: The low pressures may demand larger combustion volume, a feature that may be critical for the design of hypersonic vehicle propelled by a scramjet.

(e) Fuel needs to be injected into the combustor that has supersonic flow inside with large enough static temperatures, and much larger stagnation temperatures

(f) Avoidance of hot pockets near the walls implies that the fuel be injected from centrally located struts

7. **Interaction/Integration of Airframe and Engine**: This necessitates very long combustion chamber. In Scramjet aircraft, the entire lower body of the aircraft is engine. The front portion of the underside operates as diffuser, with rear portion providing combustion and expansion surface

8. **Design and Testing difficulties of integrated design**: we have not perfected the integrated design of airframe and engine as yet. Also, testing of integrated aircraft needs huge wind tunnel, with very high costs involved in providing power of supersonic flow simulation in the wind tunnel.

(Source : Introduction to flight by John D Anderson Jr)

Criticality of efficient diffusion and Acceleration- High Speed Combustion

Fuel needs to be injected into the combustor that has supersonic flow inside with large enough static temperatures, and much larger stagnation temperatures.

Avoidance of hot pockets near the walls implies that the fuel be injected from centrally located struts. The usual circular configuration for combustors can be sacrificed in favor of a rectangular configuration.

Typical velocities in the combustion chamber are about 1 to 1.5 km/s and the Mach numbers will be 1.4 to 2.3 for a typical combustor entry Mach number of 2.5. The residence time will be in micro-seconds.

Main problems associated with supersonic combustion are as follows**:**

- Turbulent mixing,
- Aerodynamic effects of heat release
- Non-equilibrium effects in diffusion flames.

Diffusion flame combustion: In the design of diffusion flame for supersonic combustion, the fuel is injected at the inlet parallel to the air flow. (Fuel pre injection in inlets or isolators holds considerable potential enhances mixing, flame stability, and combustion efficiency for scramjet engines. However, it is not considered for practical applications)

Turbulent Mixing begins immediately and combustion quickly follows. However, for the diffusion flame to exist the **chemical reaction time must be fast (small)** compared with the mixing or mechanical time. This fact limits the applicability of the diffusive mode of combustion to some regions of the flight corridor. The supersonic combustion process is controlled by both chemical kinetics and mixing.

Mixing layers of air at supersonic flows and fuel are characterized by **large-scale eddies** that form due to the **high shear** between both the streams. These eddies entrain fuel and air into the mixing region. Stretching occurs in the interfacial region between the fluids due to compressible shear/mixing layers, leading to increased surface area and locally steep concentration gradients. Molecular diffusion then occurs across the strained interfaces. Design of scramjet combustor must take into account the requirement that the fuel be well mixed with the air within a few microseconds.

The criticality of timing must be such that the ignition delay time plus the time to complete the reaction are less than the **residence time** of flow through the combustor. This chemical kinetic limitation can be overcome by maintaining the local static temperatures sufficiently high. The large localized heat release in a given section gives rise to shock waves which spread the heat release in the flow direction resulting in an advantage of the diffusive mode of supersonic combustion.

Aerodynamic Effect of Heat release: Results show very complex interactions between the sonic H_2 fuel cross flow injections and the airstream flowing at $M \gg 1$. A bow shock forms ahead of each H_2 injector. The interaction between bow shocks and boundary layers leads to separation zones where H_2 re-circulates. The shock structure allows the required pressure rise, thus isolating the combustion process from the inlet compression process, thus acting to prevent inlet surge or “unstart”.

Non-equilibrium effects in diffusion flames: The local heat release leads to enhanced local temperatures. Similarly, there would be non-uniform temperature distribution since the fuel sprays are introduced over parts of the cross section. This leads to non-uniformity in other quantities as well. The flow field over the vehicle at M = 10 would be reactive with significant dissociation of the air taking place.

Scramjet Engine-Construction: Scramjet engine is characterized by slow reaction times and high flow speeds ie low residence times in the engine. The engine needs larger combustion volumes; leading to integrated design of airframe and engine. In scramjet aircraft, the entire lower body of the aircraft is occupied by the engine. The front (fore) portion of the underside operates as external/internal diffuser, with rear (aft) portion providing expansion surface.

The scramjet consists of

- Diffuser (compression component) consisting of external ramp intake and engine intake

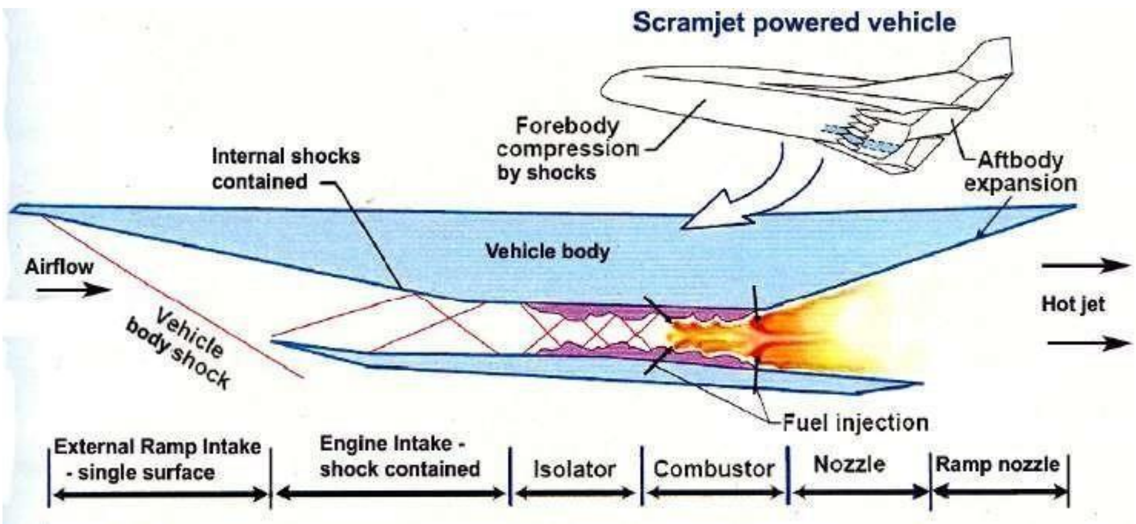
- Isolator
- Supersonic combustor
- Exhaust nozzle or aft body expansion component
- 

Fig:1.2 Supersonic combustor

Diffuser

- ➢ It consists of fore-body external intake and internal intake
- ➢ The fore-body provides the initial external compression and contributes to the drag and moments of the vehicle.
- ➢ The internal inlet compression provides the final compression of the propulsion cycle.

Since the flow upstream is supersonic, the geometry of the diffuser is entirely convergent.The oblique shock wave emanating from the vehicle fore-body obtains much of the desired compression and deceleration. The engine is designed to take advantage of the compression through shock waves and reduce the load on the diffuser. The air in the captured stream tube undergoes a reduction in mach number with an attendant increase in pressure and temperature as it passes through the system of shock waves in the fore body and internal inlet.

The air induction phenomena include

- ➢ Formation of vehicle body shock
- ➢ Formation of isentropic turning mach waves
- ➢ Shock-boundary layer interaction

- Non-uniform flow conditions

The vehicle body oblique shock becomes thinner and stronger and hugs the bounding fore-body surface more closely as the free stream mach number increases.

Flow separation & attachment: When the oblique shocks impinge upon the boundary layer, they impose an abrupt, discontinuous increase in pressure on the boundary layer immediately close to the surface. The most violent effect of the shock wave will cause the boundary layer to separate. Although, reattachment eventually occurs, it results in finite region of reversed/recirculation flow. There are situations when reattachment does not take place.

Separation of flow results in increase in pressure or form drag, increases the thickness and distortion further downstream. The increased transport of high enthalpy gases from the free stream to the boundary layer increases the wall heat transfer rates and causes hot spots. Two methods in design of air induction system are the positioning of oblique shocks avoiding interference with each other and providing blow holes to remove laminar layer turning it turbulent.

Inlet Operation:

Two modes of inlet operation are possible. They are

- Sub-critical or "unstart" condition
- Supercritical or "started" condition

Supercritical Operation of the Inlet: At slow speeds, the inlet will not capture all the free stream air and will result in causing spillage of air, contributing to spillage drag. This condition of inlet is termed as "**sub-critical**" and should be avoided. However, as the free stream mach number increases, the normal shock is swallowed inside and the flow is said to be **supercritical or "started**".The intake area is sensitive to conditions in the combustor and the design must cater for avoiding any back pressure built up which will cause flow "unstart" condition in the inlet.

Inlet Unstart: Three types of disturbances can cause inlet unstart.

- First is when the free stream mach number is reduced sufficiently below the starting value.
- Second, unstart will occur if the flow reaching the inlet face is distorted.
- And finally, unstart can occur if the back pressure from downstream ie combustor is increased. The back pressure can increase if the chemical energy release is suddenly increased or the in case of a reduction in throat area of the nozzle.

Unstart must be avoided at all costs since the condition is an extremely unsteady and violent phenomenon in which the swiftly moving shock waves can impose heavy transient loads on the structure.

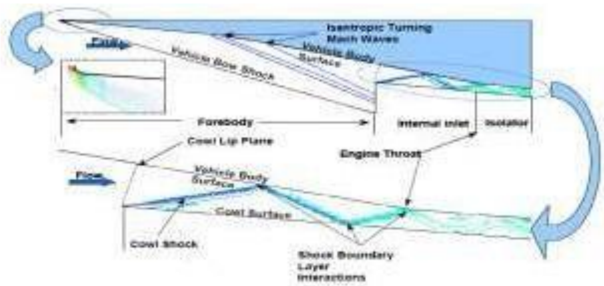


Fig:1.3 Isolator

Isolator: Isolator is constant area diffuser containing the internal shock structure, swallowed during supercritical operation of the inlet (or during operation after the inlet “started”). The isolator is inserted before the combustor to diffuse the flow further, through a shock train, producing desired flow speeds in the combustors. The function of the isolator is:

- The shock train provides a mechanism for the supersonic flow to adjust to a static back pressure higher than its inlet static pressure
- The isolator cross-sectional area may be constant or slightly divergent to accommodate boundary layer separation.
- When the combustion process begins to separate the boundary layer in the combustor, a pre-combustion shock train forms.
- The shock structure allows the required pressure rise, thus isolating the combustion process from the inlet compression process. Thus the isolator functions to prevent inlet surge or “unstart”.

Combustor: Main features include:

- Avoidance of hot pockets near the walls implies that the fuel be injected from centrally located struts.
- The usual circular configuration for combustors can be sacrificed in favor of a rectangular configuration.
- Typical velocities in the combustion chamber are about 1 to 1.5 km/s and the Mach numbers will be 1.4 to 2.3 for a typical combustor entry Mach number of 2.5

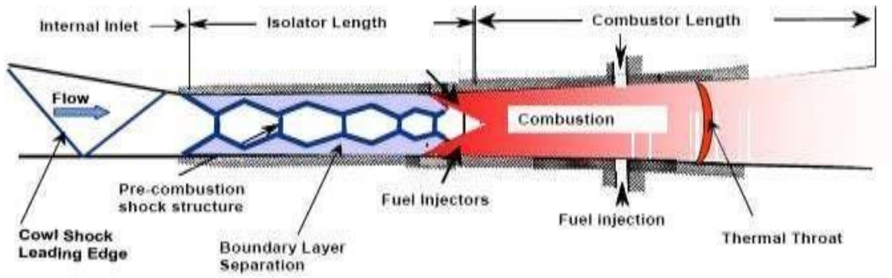


Fig: 1.4 combustor

Difficult to Control: The high speed flow makes the control of the flow within the combustor very difficult. Since the flow is supersonic, downstream influence does not propagate within the free stream of the combustion chamber.

Fuel Injection: Fuel injection and management is also potentially complex. One possibility would be that the fuel be pressurized by a turbo pump, heated by the fuselage, sent through the turbine and accelerated to higher speeds by a nozzle.

The air and fuel stream are crossed in a comb like structure with fuel struts, which generates a large interface. Turbulence due to the higher speed of the fuel leads to additional mixing. Complex fuels like kerosene need a long engine to complete combustion.

Criticality of Reaction Rates: The minimum Mach number at which a scramjet can operate is limited by the fact that the compressed flow must be hot enough to burn the fuel, and have pressure (static) high enough that the reaction be finished before the air moves out of the combustor. Additionally, in order to be called a scramjet, the compressed flow must **still be supersonic** after combustion.

Combustion limits: Two limits are very critical for the operation

- First, since when a supersonic flow is compressed, it slows down, the level of compression must be low enough (or the initial speed high enough) not to slow the gas below Mach 1. If the gas within a scramjet goes below Mach 1 the engine will "choke", transitioning to subsonic flow in the combustion chamber. Additionally, the sudden increase in pressure and temperature in the engine can lead to an acceleration of the combustion, leading to the combustion chamber exploding.

- Second, the heating of the gas by combustion causes the speed of sound in the gas to increase (through increase of $\sqrt{t}$and hence cause Mach number to decrease) even though the gas is still travelling at the same speed. Forcing the speed of air flow in the combustion chamber under Mach 1 in this way is called "**thermal choking**".

- A thermal throat results when the flow is slowed through tailored heat for causing dual-mode operation.

- There are engine designs where a ramjet transforms into a scramjet over the Mach 3-6 range, known as dual-mode scramjets.

Constant dynamic flight path: Because air density reduces at higher altitudes, a scramjet must climb at a specific rate as it accelerates to maintain a constant air pressure at the intake. This optimal climb/descent profile is called a "constant dynamic pressure path".

It is thought that scramjets might be operable up to an altitude of 75 km.

Expansion System:

- The expansion system, consists of

  a. Internal nozzle

  b. Vehicle aft body

- It completes the propulsion flow path and controls the expansion of the high pressure and temperature gas mixture to produce net thrust.

At the end of the combustion process, the air enthalpy has increased sufficiently to generate thrust through expansion in the nozzle. During the expansion process, the potential energy generated by the combustor is converted into kinetic energy. The scramjet nozzle would be of an open type, with much of the vehicle"s lower surface acting as the part of the nozzle.

A hinged flap is provided at the end of the reflecting surface to facilitate variable geometry. The hypersonic nozzles are referred to as single-sided nozzles, unconfined nozzles or simply expansion ramps. Since the flow is supersonic from entry to exit, mathematical treatment is simpler than conventional nozzles.

Scramjet Nozzle physical phenomena includes

- Boundary layer effects
- Non-uniform flow conditions
- Shock layer interaction and
- Three-dimensional effects.

Because a substantial part of the vehicle is dedicated to nozzle expansion, considerable lift and pitch moments are produced by the pressure distribution on this part of the after body, complicating the nozzle design and vehicle integration.A hinged flap is provided at the end of the reflecting surface to facilitate variable geometry. The hypersonic nozzles are referred to as single-sided nozzles, unconfined nozzles or simply expansion ramps.

Since the flow is supersonic from entry to exit, mathematical treatment is simpler than conventional nozzles. Operation of expansion system is shown below:

Isolators-A relook: Once, we complete study of the scramjet engine, before studying dual-mode ramjet-scramjet combined engines, we need to examine the behavior of shock waves in constant area ducts, isolators.

- Since the constant area flow devices produce a static pressure rise, they are called constant area diffusers.
- A supersonic flow field in a constant area duct will result in the normal shock to cause boundary layer separation, forcing the normal shock to take on an altogether different appearance.
- Two flow fields are observed while flowing through the constant area isolator "diffuser", a two dimensional flow with entry at I and exit at e, also is entry to burner.
- Please note that the back pressure in the diffuser is a due to chemical energy release in the burner, or choking of a downstream area, but may also be caused by obstructions as fuel injectors, etc.

The pattern of shock waves is based on inlet mach number as below:

- When the inlet Mach number is low supersonic 1, $>$ a normal shock train forms, the exit Mach number is subsonic.
- When the inlet Mach number is a high supersonic, the pattern is oblique shock train, with thicker boundary layer in the tube, the exit Mach number > 1.
- A rough indicator for dividing line between formation of normal and oblique shock trains is when $2 < M_i < 3$

The shock train provides a mechanism for the incoming supersonic flow to adjust to a static backpressure higher than the inlet static pressure. If the back pressure in the burner should exceed the maximum possible, the whole shock train will be disgorged and the inlet will "unstart". Drawings of isolator operation with normal shock train as well as oblique shock train are given below:

Dual-Mode Engines:

- The final application of a scramjet engine is likely to be in conjunction with engines which can operate outside the scramjet's operating range.
- Dual-mode scramjets combine subsonic combustion for operation at lower speeds, and

- Rocket-based combined cycle (RBCC) engines supplement a traditional rocket's propulsion with a scramjet, allowing for additional oxidizer to be added to the scramjet flow.
- RBCCs offer a possibility to extend a scramjet's operating range to higher speeds.

Working Principle Dual-mode Scramjet:

A pure ramjet engine operates at supersonic speeds, but with subsonic combustion, requires two area restrictions or physical throats. The first throat, at the outlet from the inlet diffuser, is required to stabilize the normal shock formation in order to deliver subsonic flow to burner. The second throat is located downstream of the burner, is required to accelerate the subsonic flow to supersonic velocities. It is important to note that flow is choked (M=1) only in the second throat. A pure scramjet engine has no physical throat. The Dual-mode engine uses “no-throat” geometry, capable ofswitching over from ramjet or scramjet mode. Employing area constrictions mean limiting the mass flow rate at high flight mach numbers.

Ramjet mode (subsonic operation)-Thermal Throat: In the ramjet mode flow must be subsonic at the burner entry. The transition from supersonic flow to subsonic flow is accomplished in the dual-mode engine by means of a constant-area diffuser called the isolator.

In order that the burner entry flow is subsonic, the flow must be choked (M=1) somewhere downstream, which causes large back pressure at p_3 burner entry. This back pressure causes a normal shock train to form in the isolator. As long as the back pressure does not exceed isolator"s ability to maintain the normal shock train, the isolator will perform as a variable area diffuser to enable subsonic flow in the burner. The function of the second ramjet throat, to choke the flow and accelerate the subsonic flow to supersonic speeds in the nozzle is provided by the means of a “choked thermal throat”. The thermal throat is brought about by choosing the right combination of area distribution and fuel-air mixing/combustion.

The heating of the gas by combustion causes the speed of sound in the gas to increase (through increase of $\sqrt{t}$ and hence cause Mach number to decrease) even though the gas is still travelling at the same speed. Forcing the speed of air flow in the combustion chamber under Mach 1 in this way is called "**thermal choking**".

A **thermal throat** results when the flow is slowed through tailored heat for causing dual-mode operation. Local heat release leads to enhanced temperatures. This increase causes increased acoustic velocity (~ $\sqrt{T}$) and reduction in Mach number even if the local speed is unaltered.

Scramjet Mode: In scramjet mode, there is no need for a physical throat either upstream or downstream of the burner. The flow is supersonic at burner entry. The isolator will contain an oblique shock train with a supersonic core flow. In this mode, the isolator will absorb or contain any pressure or thermal transients caused by the heat addition in supersonic combustion mode.

The back pressure from the burner is prevented to propagate upstream and causeunstart of the engine.

Transition from Scramjet mode to Ramjet Mode: Transition from supersonic to subsonic combustion requires a normal shock train to form in the isolator at the entry of burner. Back pressure is created in the burner by the formation of thermal throat in the burner, by either varying the area ratio in the burner or increasing the fuel flow rate to increase heat addition. Varying the area ratio to create back pressure may be accomplished by a throttling mechanism in the flow path. This process can be reversed reducing the back pressure sufficiently until flow un- chokes and supersonic flow is re-established in the burner.

Operational Characteristics-Scramjets

- For scramjet operation, the shock wave must exist in a stable form all the way through the engine and back out the rear into what is called the external nozzle.
- A shockwave- powerful enough to stand up to the pressures and stresses created by burning jet fuel will not occur until roughly Mach 3. The requirement for this „*standing wave*" limits the scramjet to Mach 3 and up.
- Combined Cycle Engines are being contemplated to complement the scramjet in order to enhance the operational envelope
- The scramjet will, in fact, substitute the mechanical throat with a **thermal throat** that results when the flow is slowed through tailored heat release.

Applications of Scramjets:

- Weapons systems -hypersonic cruise missiles
- Aircraft systems - global strike / reconnaissance
- Space access systems that will take off and land horizontally like commercial Airplanes
- Using these Scramjet technologies, along with additional ground-and flight-test experiments, will pave the way for affordable and reusable air-breathing hypersonic propulsion systems such as missiles, long range aircraft and space-access vehicles

Advantages:

1. Need not carry oxygen on board
2. No rotating parts makes it easier to manufacture than a turbojet

3. Has a higher specific impulse (change in momentum per unit of propellant) than a rocket engine; could provide between 1000 and 4000 seconds, while a rocket only provides 450 seconds or less

4. Higher speed could mean cheaper access to outer space in the future

Thrust Augmentation:

A variety of schemes for generating thrust beyond that of basic ramjet or scramjet are available for application at critical stages of the mission, when the net thrust or specific impulse of the vehicle approaches near zero for whatever reason.

The most effective thrust augmentation devices are those that naturally integrate themselves geometrically and mechanically into the existing ramjet or scramjet engine flow path. This, in general minimizes the additional volume, weight and cost required. Thus the separate turbojet engines for take-off thrust and separate rocket engines extra thrust during the mission are best choices. The emphasis of thrust augmentation is on the magnitude of thrust rather than the specific engine performance parameters. The techniques include

- The Ejector Ramjet
- External Burning
- Fuel and oxidizer enrichment

Ejector Ramjet Engine:

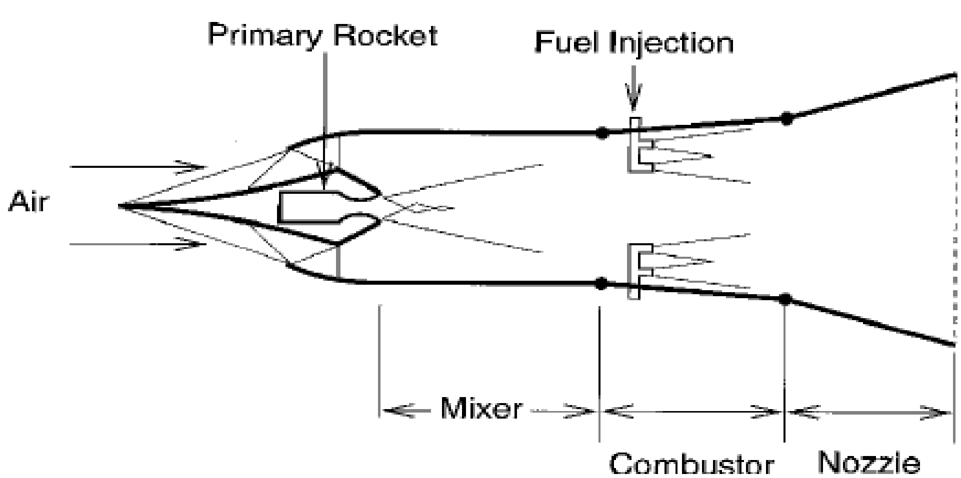


Fig: 1.5 rocket subsystem

It consists of a rocket subsystem incorporated in an air-breathing engine along with an inlet, mixer, combustion chamber and nozzle. Fuel injections sites can be located at several locations along the duct to optimize the fuel injection selection according to the requirements of the flight regime and engine operation. The ejector scramjet operates in the four modes:

- Rocket-ejector,
- Ramjet,
- scramjet,
- Rocket-only mode.

The basic property of ordinary ejectors is that they multiply original or primary mass flow by drawing a supplemental or secondary mass flow from the surrounding atmosphere.

Operation:

Rocket-ejector mode: This is an ejector cycle with the rocket acting as the primary or drive-jet. The thrust of the rocket is augmented through a jet pumping process that transfers momentum from the high-velocity rocket exhaust to the inducted air. The ejector process results in an increased total mass flow with a lower exit velocity and yields a higher specific impulse in comparison to the rocket-only operation. The rocket-ejector mode is used from takeoff through low supersonic flight speeds.

Ramjet Mode: As the flight Mach number approaches 3, the engine transitions to ramjet mode which provides a higher specific impulse in the mid-to high-supersonic flight speed range. Oxidizer is supplied by the ram air from the inlet, and combustion takes place at subsonic conditions

Scramjet Mode: Around $M = 6$, the operation of the engine is turns to the scramjet mode, when the flow remains supersonic throughout the entire engine. The engine combustion cross section must remain constant or diverge in this mode to avoid the onset of thermal choking in the scramjet. The rocket is either turned off or used as a fuel injector in both ramjet and scramjet modes.

Rocket-Only Mode: Around $M = 15$ the air density can no longer sustain an efficient air breathing cycle and the engine is switched to the rocket-only operation. The air inlets close and the rocket restarts providing thrust to insert the spacecraft into orbit. Ejectors are mechanically simple, requiring only an enclosing passage, or shroud around the primary flow, long enough to enable complete mixing with the secondary flow.

Ejector ramjets are attractive low speed propulsion candidates because of their mechanical simplicity. They can also be very easily integrated into the existing flow path.

Operating Features:

- The ejector portion of the device will have constant area and fixed geometry.
- The inlet primary flow will be supersonic and inlet secondary flow is subsonic. The ejector exit plane is sonic and choked.
- The ejector ramjet has a potential to increase the thrust above the primary flow alone, with a thrust augmentation to a factor of around 1.6-2.2 in the mach number range for which a ramjet could produce little or no thrust.

Advantages: **Increased Thrust**: The ability to utilize the rocket as an ejector increases the engine mass flow and thrust.

Reduction in Weight and Size: Since Oxidizer amount to be carried on board has reduced, weight of system is reduced. This also decreases the size of the vehicle.

Lower Vehicle Propellant Mass: Vehicle propellant mass fractions for RBCC-powered vehicles are projected to be around 70%, as compared to 90% for all-rocket vehicles.

Higher Specific Impulse due to high By-pass: As the ratio of the bypass air to the rocket exhaust mass flow increases with increasing flight speed, the specific impulse continues to increase as the cycle more closely resembles ramjet operation.

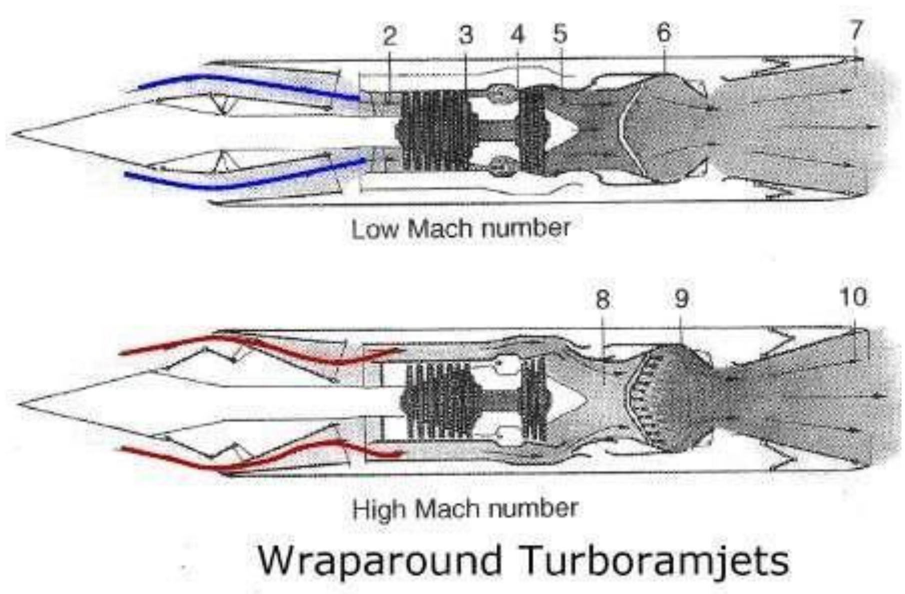
Higher I_{sp} in rocket mode: In the rocket-only mode, the use of the engine duct as a highly expanded nozzle at high altitudes increases the specific impulse of that mode of operation.

Higher T/W ratios: In the rocket–ejector mode, RBCC systems can provide vehicle thrust-to-weight ratios greater than one and are therefore capable of vertical takeoff and landing

Finally, the cryogenic fuel can be used in air-breathing modes as a **heat sink** to increase the density of the inlet airflow, thus increasing the work output.

Facilitates SSTO concept: This concept has been identified as one of the most promising propulsion system for both single-stage-to-orbit (SSTO) and two-stage-to-orbit (TSTO) vehicles.

Air Turbo-Ramjet Engine: It is basically a variable cycle engine, where during the flight itself, it changes from turbojet without afterburner, then turbojet with afterburner and then a ramjet engine.



Only Turbojet

Only Ramjet

Fig: 1.6 ramjet engine

The ramjet engine, can take an aircraft or missile to hypersonic speeds. However, ramjets do not operate at low speeds and hence, cannot take off a craft from zero speed. So, the Air turbo ramjet concept works with the turbojet help take off to some high altitude and a high Mach number, the ramjet would take over and take it to hypersonic speeds. This device is referred to as a wraparound turbo ramjet, where a ramjet is essentially wrapped around a turbojet. So, the outer annulus is essentially ramjet and the inner core is a normal turbojet. The turbojet mode operates up to Mach number around Mach 3, then the ramjet takes over up to Mach 6 or Mach 7.

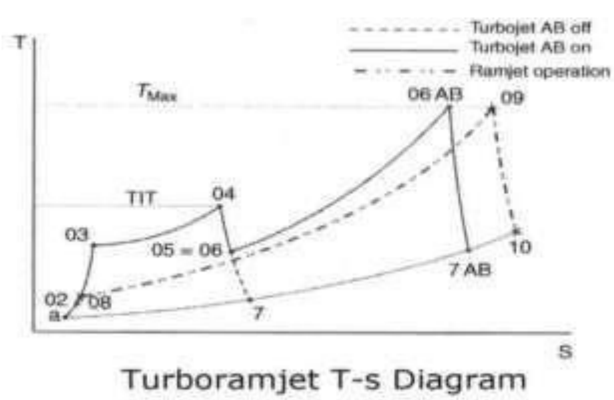


Fig: 1.7 Turboramjet t-s diagram

Air Turbo Ram Jet

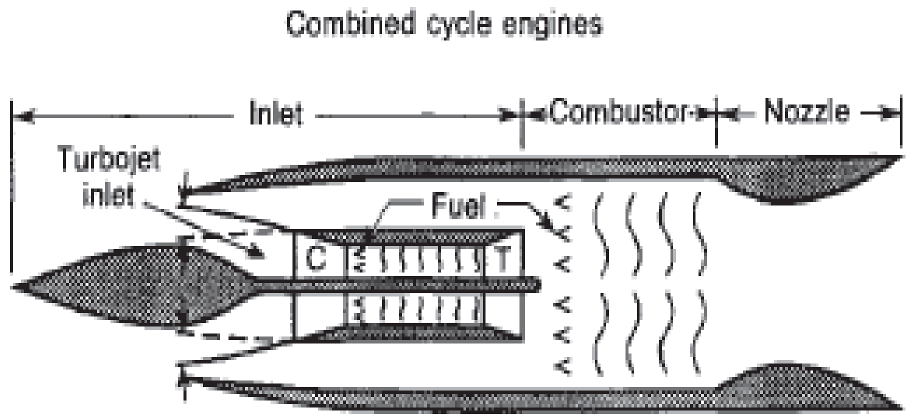


Fig: 1.8 turbo ramjet

The turbo ramjet is a hybrid engine that essentially consists of a turbojet mounted inside a ramjet. The turbojet core is mounted inside a duct that contains a combustion chamber downstream of the turbojet nozzle. The operation of the engine is controlled using bypass flaps located just downstream of the diffuser. During low speed flight, controllable flaps close the bypass duct and direct air flow into the compressor section of the turbojet. During high speed flight, the flaps block the flow into the turbojet, and the engine operates like a ramjet using the AFT combustion chamber to produce thrust. The engine would start out operating as a turbojet during takeoff and while climbing to altitude. Upon reaching high subsonic speed, the portion of the engine downstream of the turbojet would be used as an afterburner to accelerate the plane above the speed of sound.

The turbo-ramjet combustor may use hydrogen and oxygen, carried on the aircraft, as its fuel for the combustor.

Main components of Air Turbo Ramjet:

- An axial flow compressor with modest pressure ratio, commonly known as fan, provides mechanical compression of the core turbojet engine at low supersonic mach numbers. Provision must be made to bypass the air flow at high mach numbers, above 3.0.
- A power turbine driven by high pressure, high temperature gases generated in a separate combustion chamber. This turbine provides the power required by the compressor (fan). The power turbine is independent of free stream flight conditions, irrespective of the altitude of the flight. The turbine mass flow is referred to as primary flow, and it mixes and increases the main free stream air flow.
- Fuel injectors and burner for addition of thermal energy.
- A CD nozzle to complete expansion process.

Turbo ramjet rocket:

A variation of ATR concept is the addition rocket motor to ATR engine. The primary reason for adding the internal rocket engine is to supplement thrust available at both lower and higher Mach number range. The extra rocket to the core engine integrates with the overall configuration, to augment thrust levels to the core turbojet at lower mach numbers and to the ram/scram engine at higher mach numbers. The existing exhaust nozzle is designed to provide the very large area ratios demanded by the combination. A schematic diagram is given above.

Liquid Air Cycle Engine (LACE):

Liquid Air Cycle Engine is a separate class of hypersonic air breathing engine made possible by the availability of very low temperature, cryogenic liquid hydrogen fuel. The liquid hydrogen fuel has high specific energy release (heat of combustion per unit mass), good vehicle cooling capacity and also low boiling point.

The LACE gathers part of its oxidizer from the atmosphere, using liquid hydrogen (LH2) fuel to liquefy the air.

The cooling capacity of the cryogenic liquid hydrogen is used to produce liquid air (LAIR) from the atmosphere so that it can be mechanically compressed and easily and injected together with the now gaseous hydrogen in to the rocket engine, where they chemically react to provide thrust. This is a direct way of obtaining the oxygen from surrounding atmosphere rather than carrying it on board.

The process relies on fact that the temperature of liquid hydrogen is 20.4 K at 1 atm; is considerable less than that of liquid air which is78.9 K at 1 atm. The air contains nitrogen also that adds to the exhaust mass flow rate. Since the engine carries only fuel on board, the performance of LACE will generally be superior to that of pure hydrogen-oxygen rocket engine.

A Basic LACE Engine:

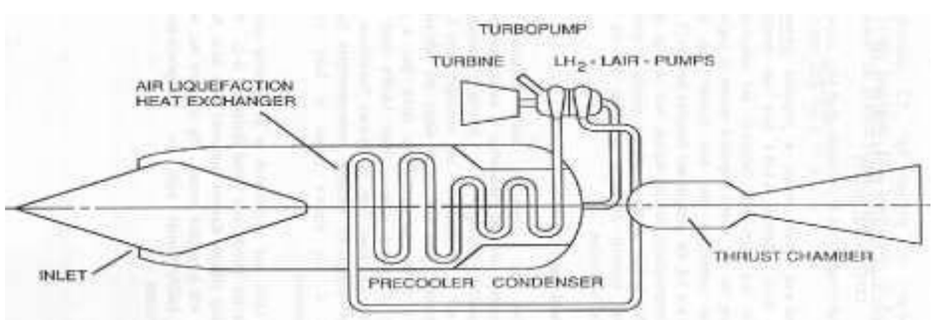


Fig: 1.9 Liquid Air Cycle Engine

Working Principle: LACE works by compressing and then quickly liquefying the air. Compression is achieved through the ram-air effect in an intake similar to that of a high-speed aircraft. The intake ramps create shock waves that compress the air. The air passed over heat exchanger, in which the liquid hydrogen fuel is flowing. This rapidly cools the air, and the various constituents quickly liquefy. By careful mechanical arrangement, other parts of the air, notably water and carbon dioxide are removed from liquid oxygen and nitrogen. The liquid oxygen can then be fed into the engine as usual. The hydrogen is so much lighter than oxygen that the now-warmer hydrogen is often dumped overboard instead of being re-used as fuel, at a net gain.

Advantages:

- The use of a winged launch vehicle allows using lift rather than thrust to overcome gravity, which greatly reduces gravity losses.
- Increases the efficiency of propellant rocket by gathering part of its oxidizer from the atmosphere.
- It lowers the take-off weight of the spacecraft considerably.

Disadvantages:

- LACE system is far heavier than a pure rocket engine having the same thrust. Vehicle will have higher aerodynamic drag and aerodynamic heating. Fuel consumption to offset the drag losses.
- LH2 tanks need heavy/large plumbing and are heavy and expensive. LOX tanks are relatively lightweight and fairly cheap. LOX is quite cheap, but LH2 is more expensive.
- Additional mass of the thermal protection system for the cryogenic fuels.

Fuel injection in Scram Jet Engine:

Design of scramjet combustor must take into account the requirement that the fuel be well mixed with the air within a few microseconds.

Turbulent Mixing begins immediately and combustion quickly follows. However, for efficient combustion, the **chemical reaction time must be fast (small)** compared with the mixing or mechanical time.

Major issues encountered in the scramjet Engine combustion are

- Combustion efficiency in converting chemical energy in to kinetic energy

- Heat transfer at low-pressure conditions in the combustor
- Low residence times in the scramjet

In the design fuel system for supersonic combustion, fuel pre-injection in inlets or isolators holds considerable potential. Pre-injection or distributed injection enhances mixing, flame stability, and combustion efficiency for scramjet engines. The fuel is injected at the inlet parallel to the air flow. Distributed and scheduled fuel injections are adopted in combined cycle engines. During the operation engines in the lower Mach number range, the flow residence times are relatively large, therefore, fuel injection is considered only in the combustion chamber.

However, as the Mach number increases, the flow is supersonic throughout the combustion chamber with very low residence times. Fuel injection must begin in upstream region, including the inlet. The large localized heat release in a given section of combustor, gives rise to shock waves which spread the heat release in the flow direction resulting in an advantage of the diffusive mode of supersonic combustion.

Following factors influence the design of combustors:

- ➢ Avoidance of hot pockets near the walls implies that the fuel be injected from centrally located struts.
- ➢ The air and fuel stream are crossed in a comb like structure with fuel struts, which generates a large interface. Turbulence due to the higher speed of the fuel leads to additional mixing. Complex fuels like kerosene need a long engine to complete combustion.
- ➢ The usual circular configuration for combustors can be sacrificed in favor of a rectangular configuration.
- ➢ Fuel injection and management is also potentially complex. One possibility would be that the fuel be pressurized by a turbo pump, heated by the fuselage, sent through the turbine and accelerated to higher speeds by a nozzle.

- It is proposed to use porous walls for fuel injection as a means both to address wall cooling and to reduce flow friction.

Distributed hydrogen fuel injection is preferred in the scramjet engine to optimize the heat release. This configuration included in-stream struts with fuel injectors that could modulate the heat addition as required by the flight regime.

The Rocket Based Combined Cycle (RBCC) or strut-jet, as it is called, is an ejector scramjet engine. It consists of a rocket subsystem incorporated in an air-breathing engine and an inlet, mixer, combustion chamber, and nozzle. It uses distributed fuel injection system with the fuel-injection sites located at several locations along the duct to optimize the fuel-injection selection according to the requirements of the flight regime and engine operation. The scramjet mode of operation is achieved through thermal choking caused careful tailoring of the fuel-injection system. Fuels used for RBCC systems have to satisfy following requirements;

- High energy density leading to high specific impulse
- Fast chemical kinetics reducing combustion time which is critical for the scramjet mode
- Provide thermal sink for cooling the incoming flow
- Hydrogen fuels are preferred over the conventional hydrocarbon fuels due to above reasons
- A number of synthetic fuels are developed recently, with increased energy output

Efficient mixing is essential for ensuring complete combustion. The inlet length can be used for mixing in case fuel is injected in to the inlet. Distributed fuel injection with integration of inlet fuel injection with combustor is considered in scramjet engines. Inlet fuel injection will also contribute to airflow compression and pre heat the fuel. Further, when liquid fuels are used, pre- combustor fuel injection would lead to secondary breakup of fuel droplets that is due to interactions with the inlet"s shock compression system. This will improve mixing and speed up chemical reaction. Considering the short residence times, direct fuel injection in to the combustor cannot ensure complete combustion.

Distributed fuel injection system offer following benefits:

- Air-fuel inter-action occurs over entire length of inlet-isolator-combustor resulting in better mixing.
- Complete combustion in shorter isolator/combustor lengths, thereby reducing engine weight and cooling loads.
- We can use combination of liquid and gaseous fuels through different sets of injectors
- Upstream fuel injection increases the residence time of fuel/air mixture

Drag In Scramjet Aircraft:

During hypersonic flight, the engine thrust is only slightly larger than the vehicle"s drag; hence efficiency of expansion process and the thrust angle relative to the flight direction become critical for the vehicle"s flight dynamics.

Spillage Drag: Spillage drag, as the name implies, occurs when an inlet "spills" air around the outside instead of conducting the air to the internal intake. The airflow mismatch produces spillage drag on the aircraft.

The inlet is usually sized to pass the maximum airflow that the engine can ever demand and, for all other conditions, the inlet spills the difference between the actual engine airflow and the maximum air demanded. Mixed compression inlets slow down the flow through both external and internal shock waves They spill air while operating at off design conditions.The minimization of external drag is an important aspect of the inlet design process.

Aerodynamic effect of Exhaust Plumes: The effect of exhaust plumes on the aerodynamic characteristics of the vehicle is usually to decrease the vehicle drag at supersonic speeds and to increase it at subsonic speeds. At supersonic speeds and above, there is often a turbulent wake area with a low local pressure at the aft end. With the action of plume, the pressure on the aft portion of the body is increased. This increases the pressure thrust and thus reduces the base drag.

Plume Drag: The plume that exits the backend of the jet engine, or a rocket, indirectly creates drag, which we call plume drag. The boundary layer around the vehicle can interact with the plume, creating a drag that tries to split the boundary layer from the vehicle. Because a substantial part of the vehicle is dedicated to nozzle expansion, considerable lift and pitch moments are produced by the pressure distribution on this part of the after body, complicating the nozzle design and vehicle integration.

Effect of nozzle exit pressure- Scramjet engines rely on external expansion of the fluid. Thus, fluid leaving the scramjet internal expansion nozzle will be highly under-expanded. The nozzle exit pressure ratio –(defined as the ratio of static pressure at the nozzle exit plane to the free stream static pressure) - is an important determinant of the after-body forces. It influences the shape and force of the footprint the exhaust plume makes as it impinges on the after-body, thereby contributing to the overall lift, thrust, and moment. Upon leaving the nozzle, the flow “expands” by turning toward the region of lower pressure. The adjacent supersonic external flow is forced to turn also, and an oblique shock wave (the plume shock) emanates from the nozzle lip. The engine exhaust plume impinges on the underbody and causes increments in lift, drag, and pitching-moments.

Viscous Drag & Pressure Drag: (ISOLATOR DRAG LOSSES)-The main sources of losses in the isolator are caused by the pressure drag and the viscous drag. At hypersonic speeds, relative heat addition to the air progressively decreases with increased flight velocity whereas the drag losses continuously increase until the heat addition can no longer overcome the drag and the air-breathing-based system reaches the extent of its flight envelope.

The performance of a scramjet engine when integrated on a hypersonic vehicle is determined by stream thrust analysis. This technique determines the propulsive forces on the vehicle. A schematic of a control volume that surrounds a hypersonic vehicle powered by a scramjet engine is shown below.

Airflow enters the control volume at the flight conditions, fuel is added to the air in the combustor and the flow exits through the vehicle nozzle. For ease of analysis, the flow exiting the control volume is represented by a one dimensional average flux of the exhaust plume. The spillage drag and plume drag have been combined into a single force called the additive drag.

Air spillage (and therefore spillage drag) decreases as the vehicle speed approaches the design point of the engine, and the plume drag varies depending on the amount of under-expansion in the nozzle. Both these forces are usually estimated through CFD analysis or through rules-of- thumb based on empirical or experimental databases.

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DEPARTMENT OF AEROUTICAL ENGINEERIN

Unit-2-Aerospace Propulsion-SAEA1502

Rocket Engine

Chemical Rocket:

Classification of Rocket Engine Propulsion Systems:

Allclassical propulsivesystems dependonconservation of momentum. Thisprincipleisusedin many ways tocreate thrust. Majority ofsystems expel mass.

Rockets can be classified based on how they are accelerated.

Either the energy comes from the propellant itself (internal energy) as in chemical reaction, or they can be accelerated using external energy source.

Some propulsion systems donot need evenany propellant at all. The propellant also need not be carried with thespacecraft.

Theperformance of the propulsive system depends on total mass of thespacecraft and onthe speed of the propellant.

Hence, all propulsion systems whichreducetheneed ofpropellant making thespacecraft lighter are considered advanced systems.

Classification of Propulsion System:

- **Type of Energy Source (Chemical, Nuclear, Solar, Electric etc)**
- **Basic Function of the vehicle (Booster/Sustainer Stage, Attitude Control, Orbit/Station Keeping etc)**
- **Type of Vehicle (Aircraft, Launch Vehicle, Spacecraft, Missile, Assisted take-off etc)**
- **Size (Sounding Rocket, Multi stage Rocket etc)**
- **Type of Propellant (Chemical, Electric, Nuclear etc)**
- **Source of Energy (Internal, External)**

Classification based on source of energy:

- Internal Energy: Chemical (solidpropellant, liquid propellant, gaseous propellant, hybrid propellant); Nuclear (fission/Fission/antimatter); Magneto Hydrodynamic Propulsion (MHD) ,Propellant-less(proton/nuclear)
- External Energy: Electric, Propellant-less(solar sail/laser), Catapults
- External/InternalEnergy:Nuclear,Airbreathingpropellant-less(tethers),Breakthrough propulsion

Classification based on propulsion system:

Rocket Engines are classified based on the Propulsion system they use. They are:

- Chemical Rocket Propulsion Systems
- Nuclear Rocket PropulsionSystems
- Electric Rocket Propulsion Systems
- Propellant-less Rocket Systems
- Break-through propulsion Systems

Chemical Rocket Engine-Propulsion: Rocket engine produces high pressure combustion gases generated by combustion reactionofpropellant chemicals usually fueland an oxidizing chemical. The reactionproductgasesareatveryhightemperatures(2500to4100°C).Thesegasesare subsequently expandedinanozzleandacceleratedtohighvelocities (100to 4300 m/sec). Since thesesgas temperatures areabouttwicethemelting pointofsteel,itisnecessarytocool or insulate all the surfaces that are exposed to hot gases.

EngineTypes:Accordingtodifferentphysicalstateofthepropellants,thetypesofchemical rocket propulsion devices is as follows:

1. **Liquid Propellant Rocket Engines:** Useliquidoxidizerandfuel,whicharefedunder pressure from tanks into a thrust chamber.

- **LiquidBi-propellantengine** usesliquidoxidizerandaliquidfuel(eg:liquidoxygenand kerosene)
- **Liquid Monopropellant** uses a single liquid that contains both oxidizing and fuel species; which decomposes into hot gas during combustion.

Theliquid propellant rockets are also classified based ontype of feed system used. They can be **turbo-pump fed liquid propellant systems** or **gas pressure fed systems**.
Pressure fed systems are usually for low thrust applications (like attitude control of flying vehicles etc), whilepumpfed systems areusedinhighthrust applications suchasspacelaunchvehicles etc.

2. **Solid Propellant Rocket Engine**:Thesolidpropellantrocketenginesburnagrainofsolid propellant within the combustion chamber or case. The solid propellant charge, called grain containsallchemicalelementsincludingoxidizerandfuelforcompleteburning. Theresulting hotgasesexpandthroughasupersonicnozzle andimpart thrust. There are no feed systems or valves.

3. **Gaseous Propellant Rocket Engines**: They use stored high pressure gas such as hydrozen, heliumetcaspropellant.Theseareusuallycoldgasenginesystemsusedforattitude control systemsforspacevehicles. Heatingofthegas byelectrical energyor by

combustion of amonopropellant improves theperformance andsuchsystems are called “**warm gas propellant rocket systems**”.

4. **Hybrid Propellant Rocket Systems**: Uses both liquid and solid propellant. It can use a liquid oxidizing propellant injected into a combustion chamber filled with a solid fuel propellant grain.

5. **Combination of Ducted & Rocket Propulsion Systems**:Aductedrocket orair-augmented rocket combines principles of rocket and ramjet engines. Theducted propulsion system provides better performance (specific impulse) thanthe chemical rocket, while operating within the earth’satmosphere.

Working Principle:

Liquid Propellant Rocket System:Liquidpropellants areusedinthis system,whichare fedinto the combustion chamber under pressure. The liquid oxidizer andliquidfuel are stored inseparate tanks.

A highpressuregas pressuretank provides pressure feedofoxidizer andfuelthrough diaphragms. Alternatively, separate pumpsmaybeusedtoprovide pressure feeding of propellants. Thepropellantsreactinthethrustchamberandgeneratehotcombustiongaseswhichare expandedin thesupersonicconvergentdivergentnozzle. The system permitsrepetitiveuse and canbestartedand shutoff,asrequired.Itispossibletooperatetherocketforlongdurations, exceeding 1hour by providingadequatecoolingofthethrustchamberandC-Dnozzle.

Aliquidpropellantrocket propulsion systemrequiresseveral precision valves, complexfeed mechanism including pumpsetc.

A schematic diagram is as follows:

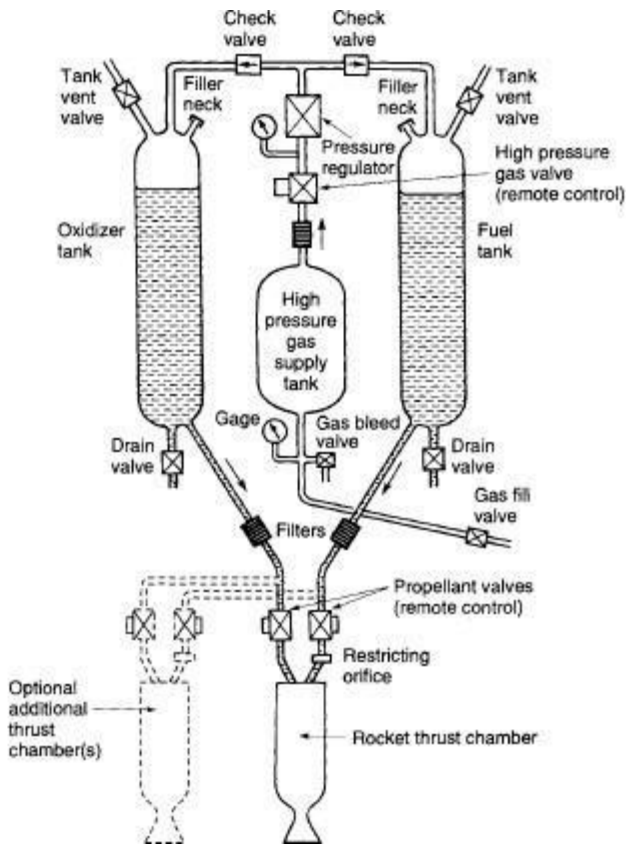


Fig: 2.1 **Solid Propellant Rocket Propulsion System**: A schematic diagram is shown below:

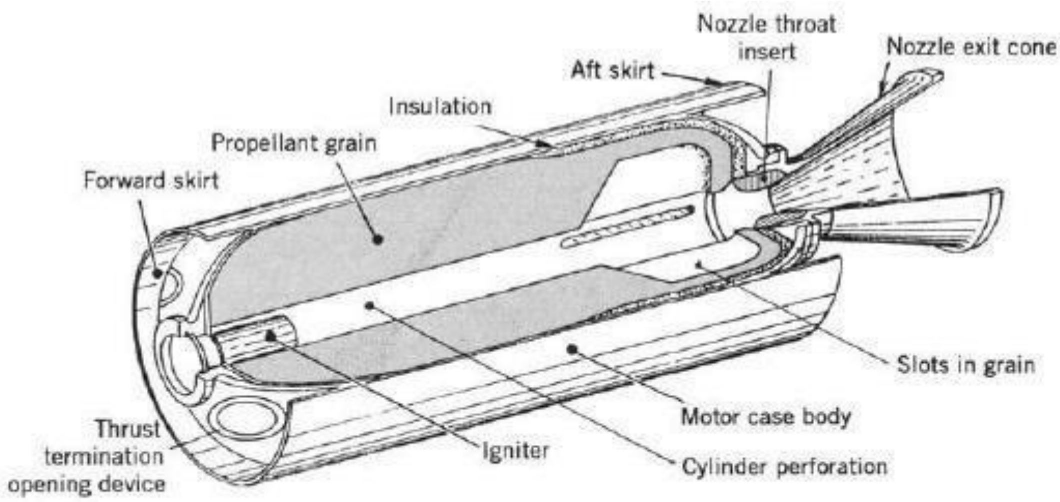


Fig: 2.2 solidpropellant

Thesolidpropellantiscontained in thecombustion chamberorcase. Thesolid propellant charge is called the grain and contains all chemical elements required for complete burning. An igniter is neededto initiatetheburningprocess.Onceignited,theburningproceedsatapredetermined rateonallexposed internalsurfacesofthegrain,tillthecompletepropellantisconsumed.Slots are providedinthe grain structure basedonvariationofburning rate. Theresultinghot combustion gases areexpandedthrough asupersonicconvergent divergentnozzletoprovidethe thrust.

There are no feed systems or valves in the solid propellant rocket motor.

Hybrid Rocket Motors:A schematic diagram is given below:

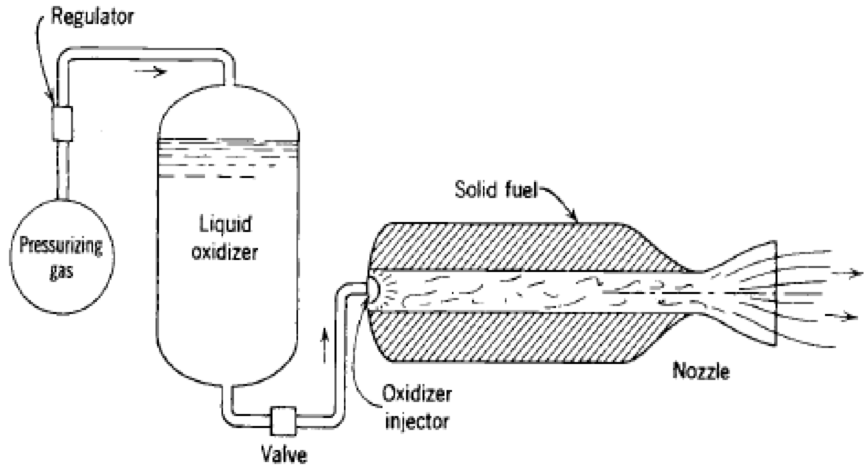
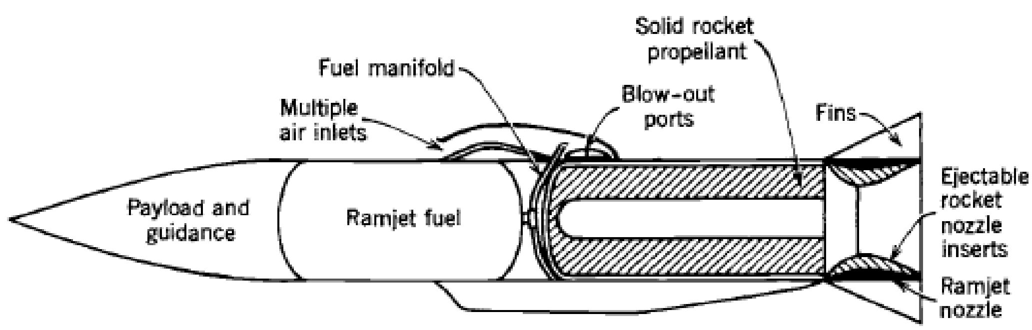


Fig: 2.2 Hybrid Rocket Motors

Hybrid rocket propulsion systems use both solid and liquid propellants. Inthe above diagram, a liquid oxidizer is held in tank, andis injected, under pressure, into the combustion chamber filled with solid propellant fuel. Thehot combustion gases areexpandedinthesupersonic convergent divergent nozzle.

Combination of Ducted & Rocket-Propulsion System: A schematic diagram is as follows:
Hybrid rocket propulsion systems use bothsolid andliquid propellants. Inthe above diagram, a liquid oxidizer is held in tank, and is injected, under pressure, into the combustion chamber filled with solid propellant fuel. Thehot combustion gases areexpandedinthe supersonic convergent divergent nozzle



The principles of rocket and ramjet can be combined sothat the two propulsion systems can operate in sequence, yet utilize acommon combustionchamber.
Initially the system operates in rocket mode, and as thesolid propellant combustion completes, theair inlettothecombustionchamberopens,forramjetoperation.Ramjetfueltanksupplies

the fuel, and the nozzle throat enlarges to accommodate enhanced ram air flow/combustion products.

Applications: Basic application of Rocket propulsion systems are

- Space Launch Vehicles
- Spacecraft
- Missiles
- Other applications

1. **Space Launch Vehicle**:SpaceLaunchVehiclesorSpaceboostersareusedtoplace spacecraft from earth intoouterspacedepending onthemission. Spacelaunchvehicles are usually multistagevehicles usingchemicalrocketpropulsionsystems.

Depending on the mission, they can be classified based on

- Number of stages (single-stage, two-stage, multistage)
- Type of propellant used (Solid, liquid, hybrid)
- Usage (expendable one time use, recoverable/re-usable)
- Size/mass of payload (manned/un-manned, military/civilian use)
- Specific space objective (earth orbit/moon-landing/inter-planetary/inter-stellar)

Solid Propellant motors orused for initial stages whereas liquid propellants are used higher stages. Gaseous propellants are usedforrocket control applications.

2. **Spacecraft**:

Depending on the mission, spacecraft can be classified as

- Earth satellites/inter-planetary satellites
- Manned/unmanned spacecraft
- Inter-stellar missions

Majority of spacecraft use liquid propellants engines, with solid propellant boosters. Electricpropulsion systems areused bothforprimary and secondary propulsion missions onlong duration space flights, inter-planetary/inter-stellar missions.

3. **Missiles**:

Missiles can be classified based on

- **Range: Strategic**(Long range ballistic missiles); **Tactical** (short rangetargets as local support to groundforces)
- **Launch Platform: Ground/surface launched; ocean/ship launched; Underneath Sea (submarine) launched**
- **Type of propellant used: Solid/Liquid** or**Combined Cycle EnginesType of Usage: Surface-to-air; Air-to surface; Air-to-air etc**

4. **Other applications:** Otherapplications are thesecondary applications whichinclude

- Attitude control
- Stage separation
- Orbital changes
- Spin control
- Settling of liquids in tanks
- Target drones
- Underwater rockets liketorpedoes
- Research Rockets

Advantages/Disadvantages of Chemical Rockets:

Solid Propellant Rockets:

Advantages:

1. Simple to design-Few or no moving parts
2. Easy to operate-Little preflight checkout
3. Ready to operate at short notice
4. Propellant will not leak, spill or slosh
5. Less overall weight for given impulse application
6. Can be stored for 5 to 25 years
7. Higheroverall density of propellant leading to compact size
8. Some propellants have non-toxic, cleanexhaust gases
9. Grain design allows use of several nozzles
10. Thrust termination devices allow control over total impulse
11. Can provideTVC
12. Some tactical rocket motors can be produced in large quantities
13. Rocketmotorscan be designedforrecovery, re-use(spaceshuttlerocket motor)
14. Can be throttled, or stoppedand re-started few times, if pre-programmed

Disadvantages

1. Explosion and fire potential is larger
2. Mostrocketmotorscannotwithstandbullet impactor beingdropped on hard surface
3. Rocketsneedenvironmental clearance andsafety features fortransport on public conveyances
4. Some propellants are very sensitive and can detonate
5. Graindamageoccursthroughtemperaturecyclingorroughhandling-limitingusefullife
6. Requires an ignition system Plumes cause more radio-active attenuation than LPRsExhaust gases are toxicincase of compositepropellants with ammonium perchlorate
7. Some propellants can deteriorate (self-decompose) during storage
8. Only somemotors can be stopped,but motor becomes disabled
9. Once ignited, difficult to change pre-determined thrust levels

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Unit-3-Aerospace Propulsion-SAEA1502

- **Liquid Propellant Rockets**

Advantages:

1. Provideshigherimpulseforgivenpropellantdensity; increases attainable vehiclevelocity increment and missionvelocity
2. Can be randomly throttles and stopped and restarted
3. Provides forpulsed (repetitive) operation. Somesmall thrust rockets allow over 250,000 times usage.
4. Bettercontrolovermissionterminalvelocity, withprecisethrusttermination devices
5. Canbelargelycheckedpriortooperationiecanbetestedforfullthrustoperationon 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. Liquidpropellants are storable in the vehicle for morethan 20 years andengine can be ready for usequickly
11. Propellant feedsystem can be designedtofeedmultiplethrust chambers
12. Plume radiation and smoke are usually low
13. Propellant tanks can be located such that vehicle stability is high

Disadvantages:

1. Relatively complexdesignwithmorecomponents. 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 feedsystem
5. Cryogenicpropellantscannotbestored forlongperiods. Storagetanksneedspecial insulation
6. Needseparate ignitionsystem (except for hypergolic propellants)
7. More overall weight for short duration, low total impulse application
8. More difficult to control combustion instability
9. Afewpropellants like RFNA (red fumingnitric acid) givetoxicvapors 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
13. Smoky exhaust plume can occur with hydrocarbon fuels

Criteria Used for Selecting of Rocket Propulsion System:

1. **Mission Definition**:Thepurposeandfinalobjectiveofthesystemwilldecidethepayload,flight regime and the type of vehicle propulsion system
2. **Affordability(cost):** ThecostofR&D,production,operation,facilitycostmustbewithin budgetary guidelines.
3. **System Performance**:Thepropulsionsystemshouldbedesignedtooptimizetheperformance.
4. **Survivability (Safety)**: All hazards must be known in advance. In case any failure, the damage to personnel, equipment, facilities and environmentmust be minimum.
5. **Reliability**: Technical risks, manufacturingrisksandfailurerisksmustbelow. Complexsystems must be avoided as much as possible.
6. **Controllability**: Thrust build up and decay must be within specified limits. Responses to control and command signals must be within acceptable limits.
7. **Maintainability:** Easy to follow maintenance procedures and quick fault diagnosis capability will keep the downtimeminimum.
8. **Geometric Constraints**: Propulsion system should fit in to the vehicle within available length and diameter. It is preferable to have a propulsionsystem with smallest volume and highest average density.
9. **Prior Related Experience**: Favorable history and relevant data of similar propulsion systems must be available.
10. **Operability**: Should be easy to operate with operating manuals available.
11. **Producibility**: Easy to manufacture, inspect and assemble
12. **Schedule**: The propulsionsystem should be capable of completing the mission in given time frame.

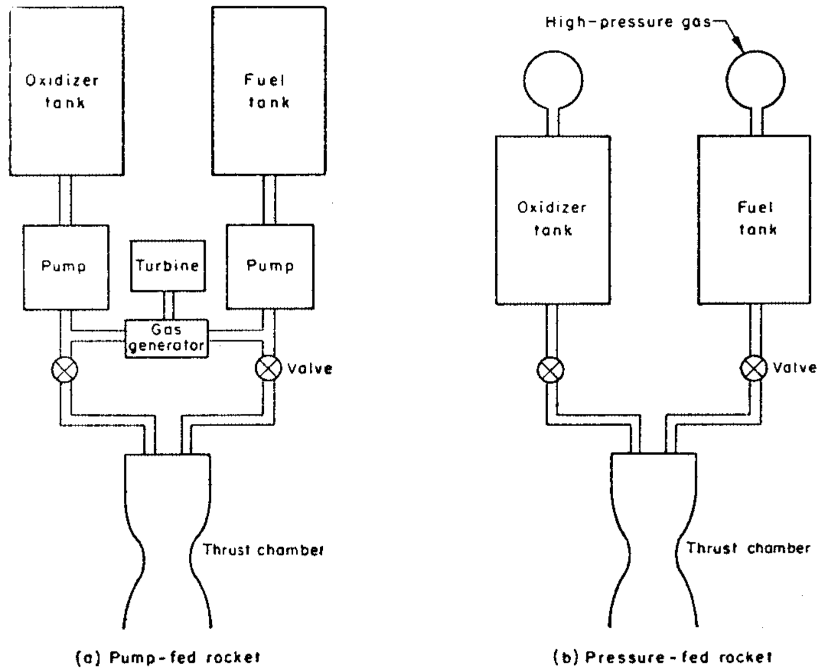

FIG. 2. Schematic of liquid-propellant rocket

Solid Propellant Rocket Motor:

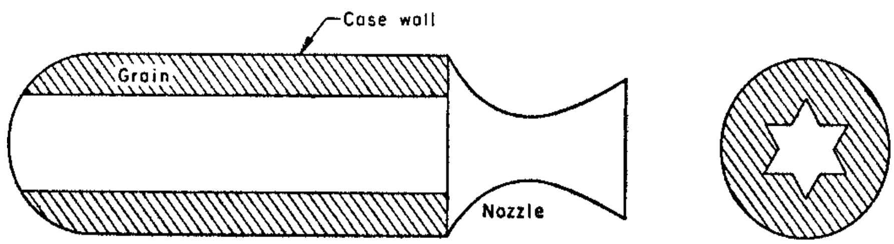

Fig: 3.2 Solid Propellant Rocket Motor

Performance:

1. **Total Impulse:** The total impulse is the thrust force F integrated over the burning time t.

2. **Specific Impulse**:Thespecificimpulseisthetotalimpulseperunitweightofpropellant.It is an important figure of merit of performance ofthe rocket system.
   For constant thrust andpropellant flow, specific impulse is

Theperformanceofrocketisdeterminedlargelybytherocket-propellantcombination andthe totalamountofusablepropellant.Theperformanceofpropellantsischaracterizedby the specific impulse, a measure of thrust producedper unit of propellant consumed per second. The unit of specific impulse is sec.

Thevelocitythatcanbeachievedbyarocketisdirectlyproportionaltothespecificimpulseofits propellants.

3. **Effective Exhaust Velocityc:** Inarocketnozzle,theactualexhaustvelocityisnotuniform over theexitcrosssection.Forconvenience,auniformexitvelocityisassumedwhich allows a one-dimensional description of the flow.

   Theeffective exhaust velocity c is the average equivalent velocity at which propellantis ejected from the vehicle. It is defined as

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

4. **Mass Ratio MR**: Themassratioofa vehicleisdefinedtobe thefinal mass (afterthe rocket has consumed all usable propellant) divided by mass (before rocket operation).

   - Thefinalmass isthemassofthevehicleaftertherockethasceasedtooperate when all theuseful propellant mass is consumed and ejected.
   - Thefinal mass includesmass ofguidancedevices, navigationalgear, payload, flight control system, vehicle structure tanks, control surfaces, communication equipment and unusable propellantetc
   - Value of MRranges betweenaround 10 % for large vehicles toaround 60 % for tactical missiles

5. **Propellant mass fraction ζ**:Thepropellantmass fraction ζindicates thefractionof propellantmass inaninitialmass .Itcanbeappliedtothevehicleorastage.

6. **The Impulse-to-weight Ratio:** Theimpulsetoweightratioofthepropulsionsystemis definedas the total impulse divided by the initial vehicle weight

Solid Propellant Types:

Three types solid propellants are in use:

- Double-Base
- Composite
- Composite modifieddouble-base

Double-Base : Consists of nitrocellulose and nitroglycerine plus additives in small quantity. It is a homogeneous mixture of two explosives (usually nitro-glycerine**i**n nitrocellulose). DB propellants are used in smaller rocket motors.

Composite:
Composite propellants are heterogeneous(physical) mixture of powderedmetal (fuel), crystalline oxidizer and polymerbinder.

Theoxidizer is usually ammonium nitrate, potassium chlorate or ammonium chlorate. The fuels used are oftenhydrocarbons suchas asphaltic-type compounds or plastics.

Composite-modified double-base:
Combination ofcomposite and double-base ingredients.

Liquid Propellant-Types: The liquid propellant contains

- Oxidizer (liquid oxygen, nitric acid etc)
- Fuel (gasoline, alcohol, liquid hydrogenetc)
- A suitable gellingagent.

A**bi-propellantrocket**unithastwoseparateliquidpropellants,anoxidizerandafuel.Theyare stored separately and are not mixed outside the combustion chamber. Majority of liquid propellant rockets usebi-propellants.

A **mono-propellant rocket**contains the oxidizing agent and combustible matter in a single substance. It may be mixture of several compounds or ahomogenous material such as hydrogen peroxide or hydrazine. Mono-propellants are stable at atmospheric condition, but decompose and yield hot combustion gases when heated.

A **cryogenic propellant rocket** stores liquid propellant at very low temperature. Cryogenic propellantisaliquefiedgasatverytemperature,suchasliquidoxygen(at-183°C)orliquid hydrogen(at -253° C). Provision forventingthe storagetank andminimizingthevaporizingloses are essential for this type of rockets.

Storable propellants (eg. Nitric acid or gasoline) are liquids at ambient temperatures and can be storedin sealedtanksforlongperiods.**Spacestorablepropellants** (eg.Ammonia)areliquidsat space environment. Their storage tanksneedspecific design, specific thermal conditions and pressure.

Some commonly used Chemical Propellants & their I_{sp}(sec) :

1. **Solid Double Base (NC + NG): 170-250**
2. **Boron metal and Oxidant : 200 to 250**
3. **Aluminum metal and Oxidant : 200 to 250**
4. **Ammonium perchlorate with nitro-polymer : 210 to 270**
5. **Lox-Hydrazine-liquid Bi-propellant : 300 to 385**
6. **Lox-Alcohol/JP4 : 250 to 270**
7. **Hydrazine-Chlorine Triflouride: 250 to 300**
8. **Hydrazine liquid monopropellant: 160 to 190**
9. **Hydrogen Peroxide-liquid monopropellant: 160 to 190**

Chemical Propellants:

Any substance which is used for propelling vehicle is a propellant. It could be a plasma, or charged particle oranysubstancethat releases storedenergy. Evencoldgas at highpressureisa propellant.

Requirements of a propellant:Inarocketvehicle, theaim istogetahigh, whichinturnneed high chamber temperature and pressure. We also need low value of molecular mass of gasses.

Molecular Mass: A chemical propellants generate high through chemical reactions ie heat release per unit mass must be large.

The atomic mass of the elements that constitute the chemical propellant must be small. The product gasses also will then be low, so that will be low.

The atomic mass ofhydrogenis 1(molecularmass is 2); helium is 4; carbon is 12, aluminum is 27 etc. Oxygen is a powerful oxidizerwith an atomic mass of 16.

From molecular mass point of view, we will not prefer any chemical element beyond Chlorine, with atomic mass of 35.

Specific Heat Ratio γ: Morecomplex molecules havelowervalue of, because the degreesof freedom are more. Therefore, we need combustion gasproductstobemore complex forlow value of complex gases will have more specific heat.

Therefore the requirements of low specific heat and low specific heat ratio are contradicting each other.

Sincethe effect of specificheatratio onthe jet velocity is lesspronouncedthan that of the specific heat, we prefer propellants with combustion products having lower value ofspecific heat at constant pressure.

Velocity increment needed for launch:

There is a distinction between velocity increment and the actual velocity of the vehicle.

Thevelocity increment is the velocity calculated from therocket equation, andis a measure of the energy expended by the rocket.

Thevehiclevelocityislessthanthis,becauseofgravityloss,andtheenergyneededtoreach orbital altitude. The difference representsthe energy expended against gravityloss andpotential energy.

Deductions fromTsiolkovsky's Rocket equation:

- Ratioofinitialandcurrent massofvehicle defines current velocity.
- Itis applicabletoanyvelocityincrement,wheninitialand finalmasses areknown.

- Theexhaustvelocity isassumedtobeconstant,andisvalidformostofthereal cases.
- Thevelocityofrocketvehicle,atanyinstantofburntime,isdependentonlyon exhaust velocity and the instantaneous mass ratio.
- Use of multi-stage vehiclesenableachievinghighervelocityincrementfordeepspace missions

Combustion Instability: Combustioninstabilityreferstofluctuationsinpressurebuildupin combustionchamberwhichmayturnsevereifthecombustionprocessisnotcontrolled. Unstable combustion, orcombustion instability resultsinoscillations occurring at regular intervals, which may increase or may die out.

Principletypes of combustioninstabilities are

- Low frequency instabilities, or chugging
- Intermediate frequency instabilities or acoustic bugging
- High frequency instability or screeching

Combustioninstabilitycanbeavoidedbyavoidingresonanceinthefeedsystems,providing acostic damping(absorbers byproviding small cavities) in thechamberand provision of injector baffleschanges in the injector geometry etc.

Reaction Control Systems (RCS):

Propulsion systems are of two types:

1. **Primary Propulsion function** –Highthrustengines,usedforlaunchalong flightpath,for orbit injection, interplanetary missions etc). Some missions require fourto six rocket units whilemore complexmannedspacecraft need 40 to 80 rocketunitsinall stages.
2. **Secondary Propulsion function** - Low thrust applications like attitude control, spin control, momentumwheel,stageseparationor forsettling offluids. Thesmallthrust rockets must givepulses of small bursts ofthrust, necessitating thousands ofrestarts. The smallattitude controlrocketsmustgivepulsesorshortburstsofthrust,necessitating thousands of restarts.

Higherthrustenginesareusedforprimarypropulsionsystemsforinter-planetaryspacecraft.

Majorityofspacecraftusedliquidpropellantengines,withsolidpropellantboosters.Several spacecraft have operated successfully with electrical propulsion for attitude control. Electrical propulsion systems willprobably be used forprimary and secondary propulsion missions onlong duration spaceflights.

AreactionControl System(RCS)is neededtoprovidetrajectory corrections(small Δvcorrections) and correctingrotationalorattitudepositionofthespacecraftandlaunchvehicles. RCSisalso called auxilliary rocket propulsion system.

Ifonly rotational maneuvers aremade, RCSistermedas attitude control system. Maneuvers

Conducted byRCS:

1. **Velocity Vector Adjustment and Minor In-flight Correction Maneuvers**: These are performed with low thrust, short duration and intermittent (pulsing) operations. RCS usesmultiplesmallliquidpropellantthrusters,bothfortranslationandrotation. The **reaction control rocket systems** in a space launch vehicle will allow accurate orbit injectionafteritisplacedintheorbitbyanotherlessaccuratepropulsionsystem. The **vernier rockets**placedonaballisticmissileareusedtoaccuratelycalibratetheterminal velocityvector for improvedtarget accuracy. Mid-courseguidance correctionmaneuvers also fall inthis category. Propulsion systems fororbit maintenance maneuversarecalled station keeping maneuvers. They keepthe spacecraft in the intended orbit overcoming the perturbingforces.

2. **Rendezvous and Docking Maneuvers**: Therelativepositions ofthe launch planet andthe target planetarecriticalfor planetary transfermission. Thespacecrafthas tomeetor **rendezvous** withthetarget planet whenit arrives atthetarget orbit. There isaspecific **time-window** for alaunch of aspacecraft that will make asuccessful rendezvous. Docking is the linking up of two spacecraft and requires a gradual gentle approach (low thrust, pulsing mode thrusters) soas not to damage the spacecraft. These maneuvers involve both rotationalandtranslationalmaneuversconductedbysmallreactioncontrolthrusters.

3. **Simplerotationalmaneuvers:** Thesemaneuversrotatethespacecraftoncommandinto specificangularpositionsoastoorientorpointatelescope,instrument,solarpanel or antenna forthepurposeofnavigation,communicationorsolarpowerreception.Sucha maneuverisalso intendedtokeeptheorientationofthesatelliteinaspecificdirection,for example if the antenna needs to be continuously pointed towards the center of earth. Then thesatelliteneedstobe rotatedarounditsownaxisonceeverysatelliterevolution.These maneuvers can also provide flight stability or correcting angular oscillations. If the rotation needstobeperformedquickly, thenachemicalmulti-thrusterreactioncontrolsystemis used.

4. **Changeof Planeof Flight Trajectory:** Thismaneuverrequiresapplicationofthrustforceina directionnormaltotheoriginalplane.Thismaneuverisperformedbyapropulsionsystem that has been rotated by the RCS in to proper orientation.

5. **Transferoforbits:** Sometimes,transferorbitscanbeachievedwithverylowthrustlevels (0.001 to 1 N), using electric propulsion systems

AnRCScanbeincorporatedinthepayloadstageandeachofthemulti-stagevehicle.RCS operates throughout the flight and provides control torques and forces. Liquid propellant rocket engines withmultiplethrusters are most commonly used. Electric propulsion systems areused on spacecraft.

LifeofRCSmaybeshortoritmaybeusedthroughoutthemissionduration(maybemorethan 10 years) as part of the orbiting spacecraft.

The vehicle attitude has to be controlled about three mutually perpendicular axes, each with two degrees of freedom (clockwise and anti clockwise)

Inordertoapplytorque,itisnecessarytousetwothrustchamberswithequalandopposite start/stop times. Thereis aminimum of 12 thrusters inatorque control system.

An RCS usually has the following major subsystems

- **Sensing devices** for determining attitude velocity and position of the vehicle
- A **control-command system** that compares actual position with desired position and issues commands for corrections
- **Devices for changing angular positions**, such as gyroscopic wheels
- **Attitude control thrust providers**

RCSsystemsarecharacterized bythe magnitudeofforce, quantity(number) anddutycycles. A dutycycle referstonumber of thrustpulses, operating time and timebetweenpulses

RCS systems can be mass expulsion types (rockets) or non-mass expulsion types. Reaction wheels or flywheels andmomentum storage devices are examples fornon-mass expulsion types. The vehicle angularmomentumcanbechangedbyacceleratingordeceleratingareactionwheel.

Thepropellants for RCS fall intothree categories-cold gas jets (also called inertgasjets), warmor heated gas jets and chemical combustion rockets. Hydrazine is most commonly used monopropellant in RCS.

Nozzles:

The function of nozzle is

- Toconvert high-pressure, high-temperature energy(enthalpy) tokineticenergy. Thrust force is derived from this conversion process.
- To straighten the flow so that it exists in axial direction

Because ofthehightemperaturesthat thenozzleexperiences, materials usedin nozzle construction are usually nickel-based alloys, titanium alloys or ceramic composites.

Under-expanded Nozzles:

If the nozzle exit pressure is greaterthan ambient pressure, the flow is considered as **under- expanded**.

- An**under-expanded nozzle**dischargesthefluidatanexitpressuregreaterthanthe external pressure
- The exit area is too small for optimum expansion.
- The expansion isincomplete.
- This condition occurs at altitudes higher than the design altitude.

Over-Expanded Nozzle:Iftheexitpressureislowerthantheambientpressure,theflowis considered to be**over-expanded**.

- Inan**over-expanded nozzle**,thefluidattainsalowerexitpressurethantheatmospheric pressure.
- The exit area at this condition is too large than the optimum area.
- This condition occurs when the nozzle operates at altitudes lowerthanthe design altitude.
- Sincethepressureinside thenozzleislowerthantheoutside pressure, there is possibility of flow separation due to adverse pressure gradient.

Governingequationsofflow:Foranalyzingtheflowthroughnozzle,theflowisassumedtobe friction-lessandadiabatic,andtheexitpressureisassumedequaltotheambientpressure.

Different possible **flow conditions** are explained with reference to the diagram below:

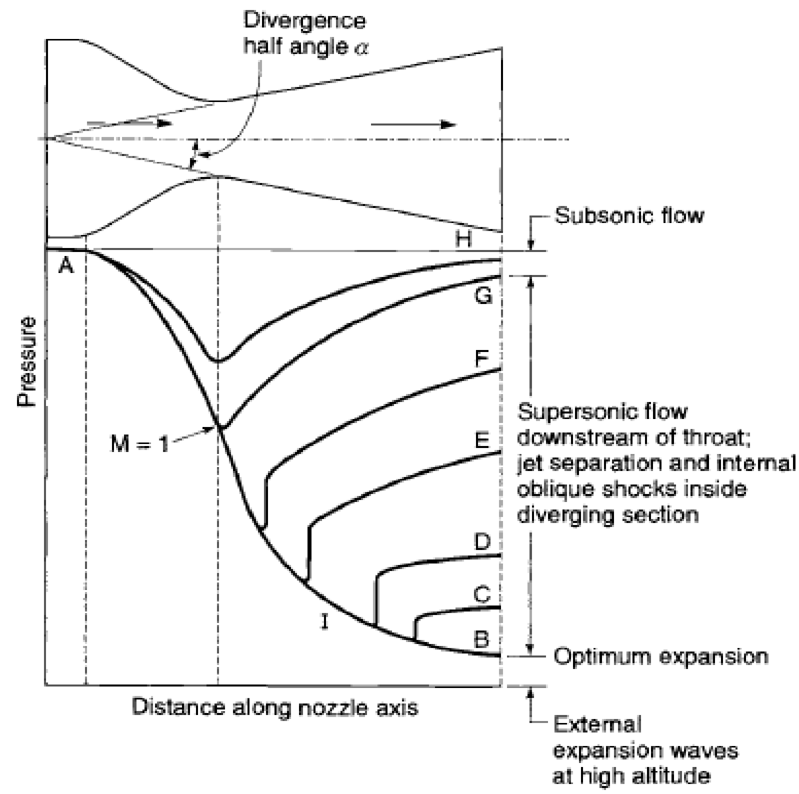

FIGURE 3–9. Distribution of pressures in a converging–diverging nozzle for different flow conditions. Inlet pressure is the same, but exit pressure changes. Based on experimental data from A. Stodala.

- Curve ABshowsvariation of pressure withoptimum backpressure at the design area ratio. (with M = 1 at throat)
- CurvesACandADshowvariationofpressurealongtheaxisforincreasinglyhigher external pressure (over-expansion). At point I, on curve AD, thepressure is lowerthanthe exit pressure andasuddenrise inpressuretakesplaceaccompanied by separation of flow fromthewalls. (Conditionwhenaircraftflies atialtitude lowerthan designaltitude)
- Thesuddenpressure riseinthe curve ADisacompression discontinuityaccompanied by a compression wave.
- Expansion wavesoccurincases whereexternal pressureislowerthantheexit pressure, ie below pointB.(Under-expansion–conditionwhenaircraftfliesatanaltitudehigherthan design altitude)

LossesinNozzle: Inactualcase,theflowisnon-isentropic.Theentropyincreasesduelosses caused by **friction in the boundary layer, flow turbulence, secondary flows due to 3-D flows, shocks**

Types of Nozzles:

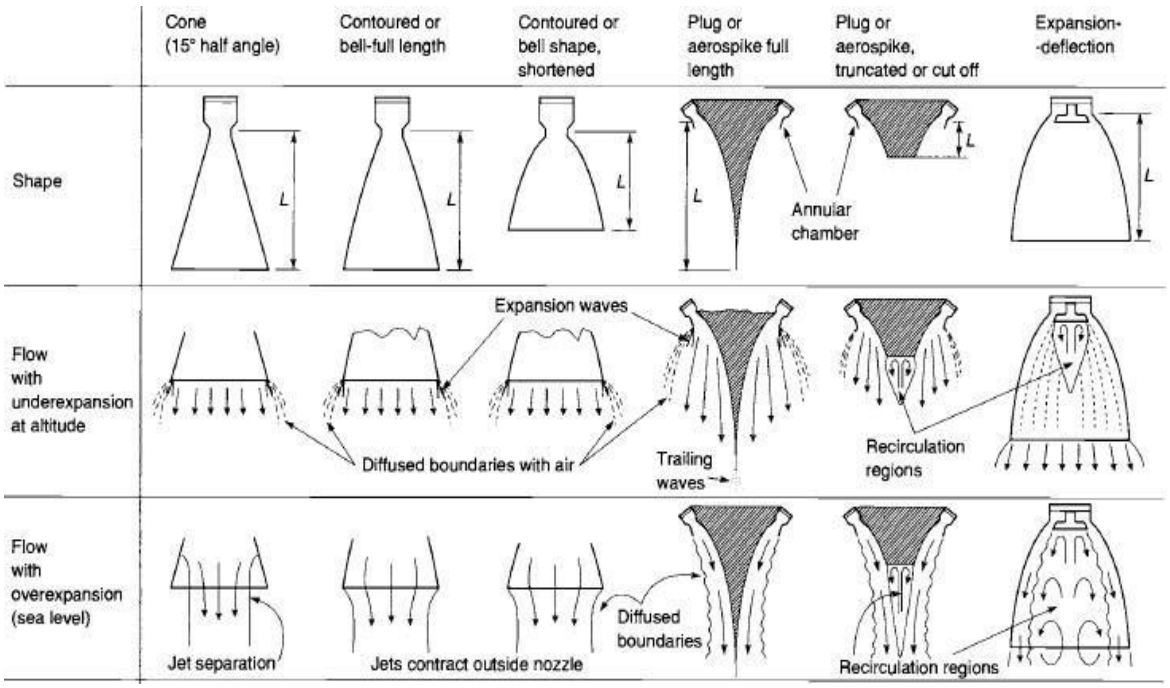

Fig: 3.3 Flow Conditions in Nozzle:

Multiphase Flow (Presence of Solid articles/Liquid droplets):

- In somerockets, thegaseous working fluid contains many small liquid droplets and/or solid particles that must be accelerated by the gas.
- This occurs in with solidpropellants andsome gelled liquid propellants which contain aluminum powderthat forms small oxide particles in the exhaust.
- It can also occur with ion oxide catalysts, or propellants containing beryllium, boron or zirconium.
- Ingeneral,iftheparticlesareverysmall,withdiametersof0.005mmorless,theywill have almost same velocity as the gas and will be in thermal equilibrium with the nozzle gas flow
- Thesolid/liquidparticles give up heattothegas during expansion inanozzle.

- Asthegases give upkineticenergytoaccelerate theparticles, theygain thermal energy from theparticles.

- Astheparticlediameterbecomebigger,thelargerparticles donotmoveasfastasthe gas and donot give up heat as readily as the small particles.

- Thelarger particles have a lower momentum and they reachnozzle exit at a higher temperature than the smaller particles.

- For larger particles, over 0.015 mmdiameter, the specific impulse canbe 10 to 20 % less than the specific impulse value without flow lag.

Chemical Equilibrium: The chemical equilibrium during the expansion process in the nozzle can be regarded as the following:

- **Frozen Equilibrium:** Thecompositionofthecombustionproductsisinvariant,thatis,no changeingascomposition.Therearenochemicalreactionsorphasechangesinthe nozzle flow. The product composition remains same from nozzle inlet to exit. This method is usually simple, but underestimates theperformance by 1 to 4%.
- Inthe frozen flow case, nochemical change occurs during expansion, thereareno rate processesatalloccurring,themoleculespreservingtheiridentity all theway.

Shifting Equilibrium:

- Instantaneous chemical reactions, phase changes occur between gaseous and condensed phases of all species in the exhaust gases.
- Thus, the product compositionshifts as the flow proceeds throughnozzle
- Theresults calculated are called shifting equilibrium performance.
- Thegascompositionandmasspercentagesaredifferentinthechamberandnozzleexit.
- Thisanalysis ismorecomplex andthevalues of theperformance parameters , are overstated by to 4%.

- Intheshiftingequilibriumflowcase,reactionsdooccur,theirrateissohigh(comparedto the expansionrate)thatconditionsadjustcontinuously tomaintainequilibriumatthe local pressure and enthalpy level.
- Withtheresult,thewholeprocesscanberegardedasreversible(andhenceisentropic)
- Theactual expansion processina rocket or ramjet nozzle is intermediate betweenthe extremes of frozen and shifting equilibrium flow.
- The equilibrium flow produces higher performance due to recovery of some of the chemical energy tied up in the decomposition of complex molecular species in the chamber - a kind of afterburning effect.

Propellant Performance: SPR Performance:

The burn rate of propellant r is related to the chamber pressure p, as given below:

$$r = ap^n$$

whereaisanempericalconstantinfluencedbythegraintemperature,andnistheburningrate exponent,calledcombustionindex. Nalsoisdependentontheinitiagraintemperature.

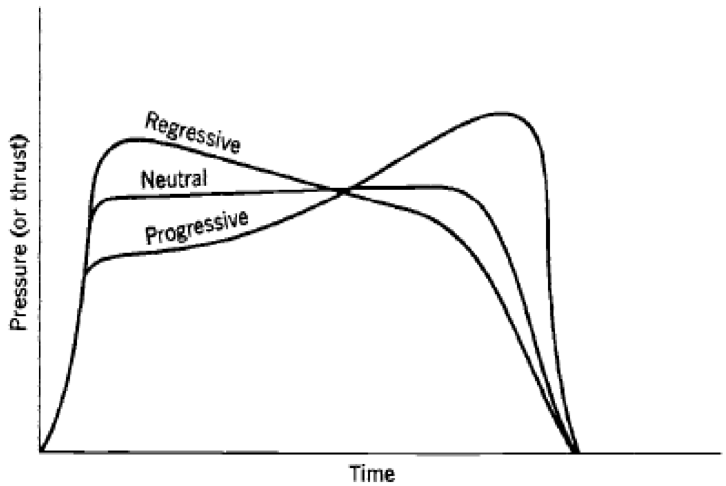

Fig: 3.4 pressure vs Time contour

- **Progressive Burning**-Thethrust, pressure and burning surface area increases with burn time.
- **Regressive Burning**-Thethrust, pressure and burning surface area decreases with burn time
- **Neutral Burning**-Thethrust, pressure and burning surface area remains constant through motor burntime.
- **Sliver**- Un-burnt propellant remaining in the casing after motor burn-out

The burning rate is a function of propellant composition. The burning rate of a composite propellant can be increased by

- adding, catalyst ormodifiers,
- decrease the oxidiser particle size
- increase oxidiserpercentage
- add plasticiser or binder to increase the heat of combustion
- imbed metal strips or metal particles

Other than propellant composition, the burning rate is also dependent on

- Combustion chamber pressure
- Initial graintemperature

- Velocity of gas parallet to the burning surface
- Combustion chamber gas temperature

Injectors:

Thefunctions ofinjectoraresimilartothoseofa carburetorofanI.Cengine.Thefunctionsare

- Injectorhastointroduceandmetertheflowofliquidpropellantsintothecombustion chamber
- Ithastoatomizethefuel,iecausethe liquidtobebrokenupintosmalldropletsinthe combustion chamber
- Ithastocausedistributionandmixingupofpropellantssuchthatacorrectproportionof mixture of fuel and oxidizer will result
- It has to ensure uniform propellant mass flow and composition overthe cross section of the combustionchamber.

Above functions are accomplished with different types of injector designs, as shown below:

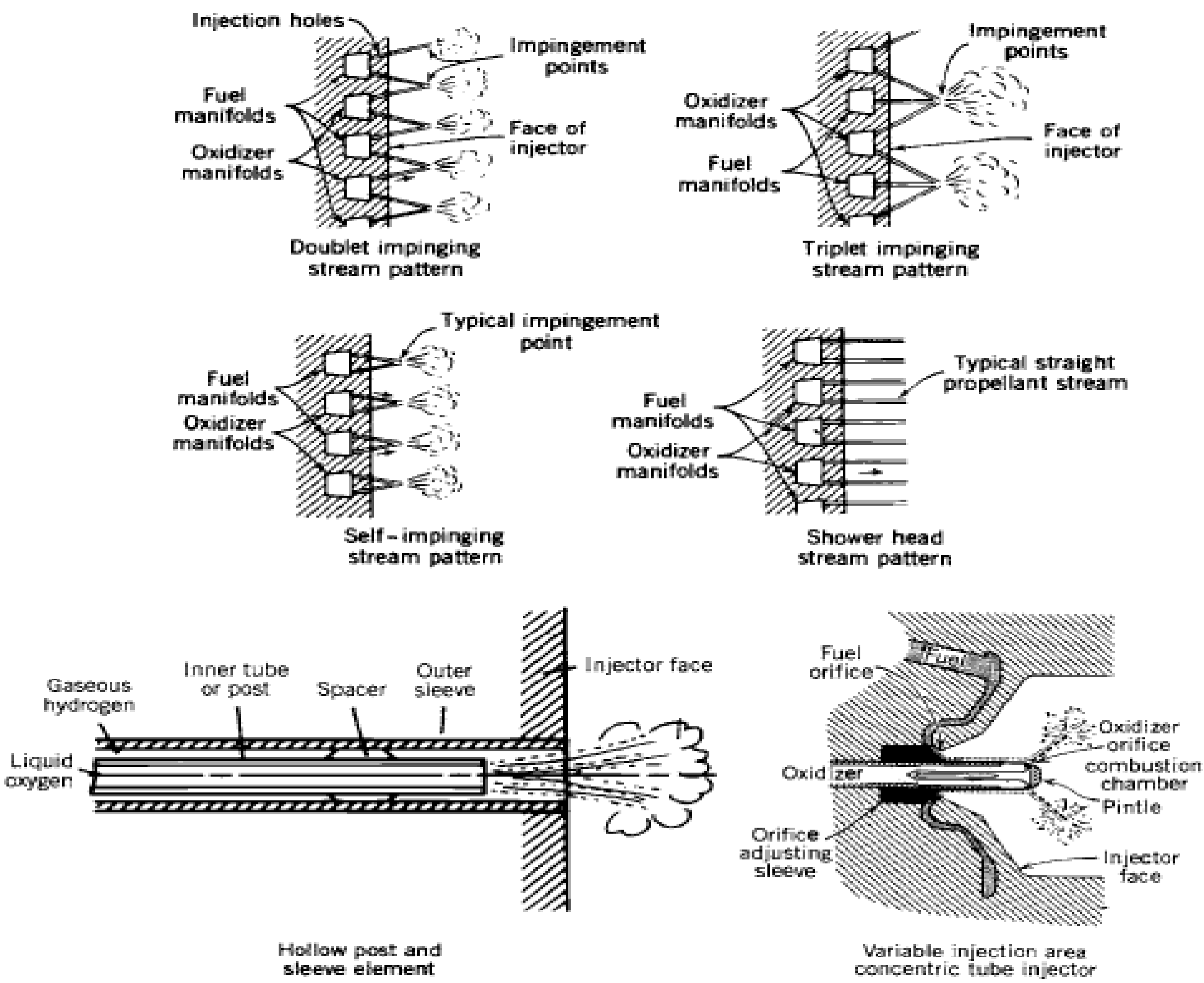

Fig: 3.5 types of injector

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Theinjectionholepatternonthefaceoftheinjectoriscloselyrelatedtotheinternalmanifoldsor feed passageswithintheinjector.Theseprovideforthedistributionofthepropellant fromthe injector inlet to all injection holes.

Alargemanifoldvolumeallowslowpassagevelocitiesandgooddistributionofflowoverthe cross sectionofthecamber.Asmallmanifoldvolumeallowslowforalighterweightinjectorand reduces the amount of “dribble” flow after the main valves are shut. However, the higher passagevelocities causeamoreunevenflowthroughdifferentinjectionholesandthuspoor distribution and wider local variation in composition.

Dribbling results in after burning, after valve closing leading to irregular combustion.

For applications needing very accurate terminal velocity requirements, the cut-off impulse has to be very small, passage volume is minimizedas muchas possible.

Impinging-streamtype, multiple-holeinjectors arecommonlyusedwithoxygen-hydrocarbon and storablepropellants.Thepropellantsareinjectedtroughanumberofsmallholesinsucha mannerthat the fuel and oxidizer streams impinge on each other. Impinging patterns can also be fuel-on-fuel or oxidizer-on-oxidizertypes.

The triplet pattern also is used in some cases.

The non-impinging or shower-head injector employs non-impinging streams emerging normal to the face of the injector. It relies of diffusion or turbulence and diffusion to achieve mixing.
However, this type requires large chamber volume and is not commonly used now.

Sheetor Spraytypeinjectors givecylindrical,conicalorothertypesofspraysheets,withsprays generallyintersecttopromotemixingandatomization.Thewidthofthesheetcanbevariesby using axiallymovablesleeve,itispossibletothrottlethepropellantoverawiderange.Thistype of **variable area concentric tube injector** is used in lunar module.

The**Co-axial hollowpost injector** isusedforliquidoxidizerandgaseous hydrogeninjectors (shownon lowerleftofabovediagram).Theliquidhydrogengetsgasifiedintheoutersleeveby absorbing heat from the cooling jackets. Thegasified hydrogen flows at highvelocity (around 330 m/sec) while the liquid oxygen flows slowly (around 33 m/sec). This differential velocity causes a shear action, which helps inbreaking up theoxygenstreamintosmall droplets.

Theinjectorassemblyshownbelow,usedspaceshuttle,has600concentricsleeveinjection elements, ofwhich75ofthemarelengthenedbeyondinjectorfacetoformcoolingbaffles,which reduces combustioninstabilities.

Factors influencing injector behavior: The approach to design and development of liquid propellant rocket injectors are based on empirical relations. The important factors that affect the performance andoperating characteristics of injectors aregivenbelow:

- **Propellant Combination**:Theparticularcombinationoffueland oxidizeraffects the characteristics suchaschemical reactivity, speedofvaporization, ignitiontemperature, diffusionofhotgasses,volatilityandthesurfacetension.Hypergolic(self-igniting) propellants generally require different designs from those required by propellants that must be ignited. Each combination requires owndesigninjector design.
- **Injector Orifice Pattern and Orifice Size**: Withindividual holesinthe injector plate, there is aoptimumperformanceandheattransferconditionforparameterslikeorificesize, angleof impingement,distanceofthe impingementfromtheinjectorface, numberof injector orifices per unit surface of injector face and the orifice distribution over the orifice plate surface. These parameters are decided experimentally or from similar successful earlierdesigns.
- **Transient Conditions**:Startingandstoppingtherocketmotoroperationrequirespecial provisions like temporary plugging of holes, accurate valve timing, insertion of paper cups over holes topreventientry ofone propellant intomanifold of other propellant etc.
- **StructuralDesign**:Theinjectorishighlyloadedbypressureforcesfromthecombustion chamberandthe propellant manifolds. During transients (starting and stopping), these pressure conditions casesevere stresses. Thefaces ofinjector areusuallyflat andneed reinforcements.Alsothestructureoftheinjectormustbeflexibleenoughtowithstand the thermal deformations causedbyheating byhot combustion gases andcoldcryogenic propellants.

  The injector design must also provide for sealing to prevent internal leaks.

Combustion Chamber:

Liquid Propellant Rocket: Thecombustionchamberorthrustchamberisacombustiondevice where theliquidpropellants aremetered, injected, atomized,mixedandburnedto formhot gaseous reaction products. These hotgases are acceleratedandejected fromthe nozzle at ahigh velocity to impart a thrust force.

A thrust chamber has three main parts, an injector, a combustion chamber and a nozzle.

In a cooled thrust chamber, one of the propellants (usually the fuel) is circulated through cooling jacketsto absorbtheheatthatistransferredfromthehotcombustiongases tothechamber walls.Thereare uncooledthrustchambers,whichuseablativematerialstowithstandhigh temperatures.

The combustion chamber wherethe burning takes place, must be designedto withstand the heat generated. The volume of the chamber must belarge enoughfor adequate mixing, evaporation and completecombustionofthepropellants.Thechamberdiameterandvolumeinfluencethe cooling requirement.

Thecharacteristic chamber length is defined as the lengththat a chamber ofsame volume would have if it were a straight tubeandhad no converging nozzle section.

$$L^* = V_C/A_t$$

Solid Propellant Rocket: Insolidpropellantrocketmotors,thepropellantiscontainedandstored inthe combustionchamber. Thestoragemaybeforlong periods ofaround15years,sothe motor casing s sometimes hermitically sealed.

The solid propellant rocket motor has the convergent divergent nozzle fixed onto the combustion chamber.

Desirable Properties of Solid Propellants:

- High specificimpulse
- Low molecular weight to provide high exhaust velocity
- High heat of formation to result in high temperatures
- Combustionproducts must contain simple light elements
- Highdensity to result in better specific energy and low size
- Simple to manufacture with few moving parts
- Re-usability ofcomponents
- Smoke-less, non-toxicexhaust

Nuclear Rocket

Power-Thrust-Energy: Thehighspecificenergyofnuclearfuelisthereasonwhich makesnuclearpropulsionidealfordeepspacemissionsincludingmannedmissions to otherplanets.

For voyagestoplanets, aspacecraftneeds to begivenaveryhighvelocityof above 11 km/s. The power in the exhaust stream will be

$$P = \frac{1}{2} m v_e^2$$

The thrust and power can be related as

$$F = m v_e$$

$$F = 2\frac{P}{v_e}$$

m is the mass flow rate.

Considering interplanatory mission with a departure velocity of 11km/s, the specific energy/power (per unit mass) required works out to 60.5 MJ/kg.

Themaximumexhaustvelocityofa**LOX/LH**engineisabout4.5km/sanditworks out to **energy per kg as 10.4 MJ/kg**. So about 6 kg of propellant is needed to be burntfor every1kgofvehiclemass,inorderto provideenoughenergytoseta vehicle off on interplanetary mission.

In comparison, the **energy contained in a kg of pure uranium 235 is 79.3 x 10^6 MJ**. A single kg of uranium 235 can provide energy to place a 1000 t vehicle for interplanetary mission.

Thehigh specific energyof nuclearfuel is a major advantage forhighenergy interplanetary missions.

Theenergystoredinnuclearpropellantsis10^7 -10^9 timeshigherthanchemical propellants. Apropulsionsystemusingnuclear energycanachieve anyspecific impulse comparable to the speed of light.

Nuclear Fuel Basics:

Nuclear processes(Fissionor Fusion) useverysmallquantities ofmatter.A working fluid, usually is coupled with nuclear reaction products.

Afterfusionorfissionofatoms,theendproductwillhavesmallermassthanthe initial atoms**. This mass defect is directly transferred in to energy based on Einstein's equation**

Fission Propulsion: Nuclear Fission is a processin which a large nucleus of an atom splits intotwosmallernuclei(lighternuclei)withreleaseofenergy.Thefission processoften producesfreeneutrons and releases averylarge amountofenergy. The splitting of nucleus is as a result of neutron bombardment.

The mass changes and associated energy changes in nuclear reactions are significant. Forexample, the **energy released from the nuclear reaction of 1 kg of uranium is equivalent to** theenergyreleasedduringthecombustionof **aboutfour billion kilograms of coal**.

Nuclear Fission:

NuclearFissionisusedinhighthrustapplications.Fissionisaprocesswherea neutron is absorbed a uranium nucleus, which causes the nucleus to split into two nuclei(ofmass abouthalfthatofuranium).Themassdefectcausesreleaseof energy, inthe formof kineticenergyofthetwofissionfragments. Thesplitting processisalsoassociated withreleaseoftwoormoreneutronsareemittedatthe sametimeasthefissionofthe nucleusoccurs.Theseneutronsgotointeractwith another nucleus and cause to split, thereby, settingup achain reaction. Since rate atwhichenergyisreleaseddependsonly ontheneutronflux,thepoweroutputof afissionsystem is controlledbyinserting materials thatabsorbneutrons.

In a controlled nuclear fission, the uranium becomes very hot, leading to melting of Uranium.Hence, tocontinue withtheenergyrelease, itis essential tocoolthe uranium extracting heat. The cooling of Uranium is accomplished using a propellant, which passesthroughthereactorandthenexpelledoutofthenozzle.

Twoisotopesof Uranium, $U^{238}$and U^{235} are available, ofwhich U^{235}has high probability of initiating fission process.

Nuclear Fusion:

- If two light nuclear cores are fused together (Eg. hydrogen), the resulting heavier nuclear element has less binding energy than the sum of the two original ones.
- The energy difference is released as heat.
- Fusionismorecomplexthanfission,sinceinfusion,inordertobringthetwo positively charged nuclear coresclose together, the energyof electrostatic repulsion has to be overcome and maintained
- Theenergyreleased in nuclearpropulsion isgovernedby Einstein'sequation $E=mc^2$
- Nearlyallgainedenergythroughthemassdefectis releasedasheat.
- While Fission and Fusion transfer onlypart of thenuclear bindingenergy in to heat, the Matter-Antimatter annihilation (eg. Proton-antiproton or hydrogen-antihydrogenetc) canreleaseallthenuclearenergy.

Sizing of the Reactor/Ensuring Sustainable Chain Reaction:

There are two approaches that will improvethechancesof sustainable chain reaction. They are

- **Enrichment of U^{235}**: It involves increasing the percentage of U^{235} in the natural uranium to a level thathighly increases the probability. The process of enrichment is complicated and costly. Natural Uranium contains very low %ofU^{235}(0.72%).AlthoughU^{238}alsoparticipatesinfissionprocess,the probability of fission initiation is very low since U^{238} requires collisions with neutronswithhigh energylevels,thusreducingtheprobabilityverylow.

- **Useofmoderator**:Thesecondapproachistoslowtheneutronsquicklyand reduce absorption of neutrons by U^{238} nucleibyusinga moderator, usually carbon or water. The moderator is mixed with the uranium atoms in a **homogeneous reactor**, orthemoderatoranduraniumcanbeinseparate blocks, as a**heterogeneous reactor**.

Theheterogeneousreactor whichusescylindricalrodsof Uraniumseparated by blocksof moderator,improvestheprobabilityofsustainedreactionhighand permitsuseofmore naturalUranium. However,thisincreasesthesizeofthe reactor, as more moderator is required.

Forspaceapplications,theneedtokeepsizelow,requiresuseofenriched Uranium. Plutoniumcan also beused in thesamewayasenriched Uranium, but the material is poisonous and highlyradioactive. Safety issues are complexto handle.

Calculating Criticality: Criticality factor relates to calculating the space fission reactor that can attain sustainable chain reaction with minimum size. The following key issues are considered while decidingthesize of spacereactor:

- Inafissionreactorusingmoderator,sufficienttraveldistancemustbe provided for neutrons to slow down adequately and avoid being absorption by the U^{238} nuclei.
- The slowing down must occur in the moderator.
- When Uranium with low enrichment is used, the Uranium is concentrated in the fuel rods, separated by blocks of moderator.
- Therefore, the size of the reactor is mainly decided by the dimensions of the moderator.
- Leakage of neutrons from the reactor reduces the neutron flux and leads to low probabilityofsustainedfission. Neutronleakage mustbelow.
- Largerreactors willhave lesser leakage thanthe smallerones.
- Heat generated byfission must be efficiently removed preventing reactor core fromoverheating.
- Propellant flow through channels passing through thereactor must be carefully designed for efficient cooling.

- The best shape for the reactor to minimise neutron leakage and provide for propellant channels is cylindrical, with height approximately equal to diameter.

The criticality factor is defined by the "four-factor formula", as given below:

$$K_\infty = \infty \varepsilon p f$$

$K_\infty$is called "multiplication factor" or "reproduction constant"

K_∞indicates the effective number of neutrons per fission that survive all the loss mechanisms and cause fission in another nucleus.

For $K_\infty < 1$, nochain reactionispossible For

$K_\infty > 1$, thechainreaction ispossible

$K_\infty = 1$isthecriticallevelandK$_\infty$willneedstobecontrolledat1forsteady production of heat in the reactor.

The subscript∞ refers to a reactor size corresponding to infinite, where neutrons cannot leak out throgh sides.

The four parameters that influence value of K_∞ are:

∞isthenumberofneutronsthatemergefromfissionofthenucleus,perincident neutron.U^{235}nucleusproduces2.44neutronsonanaverageperincident.The value of $\eta$for U^{235} is 2.07, available for furthurfission process.

The value of η must be far higher than unity for catering for loss mechanisms.

εisthefastfissionfactor,indicatestheprobabilitythataneutron isavailablefor furthur fission process. Value of ε should be 1.

Pisthe "resonance escapeprobability", which indicates chances of absorption by U^{238} nuclei before causingfurturfission process. Value of pdependson fraction of U^{238} in thefueland itsdistribution. If themoderatorslowsdowntheneutrons quickly,theirchancesofcapturearereduced,withvalueofphigh.Valueofp ranges from 0.6 to 0.8.

Thefourthparameterfisthe“thermalutilizationfactor”,indicatingprobabilityof capture of low energyneutrons after slowing down by moderator.

Reactor Dimensions/Neutron Leakage:

Asthesizeofthereactordecreases, theneutronleakageincreases,lessspaceis availableformoderator.Therefore,moreneutronsneedtobeprovidedwhich requiresenrichmentofnaturalUranium.Forverysmallreactors,almost90% enrichment of fuel is needed.

The keyfactors that determine reactor size are neutron leakage from the core, and the ability of moderator to prevent neutron absorption. Two properties of neutrons, diffusion length and slowing-down length are critical.

Diffusion length represents the way scattering in the moderator reduces the neutronflux,asthedistancefromsourceofneutronsincrease.Itisabout52cmin graphite.

Theslowing-downlengthexpressesthemeandistancetravelledbyneutrons, through moderator before reaching thermalenergies(escapingabsorption).

It is about 19 cm for graphite.

For any reactor of finite dimensions, neutron leakage will occur.

Relation between neutron leakage and reactor size is given by the formula

$$N = N_0 e^{\frac{-r}{L_r}}$$

WhereN&N_0arethenumberofneutronscrossingaunitvolumeofmaterialinat the sourceandasthedistanceincreases, situatedatadistancerfromthe isthe diffusion length.The neutron flux also varies with time, depending whether the reactor is sub-critical orsuper-critical.

Thecriticallink betweengeometryofthereactorandthecriticalityisgivenbythe "**buckling factor**".

The buckling factor is calculated based on neutron diffusion in a reactor of different shapes.Itisfoundtobeinverselyproporional to thethelength LandradiusRof the reactor.

Control:

Controlofneutronfluxandhencethepower outputisessentialforthereactor. Controlismaintainedbyusingnumberofcontrolrodswithhighabsorptioninthe core. Thecontrolrodsmoveinachannelandbeinsertedorwithdrawnfromthe core.

When fully inserted, they absorb the neutrons so that the reactor goes sub-critical andthe fissionstops.Atanintermediateposition,theneutronsareabsorbedjust enough to retain the criticality.

The control rods are connected to a neutron flux sensor with a feedback mechanism, to hold the reactorat any desired condition.

Atthestartup,therodsarewithdrawnsothatkis greaterthanoneandnetronflux and poweroutputincreases.Oncedesiredcriticallevelisreached,therodsare insertedinto theintermediateposition.Shutdownisachievedbyfullyinserting the rods in to the core.

Reflection:

Innormaloperation, theneutronsdiffusingoutofthenuclearcorewillbelostin fission processorgetabsorbed.Smallerreactorscanbedesignedtocausethe neutronsto diffusebackagainintothereactor,afterleavingthecore,spending some time scattering off the nuclei in the external moderator. Some neutrons diffusingoutof reactorcorewillparticipateinthefissionprocessandtheremaining couldbemade to diffuseback. Acorefittedwithanexternalmoderator,called "reflector" can be advantageous, in that smaller quantities of U^{235} is needed to achieve criticality.

Forspacebasedreactors,abilitytocontrolneutronreflectionprovidesacontro element. This reduces the need for internal control rods which are inconvenient in a spacereactor.

Reflector will help in

- Reducing the cost of material
- Reduce the neutron leak out of the reactor
- Better neutron density distribution in the core
- More even power distribution in the core
- Canavoiduseofinternalcontrolrodsforregulatingneutronfluxinthespace reactors.

Prompt and Delayed Neutrons:

Thefissionprocessinside thenuclear core involvesneutronsbeingreleased and travelling to the next nuclei/moderator along path. With-in the nuclear dimensions, thetraveltime is almostinstantaneous, with in afewmilliseconds. This would makethecontrol mechanismofmovingcontrolrodsin/outofthecoretoregulate neutron flux very difficult.

However, the controlprocess is helped/made effective due to presence of “delayed neutrons”. The movement of about 1% of the neutrons is delayed because formation of unstable intermediate nuclei of isotopes like iodine and bromine whichundergo decayduringthenuclearprocess,butwillcauseinducetimelag between prompt neutrons and delayed neutrons.

Thedelayed neutrons makes the controlprocess though movement of controlrods more effective.

Thermal Stability:
Themultiplicationfactorkissensitiveto temperature.Kdecreaseswhenthe temperature raises. This isdue to thefact thatdensityof corematerials increases causingthemto expand,increasingthemeandistancebetweencollisionand increases the probability of fission.

Thermal stability is a factor that makes the controlled release of fission energy easier.

As kgets more than 1, the increased release of energy due to neutronflux being more, increasesthetemperature,whichinturn,reducesthevalueofk.Thus thermal stability is established.

Therearetwofactorsatwork,whichgovernthepoweroutput.Forastablestateof the core,valueofkisone.Thepowerleveldependsontheneutronflux,whichis stable only when k equals one.

Toincreasethepowerlevel,valueofkisallowedtobecome greaterthan1.Once the desiredpowerlevelis reached,k is returnedtovalueof 1, andthereactor contimues to produce power at the new level. A decrease of power is also established in a similar way.

Nuclear Thermal Propulsion-Principle:

The engine consists of a nuclear reactor, with the propellant used as a coolant for the core. The heat generated by fission is carried away by the propellant, and the hot propellant is expanded in the nozzle.

Thecorecontainshighlyenriched Uranium,mixedwitha quantityof moderator. Higher thelevelofenrichment,difficltistocontroltheengineandcostisalsohigh. However, lowering the enrichment increases the size of the reactor.

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Nuclear Thermal Rocket Engine:

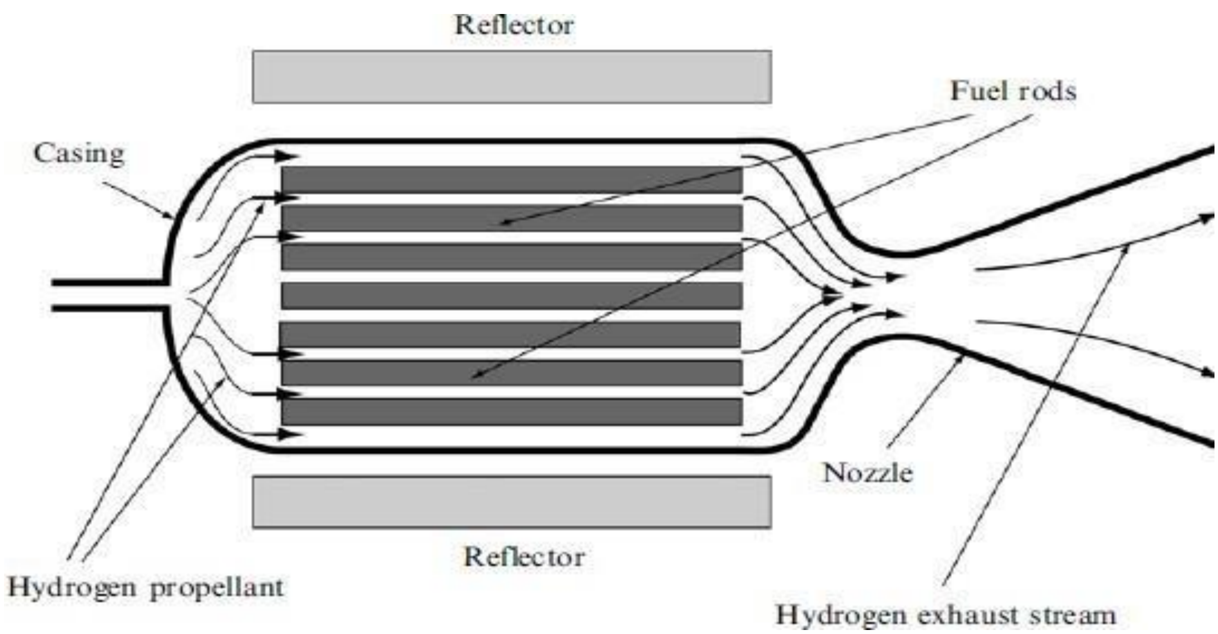


Fig: 5.1 Nuclear rocket

Hydrogen is used as propellant, which gets heated in the core, and expands in the CD nozzle.

The rate of fission and the heat production is controlled by the reflector.

Although, the nuclear thermal engine is similar to a chemical engine as far as the principle is concerned, there are issues specific to nuclear energy/materials that need to be addressed.

There are several very specific engineering details are unique to the fission engine.

Radiation and its management: Nuclear fission produces the radiation effects both duringtheoperationandafteruse.Pureuraniumbyitselfissafetohandle,sinceits half life is very high, The fission rocket engine is safe and non radioactive as long as it has not been fired. The nuclear thermal rocket engine must be launched in space.

Radiation created during operation of the engine is through neutrons, alpha/beta particles and gamma rays. During operation, the entire core is heavy with radiation flux. Beyondthecasing,there isahighfluxof bothneutronsandgammarays, which is dangerous to humans and also to electronics, both need protection during firing.

A radiation shield made up of one or more discs high-density material is mounted onthe forwardendoftheengine. Anyhumanscanbesafelyin thecabinwell forward the engine.

An additional external shield is also provided to reduce the effect of gamma-ray flux produced by the neutron capture by the internal shield.

Otherthantheforwardside,theradiationshieldisnotprovidedanywhereelseon the spacecraft.

Propellant Flow & Cooling:

The propellant flow is similar to chemical liquid engines except that the rear no injectors and need for mixing. There is a need to cool several components of the engine. The power output of the reactor must be matched by the rate at which the h e a t is extracted by the propellant and exhausted down the nozzle.

The reflector and the casing need to be cooled. This is done by passing the hydrogen propellant through channels in the reflector, Pumps are provided to ensure flow of propellant through the channels at desired rate.

Start-up and Shut-down:

Thestart up of the nuclear thermal rocket is similar to a cryogenic chemical engine. The wholedistributionsystemhastobecooleddownsothatthecoldhydrogen does not cause thermal shock in the components. Once started, the power output ofthereacorwill raiseveryquicky,in matterofseconds.Thecoolingofthecore casing by the propellant must keep pace with rapid heating.

Initiallythepressureinthechamberisnotadequatetodrivethepropellantturbo-pumps. Initially,duringstartingphase, electricalpowermustdrivetheturbo- pumps.

Once,theengineisinstableoperatingmode,thethrustcanbevaried by positioningthecontrolrods.Thepowerouputisafunctionofneutronflux.

About1%oftheneutronsproducedbyfissionaredelayed..Whenthereactoris shut down, thefissionprocessandhencethepoweroutputcontinues to be produced. So fission heating will go on for several moments measured by the half life period of the fission material.

Thus the shut down is a complicated process in nuclear fission rocket.

Potential applications of Nuclear Engines:

1. Thespecific energyofnuclear propellant is far greater than chemical propellant.
2. High Δv values can be obtained by nuclear propulsion.
3. Large incrementsof Δvare possible with low usage ofpropellant in nuclear propulsion.
4. The advantage ofnuclear rocket is intermediatebetweenchemical and electricalpropulsionwhenonleexhaustvelocity is considered.
5. Anionenginecanonlygeneratethrustoffractionofa Newton,butnuclear engine can produce thrust in hundreads of Newtons.
6. Nuclear Enginescan provide thehigh delta velocityrequiredfor interplanetarymissions to Mars, Venus andbeyond.
7. Useof nuclear enginesfor space journeys can shorten the time of journey to a greatextent.

Development Status of Nuclear Thermal Rocket:

Both US and Russia are undertaking development of nuclear thermal rocket.

The groundtestingof nuclear thermalrockethas been stopped since 1970 dueto restrictions placed on release of nuclear contaminated exhaust from the rocket.

Thereisrenewedinterestintheneedforanuclearthermalrocketengineasthe main booster for the manned mission to Mars.

Oneproposalthatis feasible,butcostlyis to testnuclearcoreinspace.And activation and safe disposal of thecoreneeds to besortedout. Thesafety issues also need to be addressed sincenuclear coreforspace applications need to use enriched Uranium.

Itislikelythatanuclearpropelledmissionwillbemountedinthenextdecade.The proposalunderconsiderationisthatafissionreactorwillprovidetheelectricity necessary for an electric propulsion. Ifthesafetyaspectsandpoliticalacceptancecanbeobtained,thenthenuclear thermal engine will take its place in the propulsion systems for space exploration.

Electrical Rocket:

Limitations of Chemical Rocket Engines:

1. **Explosion& FirePotential**:Explosionandfirepotentialislarger,failurecan be catastrophic.
2. **Storage Difficulty**: Some propellants deteriorate (self-decompose) in storage. Cryogenic propellants cannot bestored for long periods exceptwhen tanks arewell insulated. Afewpropellants like Red Fuming Nitric Acid(RFNA) give toxicvaporsand fumes.Undercertainconditions,somepropellantsand grains candetonate.
3. **Loading/Transportation Difficulty**: Liquid Propellant loading occurs at the launchstandandstoragefacilityisneeded.Manypropellantsrequire environmental permit and safety features for transport on public conveyance.
4. **Separate Ignition System:** All propellants , except liquid hypergolic propellants, needignitionsystem. Eachrestart requires separate ignition system.
5. **Smoky Exhaust Plume:** Smoky exhaust plumecan occur with some hydrocarbon fuels. If the propellant contains more than a few percent particulate carbon, aluminum or other metal, then the exhaust will be smoky and plume radiation will be intense.
6. **Need Thermal Insulation:** Thermalinsulationisrequired inallrocket motors.

ELECTRIC THRUSTERS-MISSION APPLICATIONS TO SPACE FLIGHT

(GP Sutton: Pages 660-664; Martin Turner; Pages167-217)

Limitations of Chemical Rocket Engines:

- **Explosion & Fire Potential (SPR&LPR)**:Explosionandfirepotentialislarger,failurecanbe catastrophic.
- **Storage Difficulty (SPR &LPR)**: Somepropellants deteriorate(self-decompose) instorage. Cryogenicpropellants cannot bestored for long periods except when tanks are well insulated. A few propellantslike Red Fuming Nitric Acid(RFNA) give toxicvapors andfumes. Under certain conditions, some propellants and grains can detonate.
- **Loading/Transportation Difficulty (SPR & LPR)**:Liquid Propellant loading occurs at the launch stand andstorage facility is needed. Many propellants require environmental permit andsafety features for transport on publicconveyance.
- **Separate Ignition System (SPR &LPR):** All propellants , except liquid hypergolic propellants, need ignition system. Each restart requires separate ignitionsystem.
- **Smoky Plume (SPR & LPR):** Smoky exhaust plumecan occur with some hydrocarbon fuels. If the propellantcontains morethana few percentparticulate carbon, aluminum orothermetal, then the exhaust will be smoky and plume radiation will be intense.
- **Need For Thermal Insulation (SPR & LPR):** Thermalinsulationisrequiredinalmostallmotors.

- **Difficult to detect grainintegrity(SPR):** Cracksinthegrainandunboundedareasaredifficultto detect.
- **Toxic Exhaust Gases (SPR)**: Exhaust gases are usually toxic for composite propellants containing Ammonium Perchlorate.
- **Difficult to Re-use (SPR):** If designedforreuse, themotorrequiresextensivereworkandnew propellants.
- **Difficult tochangethrustratings(SPR):** Onceignited,thepredeterminedthrustandduration cannot bechanged.

- **Complex Design(LPR):** Relativelycomplexdesign,morepartsandhencemoreprobabilityfor malfunction.
- **Sloshingin Tanks(LPR):** Sloshingintankscancauseflightstabilityproblem**.** Bafflesareneeded to reduce the sloshing problem.
- **Combustion Instability(LPR):** Difficult to More difficult to control combustion instability.
- **Zero-Gravity Start(LPR)**: Needs special design provisions for start in zero-gravity .
- **Spills & Leaks (LPR):** Spills and leaks can be hazardous, corrosive and toxic. They can cause fires. **More**
- **Overall Weight (LPR):** More overall weight for short duration, low-total-impulse applications.

- **Tank Pressurisation (LPR):** Tanks need to be pressurized by separate system. This needs high pressure inertgas storage for long periods of time.

Electric Propulsion Systems:

Structure: The basic subsystems of aelectric propulsion thruster are

1 **EnergySource**:Energysourcethatcanbesolarornuclearenergywithauxiliary componentslikepumps, heatconductors, radiators andcontrols. Theenergysourceis different from thepropellant;

2 **Conversion Devices**: Theconversiondevicestransformtheenergyfromabovesource in to electrical form at proper voltage, frequency and current suitable for electric propulsion system;

3 **Propellant System**:Thepropellantsystemstores, metersanddeliversthepropellantto the thruster;

4 **Thruster**:Oneormorethrustertoconverttheelectricenergyintokineticenergy exhaust. Thetermthruster iscommonlyusedtomeanthe thrustchamber.

Types of ElectricThrusters:Threefundamentaltypesofelectricthrustersareavailable;

1 **Electrothermal**: In this type, the propellant is heated electrically and expanded thermodynamicallywherethegasisacceleratedtosupersonicspeedsthroughanozzle, as in chemical rockets, to produce thrust.

2 **Electrostatic or Ion propulsion engine:** Inthis type,**a**ccelerationisachievedbythe interaction of electrostatic fields on non-neutral or charged propellant particles such as atomic ions, droplets or colloids.

3 **Electromagnetic or Magneto plasma engine:** In this type, the acceleration is achieved by theinteractionofelectricandmagneticwithinaplasma.Theplasmasaremoderately dense, hightemperaturegaseswhichareelectricallyneutralbutgoodconductorsof electricity.

Electrothermal Thruster: Electrothermal thrusters use thesimplestwaytoheatthe propellantwithahotwirecoil,throughwhichanelectriccurrentpass.Moreenergy canbe deliveredfromelectriccurrentifanarcisstruckthroughthepropellant,which generates higher temperature than the resistive approach and therefore produces a higher exhaust velocity.

The propellantis heated electrically by heatedresistors or electric arcs andthe hot gas is thermodynamically expanded in a nozzle and accelerated to supersonic speeds. The electrothermalunitshavethrustrangesof0.01to0.5N,withexhaustvelocitiesof1000 to 5000m/sec.Ammonium,hydrogen,nitrogenorhydrazineareusedaspropellants.

Aschematicdiagramofarcheatingelectricpropulsionsystemisshownbelow. Thearc plasma temperature is around 15,000 K.

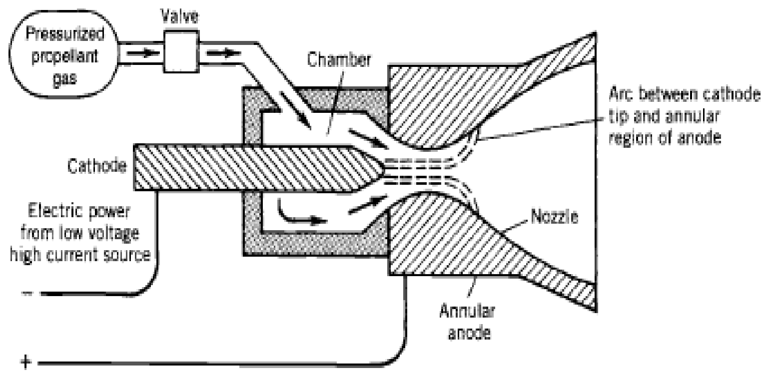


Fig: 5.2 Electromagnetic thrusters

Electrostatic and Electromagnetic thrusters accomplish propulsion through different means. They do not use thermodynamic expansion of gas in the nozzle.

Both Electrostatic and electromagnetic thrusters work only in vacuum.

Ion Rocket Engine (Electrostatic Thruster): Inanionrocketengine,aworkingfluid,like xenon, is ionized bystripping off electrons. The electrically charged heavyions are then acceleratedtoveryhighvelocities(2000to60,000m/sec)bymeansofelectrostatic fields. The ions are subsequently electrically neutralized by combining them with electrons to prevent building up of aspace charge on the vehicle. Asimplified schematic diagram of an Ion Rocket is shown below:

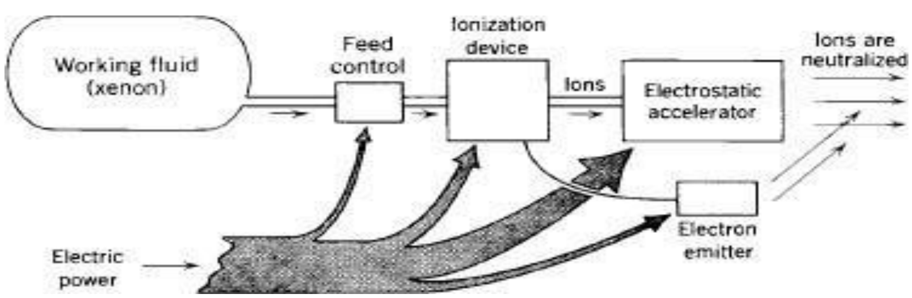


Fig: 5.3 Magneto plasma Rocket

Magneto plasma Rocket (Electromagnetic Thruster): Electrical plasma is an energized hot gas containingions,electronsandneutralparticles. Inthemagnetoplasmarocket,anelectrical plasmais acceleratedbytheinteractionbetweenelectriccurrentsandmagneticfieldsand ejected at high velocity (1000 to 50,000 m/sec).

A simple pulsed (not continuously operating) unit with a solid propellant is shown below:

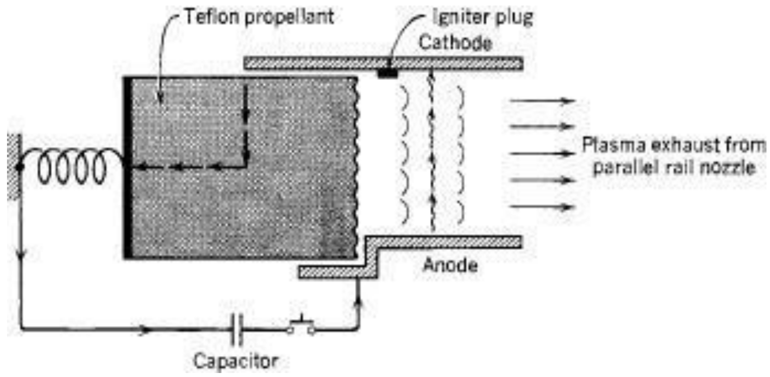


Fig: 5.4 thruster uses a parallel rail accelerator

Thethruster uses a parallel rail accelerator for self-induced magneticacceleration of a current carrying plasma.Whenthecapacitorisdischarged,anarcisstruckattheleftsideoftherails. Thehighcurrentin theplasmaarcinducesamagneticfield.Theactionofcurrentandthe magneticfieldcausesthe plasmatobeacceleratedatrightanglestoboththecurrentandthe magnetic field, ie in the direction of the rails.

Each time an arcis created, asmall amount of propellant(Teflon), is vaporized and convertedin toasmall plasmacloud.Theplasmaisthenejectedgivingasmallpulseofthrust.Thethruster can operate with many pulses per second.

The magnetoplasma rocket is used as spacecraft attitude control engine.

Performance ofElectricThrusters:

- The thrust levels of Electric thrusters aresmall relative to chemical and nuclear rockets. They have
- substantially higherspecificimpulse whichresults in longer operational life for satellites whose life is limited by quantity of propellant they carry.
- Electricthrustersgiveaccelerationstoolowtoovercomethehighgravityfieldearthlaunches. They operate best in low vacuum, in space.
- All flight missions envisioned with electric propulsion operate in gravity-free space and therefore, they must be launched from earth by chemical rockets.
- For electrical thrusters, the key performance parameter is the power-to-mass ratio ie W/kg. The power doesnotdiminish withprogressthroughthe flight, whilethemassofpropellantina chemicalrocket decreasesasthevehicleaccelerates.Thisisthekeydifferencebetween Electrical and chemical rockets.

Current Technology: Theelectricalthrusters needsubstantialquantities of power on board. All types of present dayelectricalthrustersdependonvehicle-bornepowersource-basedonsolar,chemicalor nuclear energy.

- Themass of electricgenerating equipment, powerconversionand conditioning equipment can become much higher increasing the mass of thrusters.
- This causes high increase of inertvehicle mass.

Application of Electricthrusters:Theapplicationfallsintofourbroadcategories:

1. Overcoming translational androtational perturbations: These wouldinclude
   - Stationkeepingforsatellitesingeosynchronousorbits(GEO),
   - Aligningtelescopes or antennasinLowEarth Orbits(LEO) andMediumEarth Orbits (MEO)
   - Drag compensation for satellites inLEO and MEOs
2. Increasing satellite speed to overcome weak gravitational field, for Orbit raising from LEOto a higher orbitevenuptoGEO.Circularizinganellipticalorbit;Thiswouldrequirevelocity increments of 2000m/sec to 6000 m/sec
3. Potential missions as Inter-planetary travel or Deep spaceprobes.

Electric Vehicle Performance: Thepropulsiveforcedevelopedbyanelectricthrusteristhemomentum transferred to the propellant. The Rocketequation applies to electric thrusters;

$$V = v_e log_e \ R, \text{where R is the mass ratio,}$$

R= $\frac{M_0}{M}$; M_0 isthe mass ofrocket at ignition(initial mass) and Mis mass of vehicle (final mass) Rcan alsobeexpressedasR= $\frac{M_S + M_p + M_E,}{M_S+M_E}$

where M_s ismass ofstructureincluding payload, propellanttanks and thrusters, M_p ismassof propellant, and $M_E$is mass of power supply equipment on board.

We define the power-to-mass ratio, ξ as

ξ= $\frac{P_E}{M_E}$ (W/kg); where P_E is the electric power, M_E is the mass of electric power equipment The thrusters have an η, inconvertingelectricpower to thrust, whichisexpressedas

$$\eta = \frac{mv_e^2}{2P_E} \text{ , where m is the mass flow rate, t is the burn time;}$$

$$m = \frac{M_P}{t}$$

Theexhaustvelocity v_e can be expressedas

$$v_e = \sqrt{\frac{2P_E}{m}} = \sqrt{\frac{2\xi M_E}{m}} = \sqrt{\frac{2\xi t M_E}{M_P}}$$

or theexhaust velocity can be expressed as, $\frac{M_E}{M_P} = \frac{v_e^2}{2\xi t}$

ThethrustdevelopedbythethrusterFcanthenbewrittenas,

$$F = m\ v_e = \sqrt{2m\xi M_E} = \sqrt{\frac{2\xi M_E M_P}{t}}$$

Theexhaustvelocity v_e isnotafreeparameter.Itisdecidedbythepower P_E andthemassflowratem. The massflowratem,inturndependsonburntimetandmassofpropellant M_P

Theenergycarriedawaypersecondbytheexhaustis $\frac{1}{2}\,m v_e^2$,thisisgovernedbythepowerconvertedin the thruster.

Increasing the exhaust velocity or the mass flow rate, therefore, require an increase in the power supplied to thethruster.

Highermassflowratealsoimpliesshorterburntimet. The

rocketequation can be expressed as,

$$V = \sqrt{\frac{2\xi M_E}{m}}\log\left(1 + \frac{M_P}{M_S + M_E}\right)$$

The power output P_{jet} isequal to $\frac{1}{2}\dot{m}v_e^2$. Thepower-to-thrustratio, $\frac{P}{F}$ can be written as

$$\frac{P}{F} = (\tfrac{1}{2}\dot{m}v_e^2)/\dot{m}v_e = \tfrac{1}{2}v_e = \tfrac{1}{2}g_0 I_s$$

Example: Determine the flight characteristics of an electrical propulsion thruster for raising a low earth satellite orbit. Data given is:

I_s =2000sec;F=0.20N;burntime(duration)=4weeks=2.42 $\times 10^6$ sec;Payloadmass=100kg; ξ = 100 W/kg; η =0.5

Theflightcharacteristicparametersare $\dot{m}$, M_P, P_E, M_E and Velocity increment ΔV

$\dot{m}$ =F/($g_0 I_s$), since I_s =F/$\dot{m}g_0$

$\dot{m}$ = 0.2/(2000×9.81) = 1.02 $\times 10^{-5}$

Themass ofpropellantM_P = $\dot{m}$t= 1.02 $\times 10^{-5}$ ×2.42 $\times 10^6$ = 24.69kg

The electrical power requiredis P_E=($\frac{1}{2}\dot{m}v_e^2$)/η = $\frac{1}{2}$(1.02 $\times 10^{-5} \times 2000^2 \times 9.81^2$)/0.5=3.92kW

Themass of electricalpowersystem, M_E will be

M_E = P_E/ξ = 3.92/0.1 = 39.2 kg ;

Thefinalvehiclemassafterburnout=massofpowersystem+massofpayload=39.2+100kg

Theinitialvehiclemass M_0 =final vehiclemass+propellantmass= 39.2+ 100 + 24.69 = 163.9 kg The velocityincrementΔV= $v_e log_e$ R=2000 × 9.8 ln (163.9/139.2) = 3200 m/sec

System Parameters-Interrelations:

1. **Vehicle Velocity V as a function of Exhaust velocity v_e**:TherelationbetweenVand v_e isgiven by theequation

$$V= \sqrt{\frac{2\xi M_E}{m}}\log\left(1+ \frac{M_P}{M_S+M_E}\right)$$

The exhaust velocity is given by the relation

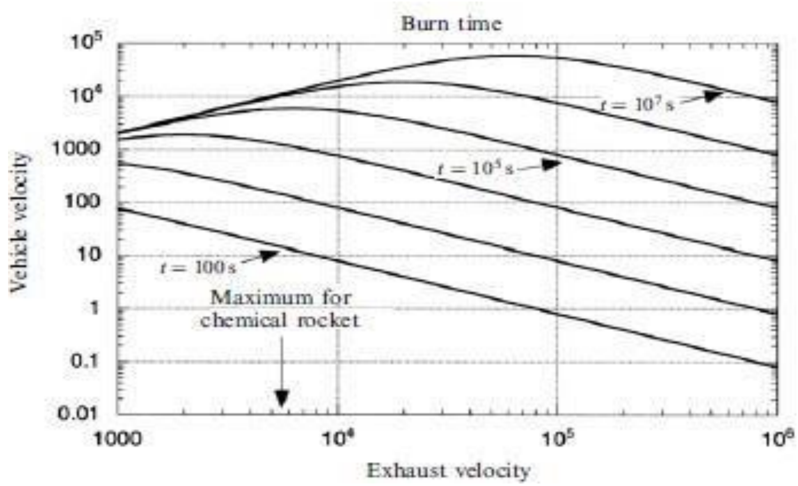
$$\frac{M_E}{M_P}=\frac{v_e^2}{2\xi t}$$

Wecanwritethe massratio as, R= $\frac{M_s + M_p + M_E,}{M_S+M_E}$which can be written as

Above equations indicate that

- Themassratio R(dryvehicle weightdividedbypropellantweight) foragivendryvehicleweight, decreases astheexhaustvelocityincreases.Thisisbecausehigherexhaustvelocityneedshigher power supply mass.
- Thismeansthatfortheelectricalthrusters,anincreasein v_e requiresanincreasemassofpower source, or dryvehiclemass, therebyresultinginnoimprovementofvehicle velocity.

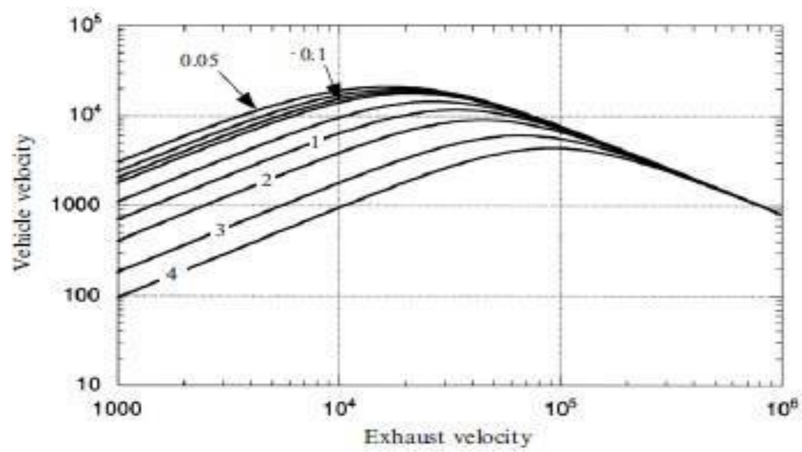
- Figure below shows vehicle velocity as afunction of exhaust velocity and burn time t



- Abovegraphassumesafixedrelationshipbetweenexhaustvelocityandpowersupplymass, with burn time as a parameter.
- Theratioofstructuralmasstopropellantmassisalsofixedat0.15,equivalenttoamassratioof 6.6.
- Itis evident that vehicle velocity does not always increase with exhaust velocity., and peaks for a certain value.
- Increasing the burn time, increases the peak value, both of the vehicle velocity and optimal exhaust velocity.
- The decrease of vehicle velocity beyond a certain point is due toincreasing mass of power supply, and hence reduction in mass ratio.
- Withthemassratiofixedfortherocket,changesinburntimeindicatechangesinmassflow rate. The exhaustvelocity for a given power dependsinversely on the massflow rate. Solow massflowratesor longburntimesarebeneficial.Also,thrustisinverselyproportionaltothe burn time, and so long burntimes and highexhaust velocities implylowthrust.
- In general, electric thrusters have low thrust values, but this is offset by their high exhaust velocities.

2. **Vehicle Velocity and Structural/Propellant mass:** Electrical thrusters aremeant for bringing savingofpropellantmass.Relationbetweenvehiclevelocityasafunctionoftheratioof payload(structural) mass topropellant mass is indicated below:

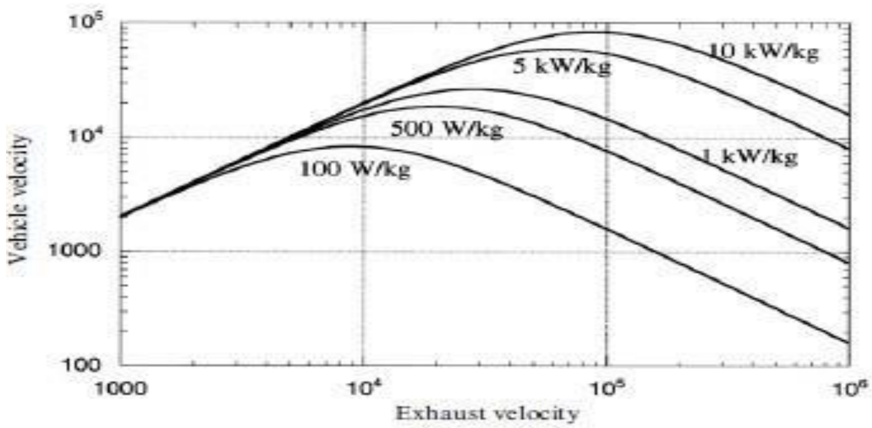
- Intheinterrelationbelow,theburntimeisfixedat1millionseconds,andthepower-to-mass ratio, ξ is fixed at 500 W/kg.
- Theratio $\frac{M_S}{M_P}$ is shown as a parameter.



It is evident from the interrelation that

- The vehicle velocity increases as the propellant mass increases
- The peak vehicle velocity shifts to the right ie peak vehicle velocity occurs at higher exhaust velocities as the payload mass increases

3. **Vehicle Velocity and power-to-mass ratio**: Vehicle velocity is plotted against exhaust velocity for varying power-to-weight ratios, in the plot below:



The above interrelation shows that

- As the power-to-mass ratio increases, the vehicle velocity increases.
- The peak vehicle velocity alsoshifts to the right, ie the peak occurs at higher exhaustvelocity as the power-to-mass ratio increases.

Importance of high Exhaust Velocity/high power-to-mass ratio:

- Highexhaustvelocitiesallows muchhigherpayload-to-propellantmassratios High
- power-to-massratioallowscrucialinobtainingthebestperformance.
- The basic characteristics of electric thrusters are **high exhaust velocity, low thrust levels and long burn times**

Electric Thrusters : Operation:

Electric thrusters can be divided into two broad categories: those that use electricity to heat the propellant, which emerges as a neutral gas, and those which use electric or magnetic fields to accelerate ions. The functional form and analysis of these two classes differ.

Resiso-jet:

Operating Principle & Components: Thebasicelectrothermalthruster,resisto-jet, consistofanozzle witha highexpansionratio,connectedtoachamberinwhichthepropellantisheatedbyahotwire through which an electric current passes. The hotgases generated by the heated propellant passes through a nozzle and are expanded thermodynamically. The expansioninthe nozzle results in a high velocityexhaust at theend ofnozzle. For high exhaustvelocity, the temperature and pressureof gases entering the nozzle should be high. Thisneeds efficientheating of propellant.

To maximize heattransferto the gas, a multichannel heat exchangeris usedtobring as much of gas volume as possible in contact with the heater.

The resistor-jet thruster is illustrated below:

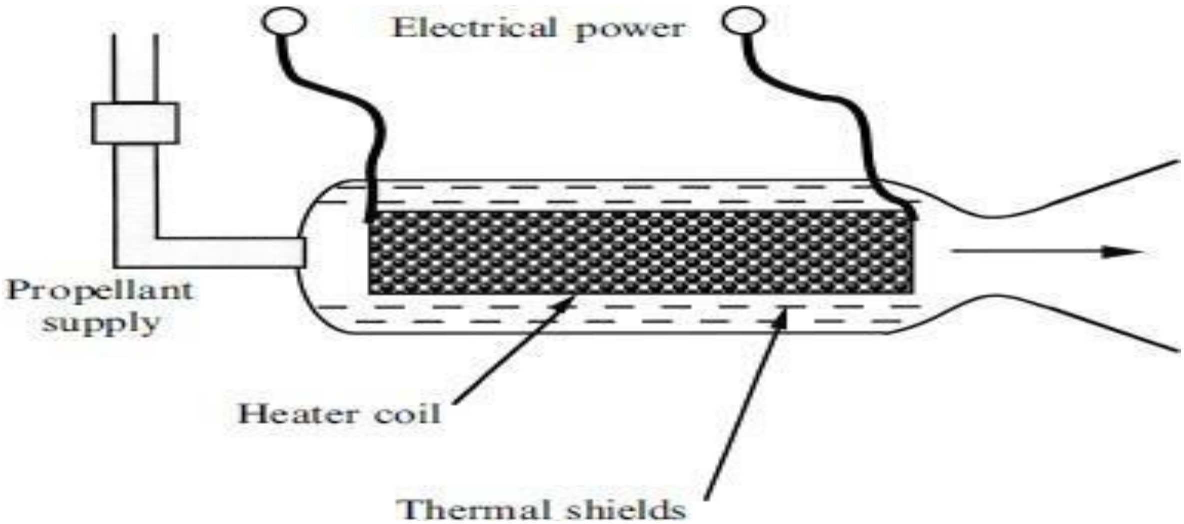


Fig: 5.5 resistor-jet thruster

System Parameters & Performance:Theexhaustvelocityiscalculatedusingthethrustcoefficient and characteristicvelocity

$$v_e = C_F C^*$$

$$\text{Where} C^* = [\gamma(\frac{2}{\gamma-1})^{\frac{\gamma+1}{\gamma-1}} \frac{\mathfrak{M}}{RT_c}]$$

- Thethrustisafunctionnozzleexitandchamberpressure(p_e & p_c)
- Sincethesethrustersareusedinvacuum, highnozzle expansionratiosareused, (around 2.25 for γ = 1.2)
- Whileinthechemicalrockets C^* dependson $\mathfrak{M}$& $T_c$for the electricthrusters, C^* mainly dependson $\mathfrak{M}$.(sincethereisnocombustionandthenozzleexittemperaturedependson powerinput andmassflowrate)
- Thenozzleexittemperatureinchemicalrocketsdependsontypeofpropellant,whereasin electric thrusters, the nozzle exit temperature is an inverse of mass flow rate.
- The melting temperature of heating element limits the maximum temperature levels in the thruster.

Example: Consider following data:

P_E=1kW; C_F =2.25; T_c= 2200 k; Propellantis hydrogen with $\mathfrak{M}$ = 2.

$$C^* = [\gamma(\frac{2}{\gamma-1})^{\frac{\gamma+1}{\gamma-1}} \frac{\mathfrak{M}}{RT_c}]\text{=4659 m/sec}$$

$v_e = C_F C^* = 2.25 \times 4659 = 10{,}483$ m/sec

- Electric thrusters can attain very high exhaust velocities,

The mass flow rate m is calculated from

$$\frac{1}{2} m v_e^2 = \eta P_E \quad \text{or } m = \frac{2\eta P_E}{v_e^2} = \frac{2 \times \eta \times 1000}{10{,}483^2} = 1.8 \times 10^{-5} \text{ kg/s}$$

- The mass flow rate of an electric thruster is very small compared to a chemical rocket The
- thrust for above thruster works out to 0.2 N, which is very small
- This means thatthe vehicle can achieve very high exhaustvelocities, but at lowthrust values, the time taken to accelerate to such high velocities is very long
- This is thefundamental difference between chemicalrockets and electric thrusters. The
- electrical efficiency can be very high at 90%

Propellantsusedcouldbehydrogen,helium,water(evenwastewatercanbeused)orhydrazene.

Disadvantages: Higherexhaustvelocities andpowerаredifficult to achievesincetransferofheatfrom filament to gas is difficult.

Arc-Jet Thruster:

OperatingPrinciple:IntheArc-Jetthruster,thepropellant gasisheatedbypassinganelectricarc throughthe flow. Temperaturesintheorder30,000-50,000k areachievedatthecenterlinewhichfully ionizes the propellant.

The anode and cathode are made of tungsten, which has high melting point. The cathode rodis pointed andis supportedin aninsulator. The insulator also holdstheanode. The anodeisshapedto create a gap withthepointed cathode,acrosswhichthearcisstruck.Thepropellantflowsthroughthisgapandgets ionized. Downstream of this arc, the anode is shaped to form a nozzle, for the expansion of the exhaust.

The propellant gas is introduced annular chamber around the cathode and swirls around it.

Thepowerthatcanbeappliedacrossanarc-jetisupto100timeshigherthanthefilamentofa electrothermal thruster. The temperature limit can be much higher.

While the propellantis ionized, the electrons and positive ions move towards anode and cathode. The cathode is struck at highspeeds, causing vaporization of the cathode material, therebylimitingit's life.

Thearcs cause concentration of energyand cause hotspotsleadingtoerosion of theelectrodes. Heat

lossesduetoionizationanddissociationarehigherthanelectrothermal thrusters.

Maximumexhaustvelocities arearound 20 km/s. Hydrogen, ammoniaandhydrazineareusedas propellants.

Power levels can reach up to 200 kW. However, heavier powersource is required than the electrothermal thrusters.

Arc-jets are best suited as station-keeping thrusters.

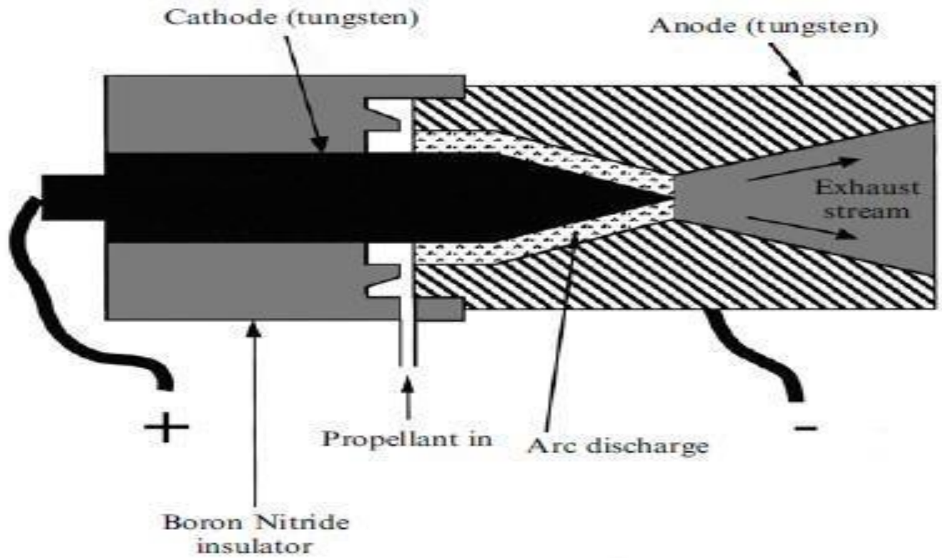


Figure 6.6. Schematic of an arc-jet thruster.

Fig: 5.6 Thermal Propulsion

Solar/Laser/Microwave Thermal Propulsion: Beamed energy, for example, a laser can be used for heatinginstead of onboard energysource. Solar/laser/microwave energysource, externalto the vehicle isused toheatupthe propellant.Theexternalbeamedenergymaybefromanearthorspacebased infrastructure. The energy is then concentrated on a heat exchanger or directly on the propellant, which isthenheatedupand expelledthroughaconventionalnozzle.Specificimpulsesof800-1200secand thrustlevels of several hundred mN arepossibleusingsunlightandhydrogenaspropellant.

A reflector is used to collect and concentrate sunlight/laser/microwave energy on to the propellant held in the chamber of the thruster.

Laser thermal propulsion offers higher specific impulse, but requires very high pointing accuracy.

Thisconceptisunderdevelopmentforusingsolarthermalpropulsiontoraisethecommunications satellite from LEOto GEO inabout 20 days. This conceptusesverylittlepropellant,savinglaunchcosts significantly.

3.1 Electrostatic Thrusters:

Performance Parameters:

- Ifthepropellantisionized,itcanbeacceleratedveryeffectivelybyelectrostaticfields.The velocity gainedforan**ion mass m** and**charge qdue totheelectric potential difference Uis given by**

$$v = \sqrt{\frac{2qU}{m}},$$

The mass flow rate is related to the current I, as

$$\dot{m} = I \times \frac{m}{q}$$

Andtheforcegenerated F, canbeexpressedas $F = I \times \sqrt{\frac{2mU}{q}}$

Forobtainingveryhighspecificimpulse,amulti-ionised,lightionwouldbeideal.However,sincethe thruster should produce high thrust, propellant with heavy ions is preferred.

3.1.2 Ion Thruster:

Working Principle: Thepropellantisionized, andthen enters aregion of strong electricfield, wherethe positiveionsareaccelerated. Theions areacceleratedpassingthroughthegridand leavetheengineas ahigh velocityexhauststream.Highestexhaustvelocities(morethan32,000m/s)areachievedby accelerating positiveionsinanelectricfieldcreatedbytwogridshavinglargepotentialdifference.

The electrons do notleave, therefore the electron current is discharged through a neutralising cathode, in to the exhaust.This wouldneutralisethespacecraft. Theelectrons dischargedcarrylittlemomentum, therefore do not affect the thrust.

Thethruster is dividedintotwo chambers. Propellant, (usually Xenon gas) enters ionisation chamberin the form of neutral gas molecules.

The cathode at the center, emits electrons, which are accelerated by the electric field. These electrons ionise propellantthroughelectroncollision. Theionisedpropellantdriftthroughthegridswithhigh potential difference and accelerate. The ions gain energy and form the ion beam with high velocities of around 32,000 m/sec.

Thrustis exerted by the departing ion stream on the accelerating grids and is transferred through the bodyof thethrustertothespacecraft. Theexhaustvelocityisgovernedbythepotentialdifference betweenthe

gridsandthemassflowrateisdirectlyrelatedtothecurrentflowingbetweenthegrids.
There is no need for a nozzle to generate thrust .

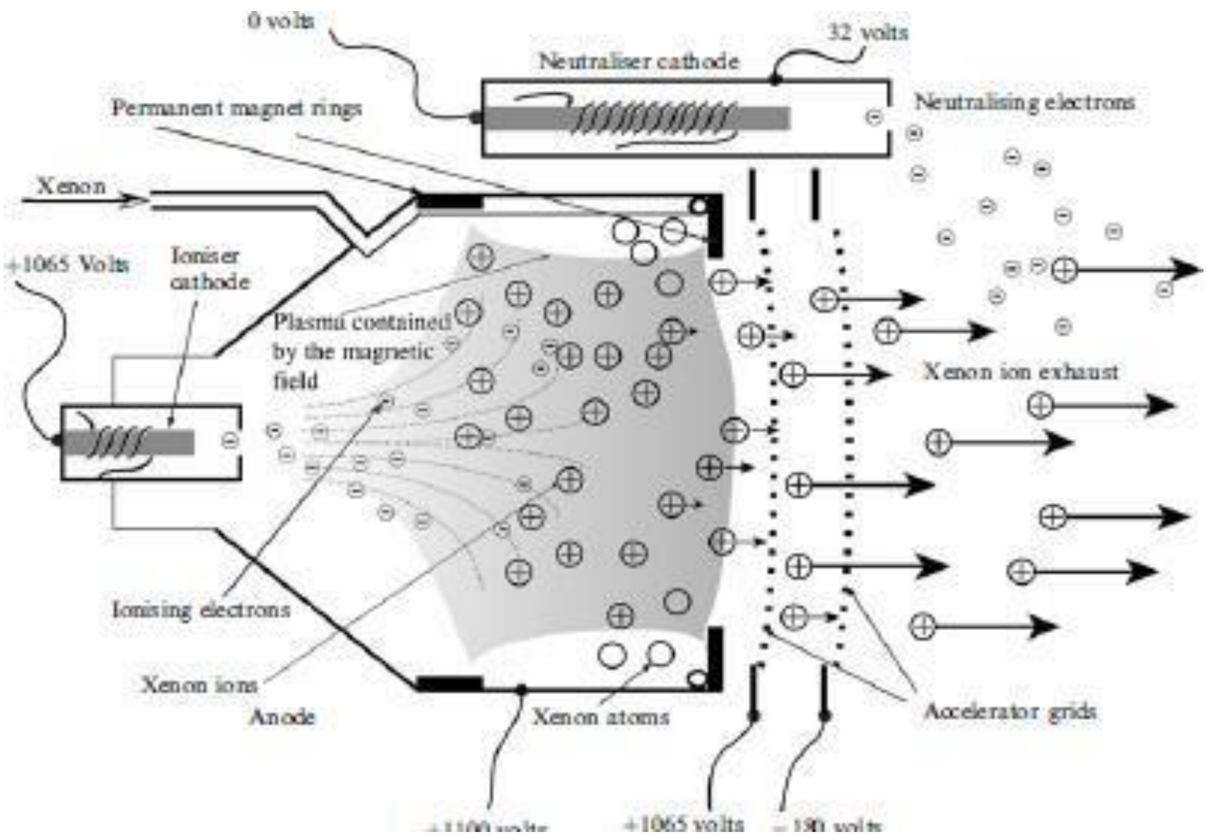


Fig: 5.7 Applications of Ion Engines:

Ionengines arebestusedforveryhighvelocityincrementmissionslikeinter-planetarymissions and station keeping.

Ion engines are not used for attitude control due to their low thrust.

Limitation of Ion Thrusters- The space-charge limit**:**

The accelerating grids have an electricfield between them, which gets partially blocked as theionsstart accelerating along thegrids.Asthe density offlow ofionsincreases,apointwill reachwhenthe accelerating field at the first grid drops to zero, because the positive charge of the ions passing through cancels the field.

This is the space-charge limit, which limits further ingress of ions and limits thrust levels.

Electromagnetic Thrusters: Thelowthrust-highexhaustvelocityionthrustersarelimitedbyspace-charge limit. Plasma thrusters(electromagneticthrusters) offerhigher thrustvalues.

Inplasmathrusters,anionisedgaspassesthroughachannelacrosswhichorthogonalelectricand magneticfields aremaintained. The current carried by the plasma(electrons andions) along the electric field vector interacts with the magnetic vector, generating a high propulsive force. The plasma accelerates without the need for area change

MagnetoplasmaDynamic(MPD)thrustersandPulsedPlasmathrusters(PPT)areconventionaltypeof electromagnetic thrusters. The Hall Effect thruster is another variant of the electromagnetic thruster.

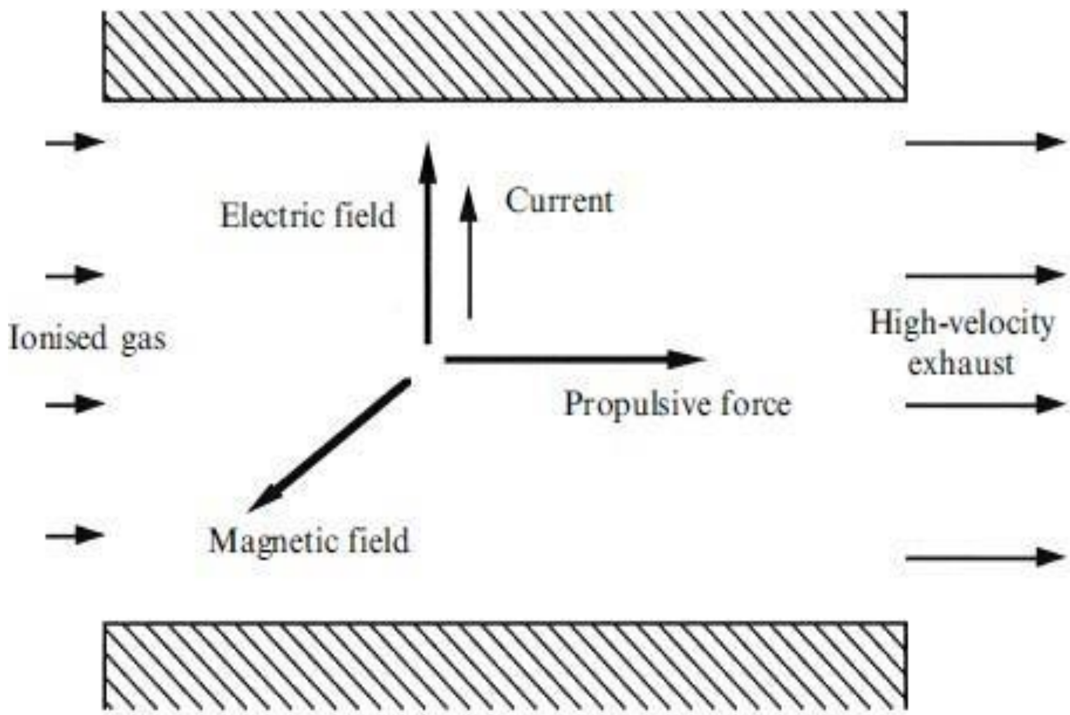


Figure 6.14. Principle of the plasma thruster.

Fig: 5.8 Pulsedplasma thruster

Pulsed plasmathruster (PPT): Plasmathrustersdonotusehighvoltagegridsoranodes/ cathodesto acceleratethechargedparticlesintheplasma,butratherusescurrentsandpotentialswhichare generated internally in the plasma to accelerate the plasma ions.

Whilethisresultsinlowerexhaustvelocitiesbyvirtueofthelackofhighacceleratingvoltages,thistype of thruster has a numberof advantages.

In the PPToperation, an electric arc is passed through the fuel, causing ablation andsublimation of the fuel.The heatgeneratedbythisarccausestheresultantgastoturnintoplasma,therebycreatinga chargedgascloud.Due totheforceoftheablation,theplasmaispropelledatlowspeedbetweentwo charged plates (anode and cathode).

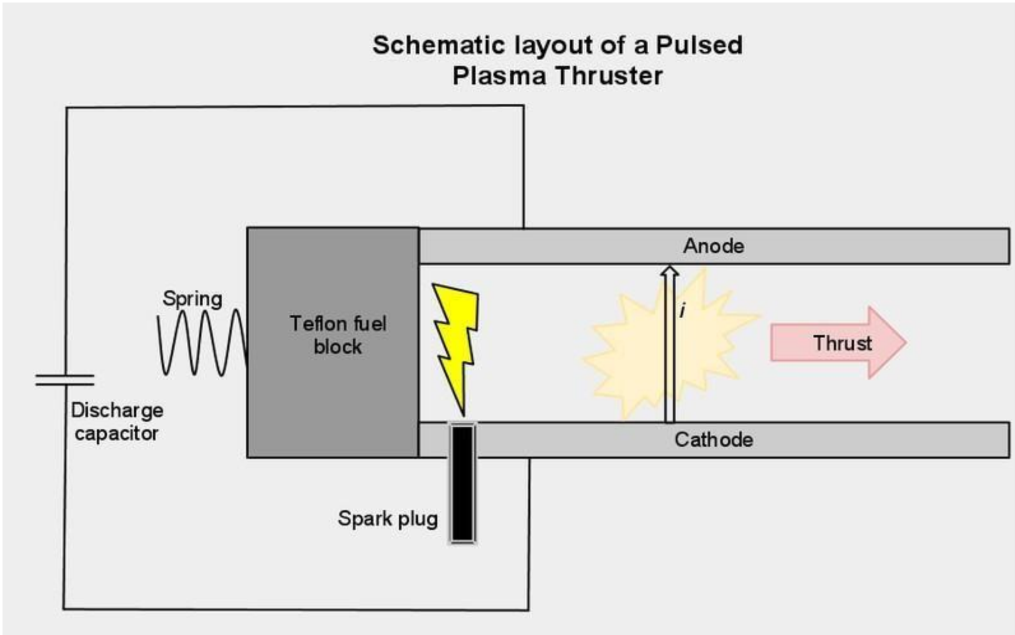


Fig: 5.9 Pulsedplasmathruster

Since the plasma is charged, the fuel effectively completes the circuit between the two plates, allowing a currentto flowthroughtheplasma.Thisflowofelectronsgenerates astrongelectromagneticfield which then exerts a Lorentzforce on the plasma, accelerating the plasma out of the PPTexhaust at high velocity.

Thetimeneededtorechargetheplatesfollowingeachburstof fuel,andthe timebetweeneacharc causes pulsing. Thefrequency of pulsingisnormallyveryhighandsoitgenerates an almostcontinuous and smooth thrust.

WhilethethrustgeneratedbyPPTisverylow,itcanoperatecontinuouslyforextendedperiodsoftime, yielding a large final speed.

Asolidmaterial,teflon(PTFE)iscommonlyusedpropellant.FewPPTsuseliquidorgaseouspropellants also.

Magnetoplasmadynamic (MPD) thrusters: MPD thrusters, also referred as Lorentz Accelerators, use the Lorentzforce(aforce resultingfromtheinteractionbetweenamagneticfieldandanelectric current) to generatethrust

Theelectricchargeflowing throughthe plasmain the presence of a magneticfield causingthe plasma to accelerate due to the generated magnetic force.

The operation of MPD thrusters is similar to pulsed thrusters.

Hall Thrusters: Hall Effect Thrusters combineastrong magneticfield perpendicular totheelectricfield created between an upstream anode and a downstream cathode called neutralizer, to create an area of high density of electrons. The electrons are trapped in a magneticfield and these electrons confined to the field are used to ionise the propellant.

The cathode then attracts the ions formedinside the thruster, causing the ions to accelerate and produce thrust .

Operation of Hall Thruster: Anelectricpotentialbetween 150 and 800 voltsis appliedbetweenthe anodeand cathode.Electronsfromahollowcathodeenteraringshapedanodewithapotential difference of around 300 V.

Thecentralspikeformsonepoleofthemagnet,andaroundtheinnerpole,anoutercircularpoleforms an annularradialmagneticfieldinbetween. Thepropellant,usuallyxenongasisfedthroughanode wherethe neutralxenonatomsdiffuseintothe channel,andionisedbycollidingwiththecirculating high energy electrons.

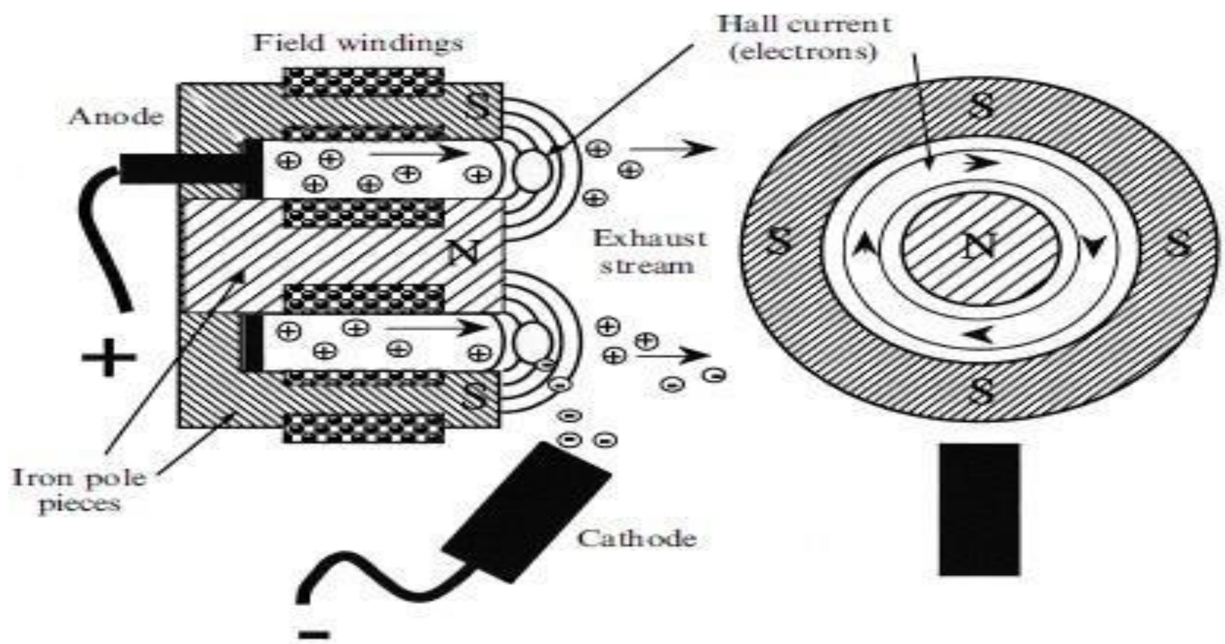


Figure 6.16. Schematic of the Hall thruster.

Fig: 5.10 Hall Thruster

The xenonions arethen accelerated by the electricfield between anode and cathode. Ions reach speeds of around 15 km/secwithspecificimpulse of1500sec.

Thrust levels are very small, around 80 mN for a 300 V, 1.5 W thruster.

Theacceleratingionsalso pull someelectrons forminga plume. Theremainingelectrons arestuck orbiting the region, forming a circulating hall current. This circulating electrons of hall current ionise almost all the propellant.

Hall thrusters can provide exhaustvelocities of 10-80 km/s andspecificimpulse of 1500-3000 sec. Most commonly used propellants are xenon, argon and krypton

The applications of Hall-effect thrusters include control of orientation & position of orbitingsatellitesandtopowerthemainpropulsionengineformedium-sizeroboticspacevehicles.

2 KW Hall Thruster in Operation

Fig: 5.11 Electric Thrusters

Applications of Electric Thrusters: The applications for electrical propulsion fall into broad categories as below:

1. **Attitude Correction (Space Station/Spacecraft)**: Overcoming translational and rotational perturbationsinorbits;**Drag compensation** forsatellitesinLowEarthOrbits;**Aligning** telescopesorantennas.**Electro-thermal (resisto-jets)** arepreferredusinglowcostpropellant like coldgasorwastewater.MPDthrustersarealsobeingconsideredforattitudecontrolof space vehicles.
2. **Station Keeping**: Forstationkeepingpurpose,savingsinpropellantmassisverysignificant. SynchronousandGEOsatelliteshavelonglifeperiodsneedextensivestationkeeping requirement. **Electro-thermal (Arc-jets) thrusters** havebeenwidelyusedforthistask.Hall thrusters and Ion engines are most suitable.
3. **Raising Orbits**: Fromlow earth tohigherorbits (up to Geostationary orbits), circularizing an ellipticalorbitInter-planetarytravelanddeepspaceprobes.Theyallrequirerelativelyhigh thrust andpowerintherangeofaround100kW,muchhighervelocityincrementsthanthose needed for station keeping. Also these corrections need to carried out in reasonablelength of time. **Hall thrusters and Ion engines** are again preferredhere.

4. **Inter-planetary missions** : These are deep space long duration applications. **Ion engines** with higher exhaust velocities are preferred.